REVIEW OF CANADIAN AERONAUTICAL FATIGUE WORK 2001-2003

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SUMMARY

This paper provides a review of Canadian work associated with fatigue of aeronautical structures during the period 2001 - 2003. All aspects of structural technology are covered including full-scale tests, loads monitoring, fracture mechanics, composite materials and nondestructive inspection.

Organisation Abbreviations Used in Text:
AMRL - Aeronautical and Maritime Research Laboratory (Australia)
ATESS – Aerospace and Telecommunications Engineering Support Squadron (DND)
BADS - Bombardier Aerospace Defence Services
CF - Canadian Forces
DND - Department of National Defence
DRDC – Defence Research and Development Canada
DTA - Directorate of Technical Airworthiness (DND)
IAR - Institute for Aerospace Research
NRC - National Research Council of Canada
QETE - Quality Engineering Test Establishment of DND
RAAF - Royal Australian Air Force
RMC - Royal Military College (DND)
SMPL – Structures, Materials and Propulsion Laboratory

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INTRODUCTION

Canadian industry, universities and government agencies were solicited for information describing their fatigue technology related activities over the period 2001 to 2003. This review covers work performed or being performed by the following organizations:

Bombardier Aerospace
- Defence Services
- Regional Aircraft

Goodrich Landing Gear

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

National Research Council of Canada
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FULL-SCALE TESTING

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

R.S. Rutledge, SMPL, IAR, NRC

The Structures Laboratory of the Institute for Aerospace Research (IAR) is conducting the F/A-18 International Follow-On Structural Test Program (IFOSTP) wing test. The set up and development of the wing fatigue test has been described in earlier ICAF reviews. The wing test is a quasi-static test that accounts for manoeuvre, inertia, and some dynamic loads. The test objective is to determine the economic life of the F/A-18 IFOSTP aircraft wing structure and where possible obtain crack growth data to support management on a safety by inspection basis, validate repairs/modifications to the structure and obtain engineering data. This test is being conducted to demonstrate that the F/A-18 wing structure has a life equal to or greater than 6000 operational hours free of catastrophic failure. It is also required to verify the period decided upon for fleet structural inspections and to yield data on maintenance schedules, component replacements such as aileron and trailing edge flap fittings and repairs for modifications where applicable.

The current plan to demonstrate the wing structural life to 6000 hours is to test until approximately 21190 simulated flight hours (SFH) followed by residual strength testing. The 21190 SFH, or 65 blocks of testing, is projected in order to apply sufficient damage to the trailing edge surface attachments of the wing box.

The status of the four IFOSTP tests under way or completed is:

- FT55 – Center Fuselage FSDADTT; Bombardier Aerospace-Defence Services – Mirabel, Canada – **16636.5** Simulated Flight Hours 5 December 2001 and Residual Strength Testing Complete July 18, 2002.
- FT46 – Aft Fuselage and Empennage FSDADTT; Aeronautical and Maritime Research Laboratory – Melbourne Australia - **23090.2** Simulated Flight Hours Wednesday 26 June 2002 and Residual Strength Testing Complete Tuesday, December 17, 2002.
- FT488/2 – Bare Bulkhead Test, Aeronautical and Maritime Research Laboratory – Melbourne Australia – failure **38135** Simulated hours July 13th 1999; and
- FT245 – Wing FSDADTT, NRC Institute for Aerospace Research, Structures Laboratory – Ottawa, Canada – **11922** Simulated Flight Hours February 3, 2003. Level 1 inspection and fuselage change.

Significant progress has been made on the IFOSTP wing fatigue test. Fatigue cycling that commenced on January 26 2001 and accelerated through twenty four hour a day testing in September 2001 provided the framework to bring the test to 11922 SFH or near two design life times by February 2003. Expeditious use of and repairs to control surfaces has allowed the testing to continue with only minor interruptions to this stage of the test. However, a large crack developed on the Y488 bulkhead in the fuselage structure just below the drag load member. This crack fractured the lower ligament of the frame equivalent to 11596 simulated flight hours of testing (but remember that this transition fuselage had significant service usage prior to use on the test). This break in the fuselage was significant enough to hasten the fuselage change and level one inspection planned between 9000 SFH and 13325 SFH. This activity should last between six to eight months depending on damages found to the test wing or modifications incorporated. Illustrations of the first fuselage removal and second fuselage installed are included in Figures 1 and 2.

The transition structure fuselage and stub wing performance has been enhanced through bonded repairs, shot peening, special fasteners and bushings and preemptive polishing of several critical areas. Using these structural repair tools has minimized delays to structural testing of the specimen. Other transition structures have been replaced in order to repair the damaged first component. To date two trailing edge flaps, two ailerons, three inboard leading edge flaps and two trailing edge flap shrouds have been prepared in order to maintain the requested schedule of the client. To date over 700 damages have been reported for the test in the structural information database.

Strain data trend monitoring has been used to identify areas where structural changes are occurring in the test specimen such as the aft closure rib and the fastener failures at two holes on the front spar at the kick rib. Use of trend strain data ultimately lead to the discovery of the failure at the Y488 bulkhead in the transition structure and cracking of the closure rib on the stub wing. Strains are monitored throughout the test block using a limited number of end levels. These end levels have been recorded in each of the thirty-seven blocks completed. Strain data was also recorded for three full blocks in pass 1, 14 and 36 and prior to block 25 a unit strain survey was carried out to aid DND and Bombardier in the interpretation of data.

Several significant failures have been discovered pertaining to the wing testing. The most significant are the: Trailing Edge Flap (TEF) wing half hinge cracking; aft closure rib cracking, hole elongations and fastener problems resulting in structural reinforcements of the closure rib and aft closure rib using bonded titanium doublers; and Inboard Leading Edge Flap (ILEF) cracking at multiple locations and confirmation of cracking in the fleet ILEF's. Quantitative fractography has been valuable in ascertaining the crack growth of the ILEF cracks.

In addition to these damages a significant non-failure due to structural modification can be reported. The holes 170-173 on the front spar of the inner wing were cold worked at 3977 SFH. This modification has seen 7945 SFH since incorporation of the modification and no damage has been detected at 11922 SFH. Therefore, if this modification is incorporated in the fleet after mid life then testing has proven adequate to obtain the design service life of 6000 hours, which is the manufacturer design service life. Subsequent testing could provide justification to extend the service life of the inner wing box should this be the most critical item.

The IFOSTP team (NRC, DND, Bombardier, RAAF, AMRL) was awarded the International Council on the Aeronautical Sciences von Karman Award for International Collaboration. D. Simpson accepted the award on behalf of the Canadian participants.

In this year the FT245 wing test challenge is to replace the transition fuselage (Figure 1 and Figure 2) and provide timely inspection data for the test wing to the clients. Following a test restart in September the challenge will be to keep the transition structures repaired. The second fuselage came fitted with several Bombardier Aerospace Defense Systems modifications. The testing of these modifications will aid in validating and obtaining engineering data for future fleet management.

Figure 1. FT-245 Wing Fatigue Test First Fuselage Removal

Figure 2. FT245 Wing Fatigue Test Second Fuselage Installation

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

J. Dubuc, Bombardier Aerospace

Bombardier Aerospace, Defense Services (BADS) was mandated by the Canadian Forces (CF) to perform a full-scale fatigue test on the CF-18 Center Fuselage. In order to demonstrate the original design life of 6000 flying hours, a total of 17,400 simulated hours were required. The fatigue cycling of the test article (designated FT55) started in May 1995 and test hours began to accumulate, interrupted by periods of extensive inspection and modification. On December 13, 2001 a large crack occurred in the Y488 bulkhead just 74 hours short of the test target time. It was decided that this was adequate and preparations began to conduct a residual strength test (RST) in accordance with the new CF-18 Lifing Policy, to be followed by a teardown inspection.

The preparation for the RST was extensive, as the test setup had not been designed for application of static loads well in excess of limit load. The hydraulic system was modified to produce higher pressure, the only way the existing jacks could apply the required loads. Modifications were made to the rig to accommodate the greater structural deflections. Safety measures were taken to contain the pieces should a catastrophic failure occur. The target loads for the RST were defined and agreed by the CF and RAAF authorities.

In order to provide some guidance to the FT245 wing test managers for the test currently taking place at the National Research Council, an additional test was incorporated into the centre fuselage RST. This involved introducing artificial cracks into a left hand inner wing, representative of the expected condition of the inner wing test article at 18,000 test hours. Demonstrating that the wing structure could tolerate large cracks would help decide whether the skin needed to be removed for modifications.

Figure 3. FT55 at Ultimate Test Load 610 m (24”) Wing Tip Deflection.

After the test article had been carefully inspected, the cracked bulkhead repaired and the specially prepared inner wing installed, the RST work up began. The systems were tested and a strain survey conducted that confirmed the new set up was fully functional. About 100 moderate cycles were applied to induce “real” crack tips at the saw cuts in the inner wing. The RST began May 31, 2002. First the wing-specific load cases were applied successfully. The centre fuselage load cases followed and all was uneventful until the June 6 application of a maximum wing up-bending case ended with a loud “bang”. This was found to be a 381 mm (15 inch) crack in the Y488 bulkhead, at a shear web near the wing attachment.

A boiler-plate repair was installed and testing resumed on July 11. With the successful application of all the remaining load cases, it was decided to push the test article beyond the required targets and see if a weakness could be found in the structure. On July 18, 2002, the centre fuselage sustained a surprising 145% of the maximum fatigue spectrum wing root up bending moment (**Figure 3**), only about 10% less than the ultimate load used in design of the aircraft.

This marked the end of loading of the FT55 test article and preparations began immediately for the teardown inspection. Inspections were begun as soon as areas were accessible or parts were extracted. The test article was completely dismantled down to the part level by February 2003. As of March, 2003, over 130 Notices of Structural Discrepancy have been produced, documenting the findings of the teardown inspections. QETE is also involved in this work, performing qualitative fractography on selected cracks to identify initiation time, growths rates, etc.

FT55 has thus certified most of the components on the Center Fuselage and demonstrated the performance of many structural modifications. The RST has served to show that most of the teardown cracks are acceptable to the airworthiness authorities and will not require time-consuming maintenance activities.

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

Spectrum Editing for a Full-Scale Fatigue Test of a Fighter Wing with Buffet Loading

R.L. Hewitt, SMPL, IAR, NRC

The spectrum editing techniques used to derive the test spectrum for a full-scale fatigue test of an F/A-18 wing with buffet loading were reported in [1]. The methodology for assessing the effects of the editing, using an experimentally verified strain-based fatigue life calculation, were also detailed.

Truncation was based on ranges and deadbands of individual sequences of section loads derived from the full loads sequence for a representative year of flying. Because the buffet added a very large number of end points to the sequences, aggressive truncation was found to be necessary in order to be able to apply the test spectrum in a reasonable time period. Truncation reduced the number of lines in the spectrum from about 20 million for the initial sequence with a 1% range truncation to about 175 000.

Even after this aggressive truncation, it was found that there were about 135 000 independent load cases. This number was reduced to less than 50 000 by a binning process so that the sequence could be accommodated by the test controller.

Overall, the lives based on manoeuvre dominated loads (e.g., wing root bending moment) were not changed by more than a few percent by all the spectrum editing processes. Lives based on buffet dominated loads (e.g. trailing edge flap lug load) increased by up to about 80%. Almost all of the increase was due to the aggressive truncation used for these loads. However, this was still a significant improvement over previous tests, which did not include buffet.
The binning process made a negligible difference to calculated fatigue lives and appears to be a useful tool for reducing the number of independent load cases. This has benefits both in terms of calculating and verifying actuator loads for the test and reducing the size of the file for the test controller.

CRJ200/700/900 Fatigue Testing

Rob Dewar, Bombardier Aerospace (BA)

The success of the first generation of BA Canadair Regional Jets (CRJ 100/200) is best illustrated in Table 1 showing number of aircraft in service and cumulative number of hours flown. This success led to the development of the second generation of CRJ 700/900s. As a result of these developments BA has been involved in an unprecedented amount of fatigue testing.

<u>CRJ100/200</u>

To augment the original CRJ100/200 structural test program, BA through the Fleet Leader Program has continued monitoring and inspection of high time aircraft in the field. The two aircraft were inspected Nov ‘01 and May/June ‘02 respectively with no significant new issues were found on the fleet leader Aircraft at 21,000 flights. All findings are communicated via the Structural Working Group.

Table 1. CRJ Fleet Statistics

As of April 2003:	100/200	700/900
Number of Aircraft in Service:	800	62
Number of Operators in Service:	24	8
Cumulative Flight Cycles	5.1 million	97,644
Average Flight – minutes	68	76
High Time Aircraft Cycles	23,580	3,595

The Fleet Leader Program has proven to be an asset in the support of CRJ100/200 fleet and BA has decided in November 2002 to provide additional fleet support by launching a New CRJ200 Full Scale Fatigue Test. The goal is to validate complete aircraft durability by test of fleet leader Aircraft 7041 (CRJ200). The aircraft was acquired October 2002 with 21,580 cycles accumulated in service. The aircraft has been stripped down and installed in test rig. Fatigue testing will commence in August 2003 following completion of instrumentation and incorporation of structural repairs. First life completion is planned for mid 2004 (80,000 cycles), second life completion mid 2005 (160,000 cycles).

Structural Airframe Testing CRJ700 & 900

The CRJ700/900 design philosophy was to develop a new generation airframe structure designed specifically for regional market, incorporating numerous structural improvements requiring extensive upfront structural analysis and validation of structural concepts via testing. Table 2 compares CRJ family structures while Table 3 provides durability goals.

Table 2. CRJ700/900 is a New Airframe

Component	CRJ200	CRJ700	CRJ900
Wing	Base	New	Common 700/900
Fuselage	Base	New	Extended & Reinforced
Cockpit	Base	New	Common 700/900
Empennage	Base	New	Common 700/900
Horizontal Tail	Base	New	Common 700/900
Aft Press Blkh'd	Base	New	Common 700/900

Table 3. Durability & Damage Tolerant Life (DADT)Comparison

Aircraft	Economic Life(Flts)	Threshold (Flts)	Repeat (Flts)
CRJ200	80,000	40,000	20,000
CRJ700	80,000	40,000	20,000
CRJ900 **	60,000	30,000	15,000

**CRJ900 Life Extension to 80,000 is under consideration.

The CRJ700/900 structural improvements include: substantially increased wing durability - wing planks design optimized through a new stringer section and a smaller stringer pitch for improved DADT, improved fuel access cutout design, improved wing plank splices. CRJ700/900 detailed static tests included: complete airframe static test, forward engine mount, aileron, elevator, rudder, ground spoiler, multi-function spoiler, inboard flap and vane, slats, winglet, cabin windows, pitch trim actuator, bird strike tests, wing leading edge, empennage and nacelle nose-cowl bird strike.

The CRJ700/900 DADT tests required 24 dedicated test rigs (Table 4). All have so far completed at least one full life.

Table 4. Test and test cycle count goal.

DADTT Test Rig	Flight Count
Fwd Fuse DADTT	82,500
Center Fuse / Wing DADTT	73,750
Aft Fuse DADTT	80,000
Slat Track 1-2 Deployed DADTT	160,000
Slat Track 1-2 Retracted DADTT	160,000
Aileron DADTT	120,000
Rudder DADTT	120,000
Multi Functional Spoiler DADTT	140,000
Inboard Flap DADTT	100,000
Inboard Slat Body DADTT	160,000
I/B Flap H/Box WS 128.00 DADTT	140,000
O/B Flap H/Box WS 264.00 DADTT	140,000
O/B Flap H/Box WS 220.00 DADTT	140,000
Winglet DADTT	160,000
Ground Spoiler DADTT	160,000
Slat to Track 1-3 Attach. DADTT	160,000
Elevator	80,000
Fwd Eng mounts	80,000
Slat #3 DADTT (Figure 4)	60,000
MLG Trunnion	29,500
CRJ900 Center Fuse / Wing DADTT	39,200

Figure 4. SLAT #3 DADT test rig.

CRJ700/900 Major Aircraft Structural Tests include:

- CRJ700 - Ground Vibration Tests & Flutter Tests, Complete Aircraft Static Test Article (CAST), Durability & Damage Tolerant Test (DADTT)

- CRJ900 Ground Vibration Tests & Flutter Tests, Durability & Damage Tolerant Test (DADTT)

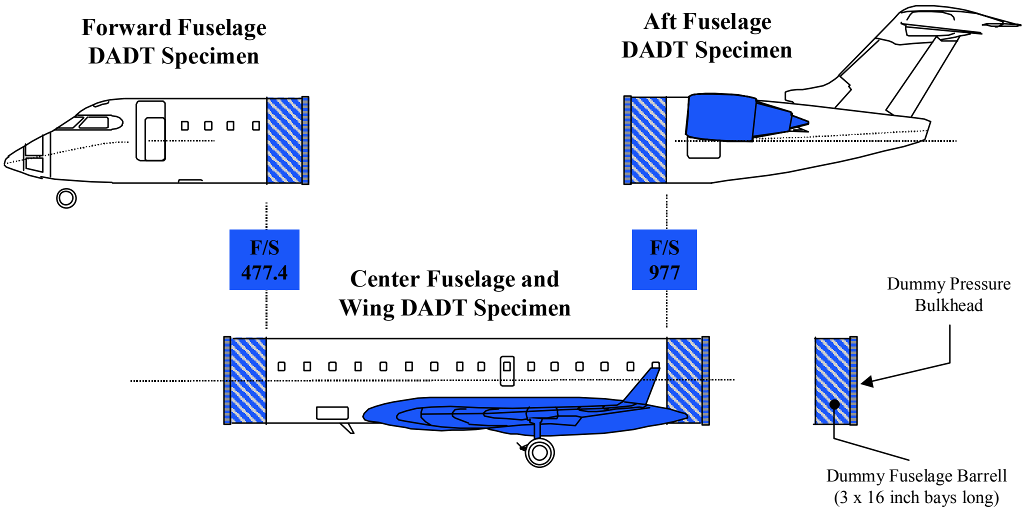


Figure 5. CRJ700 DADT test article configuration.

The CRJ700 DADT article configuration shown in Figure 5 is tested in three subsections, significantly reducing the duration of the testing. DADT test will simulate 2 life times (160,000 cycles). Artificial damage is being introduced during 2nd life to understand the damage tolerance of the structure. Repair schemes are evaluated as part of this test. Testing started in mid January 2000, approx one life of testing 80,000 cycles completed to date.

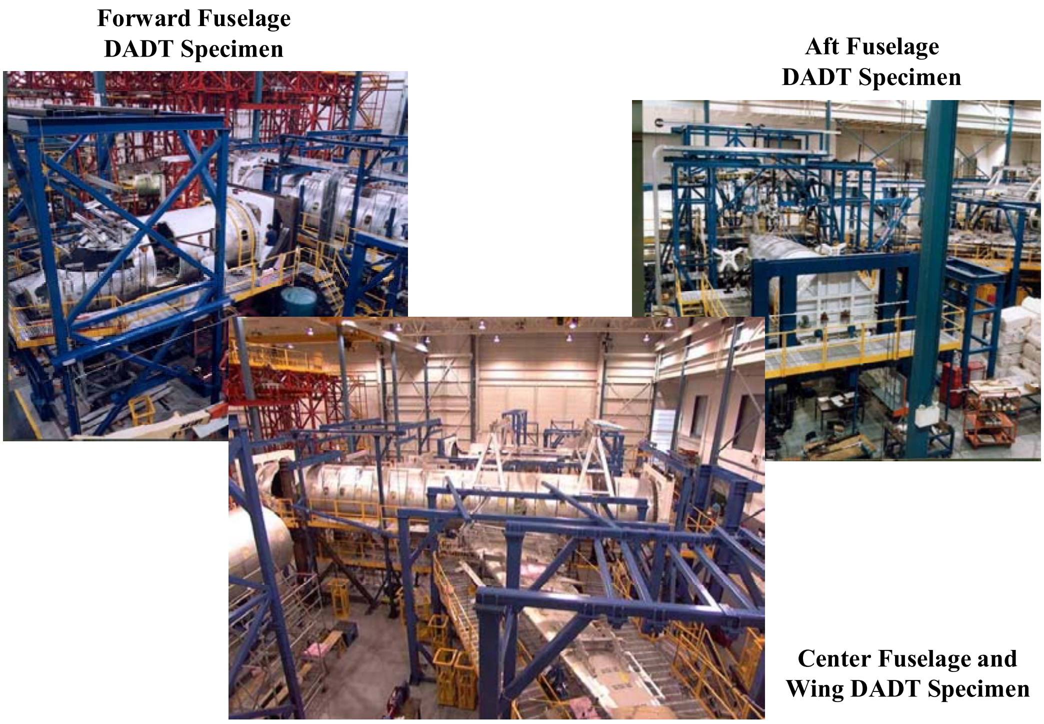


Figure 6. CRJ700 DADT tests.

The status of these tests shown in Figure 6 is as follows:

- forward fuselage test: 80,000 flights, to date no significant finding, five minor repairs

- center fuselage and wing test: 73,757 flights, MLG trunnion required modification incorporated
- aft fuselage test: 80,000 flights, findings to date: pressure bulkhead and frame FS 1108 &1126 - all modifications incorporated in production and Service Bulletins released.

CRJ900 Major Airframe Structural Testing

New CRJ900 centre fuselage & wing test article (Figure 7) accounts for the following structural CRJ700 to CRJ900 differences: additional overwing exits, additional forward cargo door, RHS service door, longer fuselage. For all other components CRJ700 test specimens will be used as a basis to validate CRJ900 DADTT requirements.

This article has completed by December 2002 34,500 flight cycles without any findings. Figure 8 shows the test article being installed in the test rig.

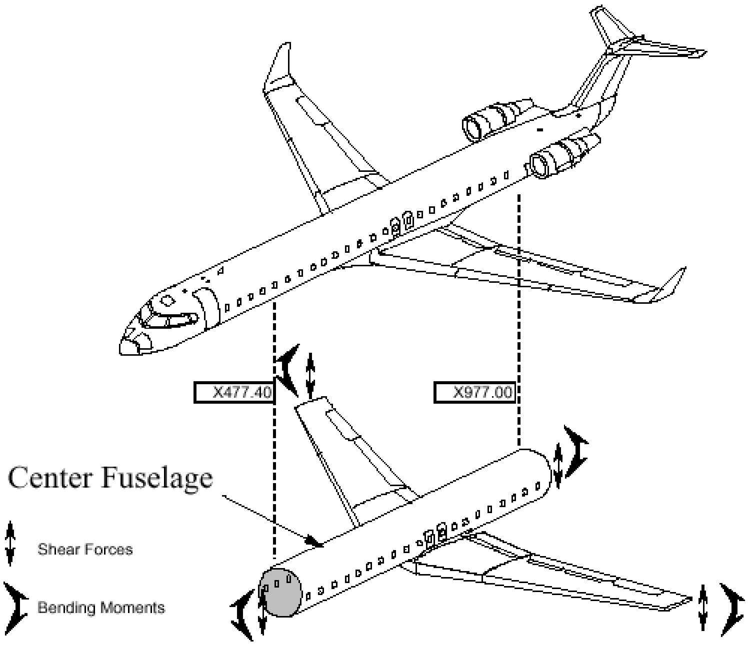


Figure 7. CRJ900 DADT test article.

Figure 8. CRJ900 DADT article in the test rig.

Structural Fatigue Testing Technology - IAR/MTS Systems Corporation

R.L. Hewitt, SMPL, IAR, NRC

Full-scale aircraft structural fatigue tests are extremely complex from a control systems viewpoint. There are usually a large number of actuators with significant interactions between them and control is made more difficult because the load cells usually move with the actuators. A model of a full-scale structural test system was developed at IAR under a contract to MTS Systems Corporation, Minneapolis, with the aim of improving the understanding of the complete structural test system and thereby reduce both the set-up and operation time for a full-scale test. This work was reported in the previous ICAF review.

The model, which was initially developed as state space systems within Matlab, has now been converted for use with Simulink to simplify its use. This allows the model to be developed by simply connecting the individual elements of the test system, as shown in Figure 9.

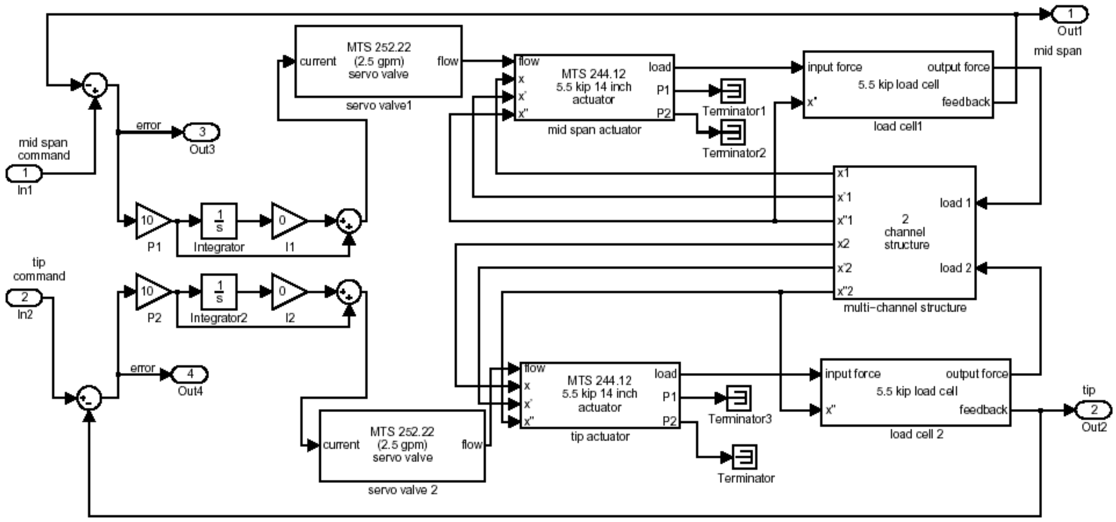


Figure 9. **Model of two-actuator test system**

The model has been used to investigate the response of multi-channel systems and to investigate some typical problems within full scale testing, such as the determination of the optimum transition times between end points [2]. It has also been used for a number of proprietary investigations aimed at increasing test speed.

IAR and MTS Systems Corporation have recently signed a collaborative research agreement to develop a full-scale structural test system test bed at IAR. This will initially be a six-channel system using a piece of real aircraft structure. It will be used both for validating multi-channel models and experimentation with different control algorithms, tuning procedures and fixturing systems.

LOADS AND USAGE MONITORING

CT114 Tutor Operational Loads Monitoring Program

J. Dubuc, Bombardier Aerospace, R. Kaufman DTA

The CT114 Tutor aircraft was originally designed, monitored and maintained according to a safe-life philosophy. The aft fuselage and empennage are still managed with a safe-life approach, although, the wing and carry-through structures have switched to a safety by inspection philosophy. The appropriate inspection intervals for identified structurally significant items are determined from on-going testing, damage tolerance analysis and in-service failure data.

The Tutor aircraft is mainly used in the Snowbirds role since 2002. There are 21 aircraft in operation with 18 equipped with an Operational Loads Monitoring (OLM) system (i.e. four (4) flight parameters and four (4) strain gauges).

With the implementation of the combined Safe Life and Safety-by-Inspection philosophy, six (6) new wing critical locations were added in the Bombardier's General Integrated Fatigue Tracking System (GIFTS). This brings to eleven (11) critical locations from the wings, center fuselage and tail components that the usage tracking is performed.

On-going OLM data processing is done and results are presented on a monthly, semi-annual and annual basis to the fleet manager for aircraft rotation, component swapping and aircraft retirement decisions.

New scatter factors for the wings, center fuselage and tail components were proposed based on the standards of the Canadian Forces lifing policy. A study was also performed to establish if it would be possible to lower the tracking factor of 1.5 used for unmonitored missions. With the results of this study a new fill-in procedure was also proposed to the Canadian Forces.

The Tutor airframe is now over 37 years old and may be flown until 2010. In order to achieve an estimated life expectancy of 2010, a major review of the current ASIP plan must be completed to identify all the structurally significant items, the appropriate inspection intervals for each role (Snowbird - solo, Snowbird - non-solo and Trainer) and the scatter factor methodology.

In-service damage has shown that the original unmodified lower spar cap flange is prone to fatigue cracking resulting in the wing severing from the aircraft in the wing full-scale fatigue test. A damage tolerance analysis of the lower main spar cap flange for the unmodified and repaired flange is being completed by the OEM to determine a suitable inspection interval. Subsequent in-service cracking after the embodiment of the repair has shown that further strengthening of the lower spar cap flange is required. The design, development and validation of a lower skin doubler will be developed by the OEM to reduce the loading in this structurally significant location.

CT133 Silverstar Operational Loads Monitoring Program:

J. Dubuc, Bombardier Aerospace, R. Kaufman DTA

Following the fleet reduction in April 2002, the CT133 aircraft is now mainly operated at the Aerospace Engineering and Test Establishment (AETE). Only four (4) aircraft remain in operation and six (6) were put in storage as spare aircraft. All aircraft are equipped with an Operational Loads Monitoring (OLM) system. Only one aircraft is instrumented with strain gauges and the others have only four (4) flight parameters.

The estimated life expectancy may be approved to 2010. These aircraft perform numerous missions varying from executive transport, training, chase plane and more recently as an ejection test bed.

The CT-133 fleet has not been monitored on a regular basis since entry in-service in 1958. An Operational Loads Monitoring (OLM) system is used to track the relative damage experienced by the wing and the carry-through components for comparison to the Lockheed full-scale fatigue test spectra. On-going OLM data reduction is performed with the Bombardier`s General Integrated Fatigue Tracking System (GIFTS) to track the aircraft usage severity. Results are gathered in the form of semi-annual and annual reports and presented to the fleet manager. Since no fatigue baseline exists for this fleet, the Canadian Armed Forces has decided to manage the CT-133 aircraft on a safety by inspection basis.

The wing is believed to be the key life-limiting structural element of the aircraft. Consequently, a coupon test program has commenced for wing critical locations WS 32 and WS 126 to ensure appropriate inspection intervals at these locations. Fracture mechanics and fatigue data on Al 2014-T6 will be determined. In the future, the OLM spectra will be compared to the Coupon Test spectra for aircraft usage tracking. This information will be used to recommend new inspection intervals when necessary.

NFTC Program

J. Dubuc, Bombardier Aerospace

NATO Flying Training In Canada (NFTC) Program is a common project of the Government of Canada and Bombardier Aerospace, Defense Services to provide all flight training, simulator training and ground based training for Canadian and foreign pilots. The CT156 Texan and CT155 Hawk are the aircraft used on NFTC.

Besides the airworthiness considerations, the economic implications play a significant role in the usage monitoring of NFTC fleets. The NFTC program was designed as a dynamic project, able to adjust the training syllabus to specific requirements of the participant countries. Therefore, such potential changes to the pattern of the training missions could lead to loads spectra significantly different of the design usage. Fleet usage monitoring programs should enable to assess the economical impact of such changes. In this respect, the Department of National Defense and Bombardier jointly created the Aircraft Fatigue Management Committee, which was commissioned to reconcile the training requirements with aircraft airworthiness and economical aspects.

CT156 Harvard Fatigue Usage Monitoring

J. Dubuc, Bombardier Aerospace

The CT156 (T-6A-1) structure is designed and certified in accordance with the damage tolerance requirements of the US FAA FAR Part 23 and also complies with USAF ASIP Specifications. The objective of fatigue usage monitoring is to continuously assess the capability of the aircraft to maintain satisfactory strength and stiffness whilst containing insidious damage, on an individual aircraft basis, in accordance with Sec. 23.575 Continued Airworthiness of FAR Part 23.

The aircraft OEM identified six Fatigue Critical Locations (FCL) that require in-service usage monitoring. All these locations have been instrumented with strain gauge sensors and a digital Flight Data Recorder (FDR) installed on each aircraft of the fleet registers strain gauges readings on a mission-by-mission basis. Periodically, FDR data is downloaded for on-ground processing.

Bombardier is currently developing an integrated software system addressing in an automated way the fatigue monitoring activities with their NFTC particularity. The system is already fully functional for the following tasks:

FDR data extraction and transcription;
FDR data validation; and
N_z and FCL load exceedances spectra derivation.

Bombardier is presently in a process of implementation of the remaining tasks (damage calculation, inspection interval prediction) and their integration in order to have a single entry point for all tasks involved in fleet usage monitoring.

CT155 Hawk Fatigue Usage Monitoring

J. Dubuc, Bombardier Aerosapce

The CT-155 structure is designed and certified in accordance with the *safe-life* approach of the United Kingdom (UK) military regulations, DEF STAN 00-970. The CT155 is used by NFTC in two distinct roles: Phase 3 Advanced Training and Phase 4 Fighter Lead-In Trainer. In order to reach the NFTC required safe-life, the mix of two training phases should be in a predefined ratio and the in-service usage spectra should remain in the limits of the design severity.

The aircraft OEM identified eight Fatigue Critical Locations (FCL) requiring in-service usage tracking. All these locations have been instrumented with strain gauge sensors and a digital Data Acquisition Unit (DAU) installed on each aircraft of the fleet registers strain gauges readings on a mission-by-mission basis (Figure 5). Periodically, the DAU data is downloaded for on-ground processing. At each FCL the DeskTop System (DTS) calculates a Fatigue Index (FI) from the peak-valley sequences of the recorded strain gauges using a stress-life approach and the Palmgren-Miner linear cumulative damage rule. Cumulative fatigue index and cumulative normal acceleration exceedances are kept at FCL level and Individual Aircraft IAC level.

To date the system has not yet achieved its projected reliability and, as an interim solution, the periodic usage analyses are based on normal acceleration (Nz) Counter Accelerometer recordings. The significance of these analyses is obviously limited to those FCL for which the fatigue usage is essentially driven by symmetric maneuvers. Nevertheless, N_z exceedances distribution and the associated FI give, at least, a relative measure of the severity of the actual training syllabus.

LANDING GEAR

Airbus A380 Wing and Body Landing Gear

Paul Vanderpol, Goodrich Landing Gear

Goodrich is responsible for the Wing and Body Gears for Airbus A380. Jointly with Airbus the company is tasked with the development of a “mature” landing gear system prior to entry into service, complimentary requirements and specifications and landing gear loads. Goodrich is incorporating new technologies with manageable risk, validated through appropriate development and test:

- High Velocity Oxy-Fuel (HVOF) coatings
- enhanced materials properties
- review of structural analysis methods

Apart from the obvious size challenge the program carries numerous technical challenges. Fatigue challenge includes the use of multiple mission experience based spectrum. Airbus and Goodrich are evaluating the ground loads using a “whole aircraft” model and rational wheel load distribution. The materials are Titanium 10V-2Fe-3Al (~30%), Steel (300M) (~62%) with other materials making up the remaining 8%. It should be noted that working with suppliers Goodrich was able to use elevated minimum strength properties for 300M Steel (Ftu = 1980 MPa (287 ksi)) and for Titanium 10V-2Fe-3Al (Ftu = 1193 MPa (173 ksi)). The figure for Ti is consistent with Mil-Hdbk-5H, but applicable to thicker sections.

In Oakville, Ontario the company has constructed a brand new test laboratory (2200 m^2). The facility will be equipped with fixtures for static, fatigue and endurance test rigs for both wing and body gear test units (Figure 10). Some of the qualification tests planned:

- Drop Test
- Static Strength Test (Limit, Ultimate, and Test-to-Failure)
- Fatigue Test (5 lives)
- Retraction/Extension Endurance Test (1 life)
- Aft Axle Steering Endurance Test (1 life) - BLG
- Lock Mechanism Icing Test
- Dressing System Test
- Actuator Component Structural Test
- Actuator Component Fatigue Test
- Actuator Component Environmental Test

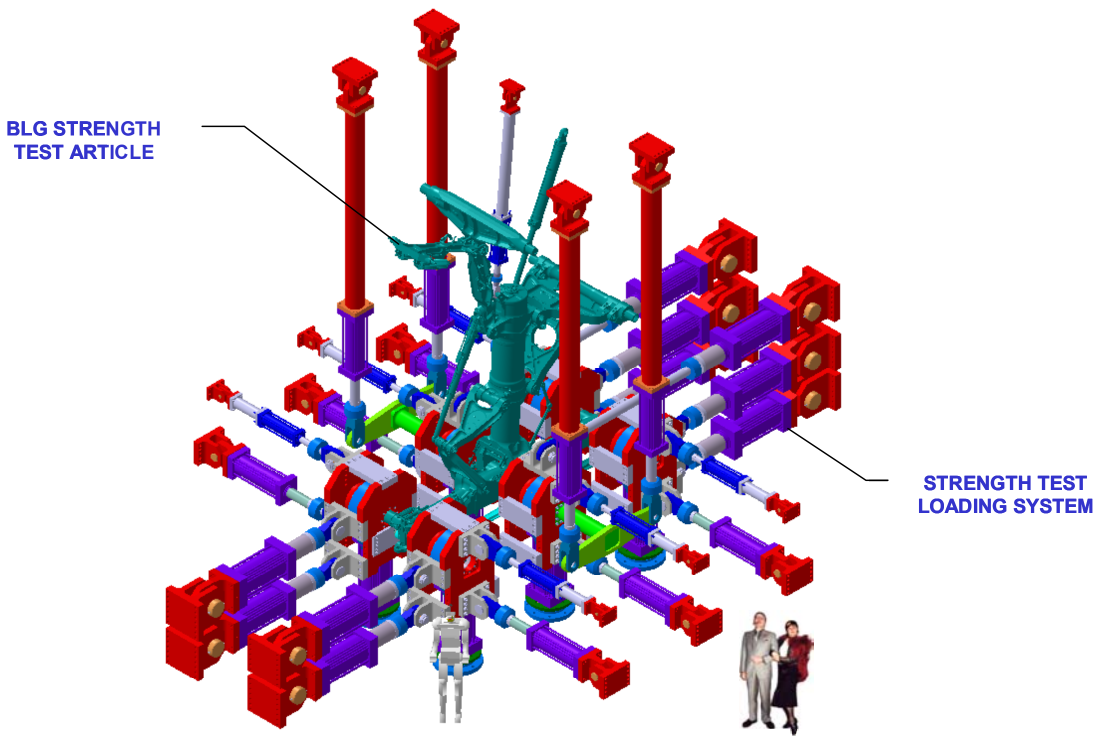


Figure 10. Body landing gear (BLG) strength and fatigue test loading arrangement (note the scale).

Replacement of Chrome Plating with HVOF (High Velocity Oxy-Fuel) coatings
Roque Panza-Giosa, Goodrich Landing Gear

The motivation for replacement of Chrome is primarily environmental - Hexavalent Cr. is a carcinogen with Chrome baths emitting Hex Cr. mist into the air. Chrome level of toxicity is higher than Cadmium and Arsenic and the current permissible exposure limit (PEL) is 0.1 mg/m^3 while for the replacement coating (WC-10Co-4Cr) the levels are reduced to 0.005 or 0.0005 mg/m^3. It is expected that many chrome plating facilities will close, price will increase and the disposal cost will rise.

Not only are WC-10Co-4Cr coating more environmentally compliant but also the are non embrittling, have higher hardness, better corrosion and wear resistance, involve shorter production cycles and a more repeatable robotic control process.

The HVOF is a thermal spray plasma process. Some of the properties of HVOF applied coatings are: high adhesive strength, low porosity, low oxide formation, excellent process for Carbides and Tungsten Carbide coatings, high degree of coating process control, high repeatability (robotics control), minimum excess required for grinding.

Typical coating materials are: Tungsten Carbide 17% Cobalt (WC-17Co), Tungsten Carbide 10% Cobalt 4%Chrome (WC-10Co-4Cr), Nickel Aluminides, Chrome Carbides and Triballoys.

Goodrich is involved in several HVOF activities:
- The HCAT (Hard Chrome Alternatives Team) includes work on the Dash-8 400 MLG Fatigue, high cycle endurance test of an actuator (seal wear) and bending fatigue of simulated pistons.
- HVOF integrity testing including fatigue and
- pin tests – aimed at A380 landing gear.

Fatigue test using hourglass, rectangular K_b and smooth gauge 4340 steel specimens showed significant improvement in fatigue endurance of HVOF coated over chrome plated specimens. The Dash 8 400 main landing gear (MLG) is the first to use HVOF coating (current gear use HVOF to a limited extent). The fatigue test of this MLG has completed 2 life times (of 5 required) without cracking, however elastomeric seals required redesign. The HVOF coated MLG test is shown in Figure 11.

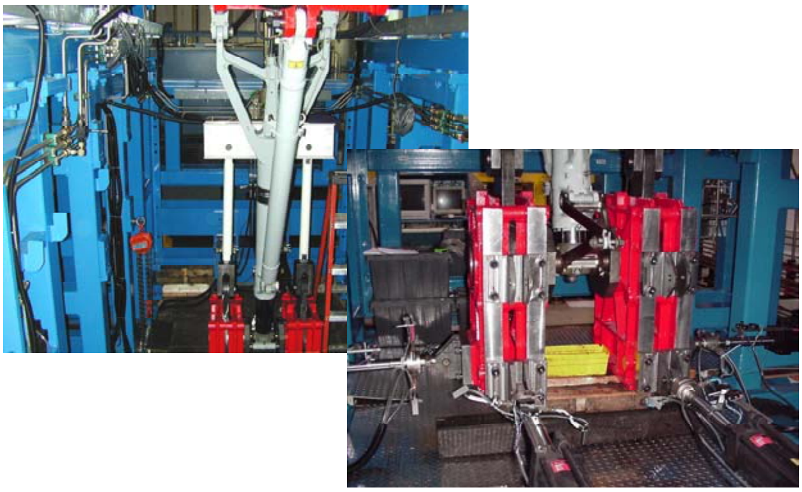


Figure 11. Dash 8 400 HVOF coated main landing gear undergoing fatigue testing.

High cycle testing of an actuator and bending fatigue of simulated pistons have shown that spalling and cracking characteristics are very different for Chrome and HVOF coatings, hence a number of tests are underway to understand this behaviour and reduce the risk on the A380 landing gear.

FATIGUE LIFE PREDICTION

Fatigue Behaviour Under Simulated Airframe Service Conditions

Dr. D.L. DuQuesnay, Department of Mechanical Engineering, Royal Military College of Canada.

The fatigue research group at RMC has been investigating the fatigue behaviour of structural materials used by DND in their military equipment. Of particular interest are aging aircraft. The overall objective is to develop fatigue-life prediction tools to enable the safe and reliable operation of fatigue sensitive structures, often for service extending beyond the original design lifetimes. In terms of specific progress it has been shown that:

The fatigue resistance of metals can be measured in the laboratory using a periodic overload spectrum. The results are insensitive to the magnitude and frequency of overloads, and the method can be applied in either load control or strain control. The former method offers the advantage of higher testing speed, and the latter the advantage of greater precision, especially at high strain levels that induce plastic deformation [3,4,5,6,7,8].

Measurements of crack opening stress levels under variable amplitude spectrum loading in small laboratory coupons have shown that the crack-opening stress level drops immediately after the application of an overload and then gradually increases with subsequently small cycles.

The crack opening stress was modelled using an exponential build-up formula, which is a function of the difference between the current crack opening stress and the steady state crack opening stress of the given cycle. The measured crack-opening stress does not vary greatly from its average value for each load history [3-5].

The experimental fatigue-lives of 2024-T351 aluminum alloy coupons were successfully modelled using the crack-opening stress level corresponding to the load cycle with an occurrence frequency of 1/200. This gave accurate and much improved estimates of average crack-opening stress for the load spectra examined, and gave reasonably accurate fatigue life estimates in comparison to the experimental fatigue lives for the tested coupons [5].

The initial experimental work concentrating on the primary structural alloy of the CF 188 (Hornet) airframe, which is 7050-T74511 aluminum alloy, has been completed. The relevant material properties were measured. Furthermore, it was demonstrated that the models described above provided accurate life prediction for unnotched components of this material compared with laboratory and field data from the IFOSTP program [6].

The combined effect of corrosion damage and fatigue spectrum loading was studied using A CC-130 load spectrum applied to pre-corroded 7075-T6511 coupons. It was determined that very small amounts of corrosion damage produce a significant reduction in fatigue life and that the fatigue life can be predicted using a crack growth algorithm. Methods of modeling the corrosion damage analytically were investigated. It was found that the depth of the corrosion “pit” was the most significant feature of corrosion damage for determining the fatigue life under variable amplitude loading. [7]

The effects of corrosion-fatigue on 7075-T651 aluminum alloy and a CSA G40.16 rebar steel subjected to periodic overload spectra were investigated. The periodic overload spectra were identical to those used for testing these materials in laboratory air. For the 7075-aluminum, two types of corrosive environments were examined (a) complete immersion in re-circulated aqueous 3.5% NaCl (ASTM simulated seawater), and (b) flow past atomized aqueous 3.5% NaCl (simulated salt fog). For the steel, only method (a) was examined because of equipment failure of the apparatus used for method (b). It was found that the simulated salt fog environment reduced the fatigue life and fatigue limit strength by a factor of about three for the 7075-alloy whereas complete immersion reduced the fatigue life by an order of magnitude. The intrinsic fatigue limit disappeared under all corrosion-fatigue conditions, meaning that ALL cycles in a load spectrum potentially do fatigue damage. These results present some serious issues that must be addressed during fatigue life assessment of structures, including DND aircraft fleets, in service in maritime and similar salt-exposure environments. The results for the steel have been submitted to an international conference [8], and a full paper is being prepared for peer-reviewed publication.

Corrosion Fatigue Analysis for CF18 Longeron Using Holistic Life Assessment Method.

Min Liao, SMPL, IAR, NRC

During the IFOST FT-245 test (see previous section), a long fatigue crack of 46 mm (1.81 inch) was discovered on the right hand upper outboard longeron (UOL) (AA 7149-T73511) of the centre fuselage at 2932 simulated flight hours (SFH), as shown in Figure 12. Corrosion pits were found in the area near the apparent crack nucleation site following paint and filler removal. In this test, the fuselage is used as a transition structure, and was obtained from a retired United States Navy (USN) F-18 aircraft that was in-service for ten years (1984-1994). Since the full-scale wing test is being conducted in an environmentally controlled facility, it is reasonable to assume that the corrosion pits went undetected after the last maintenance cycle prior to the aircraft being retired. The life of the UOL was thus composed of two parts, the first part was the life served in USN (1984-1994) which had the interaction effects of corrosion and fatigue; the second part was the life in the wing test which had fatigue effects only. A corrosion fatigue analysis was carried out to reproduce the life of the UOL [9]. This analysis used the holistic life assessment method (HLAM) and an HLAM-based computer code ECLIPSE (Environmental and Cyclic Life Interaction Predictions for in-Service Evaluations, by Analytical Process/Engineered Solutions (APES) Inc.) to take into account the interaction effects of corrosion and fatigue. Some key inputs for ECLISPE in this analysis were as follows.

- Cracking scenario – crack was nucleated from corrosion pits at the round corner of the UOL.
- Corrosion pit growth rate – described as the cube root power law, $d = C(T)^{1/3}$.
- Material models, initial discontinuity states (IDS) and crack growth rate data – IDS (as-manufactured) was estimated as 0.0073 inch using the crack growth data of a non-corroded UOL used in previous full scale CF-18 fuselage test (FT55) conducted at Bombardier Aerospace.
- The crack growth rate data were originally taken from the NASGRO material database and modified by APES to include small crack growth.
- Thickness loss effect – no thickness loss was found on the UOL, but the actual thickness of the UOL along the crack path was accurately measured using an ultrasonic technique.
- Loading spectra – determined from the strain gauges used in the wing test, which included 46,588 cycles, i.e., 326 SFH in one block.

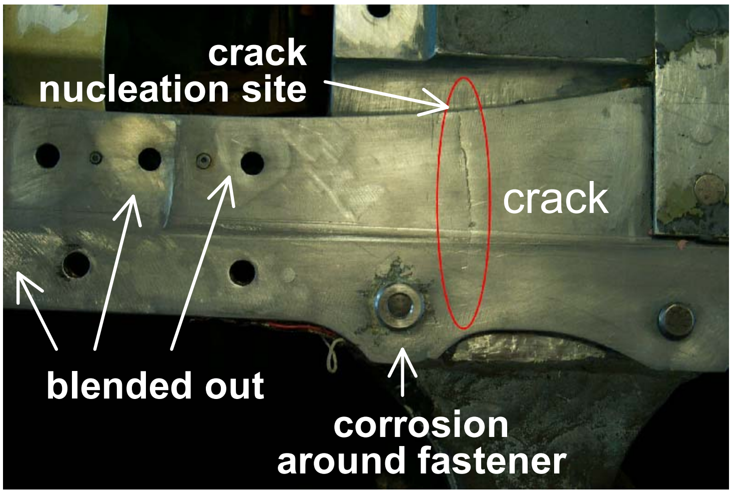
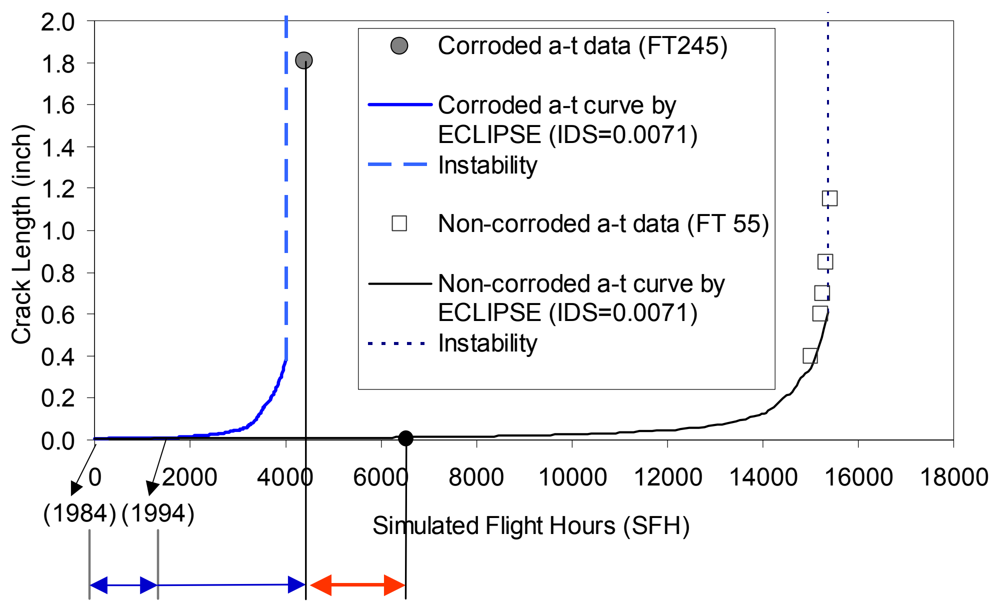


Figure 12. Fatigue crack on the right UOL of F18 central fuselage at 2932 simulated flight hours.

The analysis was carried out from the start of service life (1984) to when the crack was found during the wing test in 2001. It should be noted that this fuselage had previously accumulated 3,644 flight hours with a recorded FLEI (Fatigue Life Evaluation Index) of 0.309 in the USN (1984-1994). This usage converted to the wing test hours is approximately 1,446 SFH. The reduction is due to more aggressive loading spectrum applied to the wing test. The results from the analysis are shown in Figure 13, along with the test results for the corroded (FT245) and non-corroded UOL (FT55). The relative error between the analytical and test results is - 9%, even with no reduction to previous flight hours, the relative error is about - 40%. The good agreement between the analytical and test results indicated that the HLAM/ECLIPSE is a promising analytical tool for corrosion and fatigue analysis of aircraft structures.



1446+2932= 4378 (~6576), relative error =9%-39%.

Figure 13. Comparisons of the analytical and test results.

Participation in Corrosion Fatigue Structural Demonstration Program (CFSD)

G.F. Eastaugh, SMPL, IAR, NRC

The NRC/IAR has been a major participant in Phases 1 and 2 of the Corrosion Fatigue Structural Demonstration Program initiated by the US Air Force Research Laboratory in 1998. Lockheed Martin Aeronautics Company, Marietta is prime contractor for Phase 2, which will be completed during 2003. The program has started the development and transition into USAF and USN service of the tools and procedures needed to evaluate the risk of corrosion to the structural integrity of aircraft. Experience has shown that corrosion might exist for several depot maintenance periods before detection and can in some cases substantially accelerate fatigue crack nucleation and early growth. The tools and procedures developed in this program and successor programs will also provide maintenance managers with the flexibility to defer some corrosion maintenance to improve efficiency and aircraft availability. The basic corrosion fatigue durability (life) assessment method being used by the CFSD team is based on a holistic structural integrity design framework (also referred to as holistic life assessment methodology). This assessment method expands current USAF durability assessment philosophy and takes account of all significant, identifiable interactions between corrosion and fatigue. NRC has undertaken several experimental tasks needed to design and validate elements of this holistic software framework. These are summarized below.

To support the development of the fundamental fatigue durability model, several common aircraft materials - 2024-T3 bare and clad, 7075-T6 bare, 7075-T6511 anodized and painted, and 7079-T6 clad anodized and painted - were examined and tested to determine and characterize the internal discontinuities that are pertinent to fatigue crack nucleation and early growth. All materials were nominally non-corroded, but some were removed from aircraft and so had experienced some operational service. Fatigue tests in controlled humidity followed by post-fracture analysis were performed. From the post fracture analysis, three types of discontinuities were clearly found to be responsible for crack nucleation in the aluminium alloys studied: constituent particles, coating layers (clad and/or anodized), and extrinsic features (e.g. scratches, pits, and machining marks). Figure 14 shows an example of results for non-coated samples. It indicates that the particles that acted as crack origins were at the high end of the particle size distribution obtained through metallography and that they represented only a very small fraction of the particle population.

Many cases of corrosion occur early in a component's fatigue life. In these cases, the corrosion damage might be the primary feature that determines the site of nucleation and the fatigue life of the component. To obtain data to model nucleation and early crack growth in such circumstances, a series of tests were carried out to determine the modified discontinuity states (MDS) that caused fatigue cracks to nucleate in corroded skins. The tests were carried out on coupons fabricated from both 2024-T3 and 7079-T6 naturally corroded skins and subjected to different levels of humidity and stress. Most of the modified discontinuity states that were present were corrosion pits that varied in size but some were intergranular corrosion. The results from the different fatigue tests suggested that both high levels of relative humidity and stress lead to the early onset of crack nucleation for both the 2024-T3 and 7079-T6 pre-corroded materials. That is, as either the relative humidity or stress level increased, the number of cycles to failure decreased independently of the level of corrosion present. The results also showed that given the same test conditions (same relative humidity and stress level) the number of cycles to failure did not change significantly as the corrosion level increased. For the 2024-T3 specimens, the test results also suggested that both the relative humidity and stress level could affect the size of the fatigue-related MDS value. The tests showed that as the relative humidity increased, the size of the modified discontinuity state from which a crack could grow decreased, slightly. Also, the higher load levels triggered smaller modified discontinuity states at multiple sites as shown in Figure 15. Although corrosion was present along the faying surfaces of the lap joints, a number of the coupons failed due to cracks nucleating at the non-corroded clad layer (outer surface of the lap joints). This failure location suggests that the MDS present along the corroded gauge length in some of the coupons evolved slower than the DS associated with the cladding.

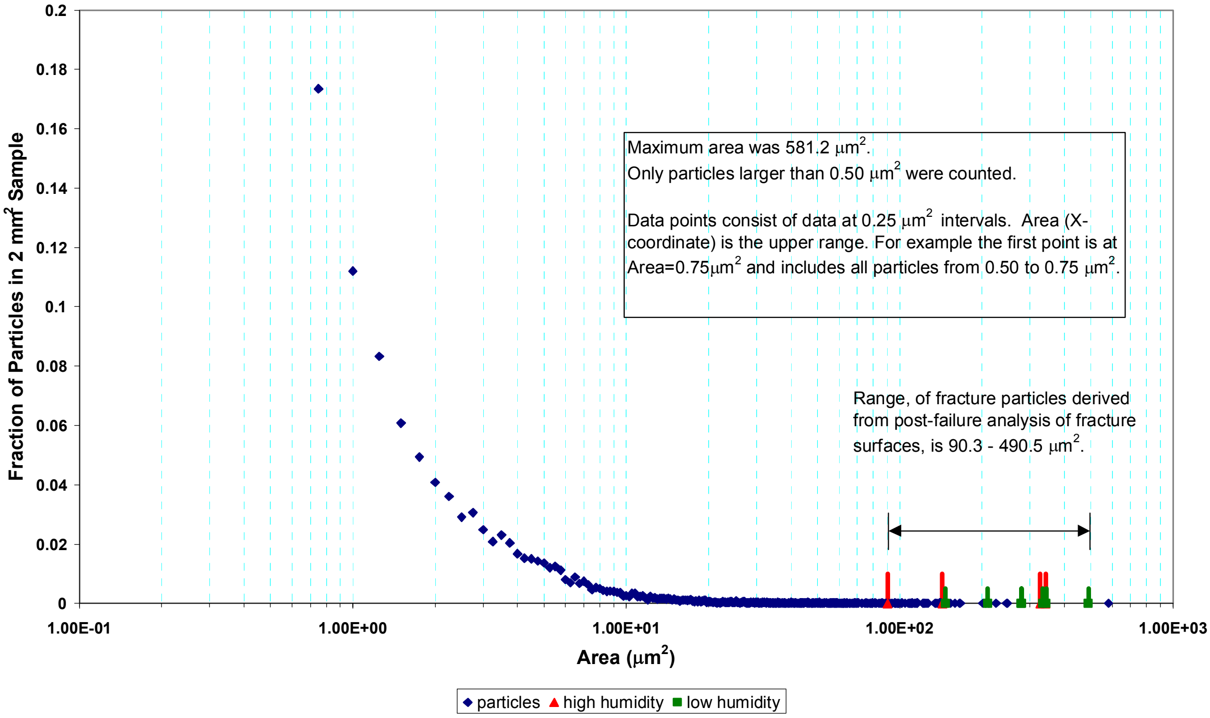


Figure 14. The area distribution of particles in the ST plane of new, thin (0.063”) 2024-T3. The particles identified in red and green vertical lines are the particles found from the fracture surfaces of the high and low humidity fatigue tests, respectively.

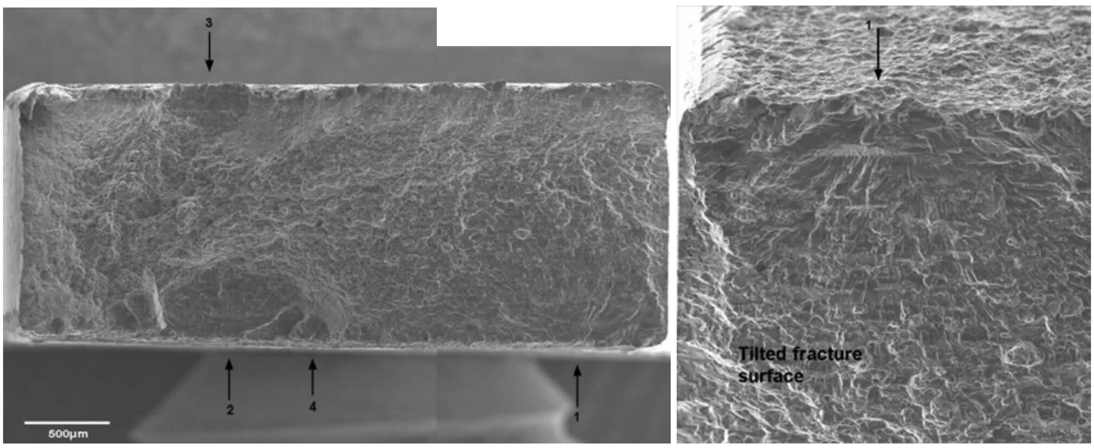


Figure 15. Coupon with 2.67% thickness loss. Multiple nucleation sites were present along the corroded faying surface (1,2,4). The predominant crack originated at a pit (1) as shown in the tilted fracture surface image. A smaller crack also nucleated from the outer surface (3).

Many cases of exfoliation corrosion have been found in aircraft wing structure, but attempts to model the nucleation and growths of fatigue cracks at such damage are in the early stages. To help provide the necessary experimental data, a study that was carried out to determine the effect that exfoliation corrosion has on the residual life of C141 aircraft wing skins, which are fabricated from extruded 7075-T6511 alloy. The results from the different tests that were carried out suggested that relative humidity did not have an effect on the fatigue life of the exfoliated specimens. The relevant data is in Figure 16. Although exfoliation was present in the majority of the specimens, a number of the coupons failed due to the crack nucleating from a discontinuity present near the corner of the specimen. These discontinuities included intrinsic particles and manufactured damage (machining marks, indents and scratches). The corner failures occurred due to a lack of a stress concentration (hole) in the vicinity of the exfoliation. For those specimens that failed in the exfoliated region, the crack nucleating features were corrosion pits and intergranular corrosion.

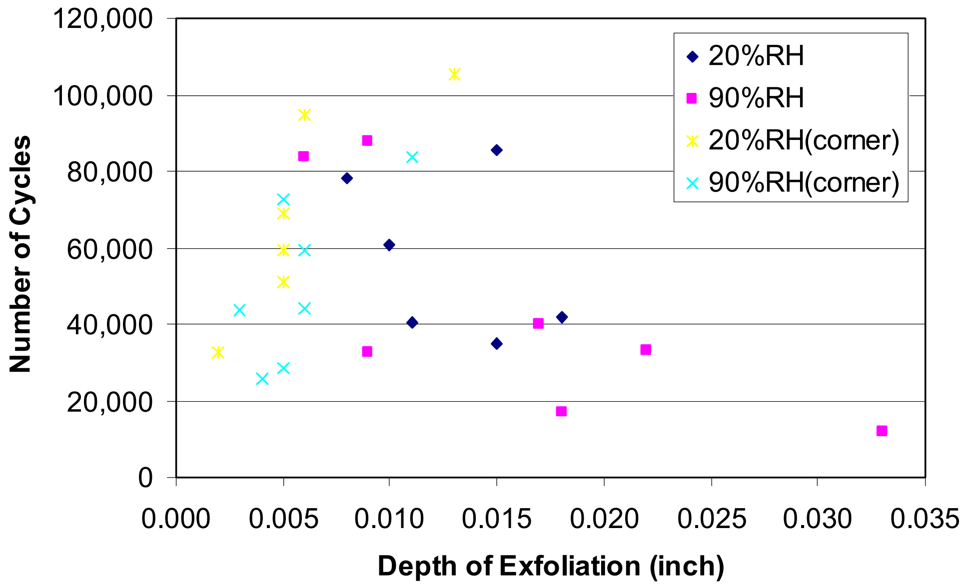
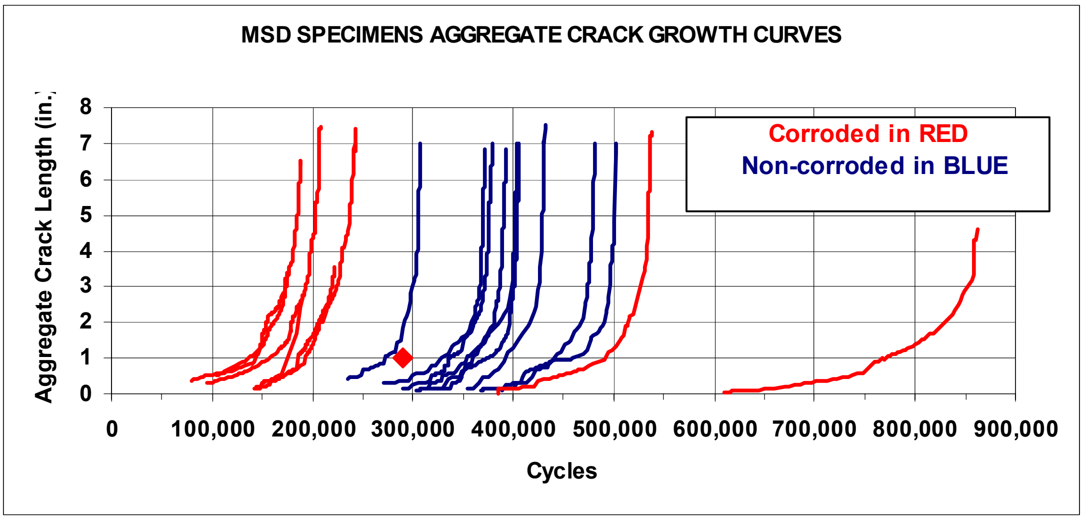


Figure 16. Graph showing maximum depth of exfoliation versus cycles to failure for tests carried out on specimens fabricated from exfoliated 7075-T6511 upper wing skins. The depth of the exfoliation was estimated using ultrasonic NDI. Some specimens failed at the corner instead of at exfoliation damage.



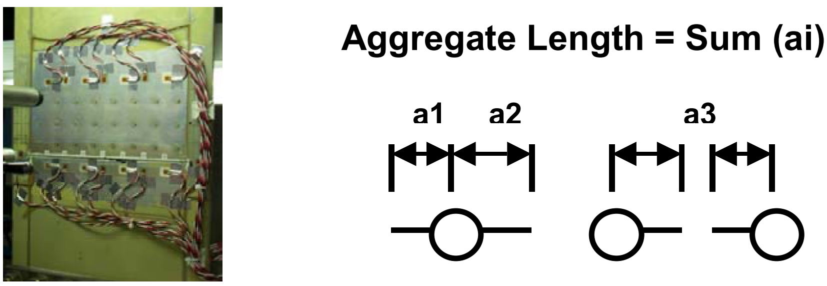
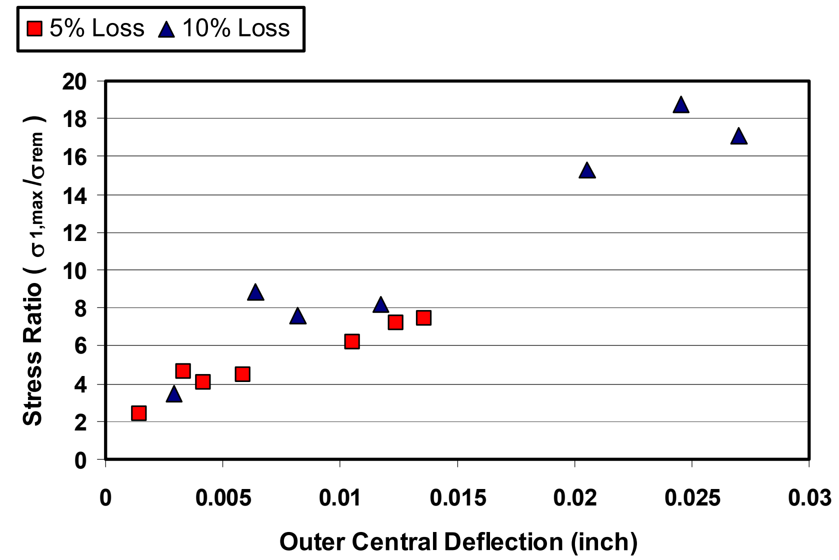
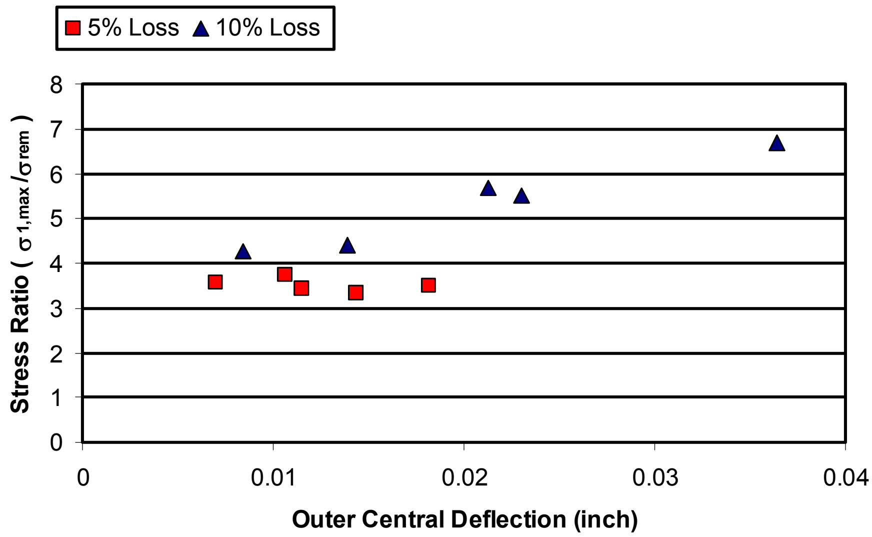


Figure 17 Aggregate crack growth curves for fatigue testing of skin splice specimens by NRC Canada.

One of the candidate categories of structure for which the CFSD project is developing a corrosion fatigue durability assessment methodology is fuselage skin splices. Such splices are fatigue critical and are one of the most widespread structural details on transport aircraft. Several types of coupon and element testing, including some of the testing mentioned above, are being used in the CFSD project to develop and validate the methodology for fuselage skin splices and comparable structural details. As part of this process, fatigue tests have been performed on a splice element in 2024-T3 alloy that is generally representative of the single shear (lap) splices on KC-135, JSTARS, C-130, and P-3 aircraft. This test specimen also allows the natural development of MSD. Several splice elements were pre-corroded in the laboratory and then fatigue tested. A control sample of non-corroded splice elements was also fatigue tested. The test program is being used to define the fatigue and corrosion fatigue failure mechanisms in fuselage splices for modeling purposes and to provide an element-level validation of the modeling strategy. The basic fatigue test results were published in [10]. Corrosion caused a substantial reduction in fatigue life and changed the failure mode. The reduction in life is illustrated in the aggregate crack growth curves in Figure 17. The paper included a comparison between the fatigue test results and life assessments of corroded and non-corroded splice elements made using the CFDA methodology and software. The analytical life assessments were made by another CFSD team member, AP/ES Inc., St. Louis, using their ECLIPSE software.



(a) Rivet spacing ratio of 1.0



(b) Rivet spacing ratio of 2.0

Figure 18. The relationship between outer central deflection and maximum stress in lap joints with different rivet spacing ratio.

Corrosion has several effects on a fuselage lap splice. One of these is the high static bending stress at the faying surface in the vicinity of rivets caused by the build up of corrosion products – known as pillowing stress. NRC has previously estimated these stresses using finite element models. In CFSD this work was continued. The aim was to develop generic tables and graphs for estimating non-corroded stress and pillowing stress from knowledge of splice geometry and nominal (far field) stress. The target aircraft was the KC-135, which has a large number of variations in basic splice geometry throughout the fuselage. The models took into account the pre-stress caused by the rivet-fastening process, the stress resulting from the internal pressure (hoop stress) and the corrosion pillowing caused by the presence of the corrosion product. The results indicated that the stress state is strongly influenced by the joint configuration (both rivet spacing and skin thickness) and so it was not possible to create the generic look-up data originally intended. However, it was discovered that it may be possible to determine the maximum stress in a joint given the maximum out-of-plane displacement, which can be measured using non-destructive inspection (NDI) techniques. The approach is illustrated in Figure 18.

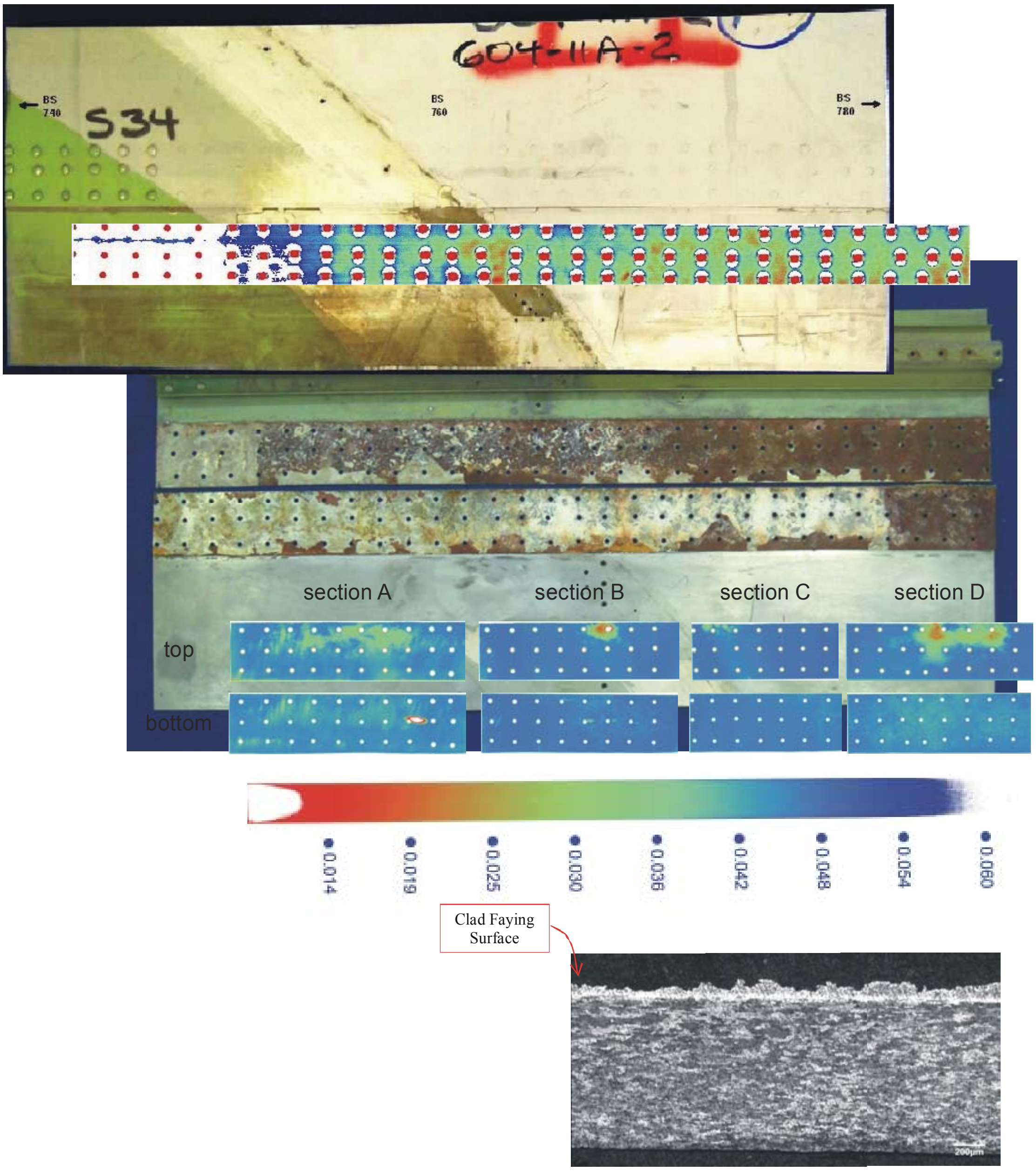


Figure 19. Examples from top to bottom of a service-retired lap joint, NDI results, a dismantled lap joint, x-ray measurements of the thickness of the layers of the joint, and a micrograph of the corrosion damage on the faying surface of the joint.

It is intended to use the corrosion fatigue modeling procedures developed in CFSD in much that same way that fatigue durability and crack growth models are currently used in determining inspection methods and frequencies. Therefore, it is necessary to develop compatible NDI procedures. In this case, the NDI procedures must be able to measure corrosion damage, and not just detect its presence. An important part of NRC's CFSD work has been to investigate various methods of measuring hidden corrosion. Fuselage lap joints are one of the most widespread and challenging categories of components in this context, and so they were used as the vehicle for the study. Service-retired lap joints were inspected and dismantled. Corrosion damage was measured after disassembly through digital x-ray techniques and metallographic examinations. The NDI results were compared to the teardown in order to determine the performance of NDI on these specimens, and to investigate correlations between NDI and structurally significant metrics of the corrosion damage. Figure 19 shows these steps pictorially in order: the selection of a service-retired lap joint, NDI, teardown and cleaning, x-ray measurement of remaining thickness, and metallography.

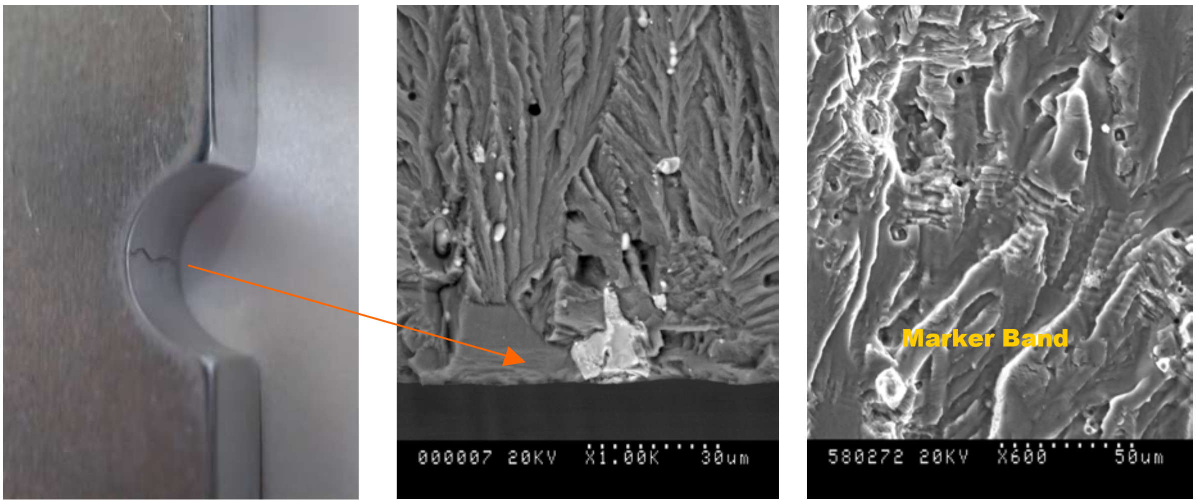
The work performed by NRC is being integrated with that performed by several other subcontractors, including AP/ES Inc., University of Utah, Northrop Grumman, and SAIC, and will be published by the prime contractor Lockheed Martin Aeronautics.

FRACTURE MECHANICS AND CRACK PROPAGATION STUDIES

Short Fatigue Crack Growth Behavior of 2024-T3 Aluminum alloys

Ali Meratri, SMPL, IAR, NRC

Nucleation and short crack growth makes up a majority of the fatigue life in aluminum alloys. In the short crack growth regime, local microstructural features such as grain boundary and size, inter-particle spacing, and texture play the primary roles. This study has recently begun in order to evaluate the various crack detection and monitoring techniques in short crack regime in AA2024-T3. Fatigue samples were also prematurely overloaded at different stages of the fatigue life in order to help develop a more basic understanding of the microstructural features that influence the growth rate and pattern of short cracks in the aluminum alloy. Figure 20 shows a fatigue crack in a single-edge notched sample nucleated from a particle as well as marker band on the fracture surface.



a) b) c)

Figure 20. Crack nucleation and early growth: a) small crack in SEN fatigue coupon (2024-T3), b) fracture surface showing a crack nucleation site, which is at a constituent particle; b) one of the marker bands used.

Prediction of the Shape of Fatigue Cracks Propagating in Thick Monolithic Aluminium Structures using Composite Bonded Doublers

Y. Bombardier, Martec Limited

Bonded patch repair technology can be used to extend the life of aging aircraft fleets and assess airworthiness of repaired structures. This technology has proven to be effective by changing load paths to avoid cracking without creating new damage in the structure; a common problem in metallic repairs.

Fracture mechanic data, such as Stress Intensity Factors (SIF), are required to determine the crack growth rate and the subsequent residual fatigue life from an initial flaw. Methodology involving finite element models was developed in the past by Martec Limited to predict the crack growth in structures with bonded repairs. Inclusion of crack shape and the direction of crack growth in the current Martec methodology are expected to predict residual life with more accuracy. The following effects are also introduced in the improved methodology: thermal residual stresses, and geometric non-linearity for single-sided patch. This research project is conducted in collaboration with the Canadian Department of National Defence, Martec Limited and Sherbrooke University. Crack shape obtained experimentally is shown in Figure 21.

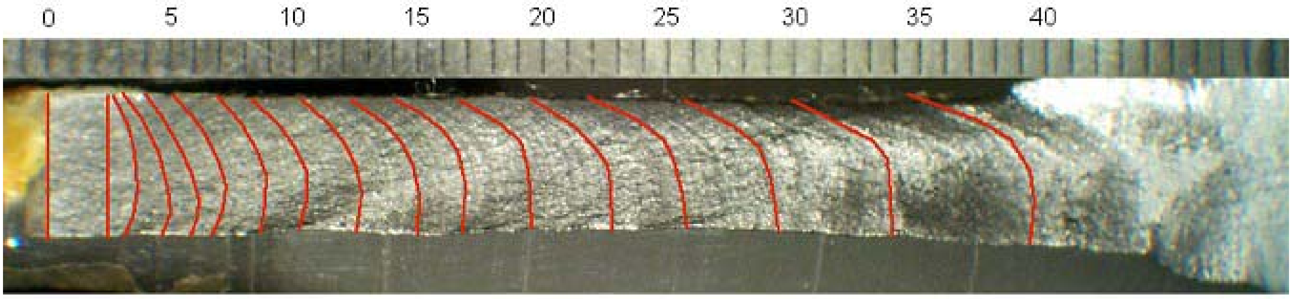


Figure 21. Influence of bonded doubler on crack shape.

Fatigue Crack Growth Prediction and da/dN Testing of Notched GLARE

X. Wu, SMPL, IAR, NRC - (Collaboration between Bombardier Aerospace and NRC)

A fatigue and fracture prediction model has been developed for fiber-metal laminates, and fatigue crack growth behaviors GLARE-3 and GLARE-4 in different orientations have been studied. The model is based on the higher-order theory that considers antiplane shear stresses in laminated panels under generalized plane stress conditions [11]. In the presence of cracks, these interlaminar stresses transfer the stresses that are lost at the crack surface in the metal layer to the adjacent prepreg layers. With the introduction of a new stress potential $\chi(z)$ and the use of the Westergaard stress potential function, $Z(z)$, where $z = x+iy$ is the complex coordinate number, a closed-form solution of the crack-tip displacement/stresses field was obtained with the consideration of the proper displacement compatibility conditions around the notch. From the two stress potentials, $\chi(z)$ and $Z(z)$, inter-laminar stresses at the interface of metal and fiber-reinforced prepreg as well as the in-plane stresses in each different layer can be obtained. A closed-form expression of the effective stress intensive factor for fatigue cracks in FML has been derived.

Fatigue crack growth rate (FCGR) testing has also been carried out on center-cracked tension speciemens of GLARE-3 at angles of 0^0, 15^0, 30^0, 45^0, 90^0 with respect to the rolling direction. Apparently, there is an orientation dependence in the FCGRs of GLARE: 0^0 and 90^0 are equivalent, but as the crack deviates, crack growth rate increases, with the highest at 45^0 at the same ΔK level. FCGR testing on GLARE 4 is also nearly completed.

At this point, the theoretical model has only been used to describe the experimental observations for 0^0. A good agreement was found in the comparison between the analytical model and test results. Analysis for the other orientations is underway.

FAILURE INVESTIGATIONS

Fractographic Investigation and Determination of the Rate of Growth of Cracks from the F/A-18 IFOSTP Full Scale Center Fuselage and Wing Tests

M. Roth, QETE, DND

QETE is performing detailed fractographic investigations to determine the rate of growth of cracks found during the F/A-18 IFOSTP full scale fatigue testing (see previous sections). During the test spectrum of variable amplitude loads simulating 325 flight hours is applied repeatedly until the desired life is reached. Knowledge of the crack growth rates is required to establish appropriate inspection intervals, maintenance actions, or modifications.

Fatigue crack surfaces are characterized by features such as striations and/or beach marks. In variable amplitude loading, striations will be associated with the higher loads. Ideally, two consecutive spectrum blocks will leave similar striation patterns. If those repeating patterns can be identified, the distance between them will provide the crack growth per spectrum block. AMRL in Australia has found that it was often possible to observe repeat patterns from the crack origin to the end of the crack using optical microscopy. QETE has been applying the same technique to determine the growth rate of cracks from the Canadian tests. Figure 22a to to d illustrate the types of observations and findings in the case of a small front spar crack (Figure 22a). Close to the origin, high magnification examination reveals a number of closely spaced lines or narrow bands (Figure 22b). The spacing between the lines, i.e. the crack advance per block, increases steadily. As the crack becomes deeper, an increasing number of the higher loads in a block form visible striations, but usually consecutive blocks leave similar sequences (Figure 22c). The coordinates of the repeat patterns are recorded sequentially starting as close to the origin as possible and travelling in as much a straight line as possible to the end of the crack. The crack depth is then plotted as a function of simulated flight hours (SFH) or blocks (Figure 22d). In an ideal world, one repeat pattern should be observed for each block from the origin to the crack front. However, because of damage to the crack surfaces only partial sequences are often recorded (Figure 22d).

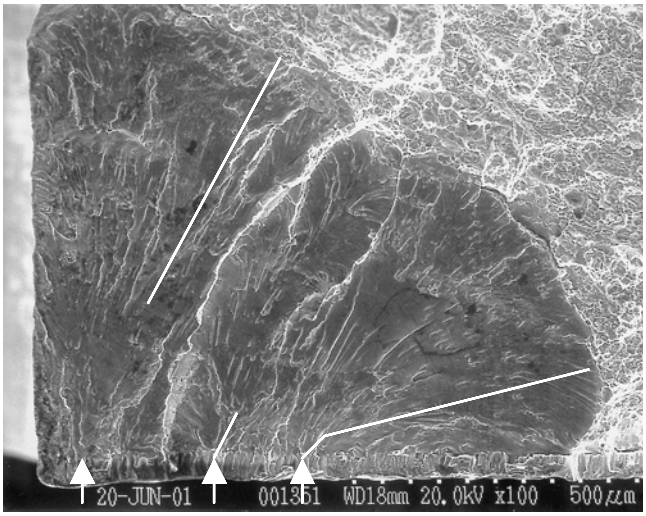


Figure 22a. Overall view of the crack. The arrows point to the primary, secondary and tertiary initiation sites going from left to right. The white lines correspond to the locations along which crack growth rates were measured (run A, run B, and run C going from left to right).

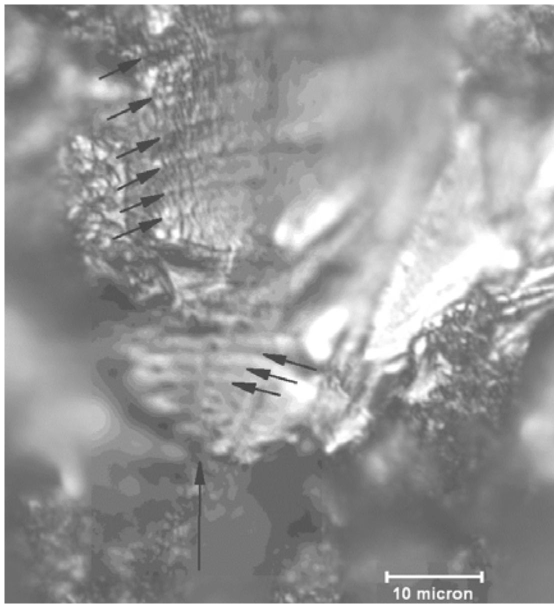


Figure 22b. The arrows point to repeat patterns close to the secondary origin at an etch pit (vertical arrow).

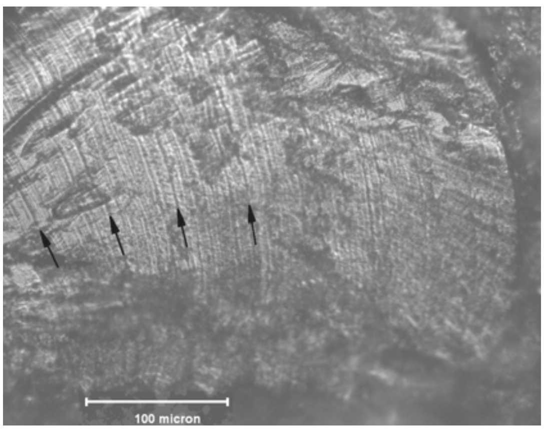


Figure 22c. The arrows point to repeat patterns in the middle of the tertiary crack.

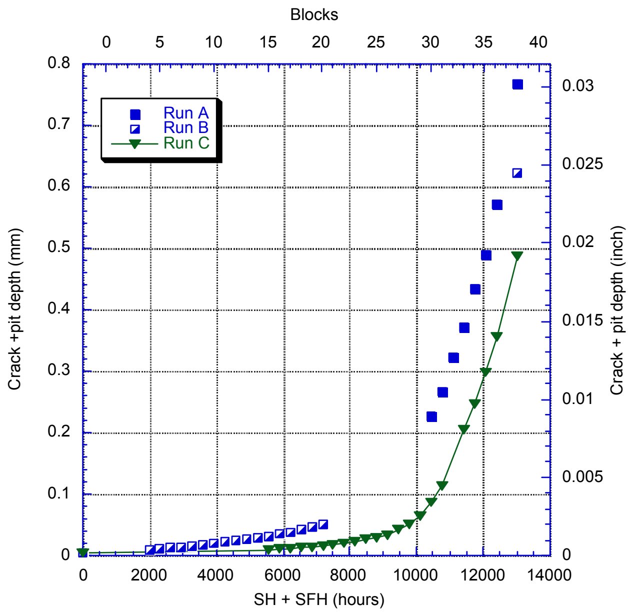


Figure 22d. Crack growth curves for the repeating patterns observed on the primary (Run A), secondary (Run B), and tertiary (Run C) crack surfaces with the crack plus pit depth plotted linearly.

Center fuselage test and wing test cracks from the following main areas have been investigated so far or are being investigated:

- Center fuselage - Y488 bulkhead [12], Y453 bulkhead, and longerons
- Wing - Front spar [13], intermediate spars [14], intercostals [15], inboard leading edge flap [16]

The main findings were that:
- Cracking often nucleated very early in the component life at small surface pits, which were formed during the etching step preceding the deposition of the ion vapour deposited (IVD) aluminums corrosion preventative coating applied to the 7050 aluminums alloy of the F/A-18 structure.
- Cracking occasionally nucleated at corrosion pits in holes of the wing, which had 702 flight hours prior to its use for the center fuselage fatigue test.
- Crack propagation was initially slow and then increased markedly usually between 6,000 and 10,000 simulated flight hours (Figure 22).

Teardowns for the P-3 Service Life Assessment Program (SLAP).

M. Roth, QETE

Canada, the US Navy, Australia and the Netherlands have embarked on a collaborative program known as the P-3 Service Life Assessment Program (SLAP) involving full-scale fatigue tests of the P-3 aircraft with the aim of extending the economic life of the P-3C and CP140 fleet to the year 2025. Apart from the development of loads Canada's commitments to this project includes:

- Fatigue testing of components simulating critical areas of the wing (WS167 and BL65) at NRC (reported here later), and
- The teardown of the right-hand wing of a RNZAF P-3B (Kestrel wing) and the right-hand wing of the full-scale fatigue test article at the completion of the test.

The results of these activities will aid in identifying fatigue related life-limiting factors and will help determine the most economic way of extending the life of the aircraft. ATESS has been tasked with both of these teardown activities. The teardown of the Kestrel wing began in April 2002 and was to be complete in March 2003. The teardown of the test article wing is scheduled to begin in May 2003 with the majority of activities completed by September 2003. The work on each wing section will include visual inspections, in-situ NDT inspections, teardown activities and post teardown NDT inspections. Fractographic examination of the cracks found will be performed by QETE. DTA and Martec Ltd. are developing teardown database.

PROBABILISTIC AND RISK ANALYSIS METHODS

Effect of Discontinuity States on the Risk Assessment of Corroded Fuselage Lap Joints.

Min Liao, SMPL, IAR, NRC

A new corrosion management approach has been proposed with the intent of anticipating, planning, and managing corrosion, which stands in sharp contrast to the present 'find and fix' philosophy. This new philosophy uses the holistic life assessment method (HLAM) to estimate the entire life cycle by addressing the interaction effects of corrosion and fatigue. The major parameters of HLAM include 'initial discontinuity state' (IDS), which describes the as-produced or as-manufactured state of the material to establish the initial analysis condition, and 'modified discontinuity state' (MDS), which describes the subsequent state of the material after it has evolved under cyclic and environmental mechanisms. To characterize the IDS and MDS presented in pristine and corroded fuselage lap joints, coupon specimens were fabricated in a previous project from pristine, artificially and naturally corroded lap joints containing three corrosion levels, 0%, 2%, and 5% average thickness loss [17]. With the aid of a SEM, the discontinuity states (DS) data were measured from the crack nucleation sites on fracture surfaces of coupons after fatigue tests, and characterized as the semi-circular cracks.

Research was carried out to investigate the statistical characteristics of the DS data, and effect of the DS distributions on the risk assessment of fuselage lap joints that contain multi-site damage and corrosion [18-19]. To accomplish this, a k-sample Anderson-Darling test was conducted to determine whether or not there was a significant difference between the natural and artificial corrosion MDS data. The results indicated there was no significant difference in MDS data and thus it could be assumed that the damage resulting from the artificial corrosion process was similar to that produced by the natural process. Consequently, the natural and artificial corrosion MDS data were pooled together to get a larger sample size (Figure 23). The best-fit distributions for the pristine and corrosion DS data were obtained using the Anderson-Darling goodness-of-fit test as well as probability plots (Figure 24). Based on the DS distributions and previous crack growth analysis, a risk analysis was performed in fuselage lap joints using the code, PROF (PRobability Of Fracture) from the United States Air Force (USAF). The analytical results were in good agreement with the test results obtained from a previous project on fuselage lap joints containing multi-site damage and corrosion (Figure 25). It was found that the DS distribution had a strong influence on the risk analysis results, and that corrosion, even at a low thickness loss level, could significantly increase the risk level of fuselage splices failure (Figure 25).

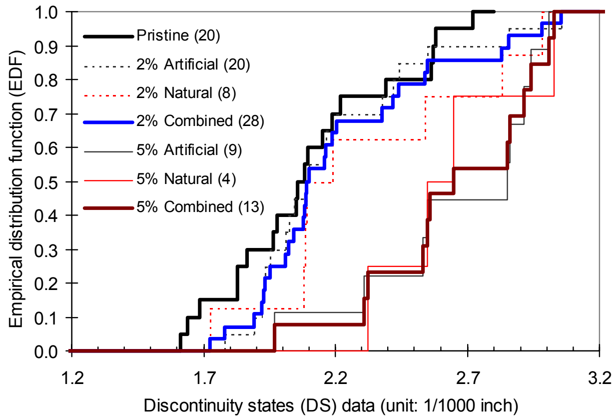
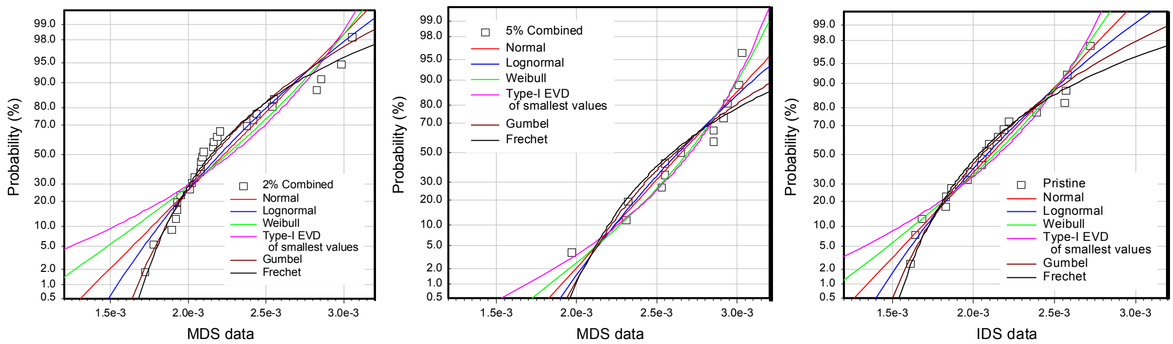


Figure 23. Empirical distribution function (EDF) of the DS data, ( ) denotes sample size.



(a) (b) (c)

Figure 24. Probability plots on normal probability papers, (a) pristine IDS data; (b) combined 2% MDS data; and (c) combined 5% MDS data.

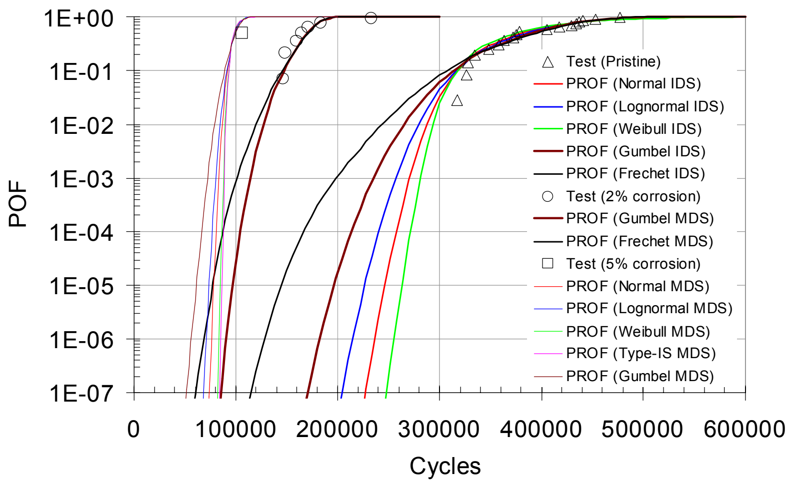


Figure 25. Probability of failure predictions for pristine and corroded lap joints using the *acceptable* DS distributions.

AGEING AIRCRAFT ISSUES

CF-188 Aircraft Structural Integrity (ASIP) and Life Extension (ALEX) Programs
J. Dubuc, Bombardier Aerospace

Bombardier Aerospace, Defence Services (BADS) has been under contract by the Canadian Forces since 1987 to conduct System Engineering Support (SESC) on their CF-18 fleet. This contract includes the conduct of the ASIP program and of all related depot level structural maintenance. Since 1998, a major life extension effort has been put in place under the ALEX program in order to ensure that the aircraft could reach its original design life of 6,000 hours, under a more stringent fatigue scatter factor of 2.5, in lieu of the original factor of 2. Under this program, Bombardier performs all the non-recurring engineering (NRE) development as well as the fleet-wide installation of structural modifications, in its Mirabel, Quebec facility. ALEX is a three-phases progressive program, which will span until 2010 at least.

In 2001 and 2002, the so-called ALEX CP1 phase has been in full production mode with over 29 discrete modifications. More than half of the fleet has been inducted through this program already. The NRE effort has been targeted at progressing towards the CP2 phase, which is to be in full production starting in May 04 when fleet leaders will be going through their second pass. At this time, the CP2 program includes 23 modifications and is still growing with the on-going ASIP effort. Concurrently, a first prototype has nearly been completed for the Center Barrel Replacement (CBR) program. Under this variant of ALEX, this sub-assembly, which contains the wing-carry-thru bulkheads is replaced in order to reduce airworthiness risks further on the fleet leaders. BADS is also currently defining modification programs for the RAAF under their Structural Refurbishment Program (SRP) based on ALEX and CBR. A first prototype of the first phase SRP1 will be conducted in Mirabel, Quebec in 2003.

The majority of the ASIP effort in 2001 and 2002 has been dedicated to defining the ALEX CP2 program based on OEM and/or IFOSTP test results and to a lesser extent fleet failures. Given its scope and complexity, Bombardier has adopted a stage-gate approach to this task. In this approach, the 1500 potential critical areas are grouped under 25 different certification sets which are geometrical entities sharing common loading mechanisms. The stages or phases of the approach are:

- Preparation phase to assess available data and engineering tools for the certification set
- Preliminary assessment phase to list and categorize all potential critical areas with respect to significance and level of maintenance (base versus depot)
- Test interpretation phase to compute safe life limits based on test and fleet results
- Airworthiness risk assessment phase to ensure prescribed modification or inspection ensures proper mitigation of risk
- Documentation phase to ensure proper trail of data and decisions in Structural Information System (SIS) and create Structural Maintenance Plan (SMP)

The test interpretation phase entails a variety of engineering tools such as finite element models (FEM), strain gauge data from full-scale and flight test as well as fatigue and crack growth prediction software. Bombardier uses the CI89 and CG90 software acquired from the OEM in the late 1980’s, integrated in an in-house developed software, Specgen. Specgen integrates all spectrum generation utilities and common stress concentration (or intensity) solutions under the same umbrella along CI89 and CG90. It allows easy access to a variety of design and test spectra and provides for linear combinations for mixed loading. In benchmarking performed with complex loading cases, Specgen yielded timesavings in order of a 20:1 ratio compared to the non-integrated software suite.

Under ASIP, Bombardier also conducts periodic fatigue tracking under the Structural Life Monitoring Program (SLMP). Bombardier produces reports on a monthly, quarterly and annual basis. The fatigue Life Expanded Index (FLEI) calculation routine in SLMP is based on Neuber’s notch, strain based approach, similar to CI89. Of the original seven locations proposed by the OEM, only the wing root control point is monitored actively and is used to schedule maintenance: Aircraft induction into the ALEX program is conditioned by the wing root FLEI, with CP1 equating to FLEI of 0.52 and CP2 to 0.65 respectively.

CF188 Y453 Bulkhead Web Taper Coupon Test Program
M. Sova, QETE

Full scale structural testing of the CF188 has highlighted fatigue cracking problems in the region of the Web Taper at the Y453 Bulkhead. BADS has developed a structural repair for this region using shot peening that will enable the component to achieve full fatigue life. QETE has initiated a coupon testing program to verify the effectiveness of this repair and to certify its fatigue life. As part of the process, the design detail surfaces were prepared using a chemical surface etching process, which simulated that implemented by the OEM for the application of Ion Vapor Deposition coatings. The process included a chemical degreasing, an acid etch (aluminum pickling), deoxidizing bath and final cold water rinse, similar to the process documented by DSTO-ARL for their component level fatigue tests of F-18 bulkheads. Preliminary results indicate that during fatigue testing to the IFOSTP spectrum pertinent to this area, fatigue cracks are developing at the predicted number of spectrum blocks. Follow-on tests to determine the effectiveness of proposed shot peened repairs to this fatigue sensitive area are in progress [20].

CF188 Wing Front Spar Coupon Test Program

M. Sova, QETE

This program was initiated by the Director for Technical Airworthiness and is also supported by BADS. QETE is conducting crack initiation tests simulating a critical location on the CF188 Front Spar loaded to the IFOSTP full scale test spectrum. Cold working of holes #172 and #173 will be evaluated. In addition, research into unique NDT methods for coupon inspection is being investigated [21].

CF188 Wing Fold Titanium Alloy Coupon Test Program

M. Sova, QETE

During the CF188 full-scale wing test (IFOSTP), in-service cracks were detected at the wing fold mechanism. To determine a suitable repair option for inclusion in the structural maintenance plan, a titanium coupon testing program has been initiated by the Director for Technical Airworthiness and is being performed by QETE and supported by BADS. The aim of this test program is to validate the component failure time/mechanism and to determine suitable repair options and implementation scenarios [22].

CC-130 Hercules

R. Kaufman, DTA

The Royal Canadian Air Force took delivery of the first C-130 Lockheed aircraft in 1960. Originally, the C-130 E and H aircraft were designed to meet the requirements of the MIL-S-5700 series specifications, which do not define durability and damage tolerance criteria. Failsafe principles were incorporated into the original design. Subsequent cracking and in-service failures have led to the identification of fatigue and fracture critical locations on these aircraft. At present, durability and damage tolerance analysis (DADTA) is not used for every CC130 primary structure repair. The current development of a CC130 loads model combined with a CC130 global finite element model and strain survey data will allow the contractor, Spar Aerospace, to perform DADTAs on all primary structure.

Future work by the Canadian Air Force includes ‘hot-spot’ strain surveys and development of preventative modifications and an associated improvement plans. To satisfy the damage and durability control plans, FEM, DTA and/or coupon tests for areas of interest using ‘hot-spot’ strain data are being conducted.

The attach (drag) angles on several CC130 aircraft were found to have cracks at more than one bow-beam location. An independent damage tolerance analysis was performed by Lockheed for this location. Cracking has also been found in the lower spar cap at WS 174 emanating from two fastener holes in the vertical flange. The severity of the cracking in the fleet has yet to be determined as the area is difficult to access. A damage tolerance analysis is being performed to determine the characteristics of the cracking in this location. A repair is also being developed.

CC-115 – Buffalo

Terence Cheung, DTA

Approval to have the Buffalo fleet life extension to the year of 2010 was granted by end of 2002. A wing full scale DADT test was contracted to the OEM in 1985 to substantiate the life extension to the year 2010 (estimated 30,000 flight hrs). Today, it is estimated that the highest-time airframe/wing fleet will only accumulate approx 24,313 flight hours maximum based on the current usage of 450 FH per year. All centre wing rear spars in the fleet have been replaced with an improved material (7075-T73) having a predicted endurance life of 37,500 FH. All six airframes in the fleet have been subjected to structural repair and corrosion survey. To date, two airframes (# 465 and 452) have been completed. The rest of the fleet is scheduled to be completed in the next two years.

The last review of the ASIP master plan was in 1985 and it will be updated in detail to identify the SSI locations, using severity factor in CPL and usage spectrum. It is important to note that the usage spectrum recorded in 1987/88 was different from the assumed wing DADT test spectrum, particularly under the 'training' and SAR' missions. It is proposed to review the fleet usage in order to qualify the effectiveness of the current ASIP program.

To address corrosion, a teardown inspection of a Buffalo aircraft stored in Trenton, ON is being considered. However, previous inspection results do not indicate major chronic corrosion problems. So far there are no major repairs on the Al 7079-T6 structural components (centre wing skins and front spar web), which are susceptible to corrosion. A bonded repair is under study as possible solution for Al 7079-T6 damage.

Cracks were found recently at the lower fuselage frames in the fleet. An angle doubler was installed by the maintenance contractor to relieve the load from the lower cracked frame. Those cracks have been carefully monitored for growth rates and the interaction between adjacent frames. OEM was tasked to investigate the possible cause of the cracks and to recommend repairs.

CC-138 – Twin Otter

Terence Cheung, DTA

The CC138 fleet ASIP uses the commercial maintenance program recommended by the OEM. There are four CC-138 aircraft flying in the CF fleet. Today, maximum accumulated flying time is 24,000 hours (850 hours annually) while the aircraft was designed for a lifetime of 35,000 flight hours. The life extension 2010 plan has not been approved yet. The third line inspection (corrosion and structures), is commencing early this year and will be completed by 2004. In 1992, the flying mission has been changed from Search and Rescue (SAR) to transport operation.

During the design and development stage, the damage tolerance concept was not applied to the CC-138. In order to comply with ASIP requirement, 55 CPLs were identified and 10 most critical locations were chosen to perform the damage tolerance analysis using the CC-138 fleet average mission mix spectra.

No major structural or corrosion problems were reported in the past years. All the structural repairs were reviewed by the OEM. An overall structural and corrosion review is scheduled for the fleet upon completion of the third line inspection. Since the usage mission has been changed in 1992, the ASIP has to be reviewed and updated with the current mission profile.

CT-142 – Dash 8

Terence Cheung, DTA

The CT-142 fleet (4 aircraft) has been approved to have a life extension to 2011. The 2021 life extension plan has been submitted for the airworthiness approval. Today, the fleet has accumulated an average flight time of approximately 5500 hours. Due to the introduction of new training syllabus (low altitude), a load survey was performed between the years of 1999 to 2001 to assess the impact to the current ASIP. The OEM was tasked to determine if there are any reductions in the inspection intervals due to the change in mission. From the DADT analysis, the OEM concluded that there is a damage ratio of 2.16 per flight between CT-142 and commuter Dash 8 airplanes. A 'half-life' threshold and repeated cut-off were also incorporated in the maintenance program as per the OEM recommendation. Currently, a special team is working to incorporate the difference in the maintenance programs between the commercial and CT-142 fleet.

CH-146 Griffon

Terence Cheung, DTA

This fleet is currently merging the structural practices of a civilian certified aircraft with the unique requirements of a military environment with a view of maintaining long-term structural integrity. This is being done through the development of an active structural integrity program aimed at ensuring compliance with military airworthiness requirements. One of the primary tasks has been the validation of current military usage with the original procurement assumption – the usage was supposed to fall within the original civilian design spectrum. This assumption is being validated through the use of parameters defined in the cockpit voice flight data recorder and the development of maneuver recognition software.

In support of the accident investigation for the Griffon crash in July of 2002, fatigue crack growth rate testing has been completed to quantify the effects of forming on the fatigue crack growth properties of AISI 301 stainless steel, material of the tail rotor primary spar. The work is being done to refine the estimates of time until failure from initial surface flaw to aid the assignment of appropriate inspection intervals. The damage limits on the tail rotor are severe; critical locations permitting no damage greater than 0.005". Probability of detection work and additional options for inspection are being investigated to define practical operational capability.

CH-124 Sea King

Terence Cheung, DTA

Within the last few years dowel pin holes on the planetary gear flange of the main rotor shaft have been elongating and subsequently developed cracks. To address this problem and to prevent cracking the manufacturer has established a repair that requires over sizing of elongated holes and shot peening of the surface. A newly manufactured shaft was assigned no life limit and the initial fatigue assessment applied no penalty to this component following repair. However, subsequent test validation saw a severe life penalty of 700 hrs assigned to a part repaired after damage beyond certain limits. Other manufacturers of similar components have not assumed this life penalty for equivalent and less conservative repair schemes. The Canadian and US operators are considering an additional fatigue testing to validate the life or consider the viability of a life extension.

Within the last five years, the Canadian Sea King fleet has been upgraded to include new T58-100 engines and new 24000 series gearboxes. Standard operating procedures quote the use of "full power" for a number of manoeuvres with upper N_R limitations governed only by autorotation at 117%. Though with the old engine/gearbox configuration, standard full power settings achieved no greater than 103% N_R, the current power plant can routinely achieve far greater main rotor speeds (108% N_R). The original fatigue assessment for the airframe and dynamic components included a small percentage of anticipated flight time at these greater rotor speeds but did not anticipate the greater standard operating conditions that are presently being experienced. To address this issue in the short term, the fleet has been limited to less than 103% N_R (drooping to 100% in flight) until the ramifications on flight instrumentation and fatigue effects on airframe and dynamic components are more clearly understood.

This fleet is also undergoing a redefinition and refocusing of the existing structural integrity program and will be addressing issues such as current usage definition.

CT114 Safety by Inspection Coupon Test Program

M. Yanishevsky, SMPL, IAR, NRC, M. Sova, QETE

The Canadian Forces (CF) CT114 Tutor aircraft fleet, procured in 1962, has been retired from its current role as lead-in fighter trainer. However, the Canadian Forces have decided to continue operating several of these aircraft in the Snowbird aerobatic role. The CT114 Tutor aircraft was originally designed, monitored, and maintained according to the Safe Life philosophy. Recent re-analysis using current aircraft/fleet usage data from on-board Operational Loads Monitoring and Individual Aircraft Tracking systems, as well as recent field detection of cracks in fuselage and wing structures, indicated that the Safe Life of the fleet in general has been consumed. The Original Equipment Manufacturer, Bombardier, and the CF Director of Technical Airworthiness and QETE organizations have collectively developed and conducted a Safety by Inspection testing program in order to keep selected aircraft from the Tutor fleet operational for the Snowbird aerobatic team using Damage Tolerance and Crack Growth principles. Since the fuselage and tail structures have recently seen a full scale durability and damage tolerance test, it was urgent to generate component level spectrum test data in order to validate that the change from the Safe Life to the Damage Tolerance philosophy for the wing attachment fittings was indeed possible. Analysis, service experience, and results from previous full scale tests identified five locations that were most critical. Three aircraft were decommissioned in order to recover components for testing purposes. Electron Discharge Machining was used to introduce damage in critical design details of these components to simulate worst case unsuccessful inspections. These components were then monitored to establish crack growth behaviour while being fatigue tested in uniaxial servohydraulic material test load frames. The components were subjected to the current Snowbird aerobatic usage spectra in order to develop/validate a Safety by Inspection program for formation and solo aircraft. To date, most locations have demonstrated that they will be able to satisfy the life extension requirement. Only one location has required to have a repair doubler designed and implement to allow the wing front spar lower cap to achieve its expected life. Additional tests will be performed on specimens where initial damage introduced was “too severe” [23].

CT133 Safety by Inspection Program

M. Yanishevsky, SMPL, IAR, NRC, M. Sova, QETE

The Canadian Forces CT133 (Lockheed T33 - Silver Star) first entered Canadian military service in the early 1950's in the role of fighter lead-in trainer. During its career, the role of the CT133 has changed several times, evolving to the current service demands for target towing, electronic warfare, executive transport, and most recently for an ejection seat test bed. Projection for the CT133 aircraft fleet is that it will be remaining in active service at least until the year 2010, requiring extension of its structural life far beyond the original design parameters. The Canadian DND has developed an action plan to ensure continuing structural airworthiness based upon Safety by Inspection and the Damage Tolerance philosophy to enable prediction of the operational life of the airframe. Operational Loads Monitoring systems have been installed on fleet aircraft, data from which were used to assess and monitor the flight loads experienced in each of the CT133's roles. This system is also being used to continually update the load spectra for analytical purposes. The wing is the key life-limiting structural element of the aircraft. Therefore, an accurate assessment of the available fatigue life in critical wing locations is required to prove / validate the continued viability of the fleet. To achieve this, DND has undertaken a coupon test programme using spar caps from retired wings to generate experimental material and sub-component level test data, that will be used to substantiate / maintain the Canadian Military Airworthiness Type Certificate (CMATC) of this aircraft. To develop the material property database, tensile, fracture toughness, and fatigue crack growth test coupons are being extracted from existing spar caps (Al 2014-T6 extrusion) from a time expired wing and a wing that has not seen service. This material can no longer be purchased and fracture mechanics data on Al 2014-T6 does not exist. To develop appropriate inspections and inspection intervals, two critical locations will be cut from 3 aircraft (6 spar caps) for generating component level design detail crack growth data, loaded to representative service usage spectra. [24].

P-3C / CP140 Service Life Assessment Program (SLAP) Support

Marko Yanishevsky, SMPL, IAR, NRC

The Canadian Forces are collaborating with the United States Navy, the Royal Australian Air Force, and the Royal Netherlands on the P-3C Orion / CP140 Aurora Service Life Assessment Program in order to determine what measures will have to be undertaken to allow Canada to continue operating the CP140 fleet of aircraft to the year 2025 safely and economically. The program consists of a wing and fuselage full-scale fatigue test, two empannage full-scale fatigue tests and several component and design detail tests. The WS167 and BL65 component specimens, Centre Crack Panel (CCP) fatigue crack growth specimens, Dome Nut Cold Work repair fatigue life coupons, and Cold Worked Hole fatigue life specimen testing, as well as teardown activities of the Kestral and post test Right Hand Wing (ATESS) and Probability of Detection NDI studies, form part of the Canadian contribution to this program. These areas and design details simulate regions on the aircraft, which have low fatigue lives based on analyses. The goal of these tests is to establish the fatigue lives of the assemblies, determine critical details of the outer and centre wing, identify where fatigue cracks will develop naturally and monitor their crack growth characteristics, and determine the effectiveness of cold working as a hole repair / fatigue enhancement option. As well, the tests are to provide a measure of the damage and crack detection capability of in-service NDT inspection techniques. Strain gauges have been installed in strategic areas to ensure that a representative strain distribution is induced in the components and to scale the loading spectrum to the same strain levels as the full-scale test.

Testing of the first WS167 and BL65 components have been completed and the data were to effectively anticipate damage in the full-scale test article (Figure 26, Figure 27). A rarely found 1 to 1 match was found between the component tests, the right "old" wing and the left "new" wing in the areas of representation. The WS167 component failed at ~3 lifetimes with 22 identified failure sites; post-test teardown inspection at the Quality Engineering Test Establishment (QETE) revealed an additional 28 sites with 35 total cracks, all below the threshold of detection without fastener removal and/or teardown. On the other hand the BL65 achieved 6 lifetimes with only 9 naturally occurring failure sites were detected during spectrum fatigue testing. To promote additional failure sites, artificial damage was introduced in three locations in the lower wing skin panel at 5 lifetimes; however, additional 2 lifetimes of subsequent fatigue loading were not able to encourage cracks to develop and propagate from these sites. The BL65 component successfully achieved 140% of the highest anticipated service load during a Residual Strength Test. Posttest disassembly of the BL65 component revealed 13 additional cracks, which were either inaccessible or hidden between layers of structure. The inspection results from these component tests emphasized the effectiveness of ultrasonic inspections for detecting damage in hidden layers. Fatigue testing of the Center Cracked Panel (CCP) specimens using spectra representing three failure critical areas were tested and the generated fatigue crack growth rate data was used effectively by the contractor to establish proper prediction of the response of the structure to fatigue spectrum testing (Figure 28). The Cold Worked Hole and Dome Nut Hole fatigue life tests are currently underway. A second WS167 component is being prepared for testing to establish the potential of cold working critical areas. In this latter test, following ~25,000 SFH of 50th Percentile US Navy usage to simulate the state of fatigue damage cold work repair will be implemented to simulate introduction of cold working in the CF CP140 fleet. This is to establish the potential effectiveness of this repair option on the P-3 / CP140 fleets.

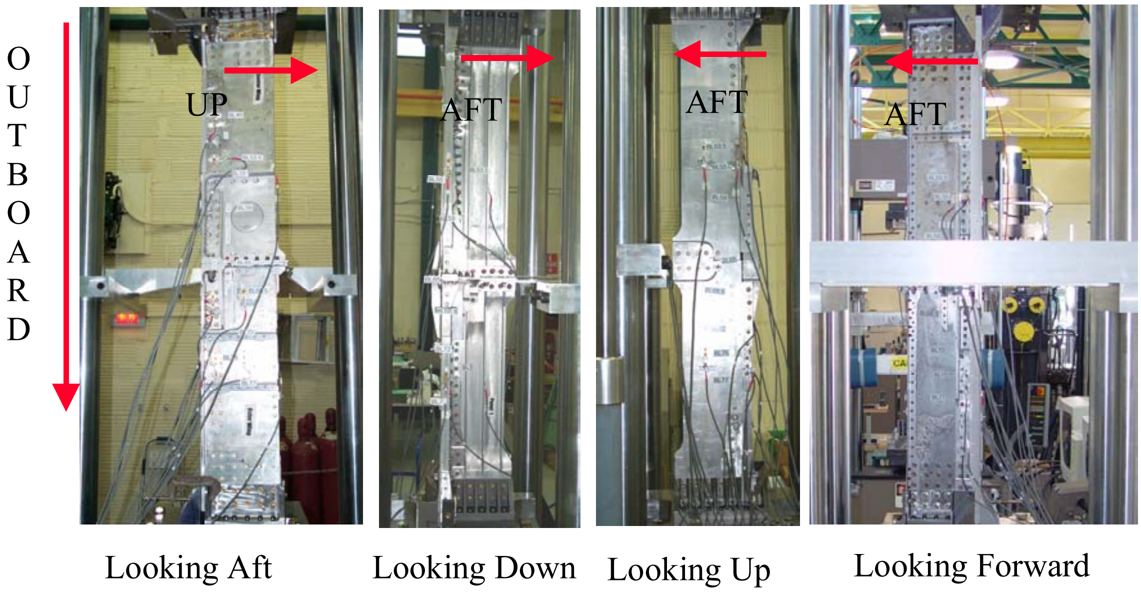


Figure 26. The BL65 main lower wing spar component installed in the loading frame.

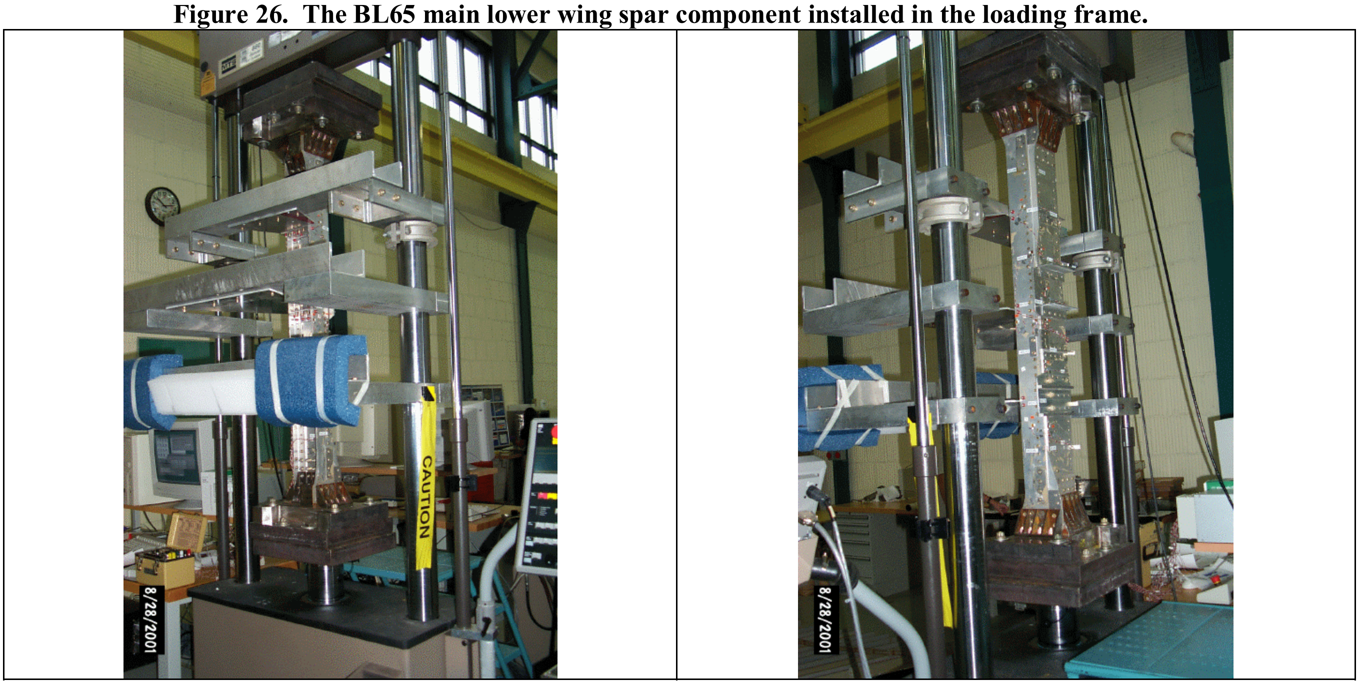


Figure 27. The WS167 component installed in the loading frame.

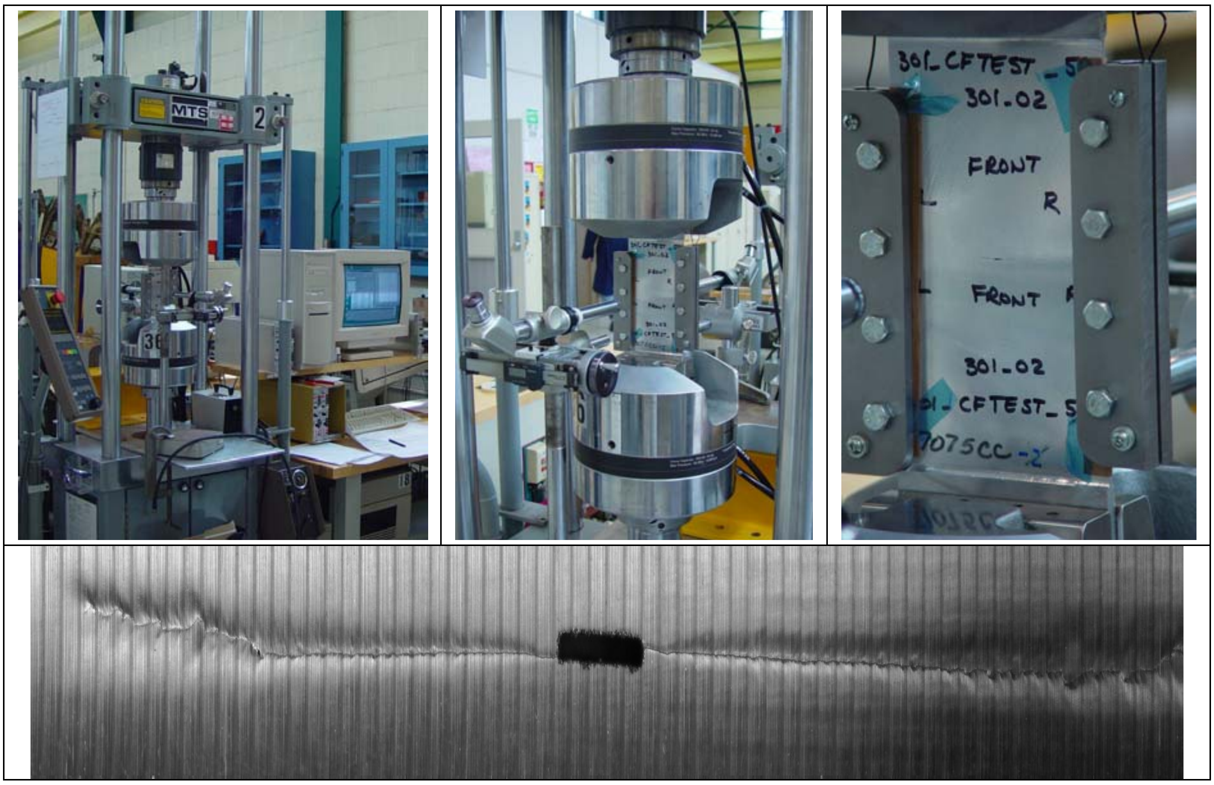


Figure 28. Centre Crack Panel tests used to establish spectrum fatigue crack growth rates for different critical locations and assessing the effects on fatigue crack growth behaviour of 50th and 85th percentile spectra. Note in the lower photo the plastic zones and tortuous crack path evident at longer crack length responsible for crack growth retardation effects (photo taken using Edge of Light).

Compression Testing of Composite Patch Repaired C-141 Wing Panels

Marko Yanishevsky, SMPL, IAR, NRC

Compression testing of Composite Patch Repaired C-141 Wing Panels is being conducted to support the USAF effort in Aircraft Corrosion Control and Coatings, to determine the ability of composite patch repairs to restore the residual strength of wing panels that have been reworked beyond their normal repair limits in order to eliminate exfoliation corrosion caused by service environmental exposure. Exfoliation corrosion is a major problem facing USAF ageing transport aircraft fleets. The efficacy of composite repairs where the structure is primarily loaded in tension has been determined for many cases; however, for structures and components dominated by compression loading little actual test data is available. These tests, although not completely answering all questions about the efficiency and reliability of composite repairs, are designed to provide data for a typical repair.

Three configurations of the C-141 Wing Panels are being tested: “pristine”, “damaged” (simulated by a centrally milled out “Z” section to a maximum depth of 80% of the panel thickness) and “damaged and repaired” (simulated by a centrally milled out “Z” section repaired with boron composite patches). The volume of material machined away for the latter two tests simulates a worst-case scenario of what may be required to remove significant exfoliation or other corrosion damage and addresses concerns about grind outs affecting bending and torsional stability. The pristine and damaged panels are shown in Figure 29 and Figure 30.

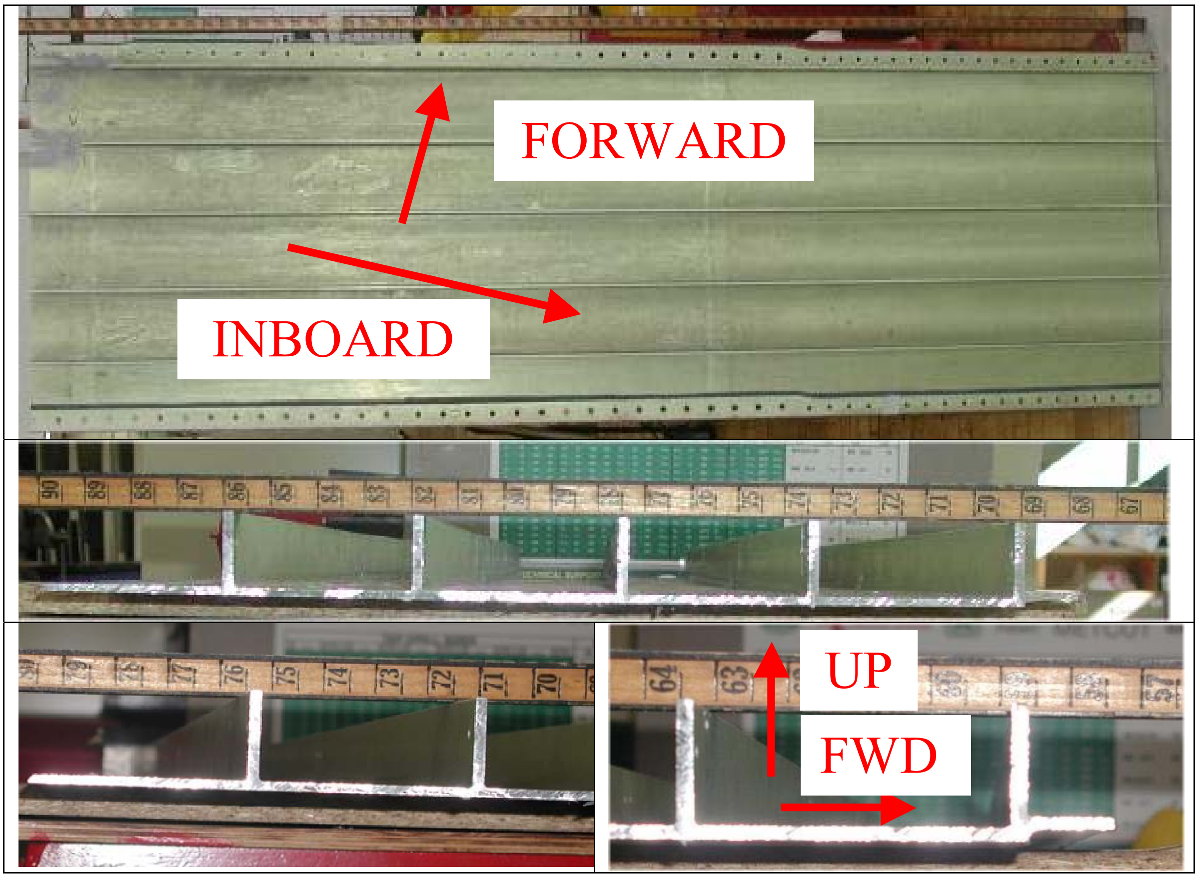


Figure 29. The "pristine" C-141 left lower wing panel #4, as received. Upper photo view is downward; the middle and lower photo views are outboard.

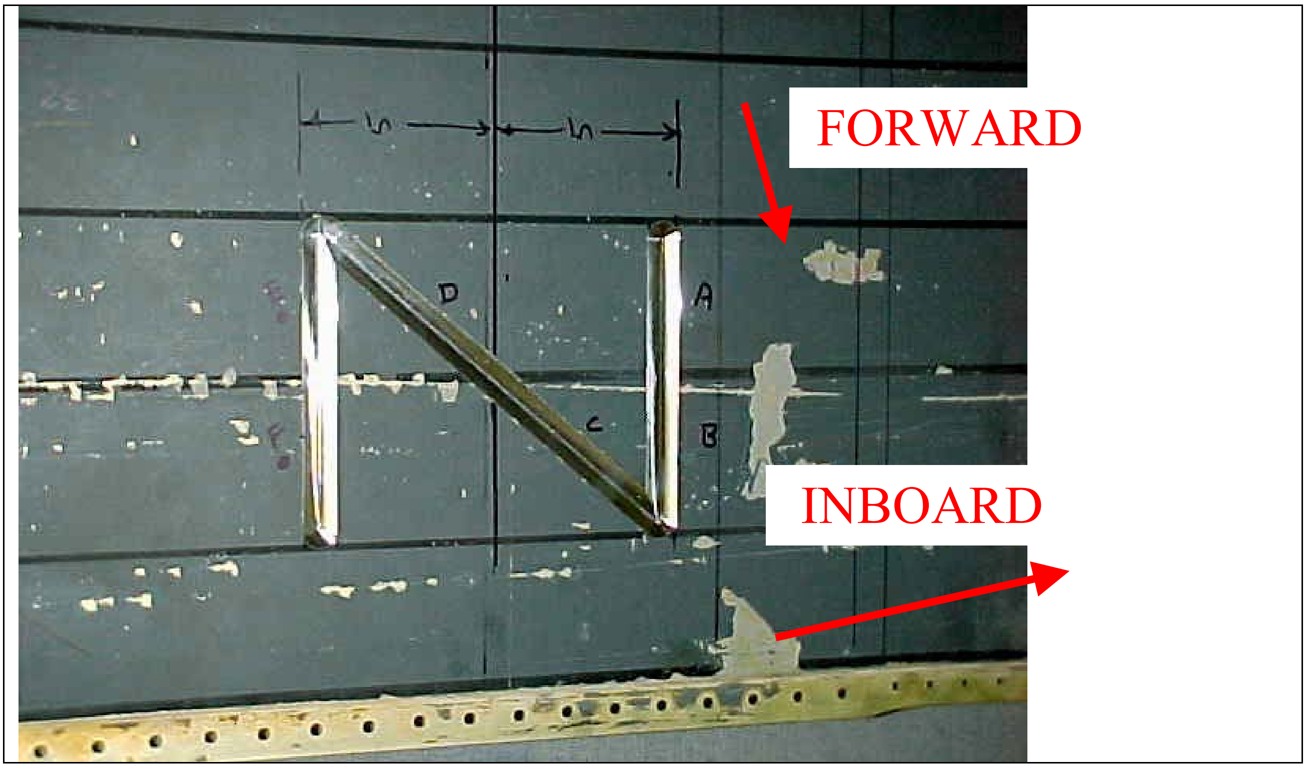


Figure 30. The milled out "Z" section that simulates the "damaged condition", view looking up.

Loading of the wing panel components is being accomplished using a custom designed loading fixture, shown in Figure 31, to allow the application of compressive loads on the ends of the wing panels. The design incorporates two rib supports attached to the panels at stations approximately 762 mm (30") apart. C-channel anti-buckling stiffeners are installed along the edges of the panel on the lower wing surface to simulate the support normally provided by adjacent wing panels, and to preclude premature failure because of unrepresentative buckling at the free edges.

Strain gauge data is being acquired to determine the strain distribution for comparison to FEM analyses. The LVDT displacement transducers are being used to measure out of plane deflections in the wing panels during testing, and to measure the overall movement between the two end fixtures.

To date tested was the setup pristine panel, with failure occurring by riser and panel buckling in the mid-plane of the panel, with most damage evident at the riser / skin interface, Figure 32 and Figure 33.

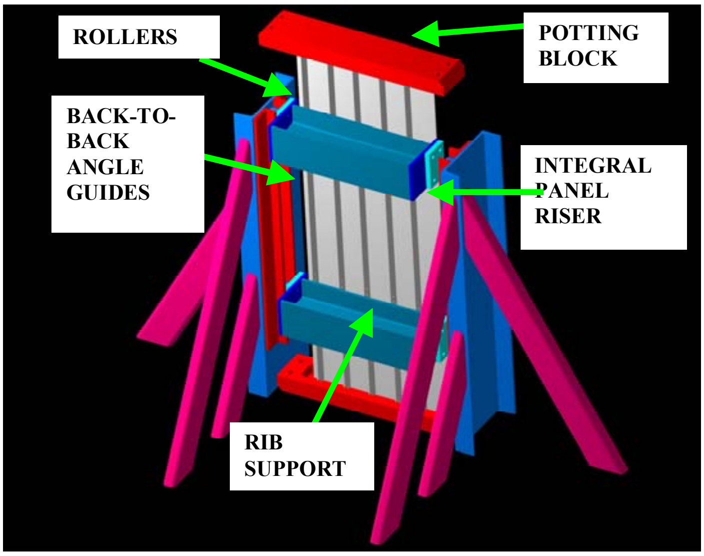


Figure 31. Overall CAD conceptual drawing of the test panel installed in the loading fixture.

Figure 32. Post Test - overall view of internal riser buckling and fracture at the riser/skin interface in the pristine panel.

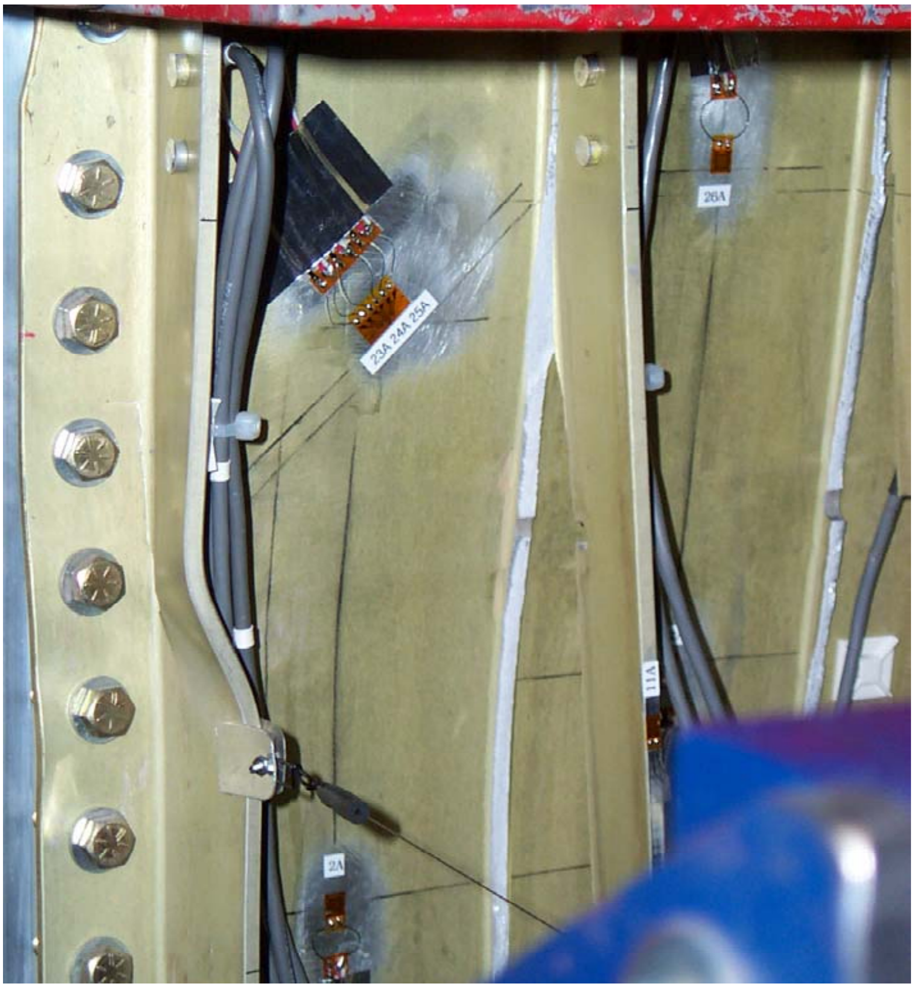


Figure 33. Close-up view of the riser buckling / fracture at the internal riser / skin interface of the pristine wing panel.

In-Situ Robotic Shot Peening

J. Dubuc, Bombardier

The technique of shot peening is commonly used for the life extension of fatigue-limited aircraft structures. However, concerns were raised about the consistency and reliability of conventional manual shot peening, especially on parts of complex geometry and where human access is limited. A new robotic shot peening process was developed by Bombardier Aerospace, Defence Services to provide a more accurate, uniform, and repeatable shot peening quality. This process, that is currently being used to extend the life of components on the CF-18 aircraft, is particularly effective for the rework of areas that are difficult to access.

A methodology was developed to facilitate the generation and verification of trajectories in a simulation tool that uses the 3D Catia model for its starting point (Figure 34). This methodology uses the NC programming tool in Catia to provide the normals to the surfaces. These normals are then corrected using proprietary software called Robot TG to form trajectories for the shot peening nozzles. The Catia model also provides the surrounding environment that is necessary to ensure collision free trajectories. This data is fed into a simulation tool that accurately represents the robot controller limitation. This combination results in a highly faithful simulation where the robot's movements can be analysed and corrected to ensure high quality collision free shot peening before deployment to the actual component.

The design and fabrication of the robotic shot peening system was as challenging as the development of the trajectory generation methodology. Since no commercial robotic shot peening systems were available on the market, our system was designed and built by our robotic team. The system is made up of 3 major components: an Japanese built robot and controller, a Swiss built computer controlled shot peening machine and a touch screen user interface that controls the robot, the shot peening machine and records all the process parameters. This interface, developed in-house, is the key to transferring the robotic technology the shop floor. It makes the system easy to use by technicians without advanced robotic training.

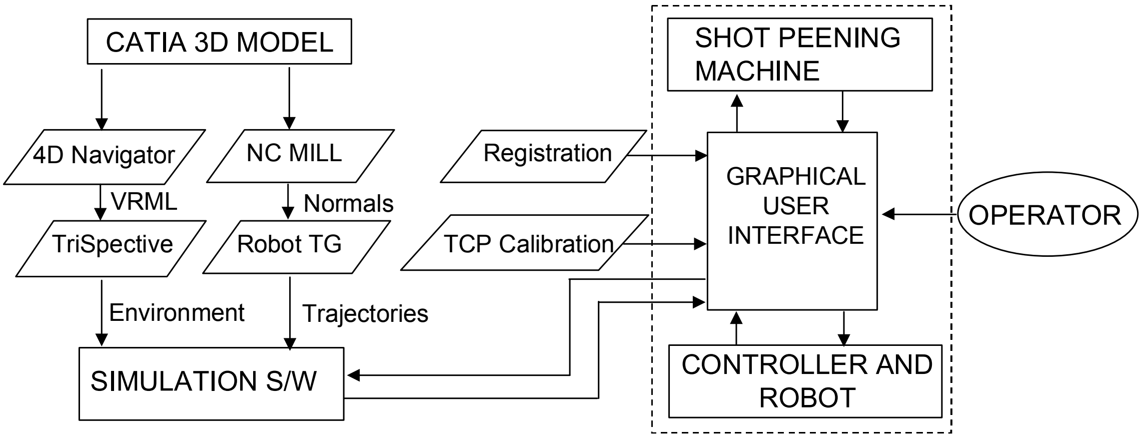


Figure 34. Trajectory generation methodology relationship with the robotic shot peening system

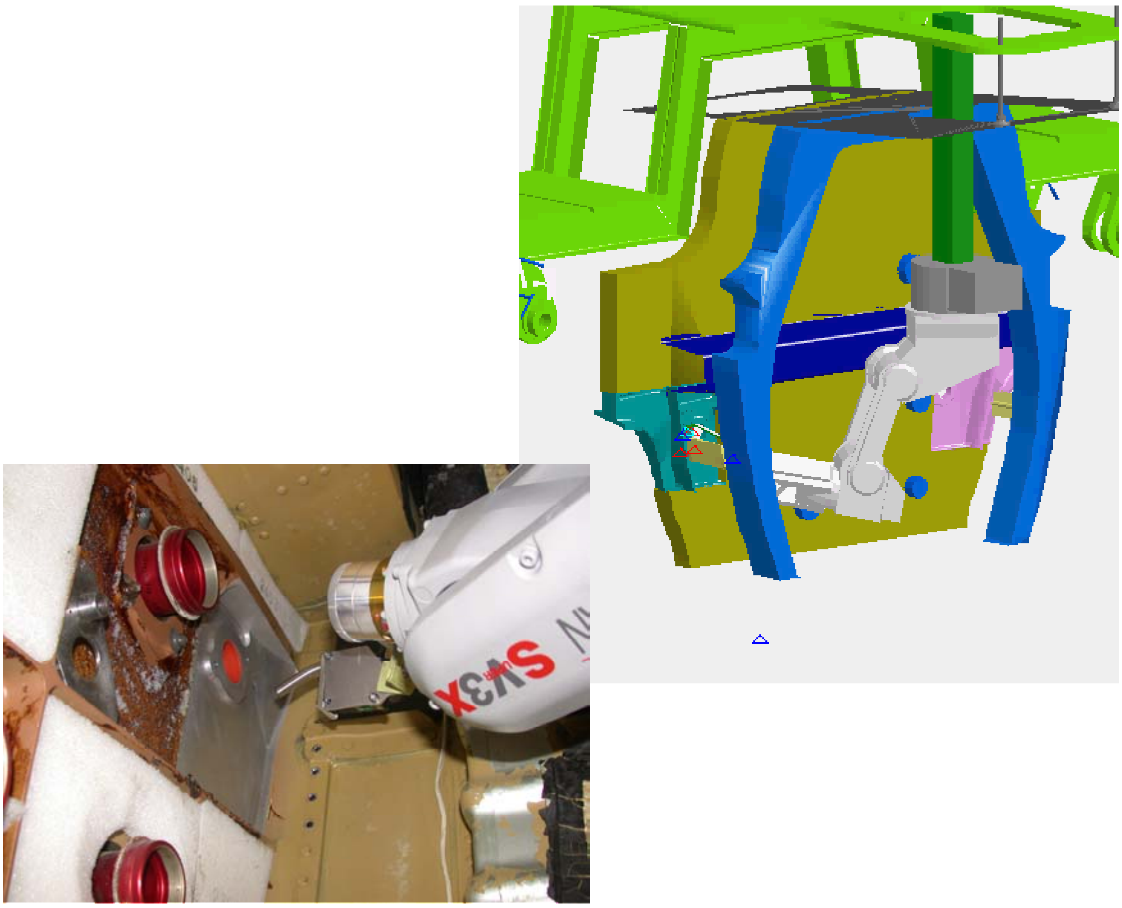


Figure 35. Robot upside down inside CF-18 fuel tank

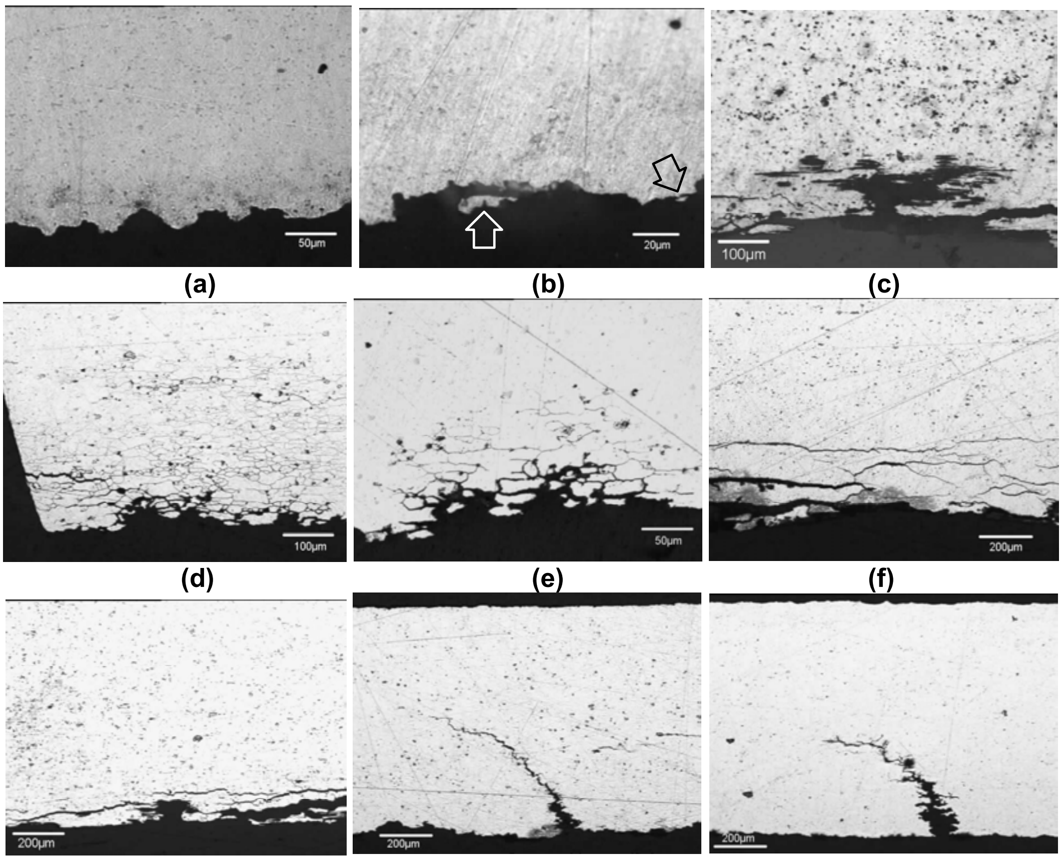
Among the critical locations on the CF-18 aircraft where the robotic shot peening is currently used is a tapered web on one of the wing carry-through bulkheads (Figure 35). Surface cracks were detected at this location on the IFOSTP FT55 full-scale test article, showing that the bulkhead is life limited. Because of the limited access, robotic shot peening was deemed the most appropriate application to reliably achieve full fatigue life. The robot must be inserted upside down into the fuel tanks to perform the shot peening. The robotic shot peening system is also used to shot peen between the lugs of the inboard leading edge flaps (ILEF). Once this component is installed in the jig, the robot performs a registration procedure with a pencil probe to ensure proper positioning. The shot peening nozzle is then installed and a tool centre point (TCP) is automatically calibrated for the job using wide beam lasers. The computer controlled shot peening machine is started by the operator using the touch screen graphical user interface. The shot peening parameters are recorded during the peening process and the system is stopped if setting are outside pre-determined limits. Several ILEFs have successfully been shot peened to date and the process is undergoing optimisation to further reduce the cycle time.

The robotic shot peening system has also been used to peen the test article of an important component on a Bombardier commercial aircraft. Once testing of this rework is complete, this modification, which includes the robotic shot peening, will be applied to several dozen in-service aircraft.

Evolution of Corrosion In Lap Joints

N. Bellinger, SMPL, IAR, NRC

A typical lap joint consists of two or more layers of sheet material with a stringer, tear-strap or frame attached on the inside surface of the second layer. The majority of jet transport aircraft fuselages have been assembled from 2024-T3 (clad or unclad) aluminum skins with 7075-T6 stringers and frames. These joints are only protected by primer and paint on some aircraft, while some manufacturers also include a layer of adhesive or sealant. Once the barrier or protective system is compromised and a corrosive medium penetrates the joint, the corrosion process could initiate at one or both of the faying surfaces. There are many types of corrosion that can occur in aircraft structures, the most common for lap joints being crevice corrosion. Lap joint age degradation can also include pitting, general surface attack, environmentally assisted cracking, or other mechanisms. To determine how these discontinuity states evolve, a damage characterization study was carried out in which ten sections were taken from naturally corroded lap joints from



(a) (b) (c)
(d) (e) (f)
(g) (h) (i)

Figure 36. Examples of various discontinuities found in naturally corroded lap joints. (a) corrosion pits found in clad layer, (b) flaking of clad layer (arrows), (c) as pits can get wider as they penetrate deeper into the bare material, (d) and (e) intergranular corrosion penetrating to varying depths, (f) and (g) exfoliation and (h) and (i) environmentally assisted cracking occurring in a sustained stress region (pillowing).

retired Boeing 747 and 727 aircraft containing various levels of material thinning as determined using an x-ray film based technique. These sections were cold mounted, progressively polished to document the discontinuities presents.

The results from this study showed that the discontinuities present in corroded 2024-T3 lap joints consisted of corrosion pits, intergranular corrosion, exfoliation and environmentally assisted cracks that occurred under a sustained stress (pillowing). Examples of these discontinuity states are shown in Figure 36. Based on this study the following process of the discontinuities evolution in clad 2024-T3 lap joints was proposed: Initially, pits form in the clad layer of the aluminum skins, which grow until this protective layer is removed exposing the bare material. Typically large sections of the cladding were removed by a number of pits coalescing. As the pits continue to grow into the bare material, intergranular attack begins to occur around the pit. These cracks tend to grow in the direction of the grain flow until they reach a free edge, either another pit or the faying surface. As the corrosion process continues the material between the pit and the free edge is removed either by flaking or by dissolution. This process of pitting, intergranular attack and flaking continues as the corrosion progresses.

Since it is unrealistic to carry out a damage characterization study to determine the surface topography for every level of thickness loss, it was decided to try and use the thickness maps from radiographs taken of cleaned lap joints available at NRC. To determine the thickness loss distribution, x-ray film taken of each cleaned skin was digitized with a resolution of 800 dpi (1 pixel=0.032 mm). A histogram plot of the thickness loss was obtained for different areas of each digitized film examined. A lognormal distribution was then fitted to the histogram data and the coefficient of variation was calculated by dividing the standard deviation by the mean. The results, shown in Figure 37, indicate that for the majority of the joints examined, as the average thickness loss increased, the coefficient of variation, which is a measure of the scatter in the surface topography, decreased. This correlation suggests that the average thickness loss, which can be obtained using standard non-destructive inspection techniques, is an important metric that can be used to obtain an indication of the surface topography. One exception to this trend of decreasing coefficient of variation occurred in the area where the environmentally assisted cracks were present since some of these crack surfaces were further corroded producing what appeared as exceptionally large pits.

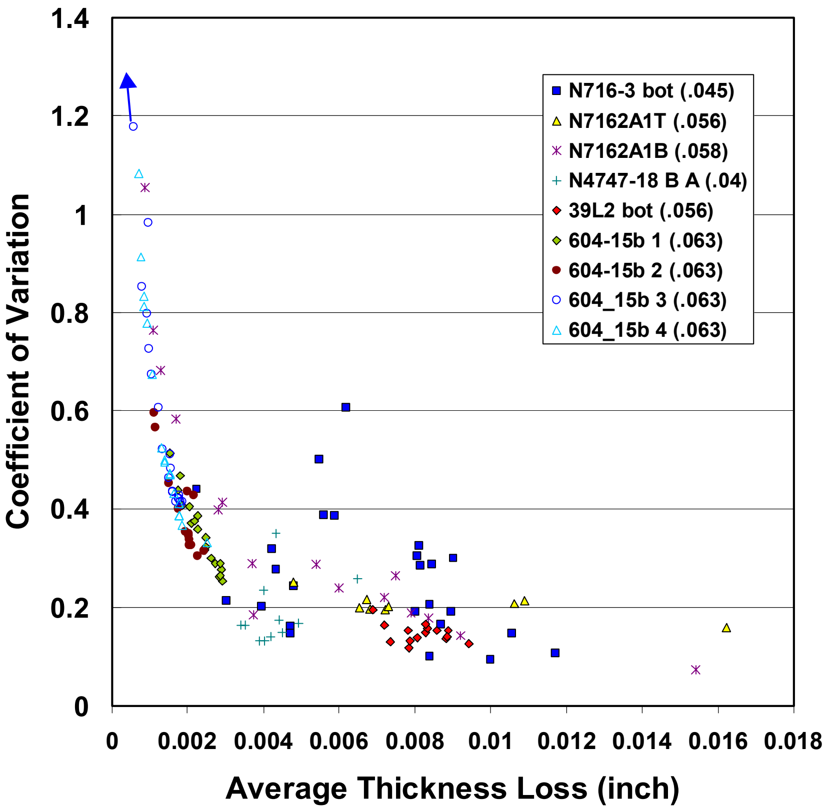


Figure 37. Coefficient of variation plot. Numbers in brackets indicate nominal thickness of lap joint.

These cracks have been found in many corroded fuselage joints. It has been shown that pillowing cracks may nucleate fatigue cracks. In a very meticulous microscopic analysis of a corroded lap joint R. Wanhill (NLR) demonstrated that these cracks may form at every rivet hole in a corroded area of a joint. Thus a situation of corrosion multi-site damage analogous to well known fatigue multi-site damage (MSD) has been identified. Given the potential for significant impact of pillowing cracks on residual strength of a lap joint focused inspections for these cracks should be implemented once an area of a joint is known to be corroded.

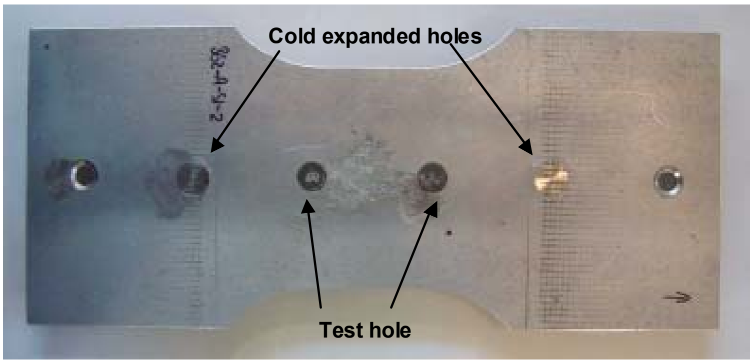
Effect Of Exfoliation Corrosion On Aircraft Structural Integrity

N. Bellinger, SMPL, IAR, NRC

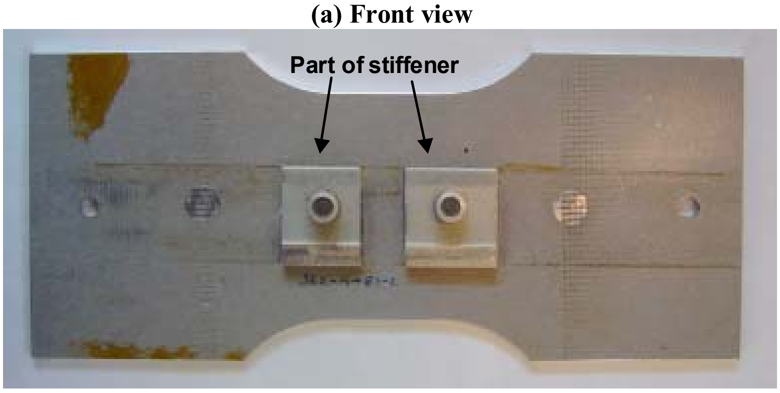
Exfoliation is generally found in upper wing skins around fastener holes at the exposed end grains. Presently in the current "find-it-fix-it" approach to corrosion maintenance when exfoliation is discovered, either through visual detection, search peening or other non-destructive inspection techniques, it must be repaired. The grinding of the exfoliated material can be carried out until the maximum groundout limit is reached at which stage the skin may require repair or replacement at significant cost to the operator. While this philosophy can result in the repair of very small levels of exfoliation (barely visible), the effect this type of damage has on the structural integrity of the upper wing is not well understood.

Fatigue Tests Results

To determine the effect that exfoliation corrosion has on the fatigue life of wing skins, a specimen that was loosely based on the ASTM E647 standard was designed to be machined from service exposed Boeing 707 exfoliated 7178-T6 upper wing skins. The stiffener, which was located along the back-face of the upper wing skins was removed leaving only a small portion of the flange around the fasteners to form a washer around the nut. Those holes that were not affected by the exfoliation were cold expanded and an interference fit steel plug was inserted into the hole to prevent premature failure, Figure 38.



(a) Front view

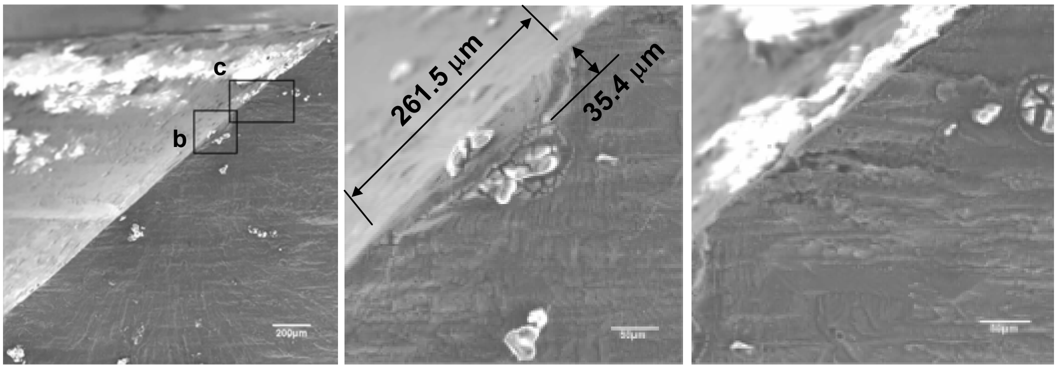


(b) Back view

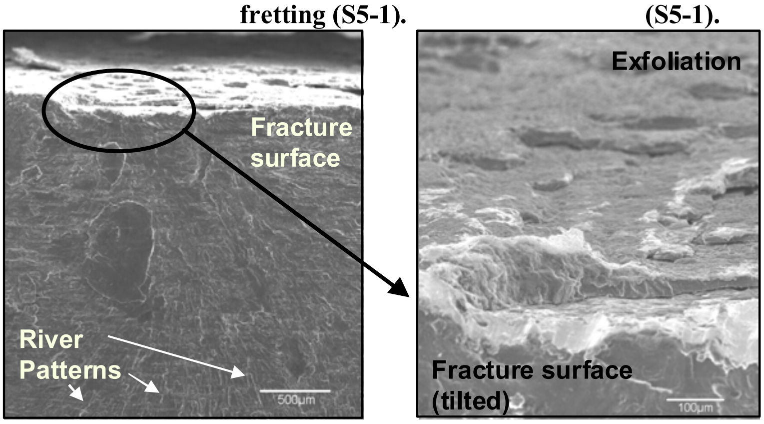
Figure 38. Optical Photograph of Test Specimen Cut From Upper Wing Skin.

All the tests were carried out under a constant amplitude compression dominated cyclic loading, with a maximum gross section stress of 82.7 MPa (12 ksi) and a minimum gross section stress of 137.9 MPa (20 ksi). The specimens were tested to failure in laboratory air at a frequency of 10 Hz and the results are given in Table 5. The maximum depth of the exfoliation was measured using ultrasonic inspection techniques that were carried out after specimen failure and was carried out from the back face of the specimen. As can be seen from Table 5 the number of cycles to failure for the exfoliated specimens was not significantly different from the non-corroded results suggesting that the current level of exfoliation might not be the critical factor governing the life of an upper wing skin.

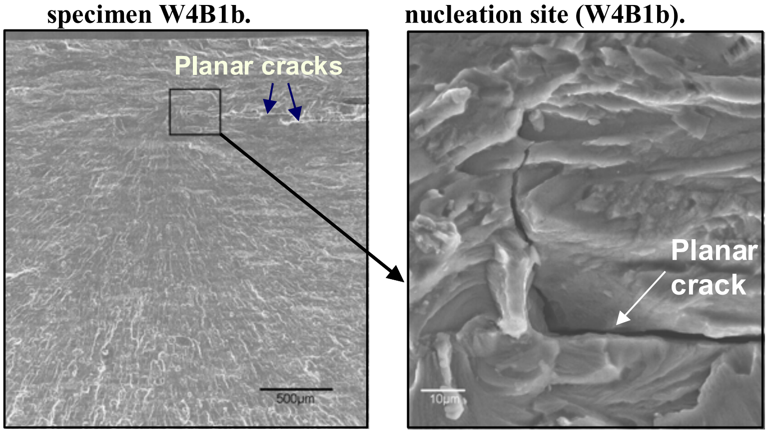
The fracture surface of each specimen was examined with the aid of a scanning electron microscope to determine the crack nucleation sites. For the noncorroded specimen, the cracking mechanism was found to be fretting that occurred along the countersink of the hole. However, for the exfoliated specimens, the cracking mechanisms were attributed to several causes, including fretting, pit-like discontinuity, intergranular corrosion and planar cracking. Examples of these crack nucleation sites are shown in Figure 39.



(a) River patterns showing crack nucleation sites for specimen 362-A-S5-1. **(b) Close-up of pit-like discontinuity, which may indicate the presence of fretting (S5-1).** **(c) Crack-like discontinuity, which may indicate intergranular corrosion (S5-1).**



(d) River pattern for primary crack nucleation site for specimen W4B1b. **(e) Close-up of crack-like discontinuity at crack nucleation site (W4B1b).**



(f) River pattern for secondary crack nucleation site for specimen W4B1b. **(g) Close-up of crack-like discontinuity (intergranular attack) at crack nucleation site (W4B1b).**

Figure 39. Examples of Crack Nucleation Sites in Exfoliated Specimens.

Table 5. Test Results of Specimens Taken From Service Exposed Boeing 707 Upper Wing Skins.

Specimen ID	Maximum Exfoliation Depth (inch)	Number of Cycles	Comments
W4B1b	0.025	182,982	Failure occurred at cold expanded hole.
362-A-S2-1	0.005	358,704	Test was stopped when single crack reached specimen edge.
362-A-S3-1	0.004	692,865	
362-A-S5-1	0.008	267,054	
362-A-S6-1	0.0	222,556	No exfoliation
362-A-S8-1	0.010	560,465	

Finite Element Modeling

A three dimensional finite element model was developed that used a "soft inclusion" technique to simulate exfoliation in order to determine the local stress distribution in the specimen. This technique was previous used at the National research Council Canada to simulate compression after impact in composite structures. Eight node brick elements were used to generate a model for each of the specimens simulated (362-A-S3-1, WB1C1 and 362-A-S5-1) using the MSC/Patran software. A typical finite element mesh is shown in Figure 40. To simplify the finite element analysis, the non-test holes were merged with steel plugs and the fastener nut in the test hole was also merged with the flange. The contact between the fastener and the plate, plate and flange and flange and fastener was simulated using the deformable contact bodies provided in the MSC/Marc software, which was used to carry out the analysis. The Coulomb friction model with a coefficient of friction of 0.2 was used in the analysis. The geometric nonlinearity caused by the finite deformation and finite slide between the contact bodies were analyzed using the 'Small Displacements/Small Strain' feature in MSC/Marc. The material was considered as linear and isotropic and all analyses were carried out using MSC/Marc. The area and depth of the soft inclusion was determined from ultrasonic inspections of the front and back faces of the specimen. The modification associated with using the soft inclusion technique was accomplished by changing the Young's modulus (E) of the elements in the affected area.

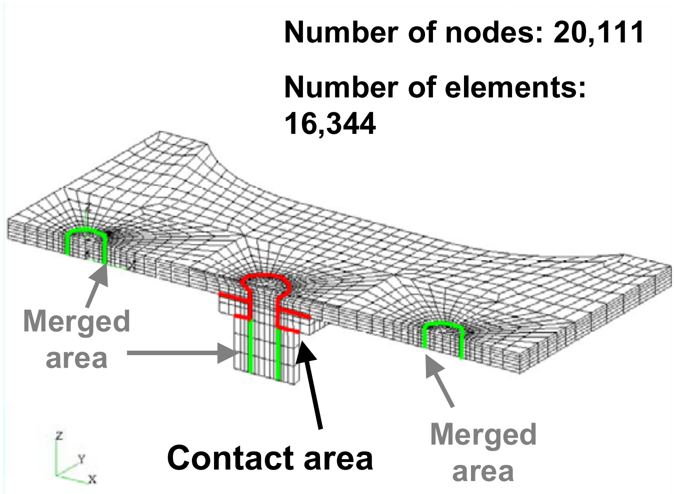


Figure 40. Section Taken Through a Typical Three Dimensional Finite Element Mesh.

An analysis that was carried out examined two scenarios; one in which the exfoliation was shallow and the other where the depth of the exfoliation was at the edge of the countersink. For both these analyses the surface area of the exfoliation was kept constant while the Young's modulus was changed to determine the increase in stress that would occur and the results are shown in Figure 41. As can be seen from this figure, as the E degradation approached 100% (complete material loss) the stress increased associated with the deeper exfoliation was higher than the corresponding stress increase associated with the shallow exfoliation. For exfoliation that had a depth of 38% of the nominal thickness, the corresponding stress increase was only 11%.

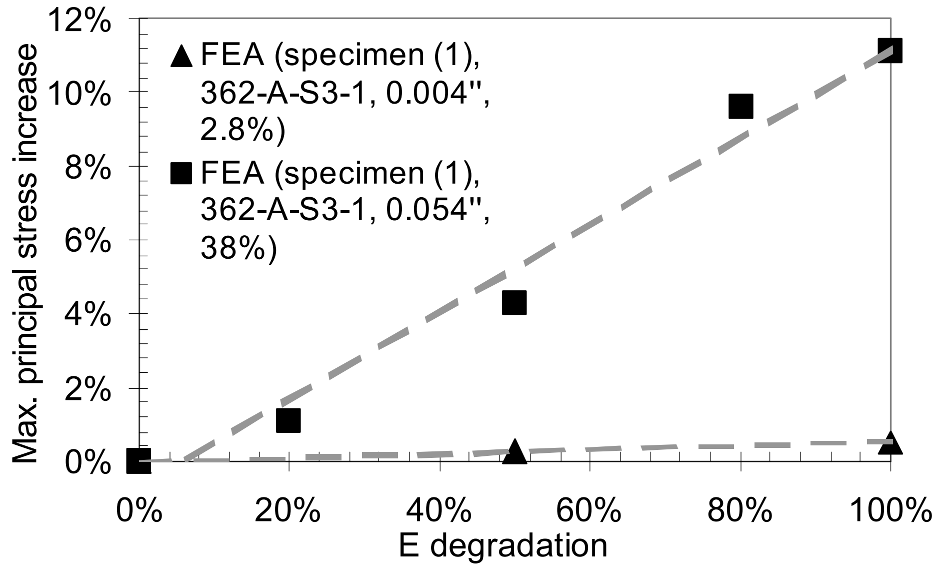


Figure 41. Maximum Principal Stress Increase With Young's Modulus Degradation and Exfoliation Depth of Soft Inclusion.

Residual Fatigue Life Analysis

As was mentioned previously, the exfoliated specimens failed due to several cracking mechanisms. Currently no single analytical model could cover all these mechanisms. Therefore it was decided that an analytical model would first focus on one of the crack nucleating mechanisms, such as a pit-like discontinuity, which was found in specimen 362-A-S5-1. The analytical model used followed the holistic life assessment methodology (HLAM). In this methodology, the life of a component is divided into four distinct phases: nucleation, short crack, long crack and final instability. The analysis may begin based on the initial condition of a component, which is determined from the Initial Discontinuity State (IDS) or at some time slice, which is determined from the Modified Discontinuity State (MDS). This IDS is a material characteristic that is related to the intrinsic material discontinuities or the intrinsic manufacturing and joining discontinuities that are present in pristine structures while the MDS describes the subsequent physical state of a discontinuity at any given time in its evolution. The MDS can evolve under all extrinsic influences such as cyclic and environmental loads. Based on physically measured IDS/MDS and mechanistic-based fracture mechanics methodologies, HLAM is capable of predicting the entire life cycle by addressing the effect of age degradation and fatigue interaction.

For specimen 362-A-S5-1, the nucleation site was found to be a pit-like discontinuity, Figure 4b, or MDS, which was caused by fretting at the countersink hole. From HLAM, this MDS evolved from an IDS at time zero through extrinsic influences such as cyclic and environmental loading. Due to the lack of information, such as the initial discontinuity state for the material 7178-T6, service loading and service environment at was very difficult at this point to determine when this pit-like discontinuity nucleated. That is, it could not be determined whether this MDS was present prior to fabricating the specimen from the service exposed upper wing skin or formed when the specimen was subjected to the compression dominated cyclic load. Therefore, a crack growth analysis, using AFGROW, was carried out using the pit-like discontinuity as the starting crack size, which was grown until failure.

The pit-like discontinuity was assumed to be a semi-elliptical crack with the dimensions shown in Figure 39b, which were measured from the scanning electron microscope image. Since the current version of AFGROW does not have a crack model for a countersink hole filled with a fastener, a crack model at a straight open hole associated with corrected beta factors was used for the crack growth analysis. The crack growth rate curve for 7178-T6 was obtained from the NASGRO database and entered into the AFGROW program. The initial crack length, which was growth to failure, was set equal to the size of the pit-like discontinuity that was present at the crack nucleation site. The analysis calculated the number of cycles to failure as 128,600, which is approximately 52% less that the test results. This conservative prediction could strongly indicate that the pit-like discontinuity was not present along the countersink prior to testing. Therefore, the time required to nucleate and grow this pit by the fretting mechanism should be taken into account when predicting the total life of the specimen.

JOINING TECHNIQUES

Fatigue of Bonded Aluminum Joints

D.L. DuQuesnay, RMC

The fatigue/adhesives group at R.M.C. has been studying the effect of surface preparation on the fatigue endurance of bonded aluminum joints. Most of the work to date has been on bare 2024 T3 and clad 7075 T6 using either FM73 or FM73M adhesive. Both single and double lap shear joints have been used. under cyclic loading (R=.5). Initial investigations compared joints made from surfaces which had been grit-blast, grit-blast and silaned, and grit-blast, phosphoric acid anodized and silaned. In the ASTM wedge test, there are significant differences between these joints. The fracture toughness after several weeks exposure would typically be 100 kJ/m^2, 600 kJ/m^2, and 2500 kJ/m^2 respectively. When single lap shear specimens were fatigued, the life of specimens with surfaces only grit-blast showed an order of magnitude drop. A much smaller drop was recorded for the other two preparations. It is should be noted that the surface preparation significantly affected the fatigue life even on specimens that had never been exposed to moisture. Examination of the failure surfaces showed that failure initiated at the surface for all of these joints.

In double lap shear experiments, the load required to produce an equivalent life increased substantially (32 MPa vs. 19 MPa to produce 10^5 cycles) but the differences between surface preparations, even in the dry state continued. Examination of the failure surfaces indicated that failure continued to occur at the surface for the specimens that had been grit-blast only. However, the failure surface moved partially into the adhesive and about the carrier cloth for the other two preparations.

A problem with the above results (and similar results in the literature) was that the large contrast in fatigue life was between the grit-blast surface preparation and the other two preparations. On the basis of the wedge test however, the former would never be considered for a real application. The contrast between the other two treatments was much smaller. Since a great deal of effort has been put into developing environmentally friendly surface preparations that exhibit zero or almost zero growth in the wedge test, it was desirable to determine whether or not wedge test performance could be correlated with fatigue life over a range of wedge test results that might actually be considered acceptable. A series of tests were performed in which one end of the specimen was tested in the wedge test and the other was then machined into a fatigue sample and the fatigue life determined at a single load (approximately 20 MPa). The results indicated that there was still a correlation between the log of the fracture toughness as determined in the wedge test and the log of fatigue life. Interestingly enough the slope of the relationship was larger for the fatigue life that was determined from dry specimens than it was for fatigue life determined while the specimen was immersed in water.

When the same series of fatigue tests were performed on the clad 7075 samples it was found that there was a pronounced tendency for the metal to fail before the adhesive. Careful examination of the failure surface seems to indicate that crack nucleation is occurring not at the adhesive-metal interface but rather at the cladding-substrate interface. This result seems to occur when the load is decreased below a certain limit and that above this failure in the adhesive or at the adhesive-metal interface occurs. Work is currently underway on clad 2024 and bare 7075 to see if this is a problem peculiar to 7075 or occurs for both clad alloys [25, 26, 27, 28, 29, 30].

Effects of welding and weld heat-affected zone simulation on the microstructure and mechanical behaviour of a 2195 aluminum-lithium alloy

M. C. Chaturvedi, Department of Mechanical and Industrial Engineering, University of Manitoba

Microstructures, tensile and fatigue properties of 2195-T8 Al-Li alloy subjected to weld heat affected zone (HAZ) simulation, and gas tungsten arc (GTA) welding by using a 4043 filler metal, with and without post-weld heat treatment, were studied. The principal strengthening precipitate in the T8 base alloy was T1 (Al_2CuLi) phase. The HAZ simulation resulted in the dissolution of T1 precipitates, and the formation of TB (Al_7Cu_4Li) phase, G-P zones and d¢ (Al_3Li) particles. When the HAZ simulation was conducted at the highest temperature of 600°C, microcracks and voids also formed along the grain boundaries (GBs). In the specimens welded with 4043 alloy, T (AlLiSi) phase was found to form in the fusion zone (FZ). Elongated TB phase and microcracks were observed to occur along the GBs in the HAZ close to the FZ interface. T1 phase was not observed in the HAZ. The post-weld heat treatment resulted in the spheroidization of primary T phase and the precipitation of small secondary T particles in the FZ, and the dissolution of T_B phase and the re-precipitation of the T_1 phase in the HAZ. Both HAZ simulation and welding gave rise to a considerable decrease in the hardness, tensile properties and fatigue strength. The hardness in the FZ was lower than that in the HAZ. Although the post-weld heat treatment improved both hardness and tensile properties due to the re-precipitation of T1 phase in the HAZ and a smaller interparticle spacing in the FZ, no increase in the fatigue strength was observed because of the presence of microcracks in the HAZ.

Investigation was undertaken to study the influence of specimen orientation on the fracture toughness and fatigue properties of welded and, weld heat-affected zones (HAZ) simulated 2195 alloy. Specimens were gas tungsten arc (GTA) welded with 4043 filler metal and HAZ simulation was done in a Gleeble 1500 thermo-mechanical simulator. The mechanical properties of the welded and HAZ simulated material were evaluated by using specimens with orientations of 0°, 15°, 30°, 45° and 90° to the rolling direction. Both tensile and fatigue tests were performed with a computerized Instron 8502 servo-hydraulic testing system. Optical and analytical electron microscopy, image analysis, microhardness, X-ray diffraction, orientation imaging microscopy (OIM), and TEM were used to characterize the microstructure

The mechanical properties of the 2195 alloy were strongly related to the specimen orientation. The lowest yield strength, ultimate tensile strength, fracture toughness, and fatigue threshold were observed to occur in the specimen oriented at 45° to the rolling direction. This anisotropy is primarily attributed to the strong brass-type texture in this alloy. The observed orientation dependence of the yield strength can be explained by means of Taylor/Bishop-Hill model, and Schmid's law together with the presence of brass-type texture can predict the fatigue-crack growth direction [31, 32].

Fatigue Behaviour of Microcast-X Inconel 718

M. C. Chaturvedi, Department of Mechanical and Industrial Engineering, University of Manitoba

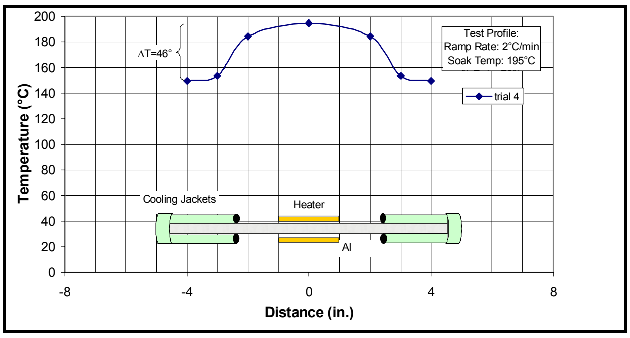
The fatigue crack growth (FCG) behavior of electron beam welded Microcast-x Inconel 718 was one of the focus of the research program. Two different metallurgical conditions were produced with pre-weld solution treatment at 1050 C and 1150 C prior to welding and thermal simulation of the heat affected zone (HAZ) by Gleeble thermo-mechanical simulator. For comparison purposes, additional testing was performed on conventionally heat treated (955°C) Microcast-X and conventionally wrought Inconel 718. The FCG rates were determined in laboratory air with a stress ratio of 0.1, frequency of 2 Hz, and at temperatures of 21°C, 350°C, and 650°C. In-phase thermo-mechanical fatigue tests were performed using the Gleeble simulator, with temperature cycled between 350°C and 650°C at a frequency of 0.1 Hz. The mixed grain size of the material produced by the 1150°C solution treatment provided a small improvement in FCG resistance over samples with smaller and more uniform grain size. FCG rates increased with temperature, and the most significant increase occurred between 350°C and 650°C as the crack growth mechanism changed from trans-granular to inter-granular. Fatigue crack propagation accelerated significantly in the fusion zone of weldments in all cases as the crack advanced inter-dendritically. The influence of HAZ micro-fissures on macroscopic fatigue crack growth rates was negligible under the conditions considered in this study. Micro-fissures would be more effective in initiating fatigue cracks than interacting with large, pre-existing cracks. The FCG resistance of Microcast-X 718 was somewhat comparable to the wrought form of the alloy at all the temperatures considered [33.

Corrosion improvement of Friction Stir Welded Aluminum Alloys

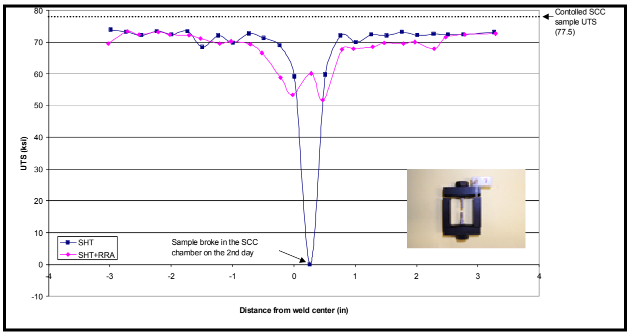
Ali Merati, SMPL, IAR, NRC

Friction Stir Welding (FSW) is a candidate process being considered for extensive use in the aircraft structures because of potential cost reductions and structural stability. The research efforts revealed that FSW joint properties, particularly corrosion performance and long term stability need further improvements to allow full potential application of the technology. In order to ameliorate the properties, the stabilization (SHT: 120°C/24hr), and Retrogression and Re-Aging (RRA) heat treatments were applied locally only to the weld region. The objective of this study is to improve the mechanical and corrosion properties of the welded joints and to reduce residual stresses by localized heat treatments.

A typical temperature profile across the weld, and the results for SSC tests on localized heat-treated 7475 alloys are shown graphically in Figure 1a, and 1b, respectively. This result indicates that for the 7475 FSW joint, the SHT+RRA combination is the most promising type of heat treatment in terms of SSC resistance. More work is needed to develop heat treatment apparatus and schedules, which will reduce the total heating time to less than the current time.



a)



b)

Figure 42. Temperature profile for localized FSW heat trial, b) SCC tensile test results after 40 days in the corrosion chamber, showing performance and residual strength for the two post weld heat treatments applied.

COMPOSITE MATERIALS AND STRUCTURES

Fatigue Performance of GLARE Riveted Splices

P. Straznicky , Carleton University

<u>Fatigue Life</u>
(collaboration between SMPL, IAR, NRC, Bombardier Aerospace and Carleton University)

Figure 43. Test rig with traveling microscope

Fatigue tests are being carried out using a riveted splice specimen developed on the basis of the work by Mueller [34. Some details of the test follow:

- 20 in wide 3-row riveted lap joint

- 1-in row and rivet pitch
- NAS1097AD4-4 rivets
- Running load of 95.3 N/mm (544 lbf/in) at 4 Hz frequency, R = 0.03

The test rig is shown in Figure 43. The specimen is instrumented with 16 strain gauges (8 each side) located nominally 25.4 mm (1”) above the upper rivet row. Fatigue crack measurements are made using a traveling microscope.

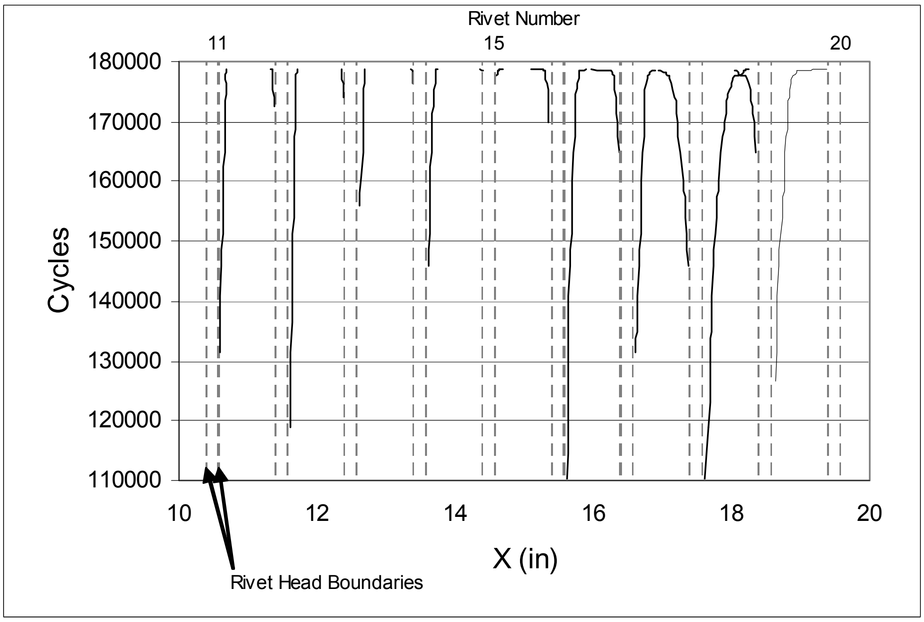


Figure 44. Progression of MSD in 2024-T3 specimen MSD-AL-004

Initially, four specimens made from Aluminum 2024-T3 were tested, the first one as a proof-of-concept, and the subsequent ones to obtain baseline data. The specimen has repeatedly produced MSD (Figure 44, Figure 45 and Figure 46), confirming its potential for studying the effects of crack interaction and link-up on the crack growth rates in riveted splices.

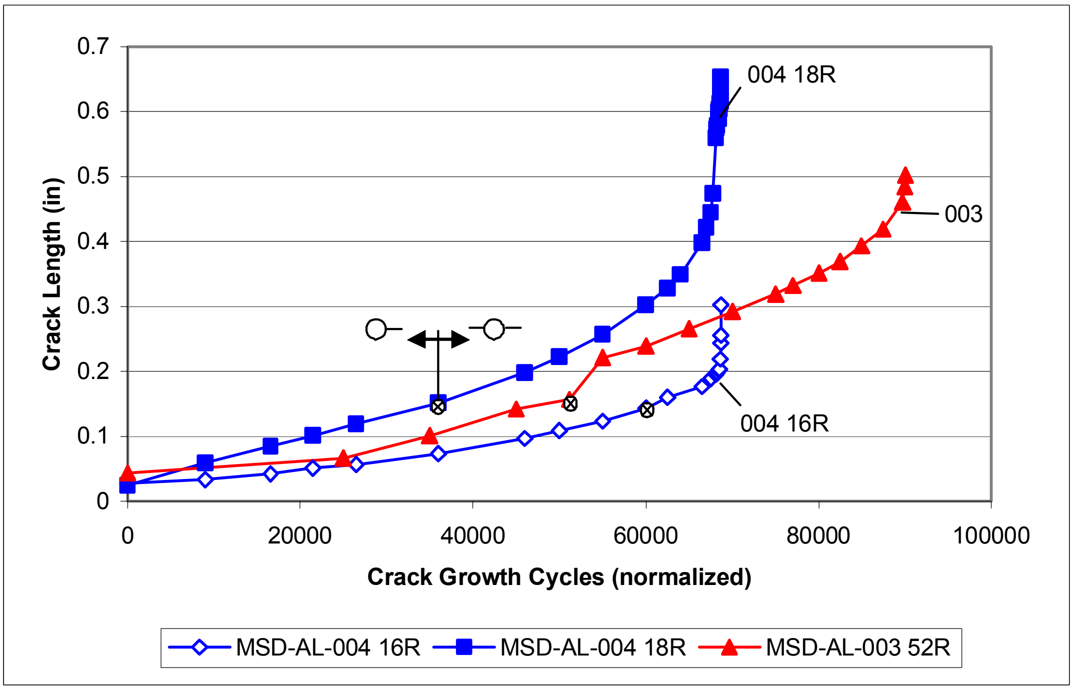


Figure 45: Crack growth curves for lead cracks in 2024-T3 specimens. Normalized to time of first detection.

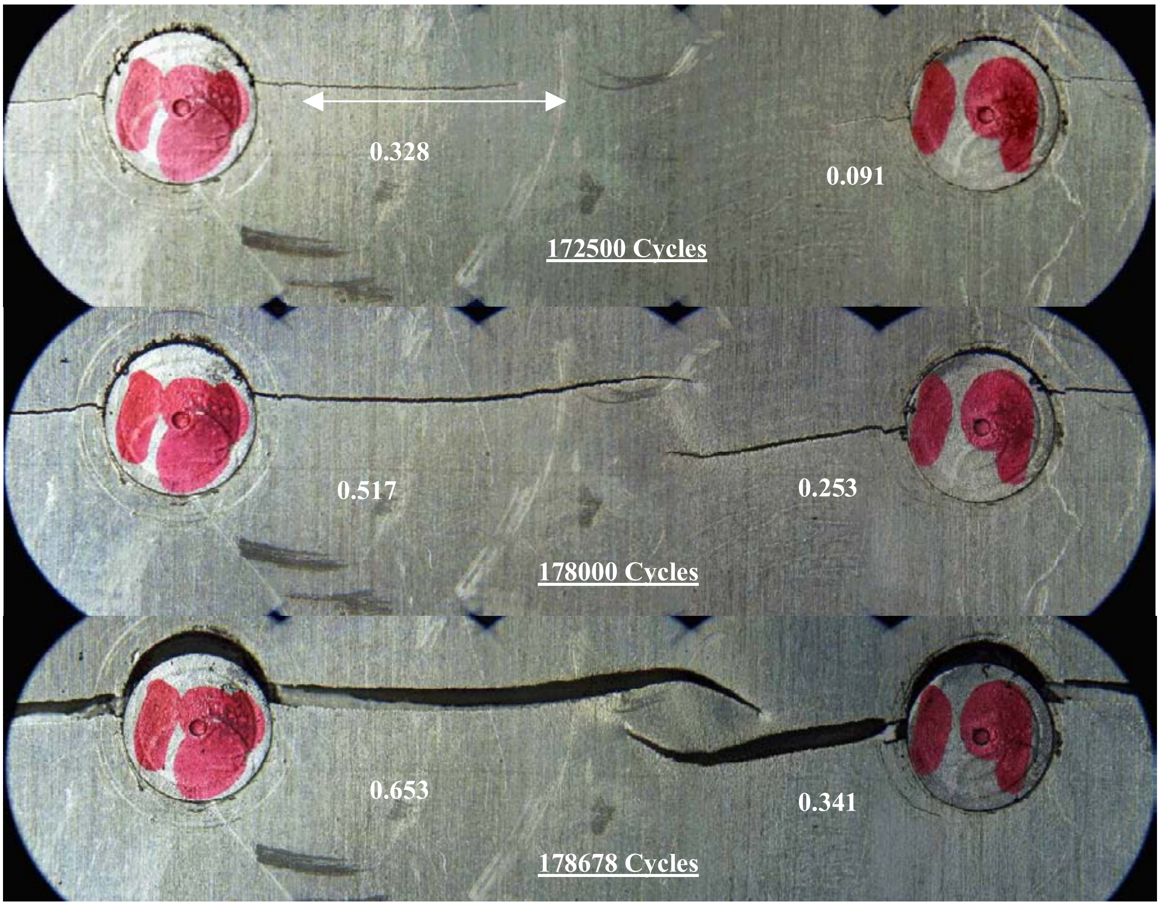


Figure 46: Crack growth and linkup between rivets 18 and 19 in MSD-AL-004

Currently, testing is continuing using 0.036 in TH GLARE3-2/1 and 0.055 in TH GLARE3-3/2 specimens. Results so far confirm the well known excellent fatigue behaviour of GLARE.

Life Extension Studies
C. Rans, Paul V. Straznicky, Carleton University

The major advantage of Glare in airframe applications is the low fatigue crack growth rate, compared to monolithic aluminum alloys. In mechanically fastened Glare splices, fatigue crack nucleation occurs early in life of the splice, and is very difficult to detect particularly in the field as it occurs in the metal plies at the faying surfaces.

The lower stiffness of the prepreg plies in Glare cause the aluminum plies to carry a higher load than monolithic aluminum alloys, resulting in early crack initiation. Furthermore, machining of the countersinks for flush-head rivets causes "knife-edges" in Glare aluminum plies. The resulting stress concentrations likely accelerate crack nucleation. If the knife-edges were avoided, this could increase the crack nucleation period thereby further increasing Glare advantage. One possible approach is the use of a formed countersink by means of dimpling.

Dimpling is currently limited to secondary structures and thin-gauge materials up to approximately 1.63mm (0.064") [35]. In aluminum alloys, fatigue life of dimpled splices is inferior to those that use the conventional countersinks, reportedly due to residual tensile stress in the outer area of the dimple, overstress during the forming process, and local stress concentration at the dimple "corner" [34]. Also, nesting of dimpled sheets at the faying surfaces changes the load transfer mechanism compared to a conventional splice.

However, in Glare, dimpling would avoid the knife-edges and, due to the small thickness of aluminum plies, would not likely cause high stresses and microcracks during the forming of the dimple. Furthermore, process variations such as reversing the sequence drill-dimple, may improve the structural behaviour by changing the residual stress distribution and by improving the surface of the drilled hole.

Therefore, an initial evaluation of dimpling in Glare is being carried out. The approach is a combination of experiment and analysis, as follows:

- Dimpling and riveting of Glare sheets, visual and metallographic inspections of joints and the surrounding sheet areas.
- Fatigue testing of single-fastener Glare coupons to investigate the effects of dimple manufacture, rivet installation and load transfer on the fatigue crack nucleation.
- In parallel, finite-element analyses of the manufacturing process (dimpling, rivet formation) to estimate residual stress distribution in Glare sheets. LS-DYNA code is used for the modelling.
- Fatigue testing of several 500mm (20") wide splice specimens with dimpled splices. The test results will be compared to the aluminum and Glare splices with countersunk rivets.
- Analysis of advantages and disadvantages of this riveting method. The disadvantages may include damage to the prepreg layers caused by dimpling, fretting failures on conical faying surfaces, and manufacturing complexities compared to the current process.

This initial evaluation will be completed by mid-2003. If it shows promising results then a more detailed program will be undertaken.

Durability Characterization of Active Fiber Composite Actuators

V. K. Wickramasinghe, SMPL, IAR, NRC

The Active Fiber Composites (AFCs) are a new structural actuator system consisting of piezoceramic fibers embedded in an epoxy matrix. The fibers are sandwiched between interdigitated electrodes to orient the driving electric field to use the primary d_{33} piezoelectric effect, as shown in Figure 47. These conformable actuators were designed to integrate directly into composite structures as active plies within the laminate to perform active shape control at high frequencies. These AFC actuators were successfully used in a 1/6-scale, Mach-scaled, active helicopter rotor wind tunnel tests by Boeing to verify vibration control through active blade twist [36]. In order to utilize AFCs in structural applications, as in the active-twist blades, it was essential to evaluate their long-term durability properties both as actuators and as structural material. Two types of long-term durability characterization tests, namely, electrical fatigue and mechanical fatigue, were conducted to extract important electromechanical properties of AFCs under simulated blade operating conditions. The characterized AFC actuators coupons shown in Figure 49 were 0.0135" thick and consisted of 0.01" diameter PZT-5A fibers with a 90% volume fraction.

Electrical fatigue tests were conducted to simulate the repeated actuation of AFCs in structural control applications. Actuators tested up to 20 million electrical cycles showed no degradation in induced strain performance. However, microscopic inspections showed evidence of damage near electrode fingers in the form of fatigue burns as shown in Figure 48, but none of the burns were large enough to cause catastrophic electrical failure in the actuator.

Mechanical fatigue tests were conducted to simulate the repeated mechanical loading expected for AFCs in structural applications. Actuators tested to 10 million cycles at the nominal load level of $1000\pm900\mu\varepsilon$ showed no significant deterioration in the modulus or actuation capability. An actuator coupon retained the stiffness and performance properties even after additional 10 million cycles at a 50% higher fatigue load level as shown in Figure 50.

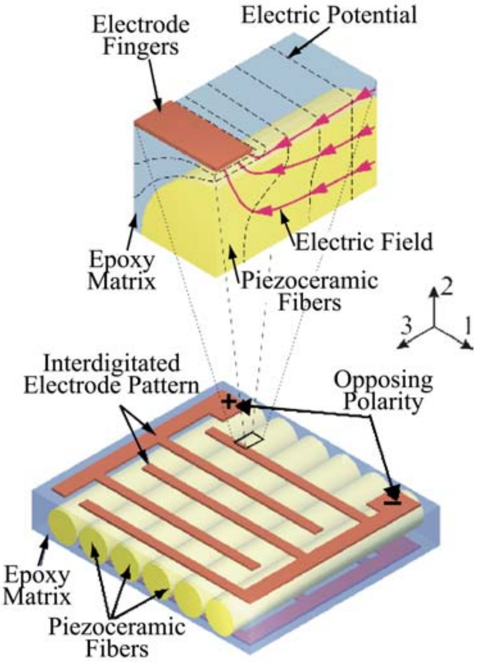


Figure 47: AFC actuator concept

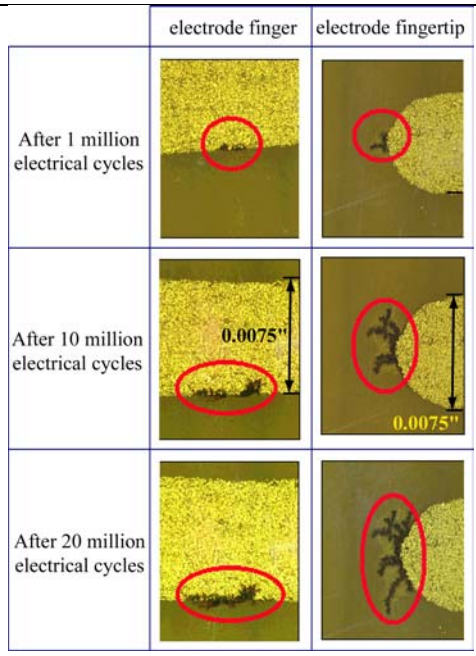


Figure 48: Electrical fatigue burns

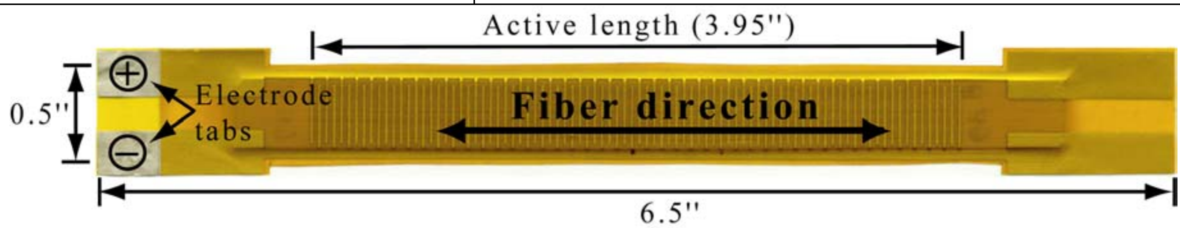


Figure 49: Baseline AFC actuator test coupon

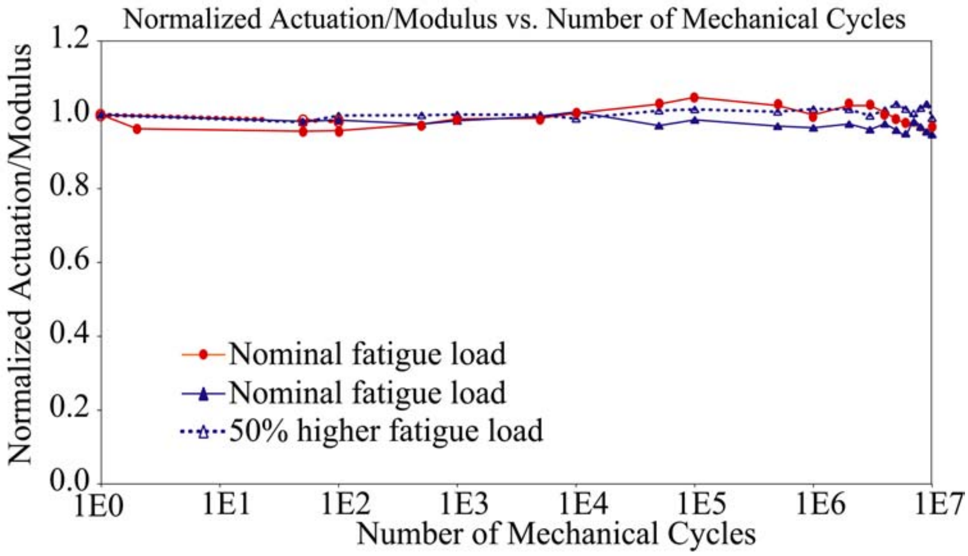


Figure 50: Mechanical fatigue test results

NON-DESTRUCTIVE INSPECTION AND SENSORS

Nondestructive Inspection of Fatigue Cracks and Corrosion
D. Forsyth, SMPL, IAR, NRC

Reliability of NDI
Research at IAR into the reliability of NDI and its impact on life cycle operation continues. NDI reliability is often characterized by the probability of detection of a flaw, as a function of the flaw size. Due to the expense of obtaining components flawed from in-service exposure, engineered test specimens are often used in experiments for estimating NDI reliability. The number of flaws and the flaw sizes to be manufactured can be optimized by a Rayleigh distribution method shown by Safizadeh [37], in order to provide the most accurate estimation of the reliability for a particular number of test specimens. This method improves over existing methods which have been published in the literature, as shown for a particular case in the Figure 51 below.

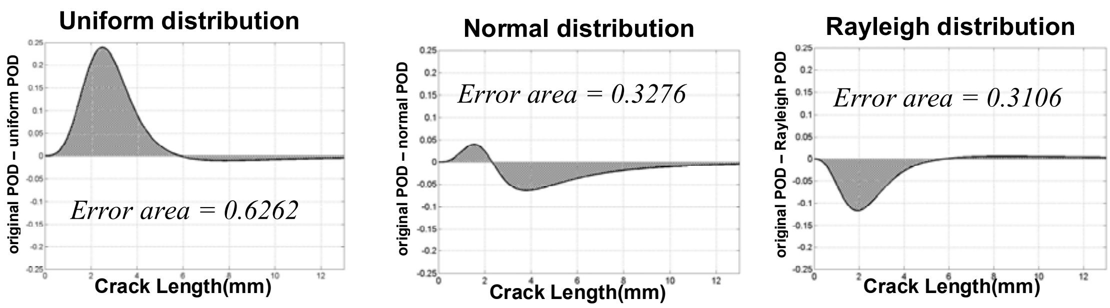


Figure 51. Errors in estimating NDI reliability from a set of engineered specimens, with the engineered flaw sizes distributed by different methods.

Data Fusion
In the case of complex structures or complex modes of degradation such as corrosion and fatigue and their interactions, often more than one NDI technique is required for complete characterization. For example, the techniques used to measure thickness loss, corrosion pillowing, and fatigue cracks in aircraft fuselage lap joints are significantly different. In order to reduce the burden on inspectors in these cases, efforts are underway to apply data fusion techniques to automate interpretation and to provide quantitative results from multiple NDI techniques.

Work at IAR has demonstrated the application of data fusion algorithms to conventional multi-frequency eddy current data, and allowed the independent estimation of the thickness of both layers of a two layer lap joint. The algorithm was developed and tested on a service-retired lap joint from a Boeing commercial aircraft, details are provided in reference [38].

The error in the data fusion result was found by comparing with thickness measured after teardown. These results were much more accurate than results from conventional NDI alone, and provided measurement of the second layer thickness which was not provided by any other NDI technique. A number of techniques were evaluated on the same specimen, and errors fit to a normal distribution. The mean and standard deviations of the errors are shown in the table below.

Table 6. Parameters of the normal distributions fit to the experimentally measured NDI error distributions.

NDI technique	mean error (inches)	standard deviation (inches)
MAUS IV ET 4kHz	0.0002	0.0022
MAUS IV ET 8kHz	0.0002	0.0020
IAR PET	-0.0001	0.0019
SAIC UT * from 0.045” thick specimen	-0.0004	0.0023
IAR fusion of MAUS IV ET - top layer -	0.0007	0.0008
IAR fusion of MAUS IV ET - bottom layer -	-0.0008	0.0011

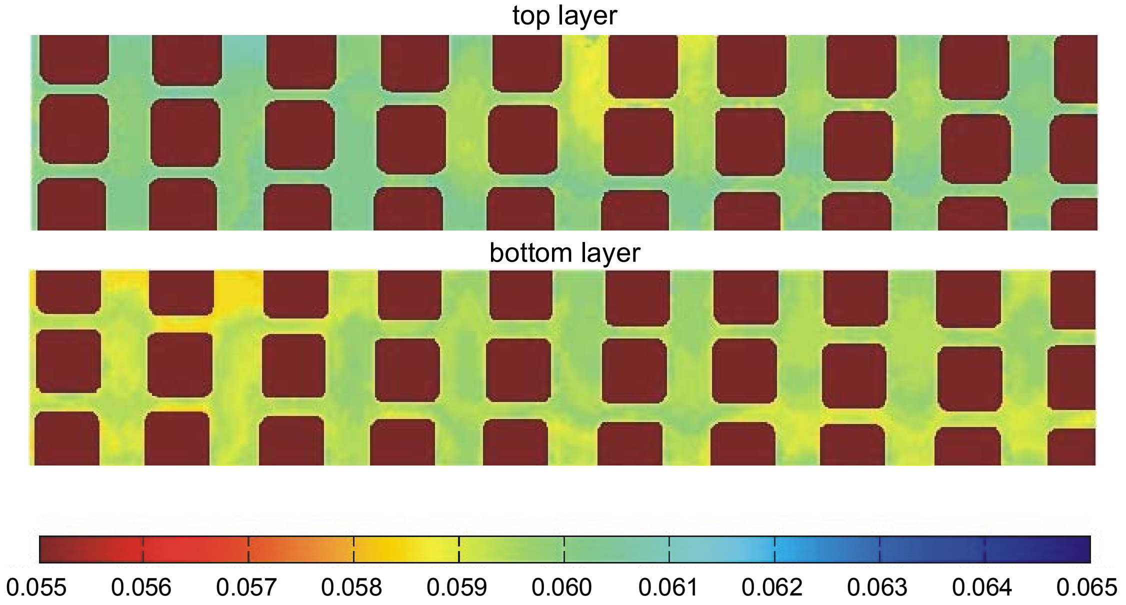


Figure 52. Results of thickness estimation on the top and bottom layers of the lap joint specimen, using data from a commercial two frequency eddy current instrument and data fusion algorithms developed at IAR.

In order to assess the importance of these errors, a simulation of the life of one bay of a lap joint in a United States Air Force K/C-135 was developed. For the criteria in this particular simulation, it was shown that the higher errors required repairs to be performed at much lower levels of damage, thus increasing the cost of operation of the component. This is shown graphically in the chart below. Details on these simulations are provided in reference [39].

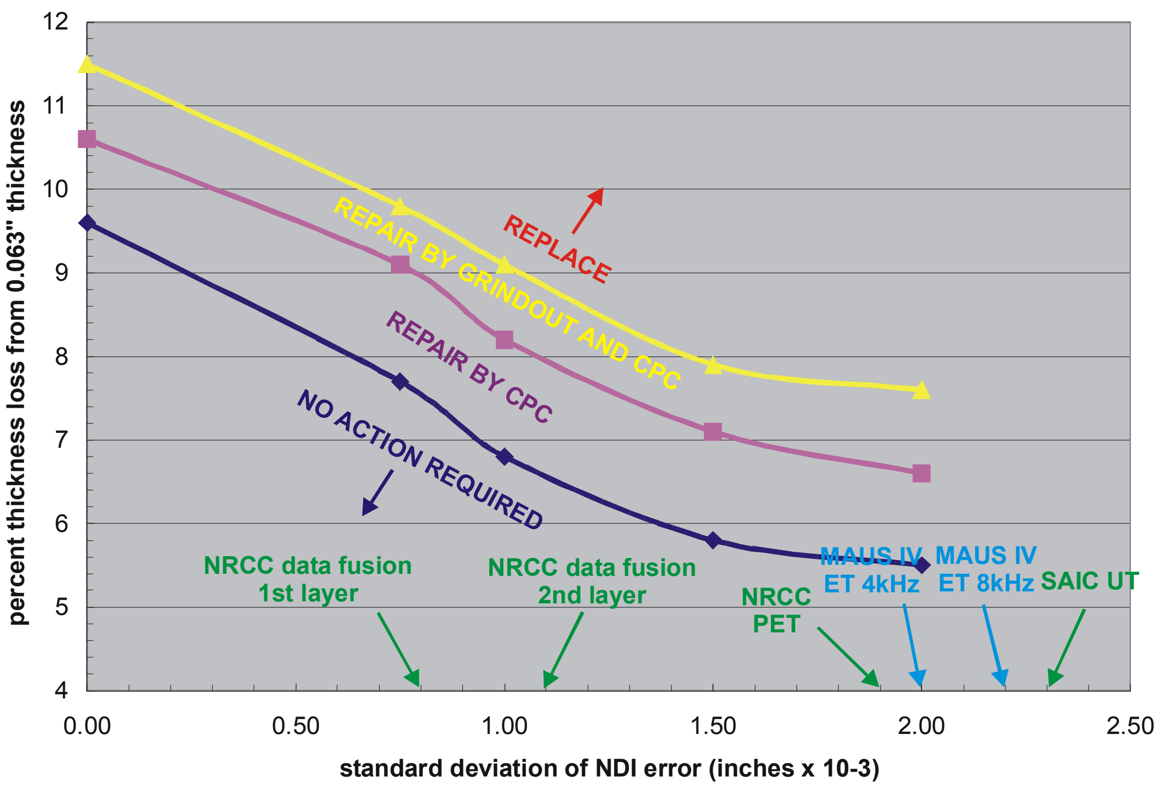


Figure 53. Effect of NDI error in measuring corrosion damage in lap joints for a particular fatigue and environmental spectra. PET – Pulsed Eddy Current Technique, UT – Ultrasonic Technique, ET – Eddy Current Technique, CPC – Corrosion Preventive Compounds.

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