A REVIEW OF RESEARCH
ON
AERONAUTICAL FATIGUE IN THE UNITED STATES

2011-2013

Compiled by
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JERUSALEM, ISRAEL

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9.1. INTRODUCTION

Leading government laboratories, universities and aerospace manufacturers were invited to contribute summaries of their recent aeronautical fatigue research activities. This report contains several of those contributions. Inquiries regarding a particular article should be addressed to the person whose name accompanies that article. The generous contributions of each participating organization is hereby gratefully acknowledged.

Government
- FAA
- NASA – Johnson Space Center
- USAF Air Combat Command
- USAF Air Mobility Command
- USAF ASC/EN
- USAF A-10 ASIP
- USAF C-17 SPO
- USAF F-16 SPO
- USAF F-22 SPO
- USAF Life Cycle Management Center
- USAF-OC-ALC
- USAF-OO-ALC
- USAF Research Laboratory – Aerospace Systems Directorate
- USAF Research Laboratory – Materials and Manufacturing Directorate
- USAF SAF/AQX
- USAF-WR-ALC
- USN-NAVAIR
- United States Forest Service

Academia
- University of Dayton Research Institute
- University of Virginia
- USAF Academy – CAStLE
- Wichita State University – Department of Aerospace Engineering
- Wichita State University – NIAR

Industry
- Alcoa Defense
- ATA Engineering
- Battelle Memorial Labs
- Computational Tools, Inc.
- Curtiss-Wright Avionics & Electronics
- Elder Research
- Engineering Software Research & Development, Inc.
- Etegent Technologies
- Fatigue Technology, Inc.
- Hill Engineering, LLC
- JENTEK Sensors, Inc.

- Jacobs ESC Group
- Lambda Technologies
- Legacy Engineering
- Lockheed Martin Corporation
- Mercer Engineering Research Center (MERC)
- MSC
- NexOne, Inc.
- Northrop Grumman Corporation
- QUEST Integrated, Inc.
- SAFE Incorporated
- Science Applications International Corporation
- Sfhire
- Southwest Research Institute
- Spirit AeroSystems, Inc.
- The Boeing Company – 787 Program
- The Boeing Company – Commercial Airplanes
- The Boeing Company – Defense, Space & Security
- The Boeing Company – F-15 Program
- The Boeing Company – P-8A Program
- The Boeing Company – Research & Technology
- The Boeing Company – Test & Evaluation
- Tom Brussat Engineering, LLC
- TRI/Austin, Inc.

References, if any, are listed at the end of each article. Figures and tables are integrated into the text of each article.

The assistance of Jim Rudd and Pam Kearney, Universal Technology Corporation, in the preparation of this report is greatly appreciated.

One of the goals of the United States Air Force is to reduce the maintenance burden of existing and future weapon systems by eliminating programmed repair cycles. In order to achieve this goal, superior technology, infrastructure and tools are required to only bring down systems when they must be repaired or upgraded in order to preserve safety and effectiveness. This requires a condition-based-maintenance capability utilizing structural integrity concepts (CBM+SI). Knowledge is required for four Emphasis Areas: 1) Damage State Awareness, 2) Usage, 3) Structural Analysis and 4) Structural Modifications (Figure 9.1-1). The following nine Technology Focus Areas are identified to provide this knowledge: 1) Non-Destructive Inspection/Evaluation, 2) Structural Health Monitoring, 3) Structural Teardown Assessments, 4) Loads and Environment Characterization, 5) Characterization, Modeling and Testing, 6) Prognostics and Risk Analysis, 7) Life Enhancement Concepts, 8) Repair Concepts, and 9) Replacement Concepts. The aeronautical fatigue research activities of this report have been categorized into these nine Technology Focus Areas, plus a tenth category titled “Overviews” that cuts across two or more of the nine Technology Focus Areas.

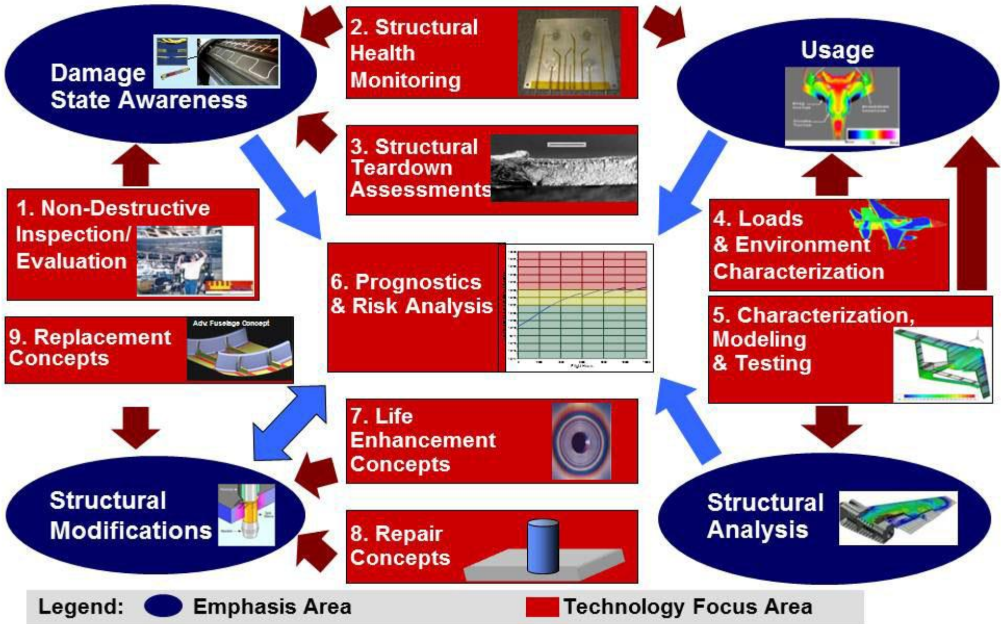


Figure 9.1-1. Condition Based Maintenance + Structural Integrity (CBM+SI)

9.2. NON-DESTRUCTIVE INSPECTION/EVALUATION

9.2.1. GMR System and Sensor Optimization for Crack Detection in Thick and/or Multi-Layer Structure

Donald Palmer, Jr., The Boeing Company – Research & Technology; Nancy Wood, The Boeing Company – Defense, Space & Security; and Charles Buynak, USAF Research Laboratory – Materials & Manufacturing Directorate

Detection of cracks in aging aircraft continues to be a major concern from a structural integrity standpoint. This is especially the case for thicker structure, where manual nondestructive inspection methods are frequently used. Often times, these manual methods require the removal of fasteners or partial disassembly in order to gain access to perform the inspection to a reasonable level of reliability. The Boeing Company has been involved with the United States Air Force relative to the development of giant magnetoresistive (GMR) sensor arrays for nondestructive evaluation applications for more than a decade. These sensors have shown to be beneficial for applications that require the detection of flaws in thick and/or multi-layer structure (Figure 9.2-1). In 2009, a prototype GMR scanning system was developed and delivered to the United States Air Force for evaluation. Based on a favorable evaluation, additional development ensued in order to optimize the prototype and produce a “productionized” system for integration into programmed depot maintenance (PDM) operations. This technical effort includes data collected on standards that show GMR sensor response as a function of thickness and flaw size (Figure 9.2-2). Also included are discussions on (1) system optimization measures taken to improve data interpretation (Figure 9.2-3) and (2) efforts to optimize GMR sensor array configurations in order to minimize scanning necessary for maximum coverage. In addition, validation results collected from wing splice areas are included.

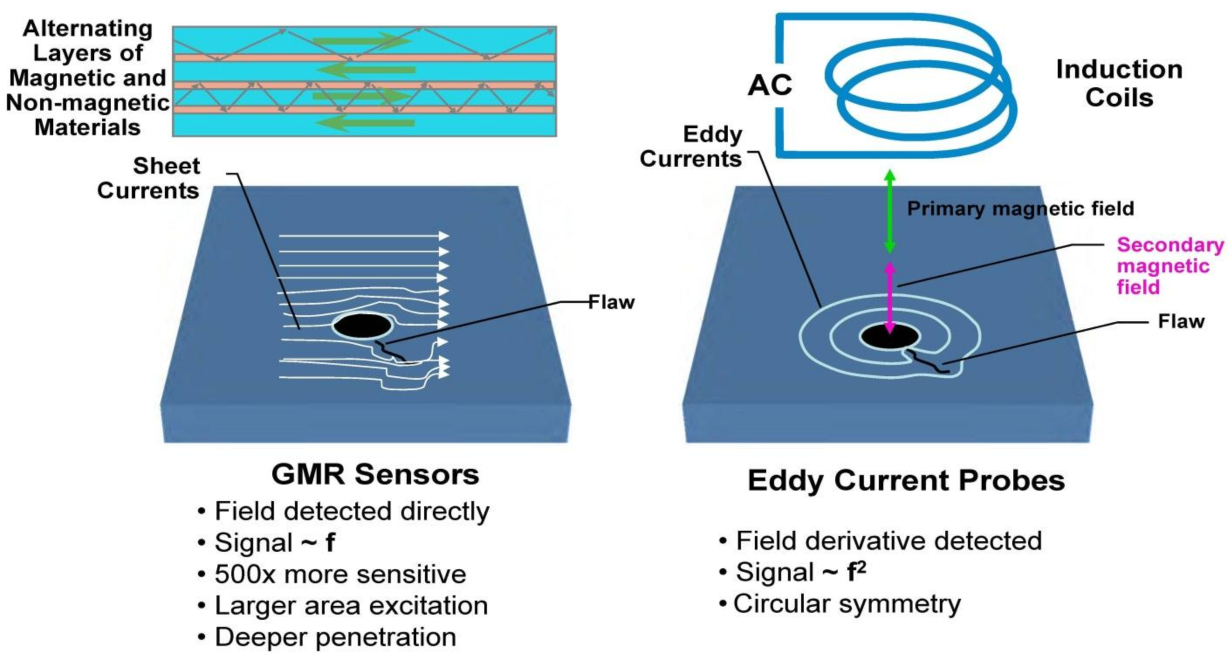


Figure 9.2-1. Magnetoresistive Sensors vs. Eddy Current Probes

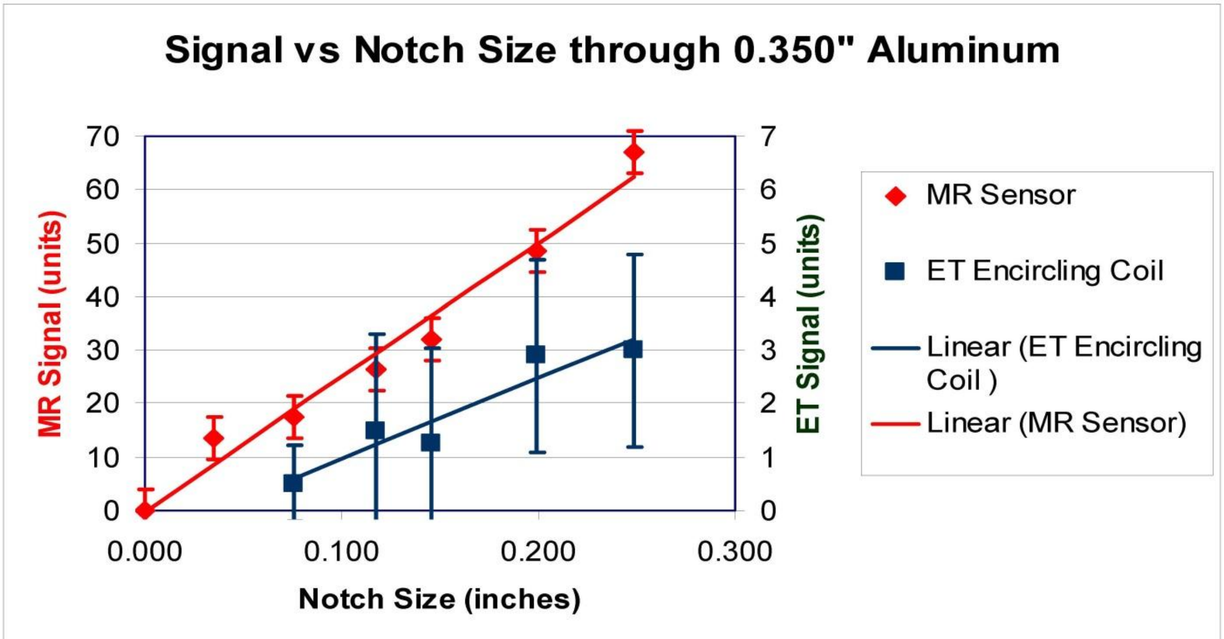


Figure 9.2-2. Quantitative Comparison Through Thick Aluminum

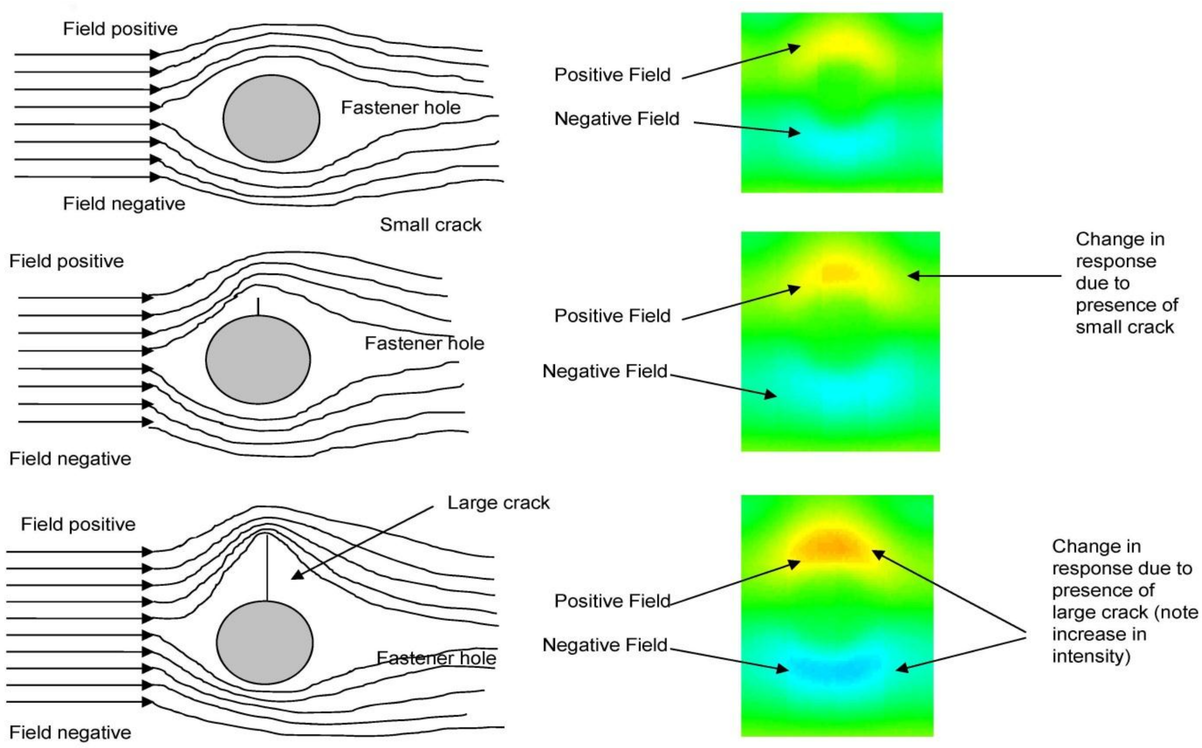


Figure 9.2-3. Magnetoresistive Sensor Image Interpretation Near Fasteners

9.2.2. POD Studies for Embedded Eddy Current Sensors

Neil Goldfine, Yanko Sheiretov, David Grundy, and David Jablonski, JENTEK Sensors, Inc.; Floyd Spencer, Sfhire; Dennis Keene, USAF-WR-ALC; and Tiffany LeMasters, Lockheed Martin Corporation

Nondestructive Inspection (NDI) for difficult-to-access locations, such as in fuel tanks and under repairs, is costly, impacts aircraft availability, and can introduce collateral damage from disassembly needed to gain access for inspections. Embedded eddy current testing (ET) methods, such as the MWM-Array by JENTEK (Figure 9.2-4), have been successfully demonstrated in hundreds of coupon tests and several full-scale tests. Recently, costs for implementation of targeted solutions have come down by an order of magnitude, making these methods practical in the near-term. However, to implement embedded ET as a replacement for NDI, requirements must be supported by an equivalent performance evaluation and associated confidence to enable integration with ASIP practices. Also, durability and reliability must be demonstrated in fuel tanks and other harsh environments that can benefit from this approach; and to justify deployment, the costs and benefits must be defined and proven. This technical effort focuses on the results of a recently completed ASIP F-16 funded environmental study (Table 9.2-1), previous durability studies (Figure 9.2-5), and a complementary Air Force Phase II SBIR Program directed at generation of POD curves (Figure 9.2-6) for embedded MWM-Rosette eddy current sensors (Figure 9.2-7) for inspection of holes, with a focus on F-15 and other Air Force platforms.

Figure 9.2-4. Example Linear MWM-Arrays

Table 9.2-1. Environmental Test Results Summary

Test	Format	Duration	Survive?
JP-5 Fuel Immersion	Sealed with Cable	17.5 days	Yes
	Sealed	35 days	Yes
	Other sleeving materials	35 days	Yes
Salt Fog Exposure	Sealed	200, 300, 400, 500 hrs	Yes
	Sealed with Cable	500 hrs	Yes
	Bare leads	200 hrs	Yes
	Bare Connector	100 hrs	Yes
Cleaning Fluid	Bare, MEK	½ day	Yes
	Bare, DS-108	½ day	Yes
Hydraulic Fluid	Bare, Royco 782	7 days	Yes

- **All sensors passed all environmental tests without any noticeable change in their electrical characteristics**

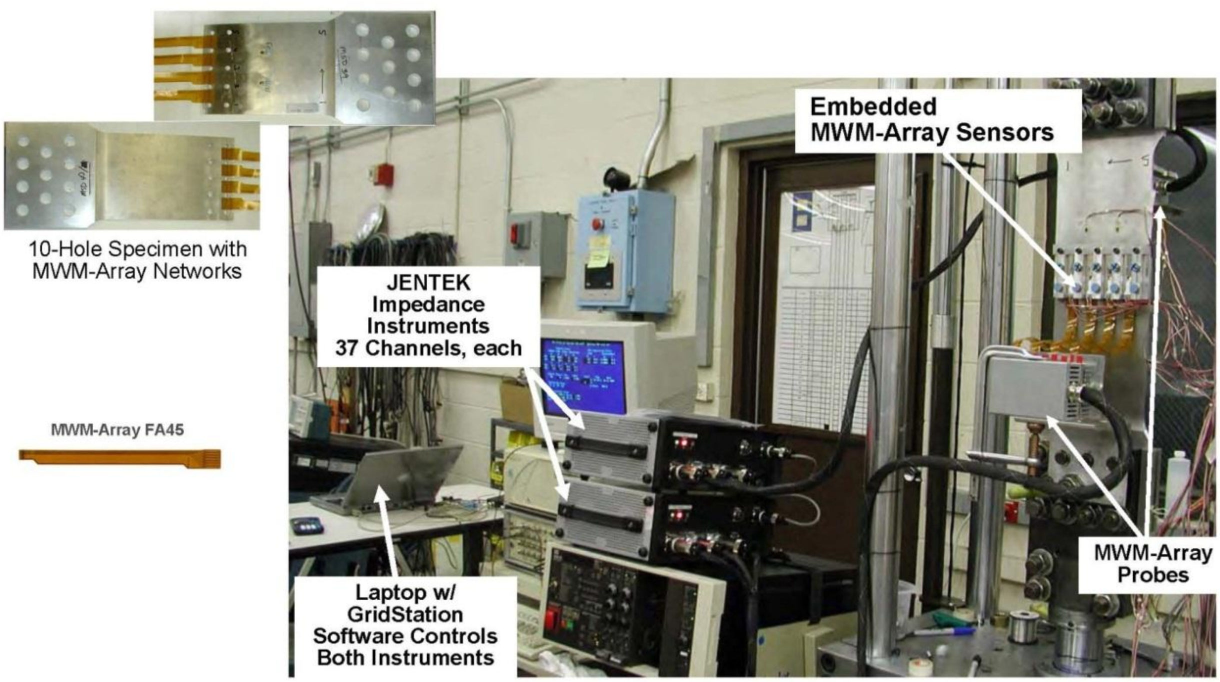


Figure 9.2-5. Durability Testing: Previous Results for Multi-Site Fatigue Testing

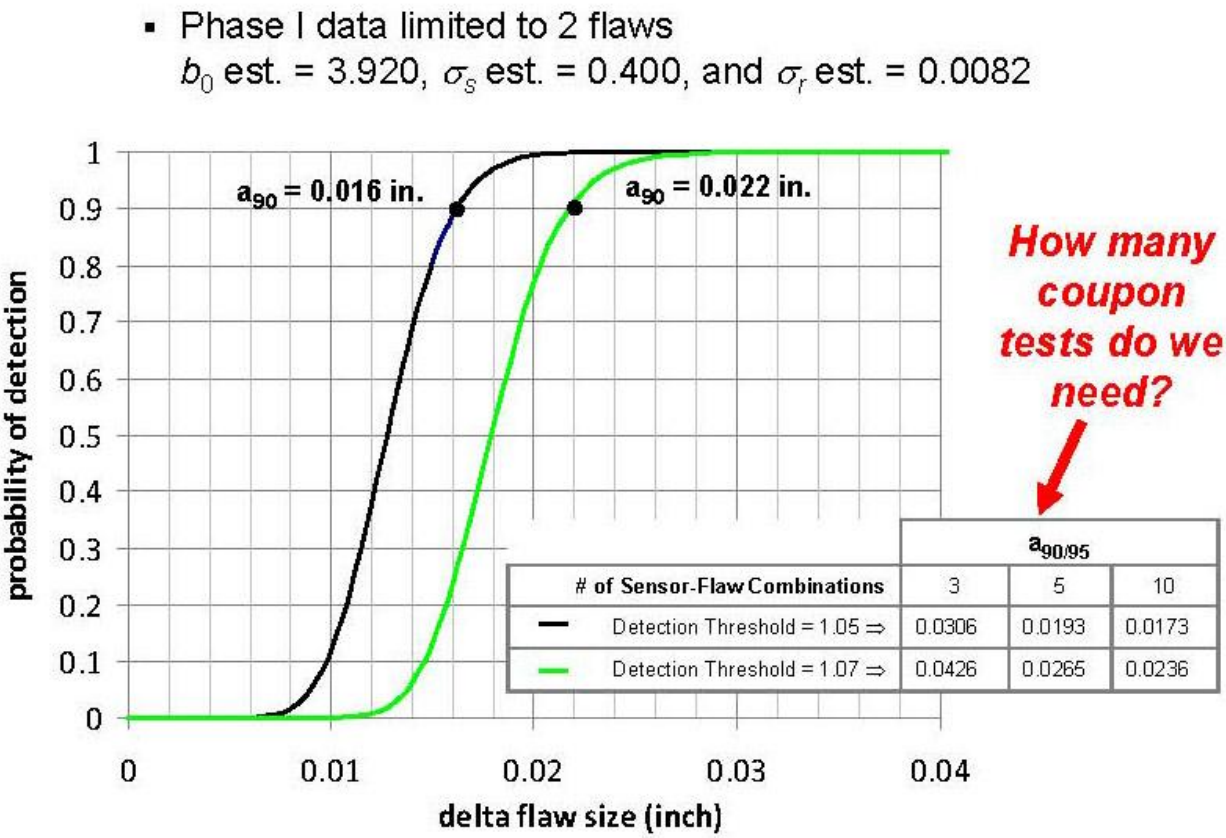


Figure 9.2-6. First POD Curves for Embedded Eddy Current Sensors Using Phase I Coupon Data

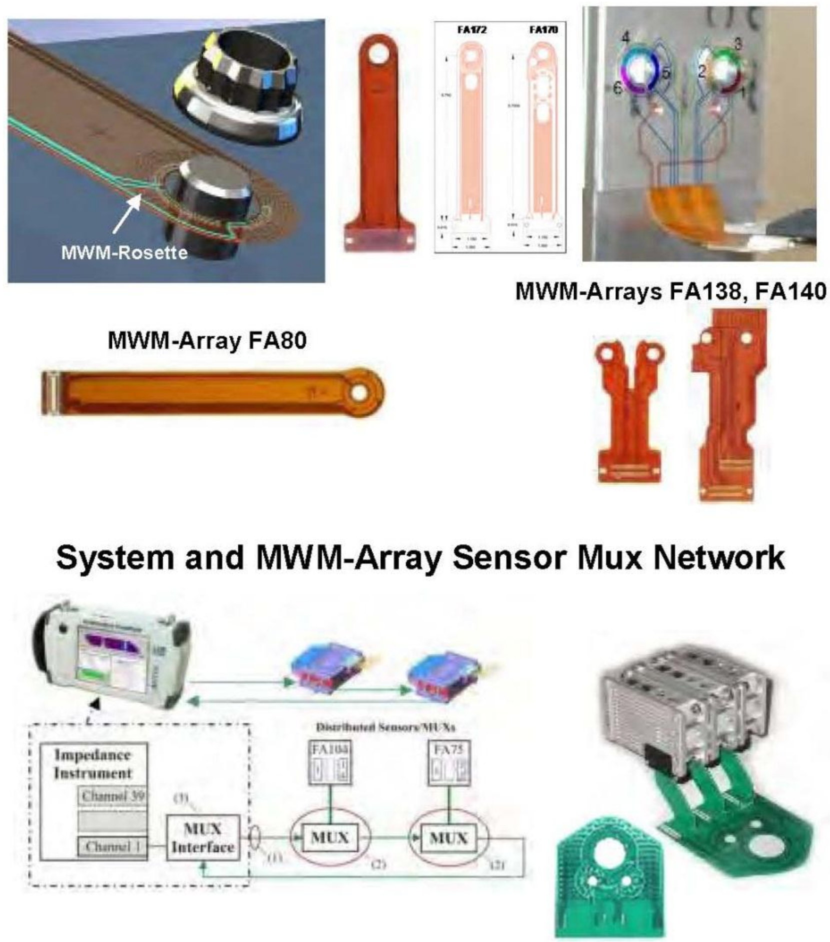


Figure 9.2-7. Example MWM-Rosettes & Integrated Solutions

9.2.3. Multi-Modal Material Property Measurements in Ti-6Al-4V Using Field Portable XRD and Optical Strain Measurement Methods

Trey Gordon, Robert Weiss, Kevin McCrary, Jim Pillers and Richard Bossi, The Boeing Company – Research & Technology

There has been an increasing demand for field assessment of material state condition and quantifying the effects of fatigue on the remaining useful life of structural components. Of particular interest are the characterization of the residual stress throughout a structure's life cycle as well as the real-time measurement of strain levels over a 3D surface of the structure. X-ray diffraction is a traditional technique for measuring residual stress and it is commonly performed on relatively small samples in a laboratory setting. Portable, commercial systems are now available that perform x-ray diffraction residual testing in the field for structures of complex geometries (Figure 9.2-8). This type of a system was optimized and used for the Ti-6Al-4V samples in this study (Figure 9.2-9). With the advent of high resolution, affordable digital cameras and computer systems, optical strain measurements using digital image correlation have become routine to measure strain over an entire part surface, not just at discreet locations as with strain gages. These systems are portable as well, and can be set up at a fatigue test facility and can be used to dynamically measure the strain levels in "real time." This technical effort describes the integration of these two methods (XRD) applied to a Ti-6Al-4V sample load and fatigue test. XRD was used at the end of each fatigue cycle and the stain levels were measured optically during the cycling. The XRD results showed a decrease in residual stress fields (Figure 9.2-10) during the testing and the optical measurements showed changes in strain levels during the cycling and at part failure.

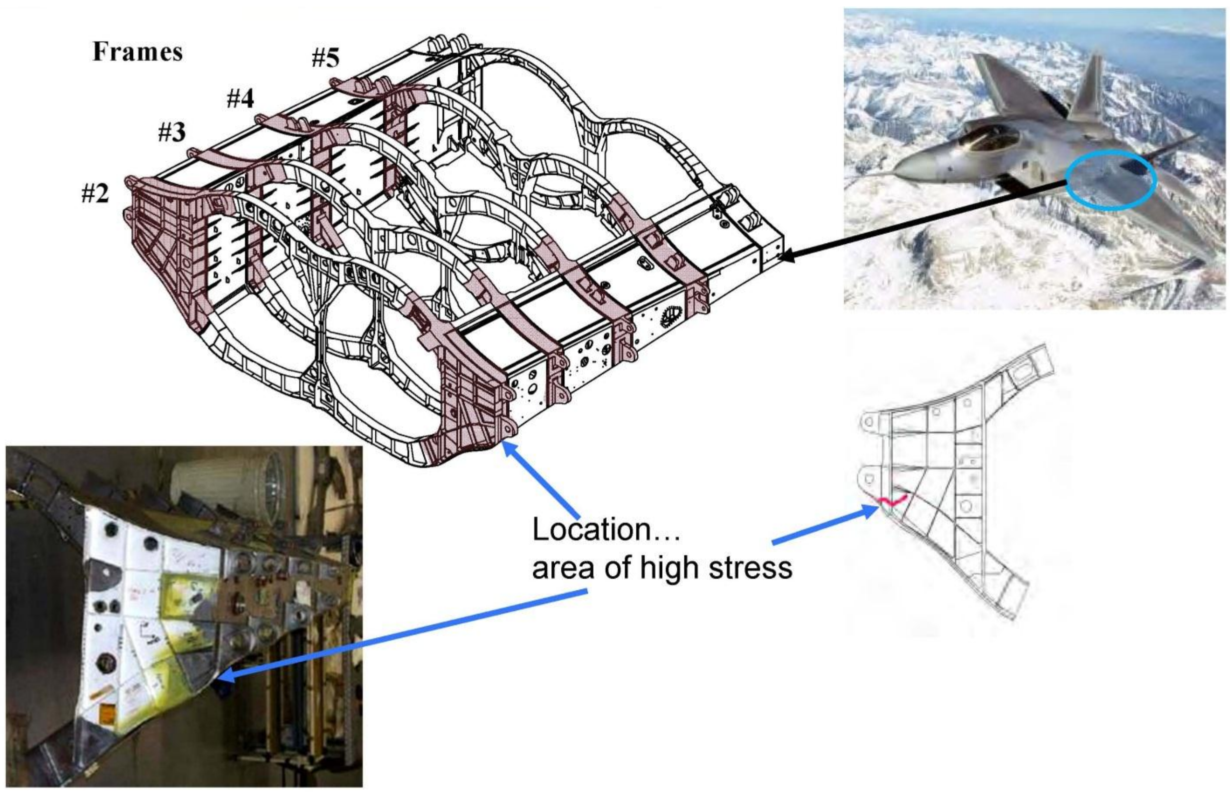


Figure 9.2-8. Titanium Airframe Application

Two Titanium Ti-6Al-4V Cantilevers

- Designed to achieve the same stress levels as airframe
- C-26 …peened from beginning
- C-27 …fatigued first for 3 cycles, then peened

Measurements

- X-Ray Diffraction baseline measured
  - Pre peening
  - Post peening
  - After each fatigue cycle block
- Strain measured with optical image correlation (ARAMIS)
  - During each block on test frame

Time of Failure

- C-26: Broke during 16th block
- C-27: Broke during 8th block

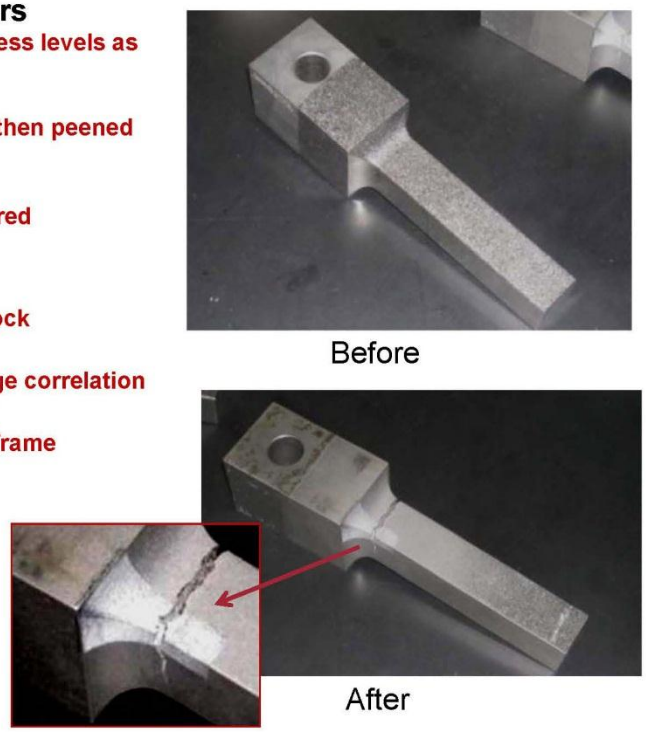


Figure 9.2-9. Evaluation of Two Titanium Samples

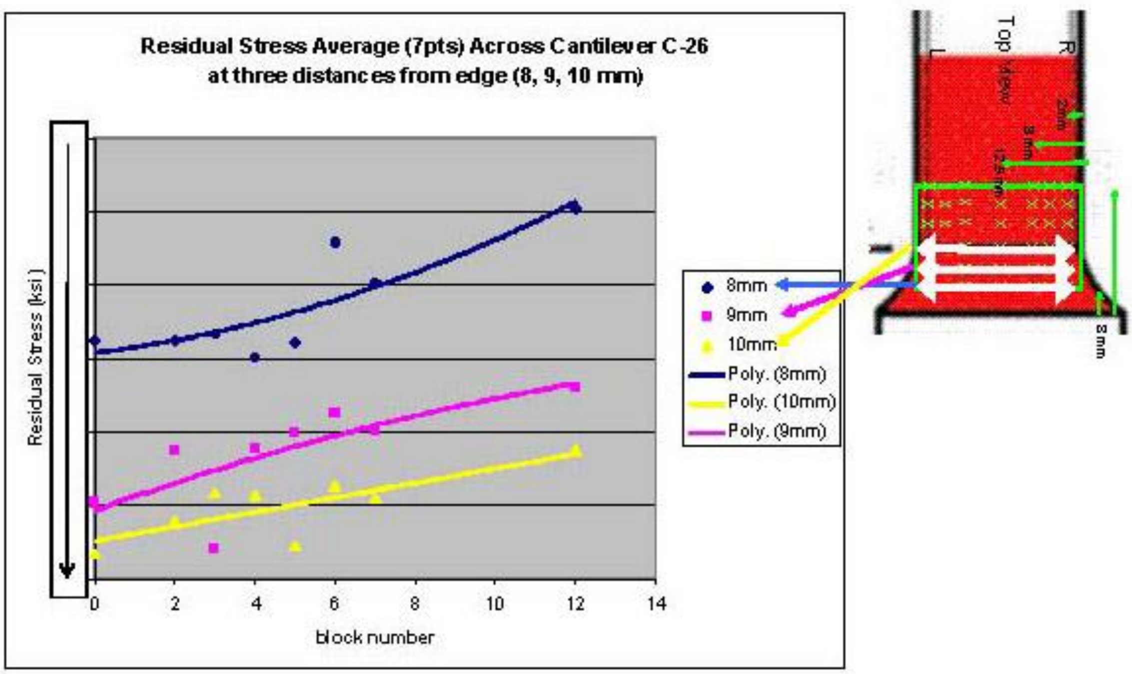


Figure 9.2-10. Successful Measurement of Residual-Stress Collapse Throughout Fatigue Test

9.2.4. Aircraft Management and Sustainment Using NDI Data Trending and Mapping Technologies

Gary Steffes, Joshua Shearer, and Steven Turek, USAF Research Laboratory – Materials & Manufacturing Directorate; Ward Fong, USAF-OO-ALC; and Thomas Sharp and Gary Cayan, Etegent Technologies

Nondestructive Inspection (NDI) methods, procedures, and enabling technologies are being used increasingly to detect flaws and defects in the United States Air Force's aging aircraft inventory. These techniques have become essential to help assure structural and functional integrity, safety, and cost effective sustainment of Air Force systems, during both initial manufacture and operational service. Advances in NDI technology and an increased reliance on NDI methods have resulted in a data explosion, but these digital data are being generated without systems in place to manage and archive the collected information. In many cases, this valuable NDI information is lost between the inspectors that collect the data and the engineers that manage the weapons systems. As part of the NDI Digital Thread for the Aircraft Production and Sustainment Program, a prototype system previously developed to collect, archive, and map NDI data will be integrated into engineering processes to provide a seamless method for actively managing aircraft systems. This technical activity will provide the details on a software system called NLign, which has the capabilities to collect, organize (by aligning to CAD models) and analyze fleet NDI data (Figures 9.2-11 through 9.2-13). These capabilities provide engineering functions with effective methods for fleet trending, improving disposition processes by providing accurate damage location to easy access to historical dispositions, and process control during asset repairs. The focus of this technical activity will be on the development and integration of the NLign software into Air Force depot processes, and applications of the software will also be provided.

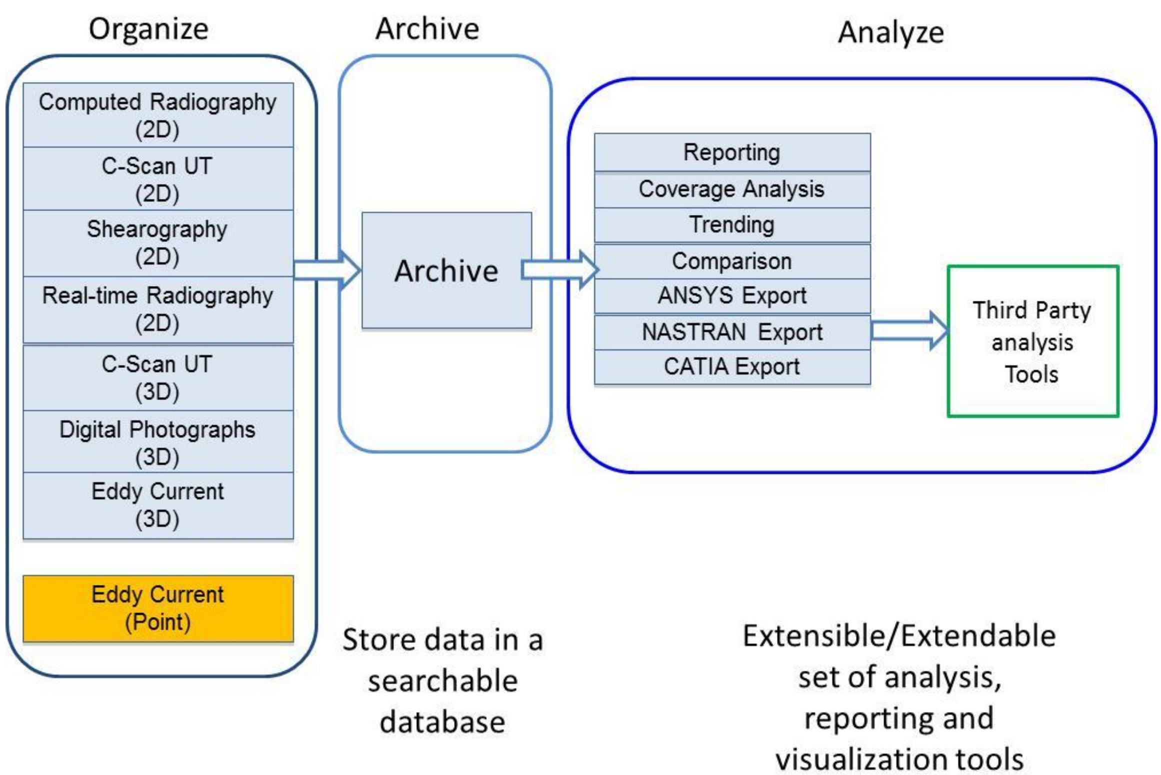


Figure 9.2-11. NLign System Overview

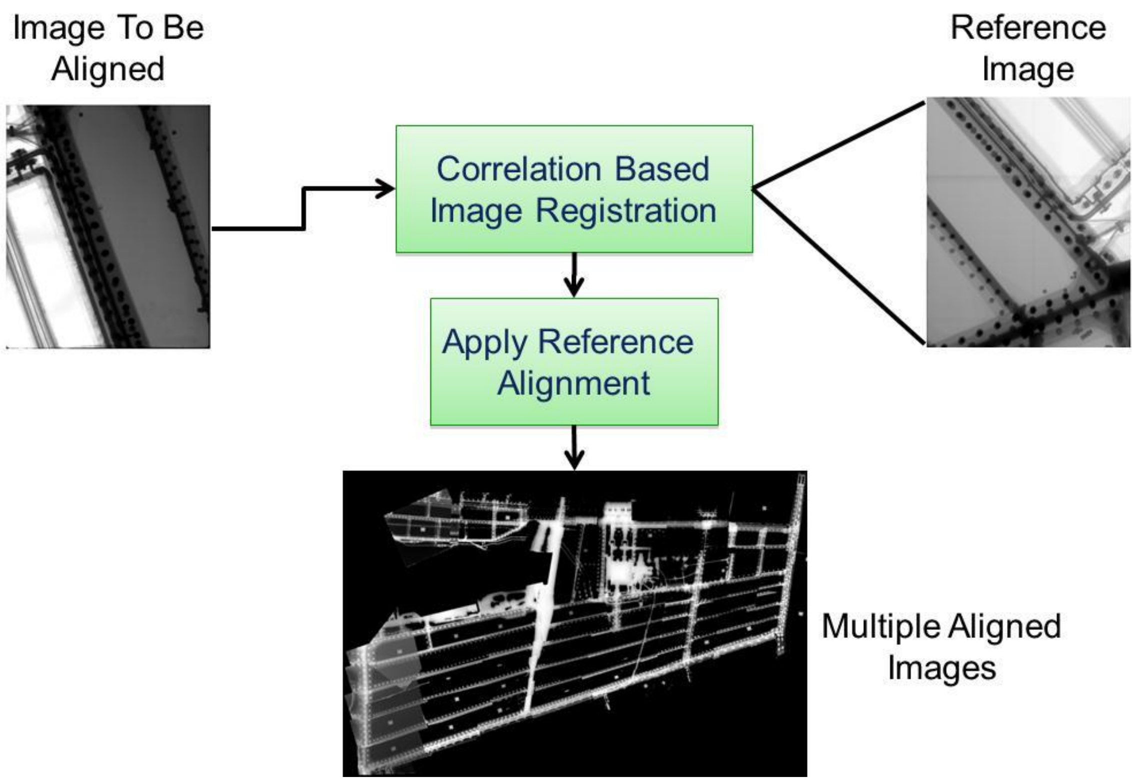


Figure 9.2-12. Example of 2D Alignment

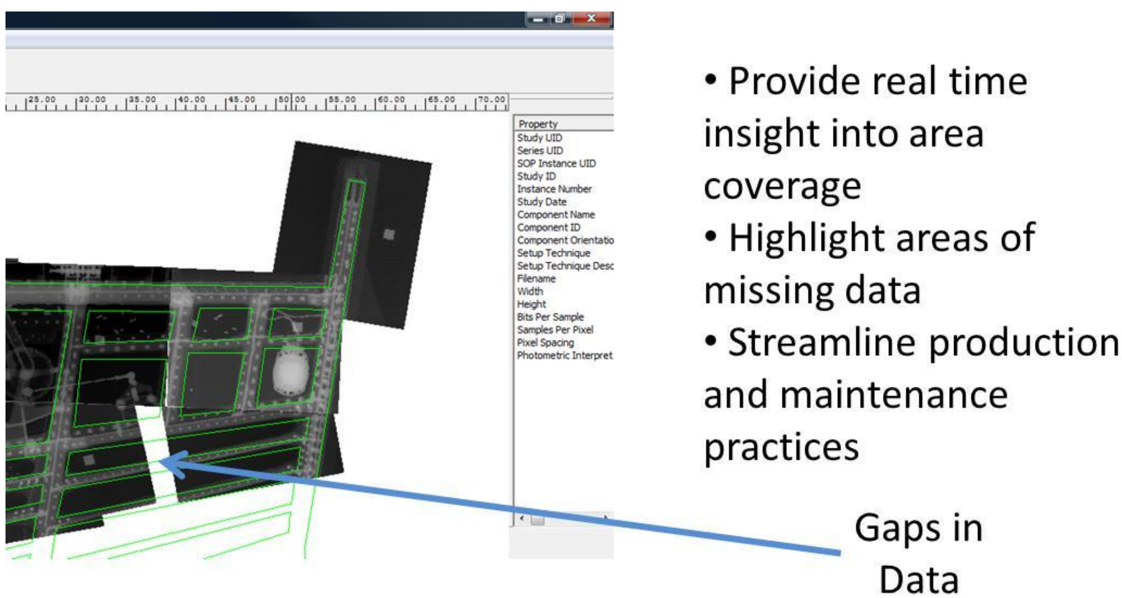


Figure 9.2-13. Analysis – Coverage Verification

9.2.5. Results of GMR Sensing Array Technique Study for the Inspection of Multi-layer Metallic Structures

Doyle Motes, David Forsyth, Mark Keiser, and Michael Mazurek, TRI/Austin, Inc.; Gary Steffes, USAF Research Laboratory – Materials & Manufacturing Directorate; John Aldrin, Computational Tools, Inc.; and Floyd Spenser, Sfhire

Recently, eddy current sensors incorporating Giant Magnetoresistive (GMR) linear sensing arrays have been developed to detect fatigue cracks in thick, multi-layered metallic aircraft structures (Figure 9.2-14). The successful deployment of these would significantly reduce depot inspection times over present man-hour intensive methods; minimize unnecessary component disassembly, repair, and/or premature airframe retirement. Several GMR modeling efforts have been conducted, and small scale experiments have been completed with specimens containing EDM notches, but larger scale validation studies have not yet been performed. As part of the GMR sensor validation program being conducted by the United States Air Force Research Laboratory, several large sets of configurable fatigue crack specimens were fabricated in different thicknesses to provide a wide range of inspection targets for candidate sensors. These specimens were mounted to a large assembly frame, simulating an aircraft wing structure, and inspected using a GMR sensor deployed on Boeing's Mobile Automated Scanner (MAUS) (Figure 9.2-15). Probability of Detection (POD) data from the inspections examining such variables as detection depth and the effects of fastener material are presented. In addition, mechanisms developed to address specimen edge effects, fastener magnetization, and signal from one crack seen in adjacent holes is addressed (Figure 9.2-16). Finally, a mockup of a splicing plate was built, scanned, and the POD analysis are discussed.

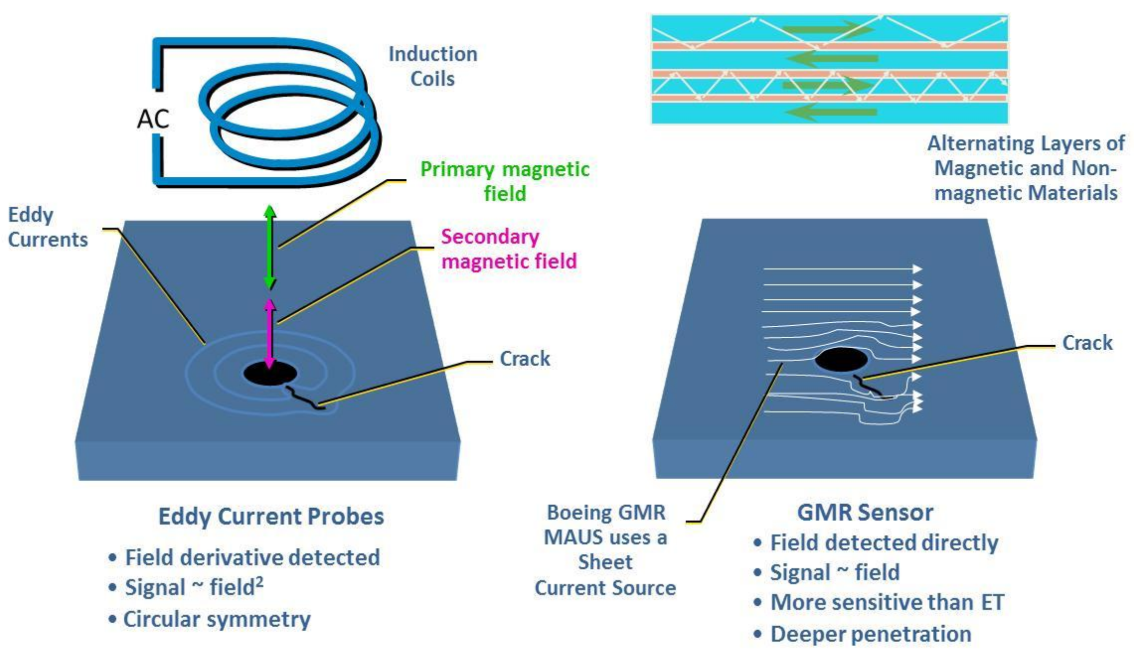


Figure 9.2-14. Eddy Current Probes vs. GMR Sensors

- Purpose: Provide AFRL with fatigue cracked specimens and POD data to evaluate a GMR array on a MAUS unit (inspection target is WS 360 on the KC-135)

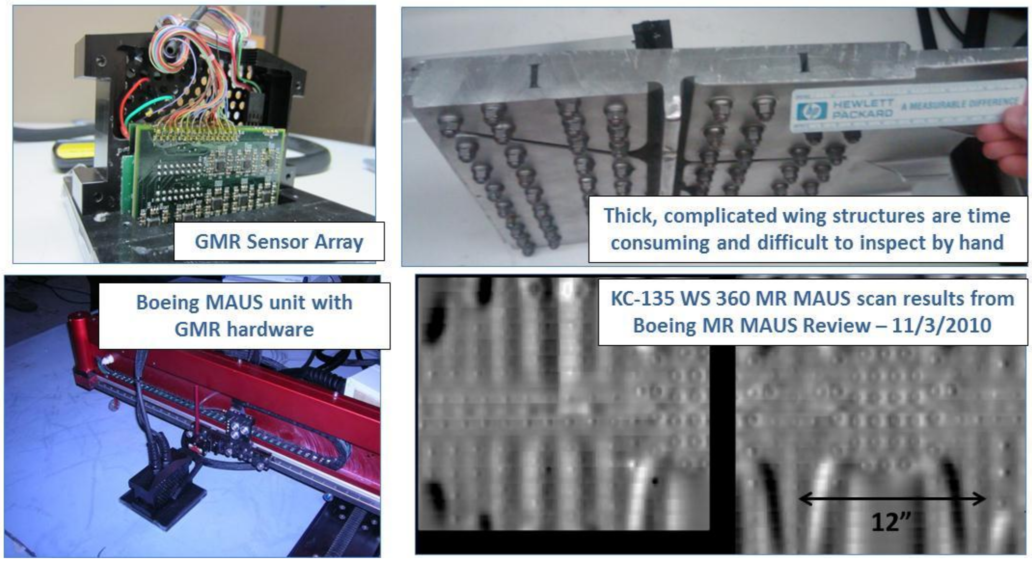


Figure 9.2-15. Purpose of Study

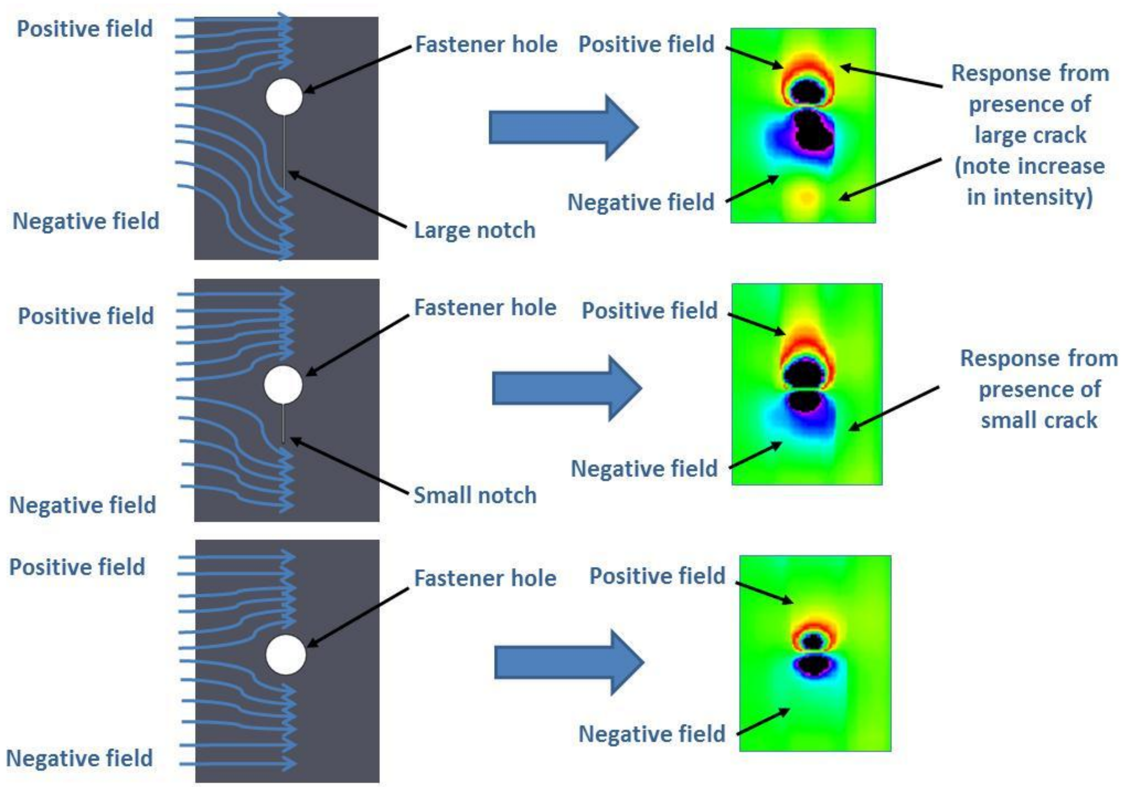


Figure 9.2-16. GMR Interpretation Near Fastener Sites

9.2.6. Challenges and Lessons From Conformal Eddy Current Probe Acquisition and Implementation

Kimberli Jones, Bryce Harris and Jacklyn Killian, USAF-OO-ALC

The latest revision of the Nondestructive Inspection (NDI) Capability Guidelines for United States Air Force (USAF) Aircraft Structures (EN-SB-08-012 Rev B) further defines the recommended NDI flaw size capabilities for computing the reinspection intervals for structures managed by the USAF Aircraft Structural Integrity Program (ASIP). USAF aircraft are expected to be in full compliance with this structures bulletin; a presentation at the 2009 ASIP Conference detailed the initial impacts to the F-16 (Figure 9.2-17), while a more recent document from the 2012 Aircraft Airworthiness and Sustainment Conference described the motivation and efforts to be compliant with the structures bulletin. The original version of EN-SB-08-012 was released in October 2008; almost four years later, significant advances have been made towards selecting and/or designing appropriate conformal eddy current probes for an F-16 NDI probe kit to be used in the field (Figures 9.2-18 and 9.2-19), as well as planning how kits will be procured, inspector training accomplished and technical orders updated. The focus of the technical activity will be on the F-16 ASIP efforts to stocklist, fund, purchase, and field the new conformal eddy current probe kits in order to be in compliance with the latest USAF NDI structures bulletin. Estimated timelines for complete probe kit implementation will be provided. These efforts were not without challenges, and the lessons learned by the F-16 will be discussed for the benefit of other weapon systems.

Figure 9.2-17. F-16 Aircraft

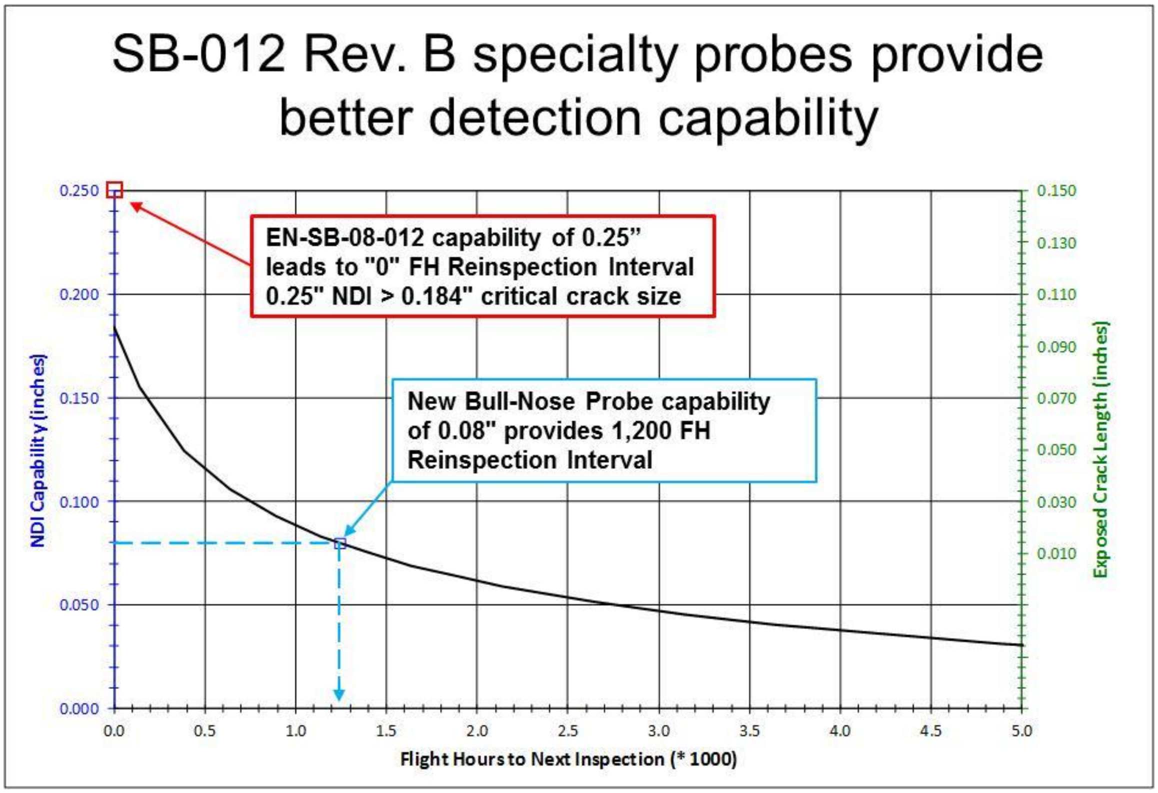


Figure 9.2-18. Upper Bulkhead Aft Vertical Stiffener Example

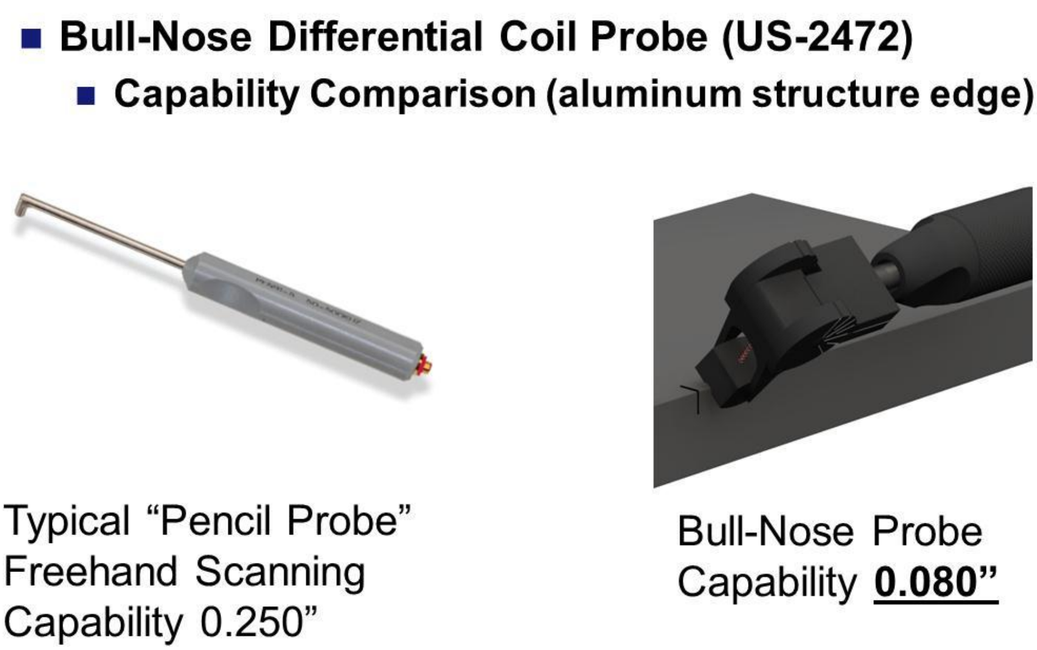


Figure 9.2-19. Probe Kit Details

9.2.7. Enhanced Magneto-Optical Imaging System for Fatigue Crack Inspection

Qingying (Jim) Hu and Phil Bondurant, QUEST Integrated, Inc.

QUEST Integrated's Magneto-Optic Imaging (MOI) eddy current system is primarily a screening tool used to rapidly detect cracks and defects in metallic aircraft skins. It is based on the Faraday Effect that describes the change in polarization from the interaction of the magnetization of the material with light propagating through the material. When a crack exists, a vertical magnetic field will be generated from the "curl" of the eddy current around the crack and results in local light polarization changes that will be visible in the optical viewing system. MOI combines the high sensitivity of the eddy current technique and the fast measurement speed of optical visual inspection into one inspection system. Benefits and features of Magneto-optic imaging include:

- Sensitivity to flaws in any orientation
- Real-time (30 frames/second) magnetic field image
- Large visualization area (up to ~50 mm circle) for a single video frame
- Imaging head can be moved in any direction
- Calibration not required
- Minimal operator training required
- Paint stripping or protective coating removal not required
- Rapid, real-time inspection of large areas, translational speeds of up 100 mm/second

Recently, Robins Air Force Base of the US Air Force has supported QUEST Integrated to enhance the MOI systems. The new design maintains all the capability of the historical MOI models (308/7 and 308/3), but with many improvements and additional features. Improvements of the new MOI+ model over previous MOI products are:

- Compact packaging that is light-weight but supports a large image (FOV) as shown in Figure 9.2-20 and 9.2-21
- An LCD display on the front panel of the controller in addition to an external monitor (Figure 9.2-20)
- Larger field of view (FOV) that allows for more rivets to be inspected and displayed on the video screen
- Assisted flaw detection (AFD) system to assist the operators in identifying rivets with flaws by marking them in the video stream to reduce operator error (Figure 9.2-20 and 9.2-22)
- Higher quality images that reduce distortion and improve contrast, with the capability of eliminating the serpentine domain lines present in the traditional MOI models (Figure 9.2-22)
- An optional portable display mounted on the handheld imager or alternatively on the wrist (Figure 9.2-23).
- Ability to operate at higher ambient temperatures as a result of a number of design changes
- Display-integrated user interface with digital control panel and video out capability
- Image processing capability – digital camera-based that allows for AFD and other upfront processing
- Flexible excitation – allows excitation to be timed with image processing for image enhancement

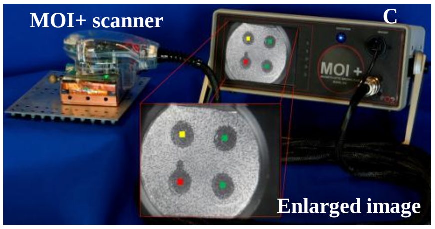


Figure 9.2-20. MOI+ System

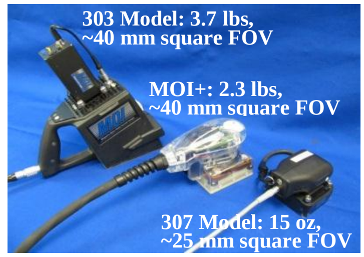


Figure 9.2-21. Comparison of Different Models

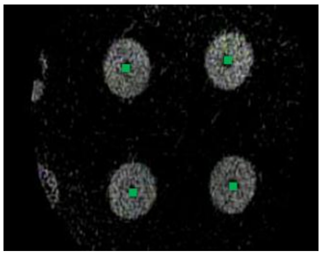


Figure 9.2-22. Enhanced Image Quality without Domain Lines

Figure 9.2-23. Portable Display for Instant View

9.3. STRUCTURAL HEALTH MONITORING

9.3.1. Demonstrating Capability Validation Protocol for In-Situ Damage Detection

Eric Lindgren and Charles Buynak, USAF Research Laboratory – Materials & Manufacturing Directorate; Enrique Medina, Science Applications International Corporation; and John Aldrin, Computational Tools, Inc.

At the 2009 ASIP Conference, the authors' presentation included the definition and outline of the protocol for validating the capability for an on-board damage detection system, frequently referred to as in-situ nondestructive evaluation (NDE) or structural health monitoring (SHM), to generate a Probability of Detection (POD) curve as part of the qualification process for such systems. A POD curve is required if these systems are to be used in the assessment of ASIP managed structures on United States Air Force (USAF) aircraft. The objective of the protocol is to define the process to quantitatively validate the capability and reliability to enable the use of such systems in the management of the structural integrity of USAF aircraft. The in-situ damage detection capability and reliability assessment includes quantifying false-positive potential and detection sensitivity variance caused by multiple factors including changes in operation environment, material, or geometry, as well as system reliability over their expected useful life.

This technical effort addresses the demonstration of the protocol to validate detection capability of an in-situ damage detection system on a representative aircraft structure. For this case study, the capability of a vibration-based damage detection method is investigated. The test fixture design provides the capability to vary critical parameters of the system with a focus on force loading boundary conditions, joint fastener torque conditions, and temperature (Figure 9.3-1). The review of this demonstration addresses the integration of external factors that affect the sensitivity of the detection capability, including variance in the structural configuration and other environmental factors, and how these factors can be accommodated in the validation process. The approach leverages previous efforts in the NDE community to incorporate validated models to minimize the amount of empirical data, time, and cost to determine the validated capability via a POD curve. The process and results for the POD evaluation of the in-situ damage detection system are presented in detail (Figures 9.3-2 and 9.3-3). In addition, the demonstration illustrates how this approach will minimize the degree of full-scale testing required for obtaining statistically meaningful damage detection assessment results. Another factor evaluated addresses variance in the POD curve with respect to the in-situ damage detection system degradation as a function of time. Thus, the feasibility of using this approach to determine the probability of detection and false-call rate is established and the necessary steps required to extend this practice beyond the demonstration stage will be discussed. This addresses one of the obstacles for implementing in-situ damage detection techniques to meet the required ASIP inspection metrics and facilitates its acceptance into USAF aircraft maintenance practices.

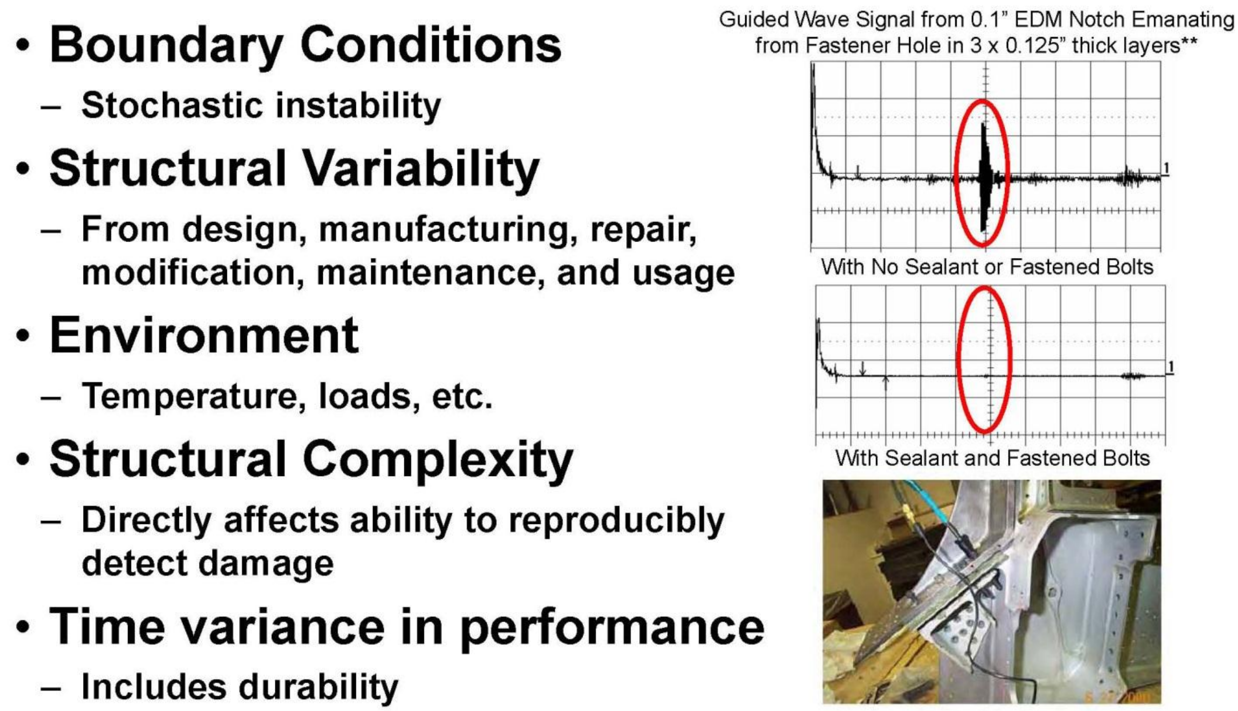


Figure 9.3-1. Factors That Need to Be Considered

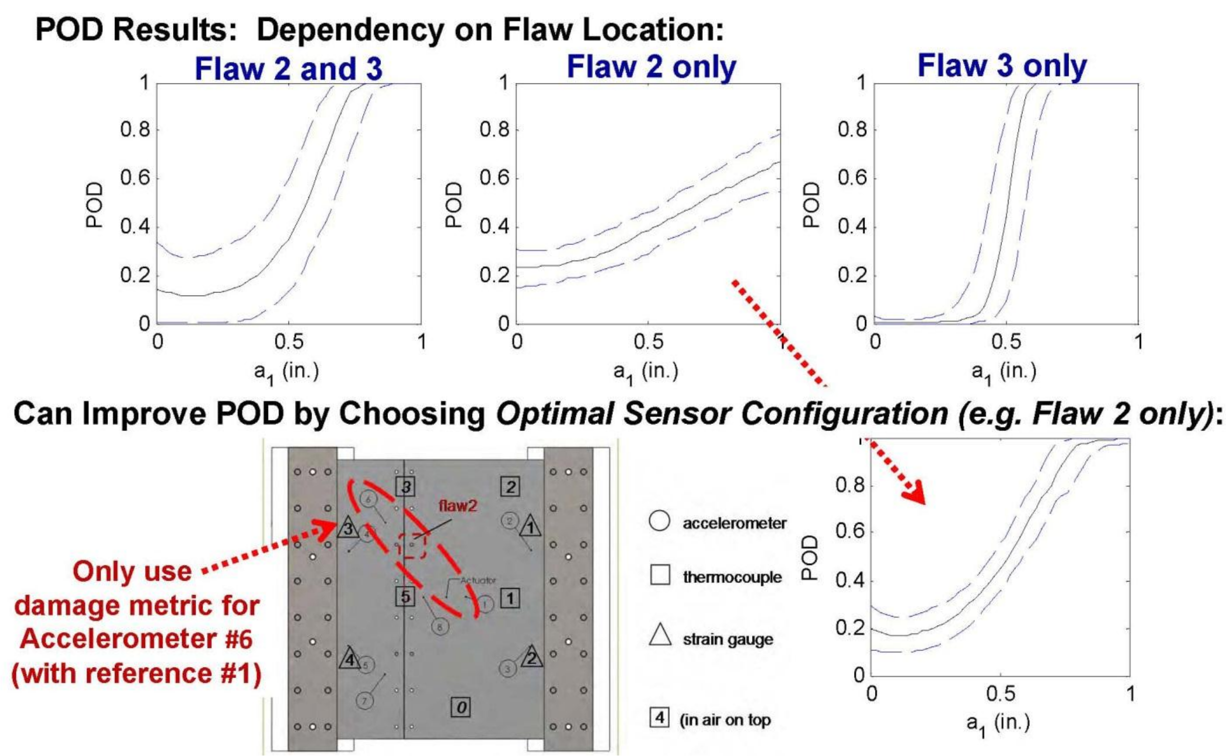


Figure 9.3-2. POD Results – Sensitivity to Flaw Location

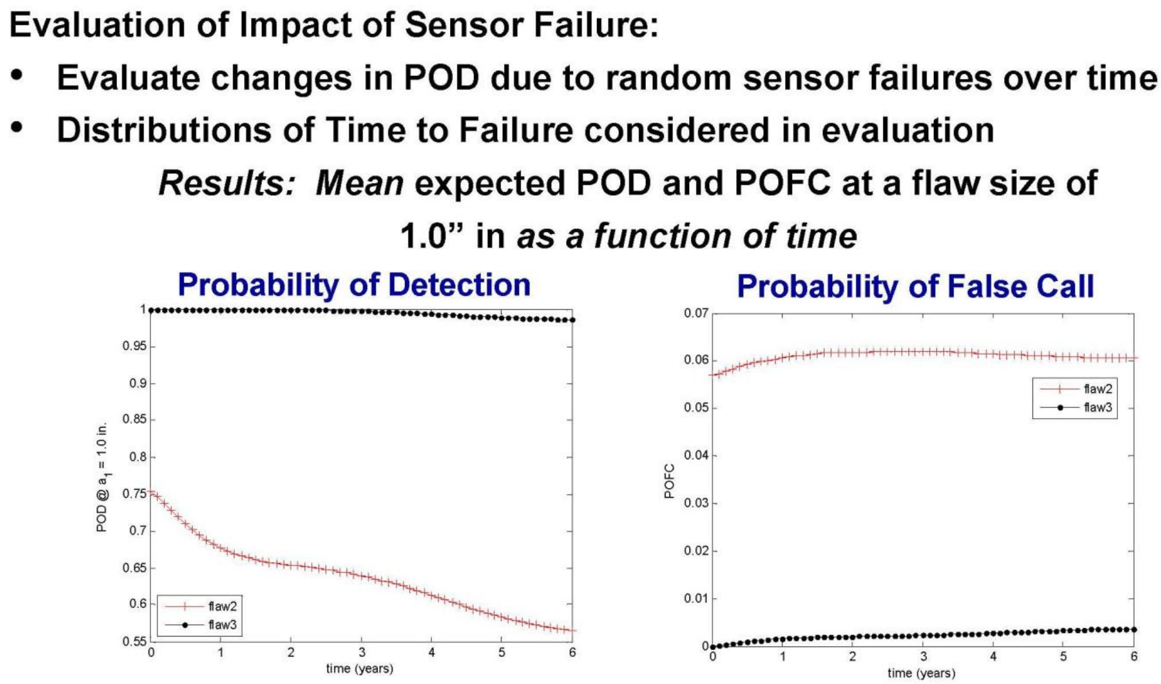


Figure 9.3-3. POD Results – Impact of Sensor Durability

9.3.2. Hot Spots Health Monitoring for F-22/F-15 Applications

Mark Derriso, USAF Research Laboratory – Aerospace Systems Directorate

Maintaining airworthiness in the presence of structural hot spots is a major focus of today’s Aircraft Structural Integrity Program. The mandate to develop Structural Health Management (SHM) strategies that make use of more first-hand information and thus allow maintenance actions to be performed based more on condition rather than schedule requires advances in technologies that allow maintainers to insure structural performance that is both timely and cost effective.

Aircraft structural components may have known “hot spots” where a particular type of damage is anticipated to occur or has consistently been observed in the field. The Air Force Research Laboratory and the Boeing Company have partnered to develop engineering tools and technologies that enable system level solutions to these problem areas. Automated inspection of these locations, or hot-spot monitoring, may offer significant time and cost savings for aircraft maintainers, particularly when the hot spots exist in areas that are difficult to access or where transitional non-destructive inspection methods will not work. The Hot Spot Structural Health Monitoring Program seeks to develop a systems-engineering approach to the design and implementation of SHM solutions for structural hot spots. The tools and technologies developed under this program are done so in the context of two specific exemplar hot spot locations on the F-22 and F-15 platforms (Figures 9.3-4 through 9.3-8).

Environmental Testing

- ➢*High & Low Temperature*
- ➢*Thermal Shock*
- ➢*Humidity*
- ➢*Overstrain*
- ➢*Connector Abuse*
- ➢*Vibration*
- ➢*Contamination by Fluids*
- ➢*Impact*

➢Other than damage resulting from direct mechanical impacts to the piezoceramic discs, no significant sensor degradation was observed

Figure 9.3-4. Mil-Standard 810 Testing

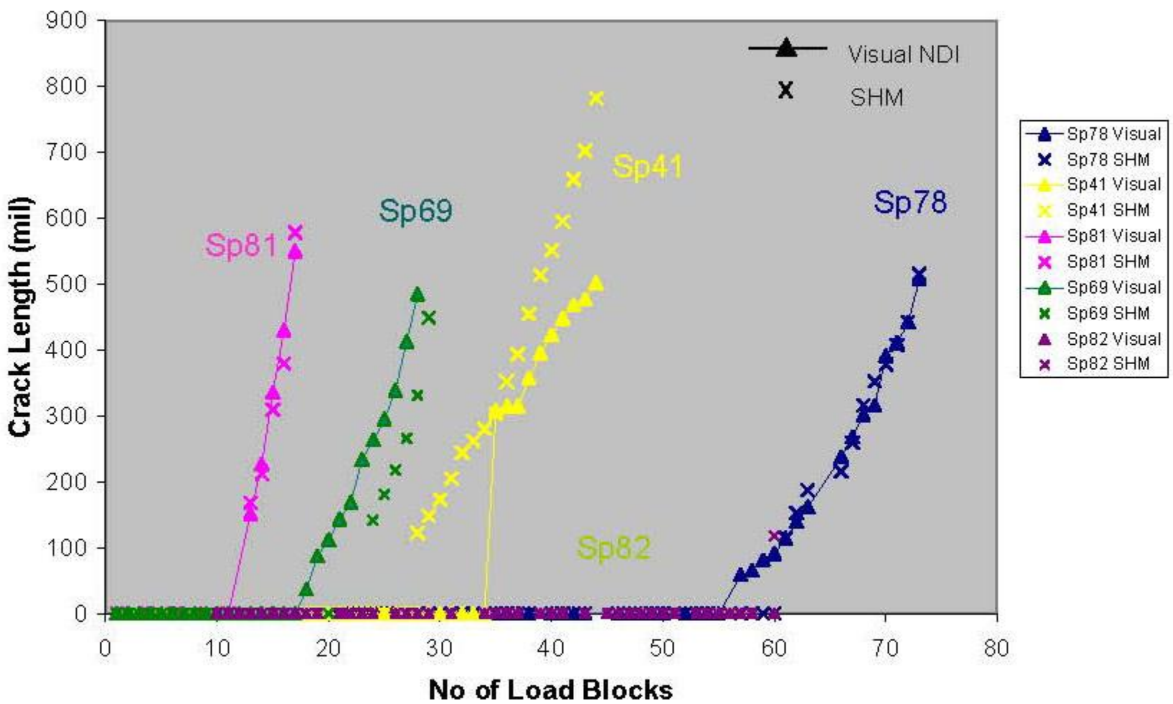
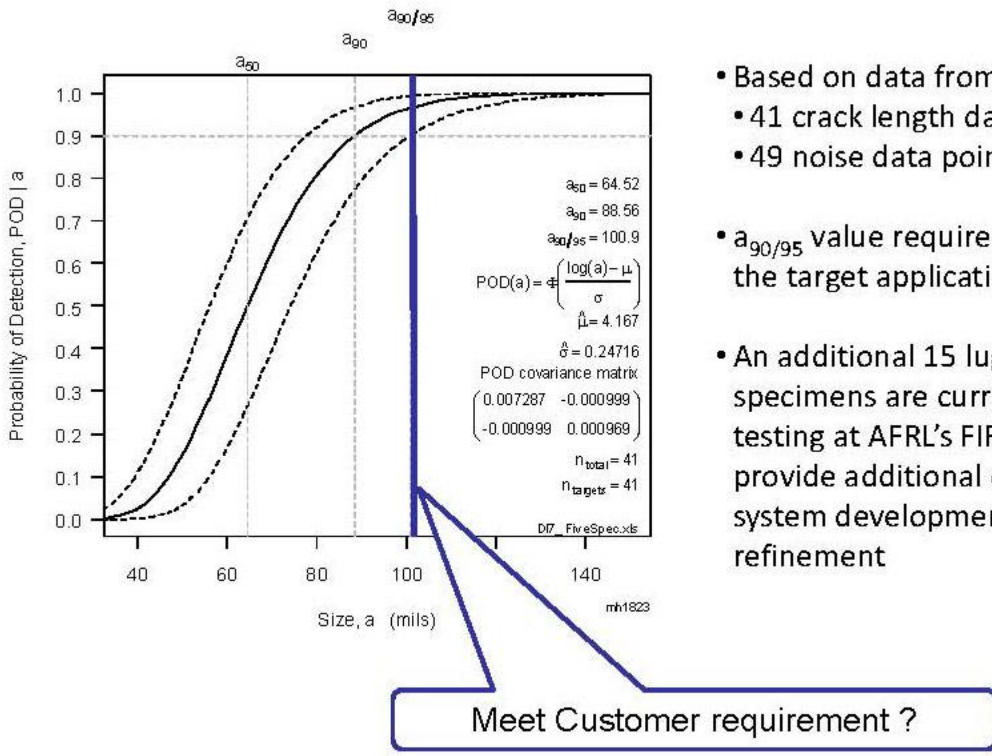


Figure 9.3-5. SHM vs. Visual NDI Results



- Based on data from 5 specimens
  - 41 crack length data points
  - 49 noise data points
- $a_{90/95}$ value required by the ASIP for the target application is 0.1"
- An additional 15 lug subcomponent specimens are currently being testing at AFRL's FIRST laboratory to provide additional data for SHM system development and refinement

Figure 9.3-6. POD Results

- 5 Subcomponent Lug Specimens tested under representative environment:
  - Variations in material property (etching, surface treatment, residual stress)
  - Flight spectrum loads with varying peak stresses
  - Variations in initial fatigue conditions at SHM installation time
  - Durability cycle up to 8 life-times

Figure 9.3-7. SHM Lug Specimen Validation

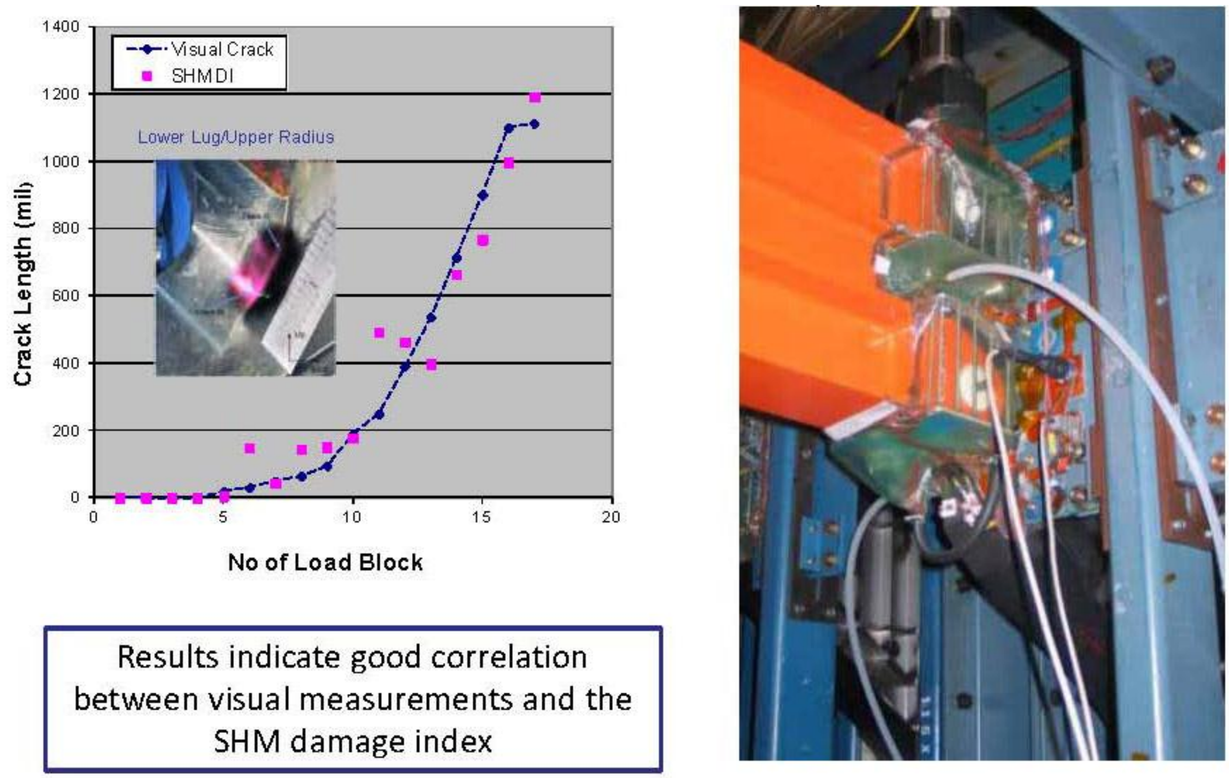


Figure 9.3-8. Full-Scale Test Results

9.3.3. Efficient Ground-Based Calibration of F/A-18E Fatigue Tracking Strain Sensors

Curtis Rands, Joshua Davis and Kevin Napolitano, ATA Engineering; and Timothy Fallon, USN-NAVAIR

Strain sensors are a critical source of data in assessing the structural health of fighter aircraft, such as the F/A-18. These sensors, permanently attached to the airframe, are monitored throughout each flight to quantify the loads experienced by the aircraft. Organizations responsible for tracking the structural life of individual aircraft use the measured data to evaluate and aggregate the damage resulting from all significant flight events. However, considerable margin must be applied to these models to account for variability in measurement sensitivity observed between aircraft unless the sensors are calibrated. Variation in sensor performance may result from differences in sensor orientation and positioning, inherent sensor “gage factors”, and aircraft build details. Today, calibration of the sensors to eliminate this uncertainty must be accomplished via a static test in an airframe test rig or through execution of prescribed flight maneuvers; the former involving significant cost and the latter achieving only limited accuracy.

This technical activity describes a portable system and efficient method developed to perform ground-based calibration of the strain sensors on the F/A-18E aircraft (Figure 9.3-9). Design, manufacture, and testing of a prototype system were funded by a Phase II SBIR program sponsored by NAVAIR. The approach was first demonstrated through analytic simulation and testing on scale aircraft models before being implemented on Navy fleet aircraft. Housed in a portable cart, the prototype calibration system includes data acquisition hardware and force application mechanisms that are automated through a software user interface. By applying and measuring a force to exercise the aircraft’s wings and empennage surfaces while monitoring structurally relevant strain sensors at these structures, the system mechanically determines the sensitivity factor for each sensor. The derived sensitivity factors can then be used to adjust flight-measured sensor data and improve the accuracy of the downstream structural life estimation models. Ultimately, the calibration process is intended to better inform maintenance decision makers, avoid unnecessary component repair/replacement, and reduce aircraft downtime while maintaining aircraft structural integrity.

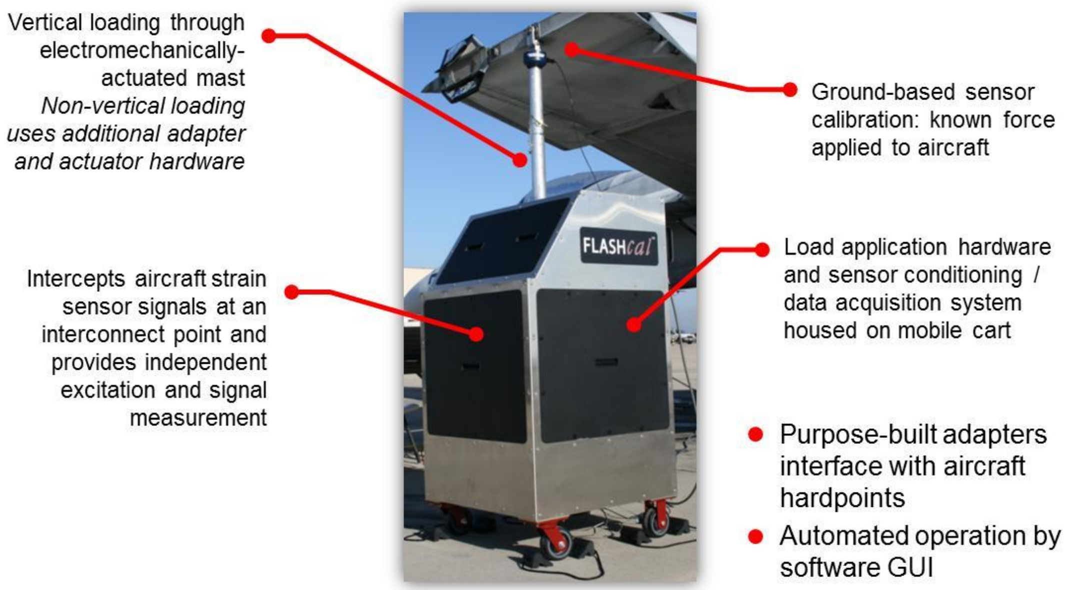


Figure 9.3-9. FlashCal™ System Provides Portable and Efficient Alternative Calibration Method

This technical activity discusses a demonstration test program utilizing the system to calibrate the sensors on a series of F/A-18E aircraft at the Navy Fleet Readiness Center Southwest. The efficient use of point loads at strategic locations while monitoring the sensors using both the system's native electromechanical actuator and loading by hydraulic aircraft maintenance jacks (to achieve a higher applied force) to obtain calibration values are discussed. Strain sensors are located at the wing root, wing fold, horizontal stabilator, and vertical stabilizer (Figure 9.3-10). Example calibration results are presented.

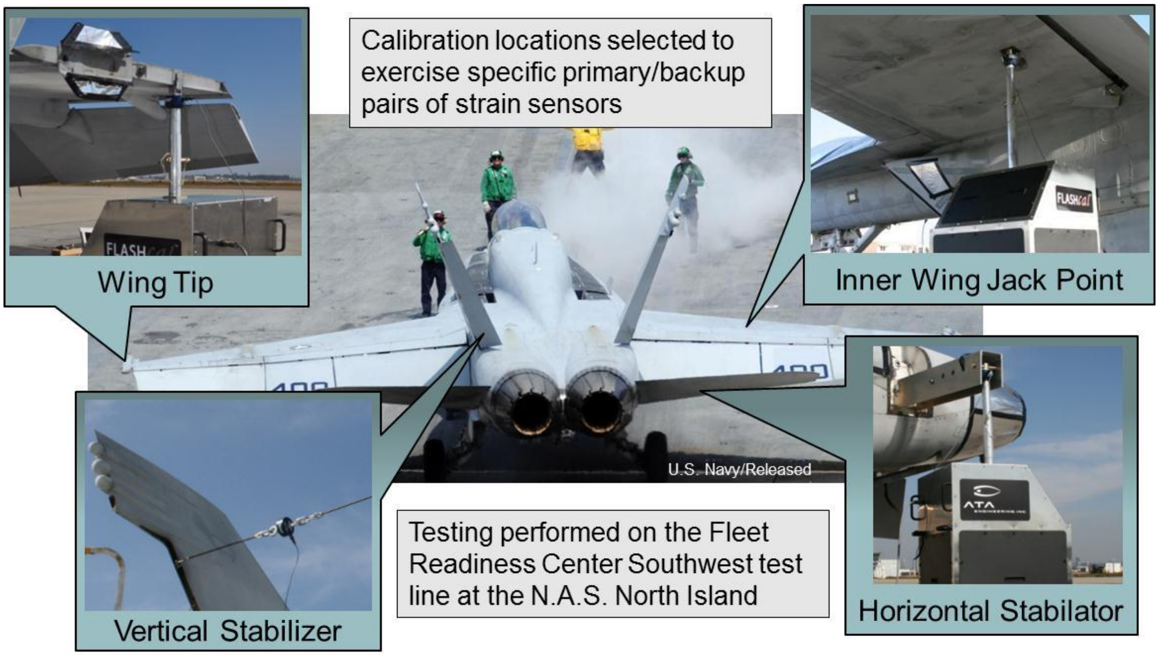


Figure 9.3-10. FlashCal™ First Implemented at Four Locations on an F/A-18E Super Hornet

9.4. STRUCTURAL TEARDOWN ASSESSMENTS

9.4.1. KC-135 Teardown Report on Aircraft 1 (AC1)

Gaddis Gann and Jeff Wilterdink, USAF-OC-ALC

The teardown program for the KC-135s has completed findings and failure analysis on one of the three aircraft planned for teardown. This technical effort highlights the following: 1) unique and innovative aspects of the program made necessary by the requirement to tear down more than one aircraft, 2) summary of a top-level view of all the findings (Figures 9.4-1 through 9.4-4), 3) sharing of details of a handful of the more meaningful findings, and 4) presentation of what steps are being planned to assess the meaning of the findings and determination of what changes to the overall maintenance plan are needed.

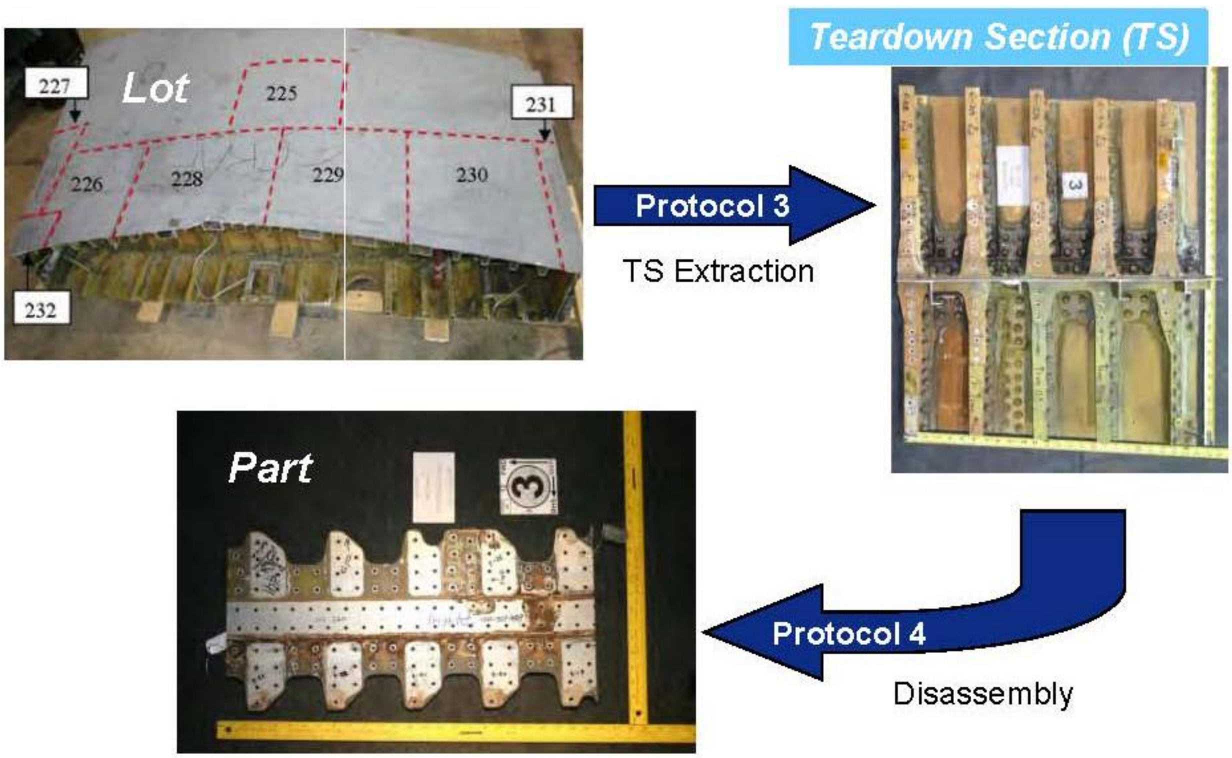


Figure 9.4-1. Lots, Sections and Parts

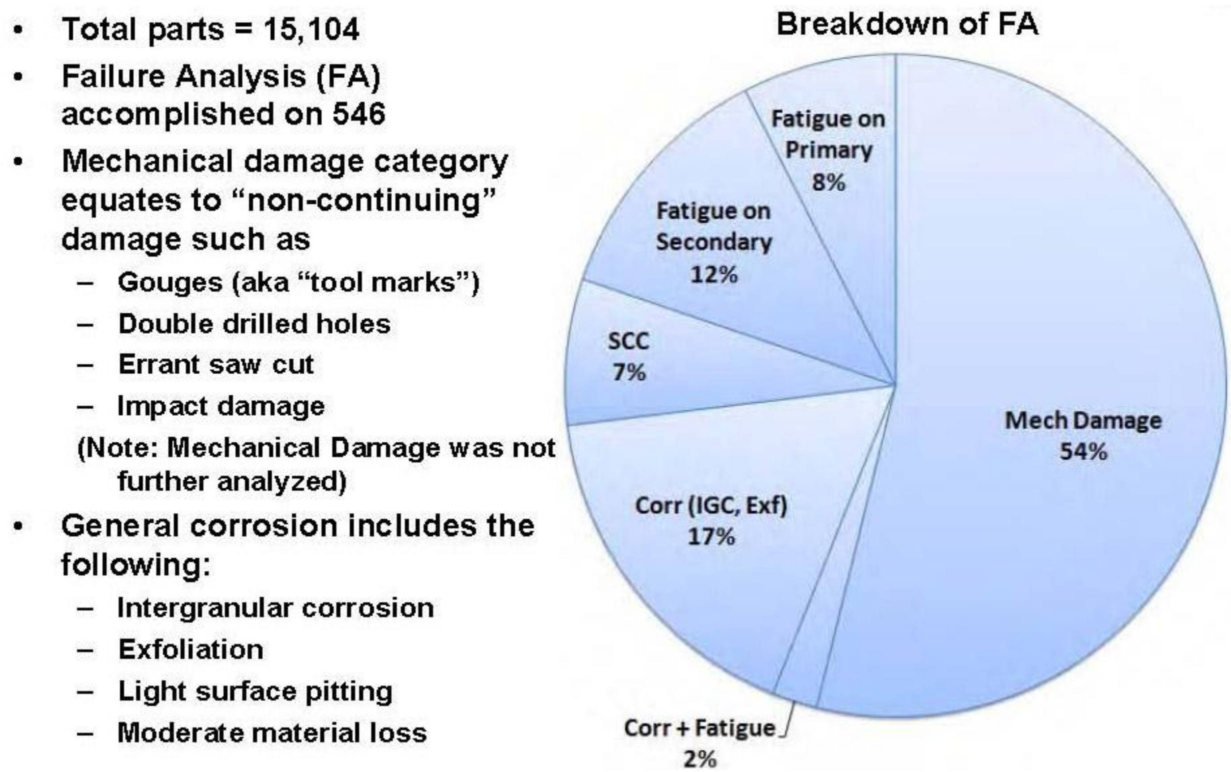


Figure 9.4-2. Distribution of Findings in Categories

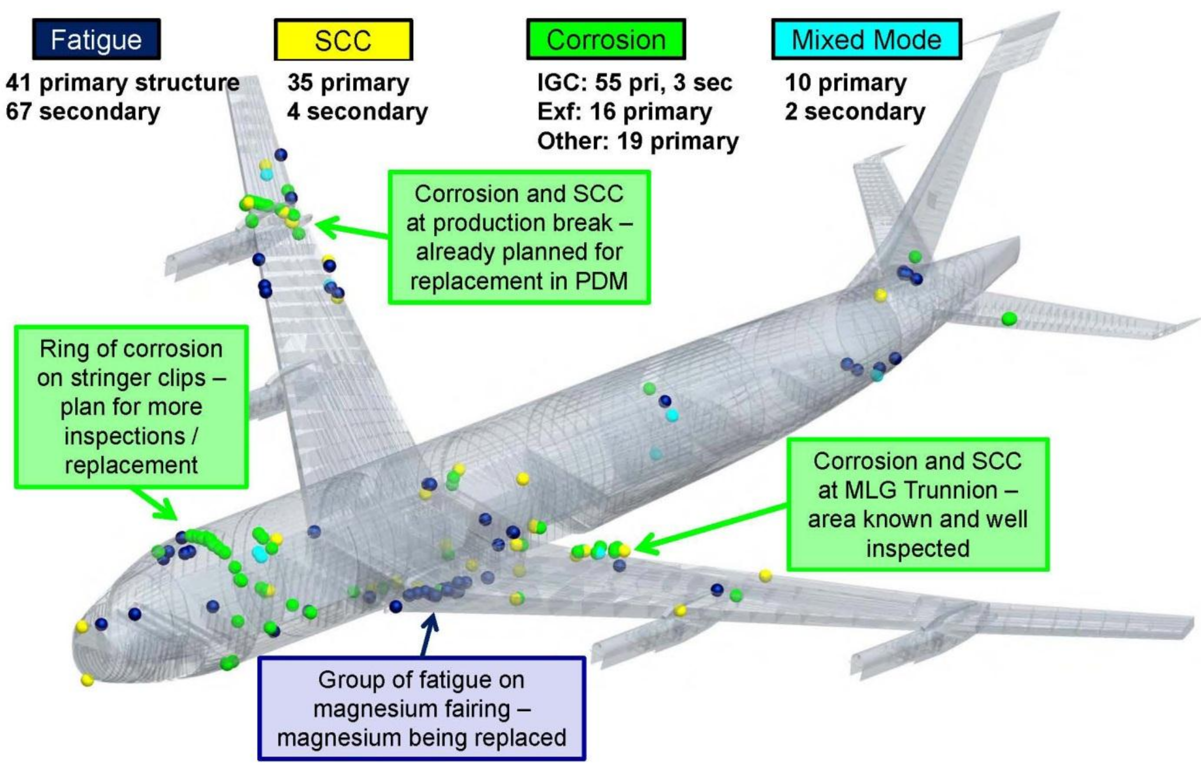
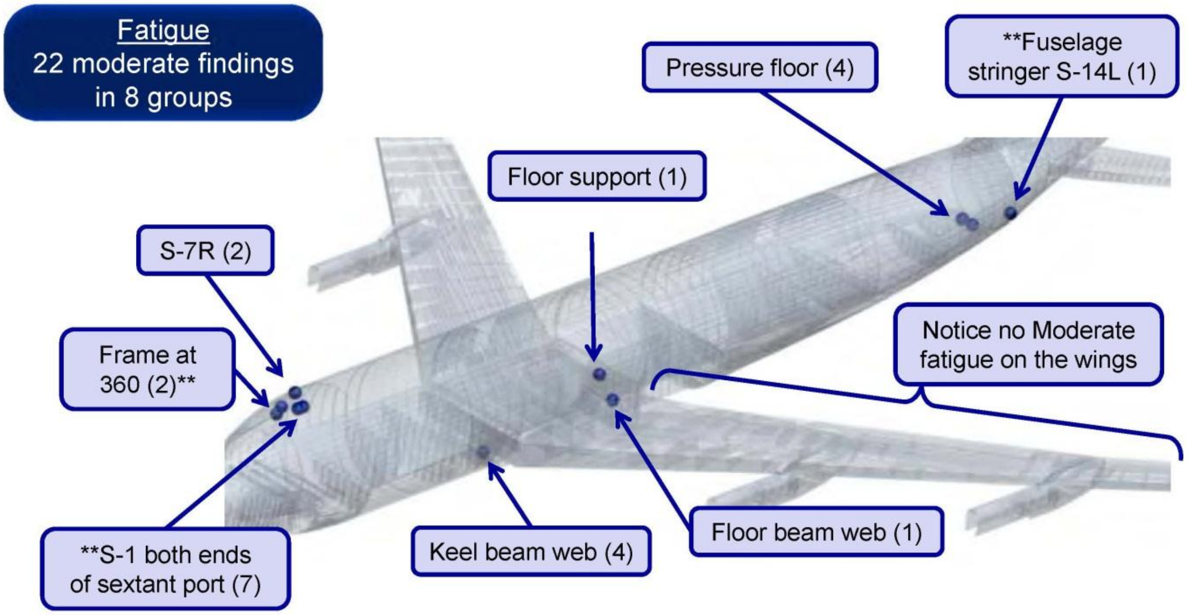


Figure 9.4-3. Locations of Failure Analyses



Type	Number Moderate	Tally of Recommendations (# out of 22 for each)					
		Special watch AC2, AC3	Improve PDM inspection	ACI	Check loads (FEM)	More Analysis (DaDTA)	Fatigue Testing
Fatigue	22	22	14	18	18	18	None

Figure 9.4-4. Moderate Fatigue Findings

9.4.2. T-38 Vertical Tail Teardown Analysis

Daniel Gardner, Northrop Grumman Corporation

The T-38 vertical tails have been in constant service with no replacement since 1961(Figure 9.4-5). In 2010, structural health evaluations were commenced to evaluate the assembly. This technical effort will describe the results of the study coupled with an overview of the analytical evaluations undertaken to determine the criticality of the results (Figures 9.4-6 through 9.4-8).

Figure 9.4-5. T-38 Talon

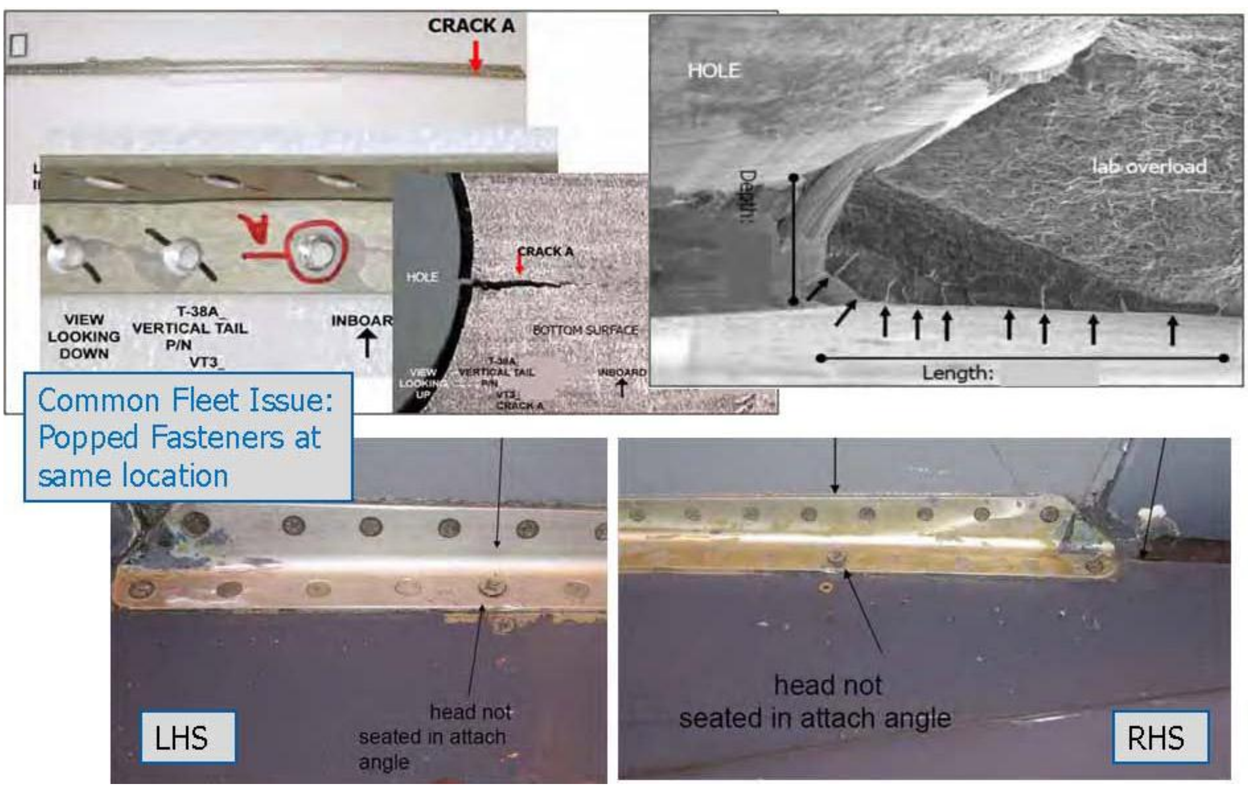


Figure 9.4-6. Attach Angle Teardown Findings

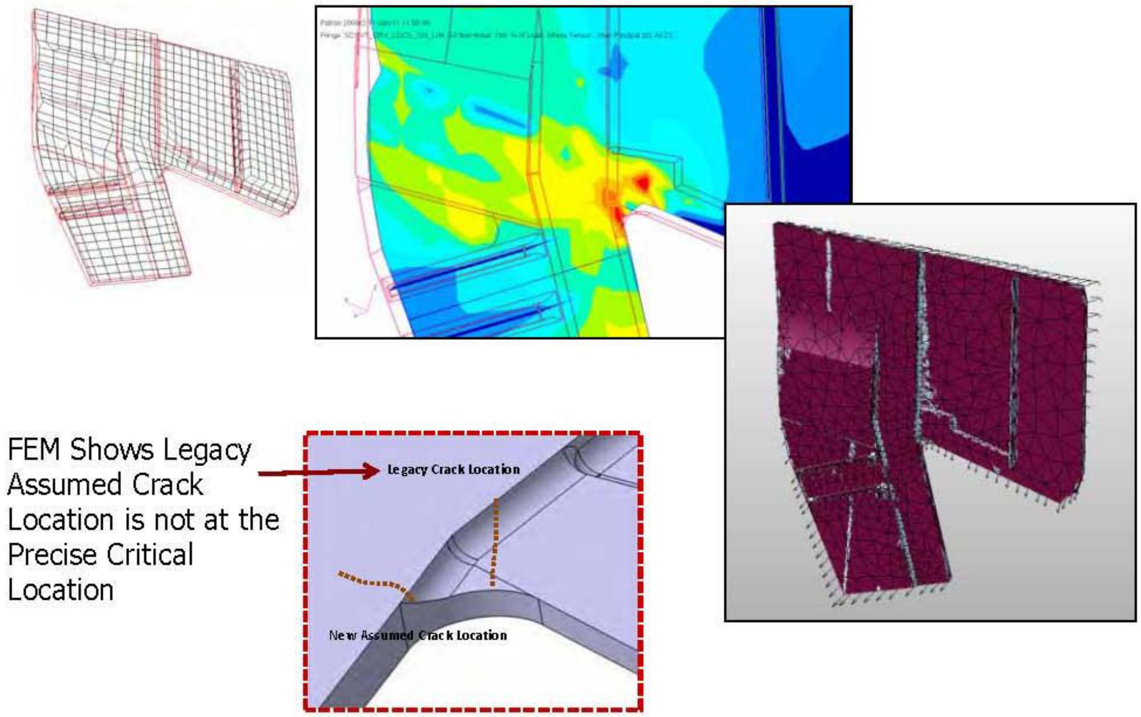


Figure 9.4-7. Vertical Tail Fatigue Critical Location

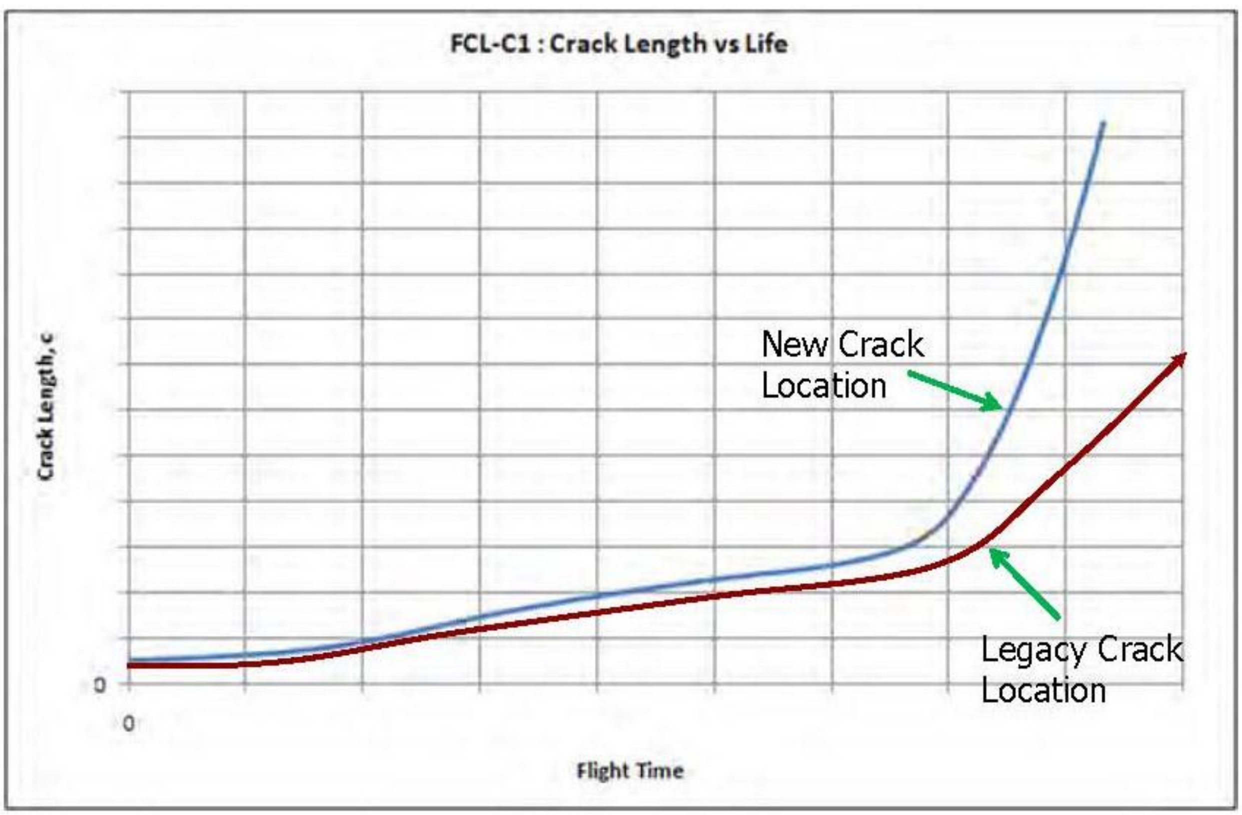


Figure 9.4-8. Crack Length vs. Flight Time

9.4.3. F-15 Structural Teardown Inspection Results

Lucas Garza, USAF Life Cycle Management Center

As principal custodian for the F-15 aircraft fleet, WR-ALC/GRM (the F-15 SPO), and primarily the ASIP manager, has the responsibility to ensure this essential fighter fleet can operate at acceptable mission capable rates and can fly the required sorties safely. As aircraft systems age and maintaining structural integrity becomes more challenging, periodic structural teardowns are often used to examine deeply embedded structural joints that are typically not observable during routine field and depot level inspections. Aircraft teardowns are also used to enable the application of high-resolution inspection techniques to critical components which are not possible with an intact aircraft. Periodic teardown inspections are an essential element of a sound fleet management strategy. Furthermore, it is vital that life extension programs contain teardown inspections that can identify new areas of concern.

The F-15 C/D airframe's design life specification was originally 4,000 flight hours and was later adjusted to 8,000 after airframe structural testing and a change in philosophy from Safe Life to Damage Tolerance Analysis (DTA). With average accumulated flight hours for the C/D model fleet of 7,100 hours and Congressional direction to extend the service life of the fleet to 2025, the SPO is now in the process of validating the sustainability of the C/D model airframe for an additional 15 years. To achieve this, the SPO directed that full structural teardowns be performed starting in 2008 as well as a Full-Scale-Fatigue-Test now underway.

S&K Technologies, LLC (SKT) has recently completed the series of aircraft teardowns which included an F-15D fuselage, an F-15C fuselage and six F-15C/D wings. This work was followed by performing microscopy on a large number of specimens containing NDI indications that were excised from parts targeted by the F-15 SPO. The approach to the F-15 teardown was similar to the methodology used during previous successful aircraft teardown projects performed by SKT, including those developed and validated during the C-5, C-130 and KC-135 teardown projects. To ensure the validity of the data collected, SKT followed Air Force and industry standard teardown protocols, assembled and managed by the Air Force Research Laboratory. The F-15 Teardown Protocols were derived from the KC-135 protocol with approval from the KC-135 SPO.

SKT's approach to the F-15 Structural Disassembly and Analysis Support Project included:

- Development of teardown data packages to identify the procedures to extract the target components from the aircraft as well as the inspection procedures to be used (Figure 9.4-9).
- Adaptation of the Teardown Data Management System to support F-15 teardown data management which provided SPO, OEM and foreign operator country engineers access to teardown findings, metallurgical reports and other pertinent records.
- Extraction of targeted structural joints and components in accordance with the approved protocols by certified SKT Technicians.
- Stripping of sealant and paint from extracted parts followed by Nondestructive Inspection (NDI) utilizing various techniques in accordance with SPO requirements.
- Metallurgical or optical analysis of all crack-like NDI indications was performed by the Israeli Air Force or SKT as directed by the SPO. With the structural teardowns complete, the inspection results are providing valuable data points for determining the effects of current usage on the F-15 airframe. The high-resolution of the inspection allows the documentation of very small defects, most of which will not affect original design life structural integrity, but become invaluable when extending the service life by potentially identifying new inspection points and may provide the basis for adjusting inspection intervals for known critical structures.

Figure 9.4-9. Extraction/Disassembly

This analysis is being used to augment the Force Structural Maintenance Plan (FSMP) and support continued structural assessment, sustainment, and mission readiness of the fleet.

This technical activity will provide a quick overview of the methodologies and protocols used during F-15 Teardown but will primarily focus on the results of the inspections and the microscopy work (Figures 9.4-10 and 9.4-11). This teardown is unique due to the number of Fluorescent Penetrant Inspection (FPI) and Bolt Hole Eddy Current (BHEC) (Figure 9.4-12) NDI indications that were ultimately excised, bisected and closely examined to identify the cause of the indication. During the series of F-15 teardowns ~540 parts were extracted and stripped, ~90,000 holes were BHEC inspected, and ~1,500 NDI indications were bisected and microscopically examined. The statistical analysis of the data recorded will provide valuable insight for the ASIP community, not only to document the structural health of the F-15, but will provide a correlation of a large number of NDI results to the actual damage state.

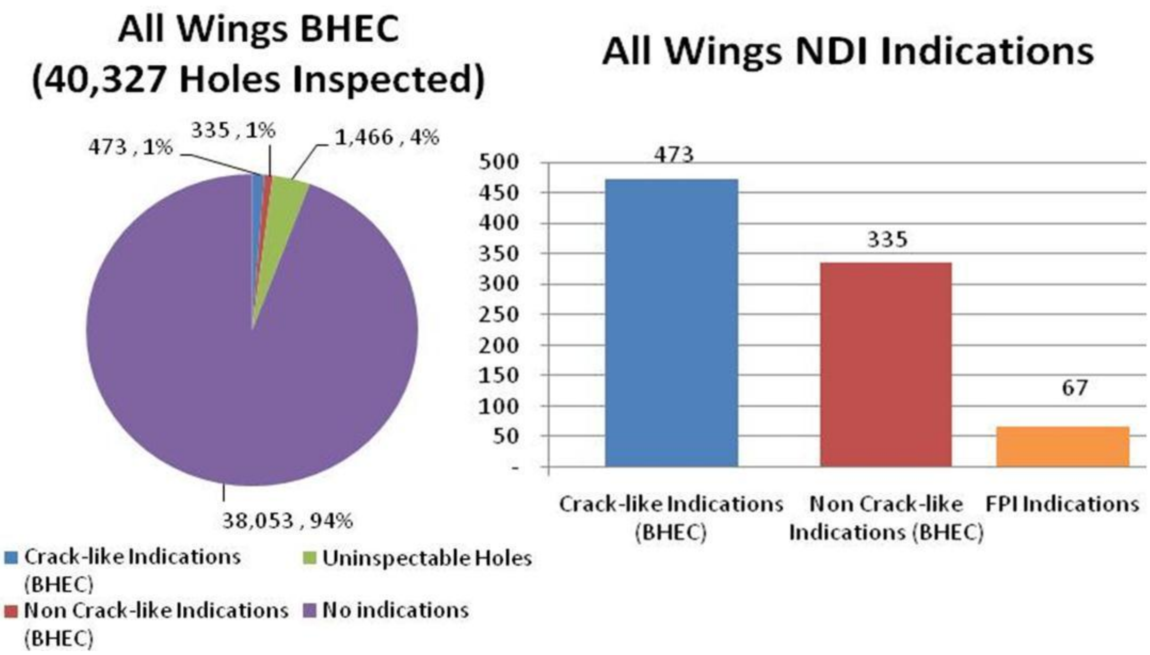


Figure 9.4-10. NDI Results for F-15 Wings

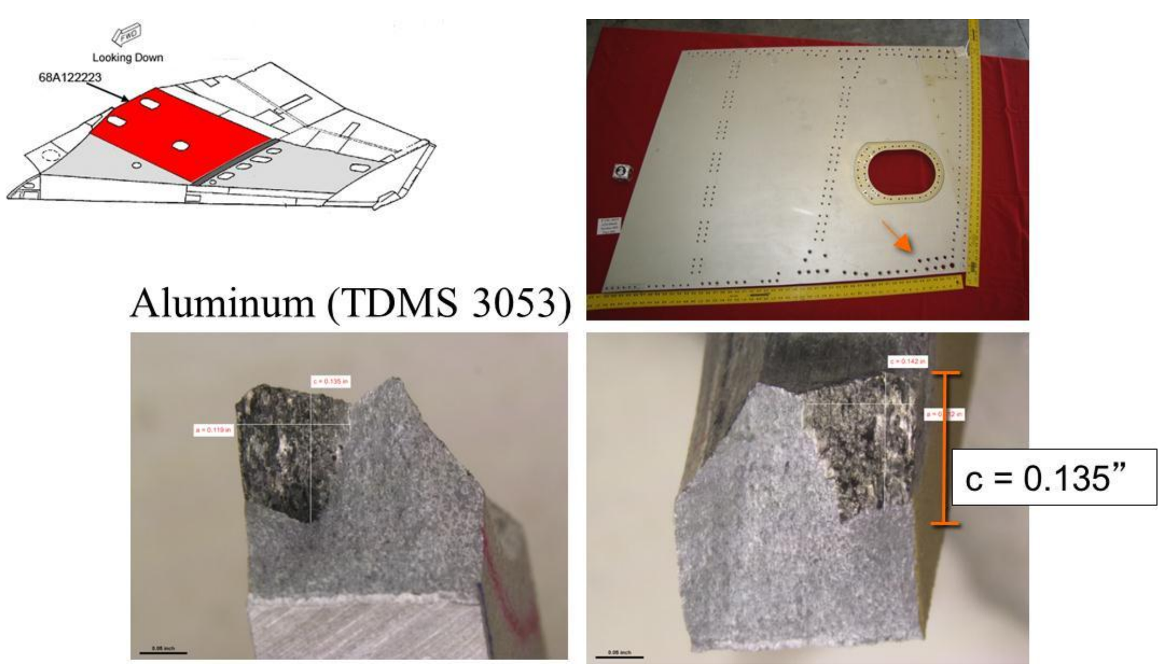
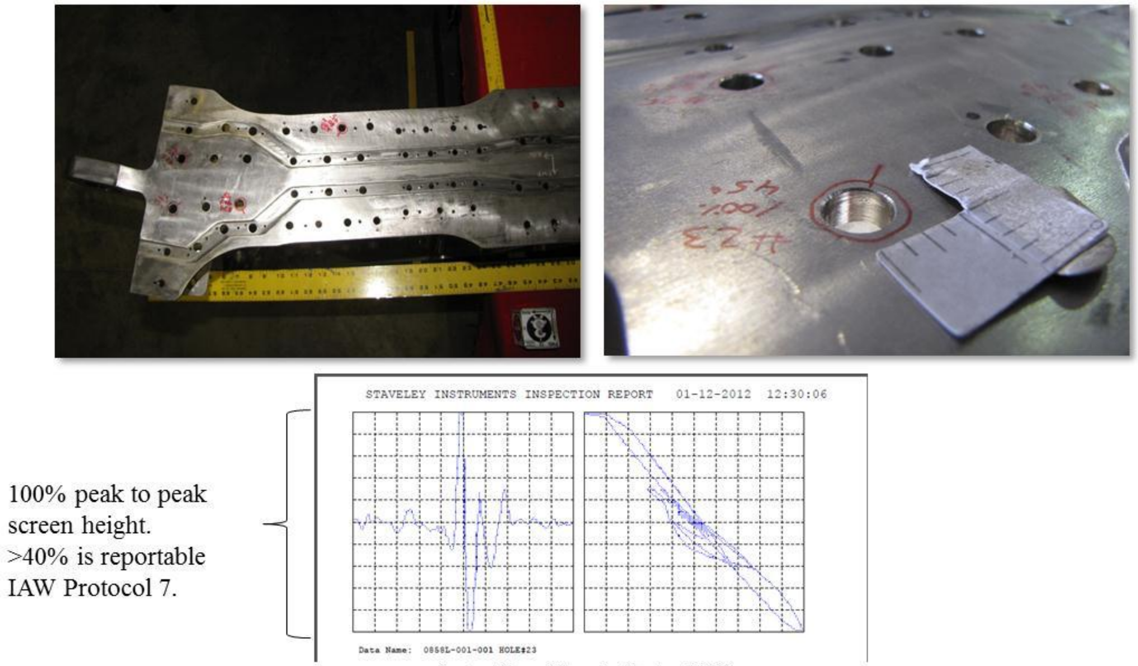


Figure 9.4-11. Results for Inboard Torque Box Upper Aft Skin



Output from Stavely Nortec 2000

Figure 9.4-12. Bolt Hole Eddy Current (BHEC) Method

9.4.4. Structural Teardown Analysis

Gregory Shoales, USAF Academy – CAStLE

CAStLE continues to assist fleet management decisions and the US Air Force Structural Integrity Program (ASIP) thorough multiple programs whose focus is assessing aging structures. Teardown analysis programs are required by MIL-STD-1530C at various points in the life cycle of all USAF aircraft. CAStLE has been part of teardown programs since 2002 and wrote the USAF best practices guide for teardown in 2008. This publication was followed by the CAStLE protocols for detailed teardown processes which captured best practices and lessons learned garnered from more than a decade of United States (US) Department of Defense (DoD) teardown programs. Since the last report in 2011, CAStLE has participated at various levels in the structural teardown analysis of six US DoD aircraft.

Beginning with planning in 2007, CAStLE has continued to execute a teardown of primary structure on three KC-135 aircraft. Teardown program execution began on the first aircraft in 2008 and completed early 2011. As of ICAF 2013 all the analysis is complete on the first two aircraft with the third aircraft's analysis in its final year. Measured by any metric the KC-135 teardown analysis program is by far the most extensive of any teardown to date. Figure 9.4-13 depicts the extent by which structural sections were removed and analyzed.

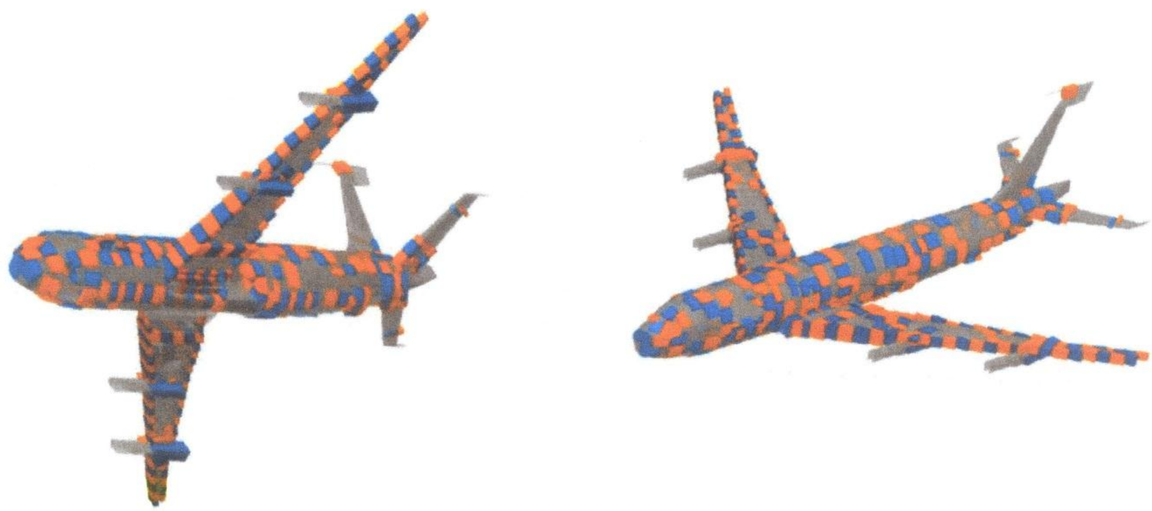


Figure 9.4-13. Graphic Indicating (by Orange and Blue Boxes) the Structural Section Removed from Each KC-135 for Teardown Analysis

Analysis of the first two aircraft has generated more than 50,000 nondestructive inspection (NDI) indications and nearly 900 detailed failure analysis (sometimes known as root-cause investigation) findings. Each and every indication and finding is carefully assessed by CAStLE, Boeing and KC-135 Program Office engineers to determine implications upon the fleet's continued service. Of particular interest to the program office are the structural health of fuselage lap joints and the performance of applied corrosion preventive compounds (CPCs). To satisfy this somewhat unique requirement, CAStLE developed specialized test equipment and protocols to perform focused lap joint evaluations. This lap joint evaluation is beyond the NDI and subsequent root-cause investigations normally associated with structural teardown analysis which is performed on the aircraft structure indicated in Figure 9.4-13. The CAStLE lap joint protocol includes residual strength and residual life testing of removed KC-135 lap joints in order to compare with the original design parameters. To evaluate the CPC performance, CAStLE adapted the electro impedance spectroscopy (EIS) test methodology to a raster scanning device. This device permits precise coating system integrity evaluations at repeatable locations on selected lap joint sections. The repeatability permits evaluation of the coating system in and around the lap joint in the as-removed condition as well as after repeated accelerated environmental exposures. Comparison between EIS data before and after accelerated exposure permits a qualitative evaluation of the CPC's potential continued ability to protect the lap joints from corrosion. Other unique aspects of this program have included the presence of a US multi-agency Oversight Committee to review process and plans and a process to qualify each program participant. Qualification is granted by CAStLE upon review of an independent site-visit-based report performed by AFRL/RX. The AFRL team's site visit is an independent assessment of the participant's ability to perform the task associated with the applicable CAStLE teardown protocols. The ongoing progress and results of this work have been presented on numerous occasions at the *Aircraft Airworthiness and Sustainment Conference.*

Beginning in 2011, CAStLE planned and completed execution of a teardown analysis on the USAF C-130 empennage. This teardown program has focused on structure in the horizontal and vertical stabilizers. In order to illuminate potential hidden damage, this program analyzed the empennage structure in its entirety. Like the KC-135 program, the C-130 empennage program inspected all components by at least two different NDI techniques—including more than 42,000 bolt hole eddy current inspections. CAStLE had previously accomplished teardowns of the critical center wing structure for the

USAF Aging Aircraft Program Office. The empennage represents one of the next most critical structural regions of the C-130. Teardown data from the current program will serve to supplement previous teardown program data as the C-130 fleet managers strive to meet USAF life goals.

CAStLE planned and is currently executing multiple teardown analysis programs for the USAF T-38 program office. One teardown is focused on assessing the critical structural elements of an entire airframe, particularly those associated with the T-38 engine upgrade program. Additional teardown analysis programs have been planned to fully inspect and analyze wing and fuselage structure tested during CAStLE's T-38 full scale fatigue test (FSFT) programs.

In late 2011, CAStLE extracted wing, empennage and aft fuselage structure from a retired Boeing 707 located at Melbourne, FL. CAStLE subsequently planned and began execution of a teardown analysis of the wing portion of that structure. This program has focused on primary wing structure and seeks to identify corrosion and fatigue damage in support of the USAF Joint Stars (E-8C) program office's fleet management decisions. Analysis for this portion of the program will be complete in mid-2013. At the request of the program office, CAStLE also completed planning for a follow-on program which focuses on analysis of the empennage and aft fuselage structure. In the wake of the 2013 USAF budgetary pressures, execution of this program is currently on hold.

Since early 2011, CAStLE has been supporting the B-1B full-scale fatigue tests. This support has included spectrum validation testing, component level test failure analysis, and planning the post FSFT teardown of test B-1B airframe. Based on an analysis of all available fleet records, CAStLE identified a prioritized list of teardown subject structure along with NDI inspection recommendations. CAStLE further prepared a selection rubric for the program office to assist their matching fiscal constraints to program technical requirements. In addition, the CAStLE teardown protocols were evaluated for applicability to the B-1B teardown plans. Unique materials in the B-1 system required the development of additional NDI inspection techniques and coating removal processes into these protocols. CAStLE plans to incorporate these new techniques/processes into the next revision of the CAStLE teardown protocols.

Lastly, CAStLE continues to share its experience and lessons learned by serving as a consultant to other DoD teardown programs. In 2012 and 2013, CAStLE provided all protocols and teardown database access to the US Navy V-22 program in support of their post FSFT airframe teardown. Selected protocols for extraction, disassembly, and coating removal were requested by and provided to the F-15 program office.

9.5. LOADS & ENVIRONMENT CHARACTERIZATION

9.5.1. F-16 ASIP Data Collection Improvements and Service Life Impacts

Bryce Harris and Kimberli Jones, USAF-OO-ALC; William Legge, Science Applications International Corporation; and Jim O'Connor and Matthew Edghill, Lockheed Martin Corporation

The F-16 weapon system (Figure 9.5-1) continues to search for opportunities for improvement in capturing valid Aircraft Structural Integrity Program (ASIP) data from the Crash Survivable Flight Data Recorders (CSFDRs). While the United States Air Force (USAF) F-16 fleet has struggled to meet the required data capture rates, recent process breakthroughs and unified support efforts have made reaching the 90% valid Individual Aircraft Tracking (IAT) requirement possible. For example, in 2011, senior USAF leadership committed to improving the cultural view of ASIP data collection through refocused enforcement of program compliance, accountability, and standardization at the wing level. These human factor directives quickly resulted in a noticeable increase across the USAF F-16 fleet in capturing valid data. A collaborative endeavor between the MAJCOM, Aircraft Structural Integrity Management Information System (ASIMIS), Lockheed Martin Aero, and the F-16 ASIP Program Office has also proven quite effective. This united front enabled ASIP engineering activities to achieve significant success in improving data validity, and also to some degree data capture. Major contributors to the validity and data capture rate improvements included: decreasing download interval times from 150 to 75 hours to shorten capture cycle, implementing a new user-friendly data processing management platform to accurately identify valid/invalid information, streamlining the ASIMIS website to simplify field-user interaction, championing changes in Air Force Instructions to enforce standardized compliance and accountability, and modifying software within aircraft download support equipment to eliminate erroneous data deletion (in work). The data obtained from the CSFDRs are critical to daily ASIP activities, especially for calculation of equivalent flight hours (EFH); these hours reflect flight severity on an individual aircraft basis and are compared to the certified service life by fleet planners. When data capture and validity rates are low, downloads are backfilled with baseline average severity usage per standard procedures. These backfilled data could be of a higher severity than actual aircraft usage, which in turn lessens the predicted life of the aircraft by assigning higher EFH for a given number of actual flight hours. Similarly, structural inspection schedules based on assumed usage severity are at risk of missing cracks, which can have safety, readiness, and cost impacts to the fleet. This technical effort will detail the mentioned efforts and illustrate the recent level of capture rate increases(Figure 9.5-2), as well as citing an example of how these improvements can impact aircraft reaching certified service life (Figure 9.5-3).

Figure 9.5-1. F-16 Weapon System

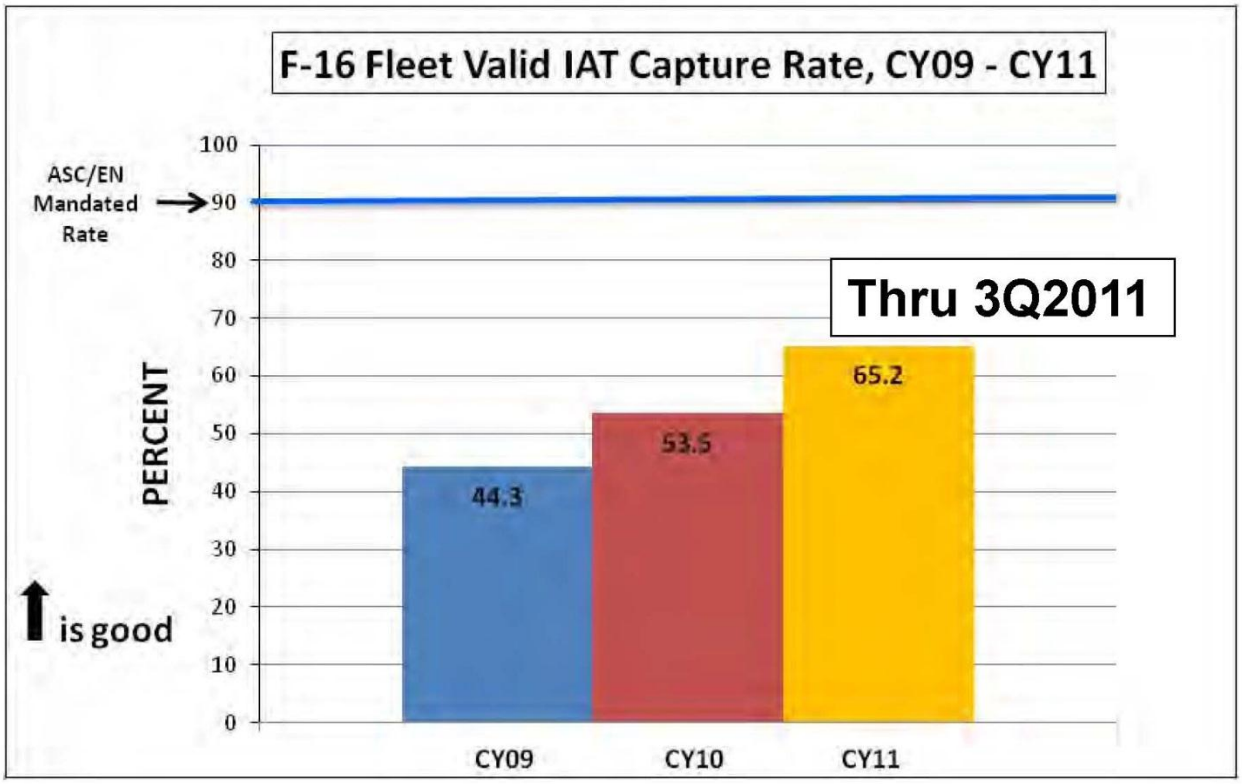


Figure 9.5-2. Yearly Capture Rate Improvement

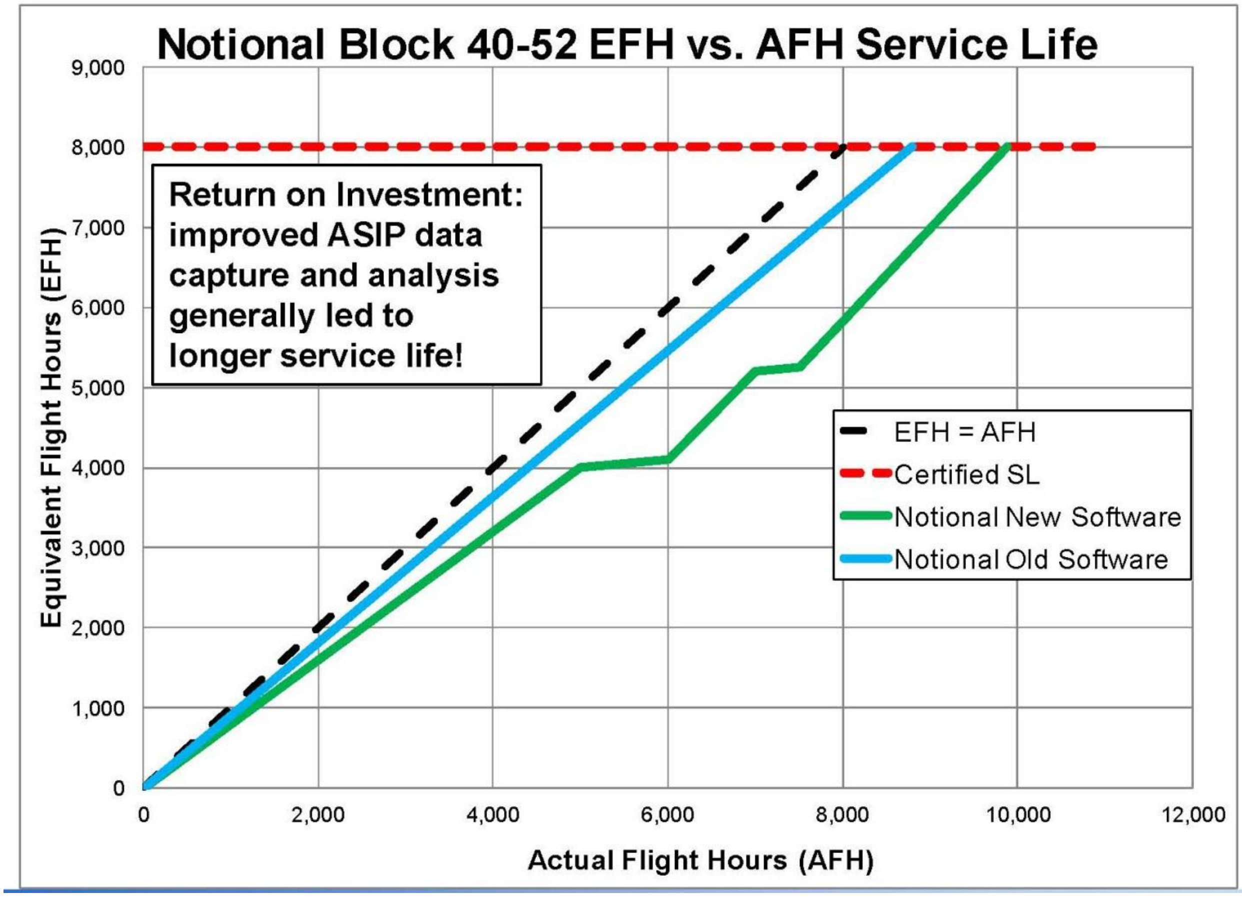


Figure 9.5-3. Service Life Effects – Improvements and Return-on-Investment

9.5.2. F-16C Block 50 Full-Scale-Durability-Test Loads Spectra Development

Bill Signorelli, Lockheed Martin Corporation

The development of the test loads spectra was a critical milestone for the F-16C Block 50 Full-Scale-Durability Test program. Starting with the analytical test spectrum, various algorithms were developed to create the different spectra needed to support the test (Figure 9.5-4). Initially, a truncation algorithm was needed to remove load cycles from the analytical test spectrum that contribute little or no damage to the aircraft structure and to help reduce actual test time (Figure 9.5-5 and Table 9.5-1). Next, an algorithm was needed to translate the truncated loads to the test fixture ram loads. The loads applied to the test article by each hydraulic ram must model the total aircraft loads of the truncated spectrum as closely as possible while maintaining the aircraft in complete balance within the test fixture. Simple comparison of component load plots and resulting internal loads from the airplane coarse finite element model were used to evaluate the ram load spectrum. However, since it is not possible to model exactly the total aircraft loads of the truncated spectrum due to limitations in the number of rams and ram capacity, another algorithm was needed to re-translate the ram loads back to the aircraft as the applied load spectrum to account for the service life differences between the truncated and the applied spectra. Finally, unique to this test as compared to previous F-16 tests, the test aircraft was not a new aircraft directly from the factory; therefore, an additional algorithm was needed to develop a past usage load spectrum. This technical activity presents an overview of the methodology employed and the challenges encountered in the development of the various load spectra that are an integral part of the F-16 Block 50 Full-Scale-Durability Test program.

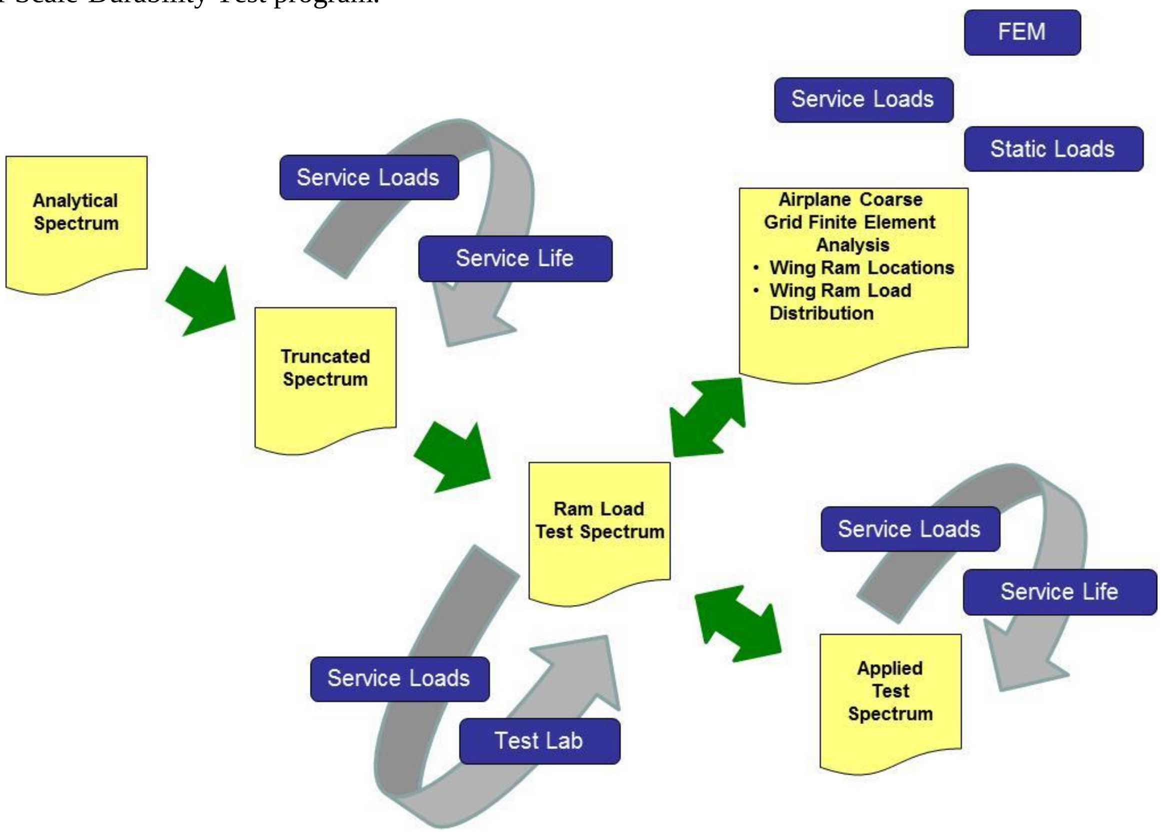


Figure 9.5-4. Durability Test Spectra Development

- **A peak-valley counting method based upon identification of local maximum and minimum load values was used to filter out insignificant load cycles**

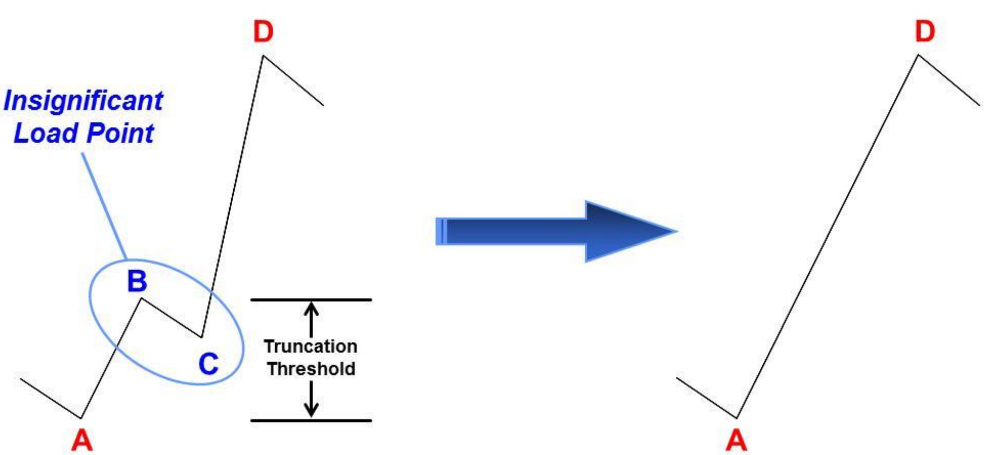


Figure 9.5-5. Truncation Algorithm Methodology

Table 9.5-1. Truncation Spectrum Development Process

- **Load range increased from 4.0 ksi value and crack growth assessments evaluated for 16 control points across airframe**
  - **A 3% difference in predicted life was deemed acceptable**

Control Point Location	Ratio	Control Point Location	Ratio
Forward Fuselage - Longeron Bolt Hole	2%	Center Fuselage - Longeron Bolt Hole	-1%
Center Fuselage - Longeron Fastner Hole	-1%	Aft Fuselage - Upper Bulkhead Flange Bolt Hole	17%
Vertical Tail Rear Spar, Shear Web Hole	3%	Aft Fuselage Panel Attach Fitting	4%
Horizontal Tail Outboard Support Beam	2%	Lower Wing Skin - Fastener Hole at LEF Actuator #3	4%
Lower Wing Skin, Bolt Hole Common to WAF	2%	Lower Wing Skin, Fastener Hole near Pylon Cutout	3%

*Ratio = Analytical Spectrum / Truncated Spectrum

- **Final truncation spectrum contains 118,976 load points**

9.5.3. Modernization of A-10 L/ESS and IATP Force Management

James Kokoris and Ronald Smith, Northrop Grumman Corporation

The A-10 aircraft (Figure 9.5-6) is a mature United States Air Force (USAF) weapons system that was originally procured during the period of 1975-84 (Figure 9.5-7). Originally designed for a lifetime of 6,000 EFH, (eventually increased to 8,000 EFH), with recorded rates of approximately 80 flight hours per calendar quarter, retirement would have begun in 2000. However, due to the accurate usage tracking provided by its continuously-monitored Individual Aircraft Tracking Program (IATP) system, some structural enhancements, and global circumstances that have kept it in demand, the A-10 has had its service life extended to the year 2040+. This significant increase has required the adoption of several new electronic systems to accommodate modernization trends in the USAF. These include the addition of a digital data bus, a GPS navigation system, satellite communications, and presently, an Aircraft Data Recorder (ADR) function to the Turbine Engine Monitoring System (TEMS). This technical activity addresses the ASIP enhancements possible through the use of the TEMS-ADR.

The original A-10 aircraft specification included a cartridge tape drive MXU-553/A recorder for Loads/Environment Spectra Survey (L/ESS), and a mechanical counting accelerometer, the ABU-15/A, for Individual Aircraft Tracking (IAT). Although the L/ESS recorder acquired a significant number of flight parameters at a relatively high frequency rate, its installation on only ~10% of the fleet, together with a high dropout rate due to its mechanical tape system, meant that less than 1% of the A-10 flight record was often recorded. Although the ABU-15/A was present on every aircraft, it was limited to recording Nz exceedances, and at only 6 different levels. The TEMS-ADR utilizes the digital technologies of flash memory, in-flight data processing, data compression algorithms, and the aforementioned digital bus to store far more flight data than could have been dreamed of in 1975.

Currently the TEMS-ADR is installed on 37 aircraft in order to fulfill the MIL-STD 1530C requirement for L/ESS, with plans to expand the coverage to 100% of the fleet in 2013.

Preliminary analysis of the TEMS-ADR data has shown excellent correlation with the legacy systems. But the larger question remains of how to efficiently store all of the anticipated information and how to utilize it effectively. With the maturation of the A-10 airframe, additional critical Control Points are being defined which require detailed spectra data in order to compute probabilities of failure to support the risk-based maintenance induction methodology and to determine lifetimes of components that have been swapped throughout the fleet. An additional challenge is that of extrapolating the previously recorded low-fidelity data to make accurate predictions for the A-10 extended service life. This technical activity addresses plans for incorporating these IATP enhancements.

Figure 9.5-6. A-10 Aircraft

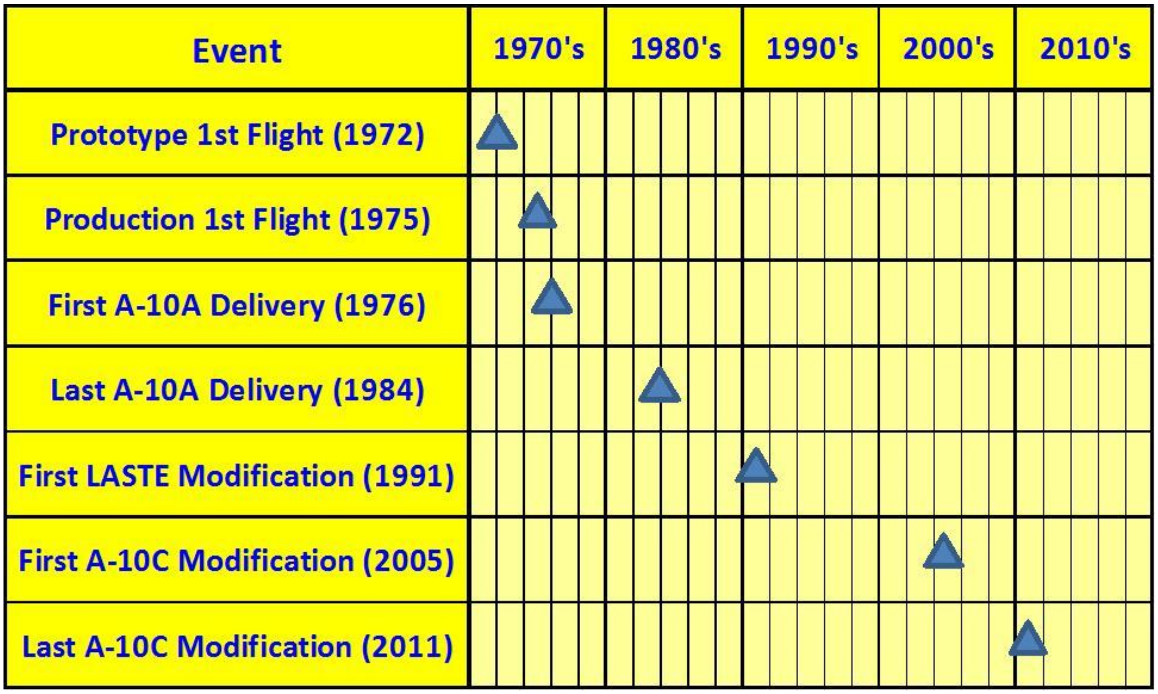


Figure 9.5-7. A-10 Program Milestones

9.5.4. Usage and Maneuver Loads Monitoring of Heavy Air Tankers

Kamran Rokhsaz and Linda K. Kliment, Wichita State University-Department of Aerospace Engineering; John Nelso, United States Forest Service; and James Newcomb, Federal Aviation Administration-Department of Aerospace Engineering

The United States Forest Service, through the Federal Aviation Administration William J. Hughes Technical Center, funded Wichita State University to conduct a survey of the operational loads experienced by a fleet of heavy air tankers. The program involved using data collected by various P2V and P3A air tankers in actual operation over several seasons. The majority of the data used for this study was collected during the 2008 and 2009 fire seasons, using a digital flight data recorder. This data set consisted of 5,316 flight files, although not all were useful due to a variety of reasons. Some results were also extracted from the data collected prior to 2008 using an older analog system. This data set consisted of 3,958 flight files from the 2007 and 2008 fire seasons.

Basic flight parameters, such as airspeed, altitude, flight duration, distance, and bank and pitch angles, were examined and presented in statistical form. Flights were divided into multiple phases (Figure 9.5-8), separating the segments when the retardant is dropped from other phases of flight. *V-n* diagrams and several coincident flight events are shown and compared with operational limits when available (Figure 9.5-9). In addition, maneuver loads were determined for various phases, leading to phase-specific exceedance charts.

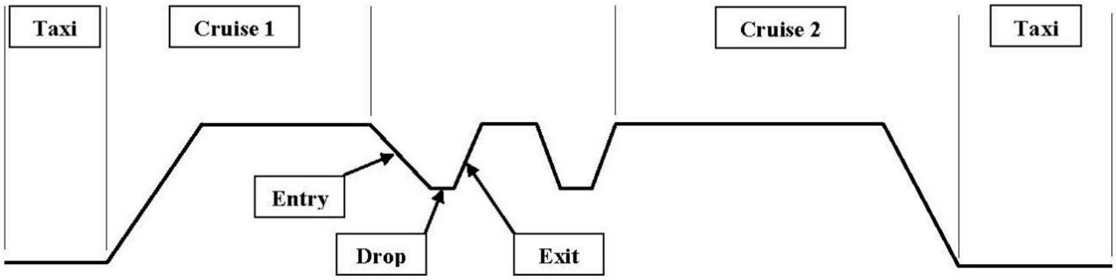
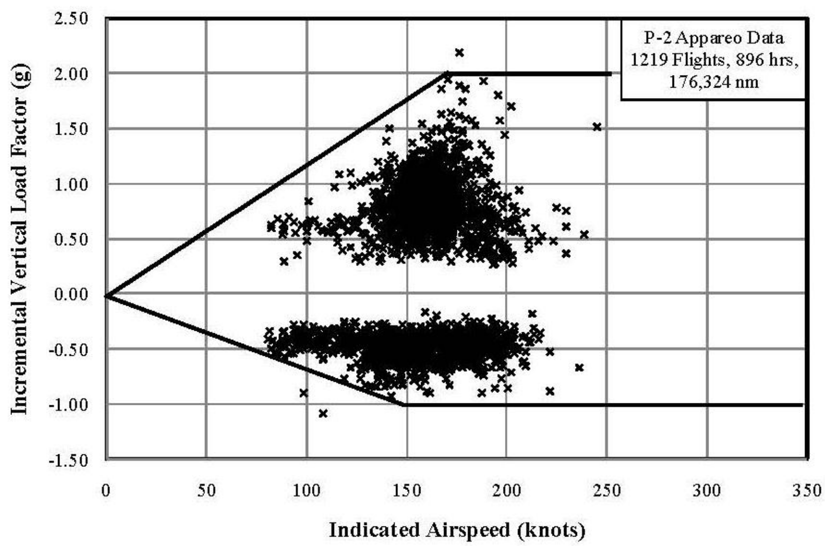
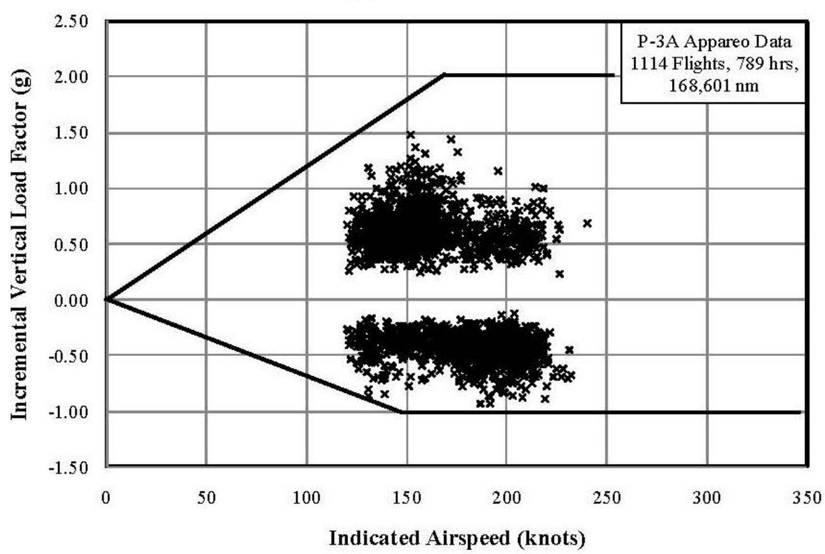


Figure 9.5-8. Various Flight Phases



(a) P2V Data



(b) P-3A Data

Figure 9.5-9. Typical *V-n* Diagrams

Cumulative occurrences of incremental maneuver load factors did not show any significant altitude dependence. Except for the drop and the exit phases, these loads occurred at lower frequencies than those stated in MIL 8866. Comparison of the incremental vertical maneuver load factors between the two recording systems showed higher frequencies of occurrence from the digital flight data recorders system. Nonetheless, the maneuver loads measured by this system were at or below those stated in MIL 8866 and well below those shown in other studies (Figure 9.5-10). Normalizing the loads by instantaneous aircraft weight showed a further reduction of the frequencies of occurrence to values well below those of MIL 8866 and presented by other references (Figure 9.5-11).

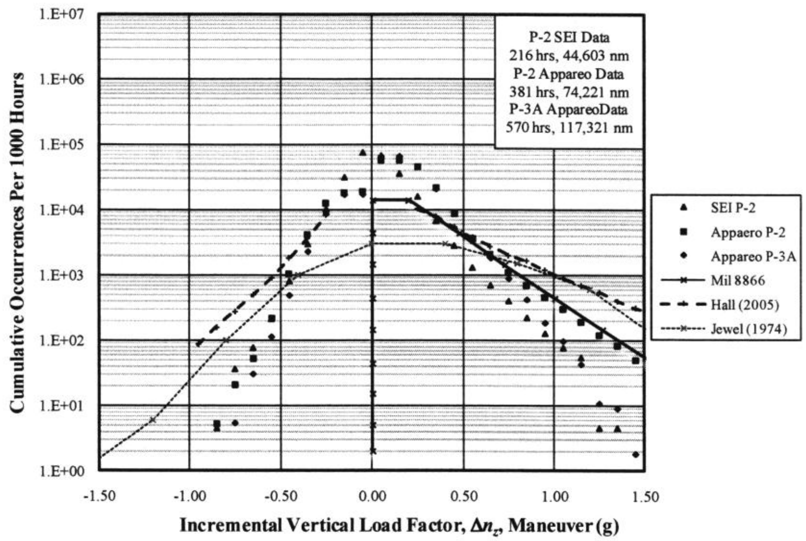


Figure 9.5-10. Comparison of Maneuver Load Exceedance Charts from Various Sources

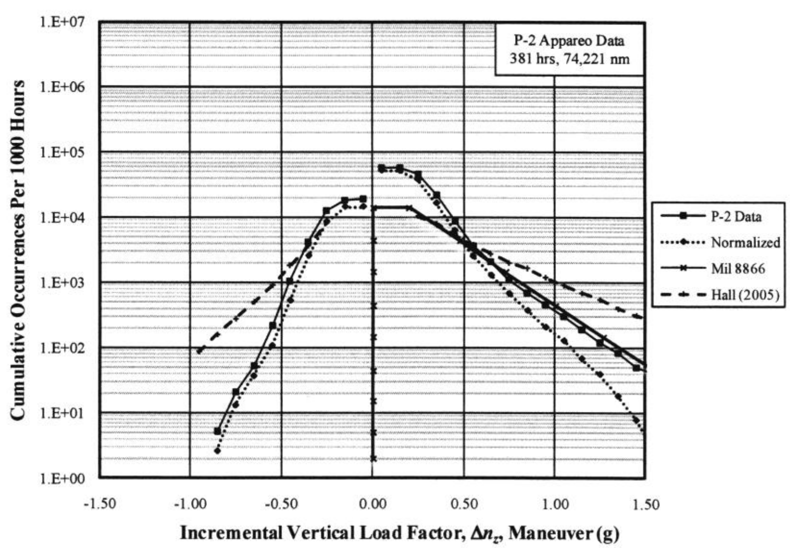


Figure 9.5-11. Exceedance Charts Normalized by Instantaneous Weight

9.5.5. An Innovative Low-Maintenance Data Acquisition Solution for Load Factor Capture

Elisabeth O'Brien, Curtiss-Wright Avionics & Electronics and Jason Niebuhr, USAF Academy-CAStLE

The USAF Academy Soaring Program trains cadets in basic soaring, aerobatics, and cross-country flying. From 2002-2011, these missions were primarily flown in several variants of the TG-10 glider. The service life of the TG-10C variant used for aerobatics was halved when the usage was determined to be more severe than originally expected. Ultimately an economic decision was made to replace the entire TG-10 fleet with the TG-16A (Figure 9.5-12). In order to better understand and monitor glider usage within the cadet training environment, a highly portable data acquisition system (DAS) has been developed for the TG-16A by the USAF Academy's Center for Aircraft Structural Life Extension (CAStLE).

Figure 9.5-12. USAFA TO-16A

In April of 2012 CAStLE started the development of a data acquisition system that captures load factor counts, is small enough to fit on the glider and flexible enough to be swapped between aircraft so as to continue capturing data when the gliders are in for scheduled maintenance (Figures 9.5-13 and 9.5-14). The data acquisition systems (DAS) were installed and started recording data in the summer of 2012 and will continue for at least a year, capturing data as the cadets progress through training (Figures 9.5-15 through 9.5-19). In addition to comparing the load factor exceedances of the different mission profiles, CAStLE will monitor the load factors over time and compare them to the OEM's load spectra to help establish the useful life of the aircraft.

Figure 9.5-13. Production Units

Figure 9.5-14. Faceplate

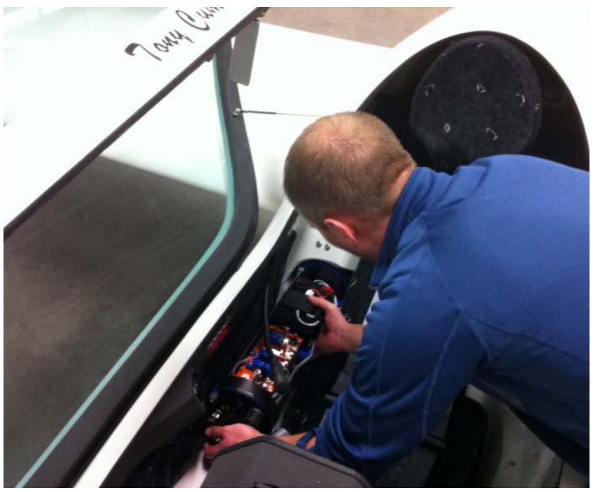


Figure 9.5-15. System Installation

Figure 9.5-16. System Installed

Figure 9.5-17. GPS Installation Behind Headrest

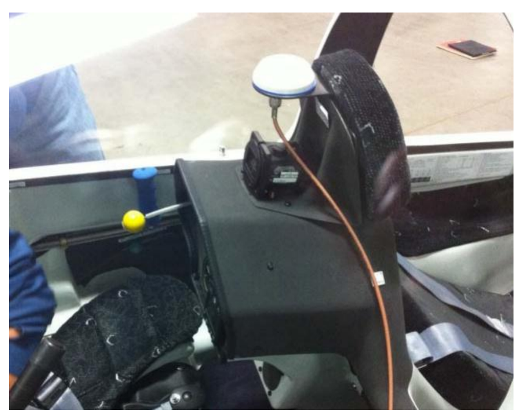


Figure 9.5-18. GPS Cable Routing

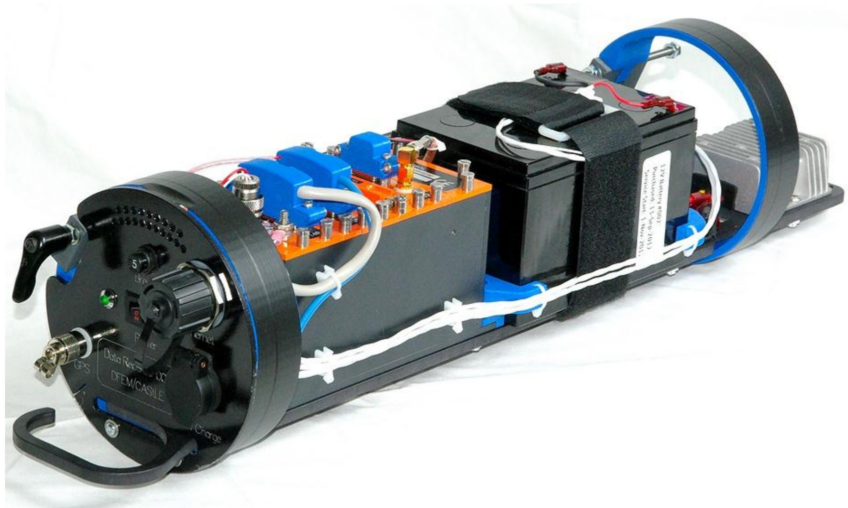


Figure 9.5-19. Close-up Production System

The units are built around a 6 slot Acra KAM-500 chassis and will record acceleration in 3 axis, temperature, and GPS parameters. The unit is completely self-contained (sans the GPS) and weighs 16.6 lbs. The battery makes up 50% of the system weight and can run for 7 to 8 hours, in line with a day of cadet training activities. The DAS is populated with four cards: an Ethernet controller, a time code generator, an A/D converter and a solid-state storage device. The Ethernet module is a 100BaseTX Ethernet backplane controller used to program the system and provide data for real-time analysis. The time code generator uses an onboard GPS receiver to provide the system with accurate time and navigation data. The A/D converter is an 8-channel module that is connected to the accelerometer and outputs temperature and three axis acceleration. The solid-state storage device is a 4GB CompactFlash card used to record the data. The Acra KAM-500 has passed environmental testing for flight qualification, therefore additional ruggedization or environmental qualifications are unnecessary to make it suitable for this application.

On-board the TG-16A glider is an oxygen (O_2) bottle holder for use in high altitude, cross country flights. Because the TG-16A gliders are being used only for aerobatic and basic training missions, the O_2 bottle holders were identified as the optimal location to install the system. As it is easily accessible from the pilot's position, turning the unit on prior to take-off and off after landing is trivial. The use of the oxygen tube means that the unit can be switched from aircraft to aircraft with no maintenance required for wiring or mounting the system. This was enabled by an innovative packaging scheme and the Acra KAM-500's rugged, compact design and low size, weight and power (SWAP) characteristics.

9.5.6. Flight Data Collection and Analysis

Gregory Shoales, USAF Academy-CAStLE

CAStLE has previously performed data collection and analysis efforts on the TG-10 sailplane and the HC-130H (USCG). The sailplane program helped identify causes of tailwheel collapse experienced by some aircraft, as well as characterize a new runway surface: AvTurf™. The HC-130H project helped the Coast Guard quantify mission severity at multiple basing locations, adding hours to the fleet of 26 aircraft.

Currently CAStLE is involved in a project that is measuring g-loading on the new TG-16A gliders flying two missions: standard and acrobatic. The purpose of the project is to determine if these two missions are similar enough (in terms of g exceedances) to allow them to be managed as one fleet, from an ASIP perspective. Another goal of the project is to compare USAFA usage to the “baseline” manufacturer’s usage. For the current project, CAStLE developed a fleet of modular units (Figure 9.5-20) to instrument multiple aircraft. These units fit in the oxygen bottle mounting tube (not currently used for oxygen), which is located near the fore-aft center of gravity of the aircraft. The units each have a triaxial accelerometer sampling at a high sample rate (f_s = 512 Hz) filtered at f_s/4 = 128 Hz. This high sample rate is being used as previous work with the glider fleet showed some higher frequency content during ground operations (mainly when the glider is towed behind the “gator” ATV on rough terrain). This sample rate may be adjusted depending upon initial results. The ACRA KAM-500 Data Acquisition Unit (DAU) includes cards for accelerometer data, GPS, and CompactFlash™ card recording. This unit is the subject of another ICAF 2013 abstract.

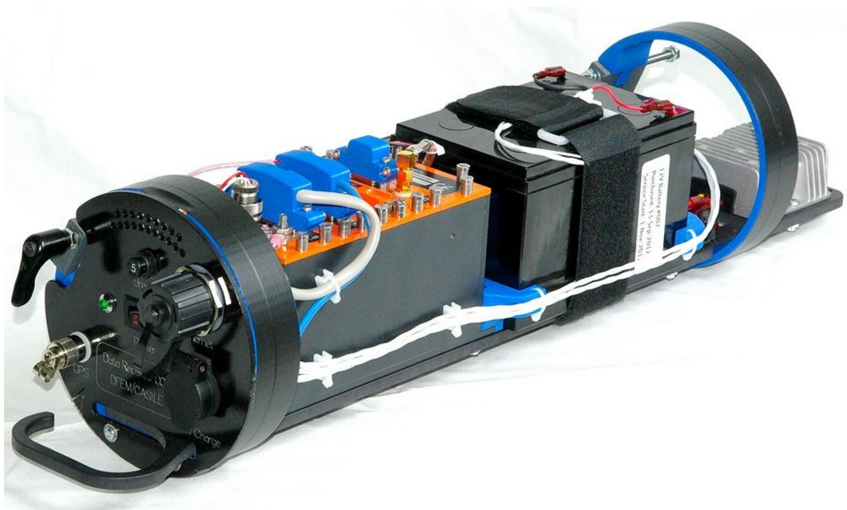


Figure 9.5-20. CAStLE Data Acquisition Module Including Battery, DC-DC Voltage Converter, and the ACRA KAM-500 DAU

9.6. CHARACTERIZATION, MODELING & TESTING

9.6.1. Overview of the Full-Scale Static and Durability Tests on F-35 Lightning II Program

Marguerite Christian and Don Whiteley, Lockheed Martin Corporation

The Aircraft Structural Integrity Program for the F-35 Lightning II is unique in that it includes dedicated full-scale static and durability test articles for each of the three variants included in the program: Conventional Take Off and Landing (CTOL), Short Take Off and Vertical Landing (STOVL), and Carrier Variant (CV) (Figures 9.6-1 and 9.6-2). These tests are a key component of the structural certification process and provide the data required to validate the structural analyses and to demonstrate the strength and stability of the airframe. The static and durability tests enable efficiencies through test consolidation and also through economies of scale. Investments made in the test fixtures and data acquisition systems coupled with efficient test protocols enable testing to progress rapidly and efficiently. Durability testing of the Horizontal Tail (HT) and Vertical Tail (VT) components is conducted in dedicated fixtures at BAE Systems in Brough, England. The CTOL HT completed the required two lifetimes of testing (16,000 test hours) in May 2011 and the STOVL HT achieved 12,000 test hours in June 2011. Testing of the CV HT is planned to start in October 2011. The CTOL and STOVL VTs are progressing through the first lifetime of testing; these tests differ from the HTs in that buffet loading is applied dynamically whereas all HT loading is applied quasi-statically. Testing of the CV VT is planned to start in early 2012. This technical effort provides an overview of the test methodologies, challenges faced and the results to date for each of the HT and VT durability test articles. The F-35 Static and Durability Test Programs, developed to satisfy the requirements of MIL-STD-1530C, continue to demonstrate the structural integrity of the F-35 airframe design and provide a model for the remaining variants as well as future aircraft programs.

Figure 9.6-1. Tri-Variant Joint Strike Fighter (JSF)

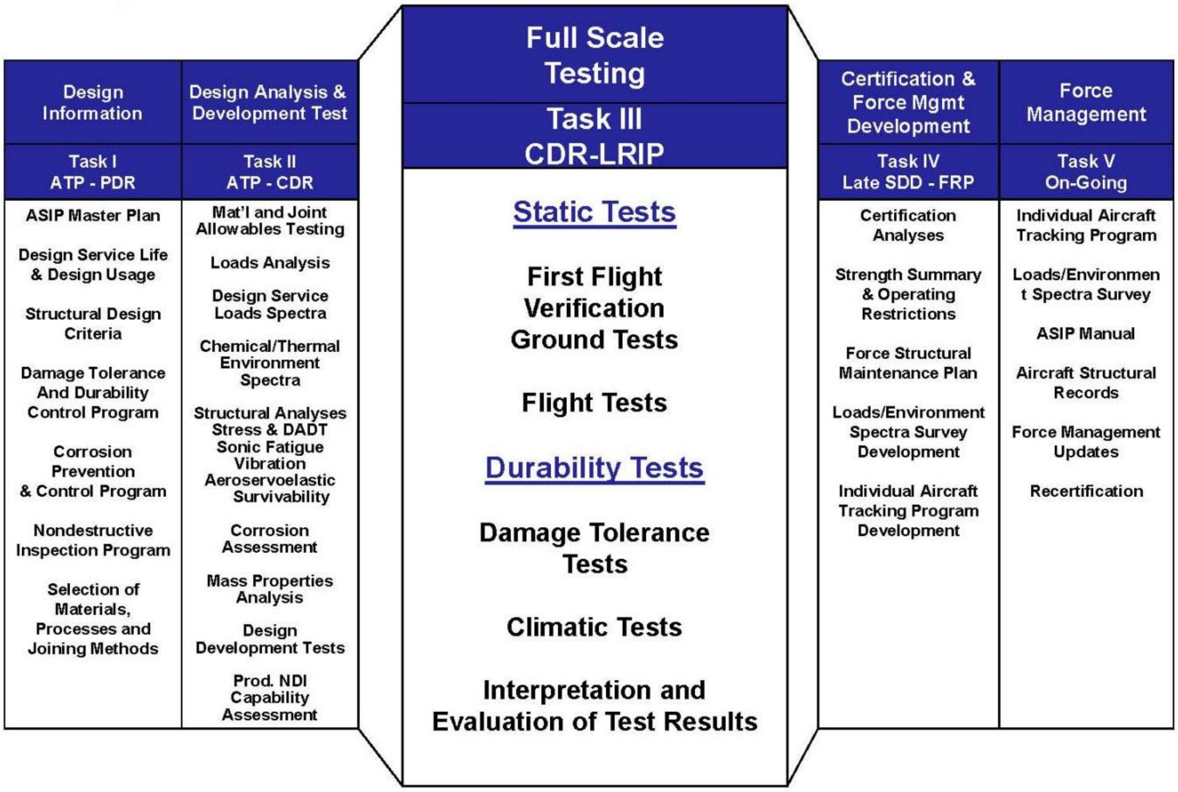


Figure 9.6-2. F-35 Full-Scale Tests and How They Relate to ASIP

9.6.2. Next Generation Crack Growth Predictions – Coupled Finite Element Modeling and Crack Growth Analysis

Joshua Hodges, USAF-OO-ALC

Traditionally, stress intensity solution development and crack growth predictions are developed independently. For standard geometries and loading, this typically works quite well. However, for complex geometry and/or loading, varying crack aspect ratios, multiple cracking scenarios, etc., this classic approach doesn't always fit. Synergies between the factors that affect the overall crack shape and growth are not necessarily captured, and thus can have a significant influence on the crack growth life.

The T-38 and A-10 analysis groups have developed a generic AFGROW plug-in that couples stress intensity development via StressCheck with AFGROW's crack growth analysis capability (Figure 9.6-3). This new capability allows AFGROW to open, update, solve, and extract solutions from parameterized StressCheck models automatically. Solutions are imported into AFGROW, crack growth is calculated, and the new crack geometry is sent back to StressCheck. This process is repeated automatically until a defined failure or stop criteria is reached. This seamless integration allows for more accurate crack growth predictions in complex situations and eliminates many of the assumptions that are required with the traditional approach.

This technical effort describes the development of the code, keys to building a proper StressCheck model, limitations of the plug-in, as well as future applications and directions of this capability. It also presents comparisons to Classic and Advanced AFGROW solutions (Figures 9.6-4 and

9.6-5). Finally, test data will be presented to help focus the experimental validation process.

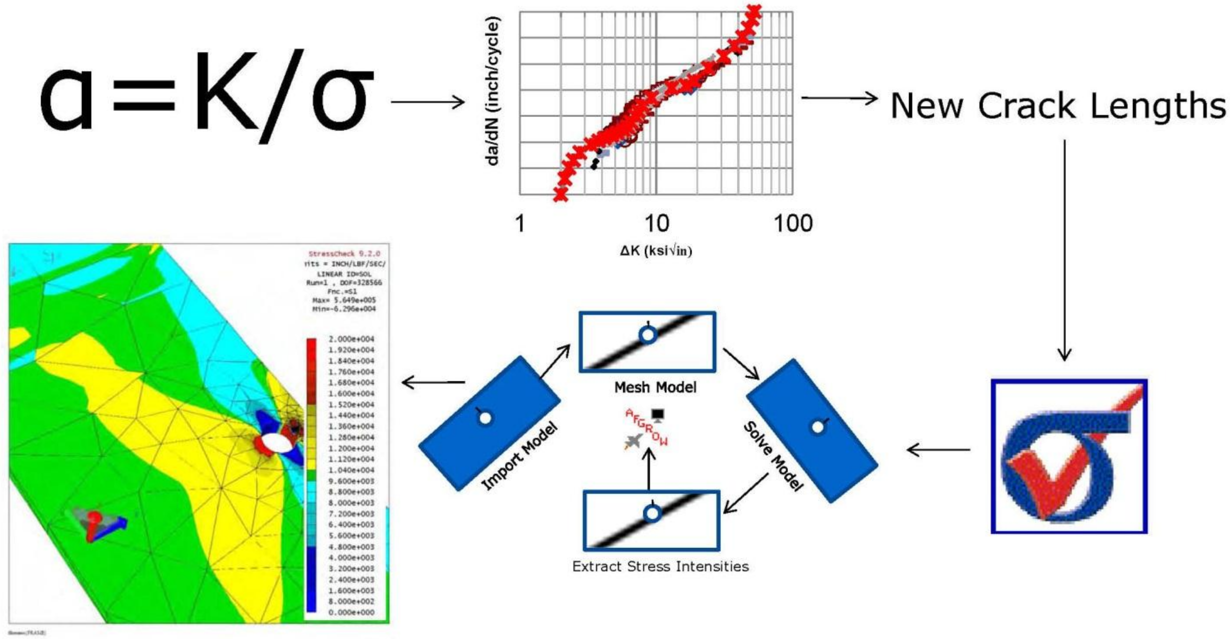


Figure 9.6-3. Coupling of Stress Intensity Development and Crack Growth Analysis

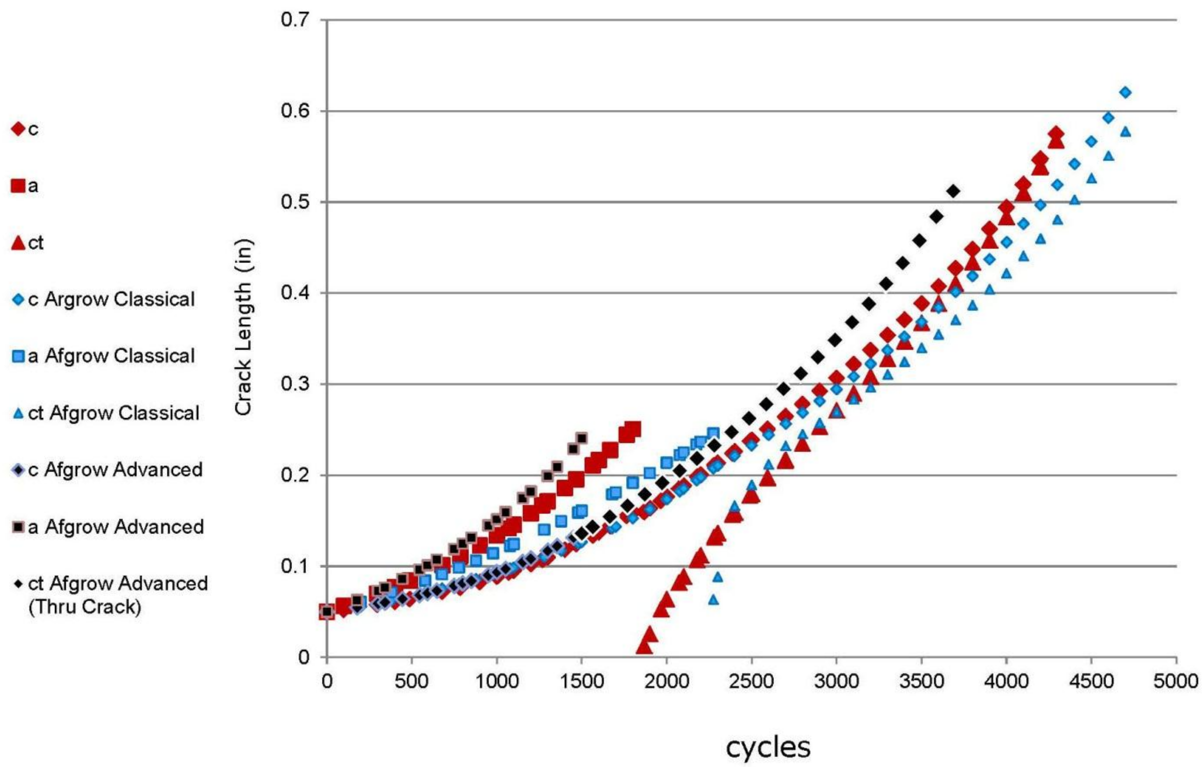


Figure 9.6-4. Comparison to AFGROW (B = 2.0, W = 5.0, t = 0.25, D = 0.25, Constant Amplitude SMF = 30.0)

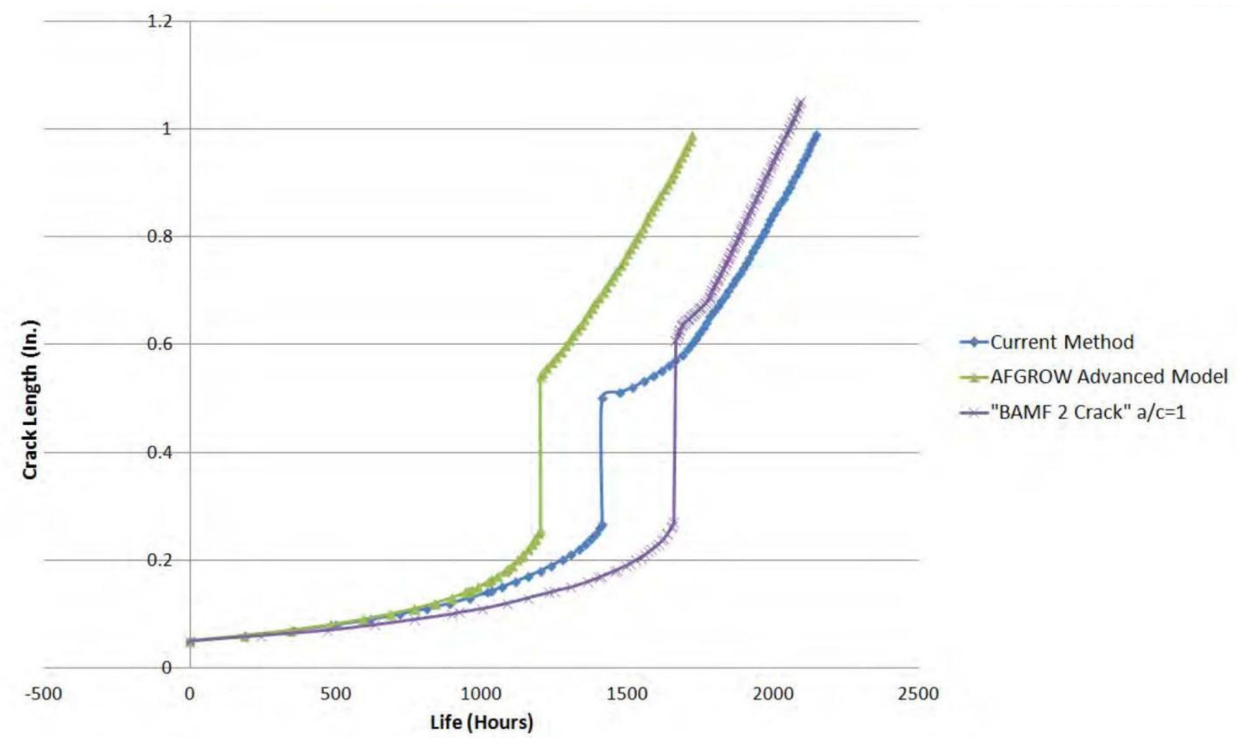


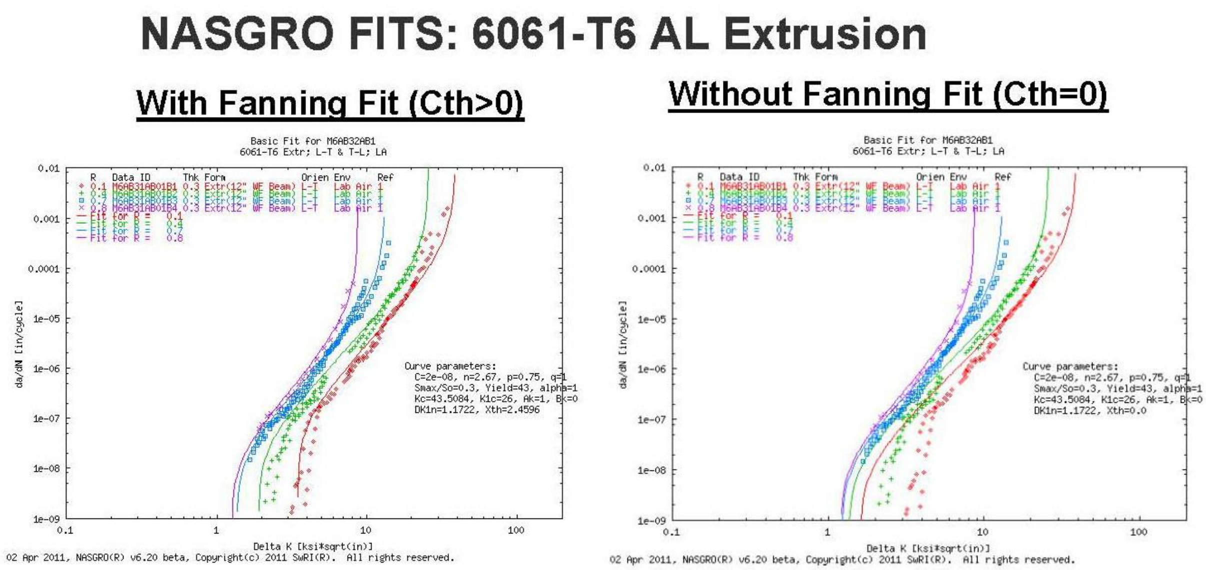
Figure 9.6-5. Comparison to AFGROW (B = 0.38, W = 2.9, t = 0.15, D = 0.204, SOLR = 3.33)

9.6.3. Crack Growth Behavior in the Threshold Region for High Cycle Loading

R. Forman, J. Figert and J. Beek, NASA-Johnson Space Center; and J. Ventura, J. Martinez and F. Samonski, Jacobs ESC Group

This technical effort describes results obtained from a research project conducted at the NASA Johnson Space Center (JSC) that was jointly supported by the FAA Technical Center and the JSC. The JSC effort was part of a multi-task FAA program involving several United States laboratories and initiated for the purpose of developing enhanced analysis tools to assess damage tolerance of rotorcraft and aircraft propeller systems. The research results to be covered in this technical effort include a new understanding of the behavior of fatigue crack growth in the threshold region. This behavior is important for structural life analysis of aircraft propeller systems and certain rotorcraft structural components (e.g., the mast). These components are often designed to not allow fatigue crack propagation to exceed an experimentally determined fatigue crack growth threshold value. During the FAA review meetings for the program, extensive discussions occurred between the researchers regarding the observed fanning (spread between the da/dN curves of constant R) in the threshold region at low stress ratios, R (Figures 9.6-6 through 9.6-8). Some participants believed that the fanning was a result of the ASTM load shedding test method for threshold testing, and thus did not represent the true characteristics of the material. If the fanning portion of the threshold value is deleted or not included in a life analysis, a significant penalty in the calculated life and design of the component would occur.

The crack growth threshold behavior was previously studied and reported by several research investigators in the time period 1970-1980. Those investigators used electron microscopes to view the crack morphology of the fatigue fracture surfaces. Their results showed that just before reaching threshold, the crack morphology often changed from a striated to a faceted or cleavage-like morphology. This change was reported to have been caused by particular dislocation properties of the material. Based on the results of these early investigations, a program was initiated at the JSC to repeat these examinations on a number of aircraft structural alloys that were currently being tested for obtaining fatigue crack growth properties. These new scanning electron microscope (SEM) examinations of the fatigue fracture faces confirmed the change in crack morphology in the threshold crack-tip region (Figure 9.6-9). In addition, SEM examinations were further performed in the threshold crack-tip region before breaking the specimens open (not done in the earlier published studies). In these examinations, extensive crack forking and even 90-degree crack bifurcations were found to have occurred in the final threshold crack-tip region. The forking and bifurcations caused numerous closure points to occur that prevented full crack closure in the threshold region, and thus were the cause of the fanning at low-R values. Therefore, we have shown that the fanning behavior was caused by intrinsic dislocation properties of the different alloy materials and were not the result of a plastic wake that remains from the load-shedding test phase. Also, to accommodate the use of da/dN data which includes fanning at low R-values, an updated fanning factor term was developed and will be implemented into the NASGRO fatigue crack growth software. The term can be set to zero if it is desired that the fanning behavior not be modeled for particular cases, such as when fanning is not a result of the intrinsic properties of a material.



NOTE: Cth is an empirical factor in the NASGRO equation that allows the choice of either fitting the threshold low-R data or a more conservative and assumed "closure free" fit which occurs in high-R load ratio testing.

Figure 9.6-6. Example of Concern in Fitting da/dN Thresholds

AerMet 100 Lab Air Data Fit – Cth = 2 .25

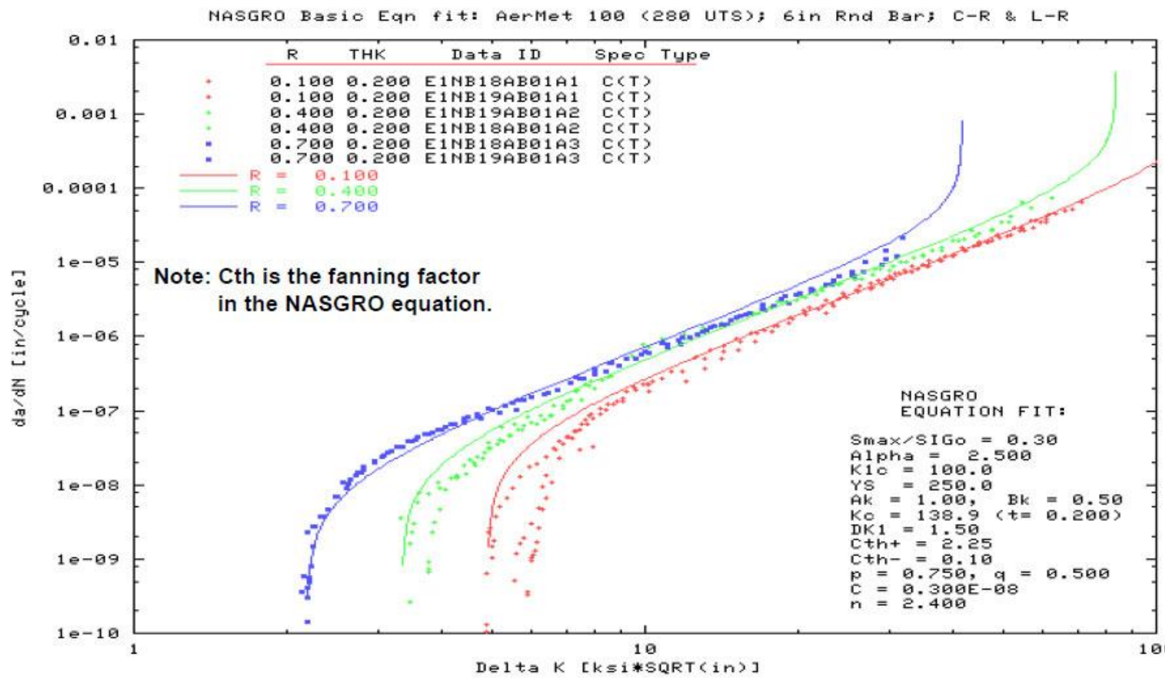


Figure 9.6-7. Example of Significant Fanning That Occurred in Lab Air Testing

AerMet 100 Dry Air Data Fit – Cth = 0

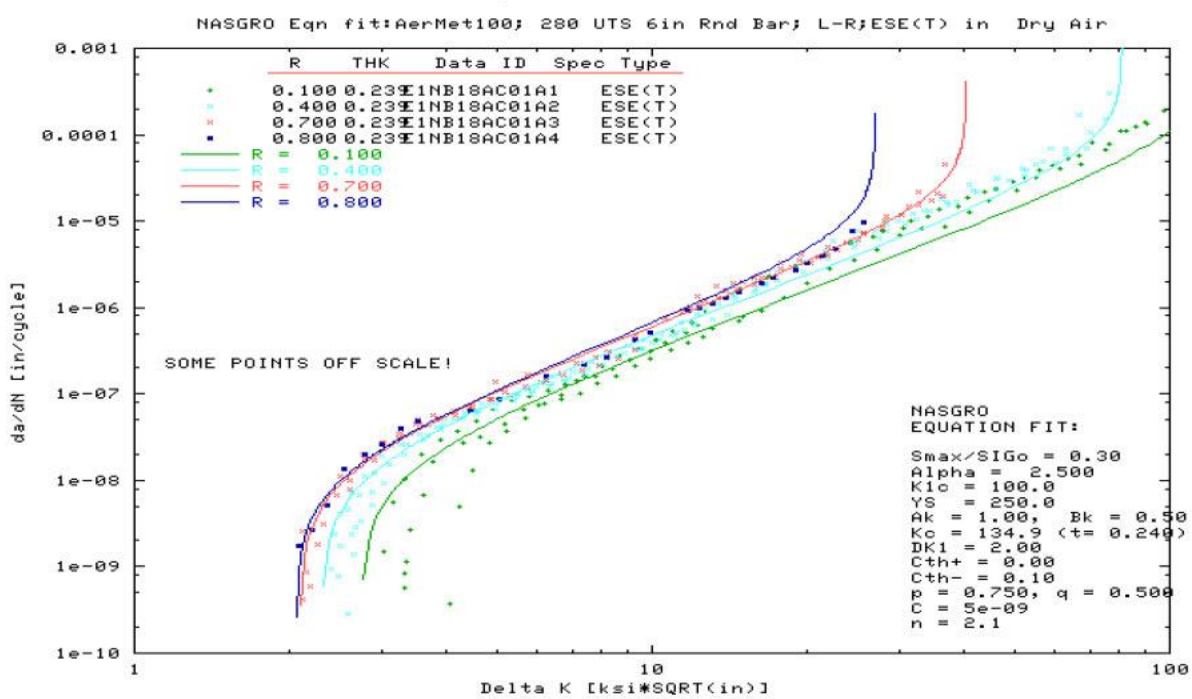


Figure 9.6-8. Example of Decreased Fanning in Dry Air Testing

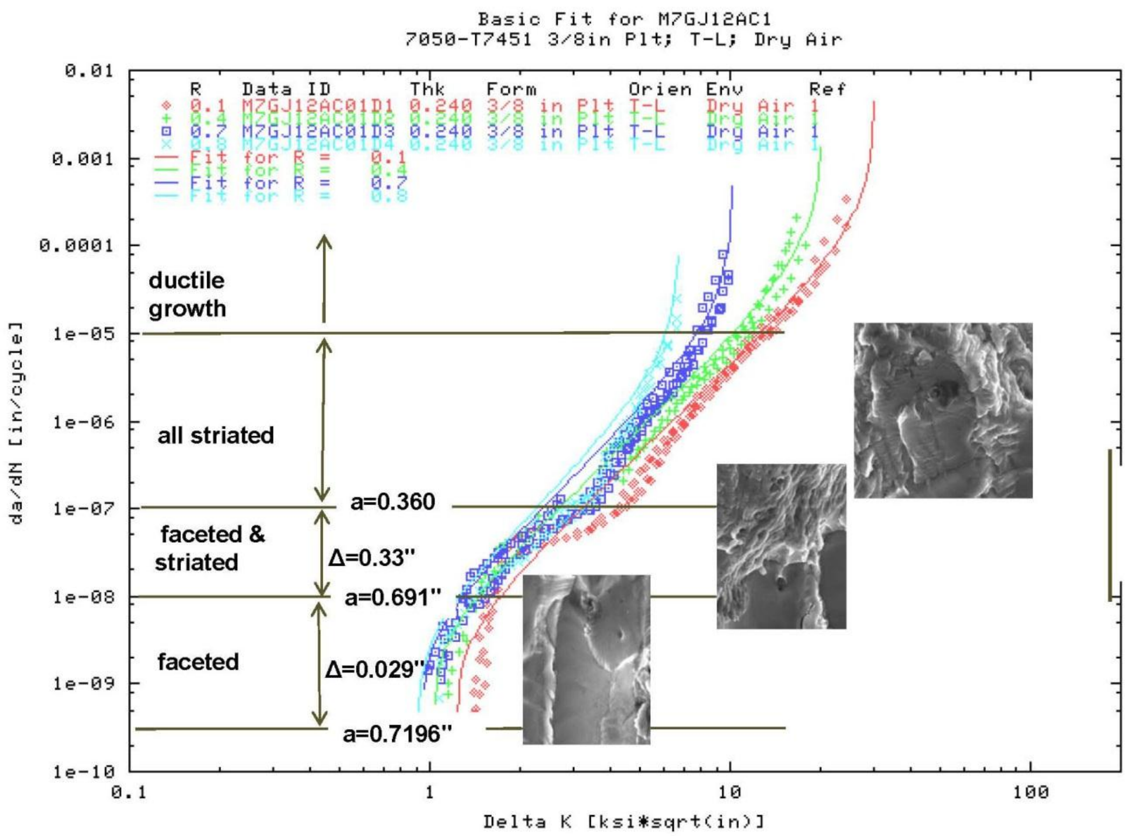


Figure 9.6-9. Variation in Crack Surface Morphology for 7075-T751 Tests

9.6.4. Durability Testing of the STOVL F-35 Lightning II

Joseph Yates, Lockheed Martin Corporation

Successful completion and correlation of findings from the full-scale durability testing of the STOVL F-35 Lightning II airframe structure is a key element of the F-35 Lightning II Joint Strike Fighter Aircraft Structural Integrity Program and the F-35 Structural Certification Plan. This is one of three full-scale durability tests and six tail component tests being performed as part of the F-35 program to verify that the airframe structure meets its durability requirements. As the F-35 Program progresses from design and development into full-scale ground and flight testing, a disciplined and rigorous approach is being applied to addressing the findings from both ground and flight testing. Careful consideration is given to identifying the root cause of each finding and the corrective actions necessary to address them. This technical effort describes the development of the F-35 durability test program for the STOVL variant (Figures 9.6-10 and 9.6-11), the development of the load spectrum for the test, the initial performance of the test and the evaluation of findings made to date in the test program (Table 9.6-1). The technical effort will also address the plans for the test going forward.

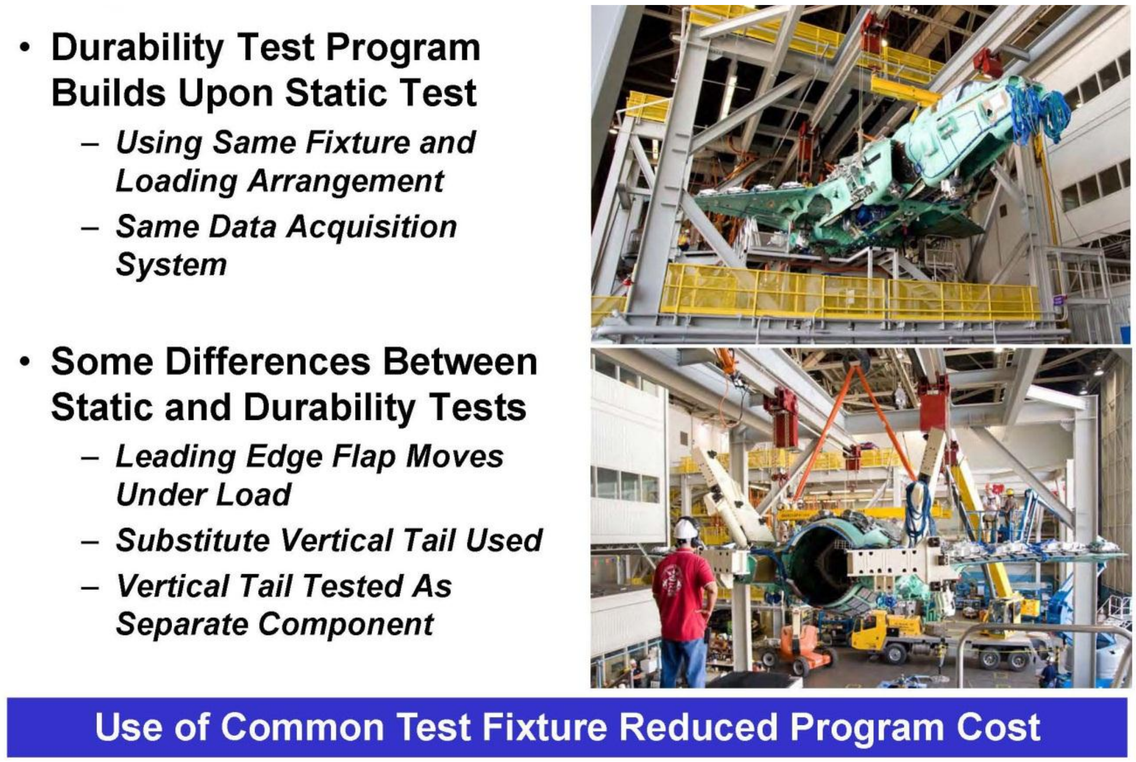


Figure 9.6-10. Full-Scale Durability Test Development

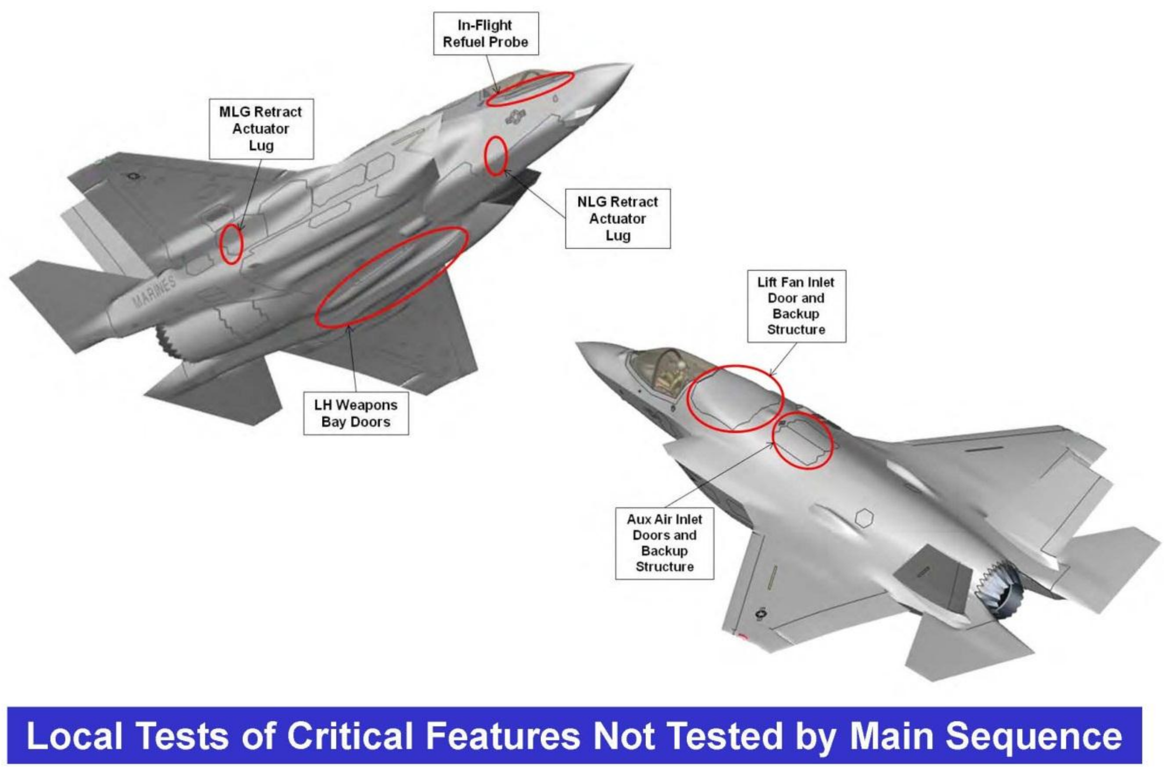


Figure 9.6-11. STOVL Local Durability Tests

Table 9.6-1. Summary of Findings to Date

Finding Location	Observation
EHAS Panel Fastener Rotation	During First 1000 Hours Block
FS 496 Bulkhead Trunnion	**Crack at 1275 hrs Maneuver**
In Flight Refueling Probe	Successful Completion of 16000 hrs
Nose Landing Gear Retract Actuator Lugs	**Lug Failure at 7182 hrs**
NLG Retract Actuator Test - Gusset	**Crack from Radius at 11000 hrs**
Nose Landing Gear Retract Actuator Lugs	**Lug Failure at 12965 hrs**
Main Landing Gear Retract Actuator Lugs	Successful Completion of 16000 hrs
Weapons Bay Door Inboard Cradle	**Cracks at Base of Lug at 11353 hrs**
Weapons Bay Door Outbd Hinge #4 (FS 450)	**Failure of Fitting at 13327 hrs**
Wing Pylon Rib Station #3	Tested Beyond Prediction
FS 503 Support Frame	Tested Beyond Prediction

9.6.5. The F-16 Block 50 Full-Scale Durability Test

Kevin Welch, Lockheed Martin Corporation

The F-16 Program is in the process of preparing for its third Full-Scale Durability Test (FSDT). Previous tests were performed for the Full-Scale Development Program (Figure 9.6-12) and for the F-16C/D Block 25/30/32 certification. The current test is in support of a potential extension of the Certified Service Life of the Block 50/52 (Figures 9.6-13 and 9.6-14) and the Block 40/42 configurations. The goal of the program is to test to a total life of 20,000 to 24,000 hours in order to support a service life extension from the current 8,000 hours to a new service life of 10,000 to 12,000 hours. The desire for the extended service life is to ensure an adequate posture for the United States fighter force structure. The feasibility of this extension has been demonstrated analytically using tools and methods honed from a large amount of heritage structural tests and a worldwide field experience of over 15 million flight hours. The FSDT program is expected to provide a physical demonstration of the practicality of a service life extension and to define structural locations requiring modification to meet anticipated service life extension requirements. This technical effort will provide a historical summary of the two previous durability tests, discuss the current need for the Block 50 FSDT (Table 9.6-2), identify the anticipated benefits of the test, describe the test plan and review the accomplishments to date.

Figure 9.6-12. F-16A Durability Test Airplane

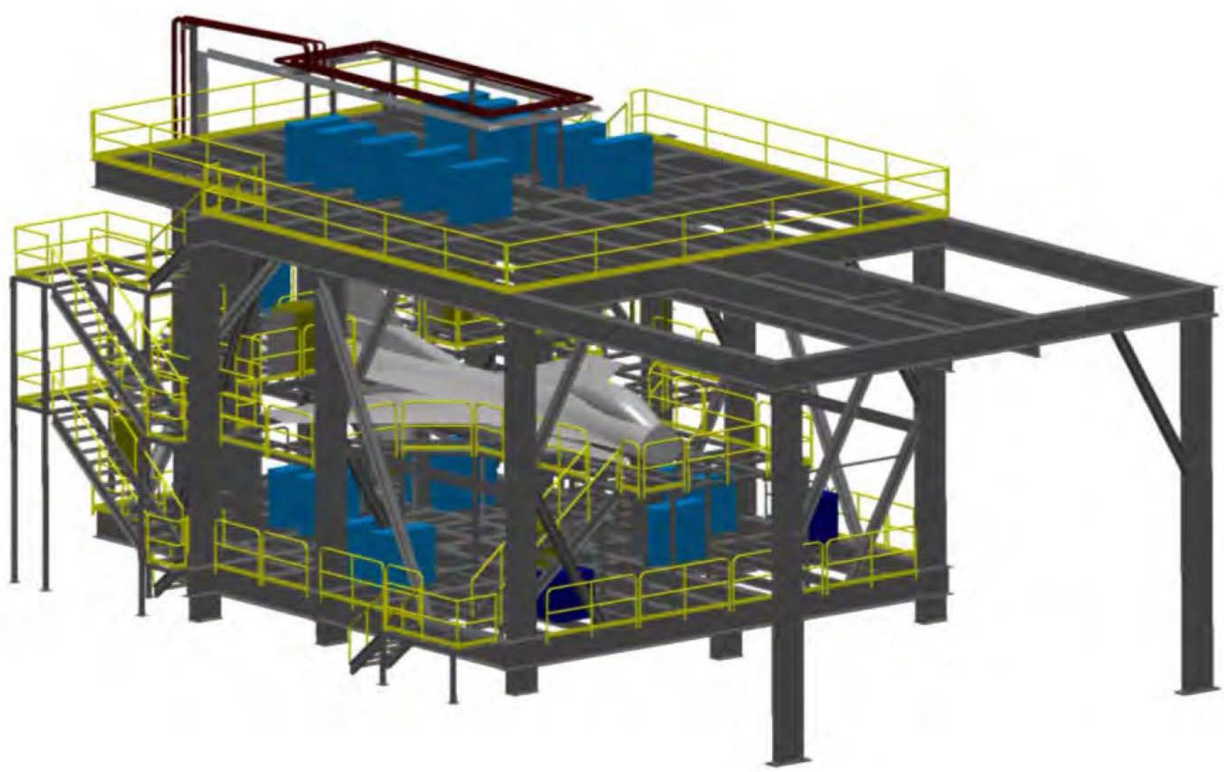


Figure 9.6-13. F-16 Block 50 Durability Test Fixture

Figure 9.6-14. Block 50 Test Article 91-0419/CC: 117

Table 9.6-2. F-16 Structural Design Criteria Severity Increased With Each Block

F-16 Type	G-level: Sym & Roll	Service Life (Hrs)	Basic Flt Des GW (lb.)	Max Des GW (lb.)	Wing Root Bend Mom (10^6 in.-lb.)	Upper Wing Skin Root Thickness (in.)
FSD	+9 / -3 +5.86 / -1	8000 hrs Requirement	22,500 (ratio:1.00)	33,000	3.168 (ratio:1.00)	0.30 (ratio:1.00)
Blk 15	+9 / -3 +5.86 / -1	8000 hrs Requirement	24,095 (ratio:1.07)	35,400	3.738 (ratio:1.18)	0.30 (ratio:1.00)
Blk 25/30/32	+9 / -3 +5.86 / -1	8000 hrs Goal	26,910 (ratio:1.20)	37,500	3.930 (ratio:1.24)	0.30 (ratio:1.00)
Blk 40/42	+9 / -3 +5.86 / -1	8000 hrs Requirement for 4 Baseline Spectra[1]	28,500 (ratio:1.27)	42,300	4.419 (ratio:1.39)	0.45 (ratio:1.50)
Blk 50/52	+9 / -3 +5.86 / -1	8000 hrs Requirement for 8 Baseline Spectra[2]	28,750 (ratio:1.28)	42,300	4.453 (ratio:1.41)	0.45 (ratio:1.50)

1 - 2 Mission Mixes and 2 Engine Configurations
2 - 2 Mission Mixes and 4 Engine Configurations

9.6.6. F-22 Aircraft Mounted Nozzle Sidewall-Testing Lessons Learned

Ken MacGillivray, USAF-F-22 Program Office

The Aircraft Mounted Nozzle Sidewall (AMNS) has been a significant cost and maintenance driver for the F-22 program. Located in the rear of the aircraft, extending aft of the engine sidewall, the part is exposed to a variety of loading regimes not typical to aircraft structure (Figures 9.6-15 through 9.6-17). Vibrationally driven high cycle fatigue and strains driven by modal shapes make analysis and testing of the AMNS difficult. With millions of dollars dumped into four redesigns, the design of this part has been anything but straightforward. The vibrational loading spectra has caused significant issues with design and was initially significantly under tested. This technical effort documents the testing lessons learned, re-design history, high cycle fatigue analysis, and lifing risk analysis methods for this unique part.

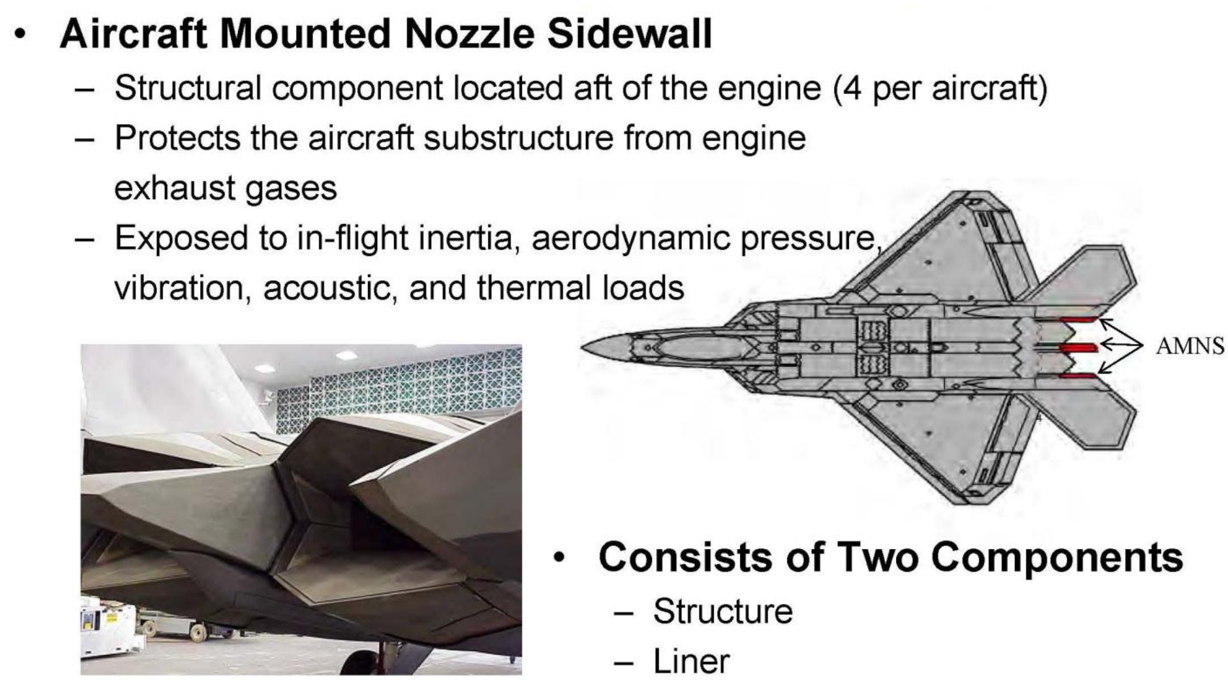
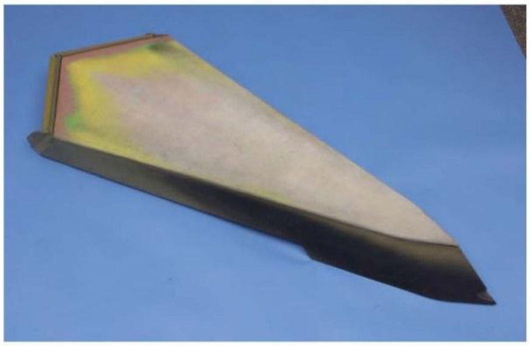


Figure 9.6-15. Location of Aircraft Mounted Nozzle Sidewall



AMNS Liner
(OML shown attached to structure)

- **AMNS Liner**
  - Contains protective high temperature LO coating
  - Mission Critical Hardware

- **AMNS Structure**
  - Attaches to aircraft via forward and aft mounts
  - AMNS OML contains liner
  - Safety Critical Hardware outboard location only

AMNS Structure
(IML side shown)

Figure 9.6-16. Components of Aircraft Mounted Nozzle Sidewall

- Aft Mount cracks were initially discovered in the field in late 2002.
- Root cause investigation showed higher than expected vibratory loads due to horizontal tail buffet, as well as, assembly tolerance issues

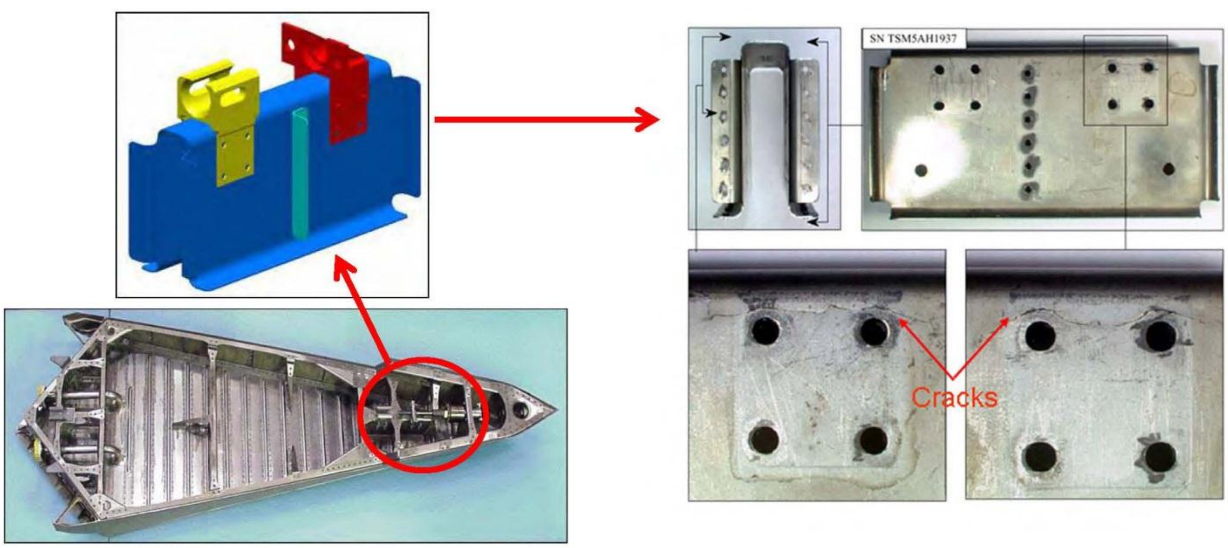


Figure 9.6-17. First Signs of Aft Mount Cracks

9.6.7. A Screening Approach for Determining Potential New F-15 Fatigue Critical Locations

Tony Bergman, Greg Strunks and Dan Bowman, University of Dayton Research Institute; and Dave Currie and Lucas Garza, USAF-WR-ALC

Fatigue-critical locations for the F-15 were originally selected using the same methods as other similar aircraft designed in that era. Critical locations were first chosen in areas of analytically determined high stresses and stress concentrations, were updated using full-scale fatigue test results, and then augmented as necessary after instances of in-service fatigue failures. Due to limitations in computational horsepower at the time the F-15 went to a damage tolerance philosophy, the number of potential fatigue-critical locations that could be analyzed for crack growth was small compared to the number of holes, radii, and other stress risers on critical structures. With advances in finite element modeling and crack modeling tools like AFGROW, it is now much easier to analyze larger sections of safety-of-flight structure. Engineers from the University of Dayton Research Institute (UDRI), in concert with the engineering staff at the Warner Robbins Air Logistics Center (WR-ALC), have been re-analyzing F-15 safety-of-flight structures in an effort to find any fatigue-critical locations that may have been missed in the original damage tolerance analysis. The key tasks in the project have included: 1) Examining the engineering drawings of all F-15 safety-of-flight structures to look for holes, radii, abrupt thickness transitions, and any other stress risers, 2) Visually inspecting parts and reviewing the results of the F-15 C and F-15 D teardowns performed by S & K Technologies, 3) Using the global FEM created by MERC and Boeing, narrowing the list of potential locations to analyze to those that are located in highly stressed areas of safety-of-flight structures, 4) Creating a theoretical stress spectrum for each location by scaling according to Nz, and 5) Performing crack initiation/growth predictions at each location using AFGROW. The results of this technical activity (Figures 9.6-18 and 9.6-19) have proved to be an efficient and effective way to qualitatively benchmark the relative severity of crack growth at a large number of locations. With further refinement of the inputs, the quantitative reliability of the crack growth predictions could be improved.

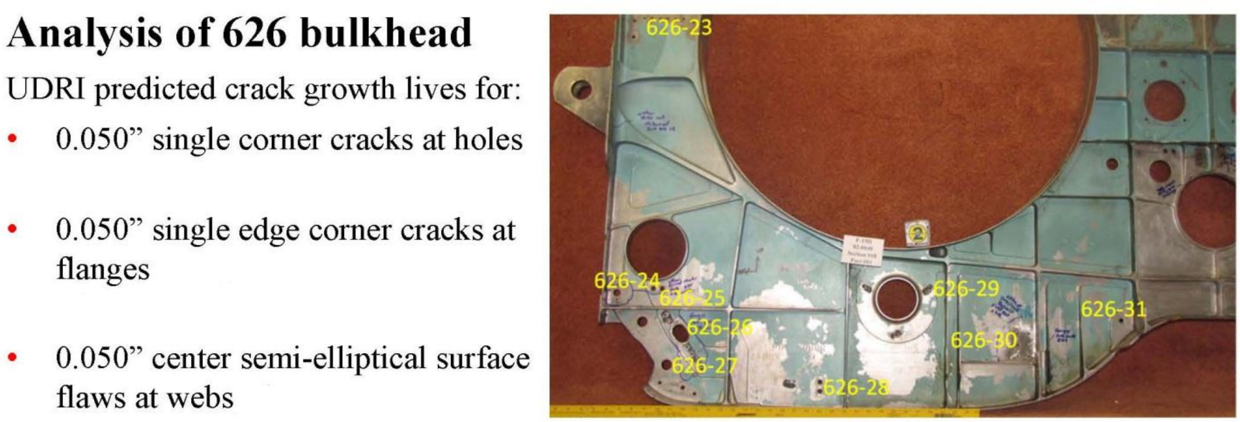


Figure 9.6-18. Crack Growth Predictions for FS 626 Bulkhead

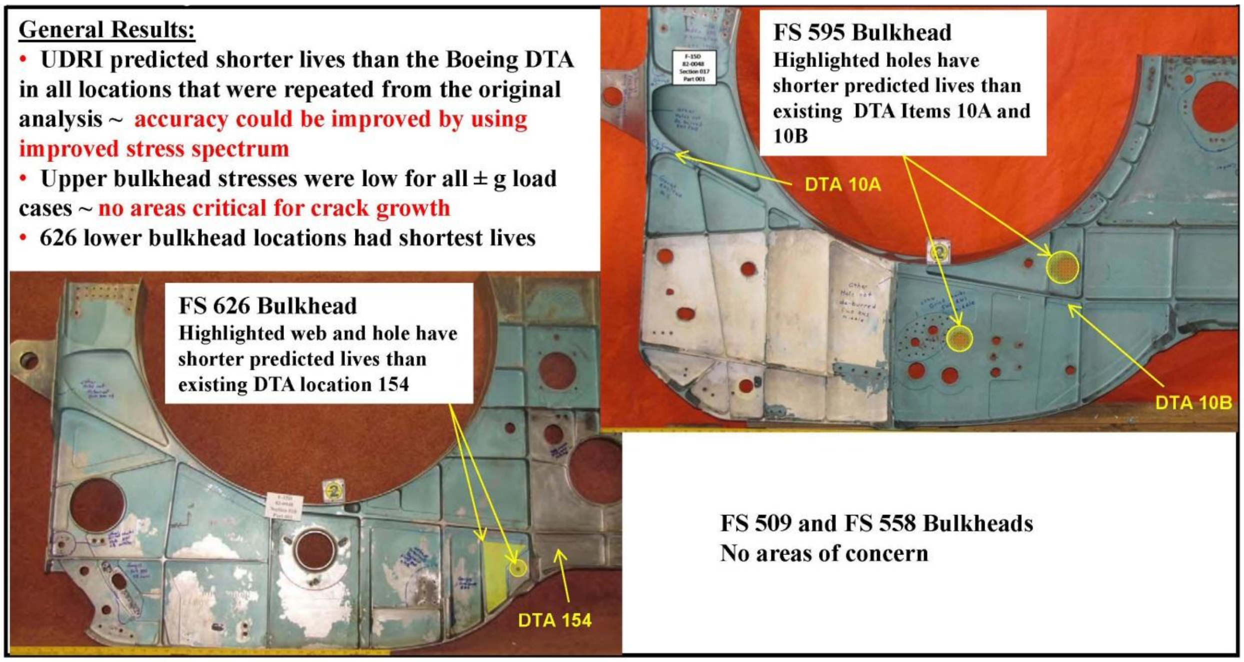


Figure 9.6-19. Crack Growth Results for FS 626 and FS 595 Bulkheads

9.6.8. The New Composite Materials Handbook 17 (CMH-17) Revision G

Yeow Ng, Wichita State University-NIAR; Larry Ilcewicz, FAA; and Rachael Andrulonis, MSC

This technical activity provides an overview of the Composite Materials Handbook 17 (CMH-17), with an emphasis on the recently released CMH-17 revision G (Figure 9.6-20). CMH-17 is a six-volume engineering reference tool, formerly known as MIL-HDBK-17, that contains more than 1,000 records of technical information for polymer matrix (Volumes 1, 2, and 3), metal matrix (Volume 4), ceramic matrix (Volume 5) and structural sandwich (Volume 6) composites. The Composite Materials Handbook organization creates, publishes and maintains proven, reliable engineering information and standards, subjected to a thorough technical review, to support the development and use of composite materials and structures. It is used by engineers worldwide in designing and fabricating products made from composite materials, particularly in the aerospace industry.

The latest revision G of CMH-17 includes ten years of new data and updates. Specifically, this technical activity covers the following items:

- Structure of the handbook, including the outline of each volume
- CMH-17 organization, including the working groups (Figure 9.6-21)
- Approval procedures for new or revised content
- Certification and statistics tutorials (Figure 9.6-22)
- How to use CMH-17 data successfully, including composite part fabrication site "process" equivalency demonstration and material & process control requirements
- Public and members' websites
- Commercial availability
- How to get involved in CMH-17 and meeting information

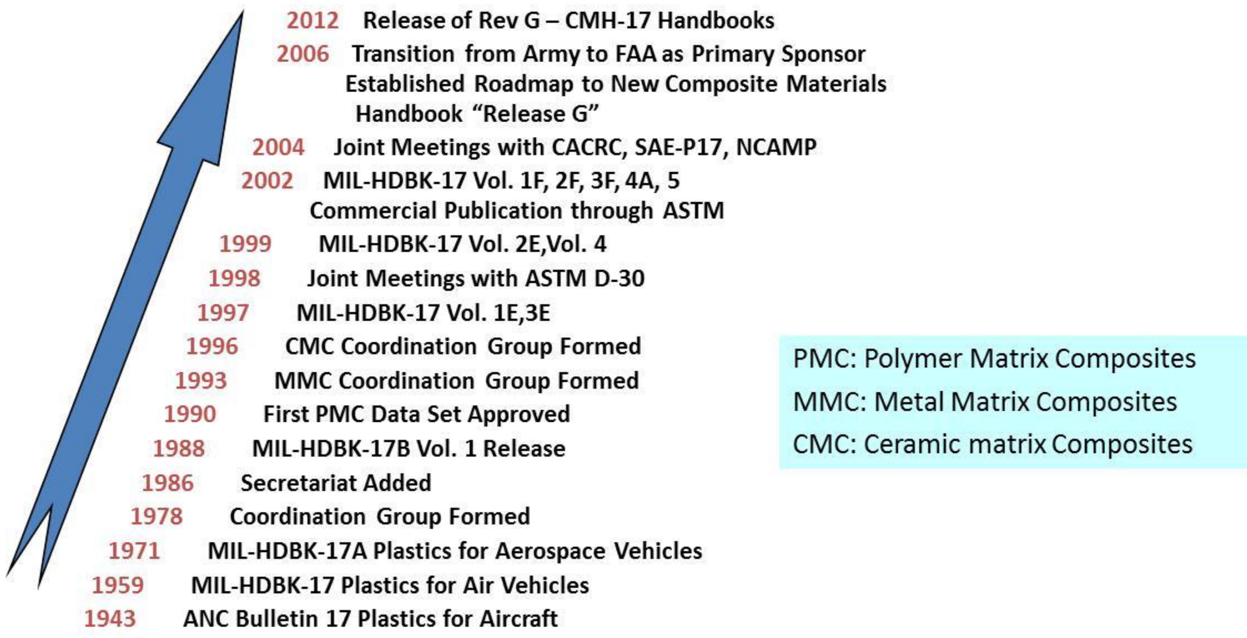


Figure 9.6-20. History of Composite Materials Handbook 17 (CMH-17)

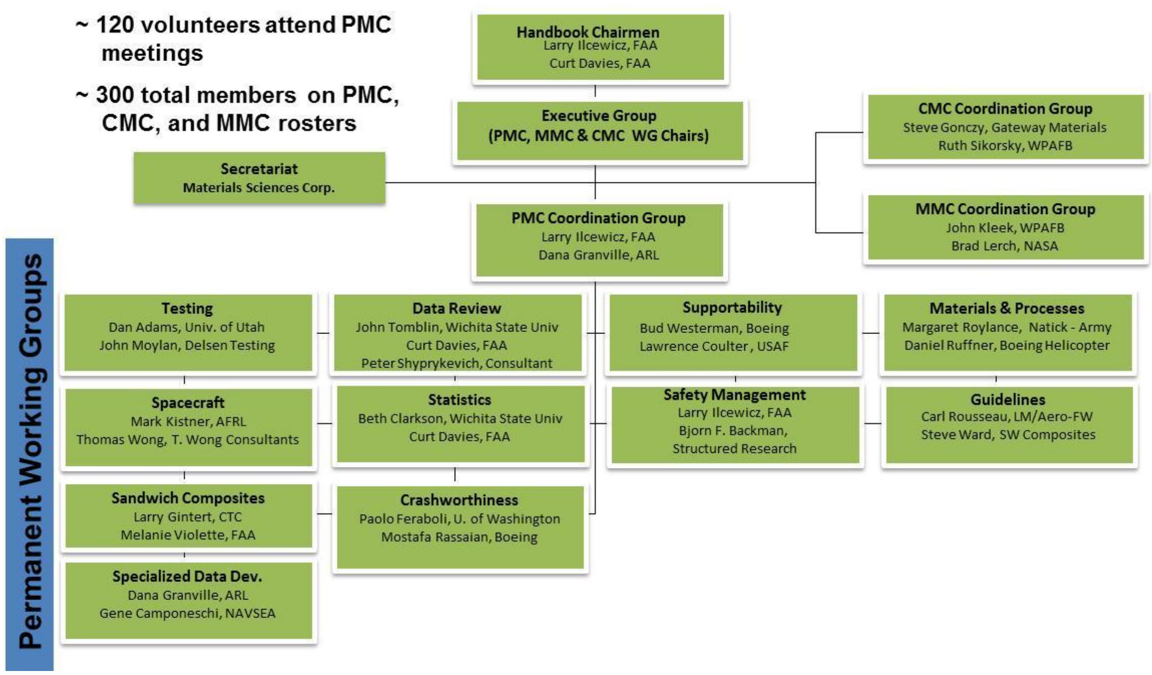


Figure 9.6-21. CMH-17 Organization

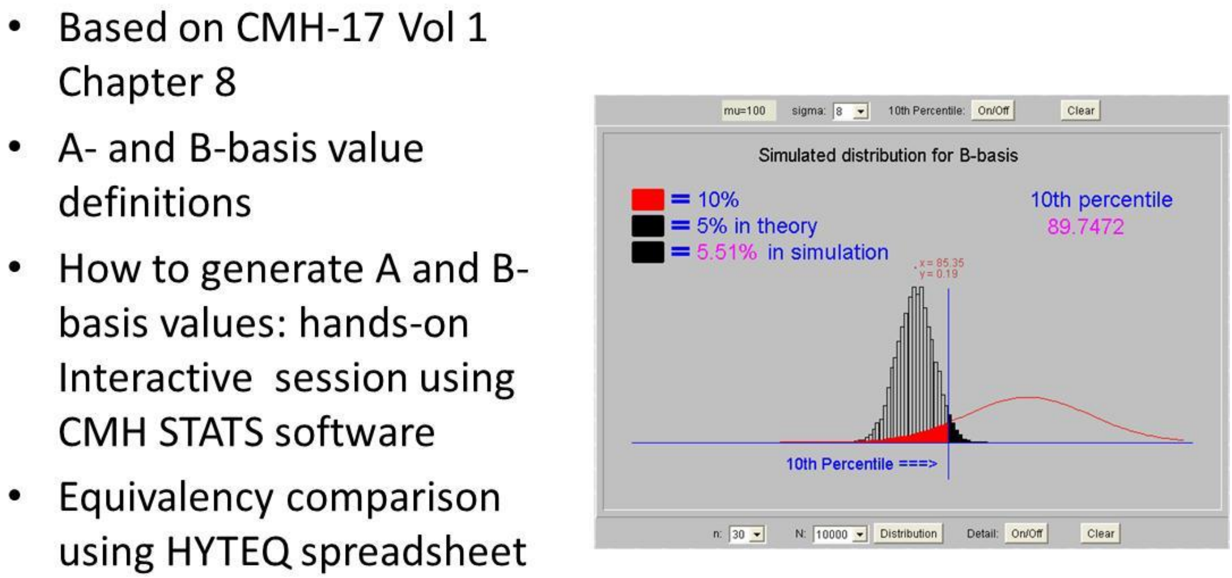


Figure 9.6-22. CMH-17 Statistics Tutorial

9.6.9. Models for Corner and Through Cracks in Support of Beta Curve Development

Matt Watkins, Ricardo Actis and Liliana Ventura, Engineering Software Research & Development, Inc.

Exact solutions for stress intensity factors (SIFs) are generally not available for cracks in the presence of complex geometric features and loading conditions. Beta factors developed to account for these features are often obtained as the superposition of the SIFs of several simpler cases, and beta curves are then compiled as a function of crack length to perform crack growth studies. An alternative to the superposition approach is the development of properly formulated models solved by the finite element (FE) method. This technical activity addresses the modeling strategy used to compute the complete beta curve across eight phases of crack growth at the CW-1 location of the wing skin of the C-130 aircraft using the commercial FE analysis software StressCheck.

For the eight phases of crack growth, a series of finite element meshes optimized for various ranges of crack lengths were created. The meshes were made parametric to interface with an automation script, to account for variability in manufacturing and repairs, and in support of uncertainty quantification. The CW-1 location (Figure 9.6-23) has significant geometric features in close proximity to the crack, including a beam cap, hat stringers, and four fastener holes along the crack path (Figure 9.6-24), leading to the requirement for both through-the-thickness and corner crack meshes. Corner crack meshes must be solved in 3D, so a technique is presented for simplifying the computational effort while maintaining high-quality results. To satisfy the requirements of solution verification, that is to control the error of approximation in the computation of the SIFs, a sequence of finite element solutions with an increasing number of degrees of freedom were obtained for each computed beta factor.

The modeling approach has several advantages over traditional beta factor development. The models are more reliable because the effects of simplifying assumptions can be evaluated and no superposition is required. In addition, effects can be easily accounted for such as traction-loaded vs. pin-loaded holes. This work was performed in support of the C-130 Service Life Assessment Program (SLAP) for the Naval Air System Command (NAVAIR) Aircraft Structures Division.

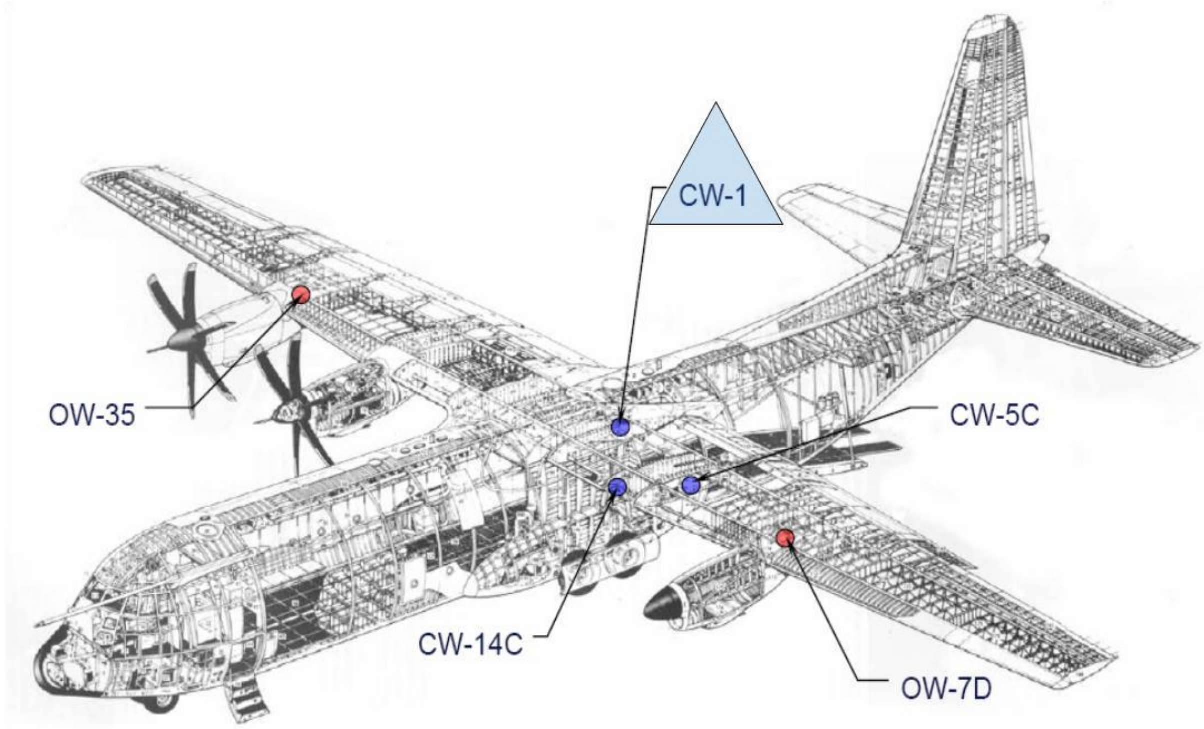


Figure 9.6-23. CW-1 Location of C-130 Lower Wing Surface

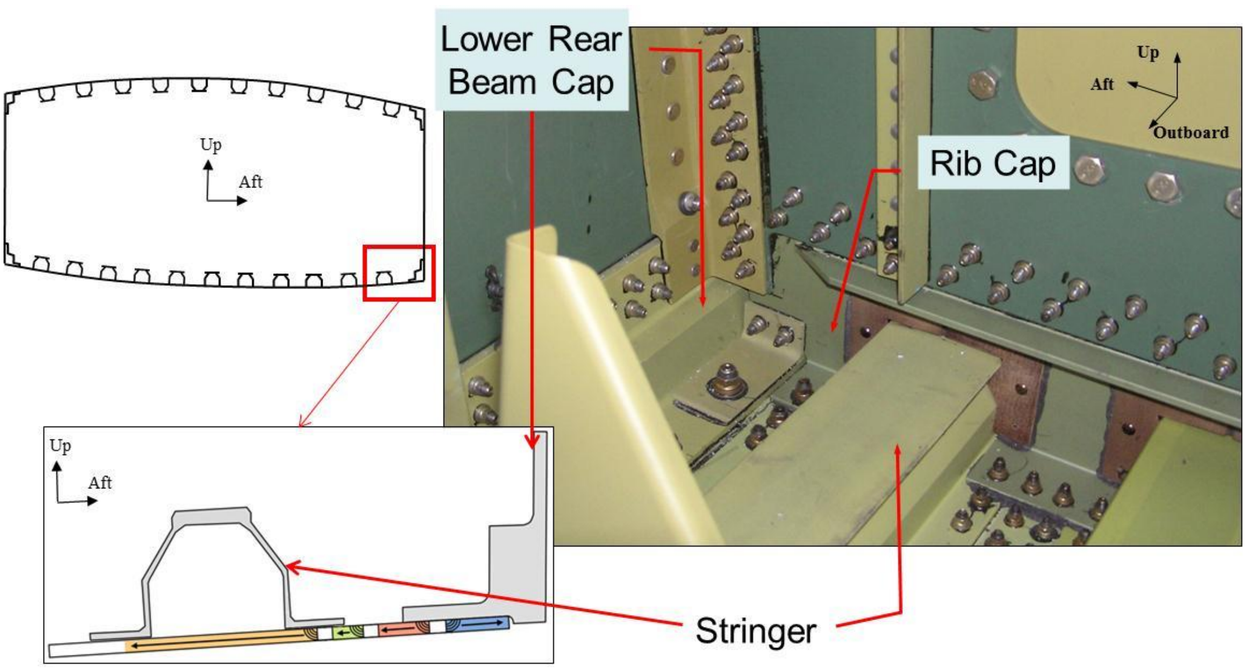


Figure 9.6-24. Local Detail of CW-1 Location of C-130 Lower Wing Skin

9.6.10. Building Block Approach to Simulated Structural Corrosion Testing

Steve Thompson, Mike Spicer, Ed Hermes, Chad Hunter, and Kumar Jata, USAF Research Laboratory – Materials and Manufacturing Directorate; and Nick Jacobs, University of Dayton Research Institute

In response to calls from the Aircraft Structural Integrity Program (ASIP) and structural design communities regarding the aging fleet and serious environmental concerns about the continued use of corrosion protection schemes, the Air Force Research Laboratory's Materials and Manufacturing Directorate (AFRL/RX) has initiated a novel multi-year building block testing program aimed at providing actionable qualitative structural corrosion data for use in airframe risk assessments (Figures 9.6-25 and 9.6-26). A wide range of variables can affect the initiation and propagation of corrosion damage on current operational airframes, including (but not limited to) aircraft design, assembly processes, materials variability, fastener types, corrosion protection schemes (coatings, treatments, etc.), environmental exposure, and loading spectra. Individually, many of these variables can be managed through data collected from coupon-level testing under tightly controlled conditions. However, operational airframes are not manufactured or operated in the same tightly controlled conditions nor are coupon tests always able to capture synergistic effects among the variables. The gap between coupon-level testing and an airframe's operational experience has limited the value and utility of much of the corrosion-related test data generated to date. Currently, ASIP managers do not have sufficient corrosion information required to translate operational and sustainment requirements into usable risk analyses. These issues include qualitative assessments of corrosion prevention methods, models to assess corrosion initiation and progression that can be interfaced to ASIP force management software and provide designers with the ability to accurately estimate the cost of sustainment during design trades, and the ability to assess corrosion in real time through the use of Condition-Based Maintenance (CBM) and better NDI methods. To achieve these goals, a new paradigm for corrosion testing needs to be developed that eliminates any material's bias and provides information on corrosion protection systems as they perform in the actual (or very close to actual) environment (stress, as well as environment). The objective of this new building block approach is to bridge the S&T gap by developing a test specimen (or series of test specimens) that can simulate airframe structural elements and simultaneously subject the specimens to conditions similar to that of an operational airframe (Figure 9.6-27). This testing effort is intended to lead to a protocol for a well-defined and accepted methodology to provide corrosion susceptibility for a range of aerospace materials, structural designs, environments, and corrosion protection schemes that may be used in future USAF and industry risk assessment efforts. This technical activity discusses the program's objectives and plan for testing.

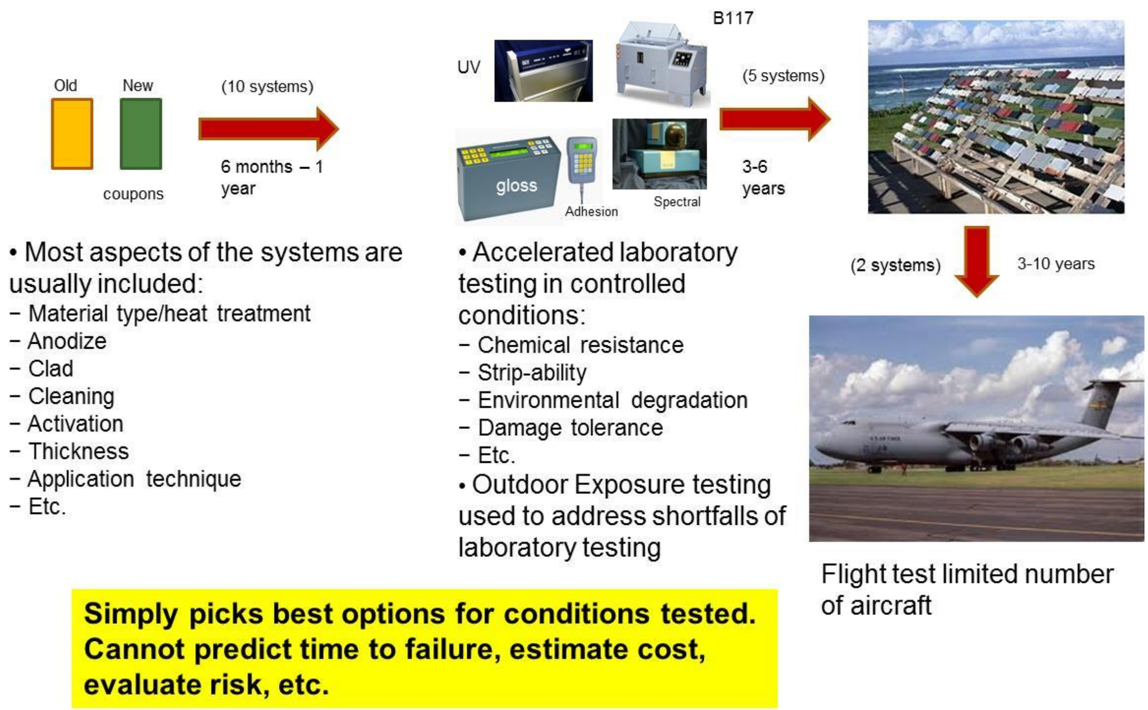


Figure 9.6-25. Design & Material Qualification: Today

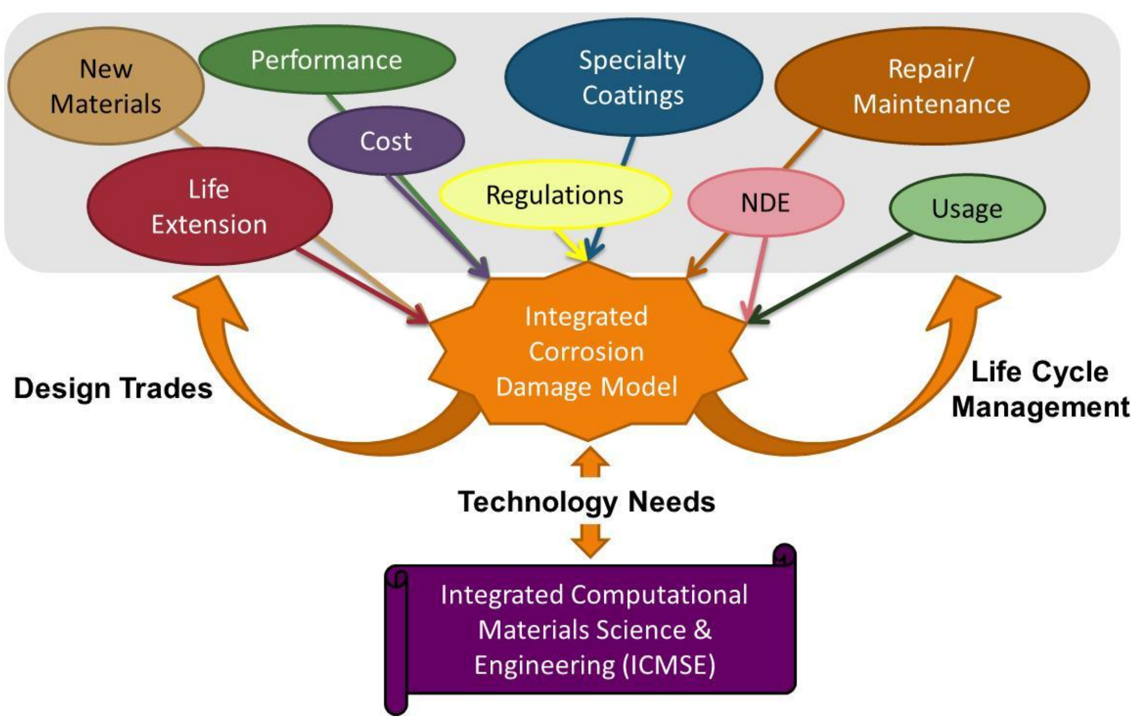


Figure 9.6-26. Design & Material Qualification: Where We Want to Go

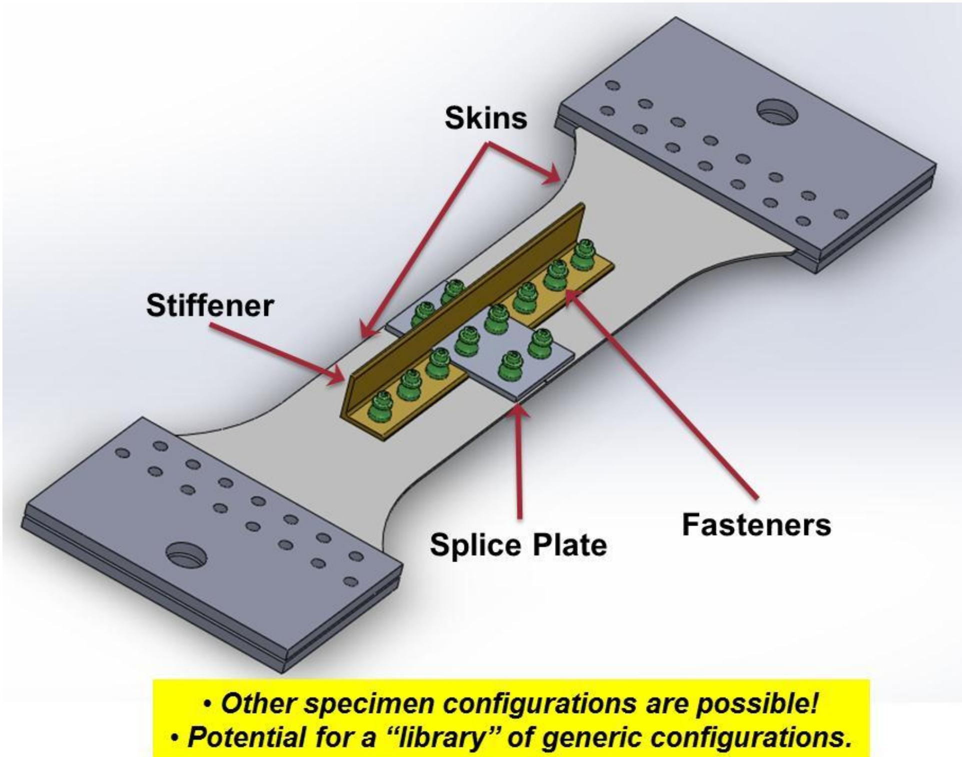


Figure 9.6-27. Test Specimen Design

9.6.11. Calculating Stress Intensity Factors for Countersunk Holes

Jody Cronenberger, Southwest Research Institute

Calculating accurate stress intensity factors (SIFs) for countersunk holes can be challenging. Southwest Research Institute has recently developed a set of SIF solutions that can be used to quickly obtain accurate SIFs for the most common aerospace countersunk hole geometries. The current solution set covers a crack growing from the base or knee of the countersink with remote tension loading (Figure 9.6-28). Crack dimensions range from very small, less than 0.005 inches, to very large with a/c aspect ratios ranging from 0.5 to 4. This technical activity discusses a unique approach used to define the limits to the solutions space (Figure 9.6-29), the finite element methods used to obtain the SIF solutions (Figure 9.6-30), the validation approach used to validate the SIF solutions (Figures 9.6-31 through 9.6-33) and an interpolation process that can be applied to quickly obtain accurate SIF solutions from anywhere within the solution space. Future plans to integrate these solutions into AFGROW and NASGRO are also be discussed

- Four possible crack initiation locations
- Knee or base of CS has the highest stresses for remote tension loading
- Only base or knee of CS is evaluated

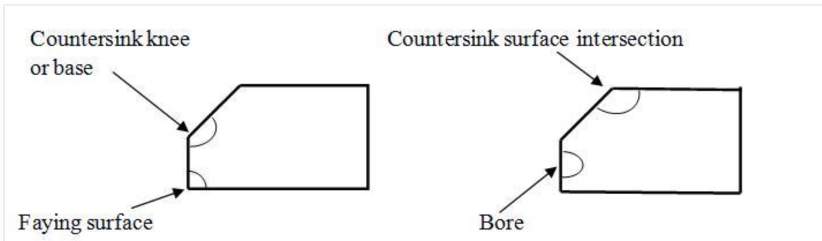
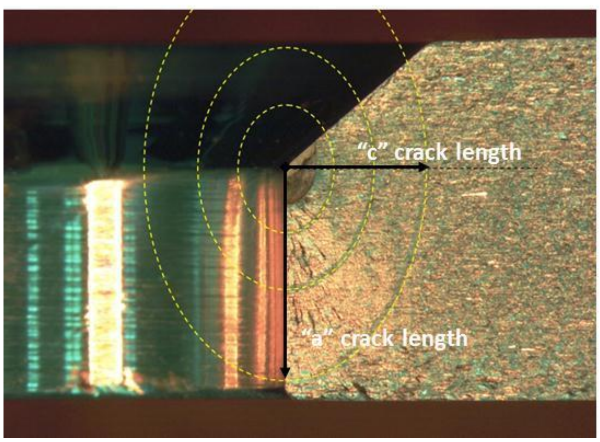


Figure 9.6-28. Solution Space Investigation

- Elliptical crack shape, "a" become a virtual length after through crack transition
- Center of ellipse does not shift as crack grows



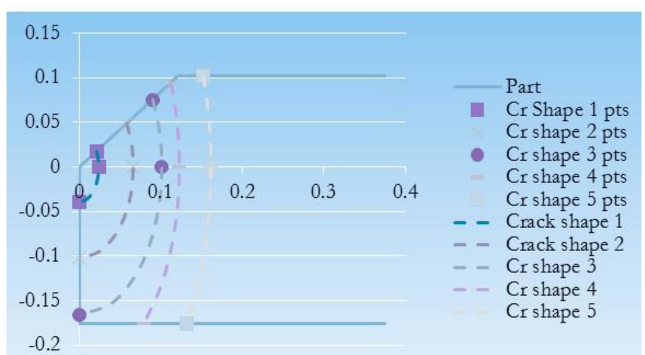


Figure 9.6-29. Elliptical Crack Shape Assumption

- The hp-version of the FEM was implemented
- Half symmetry used to reduce model DOF

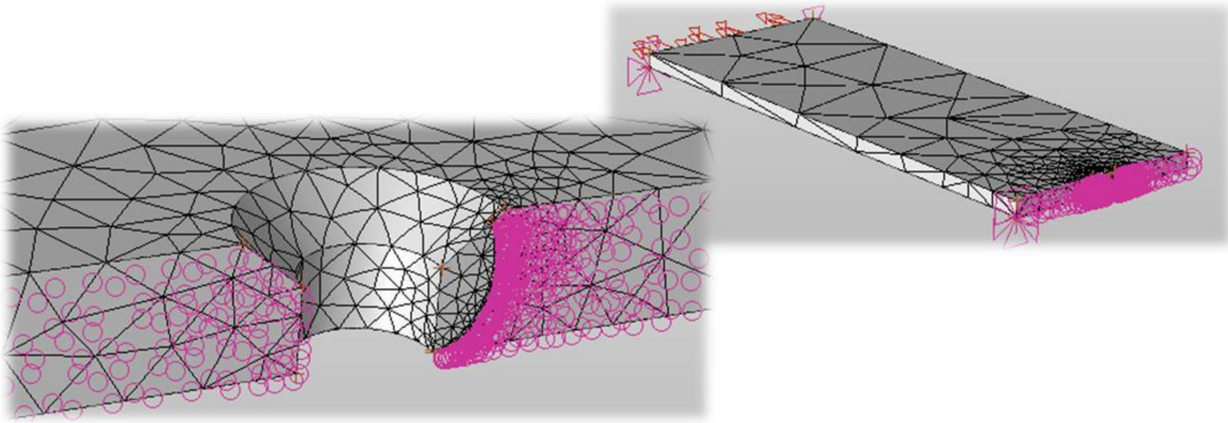


Figure 9.6-30. Computational Methodology

- Countersunk hole marker band test results showed that the ellipse approximation was valid for geometry, material and loading that was tested
- Center of ellipse did not shift as crack grew (short cracks)

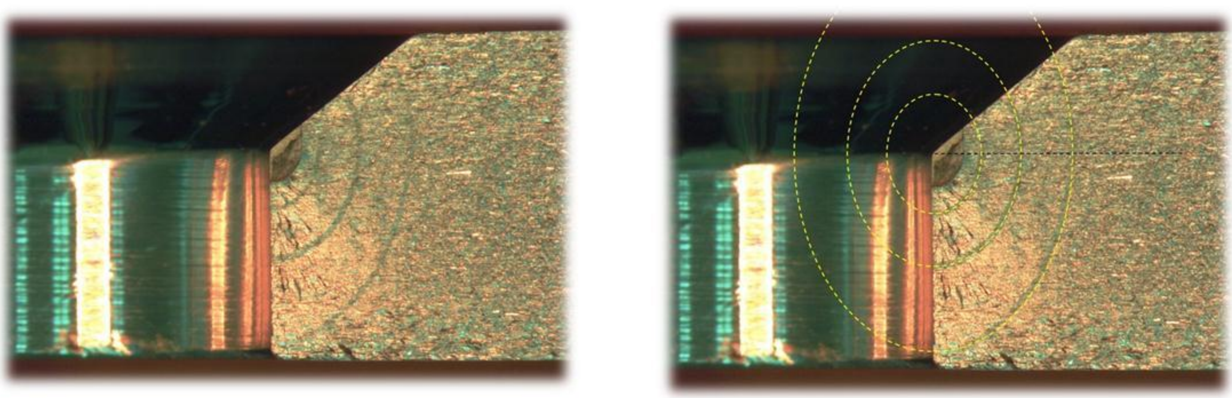
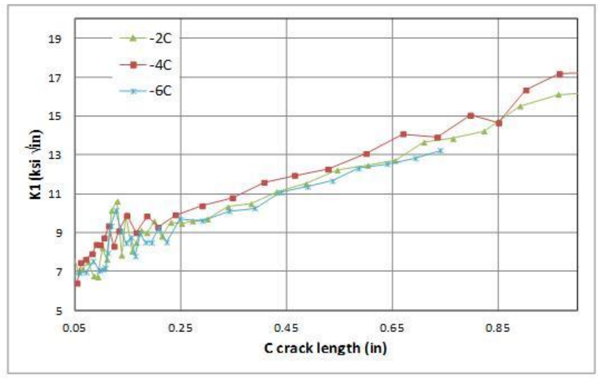


Figure 9.6-31. Elliptical Crack Shape Experimental Validation

- **Countersink hole experimental results**
- **Results show good repeatability**
- **Jitter due to through crack transition**



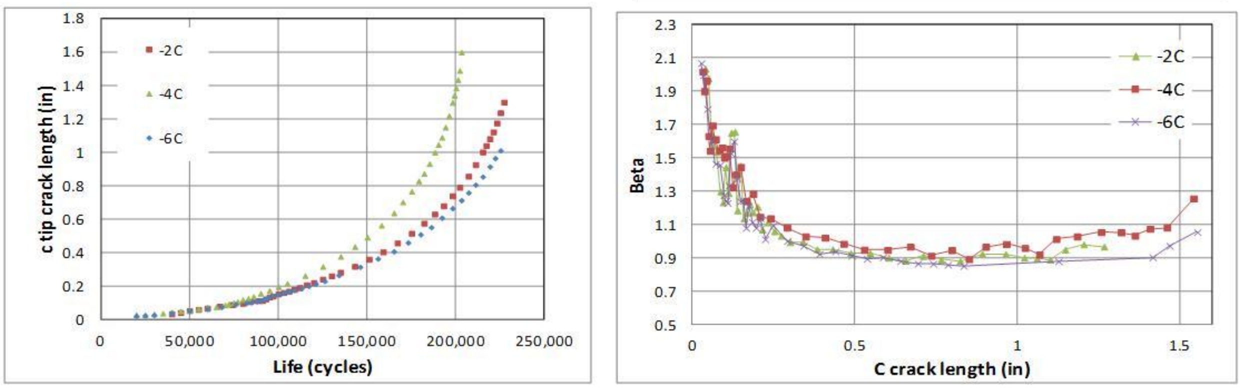


Figure 9.6-32. Experimental Results

- Experimental validation results showed good agreement between experimentally calculated K1 value and FE derived K1 values
- Material properties may be reason for slight offset observed for longer crack lengths

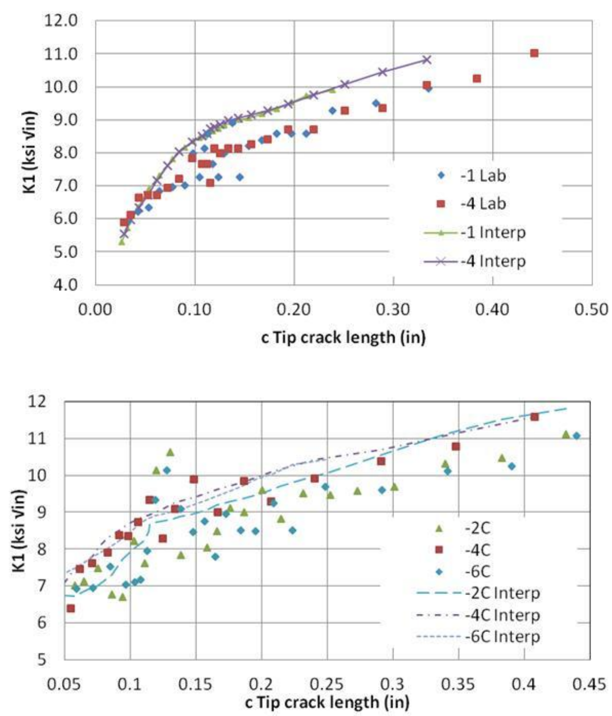


Figure 9.6-33. Analytical/Experimental Correlations

9.6.12. Characterizing a Large Aircraft Forging using a Multi-Party Integrated Product Team

Sandeep Shah, NexOne Inc.; Megan Sheebeck and James Greer, USAF Academy-CAStLE

The aft wing terminal fitting (Figure 9.6-34) of the KC-135 aircraft has experienced in-service stress corrosion cracking and occasionally requires replacement. The replacement parts are initially hand forged then die forged. The forging process is followed by machining. The part specifications require that the grain flow in these forgings conforms to the general shape of the part. Engineering examination of some sampled parts showed potentially "folded" sections of grain flow in certain areas of the forgings (Figure 9.6-35). Since this grain flow apparently did not conform to the original part specification, before using these forgings in the aircraft it was necessary to perform a thorough materials characterization. The present work involved testing specimens extracted from these suspect areas of six forgings (Figures 9.6-36 and 9.6-37) as well as from conforming areas (to act as "control" specimens). Testing was performed for static, fatigue and stress corrosion properties of the discrepant areas of the forging and comparisons made with either published values or results from control specimens to assess whether the suspect areas had degraded material properties. The test effort consisted of 90 tensile tests, 35 fatigue tests, 27 fatigue crack growth rate tests, 13 fracture toughness tests and 21 stress corrosion cracking tests. The focus of this technical activity is on the effort to tailor a testing program to a large forging like the wing terminal fitting to meet general industry testing standards using a multi-party Integrated Product Team.

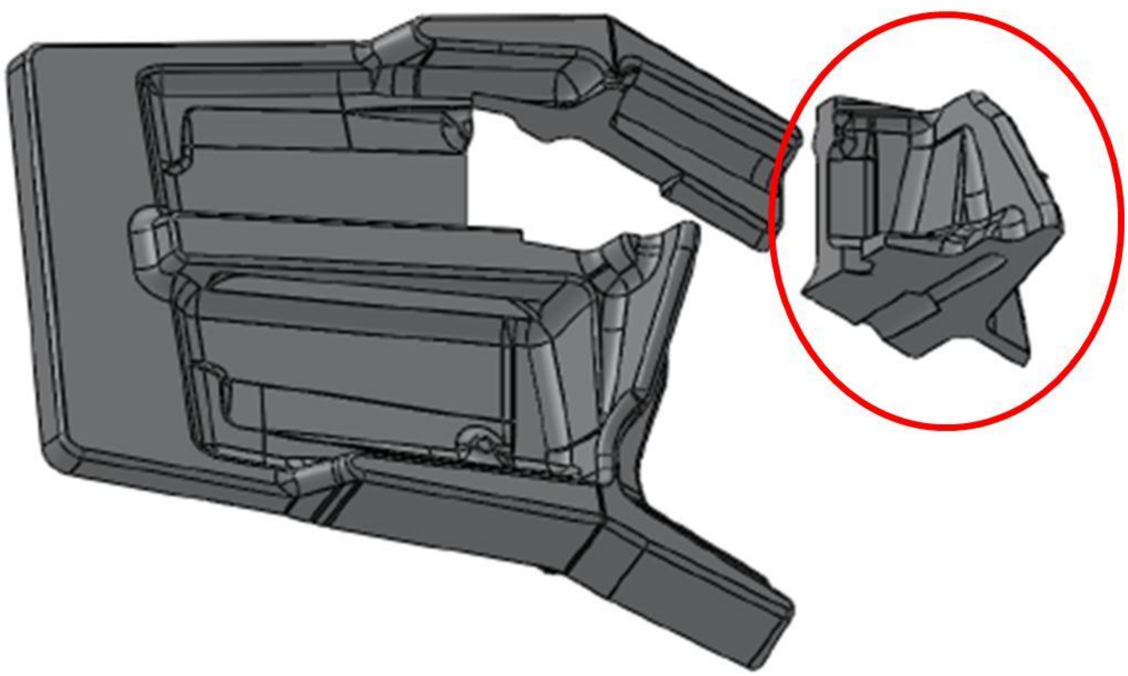


Figure 9.6-34. Primary Area of Interest

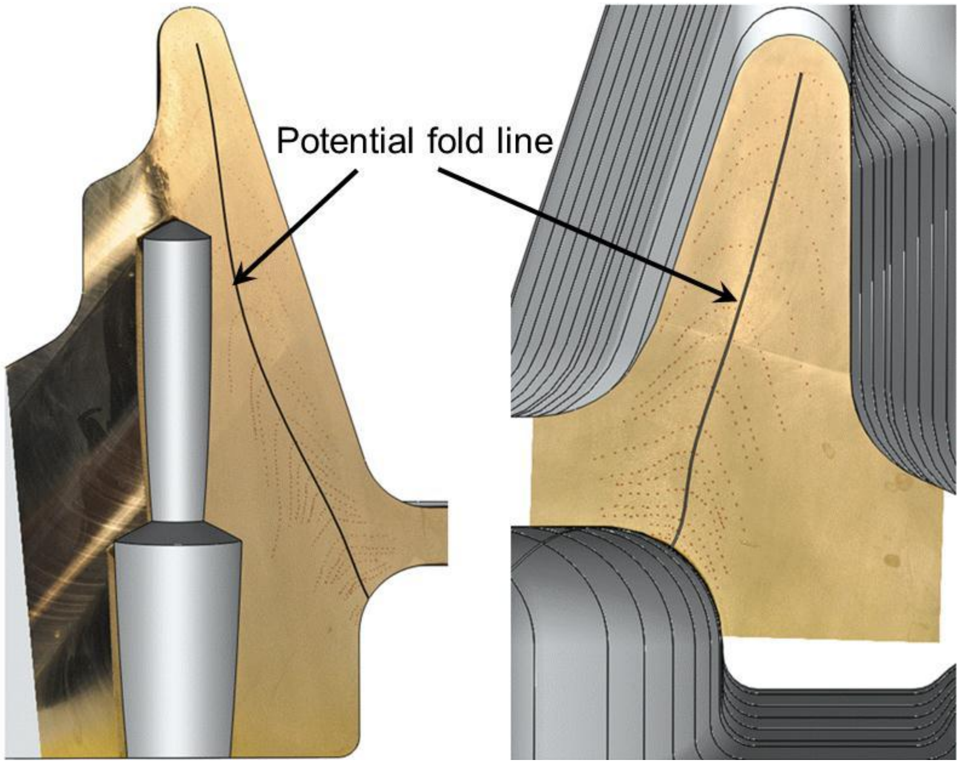


Figure 9.6-35. Discrepant Grain Flow Material

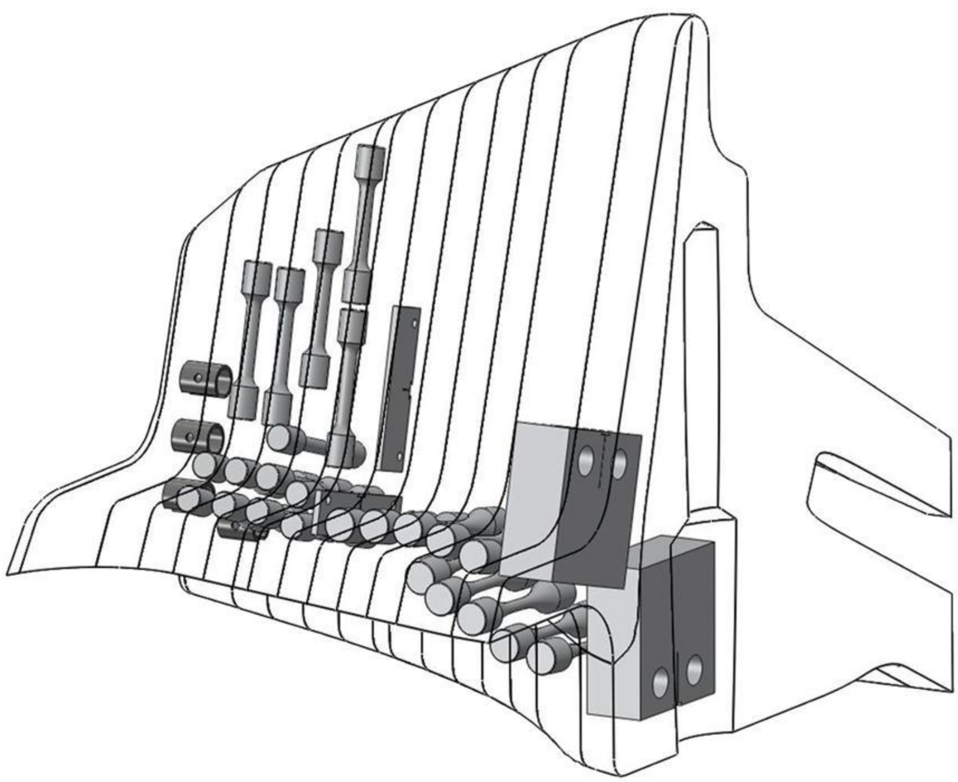
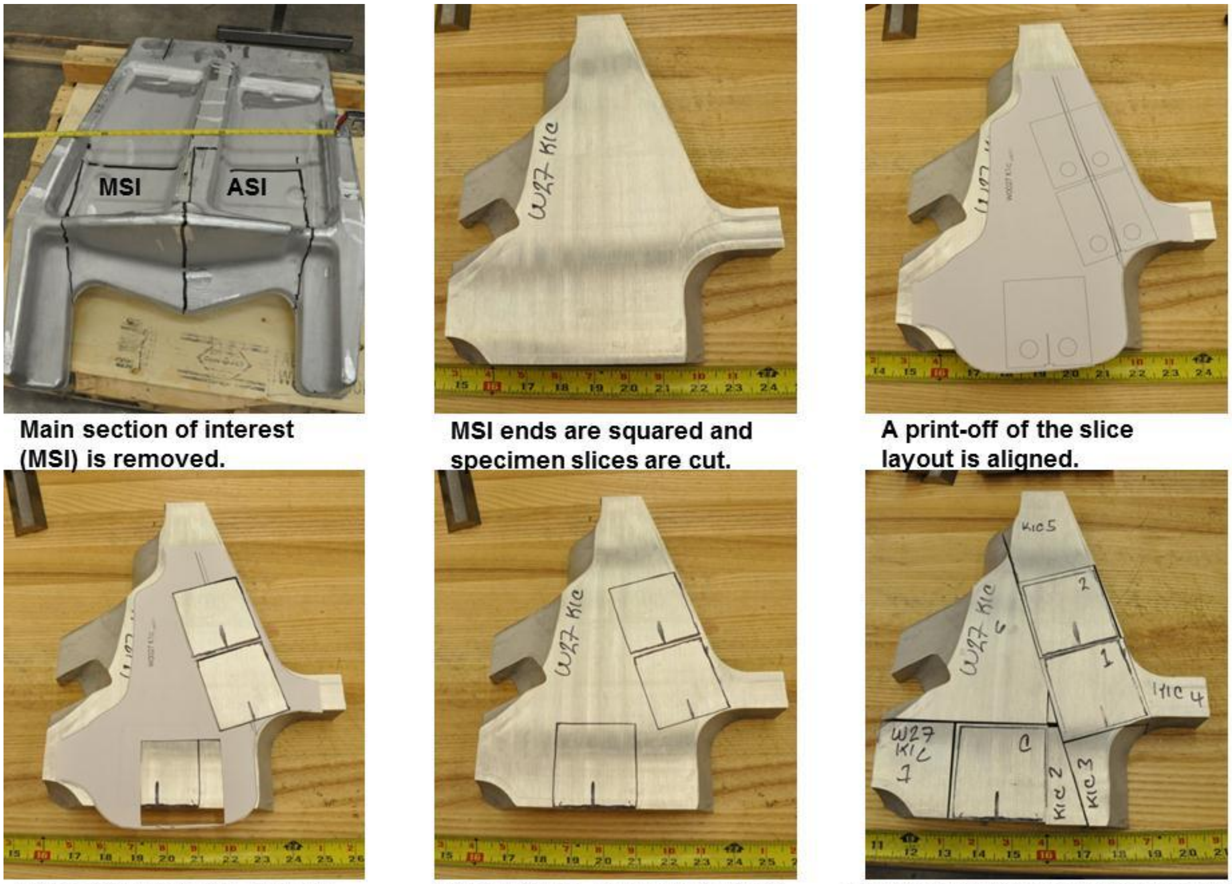


Figure 9.6-36. Combined Specimen Layout



Main section of interest (MSI) is removed.

MSI ends are squared and specimen slices are cut.

A print-off of the slice layout is aligned.

The specimens are cut out from the print-off.

The cut-outs are used to trace the specimens onto the slice.

The specimens are rough cut and labeled before final machining.

Figure 9.6-37. Typical Steps in Machining

9.6.13. Metallic Materials Properties Development and Standardization (MMPDS)

Jim Kabbara, Mark Fiesthler, Ian Y. Won, and John G. Bakuckas, Jr., FAA; Jana Rubadue, Battelle Memorial Labs

The Metallic Materials Properties Development and Standardization (MMPDS) is an effort led by the Federal Aviation Administration (FAA) to continue the Handbook process entitled “Metallic Materials and Elements for Aerospace Vehicle Structures,” (MIL-HDBK-5). The Handbook is recognized worldwide as the most reliable source for verified design allowables needed for metallic materials, fasteners, and joints used in the design and maintenance of aircraft and space vehicles. Consistent and reliable methods are used to collect, analyze, and present statistically-based aircraft and aerospace material and fastener properties.

The objective of the MMPDS is to maintain and improve the standardized process for establishing statistically-based allowables that comply with the regulations, which is consistent with the MIL-HDBK-5 heritage, by obtaining more equitable and sustainable funding sources. This includes support from government agencies in the Government Steering Group (GSG), from industry stakeholders in the Industry Steering Group (ISG) and from profits selling the Handbook and derivative products. Towards this goal, the commercial version of the MMPDS-07 was released April 2012 (Figure 9.6-38).

There has been a substantial upgrade to the Handbook with the addition of eight new metallic materials including three aluminum-lithium alloys.

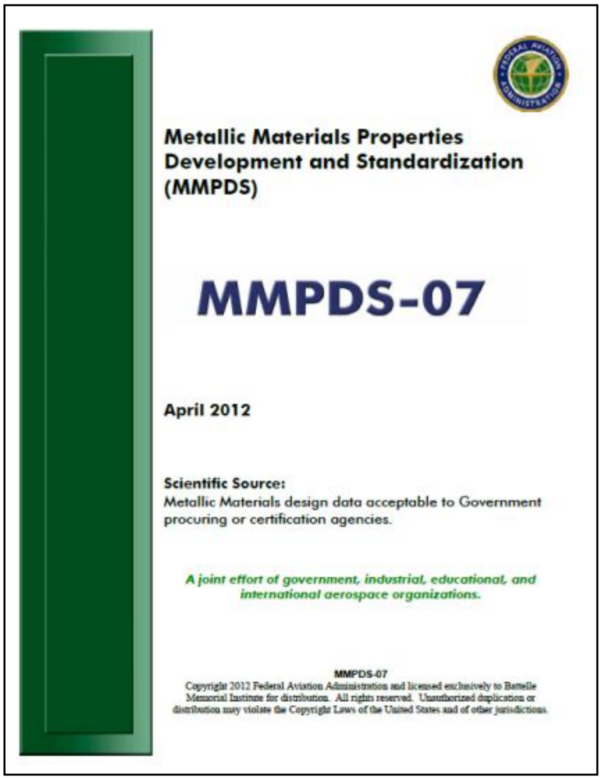


Figure 9.6-38. Cover of MMPDS-07

9.6.14. Surface Oxygenation Effects on Titanium Fatigue Strength

Mark Ofsthun, Spirit AeroSystems, Inc.

When exposed to high temperatures (> 600°F) for long periods of time (thousands of hours), titanium is susceptible to a condition referred to as Surface Oxygenation Effects (SOE). Figure 9.6-39 is an example of SOE on a titanium sheet. SOE layers are very brittle and reduce the overall ductility of the sheet metal which in turn, will reduce the overall fatigue performance of the material. In today's ultra-high performance jet engines, the long term effects of SOE cannot be ignored.

Figure 9.6-39. SOE on Titanium Sheet

Determining the effect of SOE on thin titanium is a challenge. SOE takes many hours at high temperature and when evaluating thin titanium, this process results in warping of the sheet metal. Trying to test warped specimens results in confusion where the data is a combination of SOE effects and specimen warping effects. In addition, the typical specimen has to be machined after exposure which removes the edge SOE which could be unconservative in the evaluation of the SOE's effect on the material. Spirit has developed a unique way of exposing titanium to SOE which eliminated the warping of the sheet. This innovative method involves using open hole fatigue specimens. The key is to drill the holes and perform all the hole preparation in the sheet first, as shown in the sketch in Figure 9.6-40. Then, the sheet with the holes is placed between two steel plates with areas cut out from them to allow the surfaces in the test specimen to be exposed (top and bottom). Figure 9.6-41 shows the SOE exposure set-up. After the time at temperature is complete, the sheet will remain flat and allows the individual specimens to be cut from the sheet.

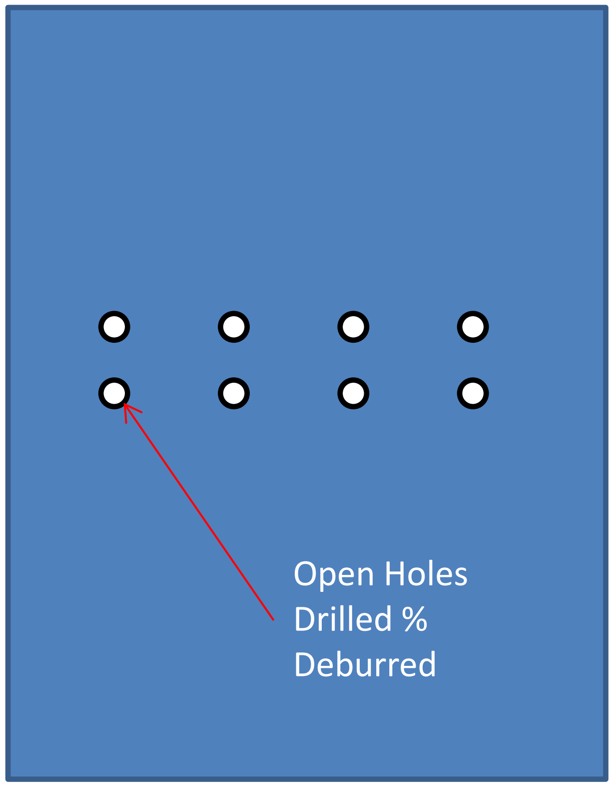


Figure 9.6-40. Sheet Before Exposure to SOE

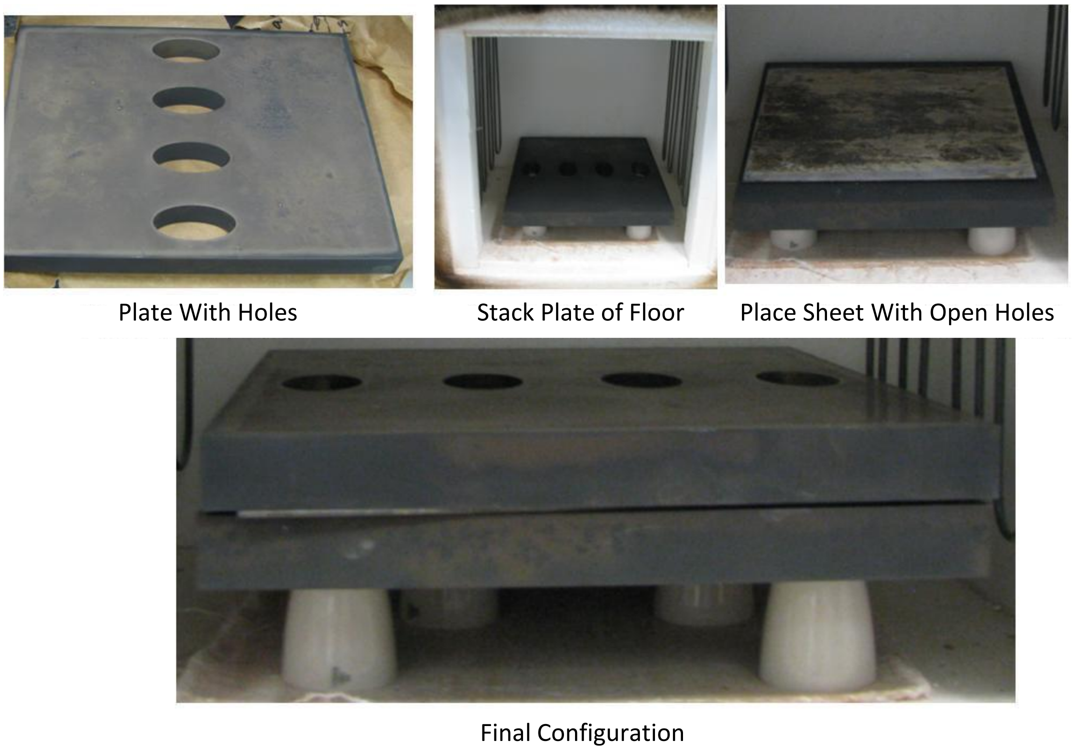


Figure 9.6-41. SOE Exposure Method

Figure 9.6-42 shows the significant effects SOE can have on fatigue. The SOE is very brittle and the brittle nature of the material leads to significant reduction in fatigue performance of the material.

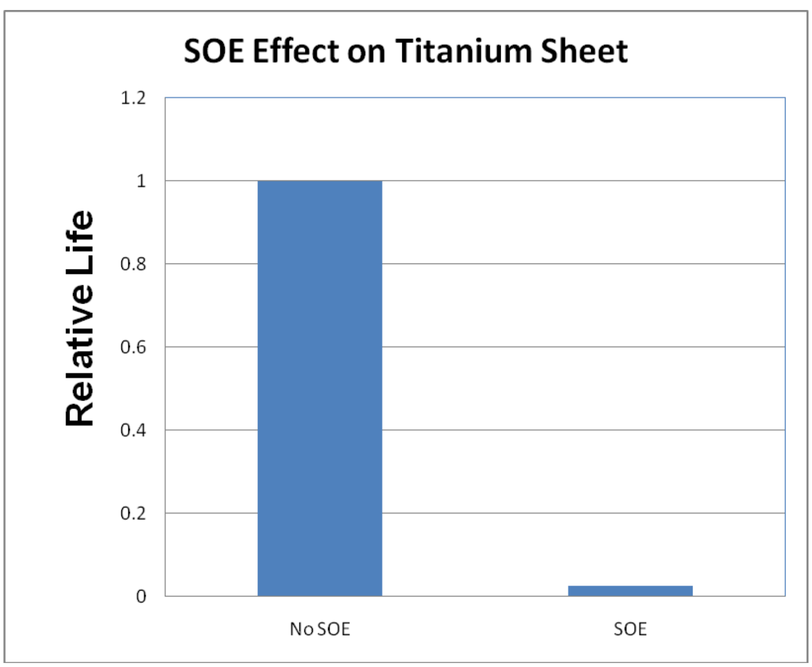


Figure 9.6-42. SOE Fatigue Results

In conclusion, when designing aircraft structure, it is imperative that service conditions are simulated as realistically as possible. Titanium materials exposed to high temperatures for long periods of time will lose fatigue performance from SOE, and this condition must be considered in the fatigue evaluation of titanium structure.

9.6.15. Drill Start Effects on Aluminum Fatigue Strength

Mark Ofsthun, Spirit AeroSystems, Inc.

The fabrication of a commercial airplane involves the drilling of hundreds of thousands of holes. In the drilling operations there are many opportunities for shop errors to occur. Drill starts are defects that arise when the mechanic starts drilling in the wrong spot or the result of using too long of a drill bit resulting in underlying structure being subject to a drill start (See Figure 9.6-43 for a typical drill start). Generally, when drill starts are detected, they are blended away and the analysis of the low K_t with the increased stresses from the blend-out is performed. However, in the case of a drill start resulting from the long drill bit or in the case of a drill start that is not detected until after it cannot be practically repaired, there is a need for data to assist in the analysis of a drill start. Analysts normally assume the drill start is conservatively approximated by an open hole analysis. Spirit undertook the task of running fatigue tests of various depths of drill starts in a 7XXX sheet material in order to substantiate this assumption.

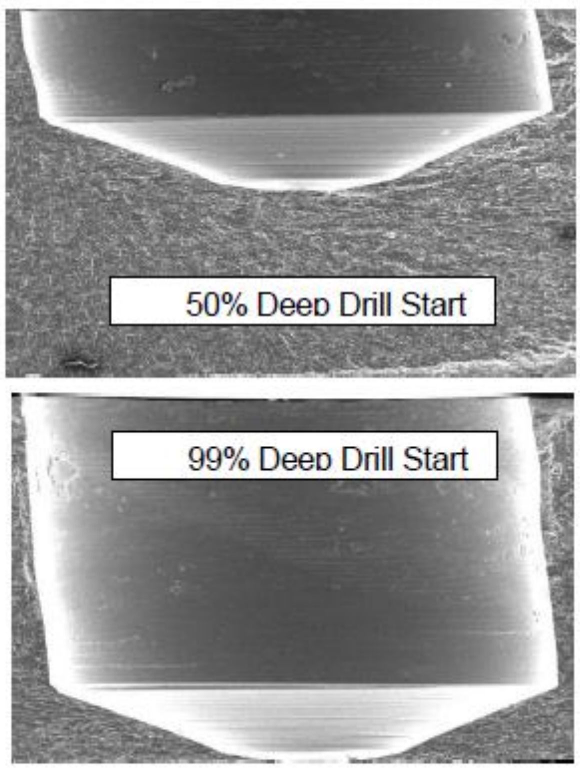


Figure 9.6-43. Typical Drill Start

The fatigue specimen was a flat rectangular sheet with 2 drill starts in the center of the test specimen as shown in Figure 9.6-44. Drill depths equal to 10%, 25%, 50%, 75%, 99% and 100% (baseline open holes) of the sheet thicknesses were tested. All specimens were taken from the same sheet and fatigue tested with the same gross area stress. The results are shown in Figure 9.6-45 which shows that the drill start is better than the open hole until the drill start reaches about 75% through the thickness of the specimen. Drill steps that are deeper than 75% actually performed worse than the open hole in fatigue. It is likely the drill start had an additional stress riser consisting of the hole and the material beneath the drill as shown in Figure 9.6-43 (99% Depth Drill Start), which may be acting more like a burr. The cracks nucleated at the bottom of the drill start.

Figure 9.6-44. Drill Start Test Specimen

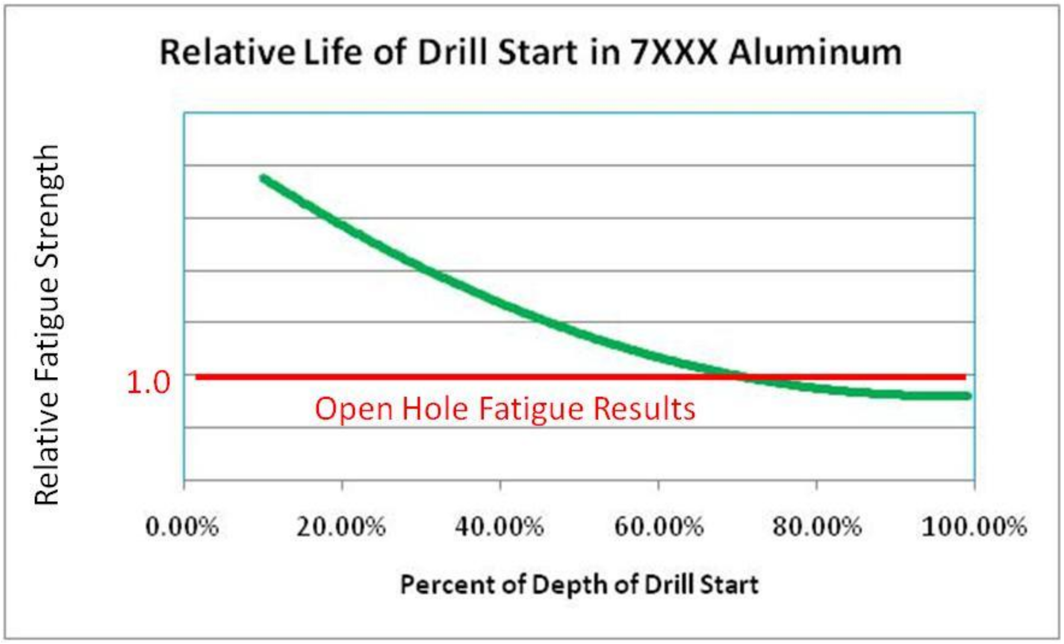


Figure 9.6-45. Drill Start Fatigue Results

In summary, drill starts may be conservatively analyzed as an open hole provided that the drill start is well below 50% of the part thickness. However, this assumption is not conservative if the drill start exceeds 75% of the part thickness. The likely interaction of the two K_t values underscores the need for test data validating assumptions made in fatigue assessments.

9.6.16. Fay Seal Effects on Aluminum Joints

Mark Ofsthun, Spirit AeroSystems, Inc.

Corrosion is a major design consideration on commercial airplanes. One important design feature used in the fabrication of commercial airplanes is the use of faying surface sealant to prevent moisture ingression into the joint. Faying surface sealant, however, can affect the fatigue performance of joints. Test experience has shown that faying surface sealant can increase fatigue performance, decrease fatigue performance or have no affect on fatigue performance. The factors determining the effect of faying surface sealant are hole fill, load level, load transfer and clamp up. In this study, low and high load transfer joints with lockbolts and aluminum rivets were used to evaluate the effects of fay seal on aluminum joints. One key feature regarding lockbolts and rivets is that they are much less likely to relax due to fay seal squeeze out unlike torqued fasteners (bolts with nuts).

Bolted Joints (Figure 9.6-46)

Fay seal can act as a sponge and effectively reduce the clamp-up and the fasteners may relax, thus possibly resulting in a significant degradation in fatigue performance of the joint as shown in Figure 9.6-46. Figure 9.6-46 is a high load transfer joint with low interference and low clearance. Figure 9.6-46 also shows that better hole fill will at least partially offset the effect of fay seal. In high load transfer joints, clamp up is important and losing clamp up from faying surface sealant will result in a loss of fatigue performance.

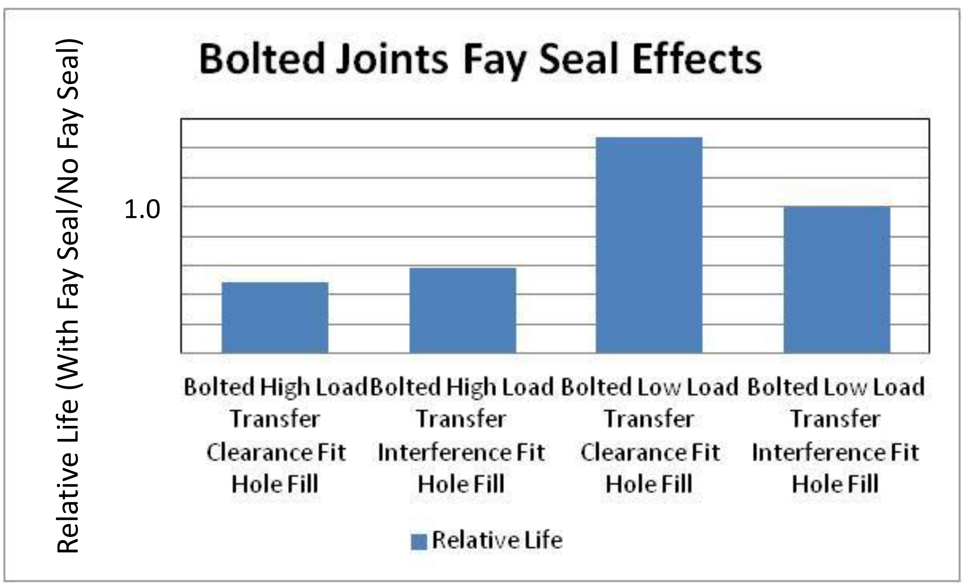


Figure 9.6-46. Fay Seal Effects on Bolted Joints

Also included in Figure 9.6-46 are low load transfer joints where fay seal joints did not actually degrade the fatigue performance. In low load transfer joints the effect of fay seal should be less and fastener clamp up effects are less.

Riveted Joints (Figure 9.6-47)

In riveted joints, there appeared to be a significant improvement in fatigue for load transfer joints with fay seal. The fatigue benefit for the riveted joint is likely due to the fay seal helping to reduce the bearing stress along with exceptional hole fill for which rivets are known. However the riveted low load transfer were not as affected except in the thick joints (joint thickness greater than the fastener diameter) where fay seal actually degraded the fatigue performance.

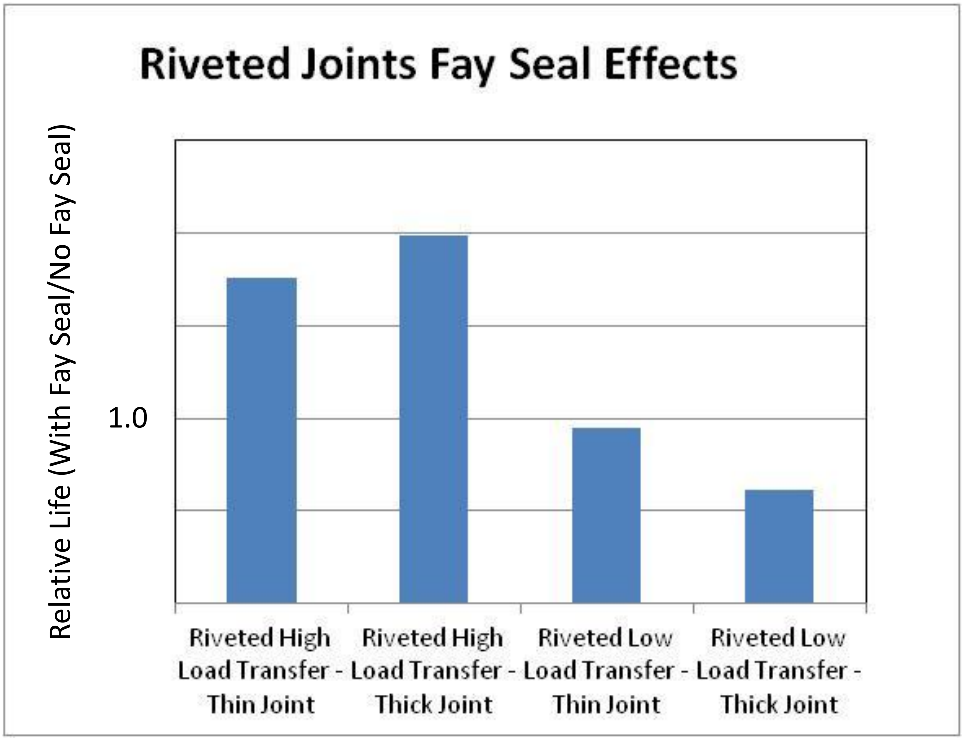


Figure 9.6-47. Fay Seal Effects on Riveted Joints

In summary, there is a need to empirically evaluate joints for the effect of faying surface sealant. The factors in determining the effect of faying surface sealant are complex with many variables. The most important factor however is the amount of load transfer and how much the fastener can relax as a result of the sealant squeezing out as the fasteners are installed.

9.6.17. Damage Tolerance Material Properties Validation

Christopher Mazur, Spirit AeroSystems, Inc.

Damage tolerance material properties are not "Allowables" similar to static strength properties. An "Allowable" is a term which implies the properties are based on sufficient test data to satisfy some statistical criteria. Federal Aviation Administration (FAA) Advisory Circular (AC) 25.571-1D, Section 5d indicates that typical (or average) material properties may be used for damage tolerance analysis of transport category aircraft. Therefore, damage tolerance test results may be published as the average value of the available public domain and/or internal company data. Most damage tolerance material properties available today in industry databases and software are based on a compilation of available literature test data. But do these databases represent present day materials used to manufacture aircraft and are the database properties based on 50-year old test data consistent with today's standards for engineering analysis?

Recent experience using industry damage tolerance properties did not meet Spirit AeroSystem's expectations. As one example, erroneous analysis results occurred where thick stock gauge aluminum plate material crack growth and toughness characteristics were superior to thin stock gauge aluminum plate material characteristics. As a result of this issue and others, a more detailed evaluation of the data within the databases was conducted by Spirit to assess the quality and applicability of the data and resultant damage tolerance properties.

A fracture toughness plot for 2024-T351 plate (L-T orientation) illustrates typical industry data supporting a damage tolerance property. Figure 9.6-48 is a plot of fracture toughness data as a function of specimen thickness along with a line representing the resultant material property. Detailed investigation of the data indicates that the data are from five literature sources published between 1966 and 1985. Middle tension (MT) panels of different widths and compact tension (CT) specimens were used to produce these data, but some specimens are not sufficiently wide to produce plane stress fracture toughness properties. These trends can be seen in Figure 9.6-48, where the wider MT panels produced much higher fracture toughness values. Spirit considers the final material property line to be conservative and severely influenced by specimen geometry.

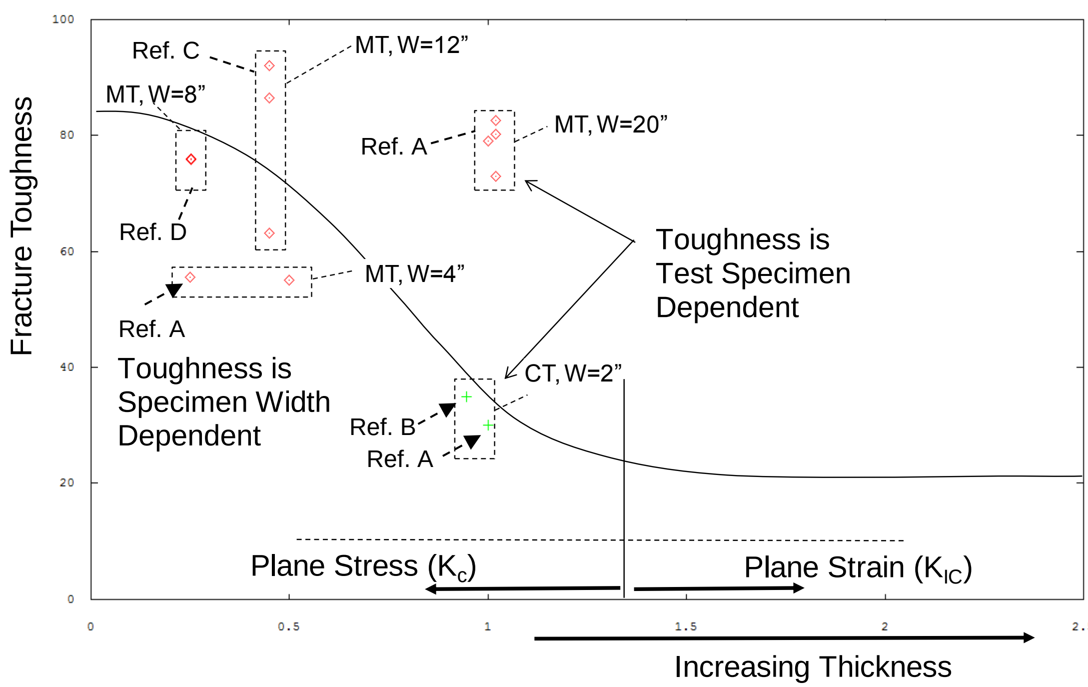


Figure 9.6-48. 2024-T351 Plate Fracture Toughness Data (L-T)

If the available damage tolerance data supporting a material property is from 1966, are those data applicable for today's material? Many changes have occurred within the aircraft metallic materials production industry including consolidation of suppliers, worldwide entry of new material suppliers, and new and/or revised material specifications. While material specifications are intended to ensure the suppliers have controls in place to produce a stable and consistent product, they do not ensure consistent damage tolerance material characteristics between producers. Having tested 7050-T7451, 15-5PH, and Ti 6Al-4V materials in the early 1990s and again in 2012, the variations in damage tolerance properties found between qualified material producers can far outweigh all other variables.

Based on assessments of the industry damage tolerance databases, Spirit concluded that it would be appropriate to use validation testing in addition to the industry damage tolerance database and material properties on commercial aircraft structure. Spirit completed a test program in 2012 to validate existing industry material properties which are highlighted in Table 9.6-3. General conclusions from the Spirit damage tolerance testing relative to the industry databases are that the industry published data tends to be conservative.

Table 9.6-3. Damage Tolerance Testing for Material Properties Validation

	Material	Specification	Gauge (in) (mm)	Supplier	Crack Growth Rate Specimen Counts	Fracture Toughness Specimen Counts
Aluminums	2024-T851 Plate	AMS-QQ-A-250/4	0.5 - 1.25 12.7 - 31.7	5	43	18
	2124-T851 Plate	AMS-QQ-A-250/29	1.25 - 6.0 1.25 - 152.4	4	49	34
	2219-T851 Plate	AMS-QQ-A-250/30	4.0 - 6.0 101.6 - 152.4	3	23	21
	7050-T7451 Plate	AMS 4050	6.0 - 8.0 152.4 - 203.2	3	30	31
Titaniums	Ti 6Al 4V Mil Annealed Plate	AMS 4911	2.0 - 4.0 50.8 - 101.6	4	29	36
	Ti 6Al 4V Mill Annealed Die Forging	AMS 4928	8.0 203.4	4	28	26
	Ti 6Al 4V Mill Annealed Forged Block	Spirit Spec.	8.0 - 10.0 203.4 - 254	3	40	35
Steels	15-5PH Solution Treated Bar (H1025)	AMS 5659	6.0 152.4	4	16	10
	15-5PH Solution Treated Die Forging (H1025)	AMS 5659	6.0 152.4	4	24	31
	15-5PH Solution Treated Forged Block (H1025)	AMS 5659	8.0 203.4	3	16	24
	15-5PH Solution Treated Plate (H1025)	AMS 5862	0.5 - 1.0 12.7 - 25.4	4	21	34
Nickel Alloy	Inconel 718 Bar (Ftu = 200 Ksi)	AMS 5663M	2.0 - 3.0 (Dia.) 50.8 - 76.2 (Dia.)	4	12	6
	Inconel 718 Bar (Ftu = 220 Ksi)	AMS 5962	1.5 (Dia.) 38.1 (Dia.)	3	22	17

9.6.18. Fatigue Study of Holes in the Radius of a Step

David Whitley, Spirit AeroSystems, Inc.

During fabrication of certain aerostructure components, a situation can arise where a hole is mislocated so that it interferes with a radius, chem-mill step, machined step, or some other similar detail. Manufacturing may use a radius filler, radius block, or spot-face to provide for proper installation of a fastener in the hole. However, many times due to access restrictions as well as other factors, it is not possible to complete these types of repairs in order to alleviate the interference and resulting stress concentration factor interaction between the hole and radius.

This research analyzes the K_t interaction that occurs between a hole and radius when common repair measures are not possible. Most common stress concentrations such as a hole in a plate are relatively well known. However, the interaction of stress concentration effects between a hole and step are not well known and the resulting impact on fatigue performance is difficult to predict. There exists a need for fatigue data that can be used to determine the analysis methods for evaluation of the interaction of machined steps and fasteners.

For the joint testing, two different specimen types were analyzed with local geometry step ups, allowing the engineer to adjust the location of the step up. Specimens were fabricated out of 7075-T7351 aluminum and typical 6/32" Ti hi-lite bolts were used in the assemblies. An example of a pre-test joint specimen is displayed in Figure 9.6-49. Additional testing was conducted to determine if radius filler for the bolts on the radius would be an appropriate repair. A stress corrosion specimen was also studied.

Figure 9.6-49. Pre-Test Joint Fatigue Test Specimen

Open hole fatigue specimens were tested to investigate any degradation in fatigue life due to a hole being placed at, or in close proximity to a nearby radius. It was determined whether or not a hole very near a radius but not necessarily interfering with the radius has any negative impact on fatigue life that would be associated with K_t interaction. The holes were placed at a series of distances away from the radii in order to properly analyze these effects. 2024-T351 plate material was used for all the open hole test coupons. All hole diameters were equal to 3/16". A picture of a pre-test open hole coupon with the hole centered directly on a tangent radius is provided in Figure 9.6-50.

Figure 9.6-50. Pre-Test Open Hole Fatigue Test Specimen

Results of the fatigue testing are provided in Figures 9.6-51 and 9.6-52. The charts show that the derived K_t modification factors increase as the distance between the hole centers and radius tangents decrease. This relationship indicates a stress concentration factor interaction between the holes and radii. The use of the radius fillers in the joint testing eliminated any K_t interaction between the fasteners and radii, returning the specimens to a baseline equivalent condition.

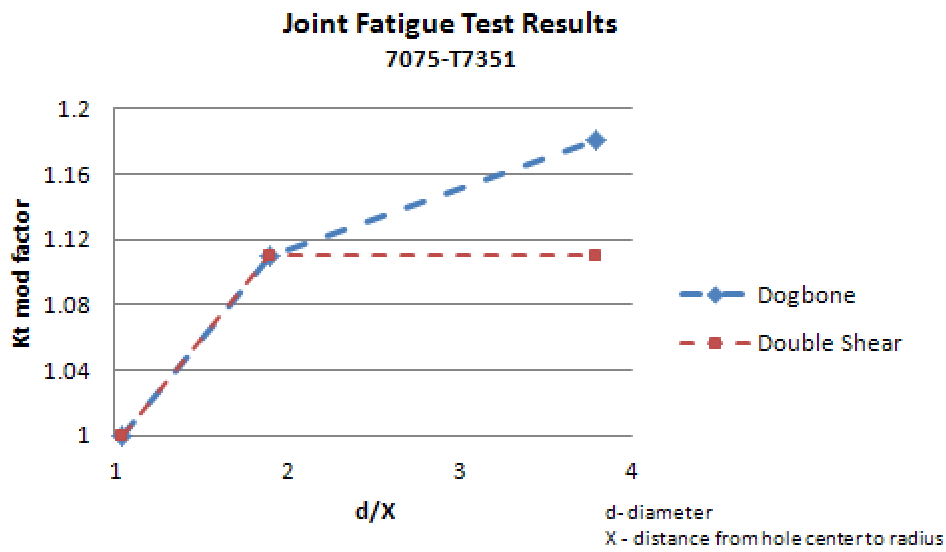


Figure 9.6-51. Fatigue Test Results for Joint Specimens

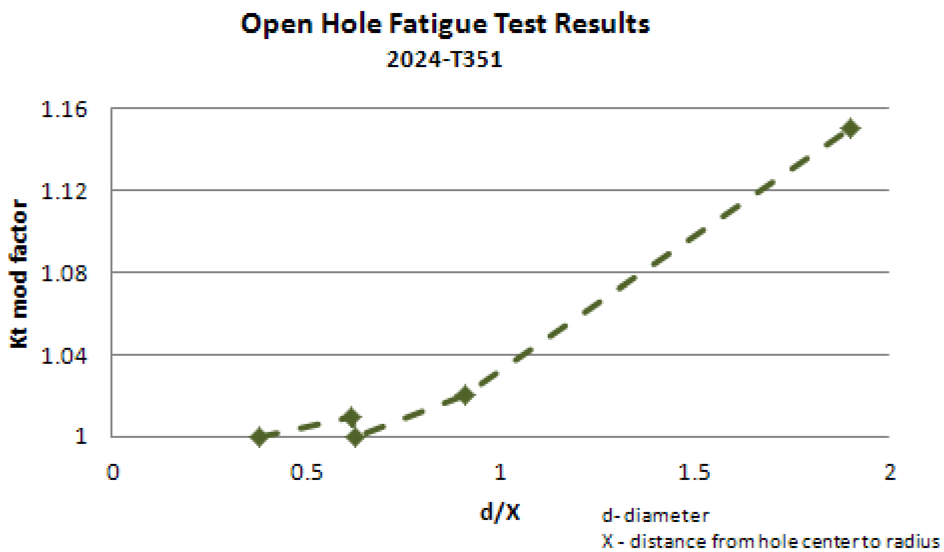


Figure 9.6-52. Fatigue Test Results for Open Hole Specimens

Typical repair measures like radius fillers, radius blocks, or spot-facing are advised for these situations. However, the modification factors and fatigue analysis methods developed here are recommended for use in analysis when common repair methods are not possible.

9.6.19. Long Term Heat Exposure Effects on 15-5PH CRES Steel Fracture Toughness

Jonathan Mowrey, Spirit AeroSystems, Inc.

Materials on aircraft structures are subjected to fluctuations in temperatures over the service life of the structure. Long term exposure to high temperatures can alter material properties just like a heat treat would. Many structures around aircraft engines need high strength along with the ability to endure temperature exposure. This investigation examined fracture toughness properties of 15-5PH CRES steel after exposure to prolonged elevated temperatures.

This evaluation included static tension tests per ASTM E8 and compact tension K_{IC} tests per ASTM E399. Specimens were exposed to a temperature of either 260 degrees Celsius or 426 degrees Celsius for a duration of 250 hours or 500 hours. Two specimen orientations (L-T and S-L) were tested.

Results of the static testing can be seen in Figure 9.6-53. From this figure it can be seen that temperature exposure at 426˚C results in an increase of the material static strength, while material exposed to the lower 260˚C sees minimal changes in relative ultimate strength for 15-5PH CRES steel.

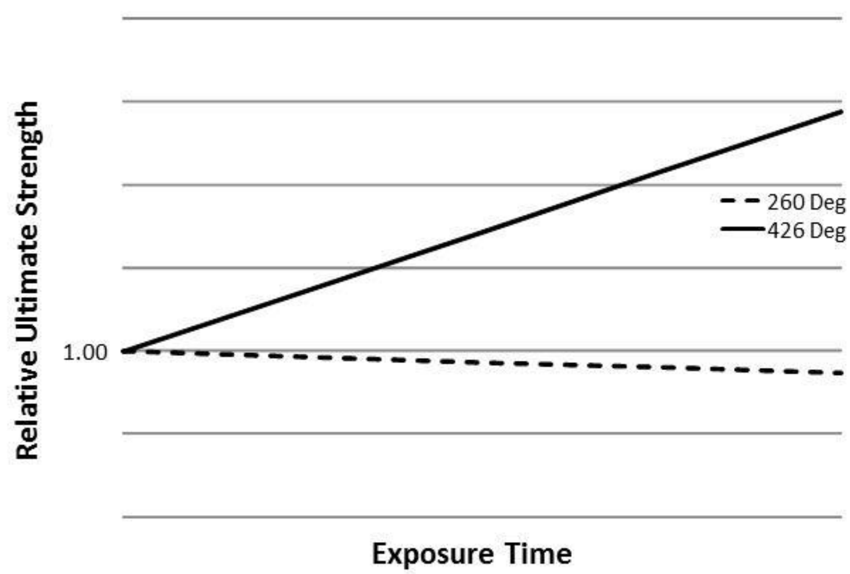


Figure 9.6-53. Static Tensile Ultimate

Plane strain fracture toughness results can be seen in Figure 9.6-54. Figure 9.6-54 shows a large degradation in fracture toughness results after high temperature exposure (426˚C). But for 260˚C exposure specimens, there was minimal loss in fracture toughness in 15-5PH CRES steel. Figure 9.6-54 also shows that fracture toughness reduction due to temperature exposure was much larger in S-L specimens than in L-T specimens.

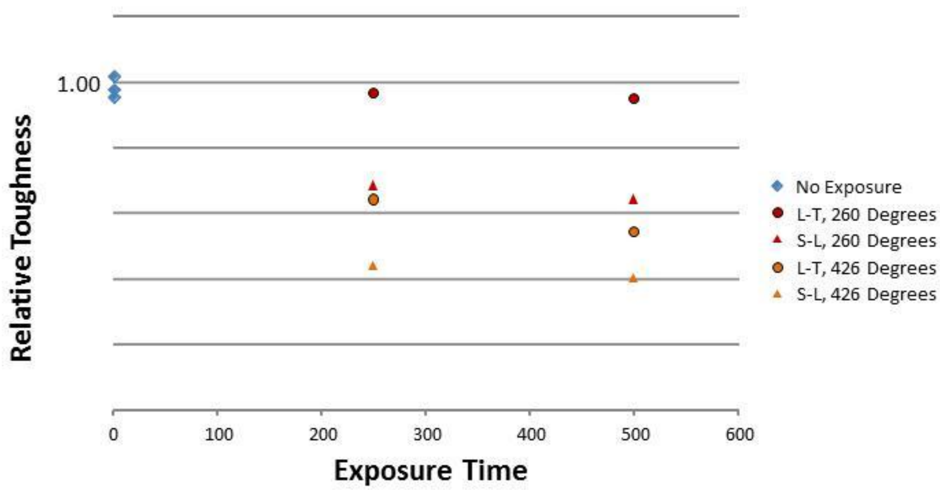


Figure 9.6-54. Relative Toughness Results

9.6.20. Fatigue Effects of Dents in Thin Aluminum

Jonathan Mowrey, Spirit AeroSystems, Inc.

Mechanical damage inflicted on aircraft can be an everyday occurrence. Damage can come from the environment in the form of corrosion and bird strikes; or it can come from accidental handling in the form of dents and scratches. In any event, damage inevitably affects the durability of the material. Inflicted dents were investigated to determine if an accurate method could be developed to predict their effect on fatigue performance of 2XXX and 7XXX series aluminum.

Various dent sizes and shapes were evaluated to develop an overall model for typical damages inflicted on aircraft. The specimen used for testing was an enlarged low K_{tn} specimen with enough room in the middle and edges to simulate dents as shown in Figure 9.6-55. Dents were inflicted upon the specimens using drop impact methods. A strong correlation between the dent fatigue effect and impact energy was observed.

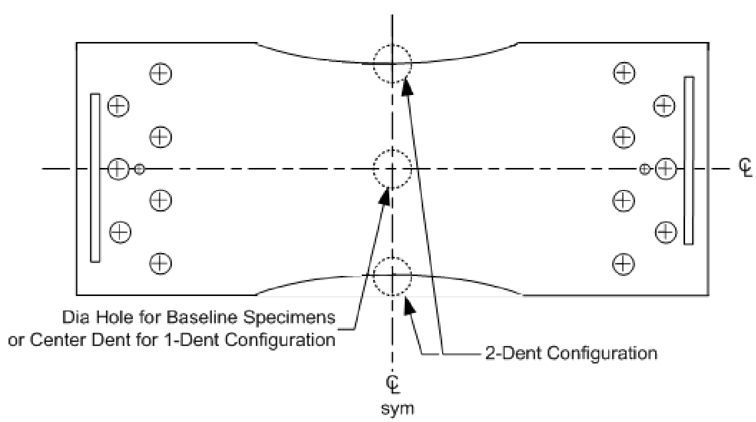


Figure 9.6-55. Specimen Configuration

Results of the fatigue tests are given in Figure 9.6-56 and Figure 9.6-57. From Figure 9.6-56, as the dent depth per thickness increases so does the amount of energy to obtain such depth. The energy calculated through each dent depth was then used to compare relative fatigue lives. Figure 9.6-57 shows the amount of fatigue life increase for each specimen compared to an open hole specimen. Fatigue life of a dented specimen can result in a longer life than an open hole up to a certain energy level as depicted in Figure 9.6-57. Data correlation of fatigue life factors based on dent energy resulted in values of R^2 between 0.72 and 0.85 (which is excellent for fatigue data).

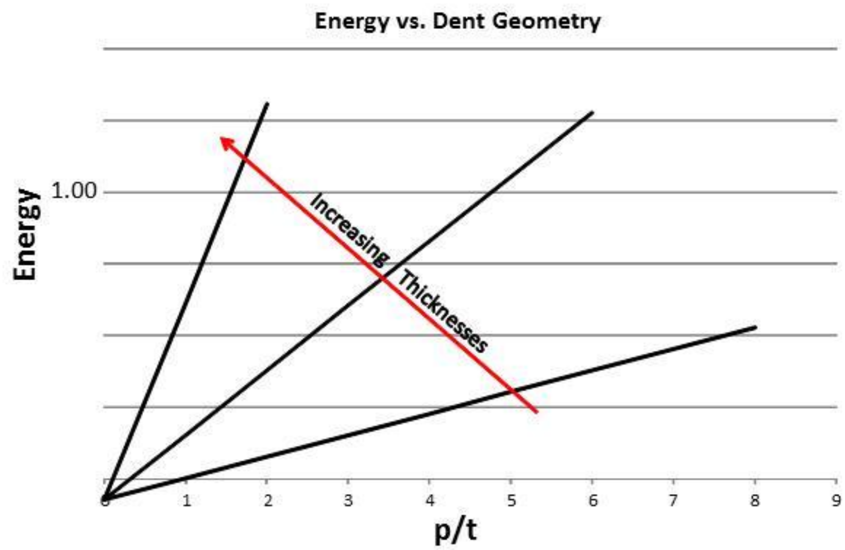


Figure 9.6-56. Energy vs. Dent Depth/Thickness

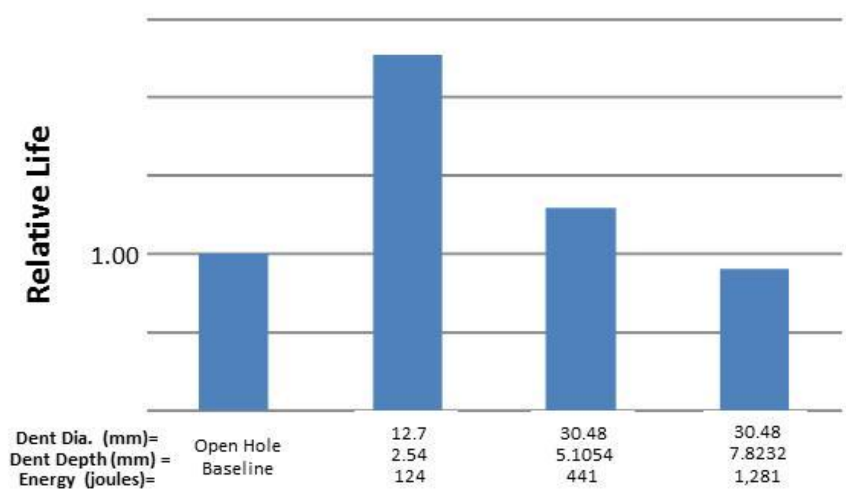


Figure 9.6-57. Relative Life for Fatigue

9.6.21. Structural Testing and Material Characterization

Gregory Shoales, USAF Academy-CAStLE

CAStLE continues to emphasize structural testing and materials characterization performed in their well-equipped Applied Mechanics Laboratory.

CAStLE is currently performing spectrum validation tests to support B-1B full-scale fatigue testing. These tests will allow for more efficient load application to the aircraft structure during testing, with the goal of reducing test time while rigorously maintaining correct airframe loads.

Material characterization was the centerpiece of a recent project with the KC-135 program office. Some wing aft terminal fittings (Figure 9.6-58) were in need of extensive testing due to suspect grain flow lines in the forged part. Characterization included tensile strength and elongation, fatigue crack growth rate, fracture toughness, smooth specimen fatigue (S-N curve), and stress corrosion cracking. All in all, over 180 tests were performed on specimens extracted from six forgings to characterize the material [1]. The Air Force Research Laboratory's Materials and Manufacturing Directorate (Mr. Steven Thompson of RXSA in particular) played a prominent role in the very successful project. Also part of the Integrated Product Team (IPT) were members of the KC-135 Program Office, Boeing and CAStLE's contracting partners, Sabreliner and NexOne, Inc.

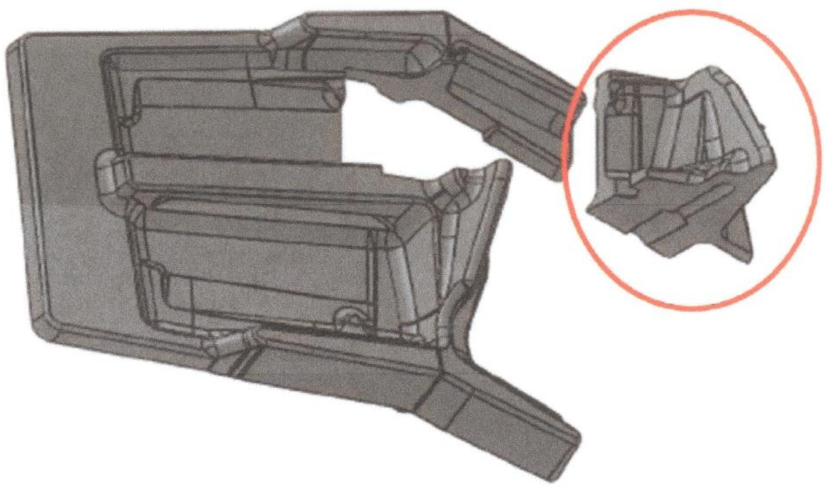


Figure 9.6-58. Wing Joint Fitting and Section of Interest Extracted for Analysis

Retardation effects are important to correct Damage Tolerance Analyses (DTA). Understanding how part material, loading and geometry interplay to slow or arrest crack growth in the presence of overloads is critical to correctly setting inspection intervals for fatigue-critical aircraft structural details. When using the AFGROW crack growth modeling tool and the generalized Willenborg retardation model, selecting the correct shut-off overload ratio (SOLR) is critical. Testing representative coupons using the correct loading spectrum is often used to obtain SOLR for a particular structural detail. CAStLE has been performing testing of 7075-T7351 wing skin material in different thicknesses in an attempt to isolate the thickness effect on SOLR testing. Multiple tests of coupons of various thicknesses (0.125, 0.375, 0.625 inches) are currently underway, and are being accompanied by fractographic analysis (Figures 9.6-59 and 9.6-60).

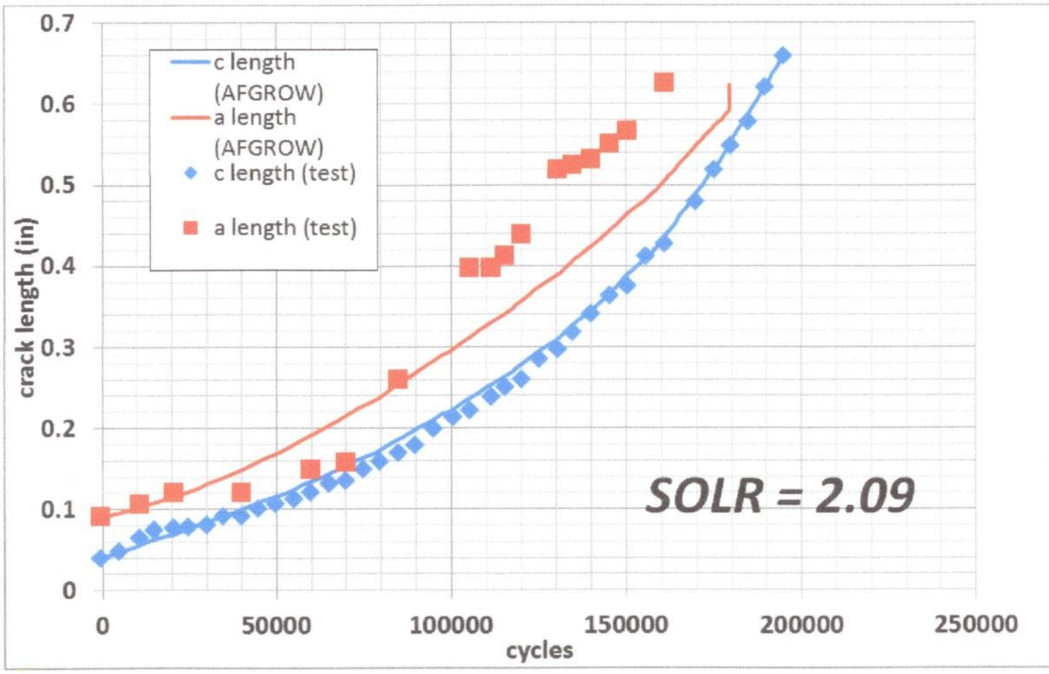


Figure 9.6-59. AFGROW Comparison with Test Data for *a* and *c* Crack Lengths for a Particular Value of SOLR

Figure 9.6-60. SOLR Specimen (Plate with Center Hole) Mounted in Servo-Hydraulic Test Machine

Corrosion testing continues to be an emphasis area for CAStLE and has multiple project goals: (1) developing a standard test specimen and protocol for evaluating the effects of inhibitors on crack growth, (2) developing a test chamber to expose built-up structural samples to load/environmental spectra that mimic in-flight conditions, (3) assessing the inhibition of crack growth by live bacteria, notably *R. Picketti* and (4) exploring novel salt-deposition techniques that deliquesce salts onto structure and provide a much more realistic test environment than the full-immersion tests often currently used. These topics are the subject of other ICAF 2013 abstracts, which provide more details on all of these efforts.

Recently, testing has been performed at CAStLE to assess the effects of using compression pre-cracking on thin C(T) specimens for obtaining near-threshold crack growth rates in aluminum. It is believed that compression pre-cracking results in a smaller residual plastic zone at the crack tip, which results in lower starting loads for decreasing ΔK testing [2].

Magnetostrictive Sensors (MsS) have shown promise for structural health monitoring, and cadets at the USAFA have done some laboratory testing using actual aircraft structure to try and assess its effectiveness. This study demonstrated the capability of a Magnetostrictive Sensor (MsS) Monitoring System to monitor crack growth in a critical aircraft structure with complex geometry. Fatigue testing was performed on two segments of fuselage longerons from a trainer aircraft. Due to their location and geometry, fuselage longerons are very difficult to inspect, requiring substantial aircraft downtime for disassembly. Validation of the MsS system’s ability to detect and monitor crack growth in aircraft components has the potential to reduce maintenance hours and increase aircraft availability. MsS sensors use a ferromagnetic material bonded to the aircraft in order to generate an ultrasonic wave induced from an electromagnetic pulse within the structure of interest. As the wave is reflected off geometries, such as holes and cracks in the longeron, the waveform data are recorded. By comparing subsequent waveform data to baseline/reference data, the system can locate and monitor crack growth. Currently, the MsS system has proven to accurately detect crack growth in land-based structures such as pipelines, bridge cables, power transmission towers and some limited applications in aircraft. These tests aimed to validate the use of MsS systems on complex aircraft structure. The crack growth data and sensor data acquired from the MsS system was analyzed by Southwest Research Institute in order to correlate sensor signal amplitude with crack growth. The data from the right hand longeron tested indicated that a crack was growing. However, due to a testing anomaly, the MsS system was unable to reliably detect monotonic crack growth. In the second test on the left hand longeron the MsS data were better correlated to the observed crack length. This test showed great promise for the MsS system’s capability to accurately detect crack growth. Figure 9.6-61 shows typical amplitude and frequency plots from the MsS system. Validation of this MsS system could lead to the development of a less-intrusive nondestructive inspection (NDI) method with similar detection capability of more traditional NDI methods.

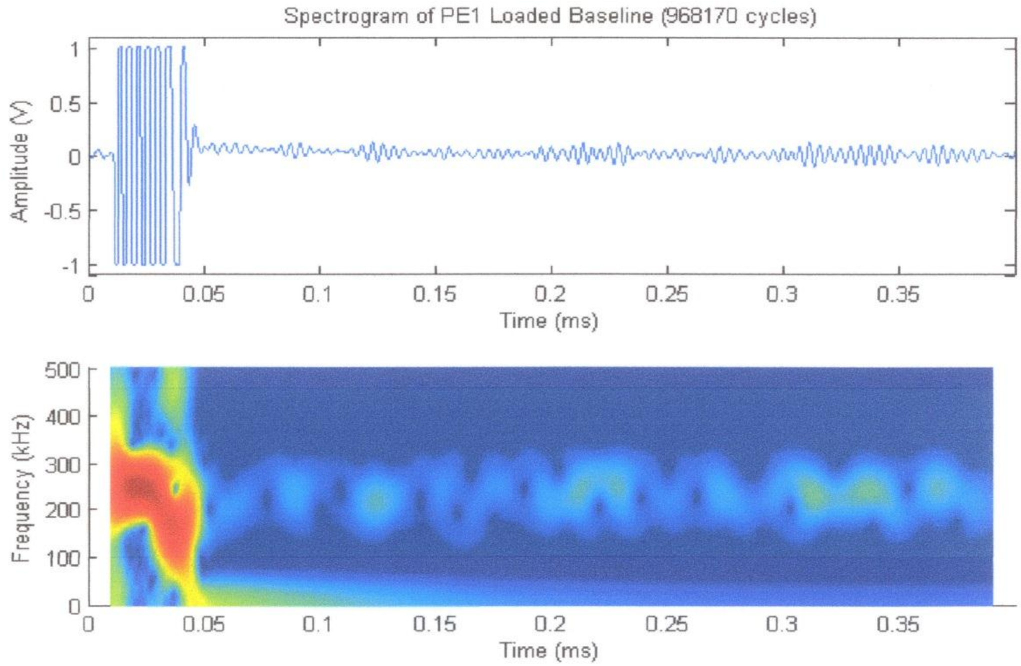


Figure 9.6-61. Pulse-Echo Acquisition for Left-Hand Longeron

References

[1] Shah, S., Seebeck, M. L. and Greer Jr., J. M., "Characterization of a Large Aircraft Forging using a Multi-Party Integrated Product Team (IPT)," in *Proceedings of the 2012 U.S. Air Force Aircraft Structural Integrity Program (ASIP) Conference*, 29 November 2012, San Antonio, TX.

[2] Walker, K. F. and Barter, S. A., "The Critical Importance of Correctly Characterizing Fatigue Crack Growth Rates in the Threshold Regime," in *Proceedings of the 26th ICAF Symposium*, 1–3 June 2011, Montreal, Canada.

[3] Esau, Z., McCullough, K., et al., "Crack Growth Detection with Magnetostrictive Sensors," prepared for submission to the *Proceedings of the 54th AIAA/ASME/ASCE/AHS/ ASC Structures, Structural Dynamics, and Materials Conference*, 8–11 April 2013, Boston, MA.

9.6.22. Stress Intensity Factors for Finite Width Plates

Matthew Hammond and Scott Fawaz, USAF Academy-CAStLE support contractors (SAFE Incorporated)

Accurate damage tolerance analyses of aerospace structure will lead to a more cost effective and safe flying environment, especially as the average age of the United States Air Force's fleets continue to climb higher. This is only possible through a complete investigation of the material properties, crack Stress Intensity Factors (SIF), and Loads/Environment Spectra Survey (L/ESS), among others. As can be seen in Figure 9.6-62, there can be a significant impact on the crack growth life predictions with only a 10% increase in the SIF values.

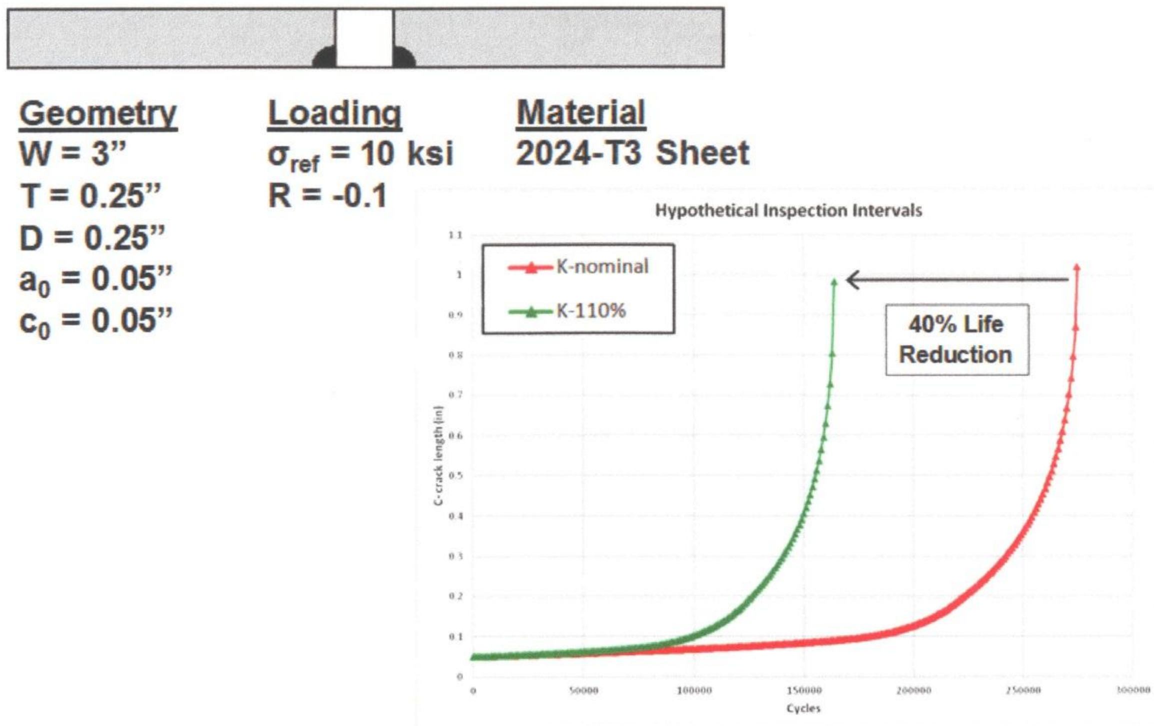


Figure 9.6-62. Impact of Inaccurate Crack Growth Analysis

The current investigation seeks to develop an extensive SIF database for elliptical corner cracks emanating from centrally located holes in finite width plates. The investigation includes uniaxial tension, out-of-plane bending, and bearing load cases. The range of crack and plate geometries is extensive. The large solution space covers the following geometries:

$1.1 \leq W/D \leq 20$
$0.2 \leq D/t \leq 20$
$0.1 \leq a/t \leq 0.99$
$0.1 \leq a/c \leq 10$

where W is the plate width, D is the hole diameter, t is the plate thickness, a is the through-thickness crack length, and c is the crack length in the transverse direction, These plate geometries cover many of the short edge distance to hole diameter ratios commonly found in aerospace structures, and encompass the following range:

$0.55 \leq e/D \leq 10$

where e is the distance from the center of the hole to the near edge.

Well-structured, fully hexahedral, finite element (FE) cracked plate models, like the one seen in Figure 9.6-63, are automatically generated and interrogated for mesh quality. Over 150,000 individual crack models are used to generate a sufficiently high-fidelity solution space as to minimize interpolation error in SIF extraction at intermediate crack/plate geometries in finite width plates.

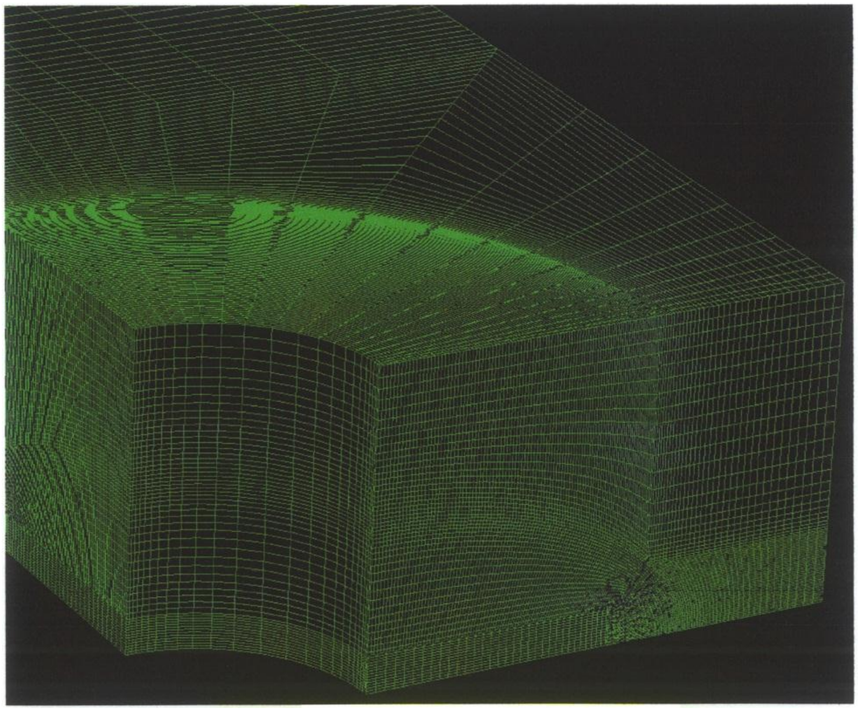


Figure 9.6-63. Well-Structured FE Mesh of Elliptical Crack in Finite Width Plate

Early comparisons between double symmetrical corner cracks, DSCC, (Figure 9.6-64) and single corner crack, SCC, (Figure 9.6-65) geometries show dramatic differences in the SIFs currently available in industry and solutions developed within this effort for finite width plates.

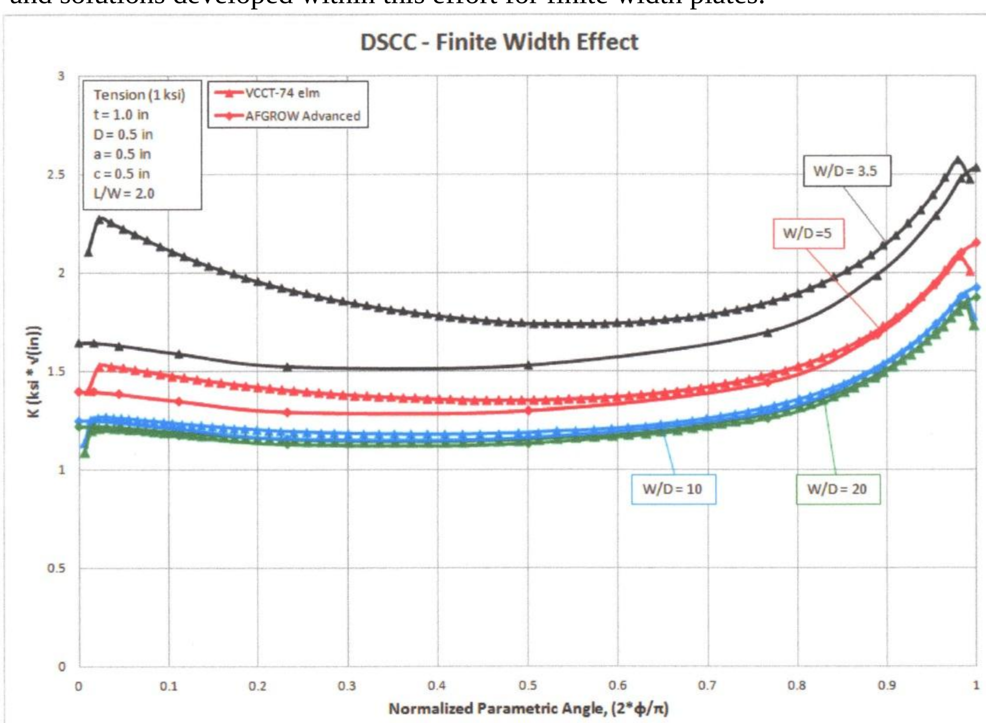


Figure 9.6-64. Variation in SIF with Decreasing Width Plates

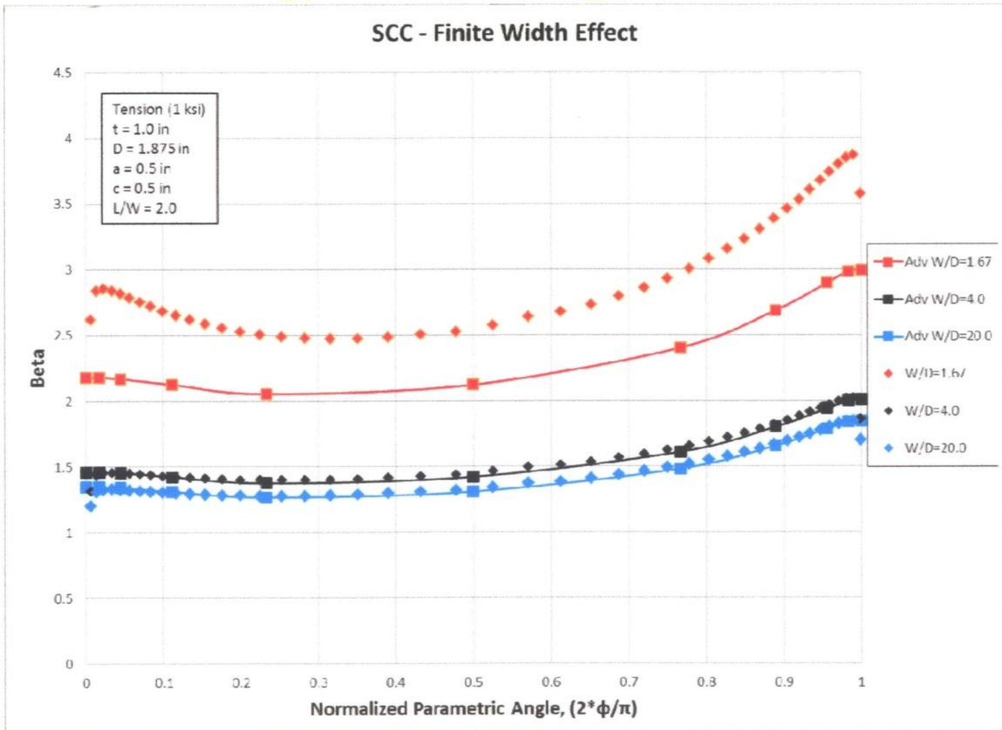


Figure 9.6-65. Variation in SIF with Decreasing Width Plates

9.6.23. The Effect of Corrosion Inhibitors on Corrosion Fatigue of Aircraft Aluminum Alloys

Sarah Galyon Dorman, CAStLE support contractor (SAFE Incorporated); Benjamin Hoff & Daniel Henning, United States Air Force

Corrosion fatigue is an area of concern for the United States Air Force (USAF) and other Department of Defense organizations. Often the USAF corrosion prevention systems include chromate containing coatings, typically in the form of chromate conversion coatings and primers. Chromate has been used successfully for many years on USAF aircraft to prevent corrosion damage. However, the environmental and personnel risks associated with chromate coatings have caused the USAF to pursue non-chromate containing corrosion prevention coatings. To fully quantify chromate replacement coatings, an understanding of the effects that chromate has on corrosion fatigue must be fully documented and understood. Some researchers have shown that chromate added to 0.6 M NaCl full immersion corrosion fatigue tests on 7xxx series aluminum alloys slows the fatigue crack growth rate substantially [1]. The limitation of this research was that the amount of chromate present in the environment was not related to leach rates of chromate from polymeric coatings [1].

The majority of USAF aircraft are protected from corrosion by polymer coatings loaded with corrosion inhibitors; for these inhibitors to slow fatigue crack propagation the corrosion inhibitors must become mobile from hydration of the polymer coating matrix [2-7]. Based on this mechanism of corrosion inhibitor release, it becomes important to know how much chromate or other inhibitor is able to leach from the polymeric coating. Testing was completed to determine the amount of chromate expected in areas of transport aircraft that experience corrosion such as lap joints and other structure. Based on these results corrosion fatigue testing in bulk solution conditions with chromate concentrations of 0.5mM and lower are planned. As chromate replacement coatings are also of interest to the USAF, the same leaching experiments were completed on molybdate containing primers and corrosion fatigue testing of concentrations of 0.05mM and lower will also be analyzed [8].

The development of the database accounting for the effect of chromate and molybdate on corrosion fatigue crack growth rates in aluminum alloys has been slowed by the presence of bacterial contamination, *Ralstonia pickettii*, in some of the full immersion tests. When the bacteria are present the fatigue crack growth rates in 7xxx series aluminum alloys are greatly reduced. Figure 9.6-66 shows the effect of *R. pickettii* on fatigue crack growth rates of a 7xxx series alloy in 0.06 M NaCl compared to high levels of chromate, a known corrosion fatigue inhibitor. The mechanism by which the bacteria are lowering fatigue crack growth rates in sodium chloride solutions is not currently understood, but is being investigated. The current hypothesis is the development of a protective film, whether oxide or some other material (biofilm), on the crack surface affects the overall corrosive environment such as desalination of the test solution.

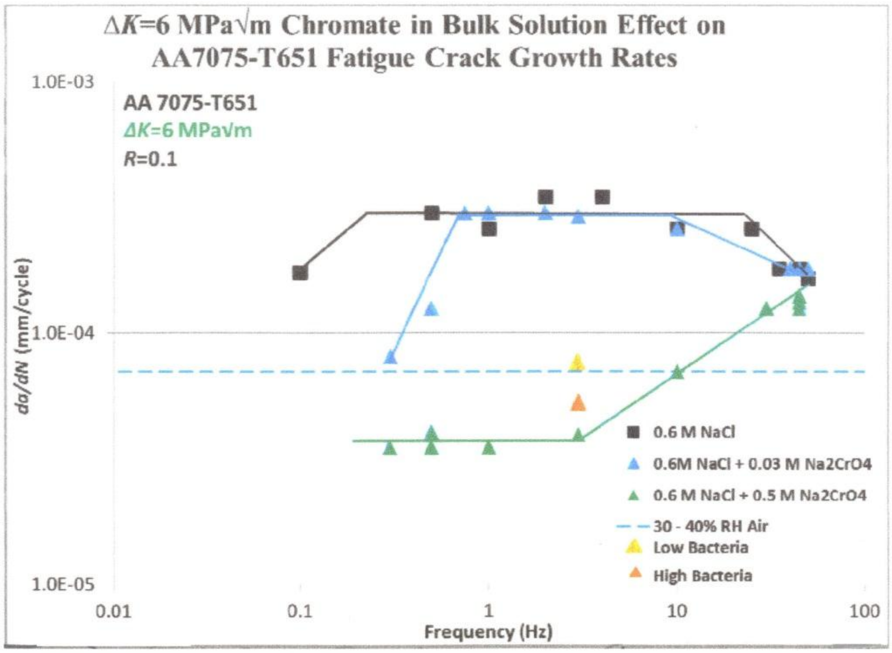


Figure 9.6-66. Fatigue Crack Growth Rate Comparison for Chromate and *R. pickettii* Tested at a Constant *ΔK*=6 MPa√m, *R*=0.1 in 0.6 M NaCl [1].

Future work will continue to investigate the effect of chromate and other corrosion inhibitors on environmentally assisted fatigue. The fatigue work will redirect towards testing under atmospheric (deliquesced salt) conditions to better mimic real world aircraft environments rather than simply full immersion corrosion fatigue testing. An understanding of the effect of chromate on environmentally assisted fatigue will be produced. This dataset will allow for a baseline comparison for chromate replacement corrosion inhibitors. The addition of the atmospheric corrosion data will provide for a more robust understanding of how corrosion inhibitors behave in real world situations.

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9.6.24. Development of Equipment and Methods for Testing Under Environmental Spectrum

Sarah E. Galyon Dorman, USAF Academy-CAStLE support contractor (SAFE Incorporated); Benjamin Hoff & Daniel Henning, United States Air Force; Erin Endres & Karen Chinnery, United States Air Force Academy

Corrosion fatigue research continues to be of great importance to the United States Air Force (USAF). Current efforts are underway to move towards more sophisticated methods of characterizing corrosion fatigue damage. In the past, most corrosion fatigue testing has been completed using fully immersed samples or samples in humid environments. Corrosion in the atmosphere is more complex than these laboratory test environments and the corrosion morphology produced is often different. In the last several years, research has moved towards atmospheric testing in which a sample has known salt concentrations deliquesced on the surface of the sample during fatigue testing [1]. The process of applying the deliquesced salt layer is time consuming and requires great precision. The environmental test chamber is also a complex system as the relative humidity must be held extremely constant to avoid variations in the deliquesced salt layer. As most life prediction models use alloy data produced in laboratory air, the life estimates are likely conservative. As the USAF looks for cost savings and to extend the life of aircraft beyond their original design, the ability to test alloys in more real-world conditions becomes more important. Being able to run an alloy or aircraft structure through an environmental spectrum in conjunction with a loading spectrum would allow for better and more complex life prediction tools. Because of the desire to move towards more accurate corrosion fatigue testing of aerospace alloys, the United States Air Force Academy (USAFA) has been working to design test equipment to aid in this cutting edge research.

Specifically the USAFA has worked to design a printer to deposit controlled amounts of salt onto the surface of a fatigue sample, shown in Figure 9.6-67. The printer allows for different salt morphologies to be deposited on a sample surface and then deliquesced for fatigue testing. While the current printer has been built to apply salt to a single sample, the user could code the software to apply salt to any number of fatigue sample geometries. This novel method of applying the environment to the sample allows for a much greater variance in the environment than previously allowed. As research continues into understanding how environments around the world vary, the salts applied can be varied and combinations of salts can be examined to understand how pollution, proximity to the ocean and other factors influence corrosion and corrosion fatigue rates.

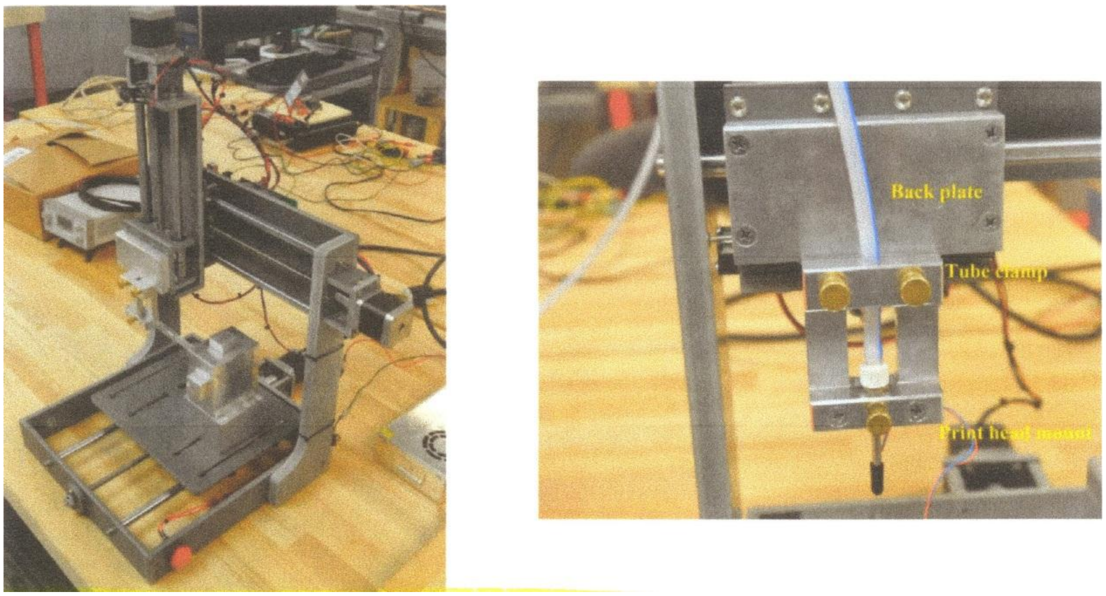

Figure 9.6-67. Salt Deposition System Platform and Close-up of Print Nozzle for Salt Spray

To examine the effect of deliquesced salt on aircraft structure, a larger more complex environmental chamber must be produced. Currently testing has been completed with small, well-defined fatigue samples that have deliquesced salt on the surface. To control the salt concentration on the sample, the relative humidity inside the environmental cell is maintained through a salt bath in the cell. Currently the small environmental chamber holds approximately 500 mL of solution when completely full. To again produce better data for the examination of the role of corrosion on the reduction in fatigue life, it would be ideal to be able to test aircraft structure or aircraft representative structure under an environmental load spectrum. To this end, the USAFA Center for Aircraft Structural Life Extension (CAStLE) is designing a test chamber for the environmentally assisted fatigue testing of aircraft representative structure. The chamber will allow for the relative humidity to be held constant to examine the effect of a salt layer on the surface of the sample along with ultraviolet (UV) light effects and ozone degradation. Figure 9.6-68 shows a schematic of the proposed chamber design. The environmental cell is being designed such that different geometry samples can be tested without redesign. Once complete this chamber will allow for the examination of an environmental spectrum in conjunction with a loading spectrum. This advance should allow for a much more comprehensive understanding of how environment degrades aerospace materials under real world conditions.

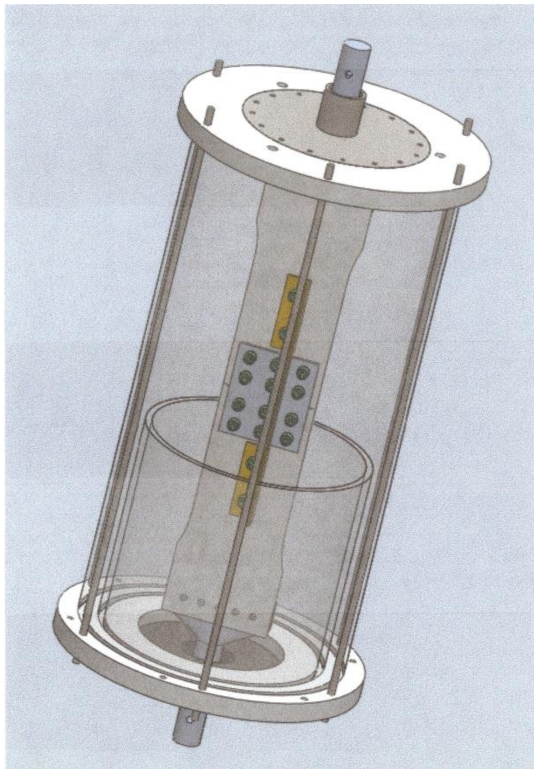

Figure 9.6-68. Schematic of Proposed Environmental Chamber for Large Scale Structure Environmental Fatigue Testing

Note: The fatigue sample shown is simply an option with the chamber being able to accommodate many differnent sample geometries. Shown sample geometry provided by AFRL.

References

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9.6.25. Effect of Low Temperature and Water Vapor Environments on the Fatigue Crack Growth Behavior of Aerospace Aluminums

James T. Burns, USAF Academy – CAStLE support contractor (University of Virginia)

Fatigue behavior is governed by both mechanical (stress intensity range, ΔK) and chemical driving forces. Loading in a moist-air environment produces atomic hydrogen (H^+) that enters the material at the crack tip and enhances damage in aerospace aluminums by one or more unique mechanisms [1]. As such the loading environment significantly affects the fatigue behavior of aluminum components but is often ignored in airframe structural management. Both the damage tolerant and safe-life approaches use material properties gathered in room temperature laboratory air environments with relatively high levels of moisture (Relative Humidity ≈ 20-70%). However, a significant portion of fatigue loading on both fighter and transport airframes can occur at high-altitude [2, 3], where the water vapor pressure (P_{H2O}) is very low and components can be at low-temperature. Critically, research has shown that loading at low P_{H2O} [4, 5] and/or low temperatures (< -50°C) [6, 7] reduces crack growth rates by an order of magnitude. Incorporating the beneficial effects of the high-altitude environments into current structural integrity management approaches may produce more accurate fatigue modeling, reduce conservatism, and reduce the maintenance inspection burden. The goal of this work is to (1) develop a database where temperature, P_{H2O}, and loading parameters vary for a legacy (AA 7075-T651) and modern (AA 2199-T8) aluminum alloy, and (2) gain a mechanistic understanding of the environmental fatigue process to inform a fracture mechanics based life prediction algorithm for variable environments.

To determine the effect of temperature and water vapor pressure on the aluminum alloys, long crack da/dN vs. ΔK data are obtained via compliance measurements during decreasing ΔK testing of compact tension specimens at R=0.5 and f=20 Hz. As significant airframe loading occurs up to an altitude of ~15,000 meters, temperatures and P_{H2O} typical of this range [8] are investigated. Testing is performed at incremental temperature values (roughly every 15°C) between 23°C and -90°C; as well as tests at 23°C within a vacuum system that sets the P_{H2O} to the equilibrium water vapor pressure above ice for a given temperature [9]. This test method will assess the role of temperature separate from the influence of P_{H2O}. The results of the testing are shown in Figure 9.6-69 (AA 7075-T651) and Figure 9.6-70 (AA 2199-T8). Both alloys show a systematic decrease in crack growth rates as P_{H2O} decreases; this is first observed at a P_{H2O}/f=17 Pa-s for AA 7075-T651 and at 8.25 Pa-s for AA 2199-T8. There is good alignment at constant P_{H2O} controlled by either vacuum or temperature for AA 2199-T8 down to -15°C. Below this temperature the growth rates diverge; specifically at high ΔK the crack growth rates align with ultra-high vacuum data, then at intermediate ΔK there is a transition regime, and at low ΔK growth rates align with 23°C results at equivalent vacuum controlled P_{H2O} values. This indicates that there is a growth rate/ ΔK dependence of the mechanism that governs the environmental cracking behavior at low temperatures.

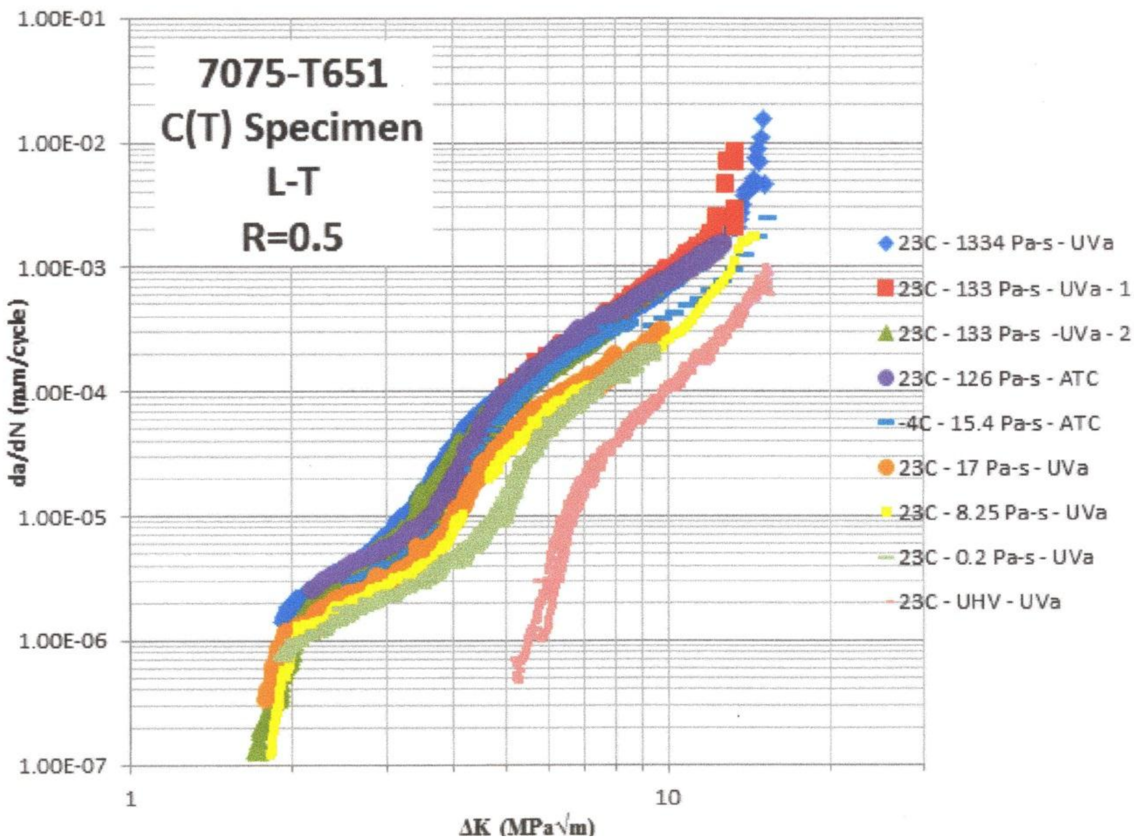

Figure 9.6-69. Crack Growth Rate versus ΔK Data for 7075-T651 Tested at R=0.5 at Various Environmental Conditions and a Frequency of 20 Hz

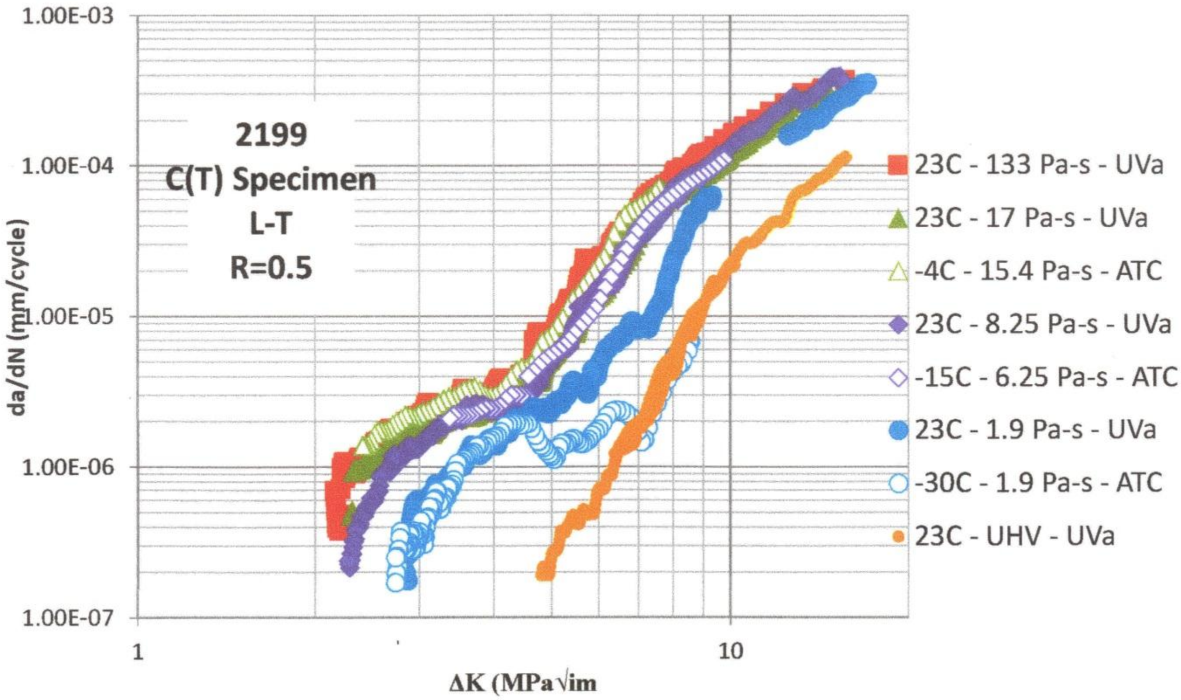

Figure 9.6-70. Crack Growth Rate versus *ΔK* Data for 2199-T8 Tested at *R*=0.5 at Various Environmental Conditions and a Frequency of 20 Hz.

Future work will continue the characterization of the environmentally dependent fatigue crack growth rates, scanning electron microscopy (SEM) characterization of the fracture surface, and collaborative efforts to study the dislocation structure proximate to the fracture surface via focused ion beam (FIB) and transmission electron microscopy (TEM) analysis. These characterization efforts will inform the mechanistic understanding and help to develop assumptions for an engineering level protocol that incorporates environmental effects into structural life predictions.

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9.6.26. Development of a Fatigue Testing Method for Analysis of the Corrosion Pit to Small Crack Transition

Divakar Mantha, USAF Academy-CAStLE support contractor (SAFE Incorporated)

Corrosion fatigue is of great concern to the United States Air Force and the Department of Defense (DoD) as equipment and aircraft platforms are being used well beyond their original design life. The cost of corrosion for DoD infrastructure and weapon systems is increasing every year due to aging of structures. As such, an understanding of how corrosion damage affects fatigue crack formation would allow for better life prediction models and the ability to test corrosion inhibitors that might slow this transition from a corrosion pit to a fatigue crack below the damage tolerant flaw size. The current research focuses on the development of a standardized fatigue test methodology to study this pit-to-crack transition.

The development of the larger pit-to-crack test protocol involves the development and implementation of a sequence of protocols, namely (1) a pitting protocol to generate consistent corrosion pits in the size range of 100 – 200 μm on the center hole fatigue specimens (Figure 9.6-71 shows the fatigue specimen and Figure 9.6-72 is a picture of a controlled pit produced by the protocol), (2) a spot welding protocol to attach the voltage probes on either side of the corrosion pit to measure fatigue crack growth rates using the direct current potential drop (DCPD) method, (3) a fatigue crack growth testing in lab air with the introduction of marker band loading to establish fatigue crack progression (Figure 9.6-73), and (4) development of an AFGROW simulation that can predict the pit-to-crack transition from crack growth rate (da/dN) data. To develop the pit-to-crack test methodology a SIPS legacy aluminum alloy and temper, AA7075-T65, is being utilized. There is a large database of fatigue data for this alloy and it is a legacy alloy often used on USAF aircraft, so it is particularly relevant to the Air Force. The pitting protocol has been finalized; the finalization of the spot welding protocol is underway along with the development of the fatigue protocol and the AFGROW life prediction tool.

Future work will apply the pit-to-crack transition protocol to determining how the corrosion damage to fatigue crack transition changes with the presence of corrosion inhibitor (chromate and chromate replacement coatings), material substitution alloys, environment and other aircraft relevant situations.

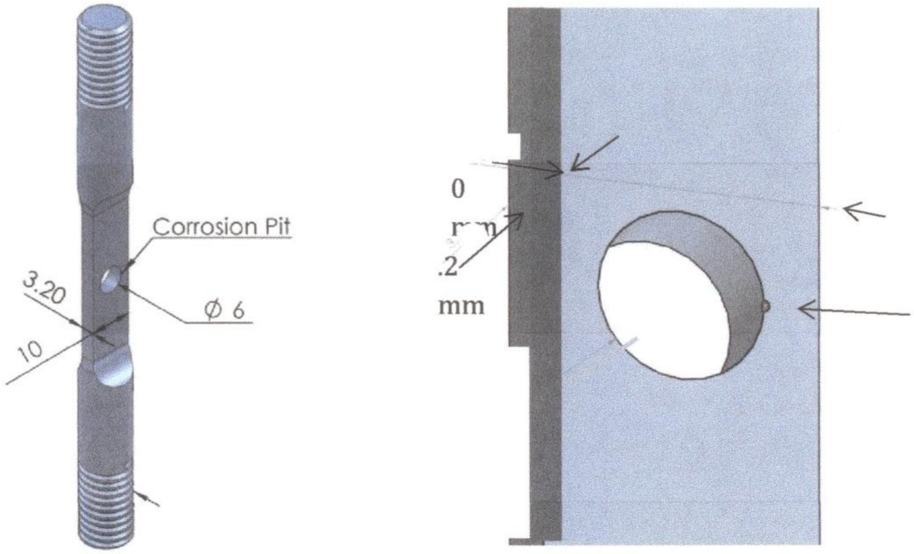


Figure 9.6-71. Corrosion Fatigue Sample with a Close up of the Pit Location

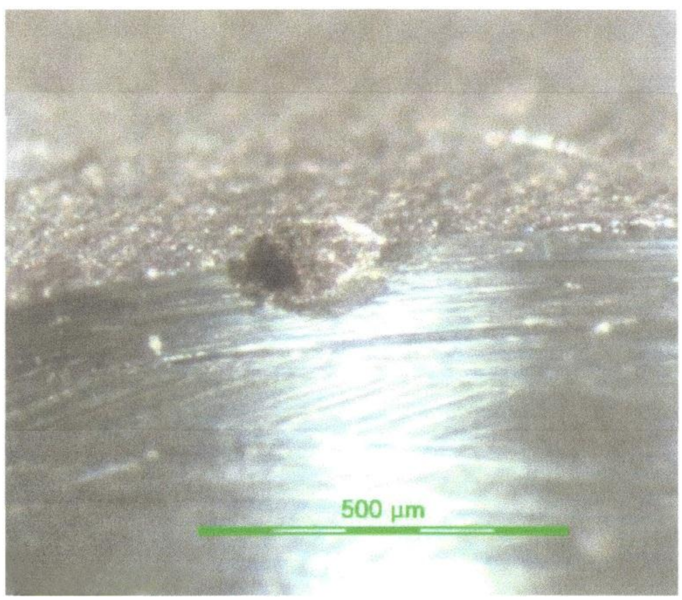


Figure 9.6-72. Corrosion Pit Produced with the Developed Pitting Protocol Along Center Hole of an AA 7075-T651 Specimen

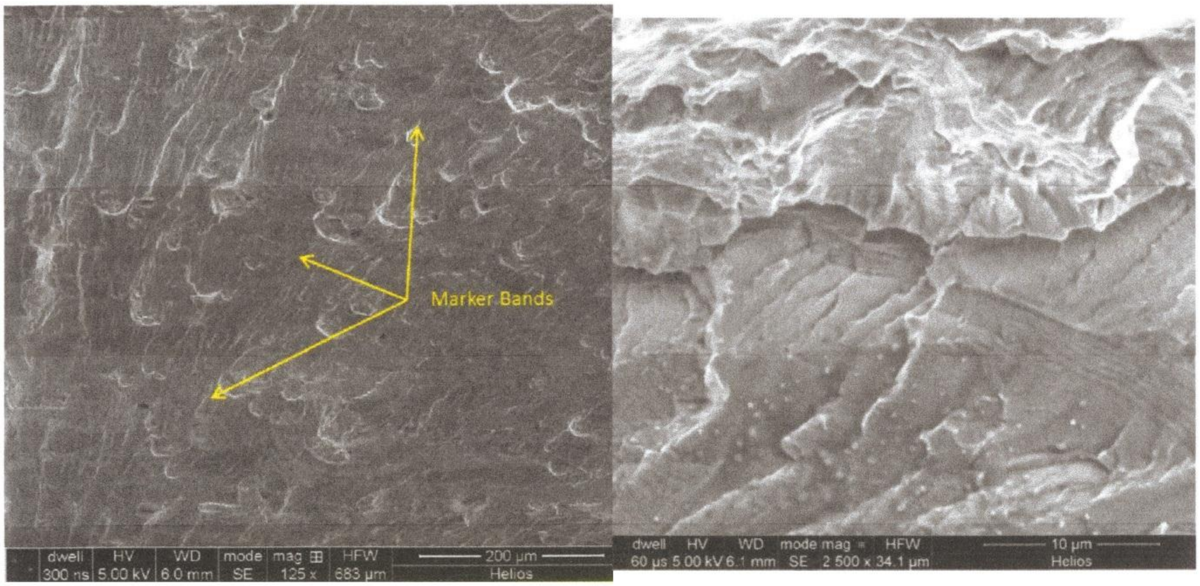


Figure 9.6-73. SEM Image of Fractured Surface of AA 7075-T651 Sample with Arrows Denoting the Locations of the Applied Marker Bands

Note: The image on the right shows a close up of the applied marker bands.

9.6.27. Modeling and Simulation

Gregory Shoales, USAF Academy-CAStLE

CAStLE continues to perform modeling and simulation efforts using the Department of Defense High-Performance Computing network of computers. These efforts are focused in the areas of high-fidelity finite element modeling and stress intensity factor (K) solution development.

Dr. Börje Andersson, under contract to CAStLE, continues to develop millions of K solutions related to the geometries of Figures 9.6-74 and 9.6-75. Development of these solutions uses the STRIPE computer code developed by Dr. Andersson while at the Swedish Defense Agency.

Case	Crack configuration	Cracks	Solut. (M)	Solut. (M)
1		1-2	7.9	7.9
3		1-2	11.0	11.0
5		1-2	7.7	7.7
2		1-2	0.7	0.7
4		1-2	0.3	0.3
6		1-2	1.0	1.0
7		1-2	0.3	0.3
8		1-2	0.7	0.7
9-32		1-2(4)	38.4	1788.0
33-34		1-2	0.9 0.7	0.9 0.7
35-36		1-2	1.8 1.6	1.8 1.6

Figure 9.6-74. Solution Space for Current K Solution Efforts at CAStLE

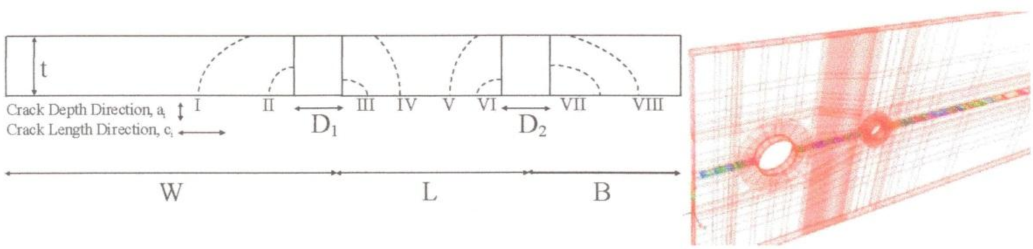


Figure 9.6-75. The Case 9-32 Description and Typical FE Mesh

Also developing K solutions under a cooperative agreement with CAStLE is Mr. Matthew Hammond of SAFE, Inc. Mr. Hammond is filling in some extreme parts of the solution space that are of interest to the crack growth community. This work is examining elliptical corner cracks emanating from holes in finite-width plates and is the subject of another ICAF 2013 abstract.

CAStLE's work on the C-130 Center Wing Box (CWB) model continues, and current efforts are focused on finishing enough of the fastener database to perform a demonstration of part of the giga-DOF model. The model is depicted in Figure 9.6-76.

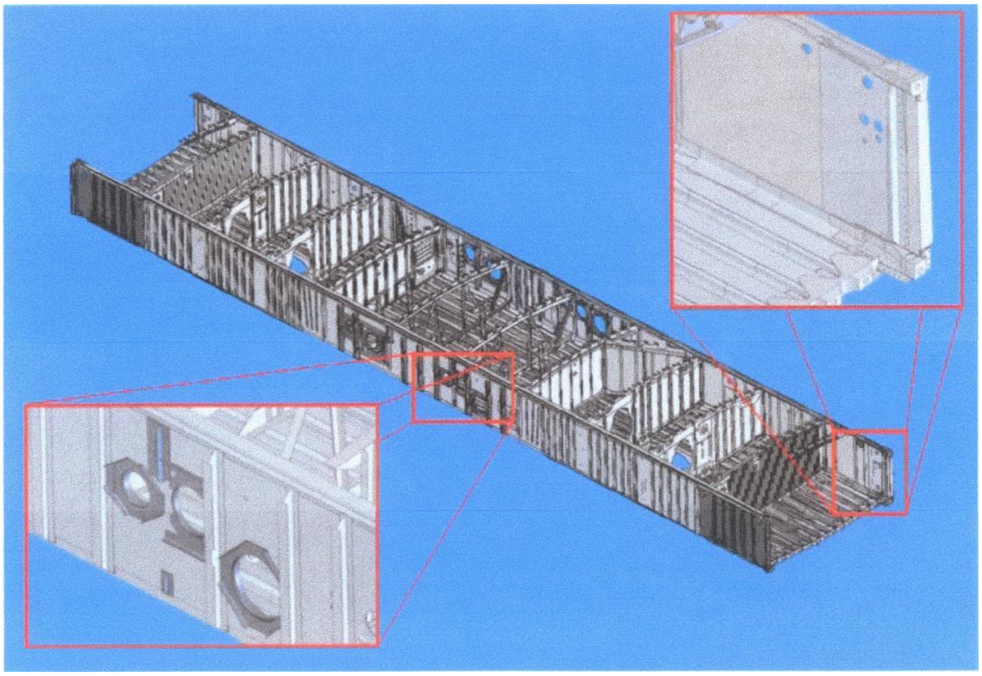


Figure 9.6-76. Detailed Giga-DOF Model of C-130 Center Wing Box

9.6.28. U.S. Navy / Boeing P-8A Poseidon Full-Scale Fatigue Test Program

J. Restis, T. Turner, and H. Wardak, The Boeing Company – P-8A Program; J. Candela, M. Edward, B, Lloyd, and N. Phan, USN – NAVAIR

The full-scale fatigue test of the P-8A for the U.S. Navy is in progress at Boeing in Renton, WA. The P-8A Poseidon is a military derivative of the Boeing commercial 737 aircraft (Figure 9.6-77). A high-level rendition of the test is provided in Figure 9.6-78.

Figure 9.6-77. P-8A Poseidon

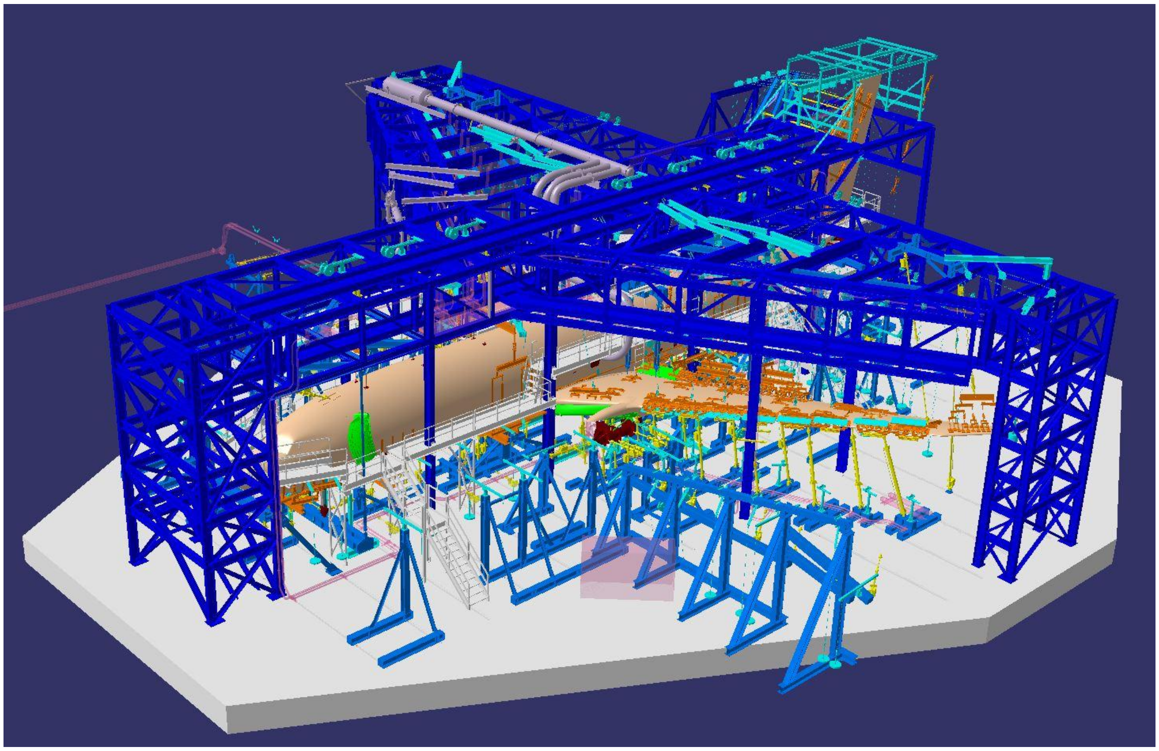


Figure 9.6-78. P-8A Full-Scale Fatigue Test Setup

The P-8A will undergo a two-lifetime fatigue test to validate the design service objective (DSO) for the airframe, per U.S. Navy requirements. The Boeing engineering and NAVAIR structures team have developed the test requirements, test loads, event tape with flight sequencing, test designs and test fixturing for the test. Fatigue and damage tolerance analysis and Aircraft Structural Integrity Program (ASIP) application are all part of the pre-testing tasks.

The test program has an aggressive schedule, taking advantage of the previous full-scale test program experience at Boeing and close coordination with the NAVAIR team. The P-8A full-scale fatigue test is one of the three full-scale tests that are currently taking place in Puget Sound Boeing; the other two are the Boeing 787 full-scale fatigue test, and a B-1 full-scale fatigue test. The horizontal stabilizer and main and nose landing gears are scheduled to be tested off-aircraft at the Boeing laboratories in St. Louis, MO.

The fatigue test spectra are based on the mission profiles and usage for which the P-8A is designed, and were defined based on U.S. Navy requirements. The spectra are appropriately clipped and truncated. The fatigue test event tape has 40 flights and close to 3,000 load cases. Test loads were applied to the S2 finite element model to compare design and verification internal loads (the design loads) with test loads. Design shear, moment, and torsion (VMT) at load reference axes (LRA) were checked to verify VMT prior to the development of the event tape. The fatigue loads meet the targeted fatigue damage requirements at fatigue critical areas on the airplane and match the analytical exceedance curve. Marker bands have been introduced to the event tape at prescribed intervals to assist the post-test tear down evaluation.

Active and passive load systems are used to apply loads with hydraulic actuators and to achieve a load balance of the airplane. Weights are counterbalanced either by physical offload fixtures or by tare loads using the test load systems. Active and passive system loads are monitored by the test control system. The control system will interlock or go into a hold if these systems exceed their limits. Hydraulic and air pressurization systems provide hydraulic power to the load systems and pneumatic power to the fuselage main cabin pressurization system. Thrust straps are designed to resist longitudinal loads and yaw moment. Whiffle tree, formers, straps and pads are used to apply specified actuator loads to the airframe. Strain and load measurement instrumentation and photogrammetry will be used at specified locations of the airframe. Fault analysis, inner and outer limits on tracking error, overloads, A&B channel compare and air system overpressure protection are all part of the testing procedures to protect the test article.

Prior to fatigue testing, the airplane completed a "high blow" (proof) test, to locate pressure leakage areas. This was followed by tuning (dump tuning and group tuning) to optimize test control system settings to achieve quality and test speed goals. Visual inspection of the airplane at the completion of all test fixtures prior to the cyclic load application was done to ensure test article acceptance for testing. Unique load endpoint validation was performed for a number of defined load cases from the event tape prior to cyclic load application. This was done to validate applied loads and strain readings in order to establish quality of actuator loads.

Cyclic load application starts with the initial block load (local) test of the universal aerial refueling receptacle slipway installation (UARSSI), weapon bay door, and engine strut, all on aircraft. Next is full-scale testing of the entire airframe, which is then followed by block load testing of the wing control surfaces on-aircraft. This completes the first lifetime testing cycle. The second lifetime and extended lifetime testing will follow the same process.

A strain survey will be performed at specified applied test cycles, and prior to the restart of testing if any major repairs are required. A detailed inspection plan has been developed based on damage tolerance analysis and the airplane maintenance plan that will be used during testing. At the completion of the two-lifetime fatigue test, a residual strength test will be conducted to demonstrate airframe capability. A comprehensive teardown is scheduled to take place following the test.

9.6.29. F-15 Full-Scale and Component Fatigue Tests

Roy Scheidter, The Boeing Company – F-15 Program

The F-15 (Figure 9.6-79) remains one of the most capable multirole strike fighters available today. During the past three decades, Boeing has produced more than 1,600 F-15 aircraft, the latest variants being the F-15E Strike Eagle and a number of export versions. The U.S. Air Force plans to fly the F-15C/D and F-15E for many years to come; no sunset dates have as yet been established.

Figure 9.6-79. USAF/Boeing F-15 (USAF Photo)

Fatigue testing is being conducted to recertify the service life of the F-15C/D and F-15E airframes. The test articles have over one design lifetime of operational service. Three tests are being run:

- F-15C Full Scale Fatigue Test (FTA7)
- F-15E Full Scale Fatigue Test (FTE10)
- F-15C/E Stabilator Component Fatigue Test (FTA8)

The first two are being performed side-by-side at a Boeing major airframe test facility in St. Louis, MO (Figures 9.6-80 and 9.6-81). Figure 9.6-82 shows the stabilator in test.

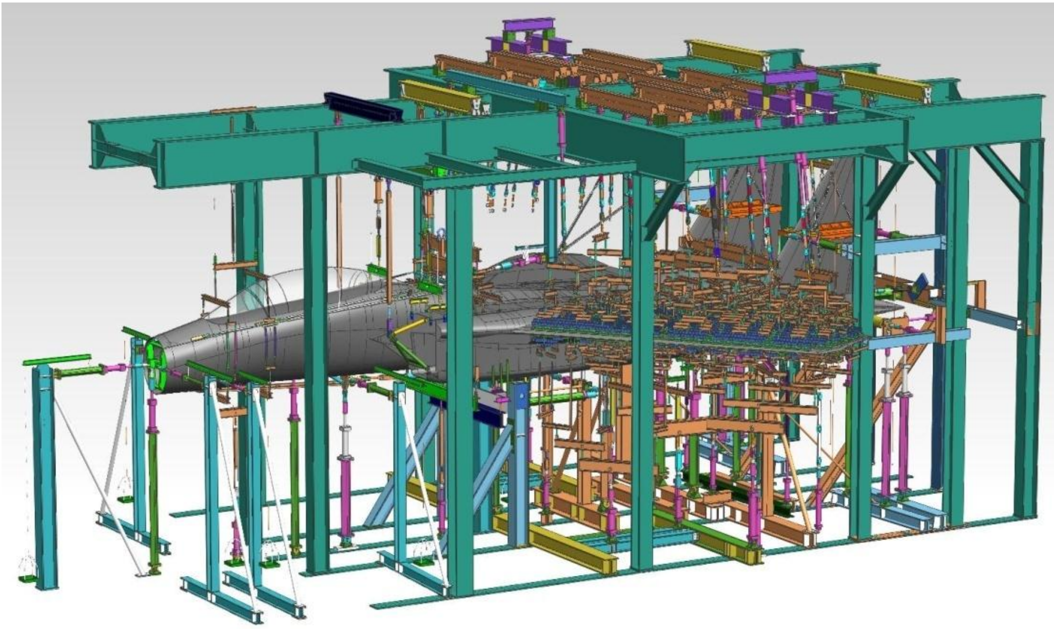


Figure 9.6-80. F-15 Full-Scale Fatigue Test Setup

Figure 9.6-81. F-15C and F-15E Full-Scale Tests

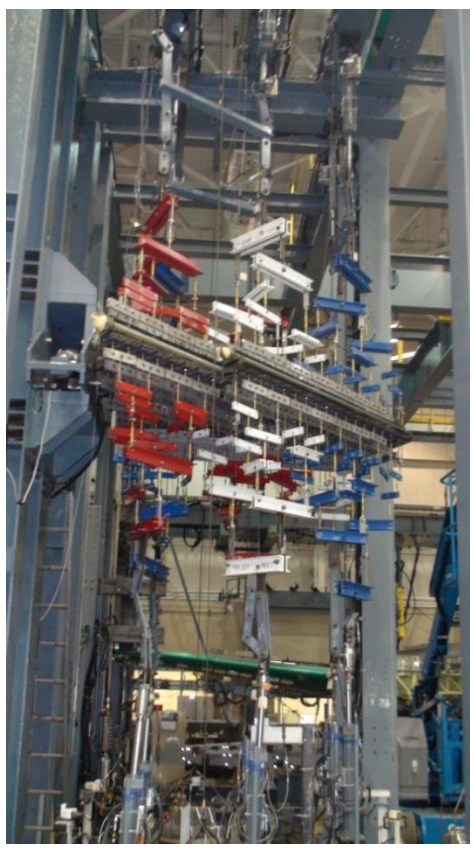


Figure 9.6-82. F-15C/E Stabilator Test

Cycling on the F-15C full-scale fatigue test began in September 2011 and is expected to continue through March 2014, after which the test fixture will be disassembled, final reports will be prepared, and the contract closeout activities will be conducted over the next four months. Wing replacement is anticipated/planned during the course of the test, after which limited teardown and analysis of the first set of wings is planned to be accomplished. Cycling on the F-15E full-scale fatigue test started in November 2012 and is expected to continue through May 2015, with follow-on activities similar in scope to the F-15C test. Cycling on the F-15C/E Stabilator Component Fatigue Test was conducted in May – June 2012, successfully achieving the test goals. Teardown and analysis of the test article is underway. In addition to extending the rated life of the airframe, these tests will be used to evaluate the performance of repairs conducted prior to, and during test and their potential application to fleet aircraft.

9.6.30. Boeing 787 Full-Scale Fatigue Test Program

S. Shaffner, The Boeing Company – 787 Program and P. Brownlow, The Boeing Company – Test & Evaluation

Full-scale fatigue testing of the Boeing 787 began on August 15, 2010 at Boeing in Everett, WA and recently completed its 60,000th flight. The fatigue test airframe is a fully structurally representative production 787 airframe with the exception of non-critical and off-airplane tested items, and is tested in a test fixture specifically designed and built for the test. A photograph of the test article and test setup is shown in Figure 9.6-83.

Figure 9.6-83. Full-Scale Fatigue Test Setup

This test is used to confirm the durability and damage tolerance characteristics of the 787 airframe structure. Additionally, the test supports verification of the proposed 787 maintenance program. And finally, the test will provide full-scale test evidence demonstrating the effectiveness of 787 airframe provisions to preclude the possibility of Widespread Fatigue Damage (WFD) occurring within the design service goal of the airplane, as required by the Federal Aviation Administration and similar regulatory agencies worldwide.

A total of 165,000 flight cycles – equivalent to 3.75 times the Design Service Objective (DSO) of the aircraft – representative of 787-8 short mission loading will have been applied to the 787 airframe at the completion of testing. The test requirements, test loads, flight profile, flight sequencing, test designs and test fixturing were jointly designed by engineers from Boeing Commercial Aircraft (BCA) and Boeing Test & Evaluation (BT&E).

The 787 full-scale fatigue test spectrum can be categorized as a "5 × 5" flight spectrum – that is, the load profile contains five unique flight types and load levels, and is applied in 5,000-flight repeating blocks. Each block contains 3,704 E-flights, 1,067 D-flights, 215 C-flights, 13 B-flights, and one A-flight, where the E-flight contains only the lowest load levels and the A-flight contains all load levels. In all, there are approximately 300 unique load cases. Prior to the start of testing, the test loads that had been developed were applied to a finite element model to compare design and test loads and validate that the applied fatigue loads will meet the targeted fatigue damage requirements at fatigue critical areas on the airplane. Pressure-only marker band loading is also part of the load profile and is applied at prescribed intervals to assist in post-test teardown analysis.

A combination of 144 active and passive load systems are used to apply loads to the airframe and to achieve load balance – about 1.5 times the number of systems used for the 777 full-scale fatigue test. Counterbalance systems and/or tare loads are employed to offload test fixturing, horizontal load systems, and portions of the test article. Vertical loads are reacted through the main and nose landing gear, and horizontal loads are reacted through thrust/drag straps and lateral body whiffle tree systems. Test loads are applied to the airframe via bonded load pads, discrete fittings, dummy airframe structure, and load formers. The test control system continuously monitors load application and is programmed to provide an audible warning, put the test in a hold state, or shut down the test should established control limits be exceeded. Similarly a separate air control system monitors the pressurization of the fuselage main cabin. Test control limits, mechanical devices, and testing procedures provide the necessary protection of the test article. In addition, approximately 2,800 channels of strain and load measurement instrumentation are installed at specified locations of the airframe and data from these transducers are collected once during each flight block to monitor the health of the airframe structure.

Regular fatigue testing operates 24 hours per day, seven days per week with regular inspections occurring every 6,000 flights. The inspection requirements are consistent with published 787 maintenance planning data and supplemental inspections for aging 787 aircraft. To date, there have been no substantial findings. Testing and teardown inspection are expected to complete in 2015.

9.6.31. Continued Development of the NASGRO Software for Fracture Mechanics and Fatigue Crack Growth Analysis

Craig McClung, Joseph Cardinal, Yi-Der Lee, and Vikram Bhamidipati, Southwest Research Institute®; Joachim Beek and Royce Forman, NASA – Johnson Space Center; Venkataraman Shivakumar, Randall Christian, Yajun Guo, and Michael Baldauf, Jacobs ESCG Group

The NASGRO® software for fracture mechanics and fatigue crack growth (FCG) analysis continued to be actively developed and widely used during 2011 and 2012. NASGRO is the standard fracture control software for all NASA Centers and is also used extensively by NASA contractors, the European Space Agency (ESA) and ESA contractors, and FAA Designated Engineering Representatives certified for damage tolerance analysis, as well as many aerospace companies worldwide. NASGRO has been jointly developed by NASA and Southwest Research Institute since 2001, with substantial financial support from NASA, the NASGRO Consortium, and the Federal Aviation Administration (FAA). The NASGRO Consortium is continuing its fourth three-year cycle (2010-2013), and is now enrolling participants for the fifth cycle (2013-2016). The international participants currently include Airbus, Alcoa, Boeing, Bombardier Aerospace, Embraer, GKN Aerospace Engine Systems, Honda Aircraft Engines, Honeywell Aerospace, Israel Aerospace Industries, Lockheed Martin, Mitsubishi Aircraft Corporation, Mitsubishi Heavy Industries, Siemens Energy, Sikorsky, SpaceX, Spirit AeroSystems, United Launch Alliance, and UTC Aerospace Systems. In addition to Consortium members, 138 single-seat and 5 site NASGRO licenses were issued in 2011-2012 to users in 19 countries.

Two new production versions of NASGRO were released in 2011 and 2012. Version 6.2, released in Sept 2011, included new SIF solutions (see Figure 9.6-84) for a through crack at an angular or elliptical edge notch in a plate (TC17), a through crack at an offset embedded slot or elliptical hole in a plate (TC18), and a through crack at an offset hole in a plate with a broken ligament (TC19). These new solutions all permit specification of remote tension and bending loads or local crack plane stresses and include extremely broad geometry ranges. Existing SIF solutions for stiffened panels and through cracks at holes were enhanced. Tabular FCG rate data capabilities were completely revised and enhanced to add new interpolation, threshold and instability options; full GUI plotting; and increased capacity. New solution algorithms with improved robustness and accuracy were implemented for inverse calculation modes (e.g., compute initial crack size given target life). A new Configuration Control module was developed that permits a company "superuser" to lock down specified options in the executable NASGRO. The current version of the FASTRAN crack growth program developed by Prof. J. C. Newman, Jr., was integrated into NASGRO in a semi-independent form. FCG data for seven aluminum alloys were added to the material properties module.

Version 7.0, released in Nov 2012, contained a wide variety of new features. New SIF solutions included unequal through cracks at an offset hole (TC23), corner or surface crack at an elliptical or angled edge notch (CC13/SC26), corner or surface crack at an embedded slot or elliptical hole (CC14/SC27), and corner/surface crack at a round hole with a broken ligament (CC15/SC28). See Figure 9.6-84 for more details. The new notch/slot solutions have the same overall geometry range as the corresponding through crack solutions in v6.2. Most weight function SIF solutions now allow different stress gradients at either tension vs. compression loads or max vs. min loads. Users can now compare two different materials graphically, calculate the number of cycles to user-specified crack size(s). Plotting and printing of stress gradient information has been enhanced. In-plane bending is now available for surface, corner, and through cracks at holes. Many other new features and bug fixes were also included in both v6.2 and v7.0.

Significant progress was achieved on NASGRO 7.1, with Alpha release scheduled for mid-2013, and Production release later in the year. New features under development include improved SIF solutions for single/symmetric corner cracks at holes, new SIF solutions for dissimilar corner cracks at holes, enhancements to several other SIF solutions, a new XML material database with powerful search capabilities, enhanced capabilities for multiple-temperature properties and analyses, user-specification of toughness as a function of crack type, an optional new interface for linking NASGRO directly with other computer programs, and others.

Southwest Research Institute has been conducting NASGRO training courses since 2006. During 2011 and 2012, SwRI trained 167 students in nine courses, including four courses in San Antonio, Texas, and five courses at remote sites including a major aircraft company, a major gas turbine engine company, NASA Kennedy Space Center, and the ESA Technical Center in the Netherlands.

Further information about NASGRO is available at www.nasgro.swri.org. POC: Craig McClung, Southwest Research Institute, craig.mcclung@swri.org, 1-210-522-2422.

New Stress Intensity Factor Solutions in NASGRO 6.2 and 7.0

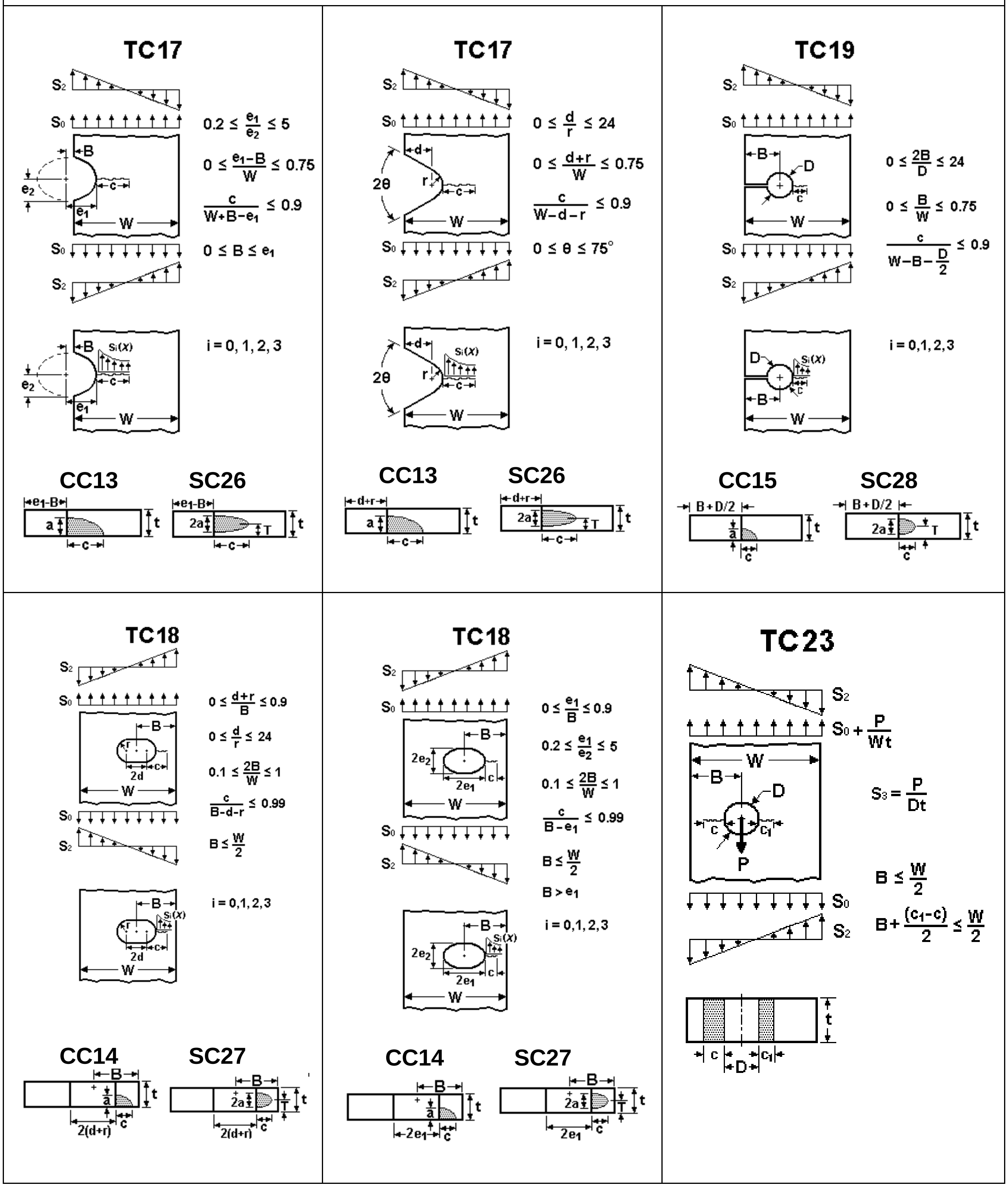


Figure 9.6-84. New Stress Intensity Factor Solutions in NASGRO Versions 6.2 and 7.0

9.7. PROGNOSTICS & RISK ANALYSIS

9.7.1. Fuselage Skin Crack Due to Engine Exhaust Impingement

Luis Diaz Rodriguez, USAF C-17 Program Office; Mark DeFazio, USAF ASC/EN; and Ko-Wei Liu, The Boeing Company

The United States Air Force C-17 large transport aircraft utilizes thrust reversers to meet stringent design requirements which entail landing and taking off on short unprepared runways. In 2005, several large visible cracks were discovered on fuselage skins running forward and aft along the edge of the longeron / stringer just forward of the wing-to-fuselage fairing on an operational aircraft (Figure 9.7-1). Visual inspections conducted on several aircraft with a high number of full-stop landings did not reveal any cracking. Similar cracking was discovered in early 2007 on three different aircraft (Figure 9.7-2). An instrumented aircraft was utilized to obtain environmental loading on the fuselage skin in the area where cracks had developed. Ground test and analysis revealed the redesigned nacelle introduces high acoustic vibration and direct exhaust impingement on the fuselage skin from the engine thrust reverser when the trust reverser is deployed during landing roll out and ground operations (Figure 9.7-3). The direct impingement load from exhaust gases is sufficient enough to cause the fuselage skin to buckle at a high frequency which causes the skin to develop a large number of small shallow surface cracks. As the number of landings increases, these shallow surface cracks grow deeper and eventually link up to form a large crack, then transition to a through the thickness / visible skin crack. A doubler repair was developed and fleet impacts were determined by using Weibull analysis to predict / project the number of expected visible skin cracks in the future prior to retrofit being completed (Figure 9.7-4). These data were used to convince leadership to fund and complete the retrofit in a timely manner. This technical activity will present an overview of this problem, data collected from a ground test program and the correlation of the Weibull analysis with fleet data.

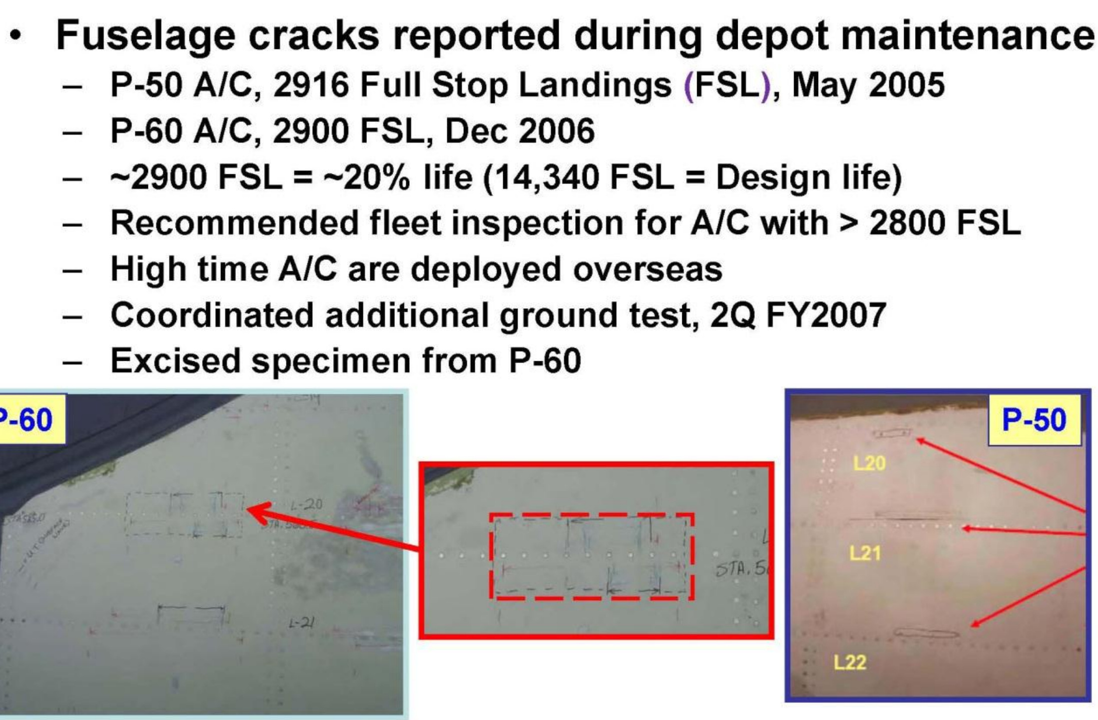


Figure 9.7-1. Observed Cracks During Depot Maintenance

- **First crack found in the field before fleet inspection**
  - P-47 A/C, 3814 FSL, March 2007, Deployed
  - P-49 A/C, 3626 FSL, March 2007, Deployed
  - NO Safety of Flight concern; airframe has large crack arrest capability
- **TCTO (draft) released 3 weeks after first crack found**
  - Inspect (NDI), repair, and reinforce skin with doubles
  - P-40 to P-75 with > 2800 FSL to stabilize the fleet
- **Fleet stabilized 4 months after TCTO release**

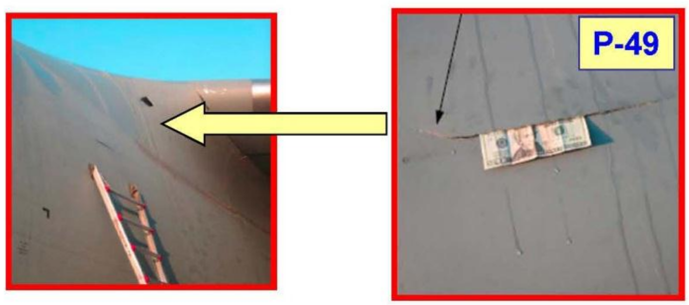


Figure 9.7-2. Observed Cracks in Field

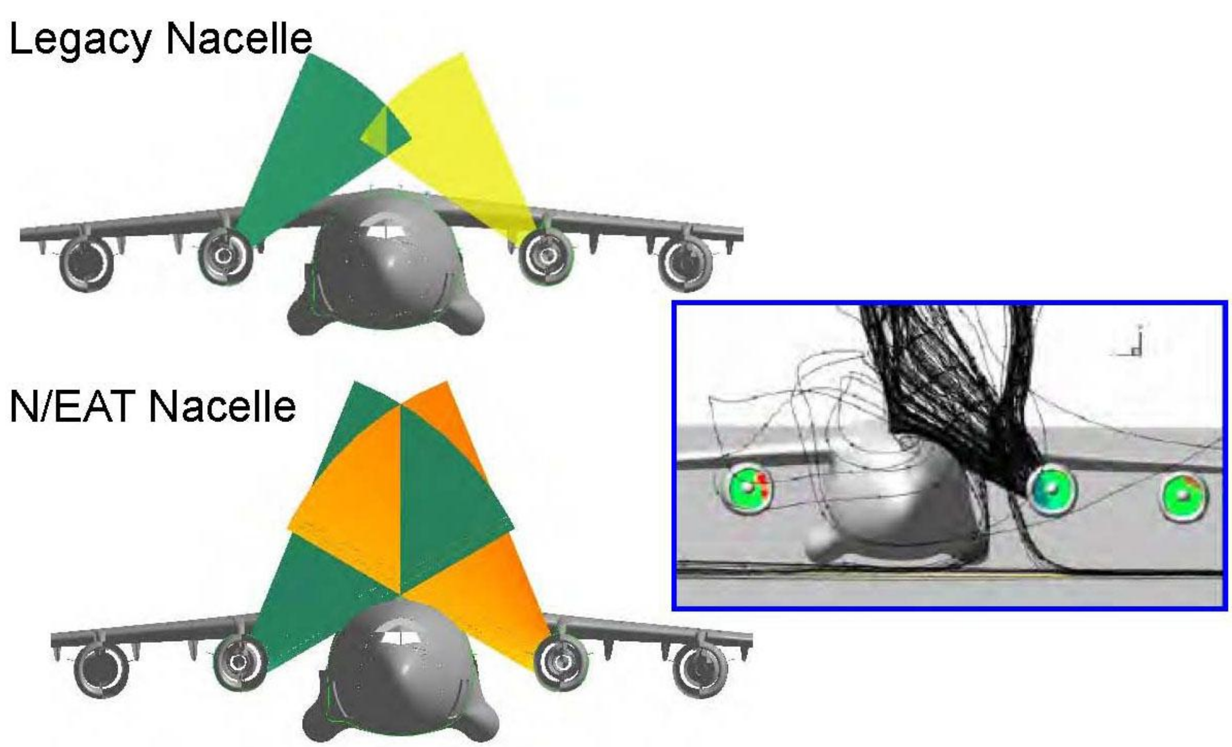


Figure 9.7-3. Change in Environment

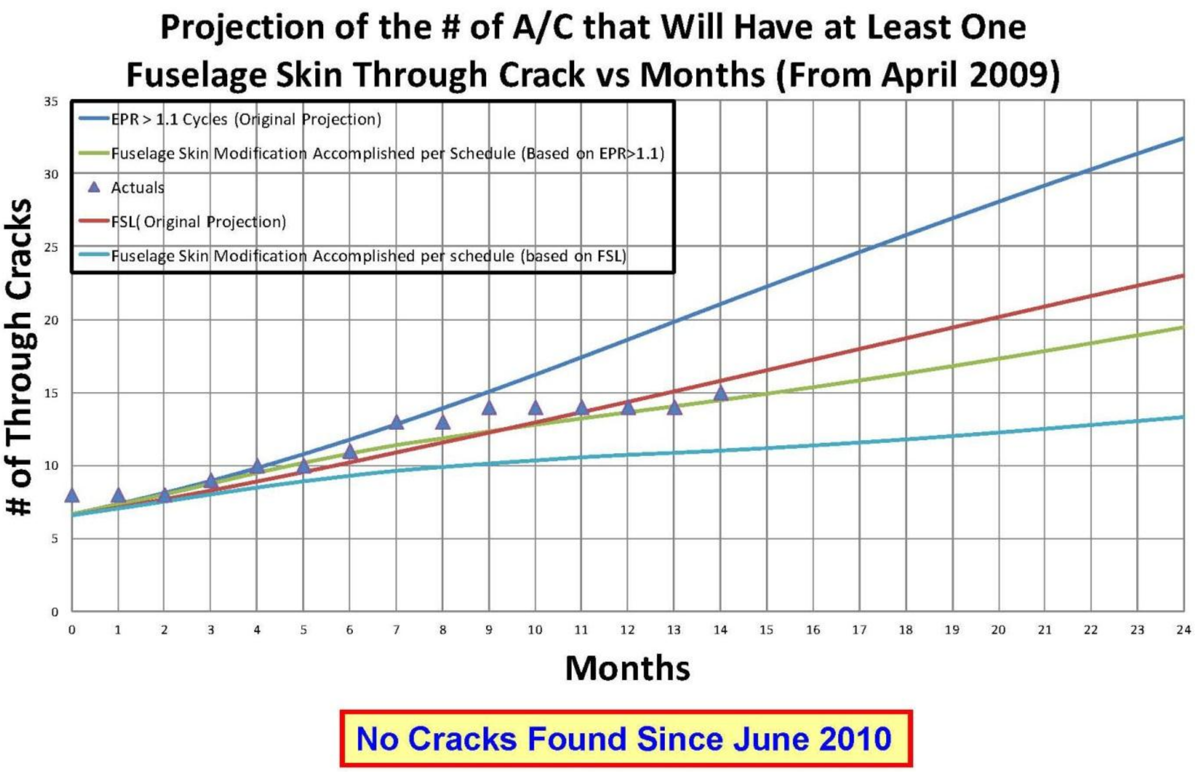


Figure 9.7-4. Crack Predictions

9.7.2. Estimation of True Indication Size from Maintenance Repair Data and POD

Zachary Whitman, Southwest Research Institute

Crucial to the validation of any risk analysis is the comparison of the predicted cracks and their sizes at the time of inspection to what is actually found during the inspection. Due to the nature of most inspections and repairs, however, small cracks or indications are not excised and broken open; thus the true size is unknown. Often, the assumed size for a repaired location that is used in an analysis is simply the amount machined off during a surface blending or hole oversize operation whereby the indication was no longer detected. This approach, however, ignores the fundamental physics of the Probability of Detection (POD) curve and the fact that any oversize or blend repair for small damage is simply a modify-until-missed approach. The result is an underestimated crack population size from the inspection and a lack of trust in the risk analysis (Figures 9.7-5 and 9.7-6).

This technical activity explains how to utilize the POD curve from the risk analysis to estimate the true range of probable indication sizes from a given inspection based upon the repair strategy. These results can be used to: a) develop the 'true' size of the original indications; b) estimate the amount of the original indication that was left behind after repair; and c) optimize the repair strategy itself. It will be shown that gradual blending or oversizing and re-inspection can actually leave a larger residual crack than a repair strategy that takes larger material removal steps. This invokes the need to optimize the repair strategy to minimize the residual crack left behind yet maximize the remaining repair potential by sizing the steps based on POD and the hole or blend limits.

- Contrast POD Probability of Detection vs. POM or Probability of Miss: POM = 1 - POD
- Repair Makes the Crack Smaller and Smaller, Re-Inspecting Until Removed <u>OR</u> Missed
- Each Repair Makes the Next Independent Inspection More Likely to Miss the Remaining Crack Since the Crack is Smaller

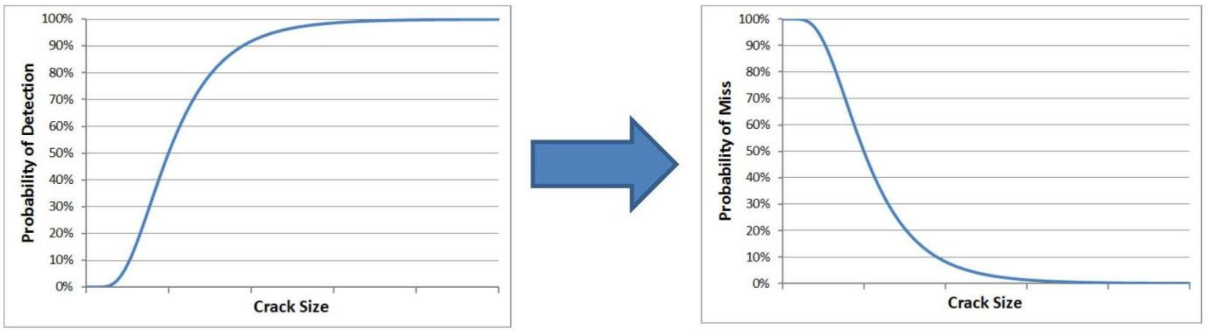


Figure 9.7-5. Contrast of Probability of Detection (POD) vs. Probability of Miss (POM)

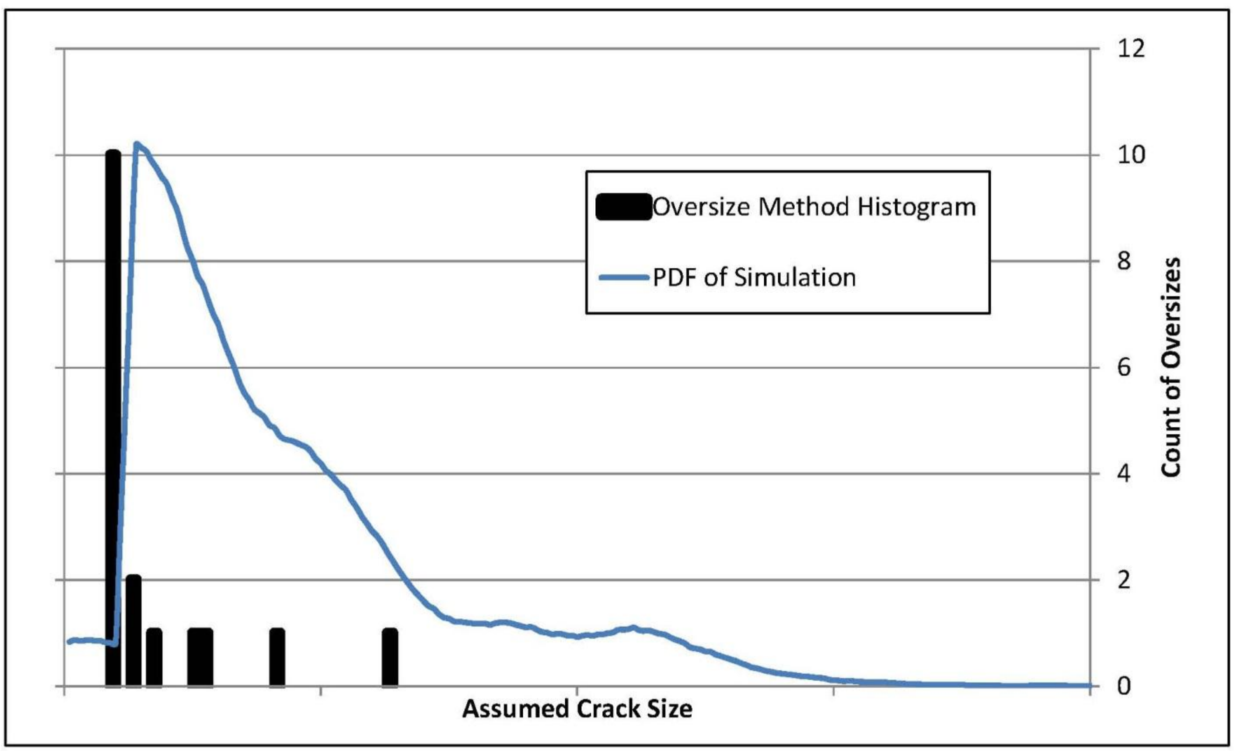


Figure 9.7-6. Comparison of Oversize Method Histogram and PDF of Simulation

9.7.3. U-2 Wing Blade Stress Corrosion Cracking Probability of Failure Analysis

Timothy Floyd, USAF WR-ALC and Javier Favela, Lockheed Martin Corporation

The U-2 airplane (Figure 9.7.7) is experiencing stress corrosion cracks in the wing to fuselage attaching structures commonly referred to as wing blades (Figures 9.7-8 and 9.7-9). Recurring inspections indicate that new cracks are developing and existing cracks continue to grow. Although all testing, analysis and fleet tracking data to date indicates that mitigation efforts of periodic inspections and cracked blade replacements at depot assures the safety of the aircraft, a Probability of Failure analysis would further confirm the efficacy of the mitigation efforts. Historically, however, stress corrosion cracking has been found to be difficult to predict analytically and empirical predictions require gathering data on the specific parts, a large and time consuming effort. This technical activity provides an overview of the process followed to quickly get an estimate of the problem's severity by utilizing the University of Dayton Research Institute's PRoF software in conjunction with some simplifying assumptions that bound the problem and provide a reasonable and yet conservative estimate of the probability of failure. The results of this analysis confirm that it is safe to fly the aircraft without any interruption to aircraft operations until replacement of the affected parts during scheduled depot maintenance. This process will show that a very complex problem, such as stress corrosion cracking, can be tackled with conservative assumptions to provide reasonable approximations of failure probability that may allow operations until a replacement part can be installed or more data can be obtained to refine the analytical solution.

Figure 9.7-7. U-2 Airplane

- Wing Blades found cracked during PDM in March 2007 by QA
- The wing blades are the structural members that connect the wings to the fuselage
- 4 blades per wing, upper and lower, forward and aft
- The thinner part of the blade attaches to the wing with 187 Hy-Loc's
- The inboard end of each fitting has 3 large lugs with 2 barrel nut holes per lug where the wing is bolted to the fuselage

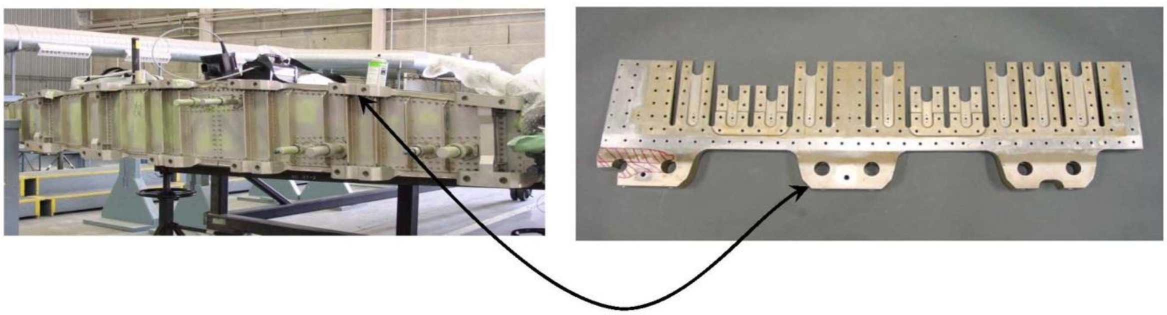


Figure 9.7-8. U-2 Wing Blades

- Review and Analyze
- Blade examined by Lockheed Martin Aero Failure Analysis Lab
  - Determine probable failure mode
  - Confirm blade complied with applicable material specifications
- Findings

1. Multiple, parallel cracks
2. Discontinuous crack through most of the cut section
3. Cracks are now known to be caused by Stress Corrosion Cracking (SCC)
4. SCC is not predictable and we've had to make some simplifying assumptions to perform the analysis shown here

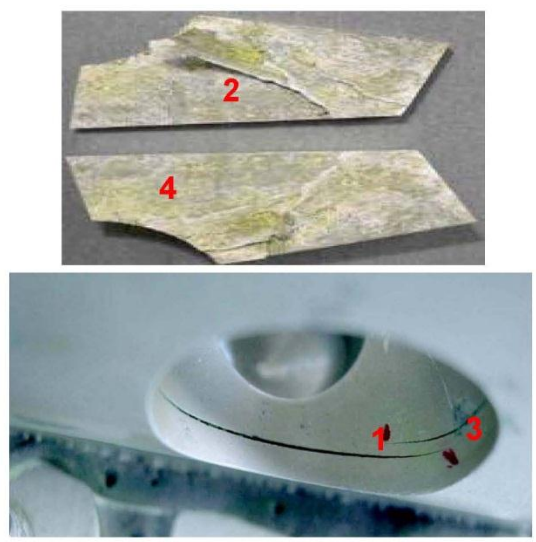


Figure 9.7-9. U-2 Wing Blade Cracking

9.7.4. WFD Rule Impact on Lockheed Martin Commercial Fleet

J.E. Ingram, R.E. Sykes, Alex Navarrette, and G.C. Watson, Lockheed Martin Corporation

In 2010, the FAA issued amendments to certain sections of the Federal Aviation Regulations (FARs) requiring Design Approval Holders of aging transport aircraft to perform assessments of the susceptibility of the affected airframes to widespread fatigue damage (WFD), and take the actions needed to preclude this damage mechanism (Figures 9.7-10 and 9.7-11). Inspection and maintenance programs based on fatigue or crack growth analyses using assumptions of single cracks forming and growing in a stable manner, or derived from tests with insufficient duration to reveal WFD characteristics, will, at some point, become inadequate to ensure structural integrity in the presence of this damage. The objective of the WFD assessment was to identify these susceptible structural locations, utilize all of the test evidence, service experience and appropriate analytical processes to estimate the onset of WFD and determine when, in aircraft cycles or flight hours, the Limit of Validity (LOV) on the existing maintenance program should be declared.

This technical activity describes the methods used by an OEM to perform this assessment for a comparatively small fleet of commercial aircraft and a limited database of service crack information. The largely probabilistic processes varied from one situation to another, depending on the age of the high-time aircraft and scope of the durability and residual strength testing conducted during aircraft development. In some cases, the WFD onset and LOV were calculated from Monte-Carlo simulations based on flaw size distributions obtained from test or service cracks (Figure 9.7-12). In other instances, these limits were derived from fatigue analyses utilizing scatter factors that were dependent on test article size, fidelity of the loading/boundary conditions and number of replicates. Finally, there were circumstances in which a sufficient number of aircraft accumulated such high numbers of cycles that statistical analysis of the crack-free performance enabled estimates of reasonable, practical limits. The methods of initial flaw size distribution, crack growth interaction, link up, and progression to loss of the susceptible load path simulated by the Monte-Carlo code are described herein, as are the processes for selecting the appropriate scatter factors for analytical estimates and methods for statistical evaluation of high-time uncracked airframes.

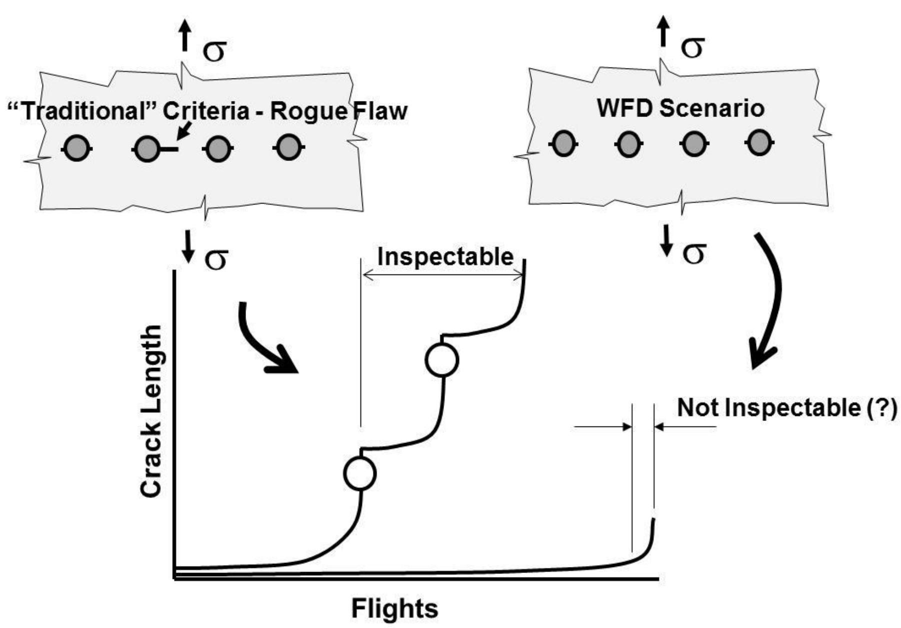


Figure 9.7-10. WFD Susceptible Structure

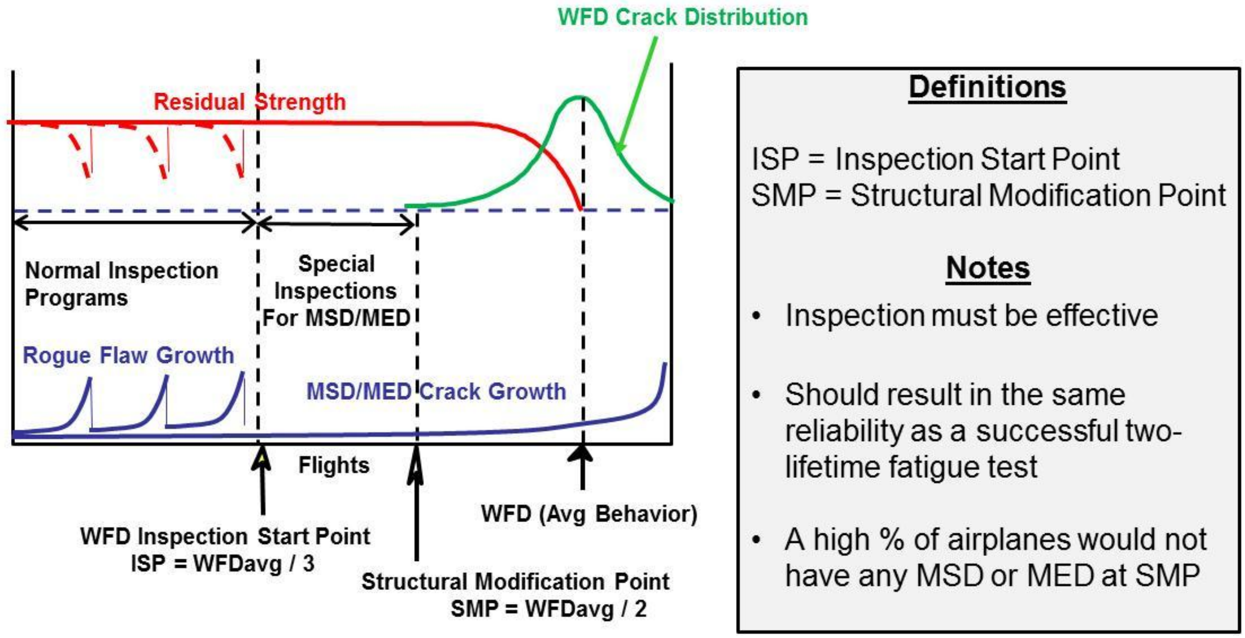


Figure 9.7-11. MSD/MED Residual Strength

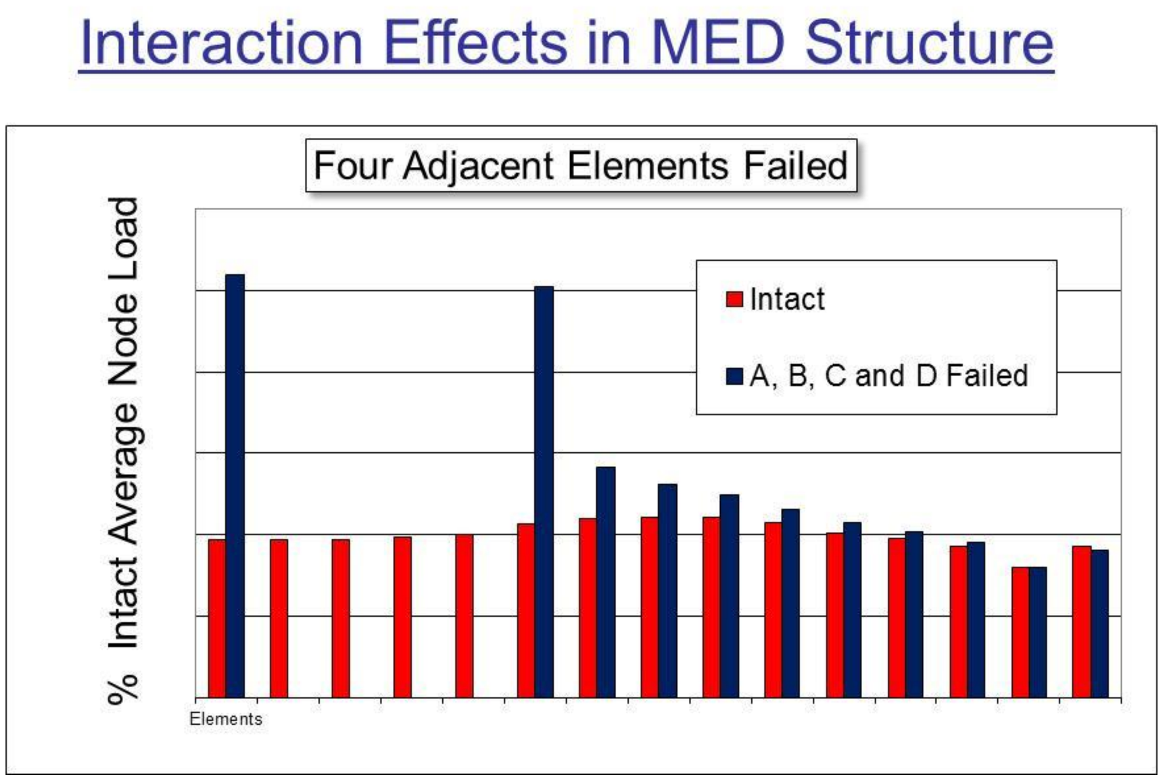


Figure 9.7-12. Monte Carlo – Residual Strength

9.7.5. Recommended Methodology Updates to Improve Single Flight Probability of Failure Estimation

Thomas Brussat, Tom Brussat Engineering, LLC

Aircraft structural reliability is measured in terms of estimated Single Flight Probability of Failure (SFPoF), the conditional probability that the structure, having survived all prior flights, will sustain the next flight without catastrophic failure. Methodology for estimating SFPoF based on the equivalent initial flaw size (EIFS) concept has become well established. A governing equation expresses SFPoF in terms of a deterministic crack growth equation and probability distributions for EIFS, fracture toughness, and residual strength load. The current methodology also incorporates effects of structural inspections and probability of detecting cracks.

To date this methodology remains largely as it was first established in the 1980's. The goal of this technical activity is to outline ready-to-implement developments that would significantly improve the fidelity of SFPoF estimates. The following effects are addressed:

- Effect of non-failure history when solving the governing equation for SFPoF (Figure 9.7-13)
- Effect of inexactness in durability and residual strength analyses (Figure 9.7-14)
- Small data sample effect on estimated probability distributions for EIFS or toughness (Figure 9.7-15)
- Effect of variations in initial crack geometry inherent in metal fatigue behavior (Figure 9.7-16)

Two effects sometimes overlooked in application of SFPoF analysis are also addressed; namely

- Effect of right-censored EIFS data to account for undiscovered cracks (Figure 9.7-17)
- Effect of assuming a conservative "Probability of Inspection" (PoI) (Figure 9.7-18)

The impact on SFPoF of considering or ignoring each of these six effects is shown for a typical example case. Some cause SFPoF to increase; some cause it to decrease. Each introduces more realism into the SFPoF calculation, and any one of these effects can be significant depending on the specific application.

SFPoF(t) is calculated as follows:*

$$\int_0^\infty \int_0^\infty \underbrace{\left[1-H\left(\frac{K_C}{\alpha(a_0,\ t)}\right)\right]}_{\text{[Failure factor]}} \underbrace{\left[\prod_{i=1}^{t-1} H\left(\frac{K_C}{\alpha(a_0, i)}\right)\right]}_{\text{[Survival factor]}} \mathbf{f(a_0)g(K_C)da_0dK_C}$$

[Failure factor] =
Conditional probability of failure in flight t

[Survival factor] =
Probability of surviving all prior flights

- **Legacy method assumes Survival Factor = 1.0.**
- **This attributes past failures to the current flight.**

RECOMMENDATION 1. INCLUDE SURVIVAL FACTOR TO AVOID "DOUBLE-BOOKING" PAST FAILURES

Figure 9.7-13. SFPoF Equation: The Neglected Factor

Recommendation 2. Consider statistical error in durability and residual strength analyses

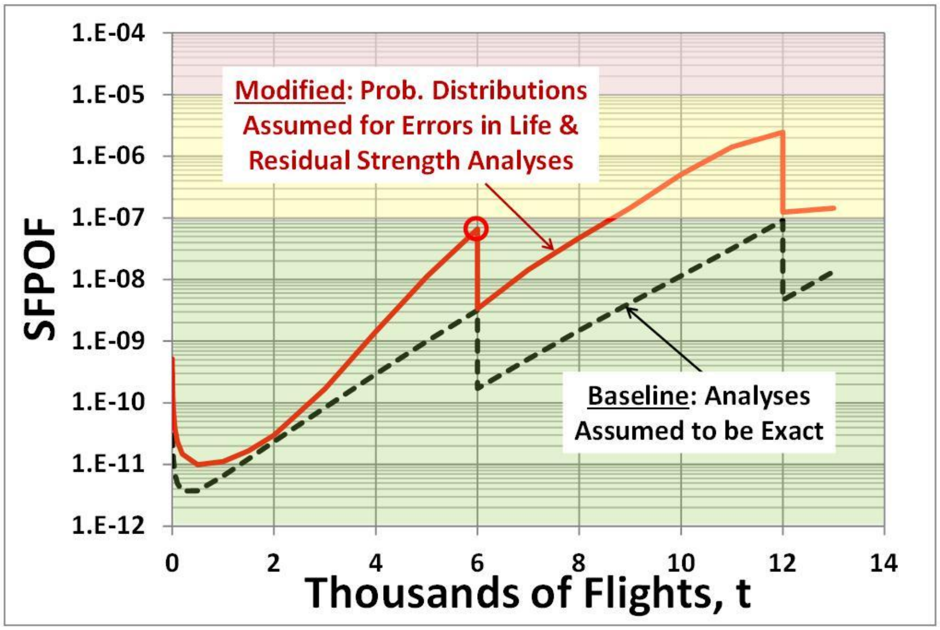


Figure 9.7-14. Inexactness in Durability & Residual Strength Analyses

Recommendation 3. Don't presume the sample and population distributions are identical

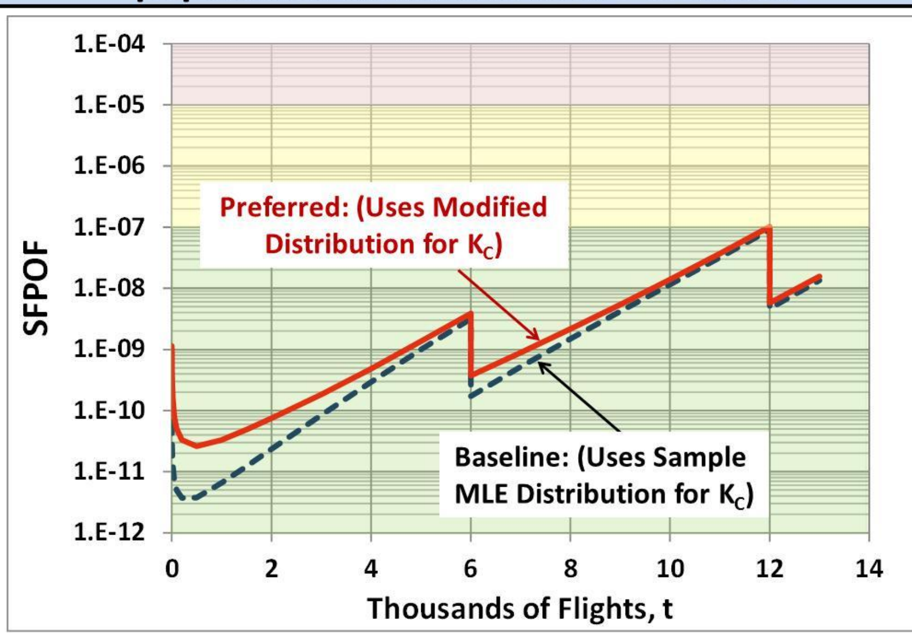
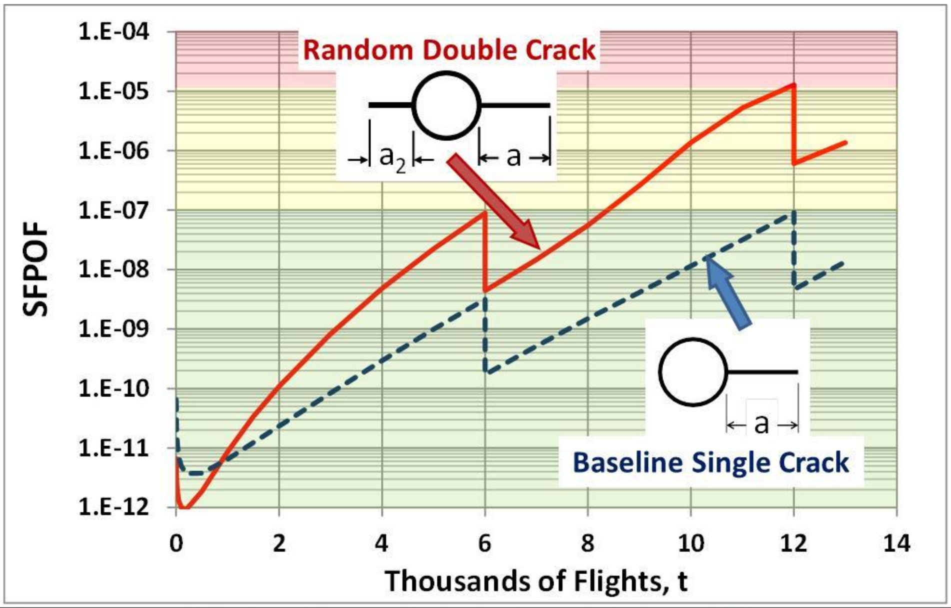


Figure 9.7-15. Is Sample Distribution the Best Choice?



Recommendation 4. Consider all possible crack geometries, each in proportion to its probability of occurrence

Figure 9.7-16. Probability-Based Double Crack at Hole

Recommendation 5. Include undiscovered cracks in EIFS distribution

- **BASELINE EIFS DISTRIBUTION:**
  - **Based only on 25 Cracks found.**
    - **How many A/C inspected?**
    - **How many undiscovered cracks?**
- **EXAMPLE: FLEET SIZE = 350 A/C:**
  - **Assume all 350 A/C inspected.**
  - **25 cracks found; 325 undiscovered cracks.**
- **ESTIMATE UNDISCOVERED CRACKS USING POD:**
  - **For every crack found, expected number of inspected cracks of same size is 1/p, where p = PoI*PoD**
  - **Assume ∑(1/p) = 350**
  - **For smallest 3 cracks:**
    - **1/p > 64 (total would exceed 350).**
    - **Assume 64 cracks each; analyze as "left-censored" data.**

a	Flights	a_0	PoD*PoI	1/p
0.785	**18780**	**3.2E-03**	**0.950**	**1**
0.369	17271	2.3E-03	0.943	1
0.330	18345	1.5E-03	0.939	1
0.191	16181	1.7E-03	0.892	1
0.190	16075	1.7E-03	0.891	1
0.118	9378	7.6E-03	0.775	1
0.110	7718	1.1E-02	0.750	1
0.091	10790	3.9E-03	0.677	1
0.082	12705	2.0E-03	0.628	2
0.075	**8582**	**6.1E-03**	**0.587**	**2**
0.060	13535	1.1E-03	0.472	2
0.053	9765	3.1E-03	0.417	2
0.049	10834	2.1E-03	0.379	3
0.048	8678	3.8E-03	0.364	3
0.040	**19259**	**1.4E-04**	**0.274**	**4**
0.029	16451	2.3E-04	0.152	7
0.026	12292	7.2E-04	0.130	8
0.022	10106	1.1E-03	0.084	12
0.021	14928	2.7E-04	0.081	12
0.020	4020	6.3E-03	0.072	14
0.016	11061	6.3E-04	0.038	26
0.013	**7925**	**1.3E-03**	**0.019**	**53**
0.012	**3865**	**3.8E-03**	**0.015**	**64**
0.007	**5278**	**1.4E-03**	**0.002**	**64**
0.003	**8162**	**3.2E-04**	**0.0001**	**64**
Totals			**10.5**	**350**

Figure 9.7-17. Effect of Including Undiscovered Cracks in EIFS Data

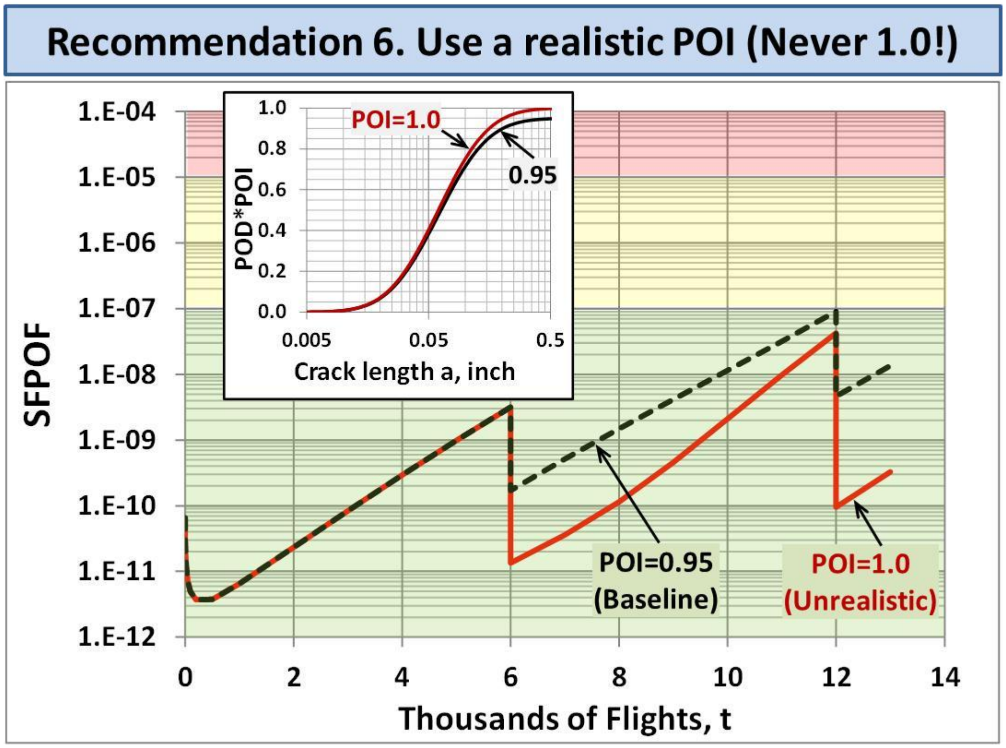


Figure 9.7-18. Use a Realistic PoI

9.7.6. Aircraft Structural Risk and Reliability Analysis Handbook

James L. Rudd, Universal Technology Corporation

The Aircraft Structural Risk and Reliability Analysis Handbook was developed under the sponsorship of the United States Air Forces' Aerospace Systems Directorate. The authors were Robert Bell, Alan Berens, Thomas Brussat, Joseph Cardinal and Joseph Gallagher; the Editor-in-Chief was James Rudd. The objective of the handbook was to provide the analytical capability for establishing probabilities of structural failure by providing a) a tutorial on basic probability and statistics, b) accepted analysis methods, and c) characterization of available input data. The handbook consists of 16 sections and five appendices (Figure 9.7-19).

Section 5.0 entitled Physics of Failure Structural Reliability Analysis is highlighted here. Subsection 5.2 considers the case of a single location with no crack. The prototype problem considered was a wing failure that occurred at 128% Design Limit Load during static testing (Figure 9.7-20). The probability that the wing would fail in service if it is not modified or repaired was determined. Subsection 5.3 considers the case of a single location with a single known crack size. The prototype problem considered was a large crack that was found at an operational base during an inspection. The chances of the structure failing if the aircraft is not flown to a repair depot for the necessary repairs was determined (Figure 9.7-21). Subsection 5.4 considers the case of a single location with a distribution of crack sizes. The risk of delaying a scheduled inspection for 500 hours in an aging fleet was determined (Figure 9.7-22).

Section 6.0 entitled Constructing and Using SFPoF vs. Flight Hours Curve is also highlighted here. The impact of fatigue crack growth, inspections, and repairs on the SFPoF is determined. Figure

9.7-23 presents an example of the impact on the cumulative distribution function a) if a crack is missed during an inspection, b) if a crack is found and repaired during an inspection, and c) of the combined impact of a) and b) above.

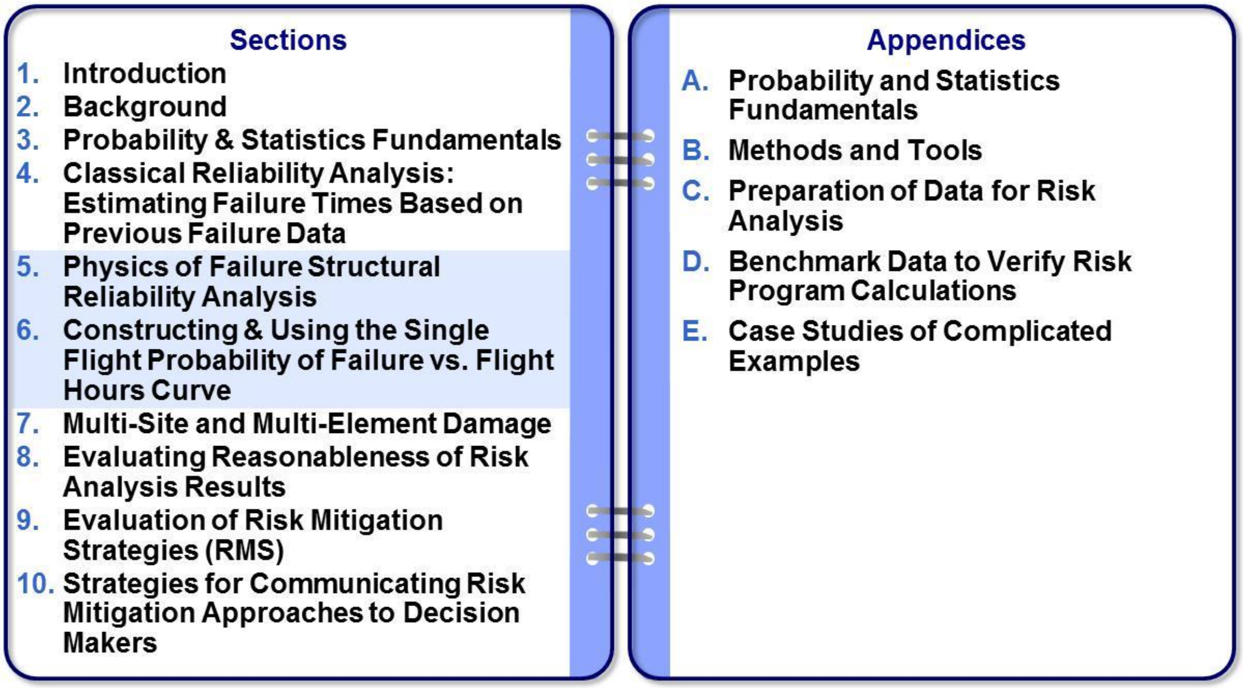


Figure 9.7-19. Table of Contents

Figure 9.7-20. F-16C Block 30 Wing Failure at 128% DLL

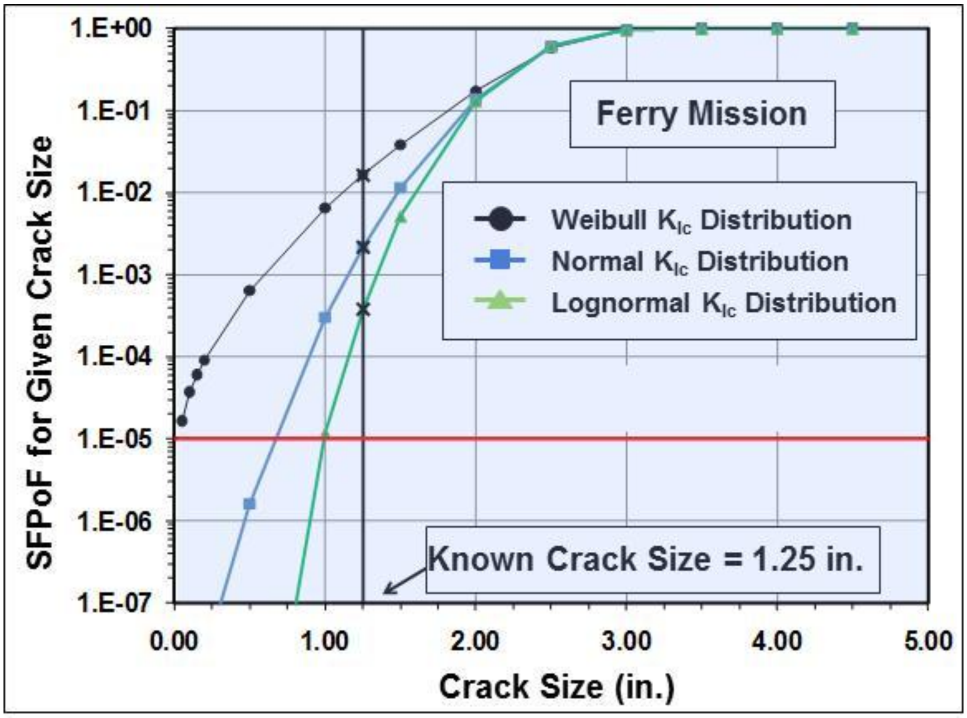


Figure 9.7-21. Single Flight Probability of Failure (SFPoF) for Known Crack Size

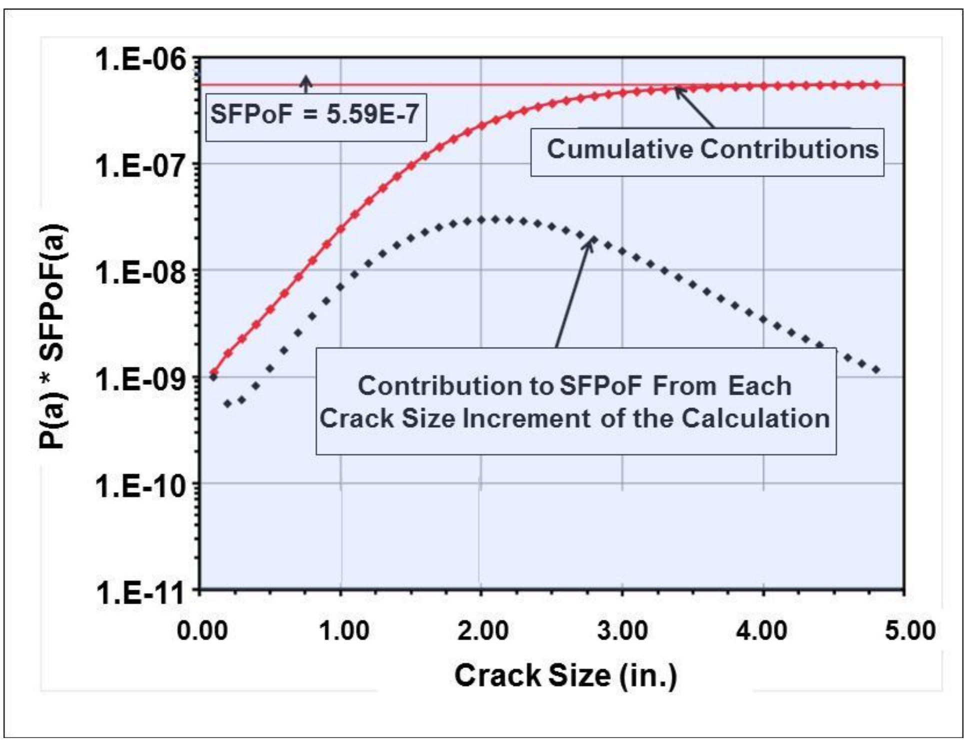


Figure 9.7-22. Cumulative SFPoF for Distribution of Crack Sizes

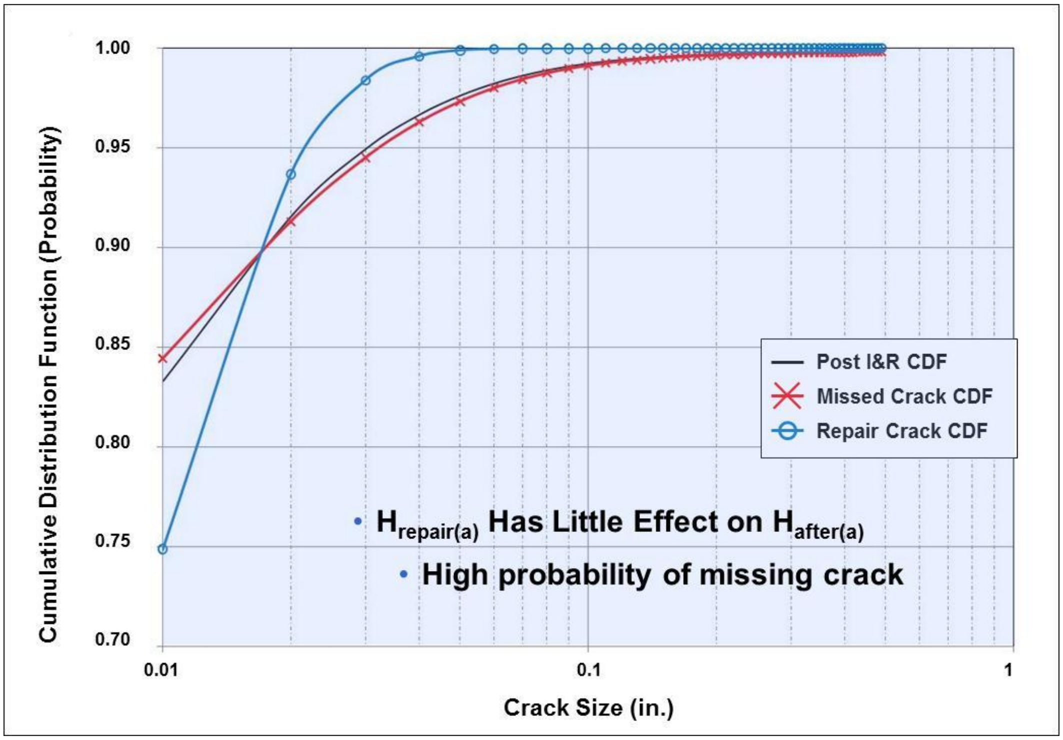


Figure 9.7-23. Post Inspection and Repair Crack Size Distribution

9.7.7. Continued Development of the Darwin Software for Probabilistic Damage Tolerance Analysis and Risk Assessment

Craig McClung, Michael Enright, Yi-Der Lee, Jonathan Moody, and Vikram Bhamidipati, Southwest Research Institute®; Simeon Fitch, Elder Research

DARWIN (Design Assessment of Reliability With INspection) integrates 2D and 3D finite element (FE) models and stress/temperature results, advanced fracture mechanics models, material anomaly data, NDE probability of detection curves, and inspection schedules with advanced probabilistic methods and a powerful graphical user interface (GUI) to determine the probability of fracture of a component as a function of operating cycles, with and without inspections. Originally developed (with substantial funding from the FAA) to address specific threats to the integrity of high energy rotating components in aircraft engines, DARWIN now includes general deterministic and probabilistic damage tolerance capabilities relevant to many applications, including airframes. DARWIN has been under continuous development since 1995, and recent advances have significantly enhanced its ease of use, efficiency, and accuracy.

DARWIN development activities during 2011-2012 focused on Versions 7.2 and 8.0.

DARWIN 7.2 includes an initial capability for automatic generation of zones for risk assessment of components with inherent material anomalies. The analyst assigns component properties (e.g., anomaly distributions, material properties, inspection schedules) directly to FEs. Once the properties are assigned, DARWIN automatically generates a zone at each FE in the user-supplied FE model. The orientation and boundaries of the life model for each zone are then computed using the automatic fracture model process introduced in v7.0. In DARWIN 8.0, the v7.2 autozoning capability was enhanced to enable multiple FEs in each zone and to identify the optimal placement of FEs in the zones to minimize the total number of zones required for a converged risk estimate. The number of zones (and associated computation times) can be dramatically reduced using the optimal autozoning algorithm.

DARWIN 7.2 also includes an initial capability for time-dependent fatigue crack growth (FCG) assessment, applicable to environmentally-influenced crack growth. The user is allowed to provide the elapsed time associated with each load step of the flight history. The crack growth life is computed using a superposition of the cycle-dependent and time-dependent crack growth rates.

DARWIN 8.0 added a new capability for overload crack growth retardation effects. This feature is based on a modified Willenborg retardation model. The new feature enables the analyst to include crack retardation effects for both cycle- and time-dependent crack growth life and risk assessments.

Computational efficiency is a critical aspect when performing risk assessment where millions of numerical simulations are often required to satisfy computational accuracy requirements. To address this issue, a new parallel processing capability was introduced in v7.2 that automatically subdivides the risk computation for simultaneous application to multiple CPUs on a single computer. The new parallel processing capability can substantially reduce the computation time required for risk assessment.

A new feature was added in v7.2 and expanded in v8.0 for modeling residual stress (RS) associated with surface treatment applications (e.g., peening). RS profiles can currently be applied directly to surface, corner, and embedded cracks in 2D finite element models, and this will be extended to 3D models in the future. The user can define and view RS profiles directly in the GUI. The residual stresses are combined with service stresses for FCG life and risk assessment using superposition.

A new bivariant stress intensity factor solution (SC29) was added in v8.0 for a semi-elliptical surface crack at an off-center hole. The SC29 solution can be used at locations where the stress gradient varies through the thickness as well as away from the hole.

In previous versions of DARWIN, XML-formatted files were used for storage of input and results data. This format was adequate for the small amount of data typically associated with routine risk assessments. However, as DARWIN capabilities have expanded, and as analysts have developed FE models with significantly more elements and load steps, the resulting file sizes have become difficult to manage using XML. To resolve these issues, a new HDF5 file format was recently implemented in DARWIN. HDF5 is a binary hierarchical file format specifically designed for complex high volume data. It supports direct random access to specific locations within a file without the need to load the entire file into memory, which significantly reduces memory requirements. HDF5 organizes data in a tree-like, hierarchical structure that is similar to the data structure used for XML-formatted data. This structure makes it convenient to navigate large files with complex data. Use of the HDF5 file format in DARWIN has significantly reduced the amount of computer memory required for execution, particularly for large files.

In development activities funded by the US Air Force Research Laboratory, interfaces were developed between DARWIN and the manufacturing process simulation software DEFORM™. These interfaces permit full-field results from manufacturing process simulations to be incorporated in predictions of fracture life and reliability. In particular, approaches were developed for modeling the effects of location-specific bulk residual stress and average grain size on crack growth behavior and fracture risk. Demonstration examples showed the practical potential for Integrated Computational Materials Engineering (ICME) to directly address component integrity and reliability.

Figure 9.7-24 shows stress contours, FCG life contours, and risk contours in DARWIN with and without the effects of bulk residual stresses arising from the manufacturing process after forging, heat treating, and machining.

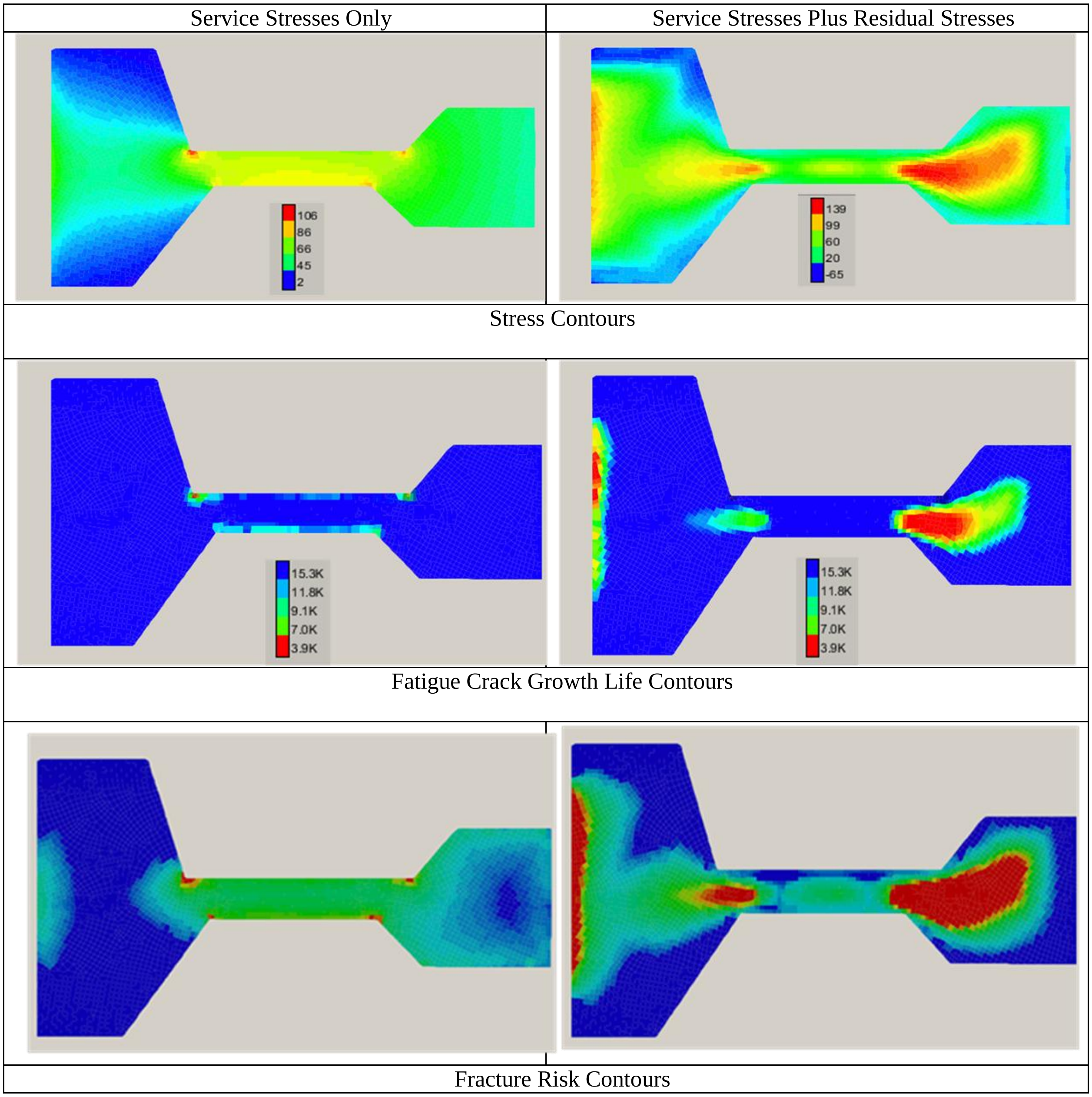


Figure 9.7-24. Illustration of the Influence of Manufacturing-Related Residual Stresses on the Overall Stress State (Top), the FCG Life (Middle), and the Component Risk of Fracture (Bottom)

For more information:

[1] "New Methods for Automated Fatigue Crack Growth and Reliability Analysis," R. C. McClung, Y.-D. Lee, M. P. Enright, and W. Liang, Paper GT2012- 69121, *Proceedings of ASME Turbo Expo 2012, C*openhagen, Denmark, June 2012.

[2] "A Tool for Probabilistic Damage Tolerance of Hole Features in Turbine Engine Rotors," by M. P. Enright, R. C. McClung, W. Liang, Y.-D. Lee, J. P. Moody, and S. Fitch, Paper GT2012-69968, *Proceedings of ASME Turbo Expo 2012, C*openhagen, Denmark, June 2012.

[3] "Integration of Manufacturing Process Simulation with Probabilistic Damage Tolerance Analysis of Aircraft Engine Components," by R. C. McClung, M. P. Enright, W. Liang, J. Moody, W.-T. Wu, R. Shankar, W. Luo, J. Oh, and S. Fitch, S., Paper AIAA 2012-1528, *Proceedings of 53rd Structures, Structural Dynamics and Materials Conference*, Honolulu, Hawaii, April 2012.

More information about DARWIN is available at www.darwin.swri.org.
POC: Craig McClung, Southwest Research Institute, Craig.McClung@swri.org, 1-210-522-2422.

9.8. LIFE ENHANCEMENT CONCEPTS

9.8.1. Full-Scale Component Tests to Validate the Effects of Laser Shock Peening

LeAnn Polin and Jeffrey Bunch, The Boeing Company-Defense, Space & Security; Peter Caruso, Lockheed Martin Corporation; and John McClure, USAF F-22 Program Office

Laser Shock Peening (LSP) is a technology with the potential to enhance fatigue life on metallic components, including primary, fracture critical, airframe structure. The capability and reliability of LSP technology has advanced to the point where it is currently being implemented as a structures retrofit on the United States Air Force F-22 program to extend aircraft service life on the wing attachment lugs, a flight critical component for holding on the aircraft wings. This is the first application of LSP for the Air Force on thick titanium structure and the first airframe application of LSP on operational aircraft. As such, the F-22 program executed an extensive and structured test plan. The durability and damage tolerance focused plan followed a scale-up method based on the Aircraft Structural Integrity Program (ASIP) building block approach, culminating in a series of full-scale component tests (Figures 9.8-1 through 9.8-3). These tests are designed to: 1) match the full-scale fatigue aircraft test results, 2) verify the durability benefit of LSP, 3) determine if the maintenance inspection schedules can be extended based on LSP crack growth results and 4) eliminate the risk of any unknown phenomena from occurring due to unanticipated effects of the LSP process on full-scale components.

This technical activity presents the analysis predictions and the results of full-scale component tests, serving as the final step in the process to certify the life improvement achieved by laser shock peening, as applied to the F-22 airframe. These full-scale test components validate the applied residual stress benefits from LSP previously measured on small test blocks, representative geometry blocks, and sub-component specimens. The analysis results demonstrate how the predicted residual stress field is incorporated into the life predictions, and test data are presented demonstrating the correlation of prediction to test. The data presented conclusively demonstrates how the durability life improvement seen on sub-component specimens translate to full-scale structure.

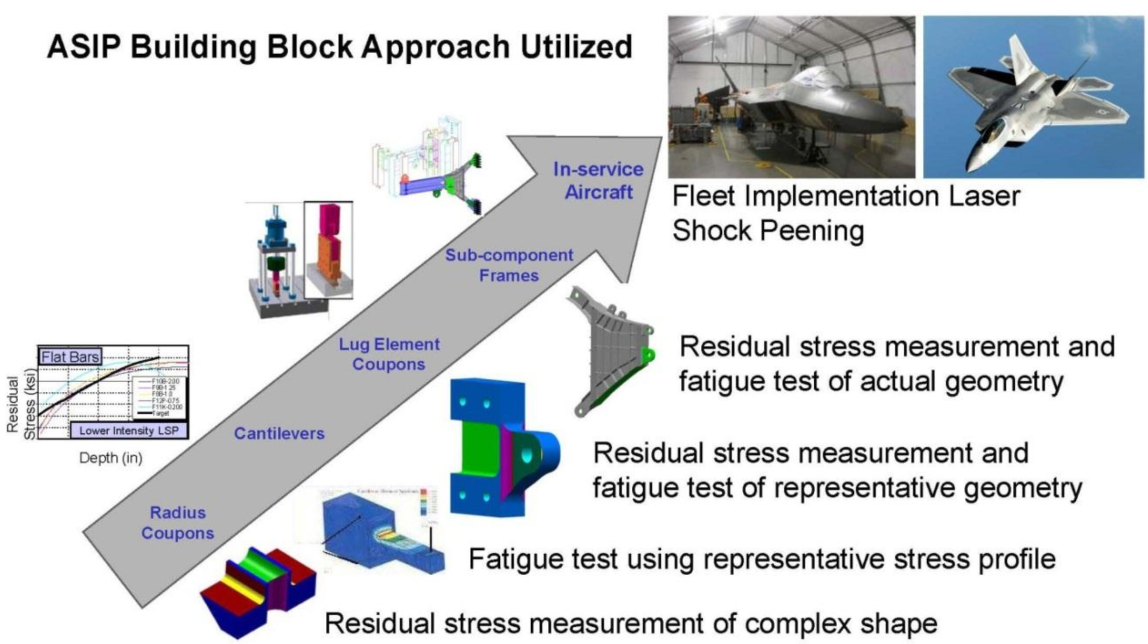
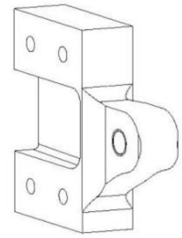


Figure 9.8-1. Test Layout

- **Fatigue Testing of Lug Elements showed:**
  - **Improvement in crack initiation time**
  - **Better than expected improvement in crack growth**
  - **LSP over GBP provides 3x greater benefit than GBP for crack initiation**



Weibull Analysis Benefit Factors (t = flight hours)	
GBP (t>0)	*9.0*
GBP (t=0)	*6.1*
LSP over GBP (t>0)	*30.2*
LSP over GBP (t=0)	*19.2*

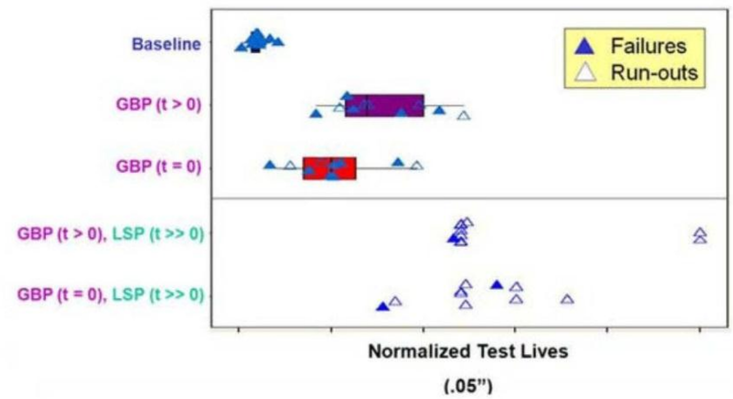


Figure 9.8-2. Lug Element Findings

- **Traditional analytical methods under predict crack growth life**
  - **Need for better analysis tools for complex geometry**
- **Demonstrates potential for fleet inspection relief**

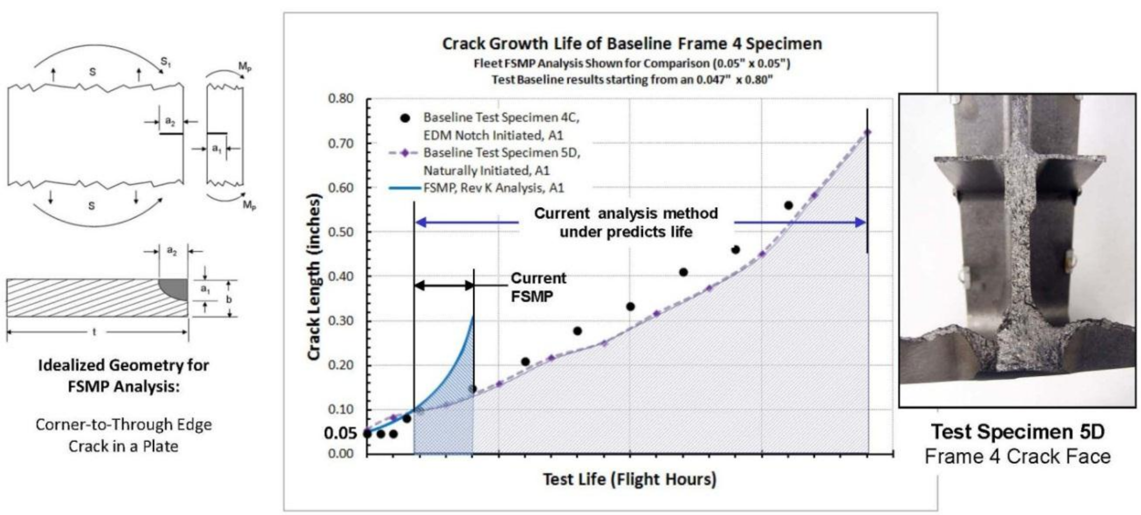


Figure 9.8-3. Baseline Test Results

9.8.2. Cold-Expansion of a Fastener Hole in a T-38 Steel Dorsal Longeron

Whitney Ponzoha, Matthew Hammond and James Greer, USAF Academy-CAStLE; Chad King, USAF-OO-ALC

Cold-expansion (CX) of a fastener hole in a steel fuselage longeron is shown to significantly retard hole-bore crack growth regardless of whether the flaw was pre-existing or introduced after CX (Figures 9.8-4 and 9.8-5). Multiple representative specimens were fabricated from 4340 steel, heat treated, and tested. All of the bore-cracked CX specimens were tested to a minimum of 3,000 equivalent flight hours (EFH) of severe spectrum loading, and none showed significant crack growth. Two other CX specimens were tested to 10,000 EFH and 14,000 EFH respectively with no significant crack growth. The only CX specimen taken to complete ligament failure withstood approximately 19,200 EFH of severe spectrum loading prior to ligament failure. The average life of a non-CX, bore-cracked specimen was about 900 EFH. An additional goal of the project was to develop beta factors and shut-off overload ratios for use in AFGROW at the CX hole, but the inability to grow cracks from these holes frustrated attempts to generate betas and SOLR corrections. However, beta corrections were successfully developed for this location in non-CX holes. In addition, varying the SOLRs used for non-CX holes showed no definitive evidence for changing the current value of SOLR used for DTA at this location (Figures 9.8-6 and 9.8-7). An interesting aspect of the cold-expansion at this hole is the residual tensile stress created at the nearby free edge by the CX process. This was cited as a concern early in the program. Indeed, experiments showed the hole CX reduced the life of the ligament between the hole and the nearby free edge by about 25% provided a 0.050 x 0.050in corner crack introduced at the edge, but in none of the test specimens did a natural crack ever nucleate at the free edge. Some nonlinear (in terms of both material and displacements) FEA was done to support the work as well, simulating the residual stresses created by cold expansion. The geometry near the hole is complicated by the presence of two satellite holes used to hold a nut plate. The FEA confirms the presence of the zone of residual tensile stress at the longeron free edge nearest the hole. This work was performed by the U.S. Air Force Academy Center for Aircraft Structural Life Extension (CAStLE) under a contract with Valdez International Corporation. The program was sponsored by the T-38 ASIP Office (Figure 9.8-8).

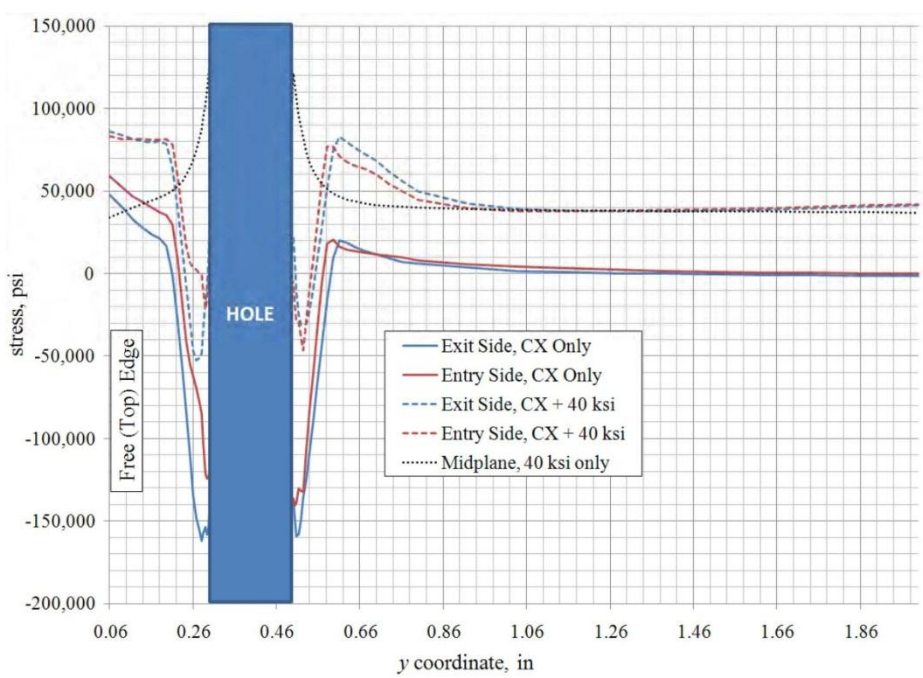


Figure 9.8-4. Residual Stress Data for Cold-Expanded Hole

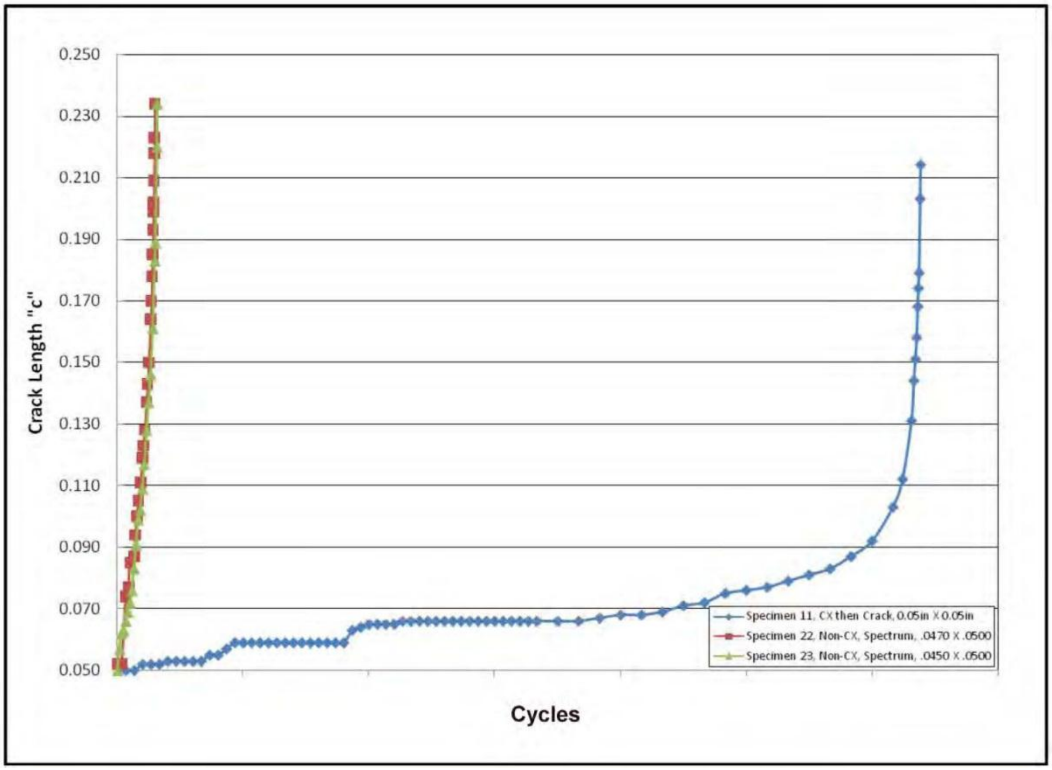


Figure 9.8-5. Crack Length vs. Cycles for Cold-Worked and Non-Cold-Worked Holes

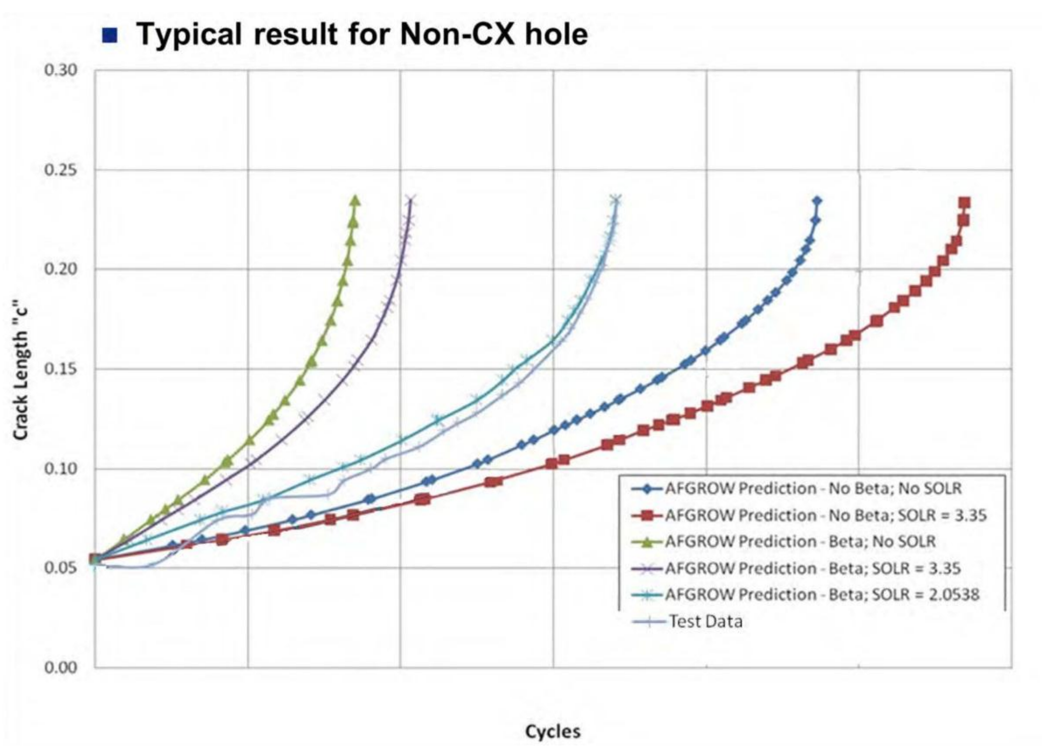
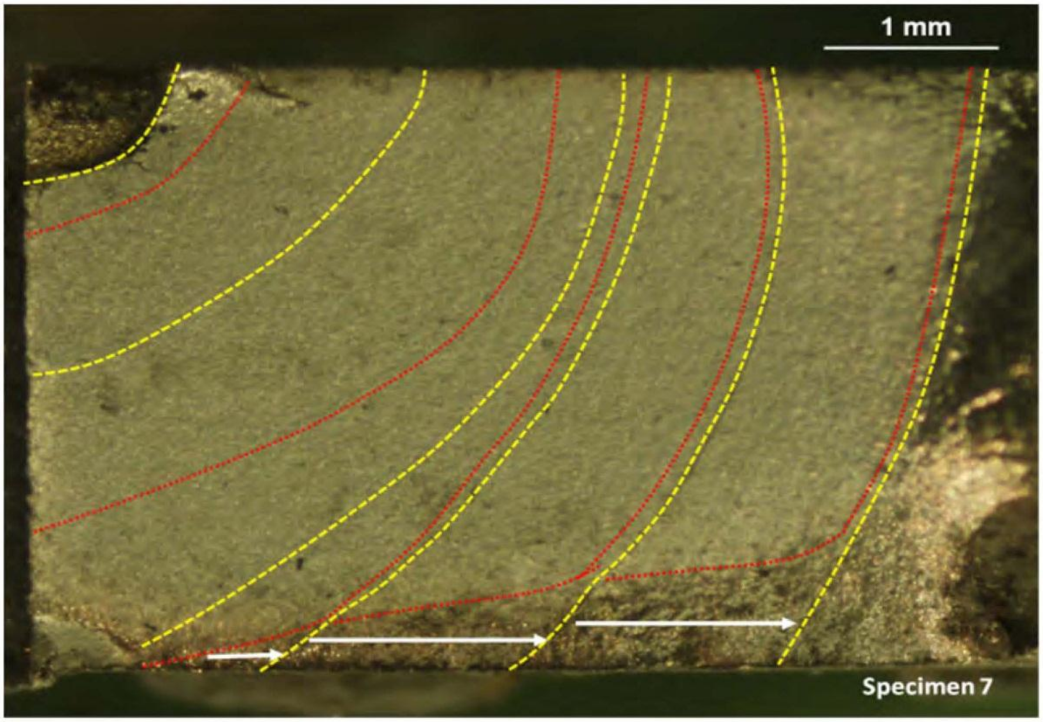


Figure 9.8-6. Analytical vs. Experimental Correlations for Several Shutoff Overload Ratios



White arrows indicate regions where peak spectrum loads caused sudden advancement of part of the crack front.

Figure 9.8-7. Crack-Front Shapes

Figure 9.8-8. T-38 Aircraft

9.8.3. The Use of Interference-Fit (Cold-Expanded) FTI ForceMate® Bushings to Repair Cracking in Primarily Compression-Loaded Bolt Holes

Bryan Nelson, Raul Macias, Keith Sundstrom, Tim Jeske, Selen Minarecioglu and Matthew Edghill, Lockheed Martin Corporation and Beverly Franada, Fatigue Technology, Inc.

The F-16 center fuselage carry-through bulkhead upper outboard flanges (Figure 9.8-9) are subjected primarily to compressive loading due to wing-up flight conditions. The presence of these high compressive loads causes local yielding of the aluminum and deformation of the upper flange bolt holes which mate the closure beam and outer skin to the bulkhead. Cracking was first observed in full-scale-durability testing and has also been reported in multiple F-16 service aircraft. Tensile residual stresses are responsible for the initiation and propagation of these cracks in a compression dominant load environment. A test program was initiated to determine if the use of interference-fit FTI ForceMate® bushings, which are cold-expanded into the holes, could repair this location and provide a service life improvement to the bulkhead flange. The introduction of additional compressive loads due to cold expansion would seem counterproductive, but it was believed that the associated hole propping effects of the interference-fit bushings would provide a significant improvement. Two separate bushing outer diameters were chosen to allow for clean-up of the current known service cracks. Additionally, testing was conducted on two different coupon specimen configurations to account for drawing fastener hole tolerances (Figure 9.8-10). Once testing had verified the significant service life benefit of the ForceMate® bushings, a non-linear elastic-plastic service life analysis was performed to validate/correlate the findings. This technical activity outlines the field experience, challenges, and final results of this testing and evaluation program (Figures 9.8-11 through 9.8-13).

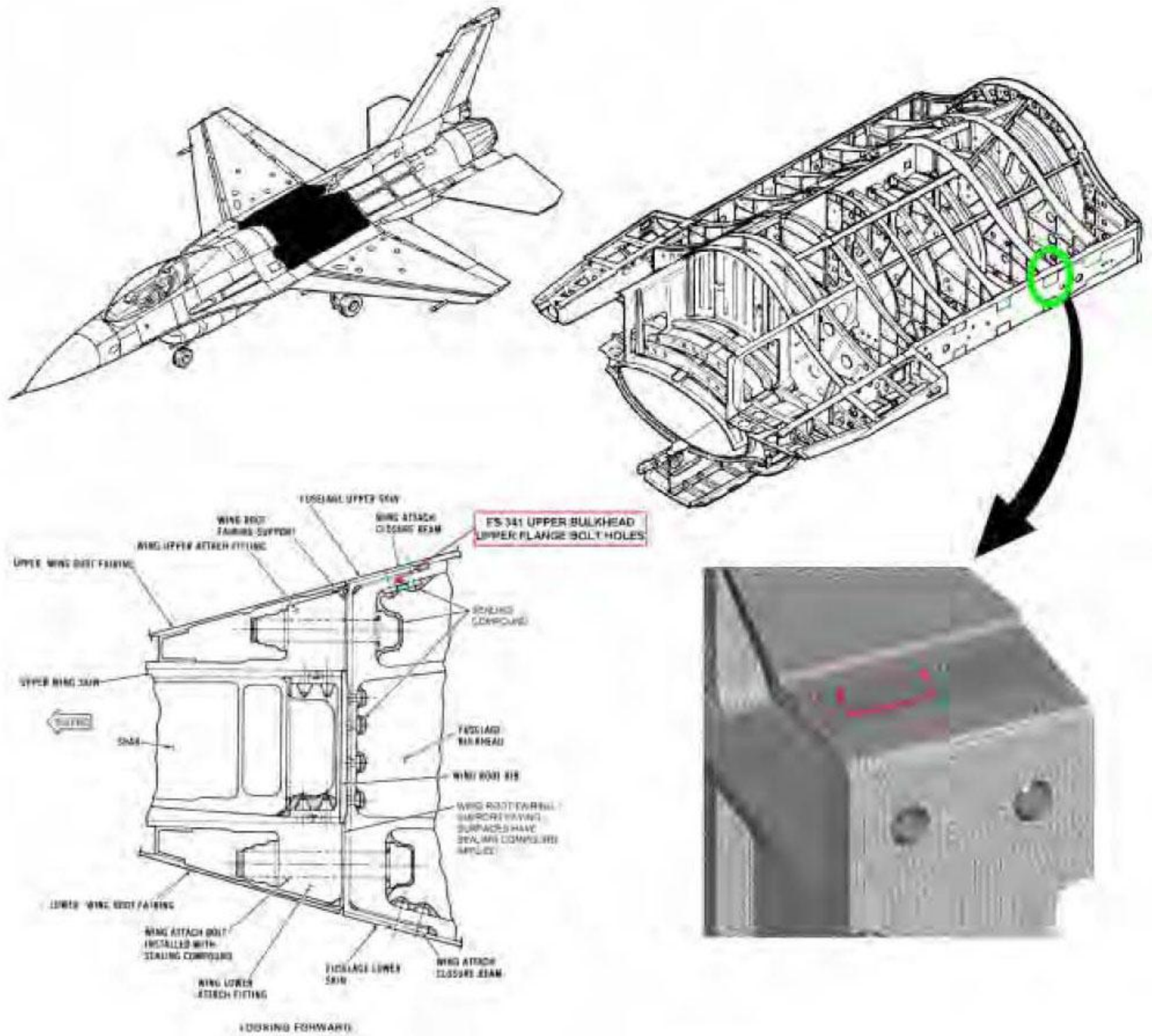


Figure 9.8-9. F-16 Center Fuselage Carry-Through Bulkhead Upper Outboard Flanges

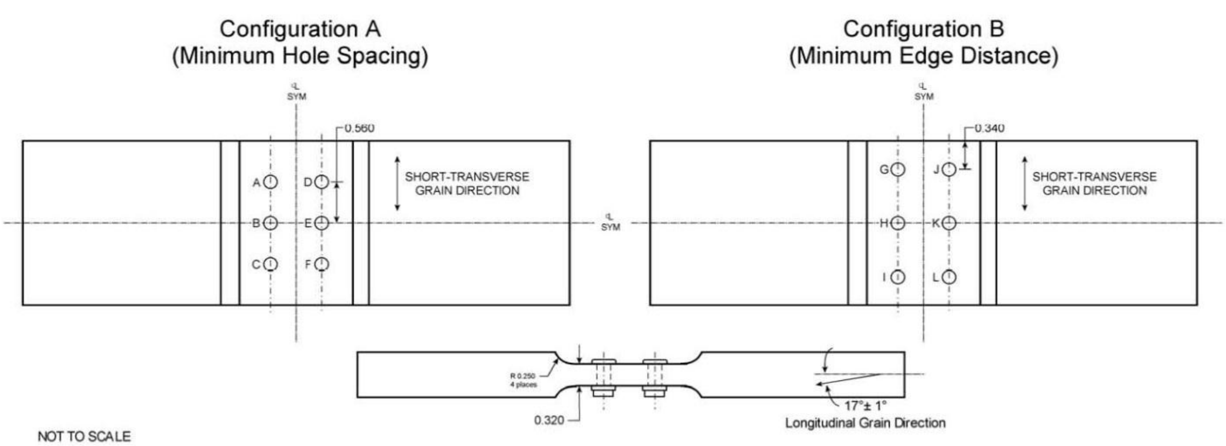


Figure 9.8-10. Test Specimen Configuration

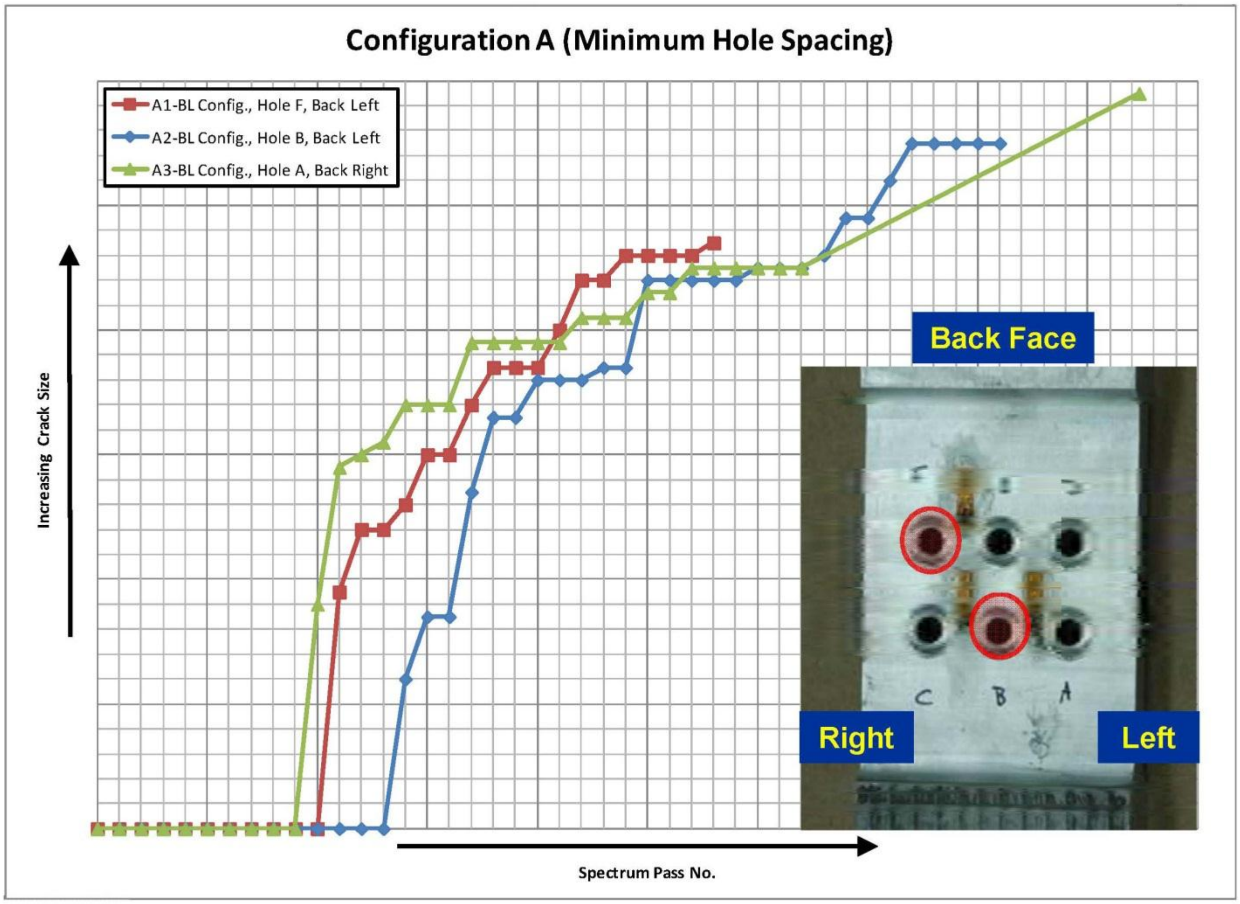


Figure 9.8-11. Test Results for Configuration A

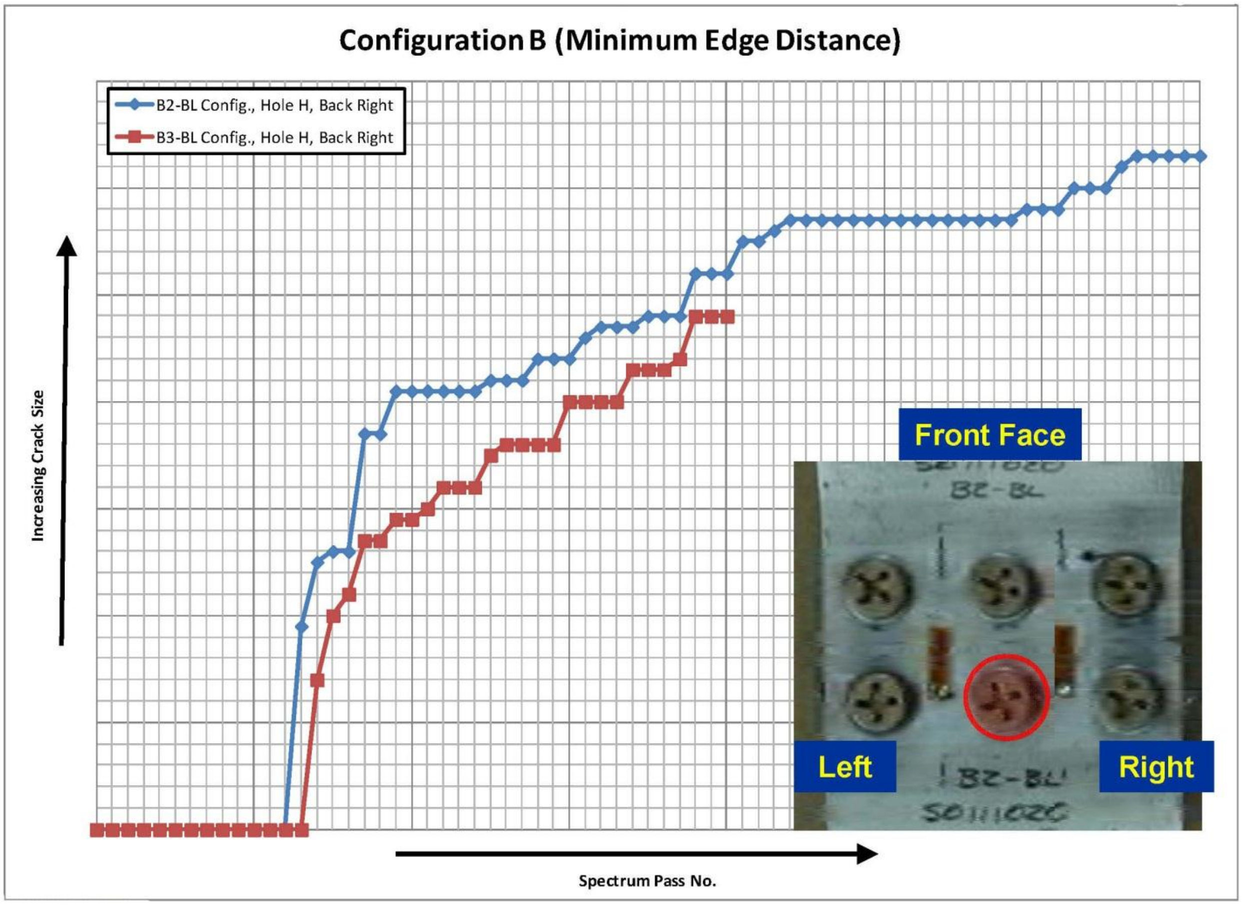


Figure 9.8-12. Test Results for Configuration B

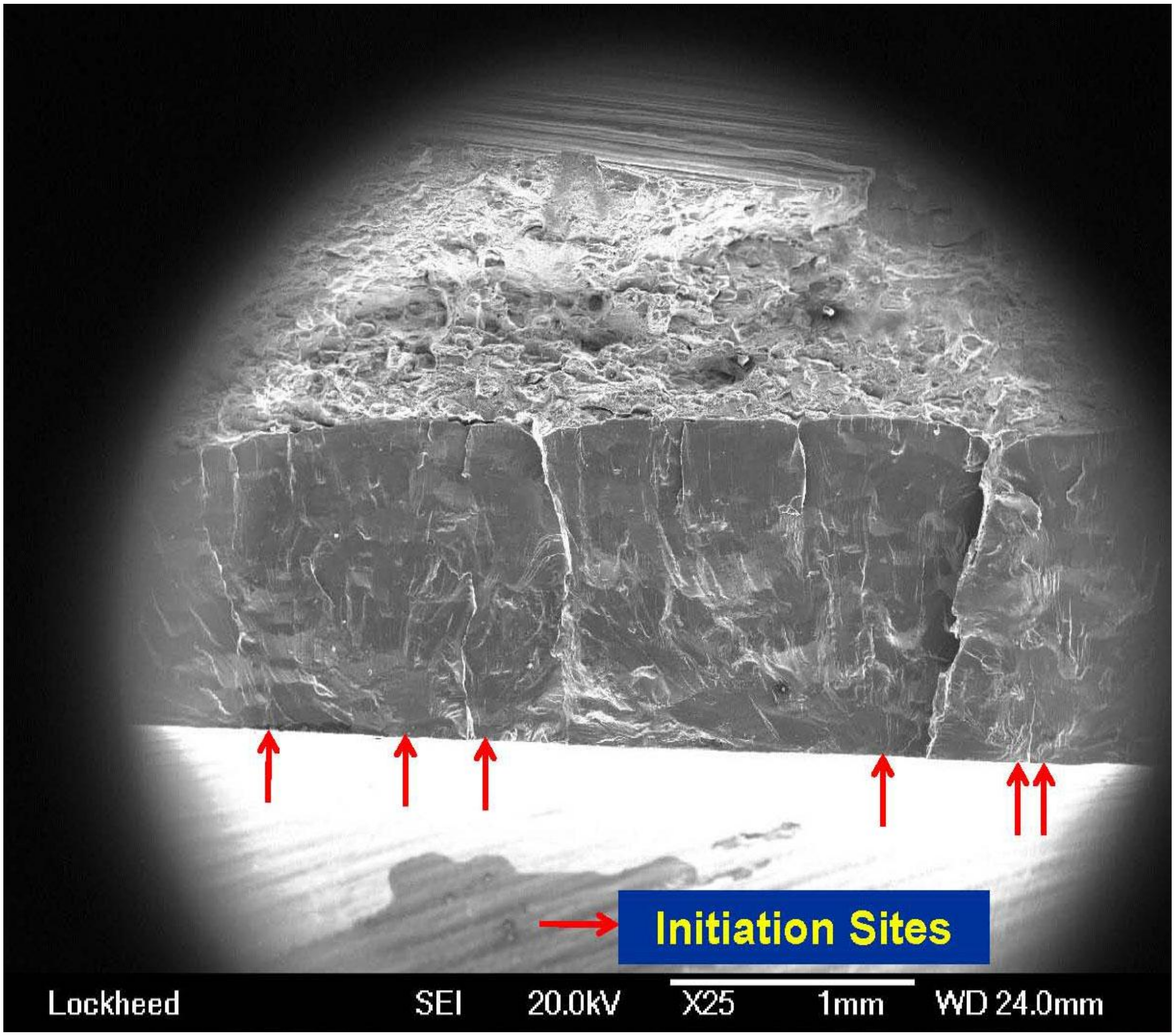


Figure 9.8-13. Test Fractopgraphy-Multiple Fatigue Crack Initiation Sites

9.8.4. Design and Analysis of Engineered Residual Stress Surface Treatments for Enhancement of Aircraft Structure

Michael Hill, Adrian DeWald, and John VanDalen, Hill Engineering, LLC; Jeff Bunch, The Boeing Company; Stephanie Flanagan and Kristina Langer, USAF Research Laboratory – Aerospace Systems Directorate

It is well established that compressive residual stresses provide improved fatigue performance and damage tolerance enhancement. To take advantage of this concept, many surface treatment processes have been developed over the past 60+ years that are capable of imparting compressive residual stress into the surface layer of a component (e.g., shot peening, cold working of fastener holes, and laser shock processing). The compressive stress near the material surface acts to slow the growth of fatigue cracks and can provide substantial benefits (e.g., longer inspection intervals, higher safety margins, reduced sustainment costs, and increased aircraft availability).

Historically, residual stress surface treatments have been developed on a case-by-case basis using an approach that is primarily based on experimental iteration. While often effective, this experimental approach is typically expensive and long in duration. Recent advancements in engineered residual stress analysis tools have shown significant potential to streamline the design process (Figures 9.8-14 through 9.8-16). This technical activity provides an overview of an analytical-based approach for design and engineering of residual stress surface treatments to improve performance of aircraft structure. The approach includes a model to predict residual stresses from surface treatment processes and subsequent fatigue analysis tools (Figure 9.8-17). The analytical approach is compared with experimental data from a recent Air Force program to enhance the performance of F-22 structure using laser shock processing (Figures 9.8-18 and 9.8-19).

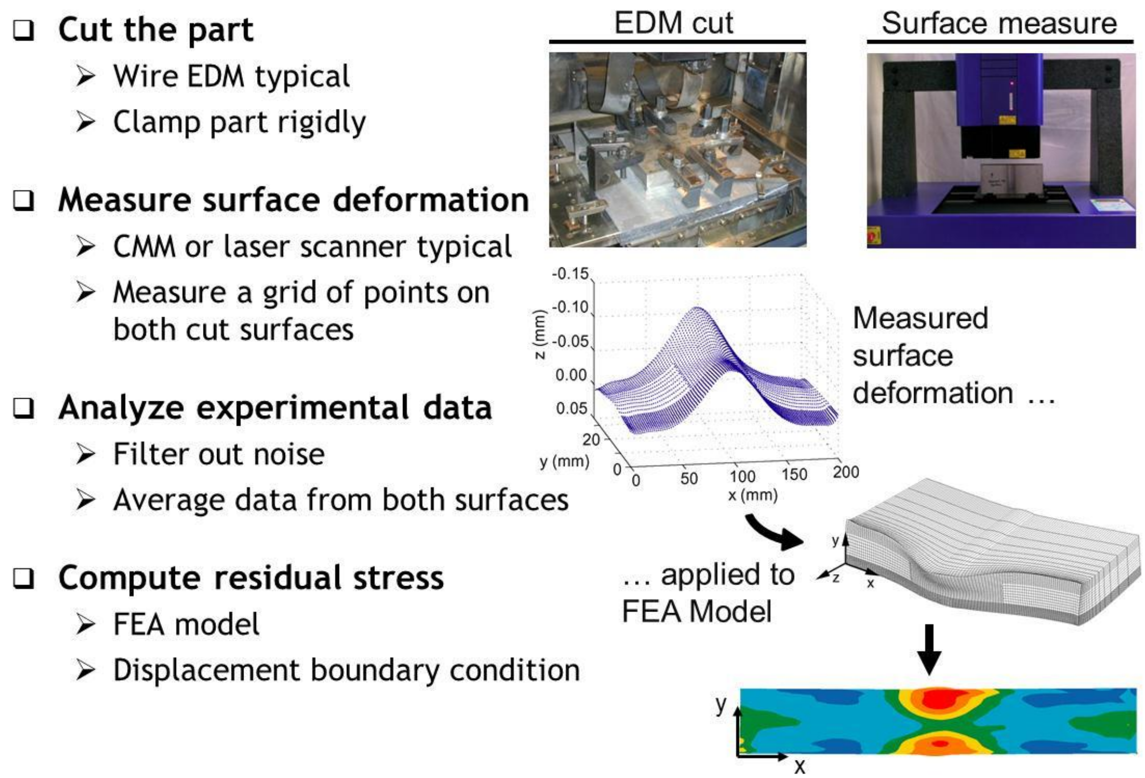


Figure 9.8-14. Contour Method Example

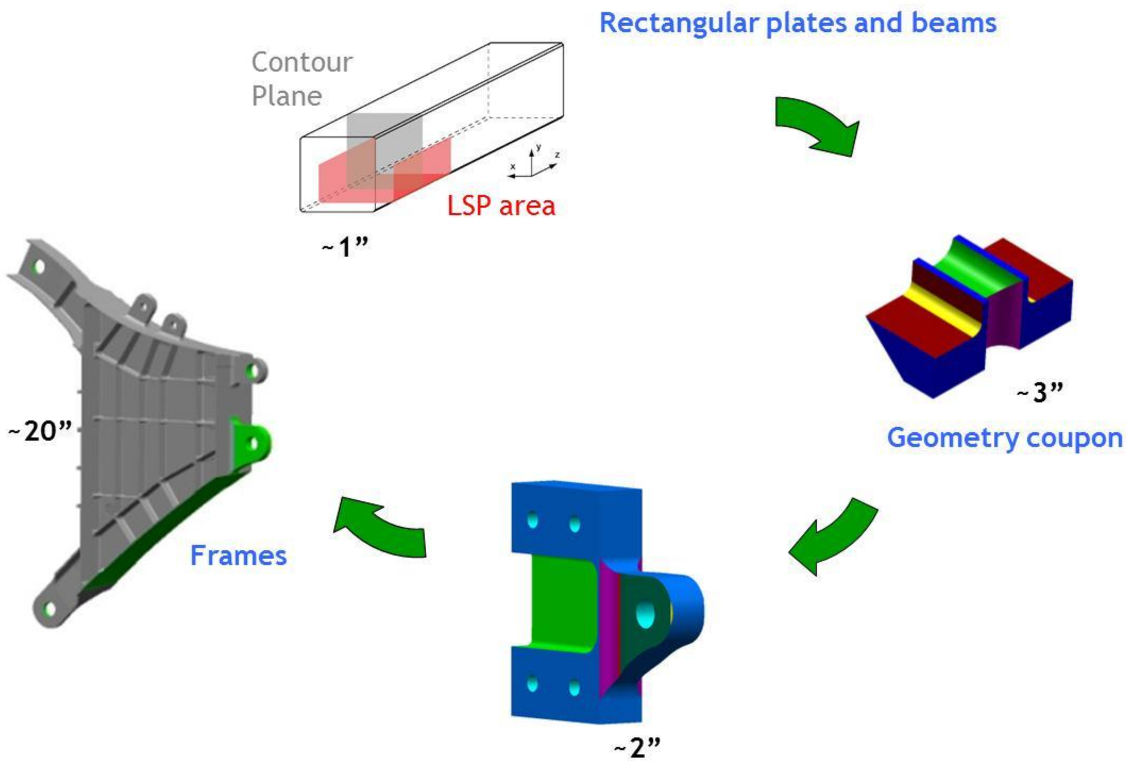


Figure 9.8-15. Contour Application for F-22 LSP Program

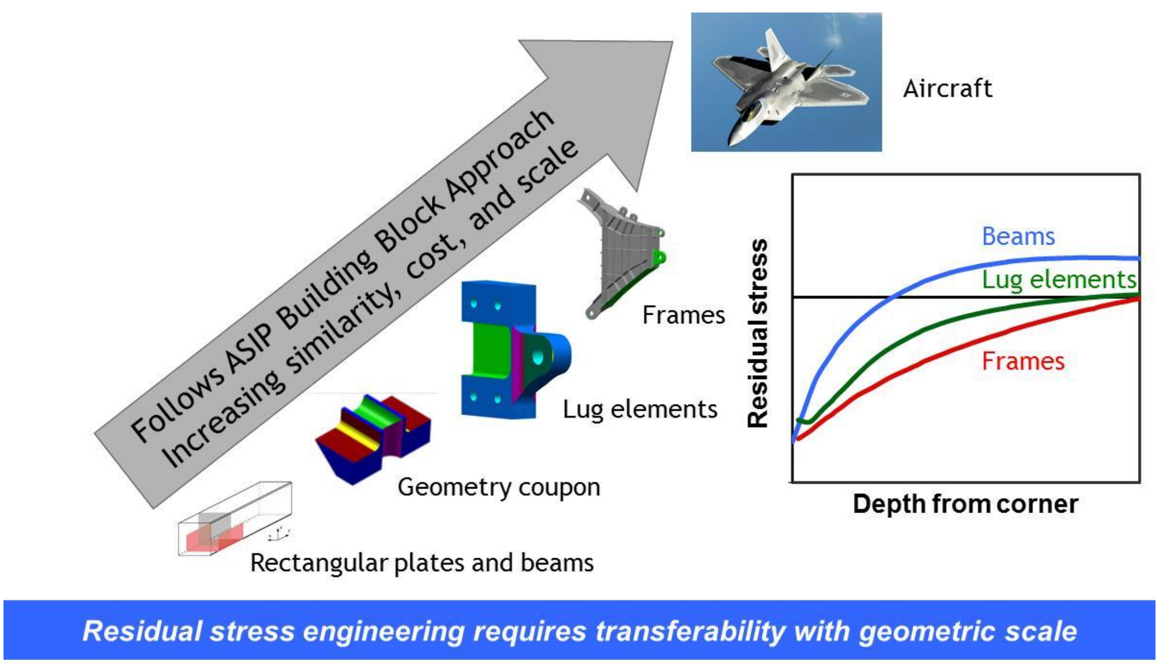


Figure 9.8-16. Contour Results: Residual Stress Changes with Geometry

- Growth occurs on same plane in Baseline and LSP models
  - Baseline prediction: planar, roughly quarter-elliptical shape
  - LSP prediction: planar, bulging shape
- Similar behavior for all Frames

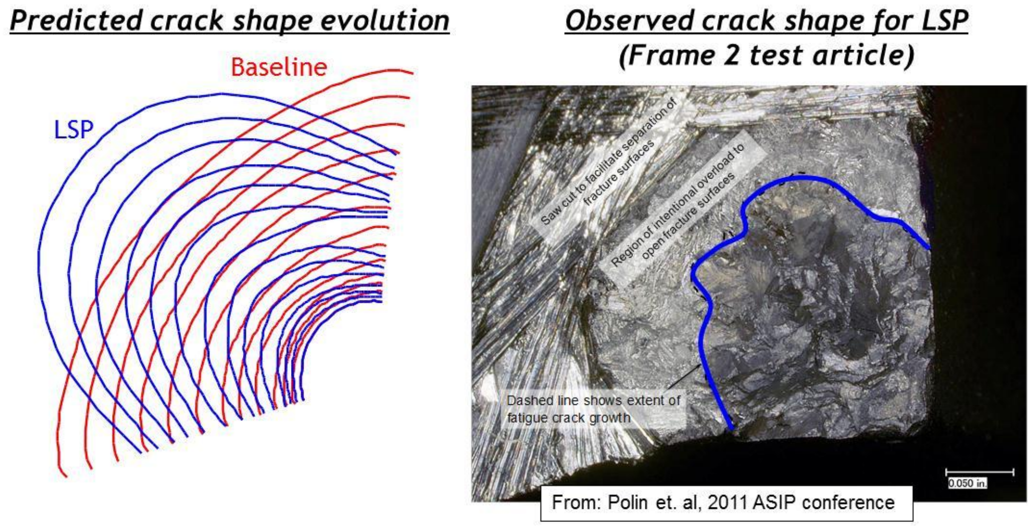


Figure 9.8-17. Predicted Crack Growth Behavior in Frames

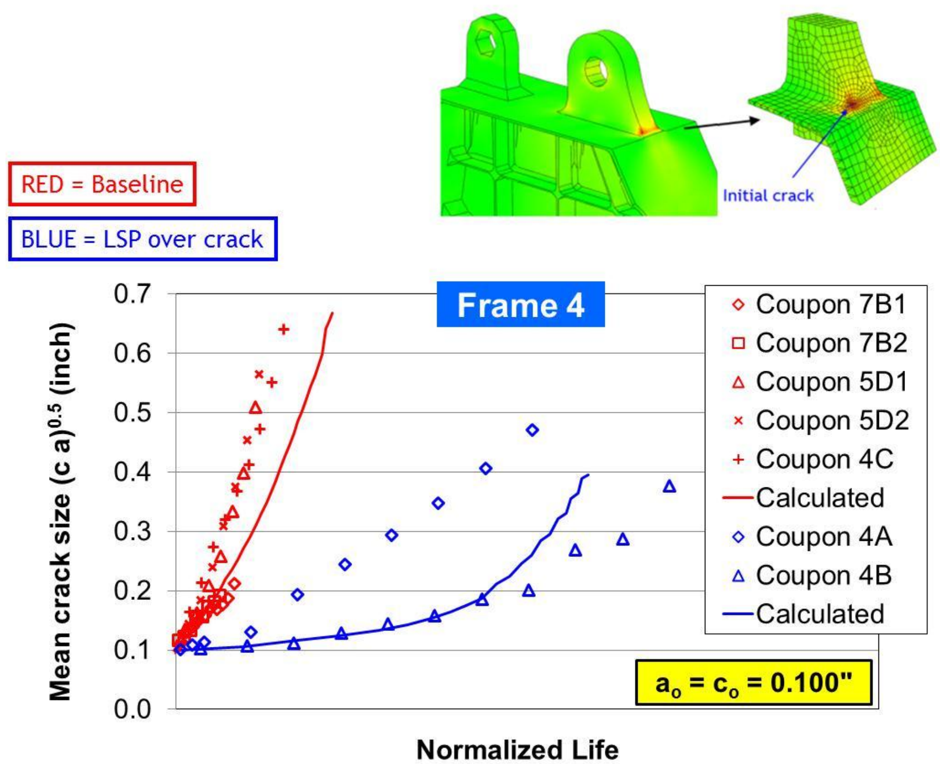


Figure 9.8-18. Frames 4 Correlation

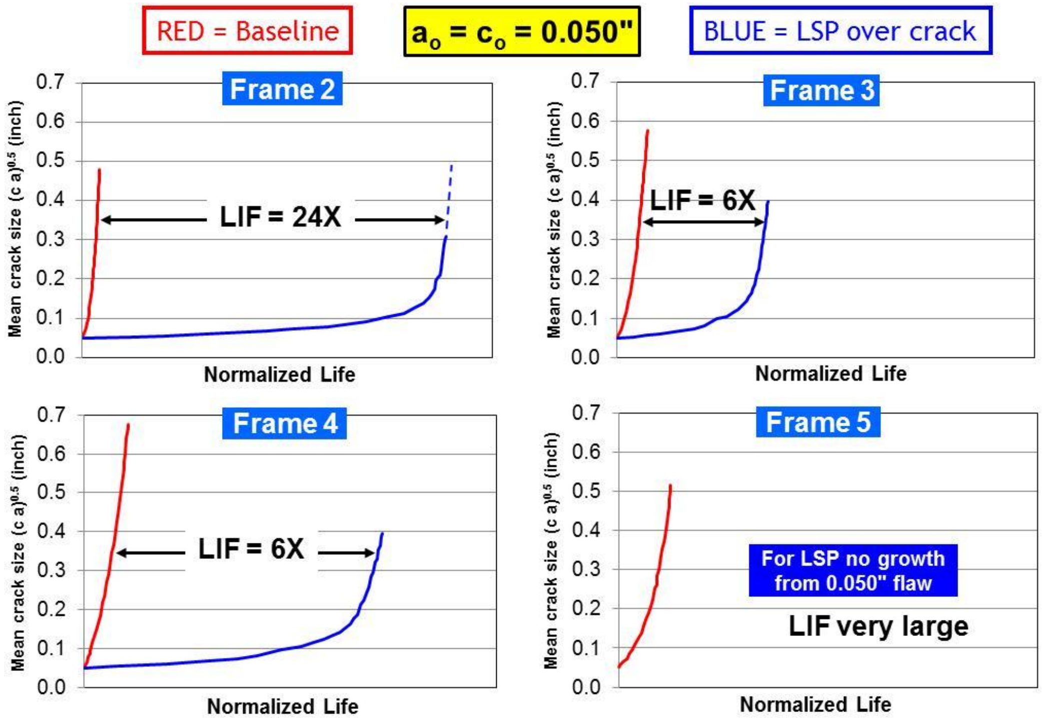


Figure 9.8-19. Predicted LSP Life Improvement Factors

9.8.5. Durability of Composite Wet Layup Repair on Leading Edge of F/A-18 Trailing-Edge Flap

Waruna Seneviratne and John Tomblin, Wichita State University-NIAR; Madan Kittur, USN-NAVAIR

The National Institute for Aviation Research (NIAR) and the Naval Air System Command (NAVAIR) is conducting a full-scale test evaluation of F/A-18 A-D inner-wings and trailing-edge flaps (TEF). The ability to use end-of-life aircraft structural components has been shown to be beneficial in many instances for the support of the existing fleet and provides a proactive approach to fleet maintenance. This program investigates the durability and damage tolerance of F/A-18 inner wing structure for the extended service life. Simulated inboard leading-edge flap (ILEF) and the outboard wing are attached to the inner-wing box for fatigue spectrum load application to assess possible damage threats. During receiving inspections prior to test, a large indentation and a major crack approximately 2.7-inch long was noted on the aluminum leading-edge of the right TEF (Figure 9.8-20). Further inspections using dye penetrant revealed that the indentation had produced several other surface and through-cracks extending about 1.5-inch on each side of the major crack. Once the spectrum fatigue loading of the TEF began, the major crack coalesced with nearby small cracks and grew further to approximately 4.5-inch in length at the end of 300 spectrum fatigue hours (SFH). The fatigue test was halted and the crack was then repaired with a wet-layup composite repair patch to prevent further crack propagation and potential catastrophic failure of the right TEF (Figure 9.8-21). Sanded aluminum surface was prepared with an application of AC Tech Sol-Gel AC-130 solution. Soon after sol-gel was cured for 90 minutes at ambient conditions, the surface was primed with BR-6700 and cured at 250°F using infrared and halogen lamps for 60 minutes. Then, a 12-ply quasi-isotropic wet layup was accomplished using AS4 plain weave fibers and EA 956 epoxy adhesive. The repair was cured, while holding vacuum, for seven days at ambient conditions. Repair integrity was inspected using pulse thermography. Then, a strain survey was conducted using a full-field photogrammetry image correlation system prior to spectrum fatigue loading

(Figure 9.8-22). The localized displacements and the far-field strain at the periphery of the composite repair patch remained consistent with the measurement prior to the repair. The repair patch survived over 3000 SFH with no indications of further damage growth or adverse strain anomalies in the surrounding structure.

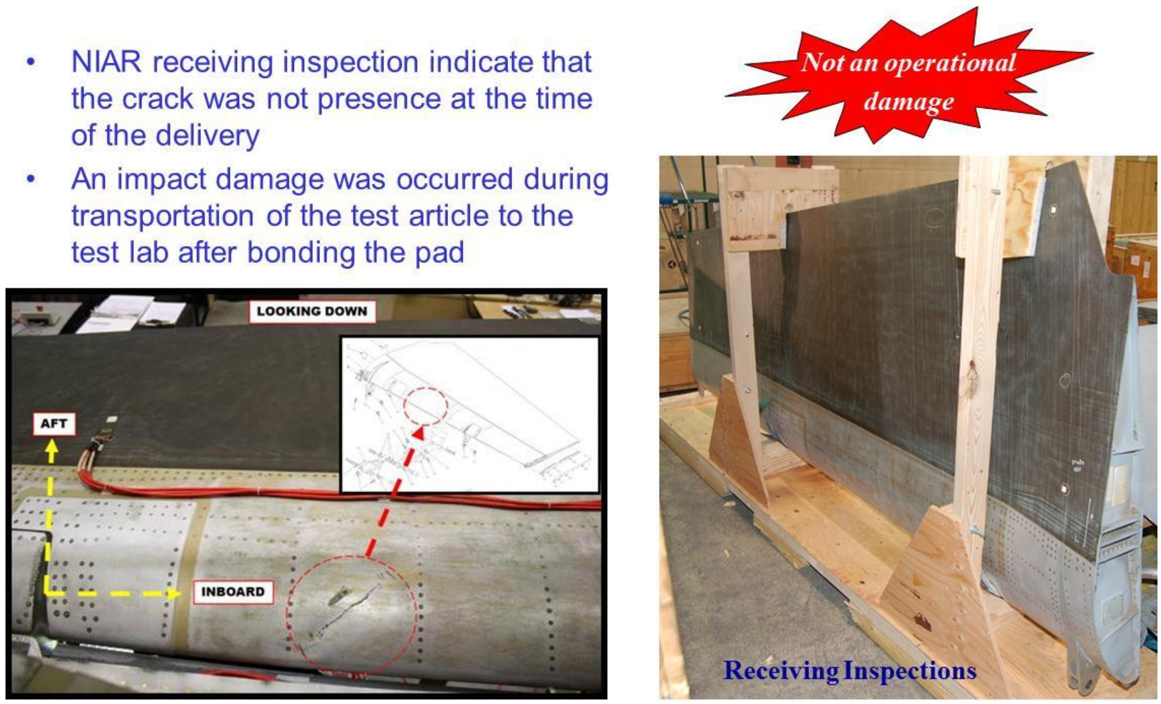


Figure 9.8-20. Genesis of Impact Damage

Figure 9.8-21. Wet Layup

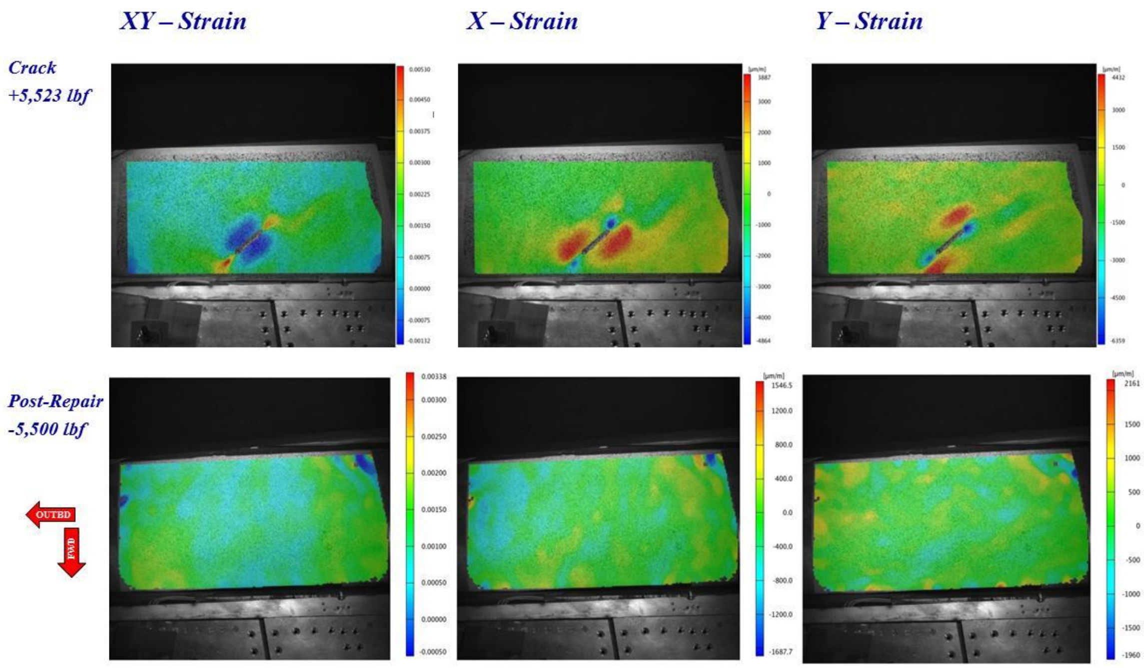


Figure 9.8-22. Photogrammetry on Crack Location (Post-Repair)

9.8.6. Redesign of the AH-64 Apache Composite Main Rotor Attachment Fittings

Len Reid, Fatigue Technology, Inc.

The development of a new Composite Main Rotor Blade (CMRB) for the AH-64 Apache Longbow Attack Helicopter (Figure 9.8-23) had three main objectives; reduced acquisition cost, reduced operation and support (O&S) cost through an enhanced field repair capability, and improved flight performance, reliability and durability, compared to the legacy Apache main rotor blade. The CMRB attachment lugs were designed for a 10,000-hour fatigue life and to be extremely damage tolerant. To achieve this, ForceMate high interference fit expanded 17-4PH stainless steel bushings were used in each of the four titanium blade attaching lugs (Figure 9.8-24). An alternate approach to the design of the blade root attachment was selected which included changing the current single piece stainless steel bushing installation to provide a removable wear bushing insert inside the main bushing that was simpler to install and also able to achieve the design performance objectives. The bushing and liner combination would be installed using the same ForceMate process as the original bushing installation to meet the original fatigue and damage tolerance capability attained in the current blade attachment. This technical activity reviews the current design configuration and the option to replace it with the thinner-walled ForceMate outer 17-4PH stainless steel bushing plus another ForceMate high interference-fit sacrificial aluminum-bronze wear bushing liner expanded into the outer bushing. This modification to the current high technology Apache CMRB will provide an O&S logistics benefit to the Army by reducing the dependence on a repair pipeline in the event of worn or galled attaching bushings, thereby providing the warfighter with a field sustainable/repairable helicopter rotor system with better sustainment, utilization and flexibility of it's assets (Figure 9.8-25).

Figure 9.8-23. AH-64 Apache Longbow Attack Helicopter

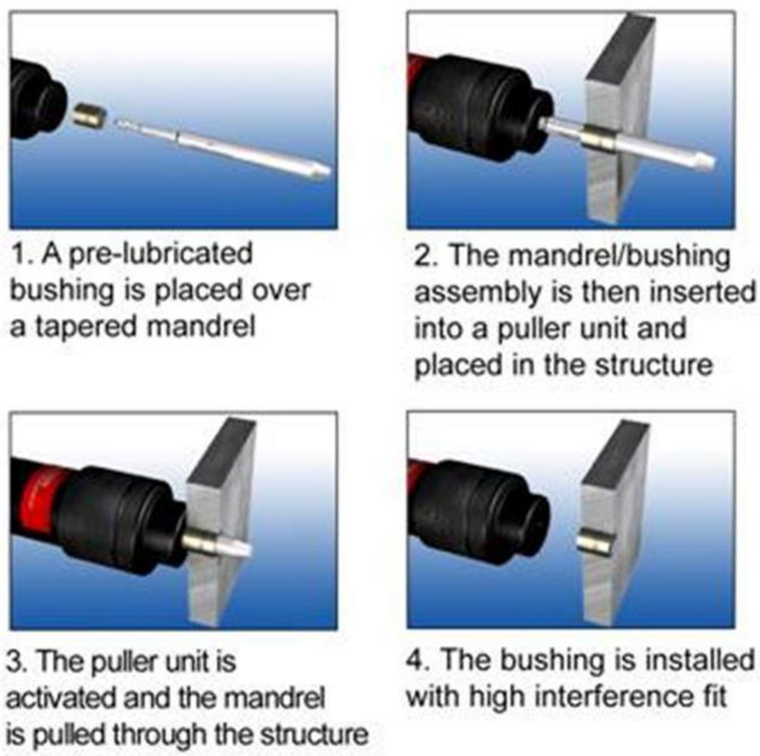


Figure 9.8-24. ForceMate Bushing Installation Method

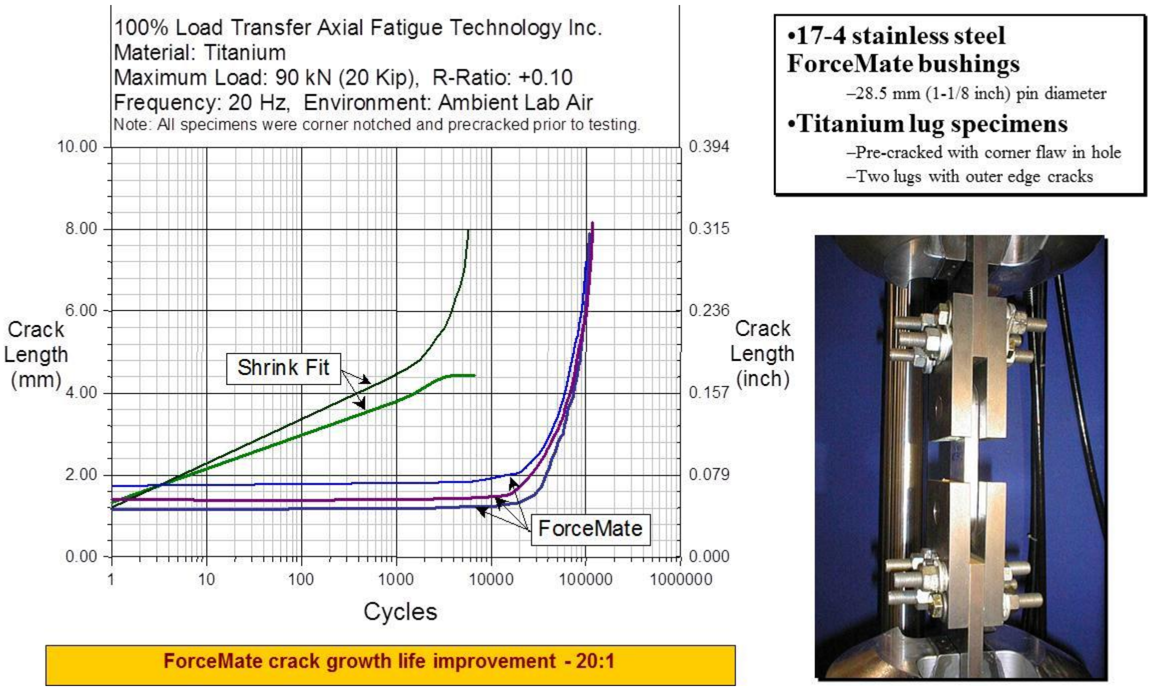


Figure 9.8-25. ForceMate Bushing Damage Tolerance

9.8.7. Investigation of Cold Expansion of Short-Edge-Margin Holes with Preexisting Crack in 2024-T351 Aluminum Alloys

Dallen Andrew, USAF A-10 ASIP

The experiments performed in this technical activity investigated the fatigue crack growth lives of short-edge-margin fastener holes (Figure 9.8-26) that contained a crack prior to cold expansion. Three configurations were used – a baseline condition consisting of non-cold-expanded holes, holes that were cold expanded, and holes containing a crack when cold expanded (Figure 9.8-27). All configurations were investigated under constant and variable amplitude loading. The hypothesis was that the cold expansion of a short-edge-margin hole with a crack prior to cold expansion will provide a significant increase in fatigue crack growth life compared to a short-edge-margin hole that was not cold expanded. The fatigue crack growth life of a cracked then cold-expanded hole was also compared to a hole that was not cracked prior to cold expansion. The United States Air Force (USAF) analytical approach used to account for the benefit due to cold expansion was compared to the experimental data and does not consistently provide conservative predictions (Figures 9.8-28 and 9.8-29).

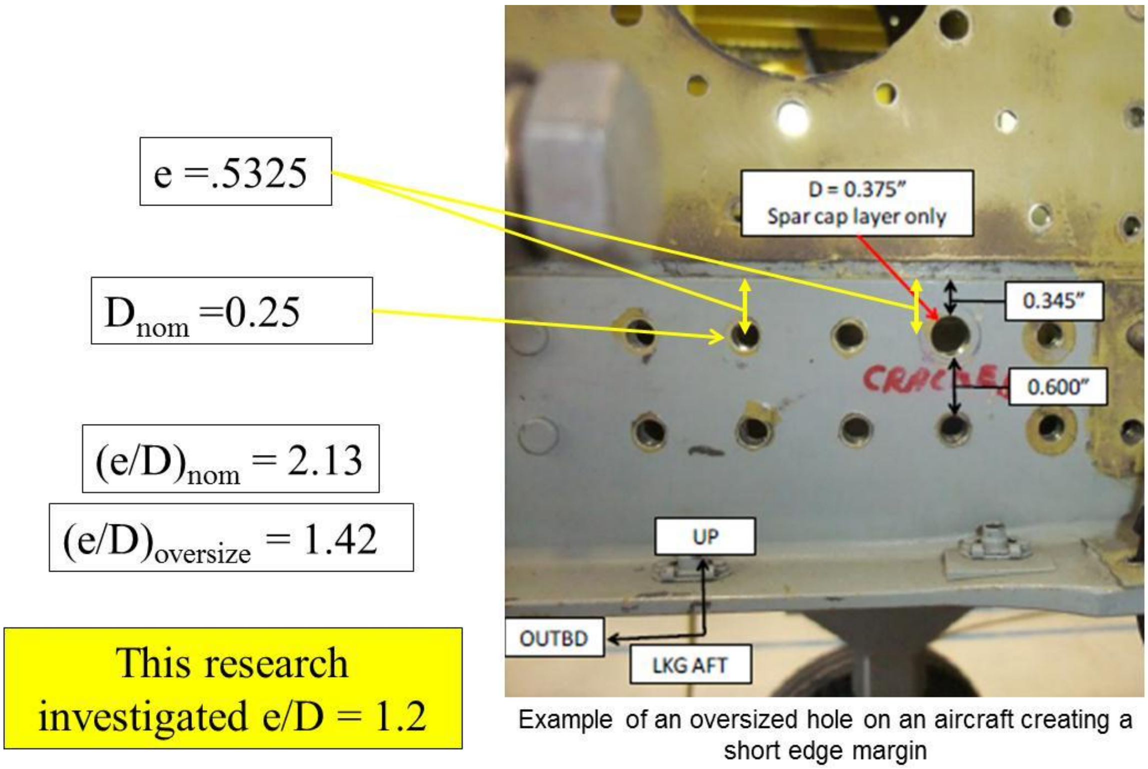


Figure 9.8-26. Short-Edge-Margin Example

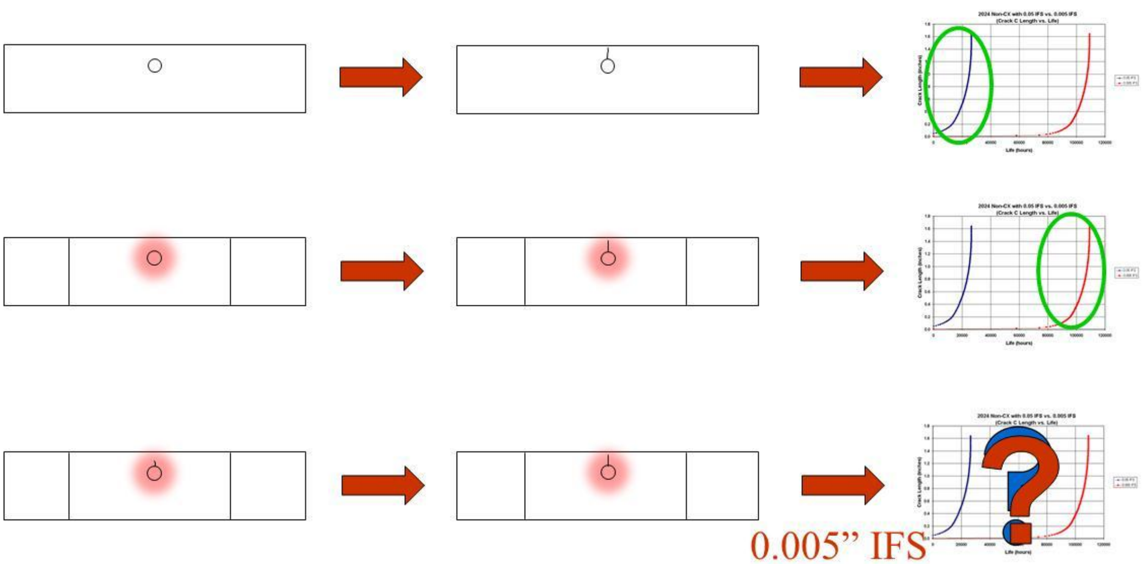


Figure 9.8-27. Specimen Fatigue Crack Growth Life Comparison

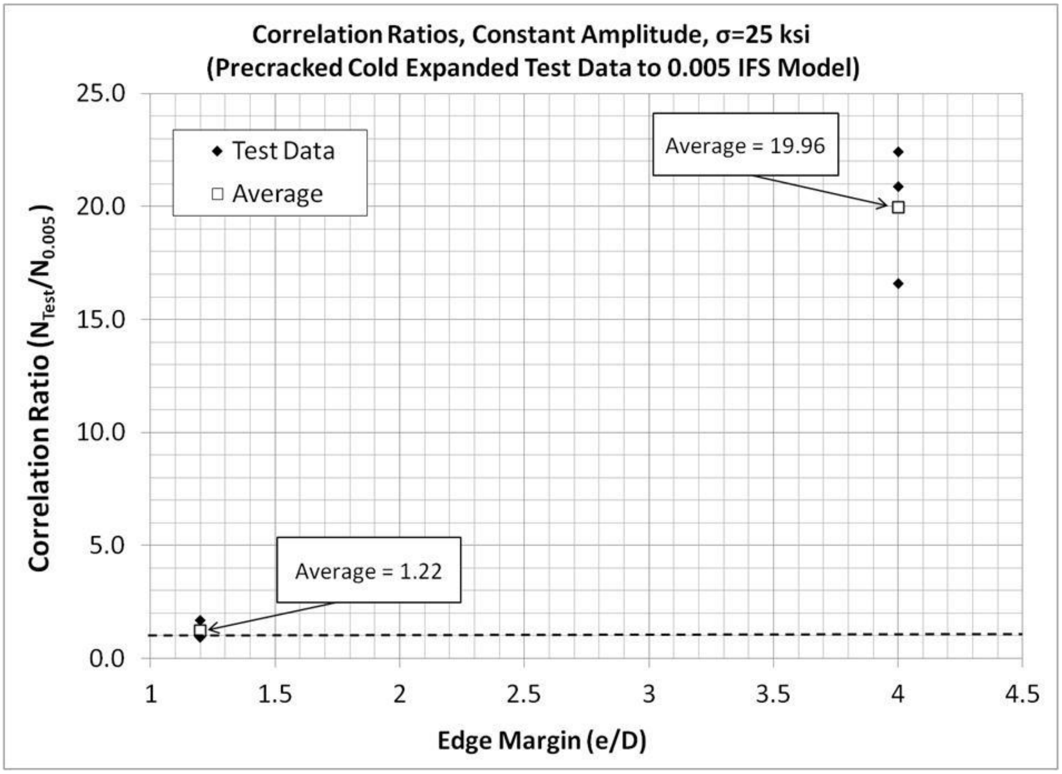


Figure 9.8-28. Correlation Ratios for Constant Amplitude Test Data Compared to a 0.005 IFS Prediction

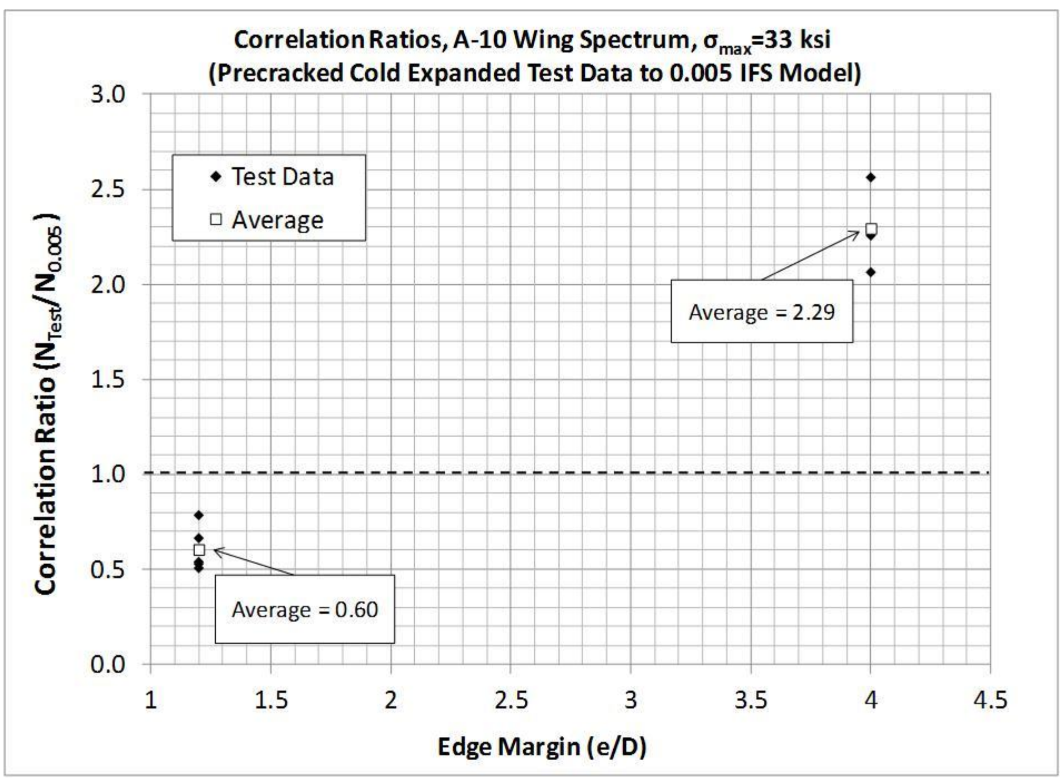


Figure 9.8-29. Correlation Ratios for Variable Amplitude Test Data Compared to a 0.005 IFS Prediction

9.8.8. Fatigue Evaluation of Freezeplug Repairs in Aluminum

Mark Ofsthun, Spirit AeroSystems, Inc.

In the complex fabrication process of building a commercial jet liner, there is a need for drilling hundreds of thousands of holes. These holes require precise sizes and accurate locations. In the course of fabricating a typical commercial airplane, there will be holes that are oversized and mis-located. When this occurs the standard repair is over-sizing the holes and using oversized fasteners. However, there are times when the oversized fasteners are not available and freezeplug repairs are considered. Freezeplug repairs are fairly common in aluminum and involve fabricating a plug from the same (or slightly stronger) material that is larger than the final hole size. The plug is immersed in liquid nitrogen to be shrunk and then quickly installed into the hole and as the plug warms, it develops a certain amount of interference. With the controlled interference fit of the freezeplug, the fatigue properties are intended to be restored.

The challenges with freezeplugs are retention over the life of the structure as well as predicting the fatigue performance of the freezeplug. Retention is simple enough by countersinking one side and capturing the freezeplug between parts or using a washer. The analyses methodology, however, is a bigger challenge. Some analysts conservatively assume the freezeplug is like an open hole. This assumption is indeed conservative, but in highly stressed components, this conservative assumption may make repair impossible and lead to costly scrapping of parts or at a minimum a more complex and expensive repair than actually necessary. Therefore, Spirit AeroSystems, conducted a very thorough fatigue test program in order to evaluate a freezeplug relative to open and filled holes.

The specimen Spirit elected to use in order to evaluate hole fill of a freezeplug was a simple dogbone (Figure 9.8-30). Baseline open holes, a slight clearance fit hi-lok and interference fit hi-lok specimens were fatigue tested head to head with freezeplugs having 0.2% to 0.4% interference fit. A sample of the fatigue data is shown in Figure 9.8-31. As expected, open holes are the poorest performing specimens in the group and interference fit hi-loks were the best. The freezeplug specimens were somewhere between the clearance fit and the interference fit.

Figure 9.8-30. Freezeplug Test Specimen

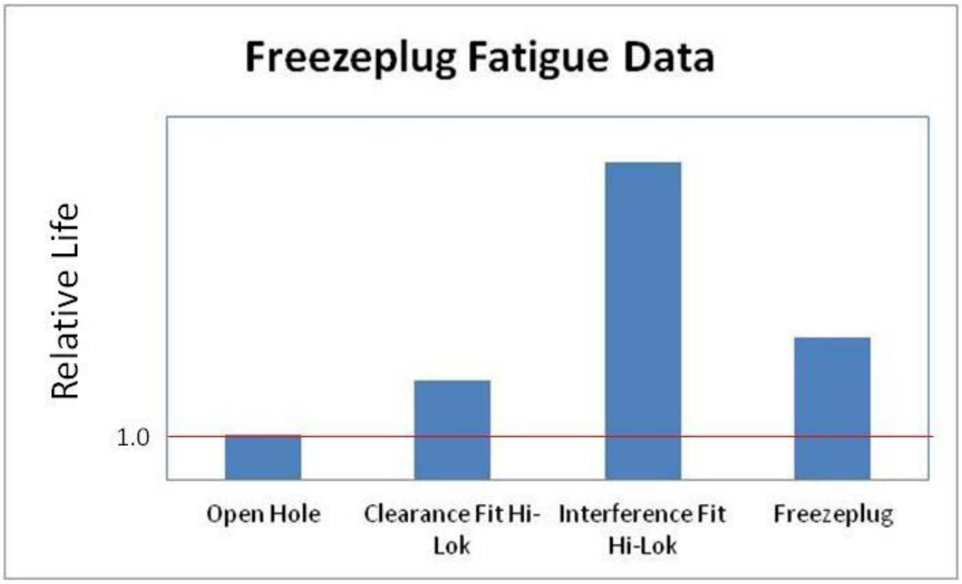


Figure 9.8-31. Freezeplug Test Results

In summary, Spirit has shown that freezeplugs with 0.2% minimum interference fit are significantly better than an open hole in fatigue and better than a transition fit bolt (slight clearance). Follow-on testing Spirit is conducting in 2013 include eccentrically placed freezeplug and joints with load transfer with freezeplug repairs.

9.8.9. Fatigue Performance of Bushing Installations in High Strength Steel, Single Pin Joints

Lee Ann Johnson, Spirit AeroSystems, Inc.

Single pin joints are used in many locations on typical aircraft structure. These joints often are fatigue critical, so extending the fatigue lives of single pin joints may be desirable to meet design goals. One factor to consider in a single pin joint is the use of bushings. More specifically, the type of installation used with bushings may affect the fatigue performance of a joint. To verify the effect in high strength steel, fatigue tests were conducted to determine if increased bushing fit would provide improved fatigue performance in a single pin joint.

Constant amplitude testing of PH13-8Mo steel lugs were performed to assess the potential influence of installing bushings into interference-fit lugs. Three levels of interference fit bushings were tested: 0.001 nominal clearance fit (baseline), 0.4% interference fit, and ForceMate® installed bushings. All specimens were tested to failure or to a predetermined maximum number of cycles with a constant amplitude loading per ASTM E466. The test set up is shown in Figure 9.8-32.

Figure 9.8-32. Lug Test Setup

Several specimens of each type were tested and the characteristic fatigue life for each type was calculated. As can be seen in Figure 9.8-33, there was little improvement in fatigue life from the interference-fit bushing tests compared to the clearance-fit bushing tests. However, there was an order of magnitude improvement in the fatigue life from the ForceMate® installed bushing tests compared to either the clearance-fit or interference-fit bushing tests.

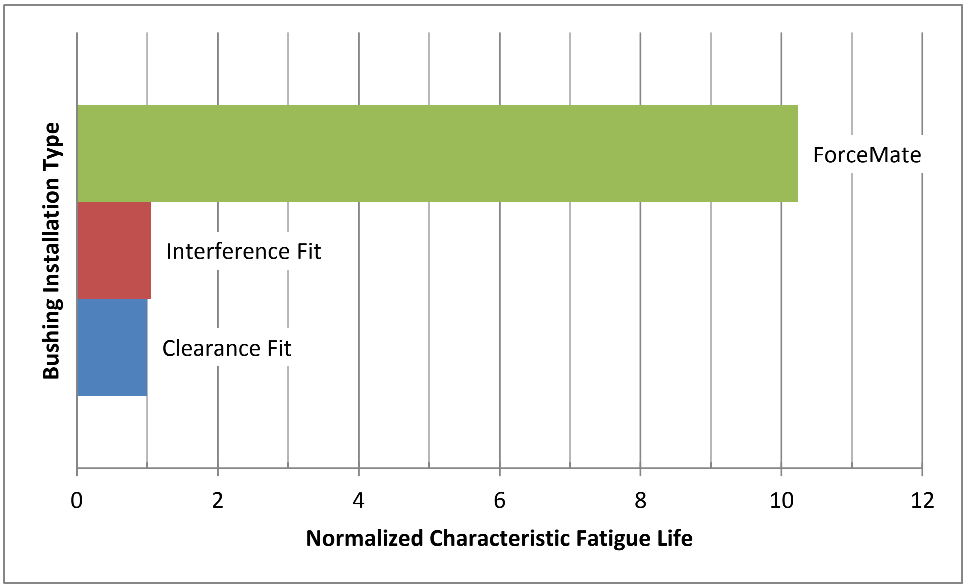


Figure 9.8-33. Fatigue Test Results

9.8.10. P-3 Propeller Life Extension and Cost Reduction

N. Jayaraman, Lambda Technologies

The P-3 aircraft operates in marine environments, resulting in potential corrosion pitting and stress corrosion cracking (SCC) in the 7076 aluminum propeller bore. Propeller maintenance practice for fitting new bushings included heavy shot peening to introduce deep residual compression to protect against SCC and fatigue failure. Reaming and re-machining operations were then necessary after peening to restore the bore finish and propeller geometry. Although P-3 propellers are designed for unlimited service life, the repeated loss of material during machining limited service to just three maintenance cycles.

To offset costs and extend the life of the P-3, NAVAIR chose Low Plasticity Burnishing (LPB®), developed by Lambda Technologies in Cincinnati, to replace the heavy shot peening. LPB induces a very deep, stable layer of compressive residual stress in the surface of a component, dramatically increasing damage and SCC tolerance to exponentially increase fatigue life. LPB leaves a mirror-like surface finish eliminating the need for reaming and machining that limited blade service life.

Complete turn-key robotic systems are installed at Pacific Propeller International, Cherry Point Marine Airbase and Warner Robbins Air Force Base to service the entire P-3 fleet (Figure 9.8-34).

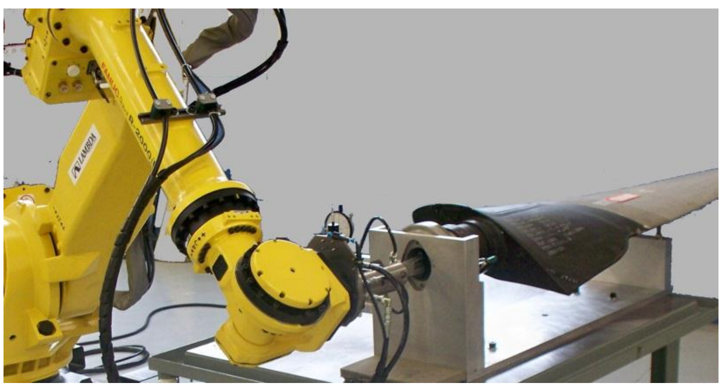


Figure 9.8-34. Robotic Processing of P-3 Propeller

By implementing LPB, NAVAIR improved the level of SCC and fatigue protection for the propeller bore while eliminating the reaming and machining steps, saving $1,000 per blade processed. Figure 9.8-35 shows the estimated maintenance cost savings with LPB. Without the reaming and machining operations, there is no loss of material in the propeller bore. This eliminates the need to scrap the part after three maintenance cycles and indefinitely extends its service life. At a cost of $35,000 per propeller, millions can be saved in cost avoidance after just a few years of implementation.

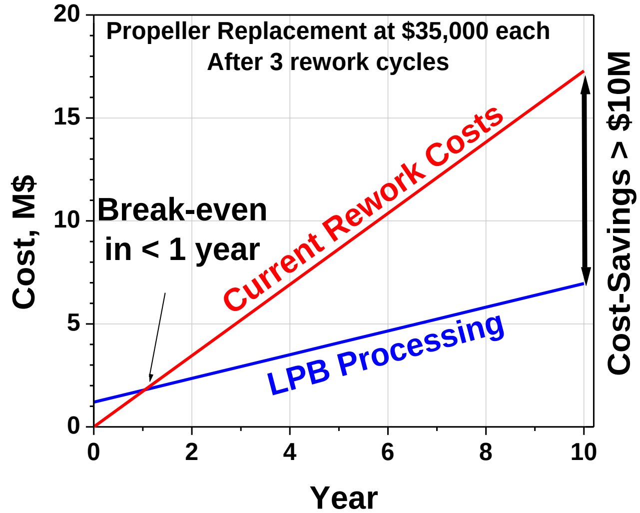


Figure 9.8-35. Estimated Maintenance Cost Savings with LPB

9.8.11. Mitigating Engine Fatigue Failure in the F402 AV8B Harrier

N. Jayaraman, Lambda Technologies

The First Stage Low Pressure Compressor (LPC1) vane in the F402-RR-408 engine powering the AV-8B Harrier V/STOL tactical strike aircraft was prone to foreign object damage (FOD) initiated fatigue failure. Safe operation required an onerous 30-hour inspection protocol and costly premature vane replacement. Even these measures did not eliminate Class A mishaps. Low Plasticity Burnishing (LPB®) improved the vane's FOD tolerance and high cycle fatigue (HCF) endurance limits to completely mitigate trailing edge fatigue failures.

LPB improves the service life of the F402 LPC1 vane by imparting stable through-the-thickness residual compression stresses that counter the applied tensile stresses in service. The vane's damage tolerance is dramatically increased and fatigue crack propagation from FOD is arrested. LPB improves damage tolerance and simplifies inspection to reduce the cost of aircraft ownership and improve fleet readiness.

LPB processing of fielded vanes increased the FOD tolerance over ten-fold from less than 0.002. to 0.020 in., with fatigue strength even greater than an undamaged new vane (Figure 9.8-36). Even with FOD up to 0.060 in. deep, the fatigue strength was 60 ksi, twice the service stress. Component fatigue testing confirmed that LPB improves the life of the vane by orders of magnitude (Figure 9.8-37).

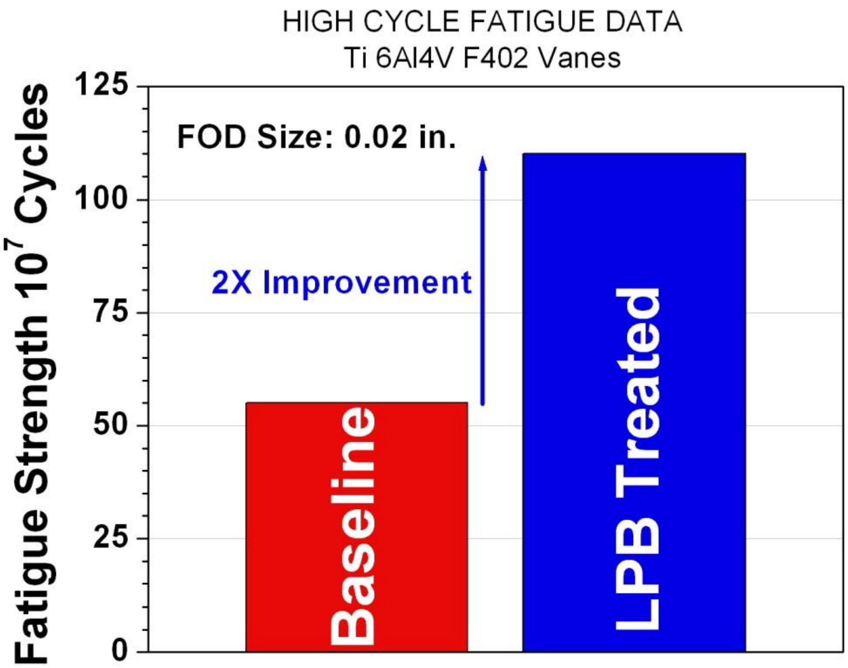


Figure 9.8-36. High Cycle Fatigue Data

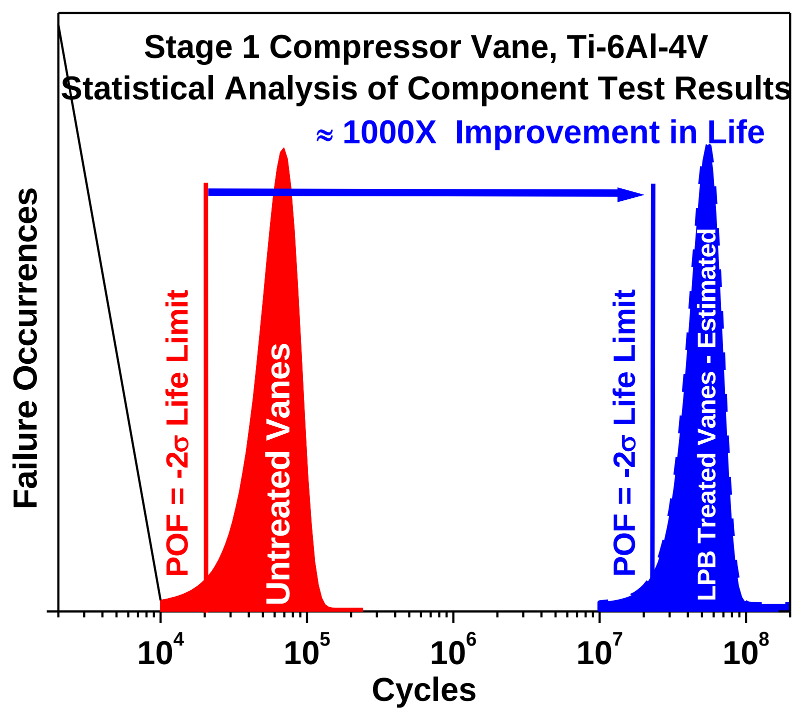


Figure 9.8-37. Statistical Analysis of Component Test Results

More than 11,000 LPB processed vanes are now flying. Not a single LPB treated vane has failed in service since production began, confirming Lambda's original fatigue design predictions. The importance of this program increases with NAVAIR's recent purchase of the UK's Harrier fleet. LPB's ability to mitigate HCF and FOD related failures supports the required fleet service life extension while reducing costs. NAVAIR estimates that more than eight million dollars will be saved on every 200 ship sets (Figure 9.8-38).

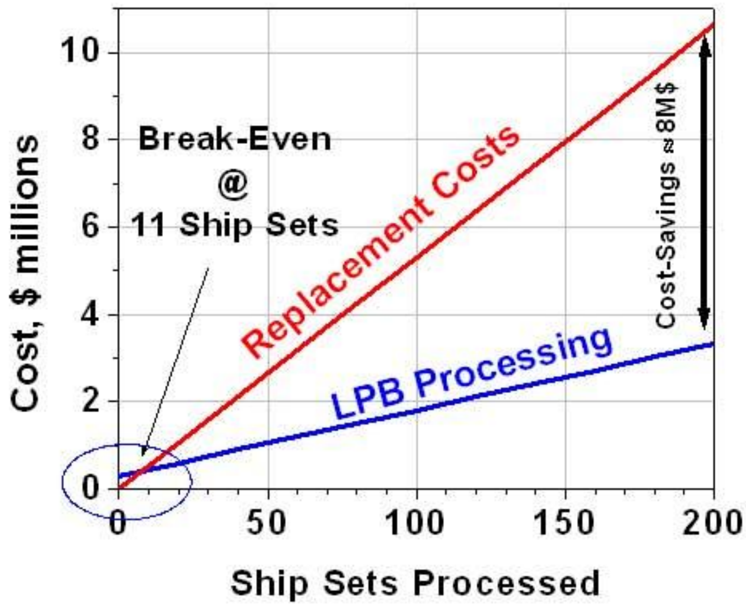


Figure 9.8-38. Cost vs. Ship Sets Processed

Reference

[1] Jayaraman, N., Prevey, P., & Ravindranath, R. (2007, 10 03). *Improved damage tolerance of ti-6al-4v aero engine blades and vanes using residual compression by design*. Retrieved from http://www.lambdatechs.com/documents/262.pdf

9.9. REPAIR CONCEPTS

9.9.1. Bonded Repair Technology

John G. Bakuckas, Jr., Reewanashu Chadha, and Ian Y. Won, FAA; Joel Baldwin, Kenneth Hunziker, and Cong Duong, The Boeing Company – Research & Technology

Under a Cooperative Research and Development Agreement (CRDA), the Federal Aviation Administration (FAA) and The Boeing Company are investigating the safety and structural integrity issues of bonded repair technology through test and analysis of metallic fuselage panels using the FAA Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility. The program objectives are to characterize the fatigue performance of bonded repairs under simulated service load (SL) conditions and to investigate tools for evaluating and monitoring the repair integrity over the life of the part.

A phased approach is being undertaken in this program. The initial phase was completed in FY 10 and revealed that properly designed and installed bonded repairs are durable under fatigue and can effectively contain large damage under severe static loads in excess of ultimate load requirements. The second phase was completed in FY 12 during which the fatigue behavior of under-designed, partially disbonded, compliant and damaged repairs to mid-bay cracks in metallic fuselage structure was characterized to evaluate repair integrity. To assess the abilities of analytical methods and monitoring systems, repair patches were made intentionally deficient to allow damage growth in the form of crack propagation and disbonding during fatigue cycling. Full-scale-fatigue tests were performed using a Boeing 727 fuselage panel. Both boron/epoxy (B/Ep) and aluminum repair patches with various anomalies were tested under SL conditions. Several patches were impacted to simulated hail and tool drop events.

The damage formation and growth of cracks and disbonds were monitored throughout the Phase 2 test using a variety of nondestructive inspection (NDI) methods, including visual inspections, eddy current, flash thermography, resonance ultrasonics, and computer-aided tap tester. In addition, a prototype piezoelectric-based structural health monitoring (SHM) system was used to assess and demonstrate its capabilities in determining the condition of damage in the repair patches. Full-field strain and displacement measurements on the patch and the surrounding regions were obtained using the digital image correlation (DIC) method. Test and analysis results correlations were conducted to further calibrate models and demonstrate the capabilities to design and analyze bonded repairs.

In general, the ability to detect growing flaws under bonded patches and monitor the effectiveness of the repair using NDI and SHM was demonstrated, Figure 9.9-1. Several parameters that affect repair performance were assessed including under-designed, partially disbonded and impacted repairs. Results revealed that fatigue performance of a repair in effectively containing damage reduces as the repair quality degrades, Figure 9.9-2. Impacting caused both crack extension and disbonding in the repairs; however, there was no subsequent damage growth during fatigue cycling as indicated by NDI and DIC, Figure 9.9-3. Model predictions were in good agreement with crack growth test results for the majority of repair patch configurations. However, the test and analysis correlations revealed further work is needed to better account for thermal residual stresses and thick-patch configurations.

The final third phase of this program will focus on the effects of environment on the durability and fatigue performance of bonded repairs and on further improvements to analytical models.

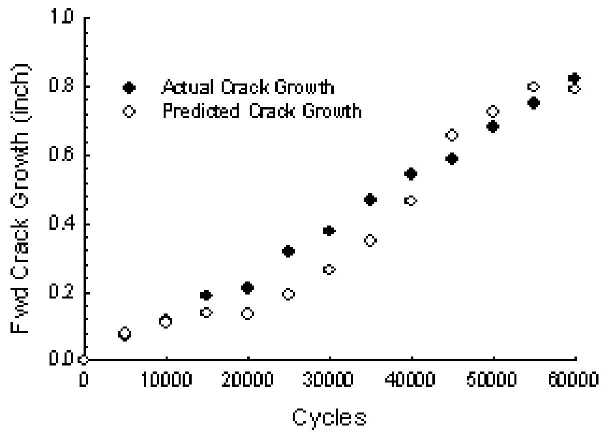


Figure 9.9-1. Monitor Crack Growth Using SHM

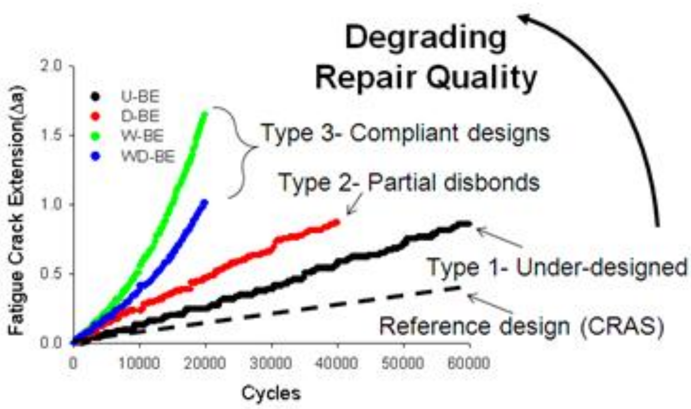


Figure 9.9-2. Effect of Repair Quality on Performance

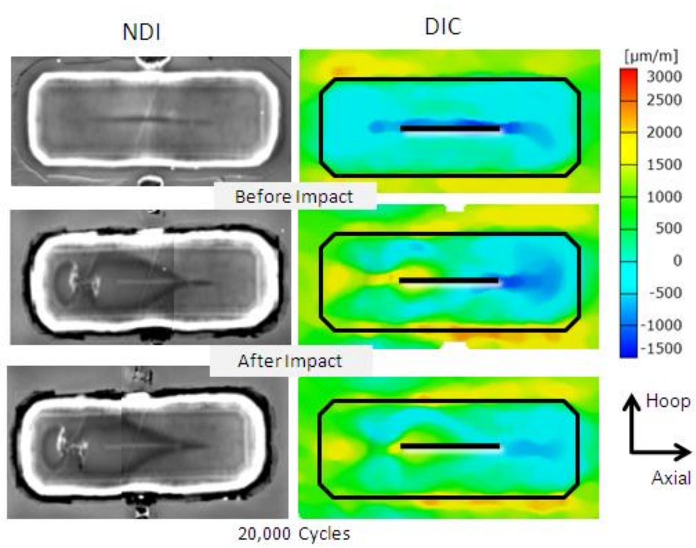


Figure 9.9-3. Monitor Impact Damage

9.9.2. Modification of the FAA's Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) Facility for Mechanical and Environmental Loading of Fuselage Structure

John G. Bakuckas, Jr., Yongzhe Tian, and Ian Y. Won, FAA; Kelly Greene and Carlyn Brewer, The Boeing Company – Research & Technology; Gregory Korkosz, Legacy Engineering

In a joint effort, the Federal Aviation Administration (FAA) and The Boeing Company are investigating the safety and structural integrity issues of adhesive bonded repair technology through test and analysis of metallic B727 fuselage panels using the FAA Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility. The program objectives are to characterize the fatigue performance of bonded repairs under simulated service load (SL) conditions and to investigate tools for evaluating and monitoring the repair integrity over the life of the part.

A phased approach is being taken involving the testing and analysis of several panels containing boron-epoxy and aluminum bonded patches. A variety of loading and environmental conditions, damage scenarios and repair conditions are being considered. The first panel tested provided baseline data and was completed in 2010. The results revealed that properly designed and installed bonded repairs are durable under fatigue and can effectively contain large damage under severe static loads in excess of ultimate load requirements.

The second panel tested was completed in 2012 in which repair patches were made intentionally deficient and contained defects to permit damage growth. In general, the ability to detect growing flaws under bonded patches and monitor the effectiveness of the repairs was demonstrated using several non-destructive inspection (NDI) methods and a prototype piezoelectric-based structural health monitoring (SHM) system. Model predictions were in good agreement with crack growth test results for the majority of repair patch configurations. However, the test and analysis correlations revealed further work is needed to better account for thermal residual stresses and thick-patch configurations.

The prior two panels were tested at lab temperature ambient conditions. In the next phase of this program, focus will be placed on assessing the effects of environment on the durability and fatigue performance of bonded repairs and continued efforts to investigate tools for evaluating and monitoring the repair bond integrity. A third B727 panel will be fatigue tested under combined mechanical and environmental loading. Major modifications were made to fully integrate an environmental system with the FASTER fixture to apply synchronous mechanical-temperature and humidly load profiles (Figure 9.9-4). With this new enhancement, fuselage panels can now be tested under a variety of operating environments ranging from hot-wet (165°F and 85-95% humidity) to extreme cold (-65°F) conditions.

Results from this on-going collaboration will be published as they become available. In general, repair configurations from the first two panel tests will be retained for comparison purposes to assess the effects of hot-wet environmental conditions on repair performance. The effectiveness of several NDI and an updated SHM systems to monitor damage growth in the repairs will be gauged. Improvements in predictive tools to design and analyze bonded repair will be demonstrated and verified though test and analysis correlations. Data from this program will be used to assess methods to quantify bonded repair integrity.

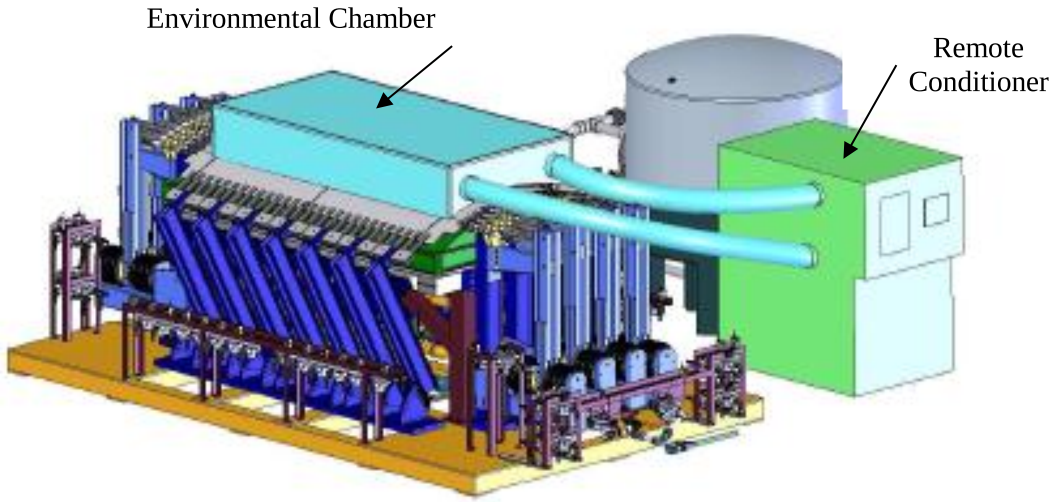


Figure 9.9-4. Modification to FAA's FASTER Fixture for Mechanical and Environmental Loading Capabilities

9.9.3. Durability of Composite Wet Layup Repair on Leading Edge of F/A-18 Trailing-Edge Flap

W. Seneviratne, S. Tomblin, T. Cravens, M. Tran, C. Saathoff and B. Saathoff, Wichita State University –NIAR

The National Institute for Aviation Research (NIAR) and Naval Air System Command (NAVAIR) is conducting a full-scale test evaluation of F/A-18 A-D inner-wings and trailing-edge flaps (TEF). The ability to use end-of-life aircraft structural components has been shown to be beneficial in many instances for the support of the existing fleet and provides a proactive approach to fleet maintenance. This program investigates the durability and damage tolerance of F/A-18 inner wing structure for the extended service life. Simulated inboard leading-edge flap (ILEF) and the outboard wing are attached to the inner-wing box for fatigue spectrum load application to assess possible damage threats. During receiving inspections prior to test, a large indentation and a major crack approximately 2.7-inch long was noted on the aluminum leading-edge of the right TEF (Figure 9.9-5). Further inspections using dye-penetrant revealed that the indentation had produced several other surface- and through-cracks extending about 1.5-inch on each side of the major crack.

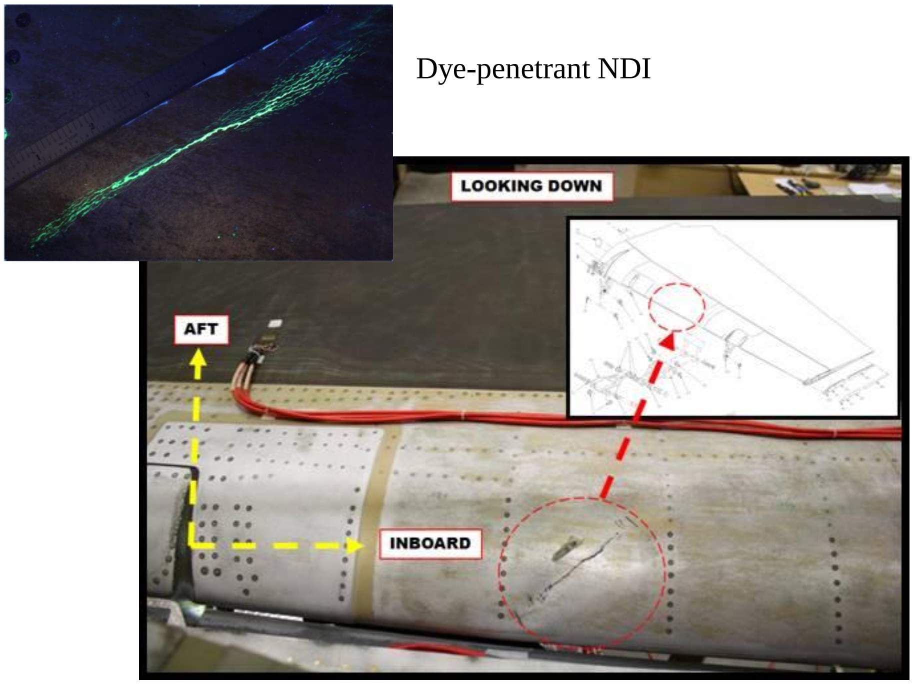


Figure 9.9-5. Leading-Edge Crack

Once the spectrum fatigue loading of the TEF began, the major crack coalesced with nearby small cracks and grew further to approximately 4.5-inch in length at the end of 300 spectrum fatigue hours (SFH). The fatigue test was halted and the crack was then repaired with a wet-layup composite repair patch to prevent further crack propagation and the potential catastrophic failure of the right TEF. Sanded aluminum surface was prepared with an application of AC Tech Sol-Gel AC-130 solution. Soon after the sol-gel was cured for 90 minutes at ambient conditions, the surface was primed with BR-6700 and cured at 250°F using infrared and halogen lamps for 60 minutes. Then, a 12-ply quasi-isotropic wet layup was accomplished using AS4 plain weave fibers and EA 956 epoxy adhesive. The repair was cured, while holding vacuum, for seven days at ambient conditions. Repair integrity was inspected using pulse thermography. Then, a strain survey was conducted using a full-field photogrammetry image correlation system prior to spectrum fatigue loading. The localized displacements and the far-field strain at the periphery of the composite repair patch remained consistent with the measurement prior to the repair. The repair patch survived over 3,000 SFH with no indications of further damage growth or adverse strain anomalies in the surrounding structure (Figure 9.9-6).

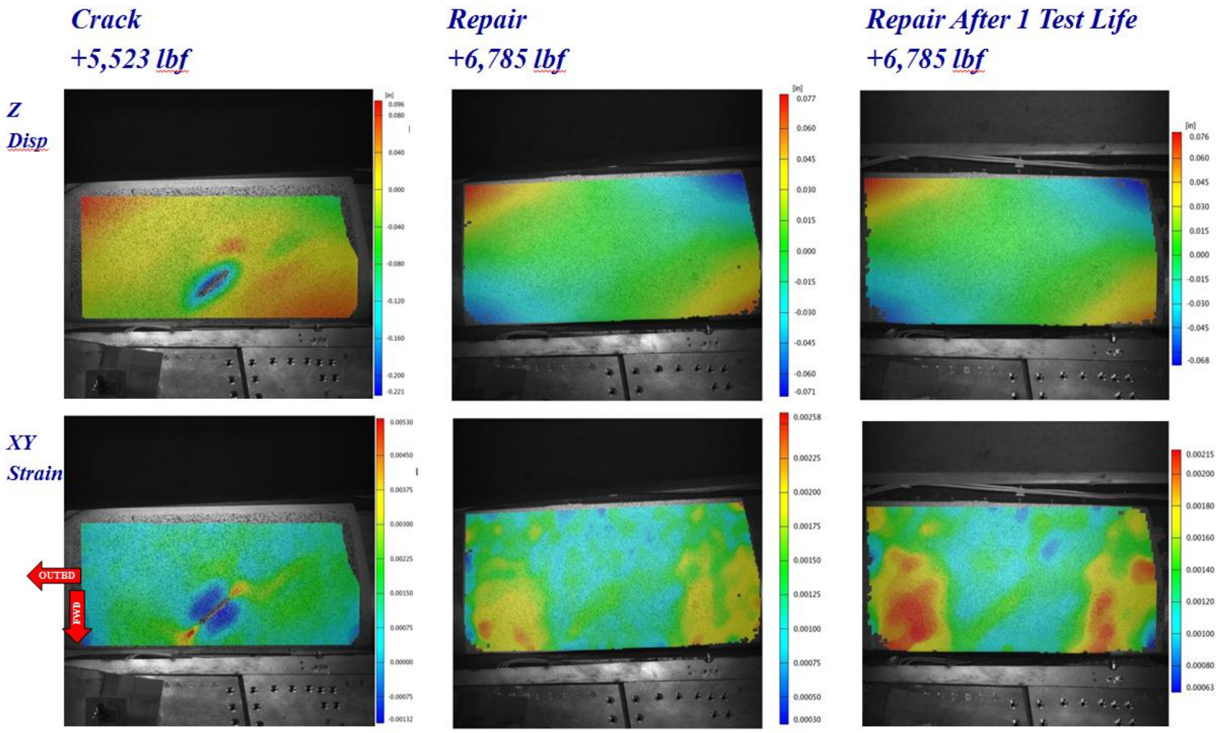


Figure 9.9-6. Photogrammetry Full-Filled Strain/Displacement Measurements on Repair Location

9.10. REPLACEMENT CONCEPTS

9.10.1. Advanced Hybrid Structures Core Technology Development Program

Charles Saff, Dave Heck, Scott Fields, Haozhong Gu and Dan Muntges, The Boeing Company – Research & Technology; Ed Forster and Stephanie Flanagan, USAF Research Laboratory – Aerospace Systems Directorate

Fiber metal laminates (FMLs) offer great promise for significant damage tolerance benefits for aircraft structures at very small weight and cost penalties. They offer around five times greater crack growth life than conventional metals at about twice the cost and they provide twice the damage resistance of composite materials at half the cost. The trades for damage tolerance limited structures can be compelling.

This technical activity will provide coupon and design detail data substantiating the promise of the material system, offering some example trades completed recently and discuss in some detail the analyses performed and analysis tools used to develop these trades. Moreover, the test data will be correlated to predictions made using the analysis tools to provide validation of the tools and thus the trade studies as well (Figure 9.10-1 through 9.10-3). Strength and crack growth data from open holes and single and double lap shear joint specimens will be provided (Figure 9.10-4).

This analysis development and validation was performed under the Advanced Hybrid Structures Core Technology Program being performed by Boeing and Lockheed-Martin for the United States Air Force Research Laboratory (AFRL) under the leadership of Dr. Ed Forster. Lockheed-Martin's portion of the program is focused on countersunk fasteners while Boeing's effort is focused on protruding head fastener systems. This is related to the applications on which each contractor focused their trade studies. Boeing's trade studies were the result of an on-going in-kind Independent Research and Development (IRAD) effort at Boeing that is focused on evaluating the ability of the tools developed for GLARE under the AFRL program to predict the behavior of other FML materials.

Boeing's approach to the development of FML tools differs from most in that Boeing started with a composite analysis tool background and tried to develop a fundamental understanding of the behavior of the FML as a composite structure. Once the stiffness, strain and strength behavior of the material was understood, a strain-based crack growth criterion was applied using standard fatigue crack growth analyses to predict crack growth in the structure using AFGROW. Results were reasonable and useful for performing trade studies. This is a bit different from the Fiber Metal Volume (FMV) approach favored by Delft and Alcoa, and a little more difficult to implement since metallic 'plies' don't respond to load as unidirectionally reinforced 'plies' do in composites. And the delamination that accompanies crack growth plays an unknown role in the modeling necessary for detailed analyses. The benefit of the composite analysis tools is that, once validated, any metal in combination with any fiber system can be predicted using the constituent material properties.

Under a combination of AFRL and Boeing funding, Boeing has also developed spreadsheet-based tools that have incorporated these fundamentals into strength, stiffness, and stress intensity solutions that are functions of crack length, hole geometry and effective width as well as hybrid materials, thicknesses and layups. These solutions are meant to be more design tools than in-depth mechanics or physics-based solutions.

Tensile Strength

- Test Data – 445 MPa
- ISAAC Prediction
  - 490 MPa – before modification
  - 443 MPa – after modification

Test Data

Specimen	σ_{ult} (MPa)	ε (%)	E (GPa)
Dry GF/E	460 ± 17	2.20 ± 0.04	27.4 ± 1.1
Wet GF/E	360 ± 6	1.80 ± 0.03	24.7 ± 1.2
Dry Glare	445 ± 23	1.70 ± 0.10	53.3 ± 1.0
Wet Glare	441 ± 27	1.90 ± 0.10	52.7 ± 1.0

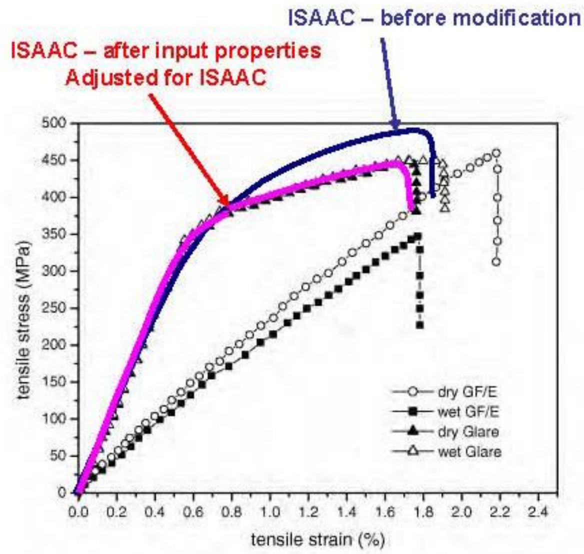
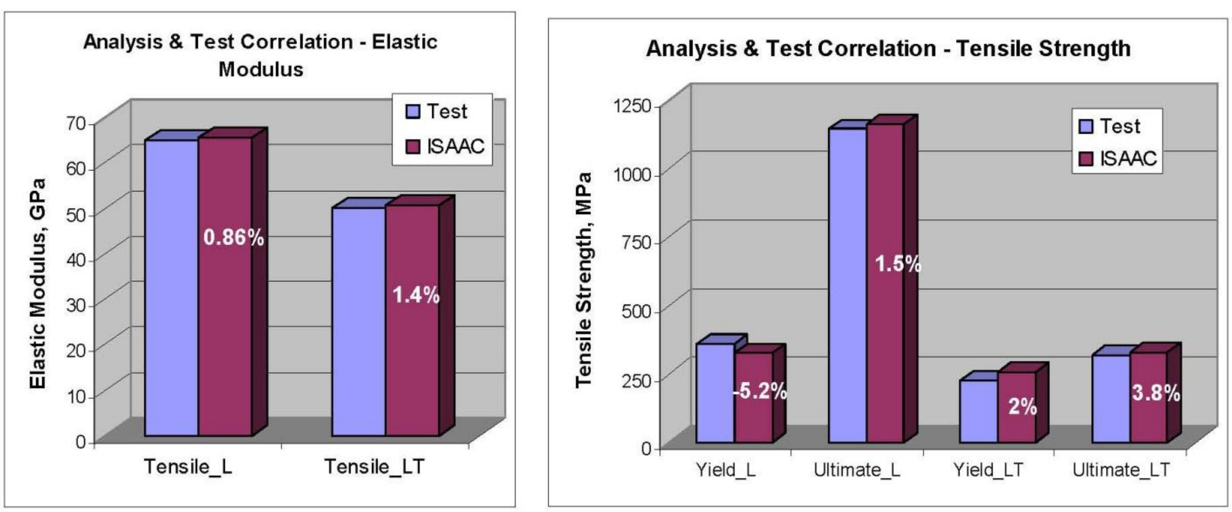


Figure 9.10-1. ISAAC Works Well for FMLs



- Largest deviation in modulus prediction of 1.4% at LT direction
- All ultimate strength predicted have less than 4% deviations
- Yield strength prediction has largest deviation of 5%

Figure 9.10-2. ISAAC Correlations of Tensile Modulus/Strength

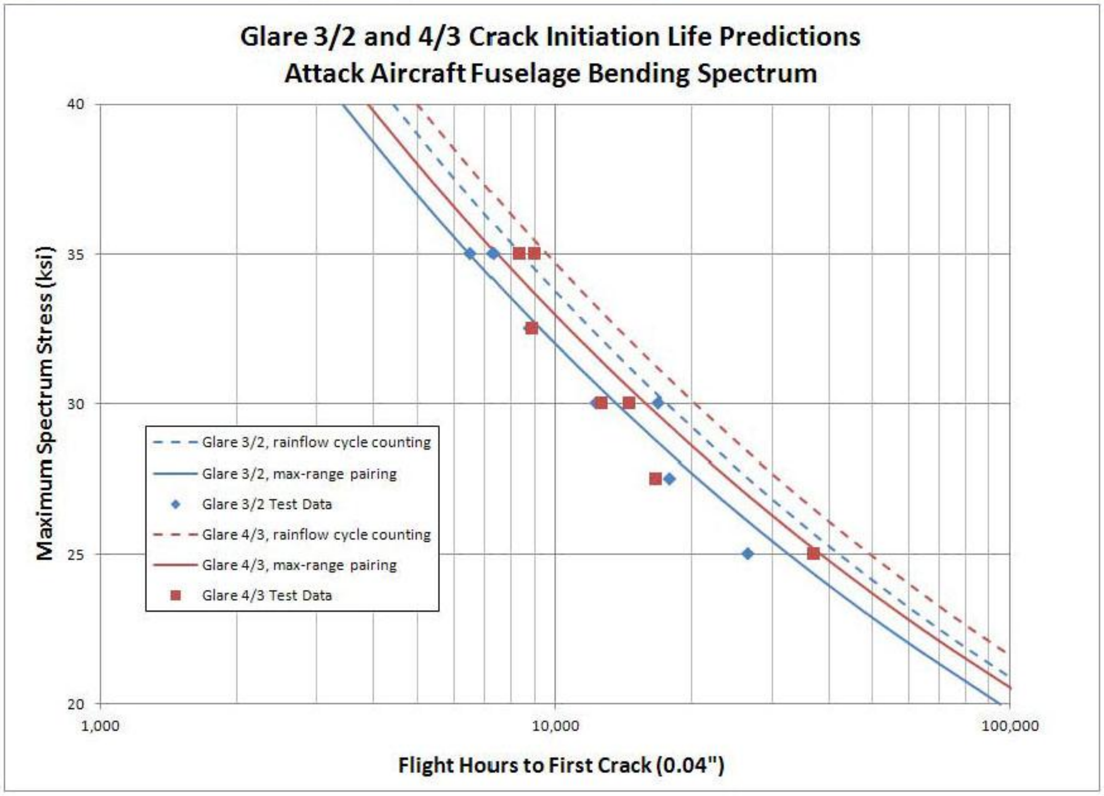


Figure 9.10-3. Spectrum Life Predictions vs. Test Data

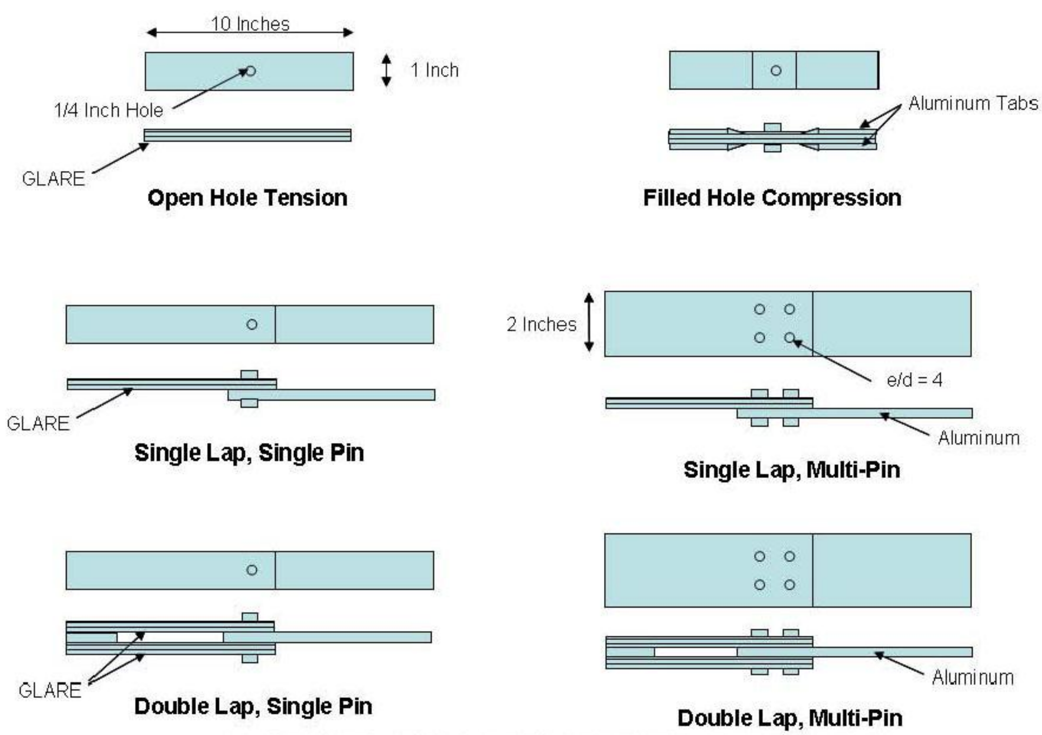


Figure 9.10-4. Test Specimens

9.10.2. Development, Validation and Demonstration of Advanced Hybrid Structures (AHS) for T-38 and A-10 Lower Wing Covers

Frank Di Cocco, Alcoa-Defense; David Heck, The Boeing Company – Research & Technology; Eric Fodran, Northrop Grumman Corporation

Advanced Hybrid aerospace Structures (AHS) are based on intelligently combining modern aluminum alloys and Fiber Metal Laminate (FML) technologies to enable extraordinary damage tolerance performance and affordability improvements over current state-of-the-art capabilities. Building on the trade study results of MAI (Metal Affordability Initiative) Task II, the MAI Advanced Hybrid Structures Task III program partners of Alcoa Defense, Boeing Research & Technology and Northrop Grumman designed, built and tested AHS lower wing covers for the T-38 and A-10.

This technical activity will show the process used to develop and evaluate AHS designs, establish design allowables, develop manufacturing plans, validate design allowables, and build demonstration articles for both the T-38 and A-10 lower wing covers. Both lower-wing-cover designs build on available FML technologies, but significantly expand the design envelop beyond the existing GLARE solutions by incorporating advanced aluminum alloys (e.g. 7055-T762 sheet on the A-10), thicker aluminum sheet gauges (0.050” thick 7475-T761 Al sheet on the T-38), and tailored layups. The design allowables needed to develop these solutions were estimated using a combination of empirical metal volume fraction and FEM tools and later validated with coupon testing. The final design solutions were able to achieve 10% weight savings on the T-38 (Figures 9.10-5 and 9.10-6) and 25% on the A-10 (Figures 9.10-7 and 9.10-8) with dramatically improved damage tolerance and service life, thereby significantly reducing life cycle costs. Large-scale-demonstration panels of both lower wing covers are being built to demonstrate and validate the manufacturing flow path and provide input for the manufacturing cost and life cycle cost studies.

The MAI Advanced Hybrid Structures Task III program demonstrated weight, damage tolerance, and cost improvement potential of AHS concepts for lower wing covers on two example USAF legacy aircraft. An assessment of the United States manufacturing base for AHS determined that most of the required production processes are at TRL9 and MRL10. The Program elevated the MRL for AHS technology to MRL 6 or greater and further mitigated aerostructural risks in accordance with ASIP design requirements.

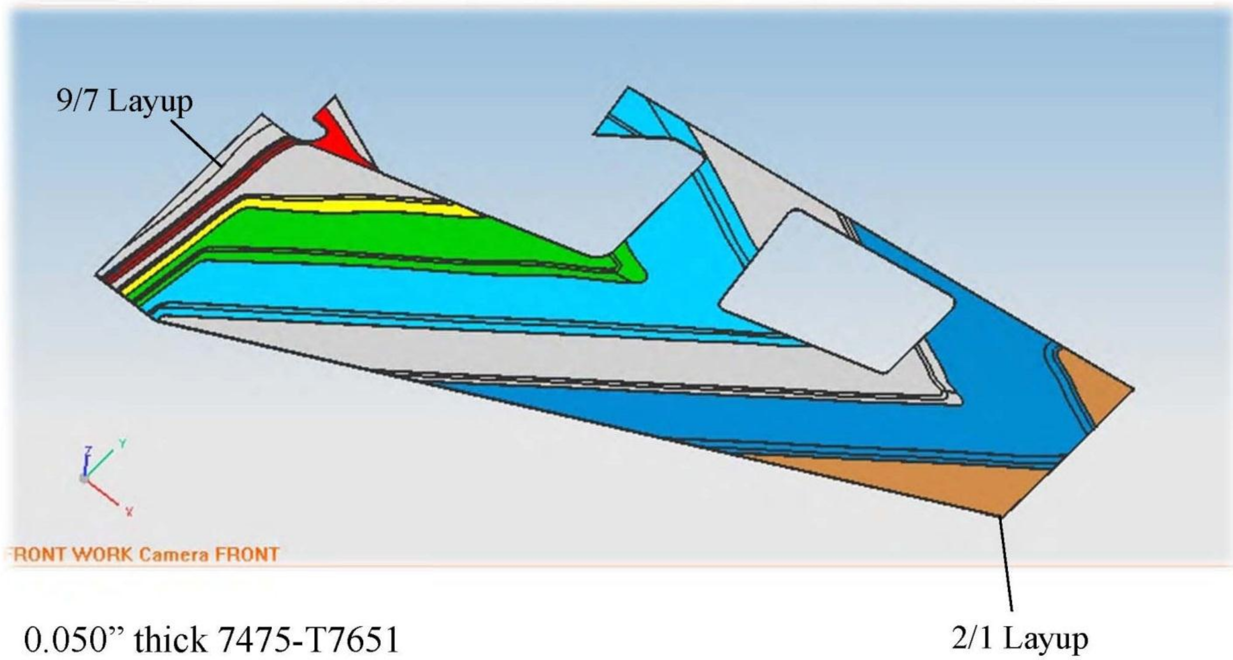
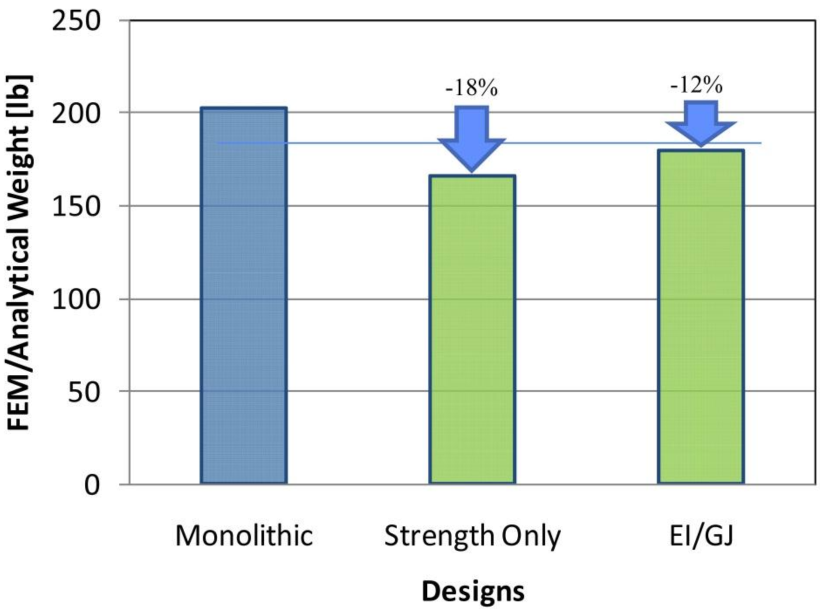


Figure 9.10-5. T-38 Manufacturing Demonstrator

- **Analytical weight savings potential of ~10% with AHS**



- **Conservative engineering configuration employed to meet incumbent IML interface and wing profile as well as reduce manufacturing cost.**

Figure 9.10-6. T-38 Weight Savings

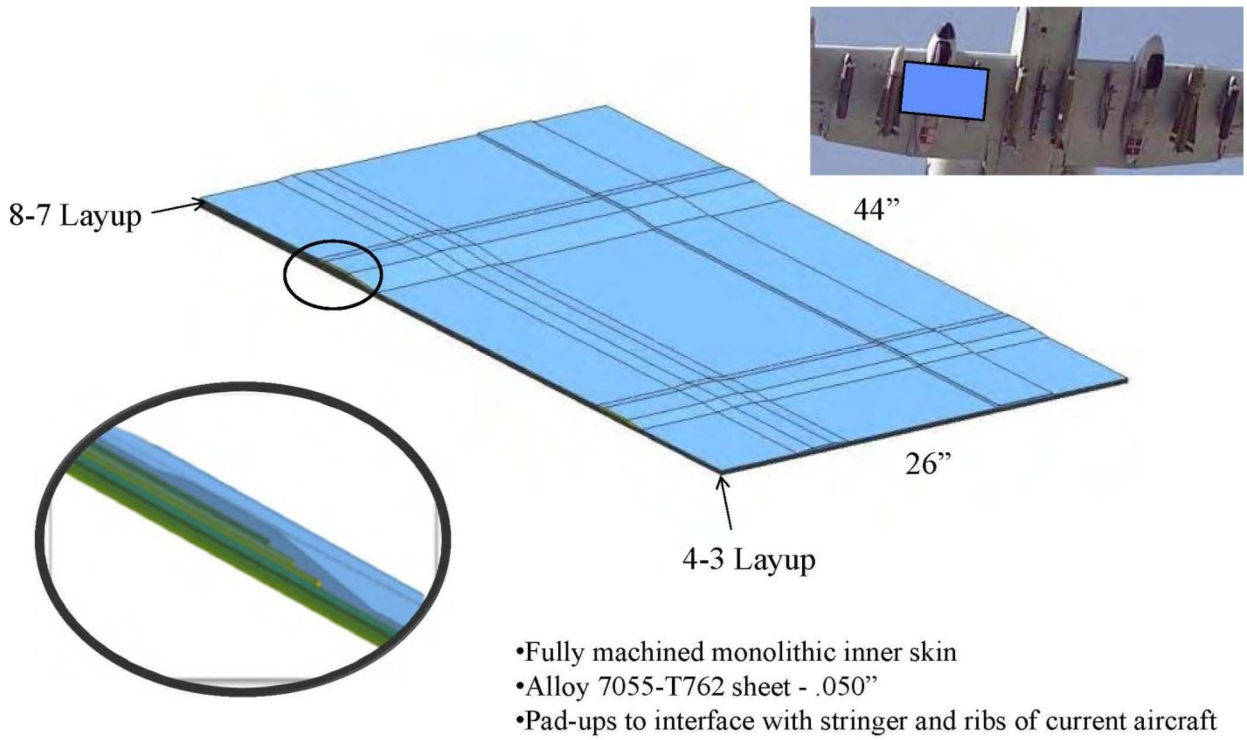


Figure 9.10-7. A-10 Manufacturing Demonstrator

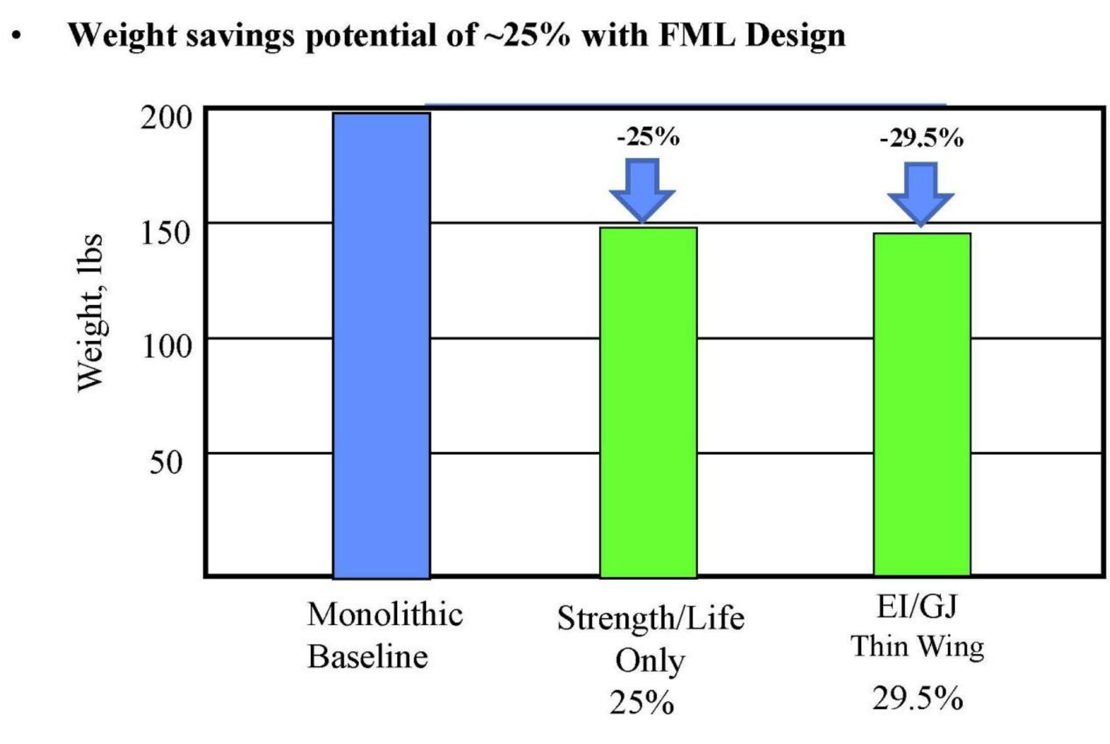


Figure 9.10-8. A-10 Weight Savings

9.10.3. Advanced Hybrid Structures for C-130 Life Enhancement (HyLife): Aluminum vs. Fiber Metal Laminate Complex Joint Head-to-Head Test Results

Doug Miller and Hank Phelps, Lockheed Martin Corporation

Lockheed Martin Aeronautics is currently working a USAF program called Advanced Hybrid Structures for C-130 Life Enhancement (HyLife). The goal of this program is to demonstrate the potential damage tolerance and residual strength benefits of Fiber Metal Laminate (FML) structures by testing complex joints that exist in transport aircraft wings. A representative C-130 wing joint was selected to demonstrate the potential benefits of FML materials under the HyLife program. This technical activity covers the results of the Aluminum vs. FML head-to-head complex joint durability and damage tolerance testing. Using design data and guidelines developed under HyLife, GLARE FML joints were designed that are representative of the outer wing root rainbow fitting-to-skin joint, which is a source of considerable maintenance issues for the Air Logistics Centers. The FML version of the C-130 rainbow fitting to lower skin joint (Figure 9.10-9) was designed using existing strength design loads and a severe usage fatigue spectrum. The fatigue life requirement used to design the joint was two 30,000 flight hour lifetimes, or 60,000 flight hours. A test specimen was designed that replicates the FML skin/stringer panel to rainbow fitting joint (Figure 9.10-10). In addition to the FML joint specimen, an all-7075 Aluminum specimen representing the baseline configuration was designed and fabricated. Since the spectrum used for design was more severe than the baseline C-130 spectrum, the Aluminum specimen was sized to meet the 60,000 flight hour (2 lifetime) goal. This was done to make sure the head-to-head testing was not biased in favor of the FML specimen. The test results for the baseline Aluminum (Figure 9.10-11) and FML specimens (Figure 9.10-12) are compared to analytical predictions. The results generated from this test program (Figure 9.10-13) will be used to validate FML analysis methods and to verify the durability and damage tolerance benefits of FML structures.

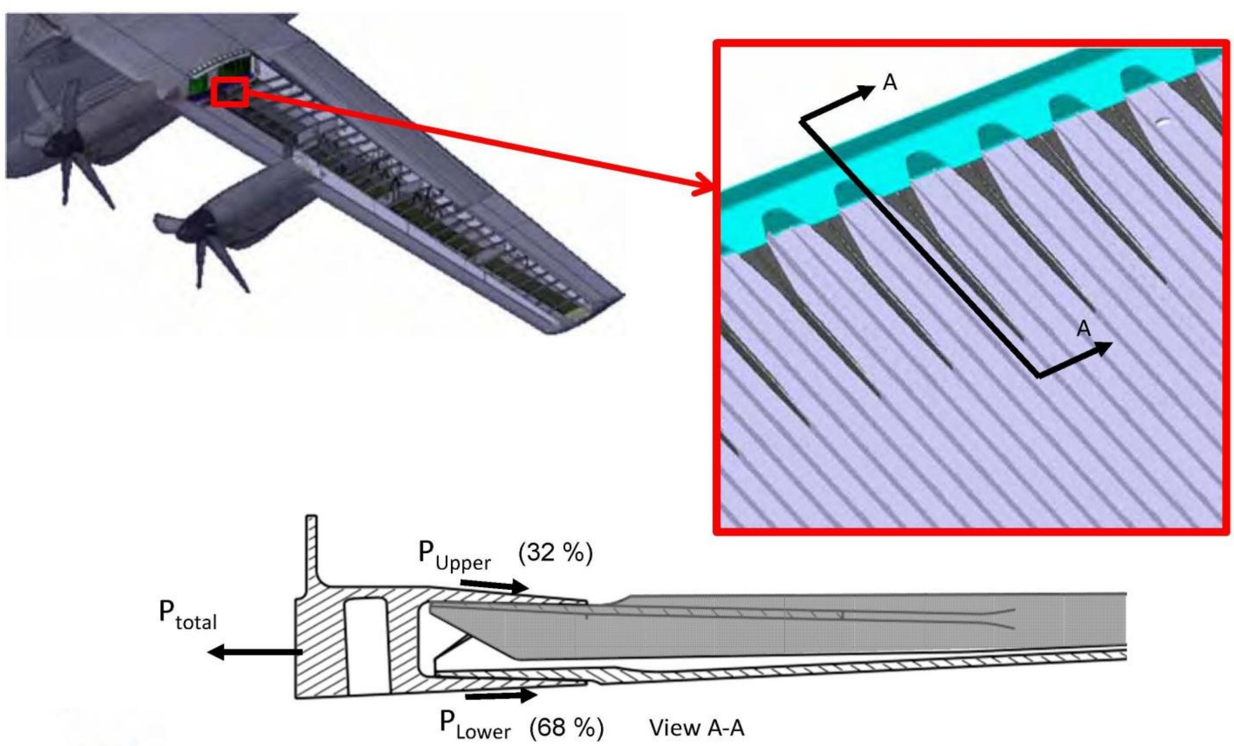


Figure 9.10-9. C-130 Outer Wing Rainbow Fitting to Skin Joint Configuration

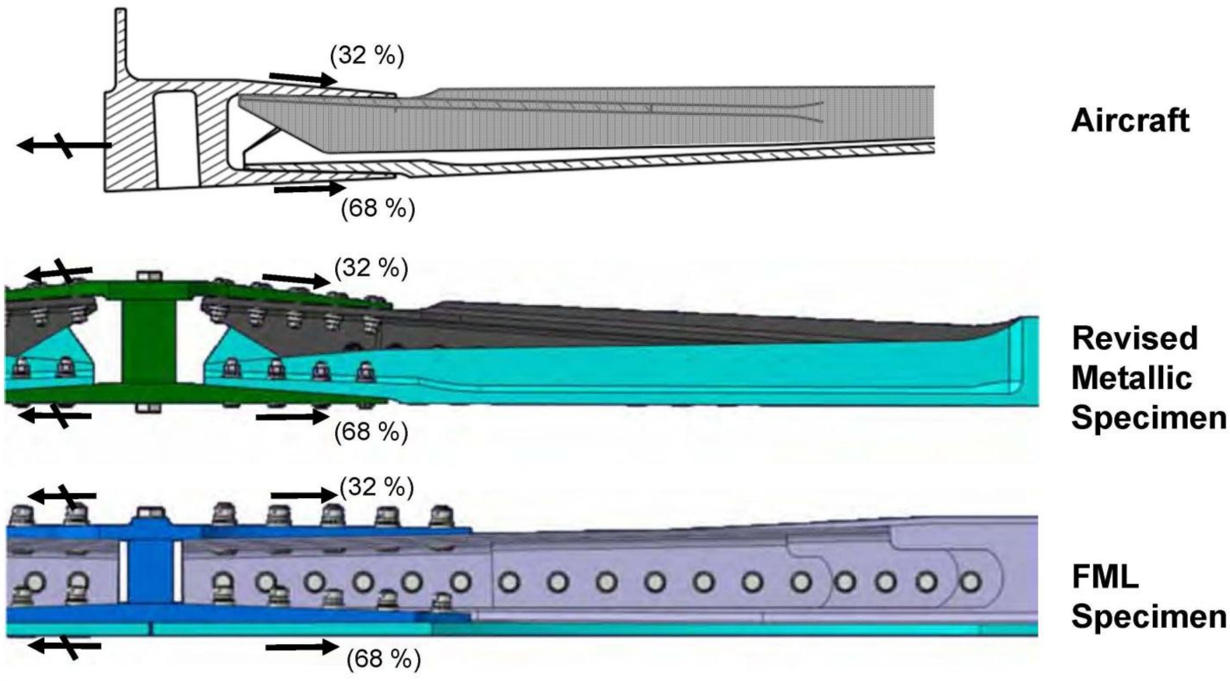


Figure 9.10-10. Specimens Designed to Match Load Distribution to Rainbow Tangs

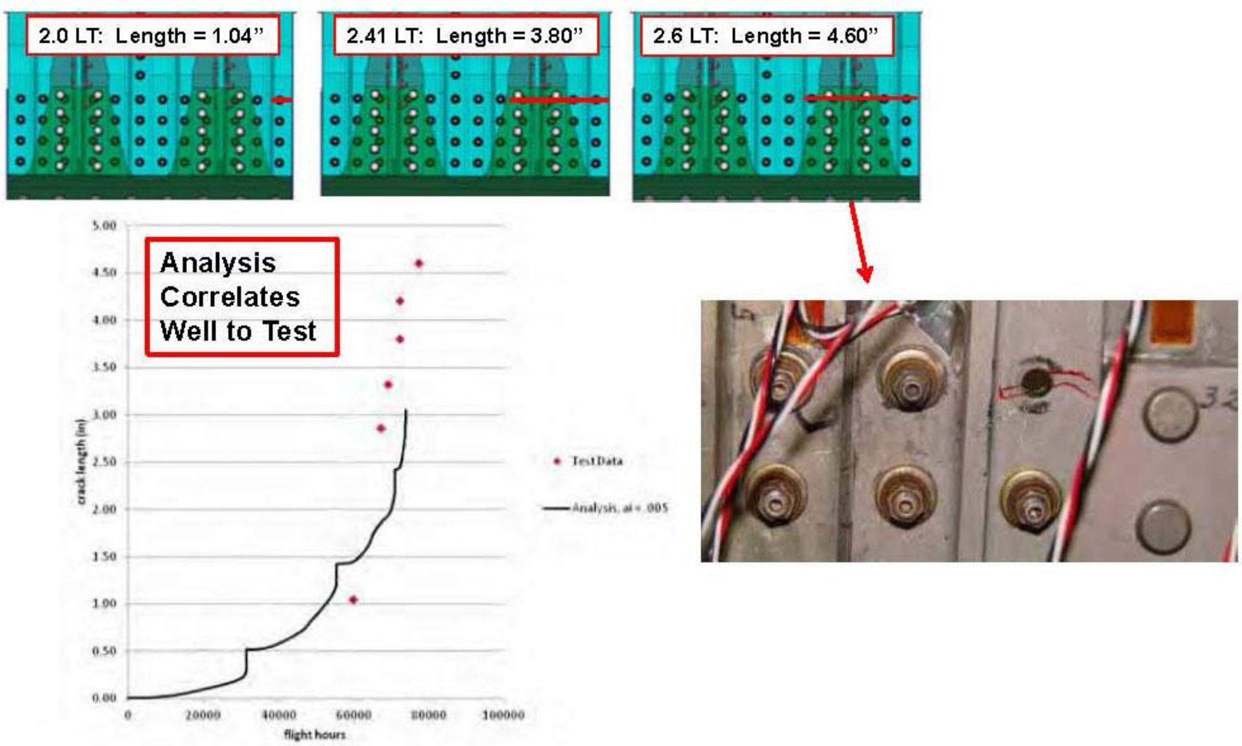


Figure 9.10-11. Fatigue Test Results for Metallic Specimen

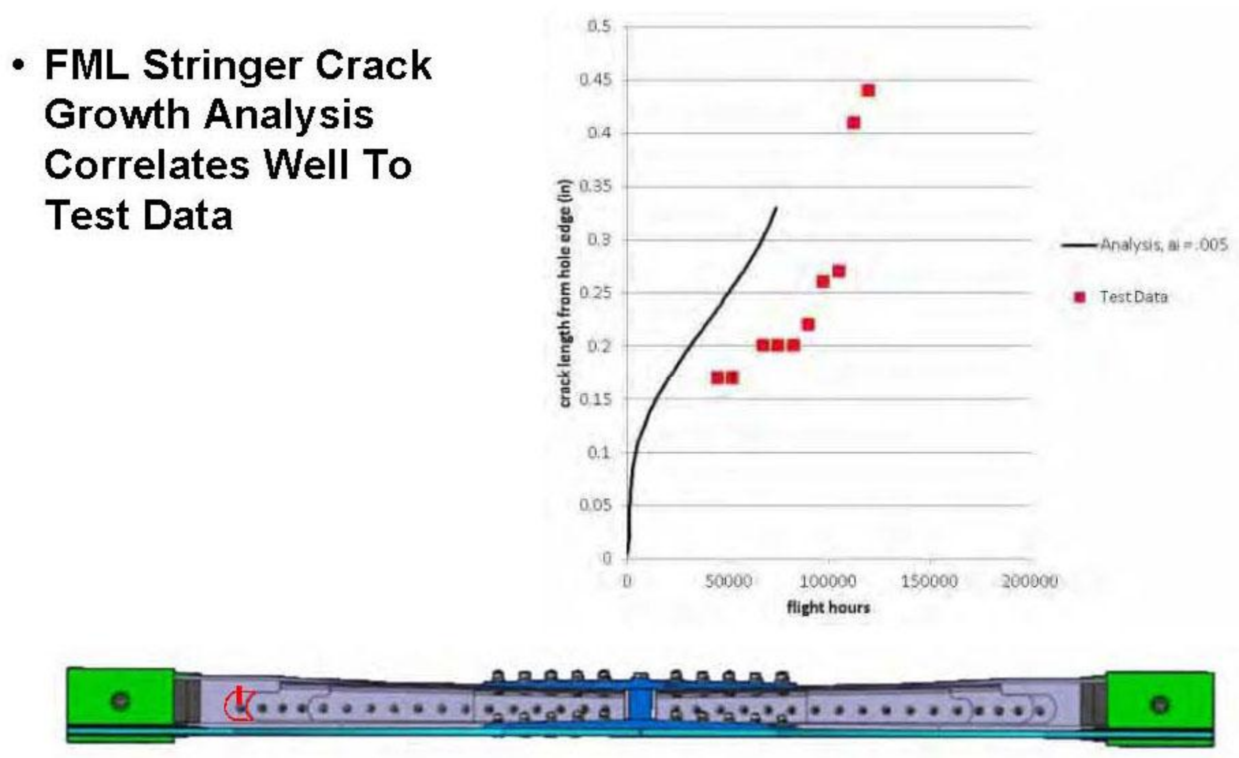


Figure 9.10-12. Fatigue Test Results for FML Specimen

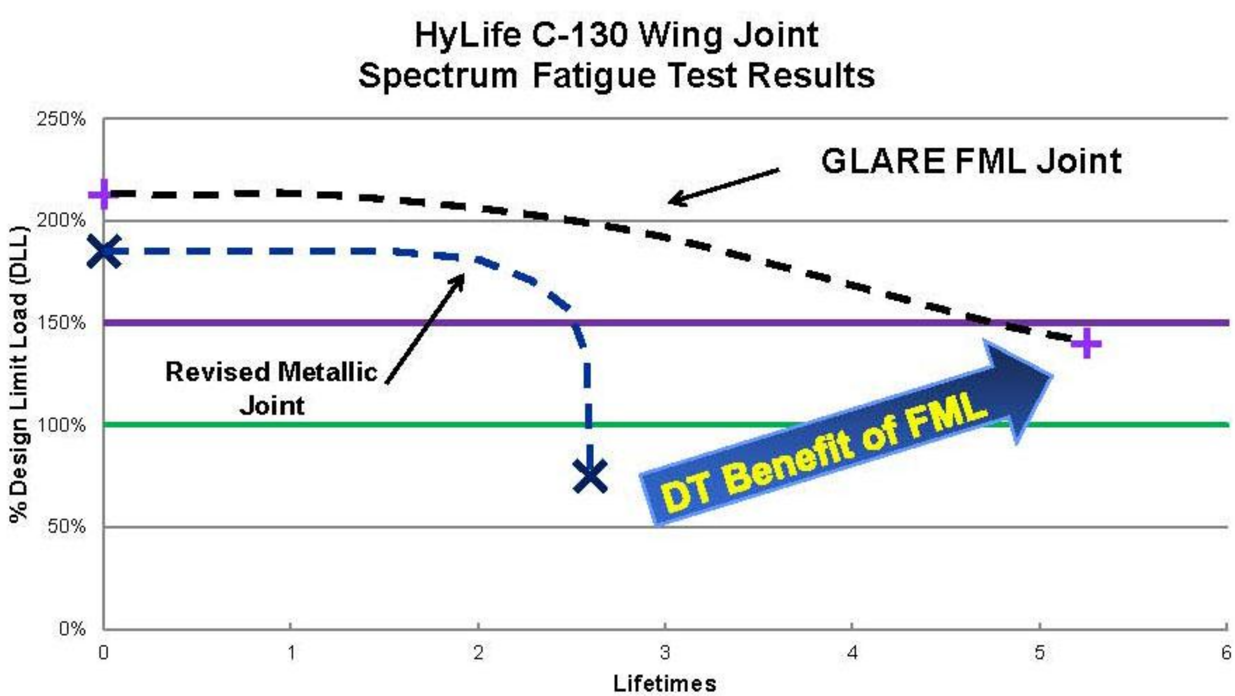


Figure 9.10-13. Test Results Summary

9.10.4. C-5 Cargo Floor Fitting Program Review as a Template for Future Life Extension Programs of the Legacy Fleet

Frank Shoup and Frank DiCocco, Alcoa Defense; David Havens, Lockheed Martin Corporation; and Michael Falugi, USAF Research Laboratory – Aerospace Systems Directorate

Corrosion and fatigue continue to plague aging aircraft, particularly in the military where aircraft are typically flown well beyond their design service lives. While the repair depots have learned how to repair corroded and fatigued aerostructures, this often provides only a temporary fix which requires continual monitoring and rework. As the aircraft fleets continue to age, repairs often become more frequent and extensive.

Recently newer alloys such as 7055 and 7085 as well as new 2xxx alloys have been developed with updated tempers that provide excellent corrosion and fatigue resistance while exhibiting higher strengths than the incumbent materials. These alloys have been fully characterized and have A & B basis allowables for many thicknesses, tempers, product forms, and process conditions. Much of the initial characterization was completed under the auspices of USAF Research Laboratory's Materials and Manufacturing Directorate in support of a "drop-in" replacement alloy.

Too often parts made of high strength alloys were originally forged or extruded. Because the requisite forging or extrusion dies are no longer available, it has become common practice to replace these parts by machining them from thick stock. While the alloy chemistry and heat treatment might be the same as the original forging or extrusion, the resulting part properties can be reduced. This has been the case with the C-5 (Figure 9.10-14) cargo floor fitting. Frequent distortion and cracking issues have been observed with the machining of replacement fittings from thick stock. Also, the current 7075-T6 replacement fittings have been susceptible to stress corrosion cracking in service.

This technical activity reviews a program which was conducted to solve the issues with the problematic cargo floor fittings on the C-5 airframe. It also demonstrates how this C-5 program was used as a template for future life extension programs of the legacy fleet (Figure 9.10-15). The C-5 fitting program was funded through the Air Force Research Laboratory and managed by the Lockheed Martin Corporation. The goal of the program was to design a new Alcoa 7085-T7452 signature stress relieved (SSR®) die forging which incorporated not just one but fifty-two different cargo floor fittings on the airframe. Therefore, one forging source could be used to manufacture any one of fifty-two different fittings. Also, the Alcoa trademark SSR® process reduces the negative effect of inherent residual stresses on the manufacturing and in-service life of the fittings. The results of this program have exceeded expectations. The new forgings are much less expensive and enable a final part fabrication reduction of 80% over the current process. This results in large savings for sustainment. Also, the 7085-T7452 SSR® material provides improved performance over the incumbent material.

The success of the C-5 fitting program has been used as a template for a much larger, fleet wide, program to address similar problematic components and supply chain issues. The "Advanced Aerospace Technologies for Modernizing the Aging Fleet" is a multi-year program funded by Air Force Research Laboratory and managed by Alcoa Defense. The scope of this program will be introduced as an example approach to provide similar near-term solutions to sustainability issues which are costly and decrease weapon system readiness.

Figure 9.10-14. C-5 Aircraft

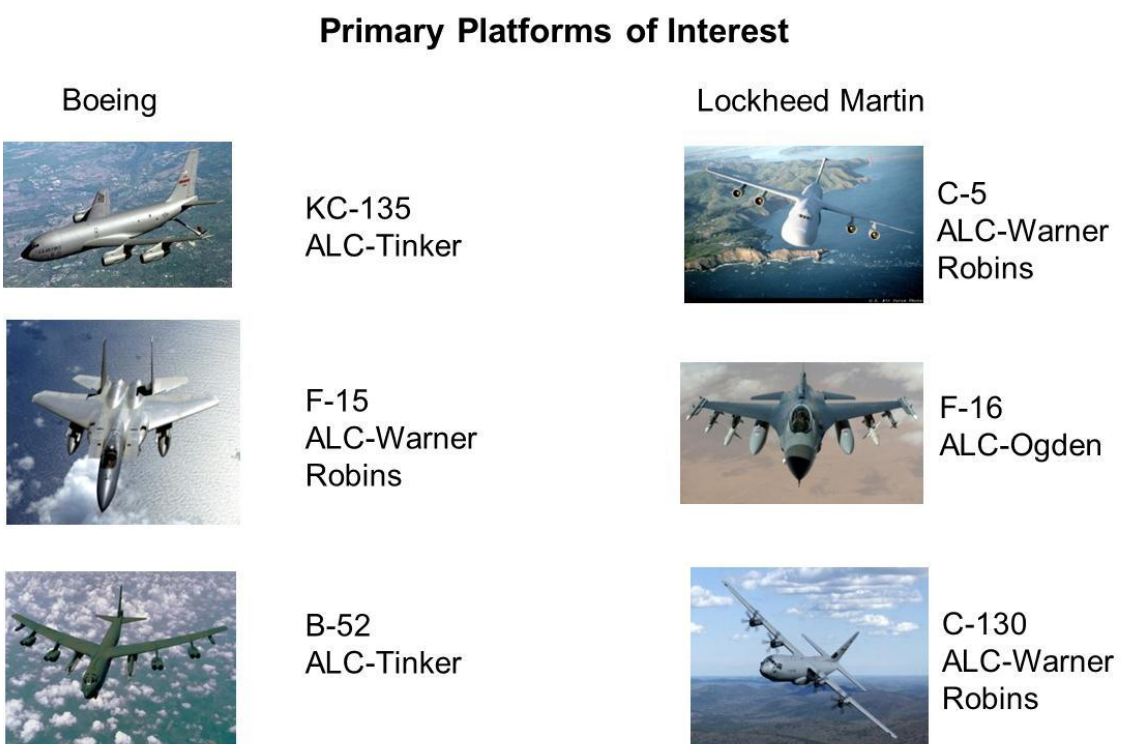


Figure 9.10-15. Legacy Aircraft Structures Modernization Opportunity

9.10.5. Future Transport Fiber Metal Laminate Complex Wing Joint Test Results

Hank Phelps and Doug Miller, Lockheed Martin Corporation

Under the Advanced Hybrid Structures for C-130 Life Enhancement (HyLife) contract research and development program, Lockheed Martin Aeronautics has developed mechanical properties for Glare fiber metal laminates (FMLs) and generated FML wing joint designs for both C-130 and future military transport aircraft. This technical activity will cover the results of spectrum fatigue and static testing of a highly loaded future transport aircraft FML wing splice joint.

The future transport joint test specimens represent a configuration where the lower surface spanwise load is spliced between wing sections via a shear joint as opposed to a tension joint (Figure 9.10-16). Tension joints like the C-130 rainbow fitting joint are a source of considerable maintenance issues for the Air Logistics Centers (ALCs). The shear joint consists of an FML skin with bonded FML stringers that run out as the stacked FML splice plates build up. The joint specimen was designed using estimated future large transport strength design loads and the mini-TWIST wing lower surface fatigue spectrum. The fatigue life requirement used to design the joint was two 30,000-flight-hour lifetimes, or 60,000 flight hours. The sizing criteria for the joint was net section yield and bearing strength.

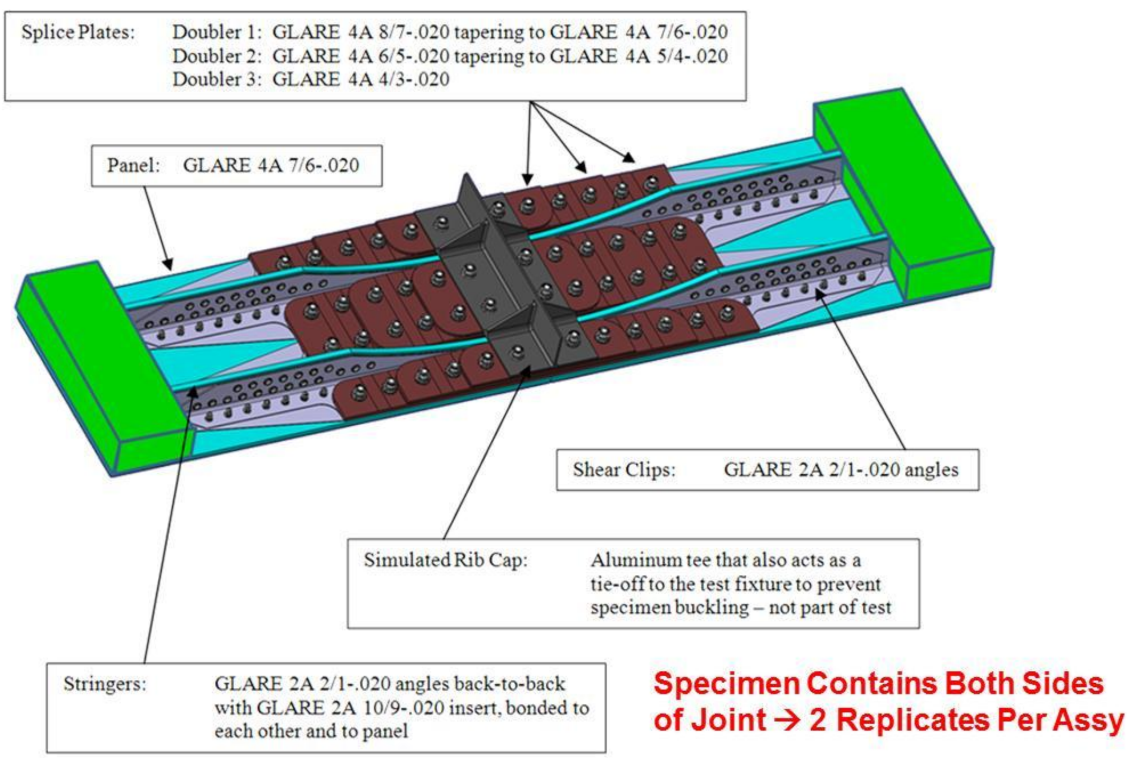


Figure 9.10-16. FML Future Joint Specimen

The joint specimen was subjected to two lifetimes of spectrum loading with periodic inspections and strain surveys. The specimen was then removed from the load fixture and a detailed inspection was performed, including removal of select fasteners for inspection (Figure 9.10-17). Anticipated cracking was found at several of the holes. The specimen was then tested for three additional lifetimes, in which the cracks never grew large enough to be visible from the outside surface. After five lifetimes the specimen was residual strength tested, and the joint demonstrated greater than ultimate strength capability (Figure 9.10-18).

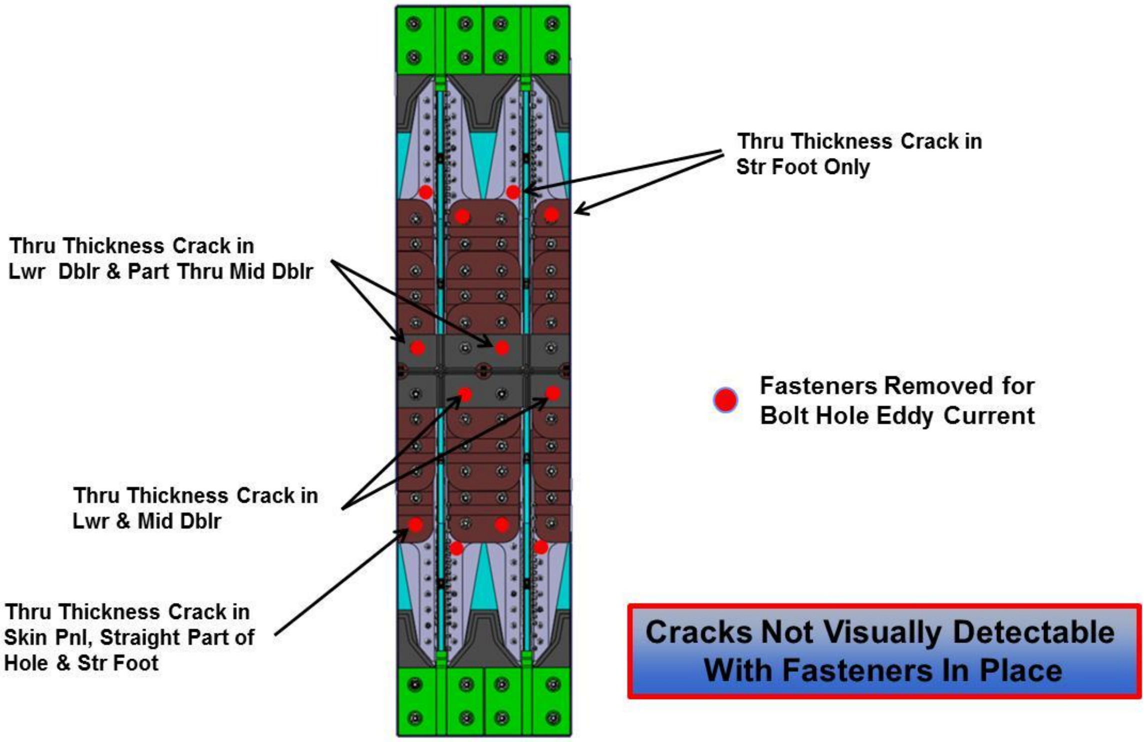


Figure 9.10-17. NDI Findings for FML Future Joint Specimen

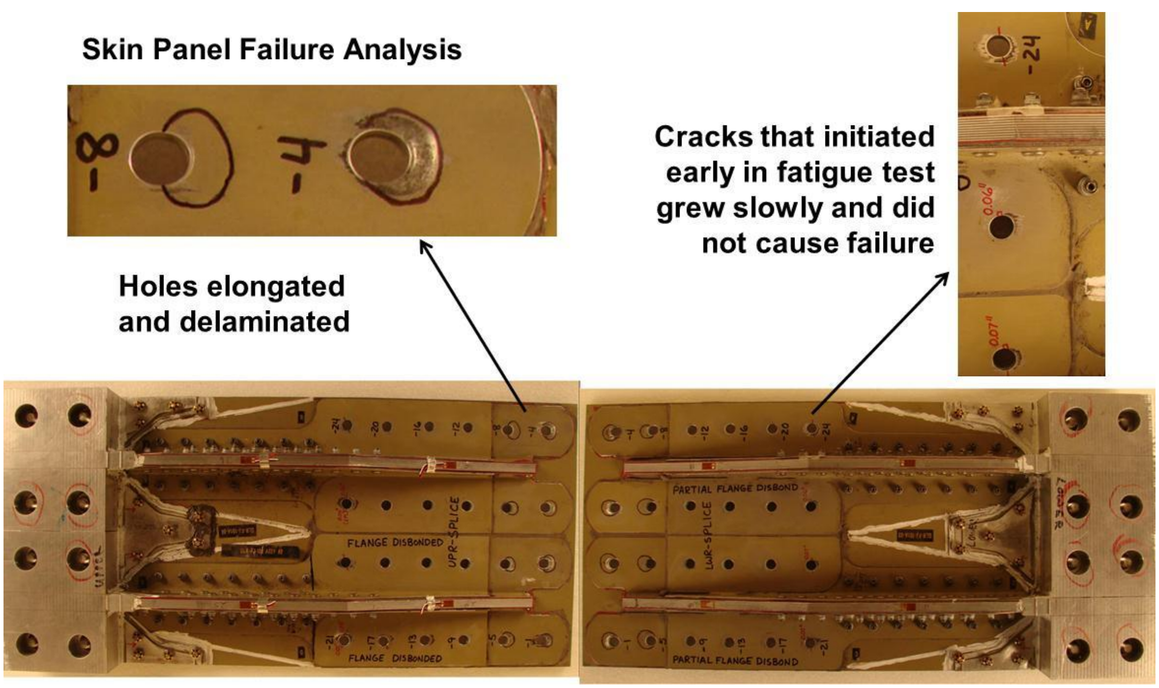


Figure 9.10-18. Residual Strength Test Results

This technical activity shows the results of the spectrum fatigue and static strength testing (Figure 9.10-19) of a highly loaded wing splice joint, demonstrating that the joint had ultimate strength capability after five lifetimes of spectrum loading with multiple cracks. The main conclusions drawn from this testing are that sizing an FML joint to static yield strength criteria ensures greater than ultimate capability, providing long damage tolerance life, and that small cracks that initiate early do not degrade the ultimate strength capability.

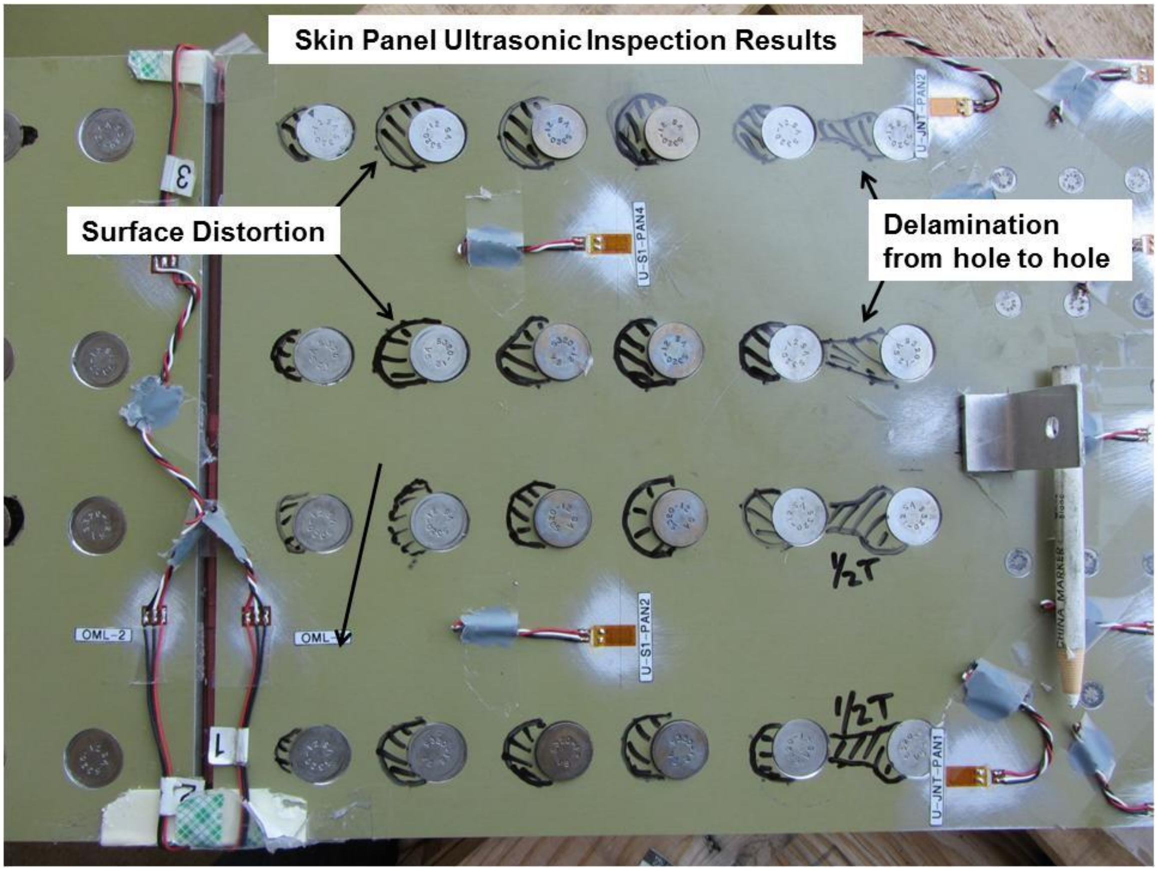


Figure 9.10-19. Static Strength Test Results

9.10.6. Incorporating Aluminum Hybrid Materials to Facilitate Life Extension in Legacy Aircraft

Michelle Creps, USAF-OO-ALC; David Hart, USAF Research Laboratory – Aerospace Systems Directorate; and Craig Masterson, The Boeing Company – Research & Technology

Test results demonstrated the A-10 upper fuselage longeron plate, which is a fatigue and fracture critical part, to be in need of replacement to meet the fleet service life extension goals. The Advanced Hybrid Structures (AHS) program is a case study investigating the technical and cost feasibility of using a fiber metal laminate (FML) design as a life enhancement solution for upper longeron plate (Figure 9.10-20).

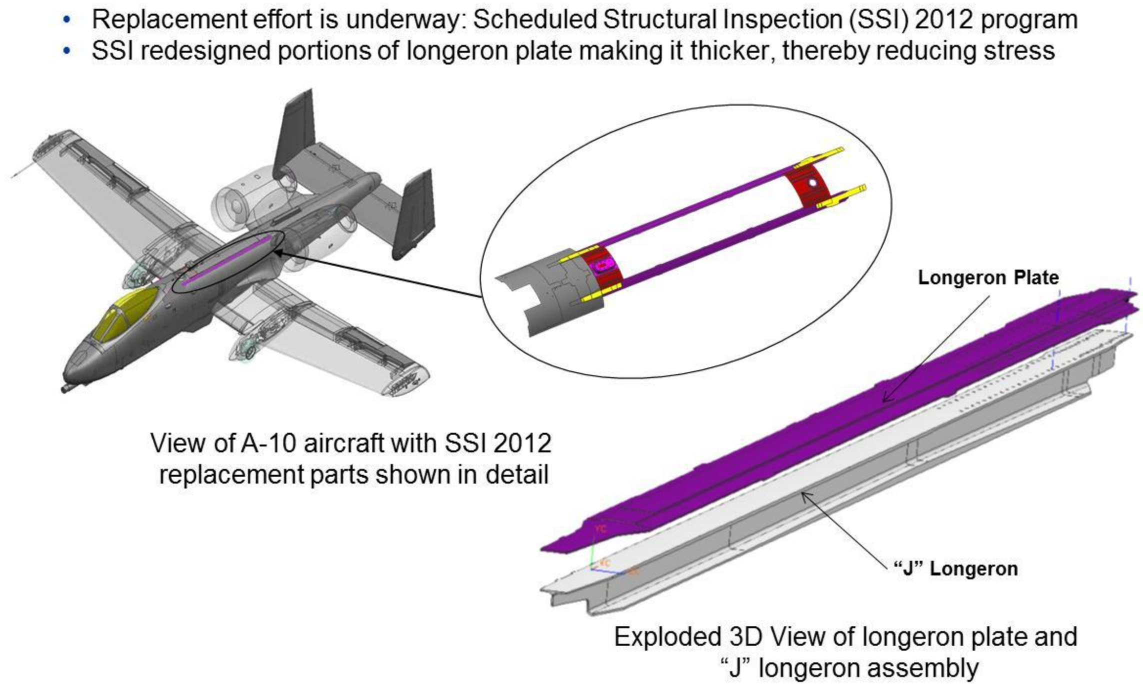


Figure 9.10-20. Replacement of Upper Longeron Plates

The AHS program includes major technical tasks to design a replacement FML longeron plate, validate analysis with a subcomponent test, manufacture a full-scale FML longeron plate and provide an installation guide specific to the installation of a FML plate. The goal of the AHS program is to establish data sufficient to transition FML technology to the USAF to meet sustainment challenges related to fatigue cracking. Development work is centered on key technology transition criteria: stabilization of materials and processes, producibility (Figure 9.10-21), mechanical property characterization (Figure 9.10-22), structural performance predictability, supportability and return on investment.

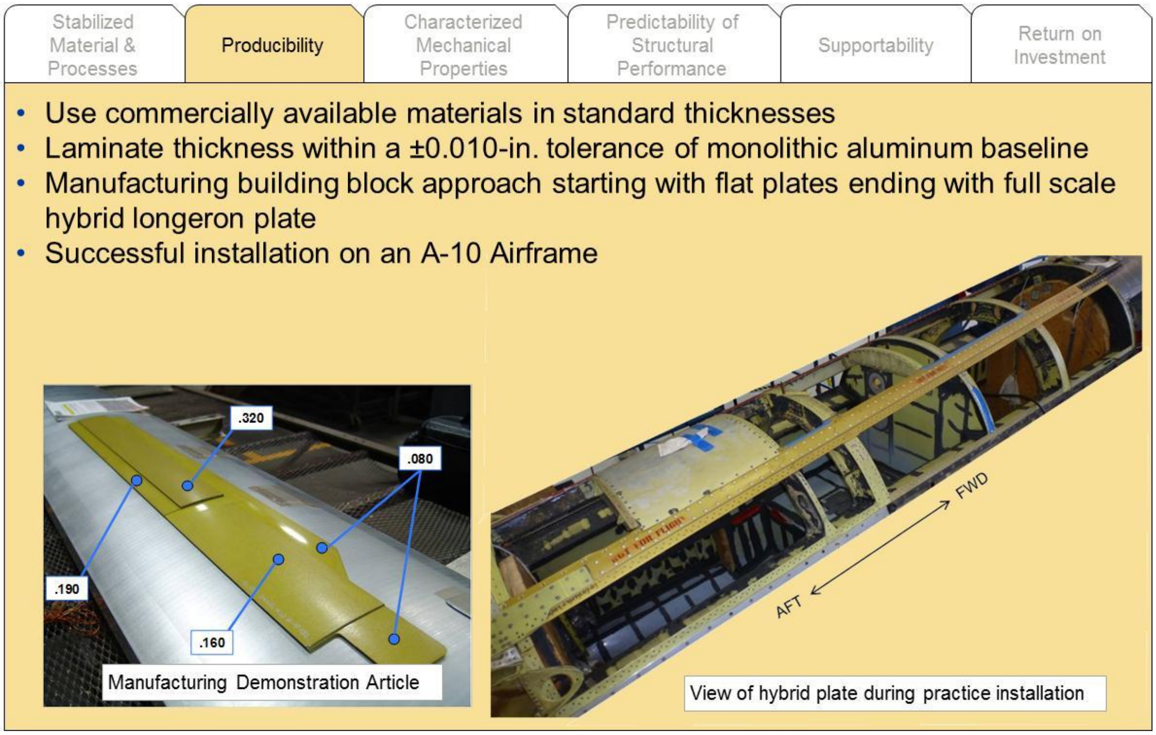


Figure 9.10-21. Producibility

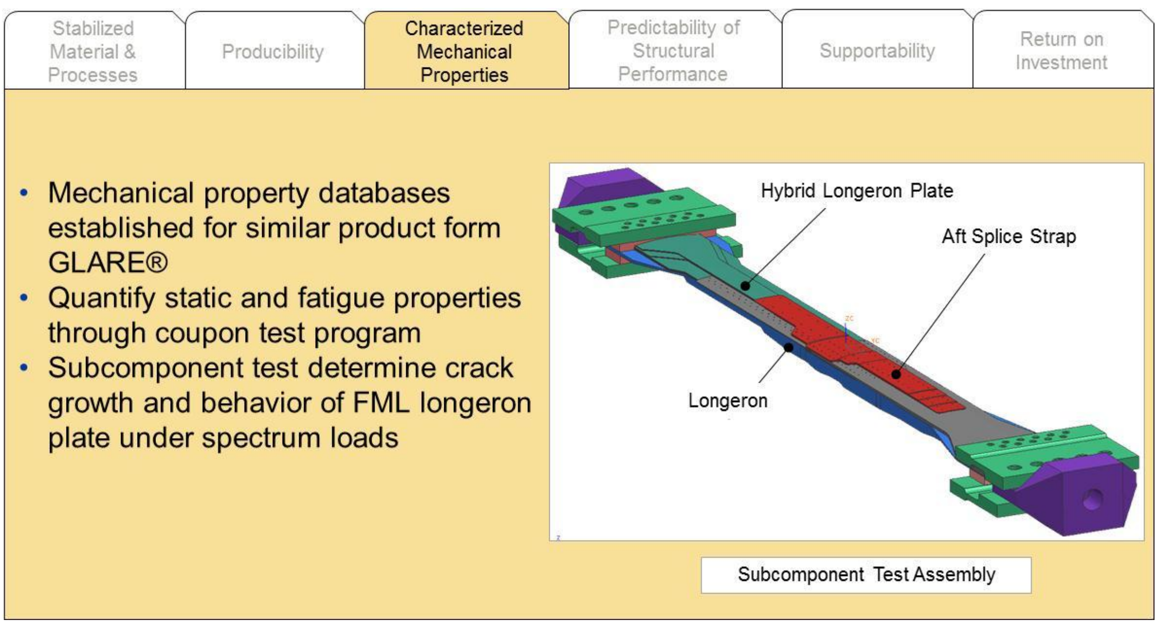


Figure 9.10-22. Characterized Mechanical Properties

9.10.7. Assessment of Advanced Aluminum-Lithium (Al-Li) for Primary Structure

John G. Bakuckas, Jr., Kevin Stonaker, Ian Y. Won and Mark Freisthler, FAA

The latest generation of advanced aluminum-lithium alloys (Al-Li) purports to offer potential weight savings compared to conventional aluminum alloys, while maintaining mechanical performance (Figure 9.10-23). Al-Li alloys, such as 2196 and 2198, have a lower density, higher modulus, and improved resistance against corrosion over the widely used 7xxx and 2xxx series alloys. Bombardier's plans propose to use these Al-Li alloys for over 20% of their new C-Series aircraft, ranging from the fuselage skin to assorted internal structures.

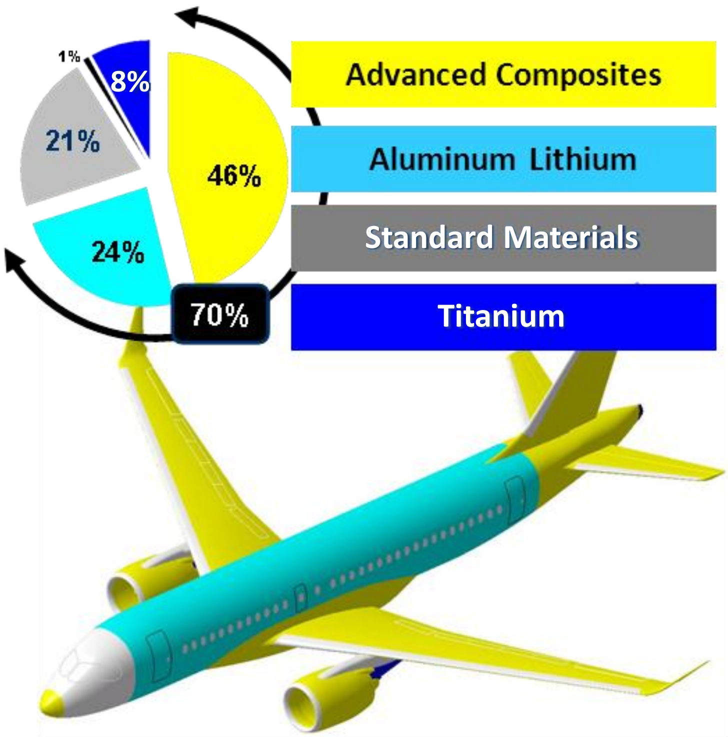


Figure 9.10-23. Application of Al-Li to Fuselage

Previous generations of Al-Li exhibited material behavior, such as low elongation and ductility, low transverse toughness, inadequate thermal stability and fatigue properties, which made them less desirable for many aerospace applications. However, the latest generations of Al-Li alloys are reported to minimize these effects while also providing much valued weight savings. This program serves as an independent assessment of static, durability, and damage tolerance capabilities of 2196-T8511 and 2198-T8 Al-Li alloys. This assessment is intended to provide the Federal Aviation Administration (FAA) with sufficient information on any unique behavior these alloys may have which may need special attention during a certification program. It is not the intention of this program to provide any detailed design information.

Resources and expertise are being leveraged with several organizations including Constellium, Bombardier, National Aeronautics and Space Administration, Naval Air Systems Command, University of Dayton Research Institute and the FAA. The 2196-T8511 alloy was supplied by Bombardier in the form of I- and T-shaped extrusions and is being tested at thicknesses of 0.06″, 0.12″, and 0.2″. The 2198-T8 alloy is supplied by Constellium in the form of sheet material and is being tested at thicknesses of

0.071″, 0.125″, and 0.25″. Each material will also be tested at three major grain directions (L, LT, and 45°) plus two mid-points (22.5° and 67.5°) to assess the extent of anisotropic behavior in the materials.

Based on experience on earlier generations of Al-Li alloys, several properties are being assessed and compared with baseline 2024-T3 and 7075-T6 alloys, including static properties, fatigue life and fatigue crack growth performance, corrosion resistance, machinability, as well as the damage and durability aspects. The initial phase of testing reveals that the static properties of 2198, namely, tensile yield and ultimate strengths are above MMPDS published A and B values. In addition, anisotropic behavior is more pronounced in thicker gage material particularly in the 45° direction, Figure 9.10-24. Measured fatigue crack growth rates for the 2024-T3 and 2198 alloys were similar in the threshold and mid-range regions. The 2198 alloy displayed longer fatigue life in all grain orientations than the 2024-T3, material (Figure 9.10-25).

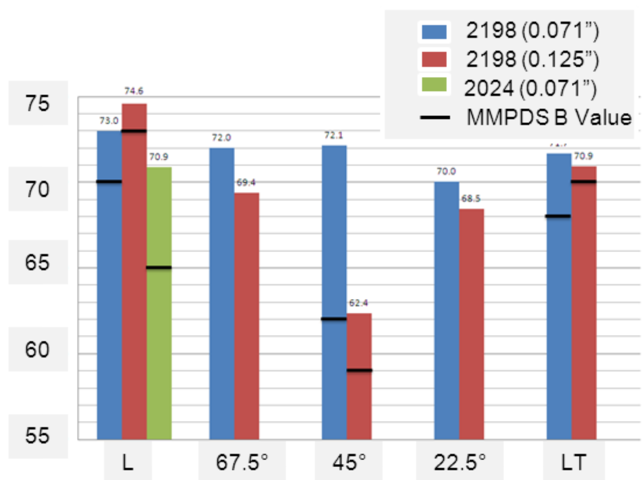


Figure 9.10-24. Static Properties

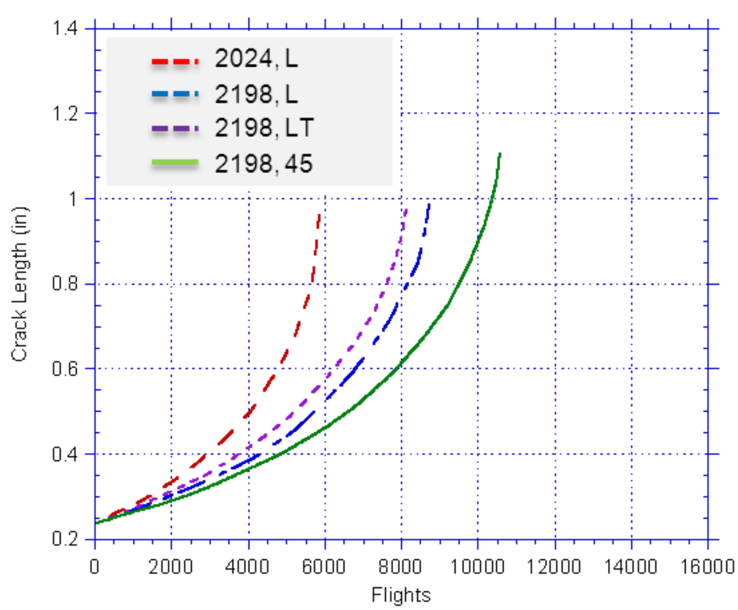


Figure 9.10-25. Fatigue Properties

A major output from this program will be the development of a knowledge base to allow FAA engineers to be aware of any unique material properties of Al-Li alloys to ensure the safe and efficient implementation and application of Al-Li material for airplane primary structure.

9.10.8. Fatigue Evaluation of Low Conductivity 7050-T7452 Forged Block

Mark Ofsthun, Spirit AeroSystems, Inc.

In the past 15 years the manufacturers of commercial airplanes have turned to machining monolithic structures from large blocks or thick plate. The part count reduction has led to significant cost savings over built-up structure. However, monolithic structure relies on machining large parts into relatively thin parts (see Figure 9.10-26). In a recent study conducted by Spirit AeroSystems, it was noted that when machining parts from 7050-T7452 forged block, if the cutter dwelled too long at a point then heat would build up and locally the part would experience some lowering of the material's conductivity (about a 25% reduction in conductivity). Lower conductivity is a significant indicator of lower static properties. It is unknown if lower conductivity has any impact on the material's fatigue properties. Therefore, Spirit conducted a test on material with known local areas where conductivity was lower by 25%.

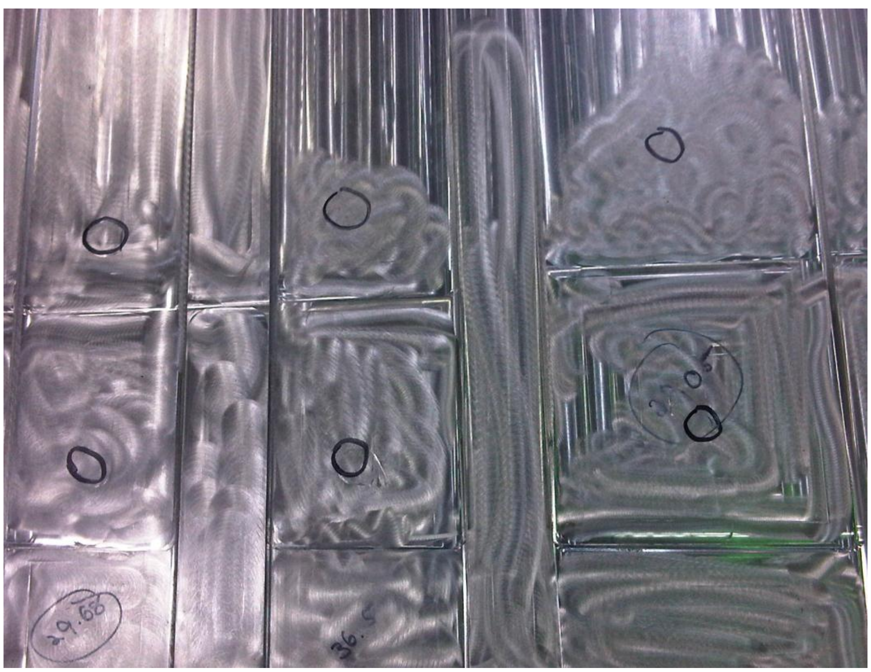


Figure 9.10-26. Monolithic 7050-T7452 Forged Block Fatigue Test Specimens

Spirit's investigation was comprised of static tension specimens with 100% of the cross section in the low conductivity as well as specimens with low conductivity that covered about 70% of the specimens' cross section. The results of the static tests indicated that yield strengths were affected more than ultimate strengths and that the size of the low conductivity zone affected the static strength (See Figure 9.10-27). In addition to the static evaluation, spirit conducted fatigue tests of flat notched specimens with a K_t of approximately 1.5. Specimens with the low conductivity right on the edge of the notch and baseline (normal conductivity) specimens were tested head to head. Figure 9.10-28 shows the fatigue coupon and where the low conductivity zone was located. The results of the fatigue testing are provided in Table 9.10-1 which shows that the low conductivity had no effect on the fatigue life of the 7050-T7452 forged block.

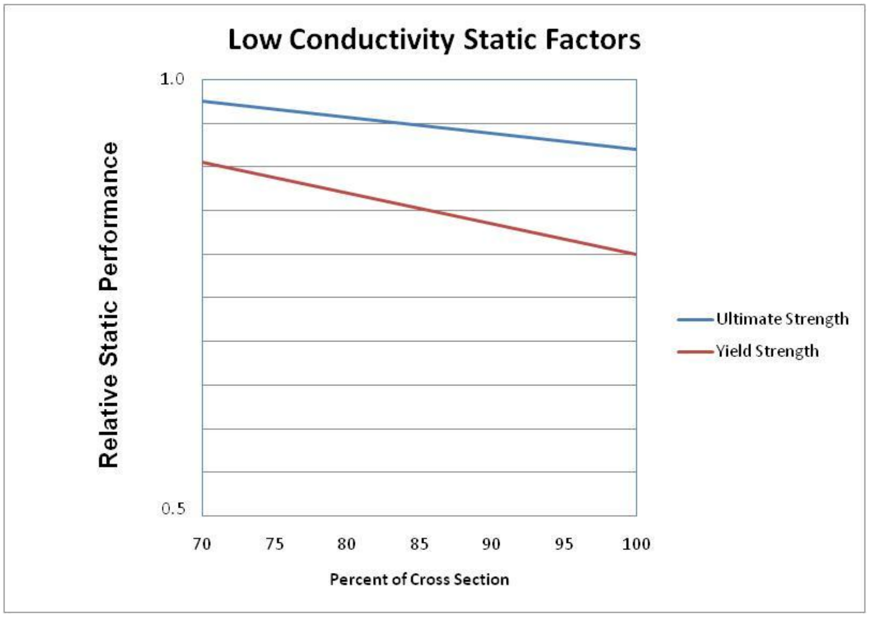


Figure 9.10-27. Static Test Results of Low Conductivity 7050-T7452 Forged Block

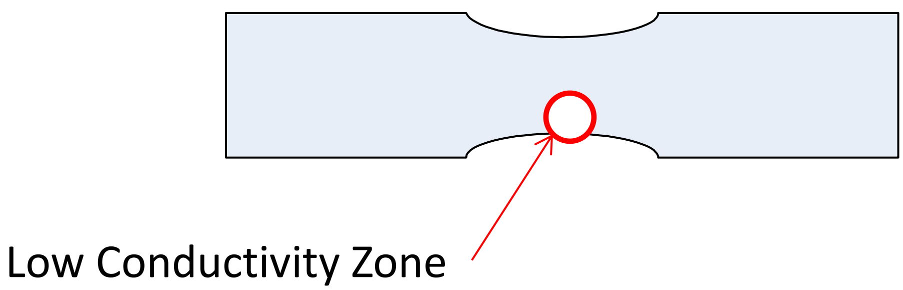
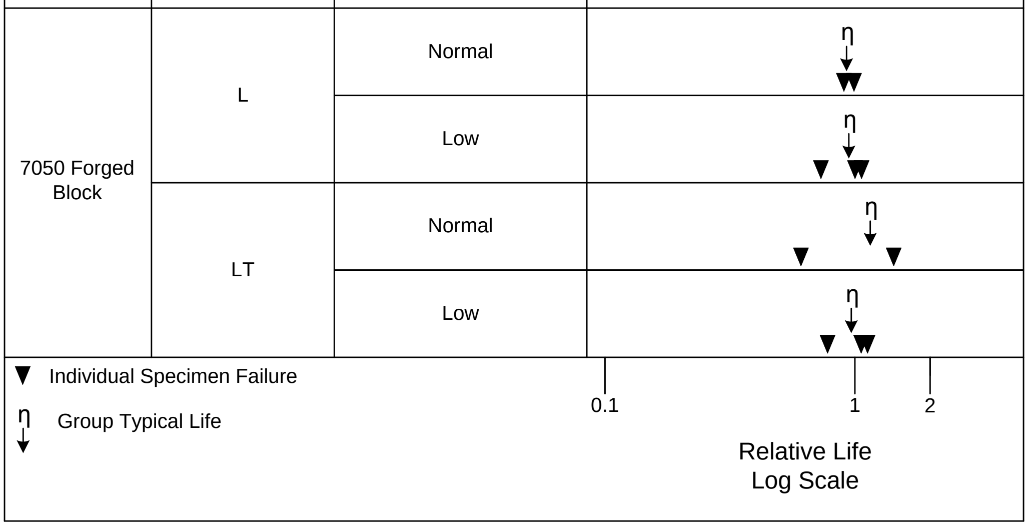


Figure 9.10-28. Fatigue Test Specimens

Table 9.10-1. Fatigue Test Results

Material	Grain Direction	Normal / Low Conductivity	Kt = 1.5 Fatigue Results
7050 Forged Block	L	Normal	
		Low	
	LT	Normal	
		Low	



The lessons learned in this study was that the machining of monolithic parts needs to be controlled to minimize the potential of overheating the material which can have detrimental effects on the material’s yield strength and to a lesser degree, the material’s ultimate strength. Interestingly, the fatigue properties were unaffected for this material. It may be that the ductility increased enough to overcome the reduction in static strength.

9.11. OVERVIEWS

9.11.1. Certification of Structural Integrity F-35 Lightning II Program

Carl McConnell, Lockheed Martin Corporation

The F-35 (Figure 9.11-1) Structures Certification Plan supports Initial Flight Clearances and Final Airworthiness for all three major variants on the JSF program through the preparation, review and approval of a myriad of certification-level products (Table 9.11-1 and Figure 9.11-2). These products range from Building Block Test Reports which provide materials allowable data to drawing release Stress and Durability Analyses. These products also include items such as Final Loads Reports written based on flight test results and Full-Scale Vehicle-Level Test Assessment Reports which verify the designs' Structural Strength and Durability, evaluate the validity of the predictive analysis models and show the correlation of the Structural Analyses which are used for the Force Structures Maintenance Plans. This technical activity presents an overview of the F-35 Structures Certification Plan and Schedule, accomplishments to-date, and an outline of the remaining structural engineering tasks necessary to support the completion of the F-35 Systems Demonstration and Development program.

Figure 9.11-1. F-35 Aircraft

Table 9.11-1. F-35 ASIP Follows MIL-STD-1530C

Full-Scale Development			Force Management	
Task I Design Information	Task II Design Analyses & Development Testing	Task III Full Scale Testing	Task IV Certification & Force Management Development	Task V Force Management Execution
ASIP Master Plan Design Service Life and Design Usage Structural Design Criteria Durability & Damage Tolerance Control Program Corrosion Prevention and Control Program Nondestructive Inspection Program Selections of Materials, Processes, Joining Methods & Structural Concepts	Material and Joint Allowables Loads Analysis Service Loads Spectra Chemical/ Thermal Environment Spectra Stress Analysis Damage Tolerance Analysis Durability Analysis Corrosion Assessment Sonic Fatigue Vibration Analysis Mass Properties Analysis	Static Tests First Flight Verification Ground Tests Flight Tests Durability Tests Damage Tolerance Tests Climatic Tests Interpretation & Evaluation of Test Results	Certification Analyses Strength Summary And Operating Restrictions Force Structural Maintenance Plan Loads/Environment Spectra Survey Methodology Individual Airplane Tracking Program Methodology Individual Aircraft Tracking Program Development	Individual Airplane Tracking Data Loads/Environment Spectra Survey ASIP Manual Aircraft Structural Records Force Management Updates Recertification

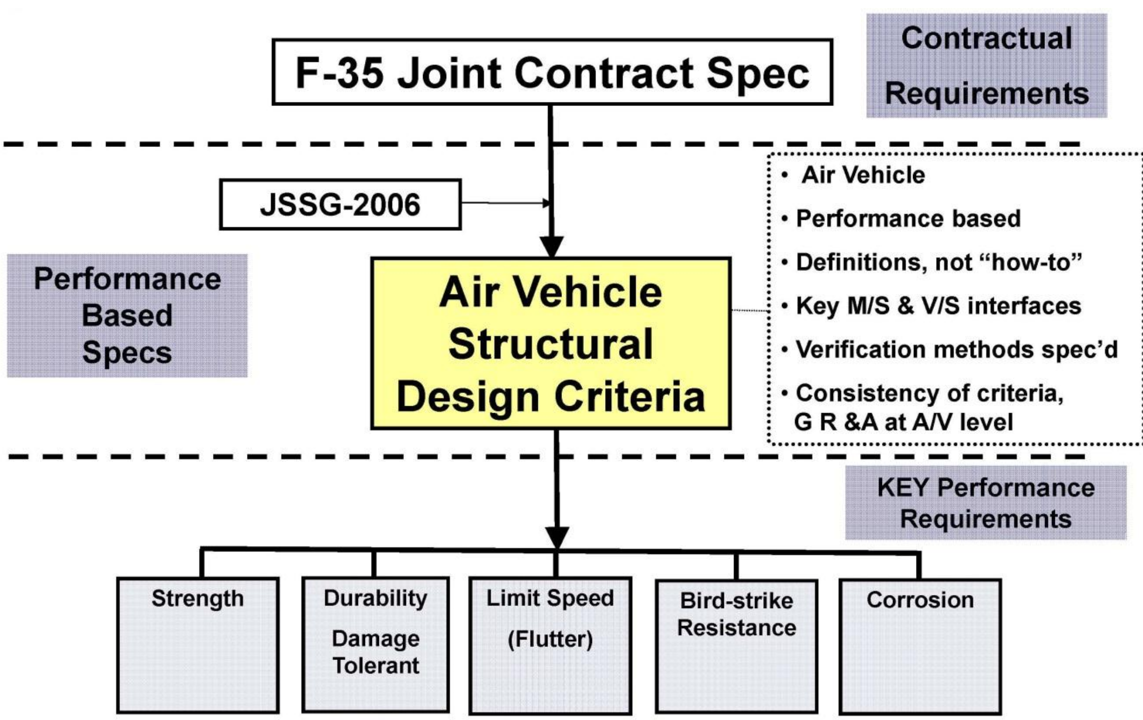


Figure 9.11-2. Structural Integrity Requirements Flow-Down

9.11.2. F-22 Force Management Lessons Learned: The First Five Years

John McClure and Robert Bair, USAF F-22 Program Office; Suresh Patel and Christopher Black, Lockheed Martin Corporation

As the F-22 fleet ages and production ends, the F-22 Program has learned many different lessons from the first five years of the fleet's usage. Some critical areas covered in this technical activity include data capture, fleet usage, and maintenance. Data capture is a very important part of sustaining the fleet. If the majority of the data related to fleet usage can be captured, inspection intervals can be accurately determined and aircraft availability can be optimized. This technical activity discusses various shortfalls of data capture in the F-22 Program, such as how significant data is lost when aircraft make unanticipated trips, such as TDYs, base visits, and air shows. In hindsight, data capture could have been significantly improved by planning for different data capture shortfalls like this. Another area covered in this technical activity is fleet usage. Fleet usage is a crucial part of force management and many different fleet usage trends could have been investigated with the F-22. For instance, most fleet's training aircraft are flown much more severely than the rest of the fleet (Figure 9.11-3). This technical activity will discuss this usage variation, and how two different fleet baseline spectrums could be used. One spectrum would incorporate the extreme usage of the training fleet, while the other spectrum would take into account the more benign usage of the rest of the fleet. Another factor of fleet usage this technical activity details is the base airspace where the fleet is stationed. Usage can be more severe at bases that have a small airspace near the end of the runway, causing fully fueled aircraft to quickly turn, causing higher NzW loads and therefore more severe usage. Different fleet usage parameters such as these can be adjusted to decelerate aircraft aging. Many significant lessons were also learned and are detailed in this technical activity in the area of maintenance for the F-22 fleet. A good example of this was Non-Destructive Inspections (NDI). The F-22's stealth characteristics make it very difficult to access the airframe, so any type of NDI requires a significant amount of aircraft downtime (Figure 9.11-4). Ideally, the aircraft design could have focused on accessibility by using different techniques such as form in place (FIP) panels that allow greater accessibility into the aircraft (Figure 9.11-5). Also, different NDI techniques could have been developed during design instead of waiting until after the full-scale-fatigue test (FSFT) was completed and aircraft had already been fielded.

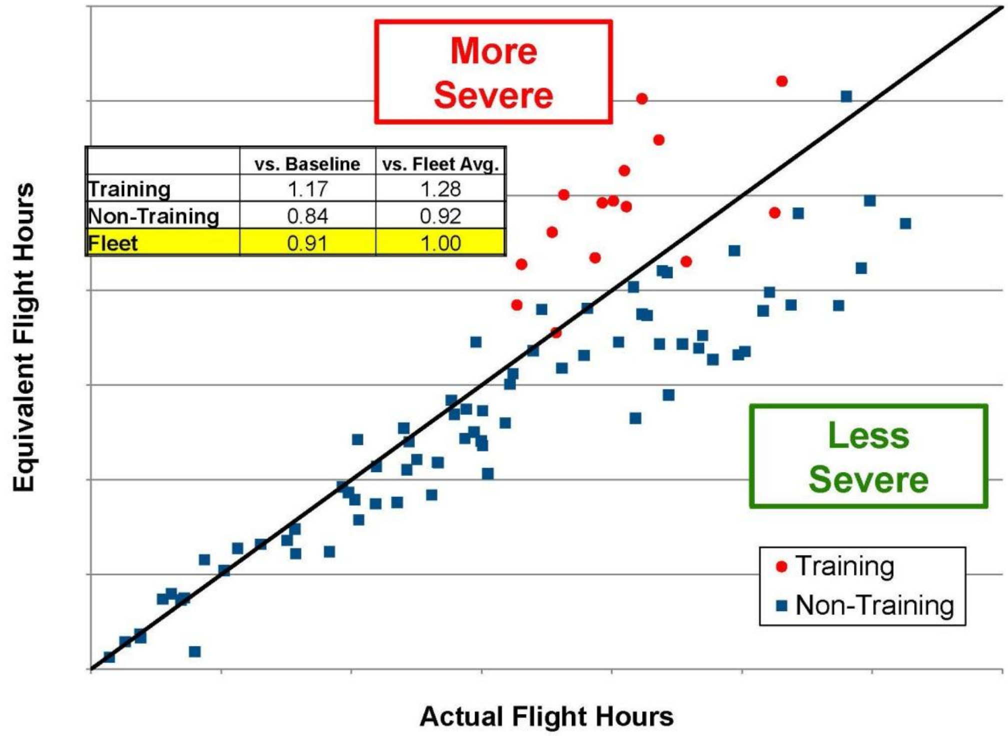


Figure 9.11-3. Fleet Usage Severity

- **Specialized equipment needs to be developed**
  - **New NDI probes developed to replace legacy probes**
  - **Helps address geometry and flaw size challenges**

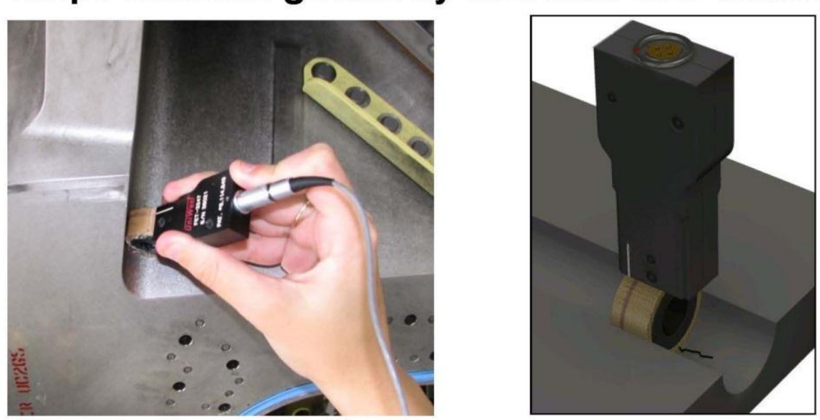
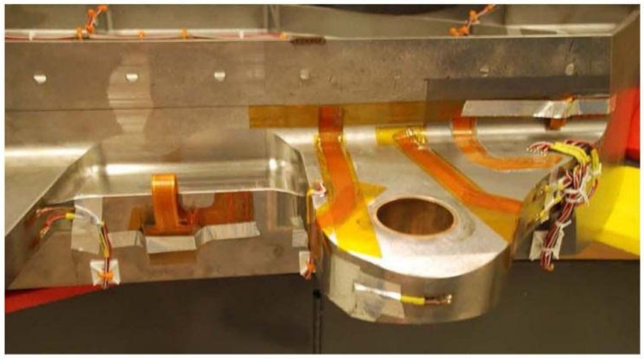


Figure 9.11-4. NDI Lessons Learned - Geometry

- **Development of SHM technologies**



- **Incorporate Form in Place (FIP) Panels**
  - **Planned for 17 FIP panels initially**
  - **Added 5 before EMD ended (23 total)**
  - **1 FIP recently added (24 total)**
  - **11 more in work (35 total)**

Figure 9.11-5. NDI Lessons Learned – Accessibility

9.11.3. ASIP: An ACC Perspective

Robert Norcross, USAF Air Combat Command

Fiscal constraints are once again creating change. This change has sparked a great interest in the capabilities of the Aircraft Structural Integrity Program (ASIP) to help determine what we need to do to keep aircraft in service longer. Studies have been championed by Air Combat Command and the Air National Guard to see how ASIP is being used to change the way we fly the aircraft in order to retain the aircraft until future aircraft are fully fielded. ACC is committed to revitalizing the ASIP and Individual Aircraft Tracking (IAT) programs and has appointed a MAJCOM ASIP POC for command-wide ASIP involvement. The initial steps in ACC revitalization efforts began with a Six Sigma 8-Step problem-solving event for low IAT data capture rates that led to policy changes that include monthly reporting of IAT valid data capture rates and guidance on downloading recording hardware during extended aircraft down time and prior to hardware cannibalization. Communication with the units is key to the revitalization effort, so there is a better understanding of the use of ASIP/IAT data and that ASIP is an active program and not a passive program.

9.11.4. Leveraging USAF ASIP to Enable Mobility Air Forces

Jason Avram, USAF Air Mobility Command

Mobility Air Forces are a key enabler to provide the right effects at the right place at the right time, both for the defense of the nation and for humanitarian support (Figure 9.11-6). This critical MAF mission demands high weapon system effectiveness and operational flexibility to meet its worldwide responsibilities, often requiring AMC to adjust on the "fly" to new mission requirements (humanitarian relief, Libyan operations, volcanic ash, etc.). As the MAF fleet continues to age (Figure 9.11-7), especially in a fiscally constrained environment, fleet Aircraft Structural Integrity Programs (ASIPs) are critical enablers to ensure flexibility and accuracy in sustainment decision-making, allowing all second and third order effects to be accounted for when determining whether situations call for structural inspect and repair, or planned structural modification. Strict adherence to ASIP, along with close coordination between AMC and ASIP managers, is key to capture aircraft usage data, understand and predict effects from actual aircraft usage, and ensure proper sustainment practices are in place and effective to enable mission effectiveness for years to come, often past aircraft original design service lives. AMC is committed to maintaining a robust MAF ASIP with a Command Engineer on staff to provide leadership and engage with SPO ASIP managers to ensure coordination and appropriate MAJCOM visibility of key ASIP concerns. Additionally, AMC weapon system managers work to educate field units on the importance of adequate ASIP usage data capture for understanding and properly managing the effects of the aircraft operational environment on overall service life (Table 9.11-2).

Figure 9.11-6. AMC Mission

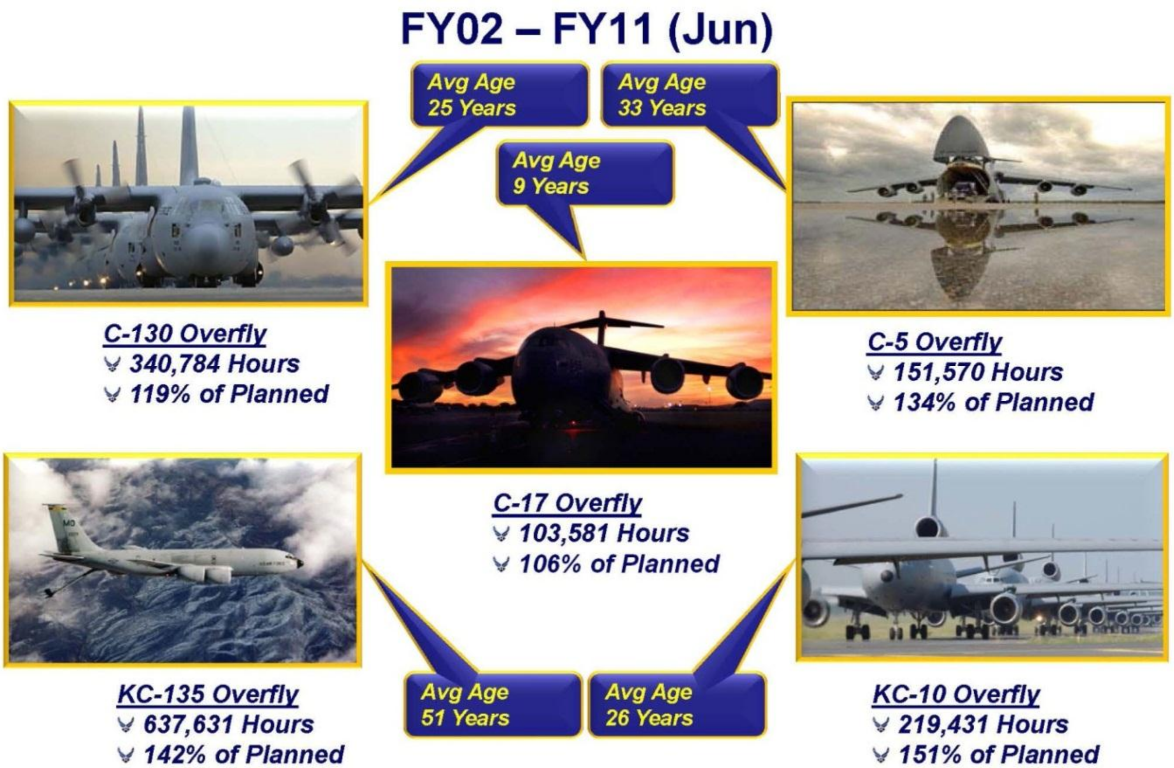


Figure 9.11-7. Demand on Mobility Fleet

Table 9.11-2. Snapshot of AMC ASIP Usage Data Collection

- **BLUF: Except C-130's, AMC ASIP Usage Data Capture Rates are Healthy Compared to Standards; Mitigating Actions are In Place for Minor Deficiencies**

	2011 ASIP Review Data	
WEAPON SYSTEM	L/ESS Status (Rqmt-20%)	IATP Status (Rqmt- 90%)
C-130E/H	Red - Not implemented	Green - 97% Capture Rate
C-130J	Green - 70% Capture Rate	Yellow - 70% Capture Rate
C-5A/B/C	Green - large data history and historically stable mission criteria	Green - 97% Capture Rate
C-5M	Green - As C-5M fleet grows, requires L/ESS funding to maintain 20% of fleet data capture; denied funding in FY13 POM.	Green - 97% Capture Rate
C-17A	Green - well above 20% limit.	Yellow - 87% Capture Rate
KC-10	Green	Green
KC-135	Green	Yellow - 87% Capture Rate

9.11.5. The "WFD" Rule – Have We Come Full Circle?

Robert Eastin and Walter Sippel, FAA

Fatigue evaluation requirements that must be complied with to gain type design certification of a civil transport category airplane are contained in Title 14 Code of Federal Regulations (14 CFR) 25.571. The primary objective of the requirements has always been to prevent catastrophic failures due to fatigue during the operational life of the airplane. However, over the years the requirements have changed as the fatigue knowledge base has increased. The most recent changes to the § 25.571 requirements are embodied in the Federal Aviation Administration's (FAA) widespread fatigue damage (WFD) rule, which is the subject of this technical activity (Figure 9.11-8). The WFD Rule was published as a final rule on November 15, 2010, and became effective on January 14, 2011. It is a result of years of experience and lessons learned while striving to maintain continued airworthiness relative to the threat of normal metal fatigue in transport category airplanes. This technical activity reviews some of the key events and fatigue management requirements that preceded the WFD Rule to illustrate how fatigue management approaches have evolved over the last 55 years. It then provides a detailed overview of the WFD Rule, which includes supporting definitions and concepts. Three components compose the WFD Rule. The first includes new requirements for certification of new transport category airplane designs. The second and third include new requirements for design approval holders and operators of certain existing airplanes, respectively. This technical activity compares the current damage-tolerance requirements to prior requirements for certifying transport category airplane designs. The comparison shows that the scope has increased dramatically from what was required 23 years ago (Figure 9.11-9). Section 25.571 now requires applicants to demonstrate by full-scale-fatigue test evidence that WFD will not occur during the operational life of the airplane. Furthermore, the primary means for managing normal fatigue cracking is by replacing or modifying structure rather than by solely relying on special directed inspections. This technical activity concludes that, with some qualification, we have returned to the fatigue management philosophy that we started with over 55 years ago wherein the likelihood of significant fatigue cracking in service is minimized by proactively limiting the allowable operating life of the structure.

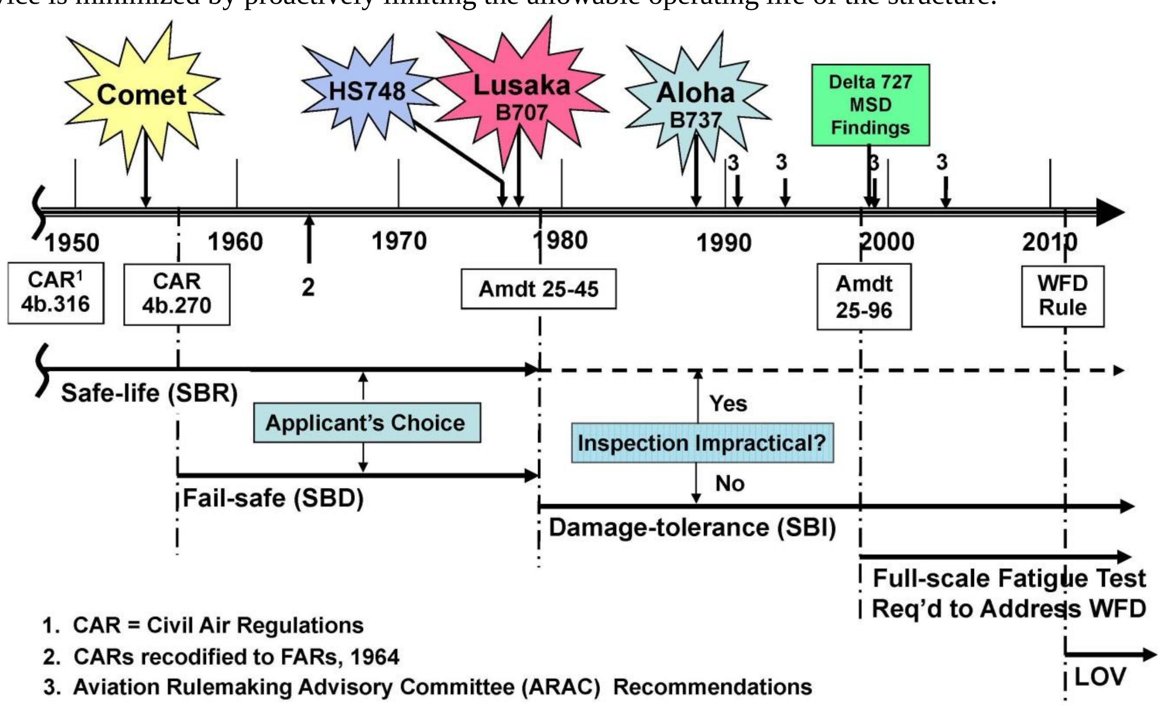


Figure 9.11-8. Evaluation of Part 25 Fatigue Requirements

89,680 total flight cycles (DSG = 75,000 flight cycles)

Figure 9.11-9. Aloha Accident in 1988

9.11.6. SLEP Planning: Lessons Learned in Light of Production Shutdown

Christopher Ifft and George Crosthwaite, USAF F-22 Program Office

The F-22 production line has been limited to 195 aircraft, with 8 already out of service (retired, MXG trainers, or lost) (Table 9.11-3). Based on current fleet usage, the 8,000-hour service life will result in the fleet being retired by 2045. In an effort to provide future planning support to Air Combat Command and the program office, an effort was initiated to look at future F-22 SLEP requirements, especially considering the difficulties presented by the loss of an active production line to support future activities. The preliminary SLEP evaluation looked at the challenges to extend the service life to either 10,000 or 12,000 hours for aircraft effectivity 63-195 which was determined to be the most economical portion of the fleet to extend life. The earlier aircraft (4010-4062) were aircraft that were produced concurrent with the original F-22 Full-Scale Fatigue Test (FSFT) test and as a result have many more life shortfalls than later aircraft that makes them more difficult to modify to achieve a longer service life. It is the purpose of this technical activity to provide an overview of how the SLEP evaluation for the F-22 examined and planned requirements to support a SLEP extension for additional service life.

An F-22 SLEP would require a new life assessment on ~3600 control points with repairs due to margin shortfalls, corrosion, or MRB actions, in addition to running original durability and damage tolerance (D&DT) analysis past the current 8,000 flight hours. This technical activity will present the methodology to address these locations and rerun the original D&DT analysis to support the 300+ locations that would need to be repaired/modified as part of a SLEP program. Additionally, the technical activity will present the planning and requirement for over 10 years of testing for another FSFT, coupons/allowables test, and dynamics test that need to be completed to verify the new D&DT analysis and support the locations that require repair/modification. The technical activity will also present the scheduling challenges and timing of these tests and repairs/modifications that is required to prevent wide spread fatigue damage to achieve the required SLEP life. This technical activity will present how the F-22 was tackling those challenges in order to maintain a capable combat force into 2055 and beyond.

Table 9.11-3. Decrease of Fleet Size Over the Years

Proposed Fleet size	Year
750	
648	1990
438	1994
339	1997
277	2003
183	2006
187	2008

- **Production officially shutdown in 2009**

9.11.7. Fatigue Life Assessment of F/A-18 A-D Wing-Root Composite-Titanium Step-Lap Bonded Joint

Waruna Seneviratne, John Tomblin and Travis Cravens, Wichita State University – NIAR; Madan Kittur, USN-NAVAIR

The F/A-18 wing-root structure consists of AS4/3501-6 carbon/epoxy composite stepped-lap joint bonded with FM-300 film adhesive to a titanium splice fitting. This is one of the key examples of bonded primary structure certified and deployed on an air vehicle in the United States. Since it transitions from the composite wing skins to a titanium fitting for attachment to the fuselage, it is a complex joint in many ways. This effort is designed to evaluate the residual static strength and remaining life of this joint area after one lifetime of aircraft service and to evaluate the service life remaining based on the usage history (Figure 9.11-10). Spectrum loading representing fleet usage with load-enhancement factors is used for cyclic testing to determine the remaining life of the step-lap joints. This effort also supports the life-extension efforts to evaluate the remaining life of the structure. Furthermore, the tests are designed to address one of the biggest concerns with the aging aircraft fleet–the unknowns that emerge with little or no warning, raising the concern that an unexpected phenomenon may suddenly jeopardize an entire fleet's flight safety, mission readiness, and/or support costs. Over sixty 25-inch long tapered dog bone test specimens were extracted from eight decommissioned F/A-18 wing skins (A-D configuration) (Figure 9.11-11). The majority of the testing was conducted in room temperature ambient conditions, while a set of specimens were conditioned in salt-fog environment prior to fatigue. Overall, the end-of-life static test data are comparable or higher than the test data reported for pristine specimens. Fatigue tests were conducted using tension- and compression-dominant fatigue spectrums that contained 6,000 spectrum fatigue hours per lifetime (Figure 9.11-12). Fatigue loads were enhanced by load severity factors ranging from 1.15 to 1.60. All runout specimens were evaluated for residual strength. Both static and fatigue results indicated that the service history including the environmental exposure has not degraded the structural integrity of the bonded step-lap joint.

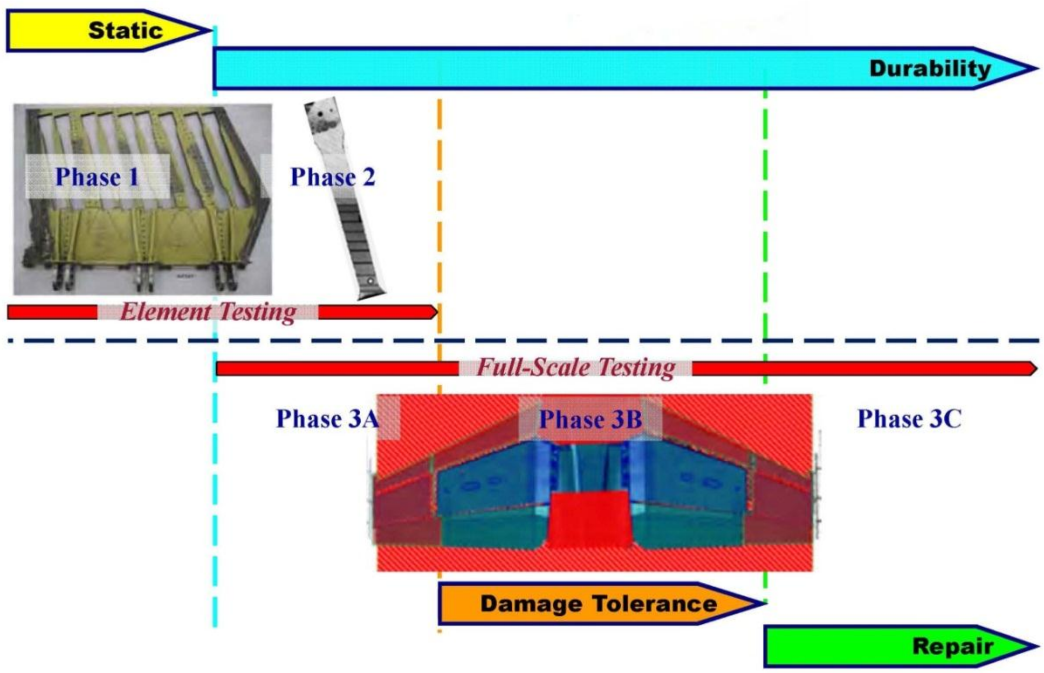


Figure 9.11-10. Test Program Overview

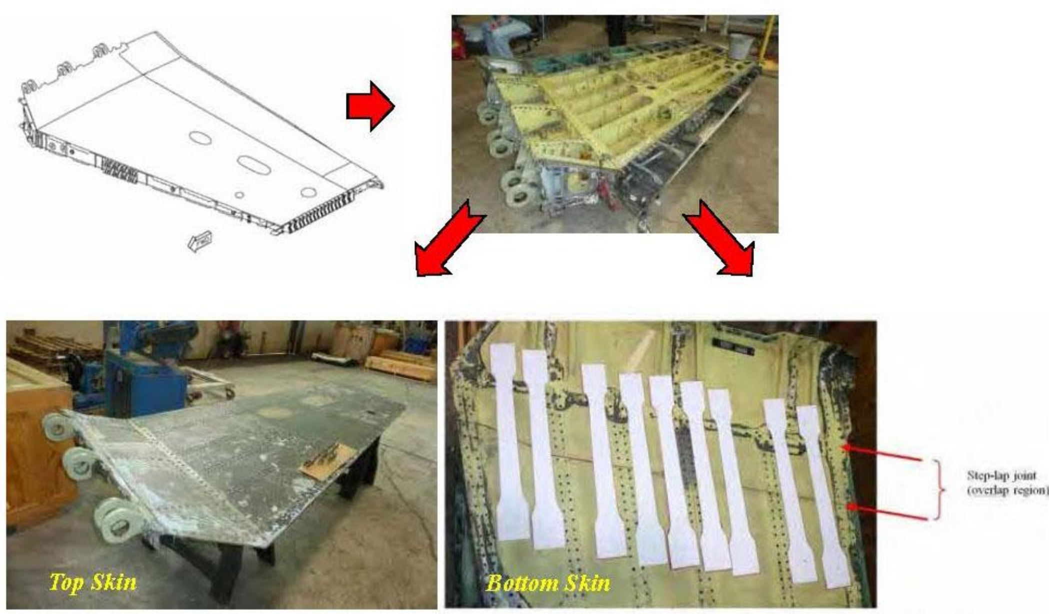


Figure 9.11-11. Wing Skin Removal and Specimen Extraction

Figure 9.11-12. Full-Scale Test Fixture

9.11.8. F-22 Corrosion Management Methodology

Phillip Young and Robert Bair, USAF F-22 Program Office

The F-22A Raptor (Figure 9.11-13) will be the backbone of the United State Air Force's (USAF) tactical fighter fleet through the middle of the 21st century. The fact that the F-22 is nearing the end of production at 187 aircraft underlines the importance of achieving full performance and life from every aircraft. A common detriment to the service life of the USAF fleet is the onset of corrosion, and in some cases, a very aggressive form of corrosion designated galvanic corrosion (Figure 9.11-14). Galvanic corrosion was first observed on aluminum panels on the F-22 in the spring of 2005 (Figure 9.11-15). The galvanic corrosion occurs as a result of the interaction between the materials utilized in low observable (LO) coatings and the aluminum structure of the F-22. By 2007, instances of galvanic corrosion on aluminum substructure were observed and presented a very real danger.

This technical activity presents the background and methodology behind the implementation of corrosion mitigations and the establishment of corrosion grounding dates to preserve aircraft safety and maximize aircraft availability. The technical activity recounts how the corrosion mitigations were established for fracture critical (FC) substructure affected by galvanic corrosion; mechanical isolation barriers were implemented as an interim solution while new LO materials were being developed to eliminate the galvanic dissimilarity. Critical remaining material thickness (RMT) calculations were found for the FC substructure. Corrosion growth rates were determined based on measurements of corrosion pit depths from aircraft substructure in depot (Figure 9.11-16). This technical activity details how the corrosion grounding dates were established using the RMT and corrosion rates. Special attention is then given to the sustainment of the fleet in regards to corrosion, to include how each individual aircraft is tracked via a corrosion age and how that shapes the scheduling of each F-22 into depot to complete these critical corrosion mitigations.

Figure 9.11-13. F-22 Raptor

Figure 9.11-14. Examples of Corrosion

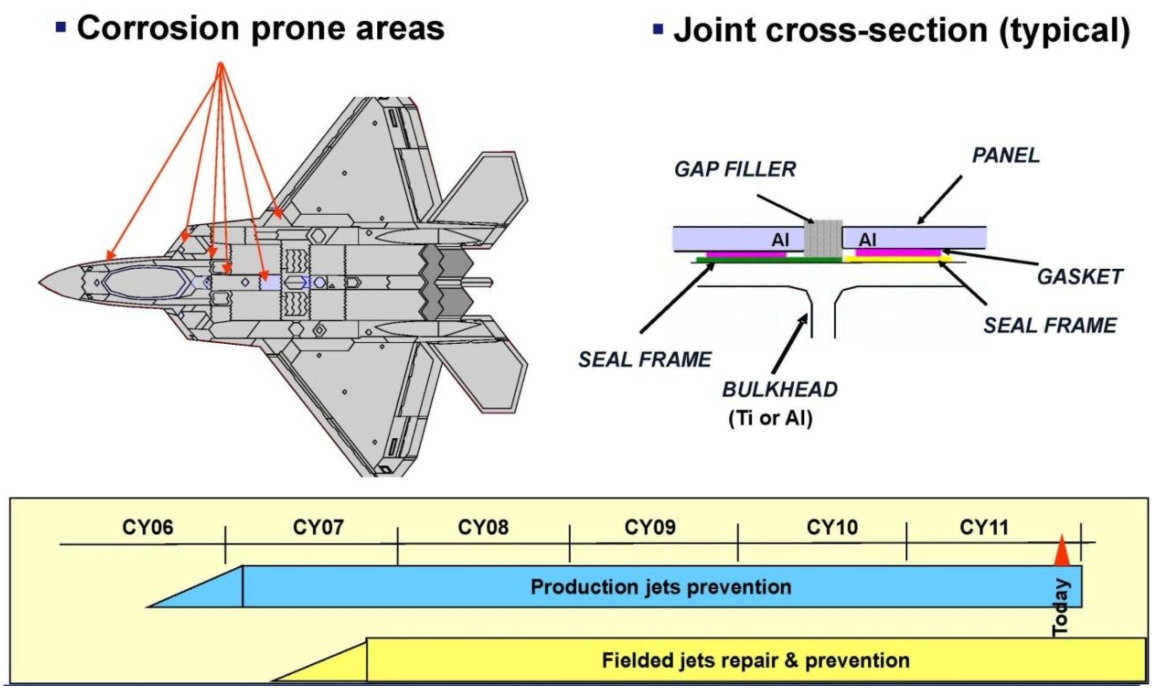


Figure 9.11-15. Locations of Corrosion

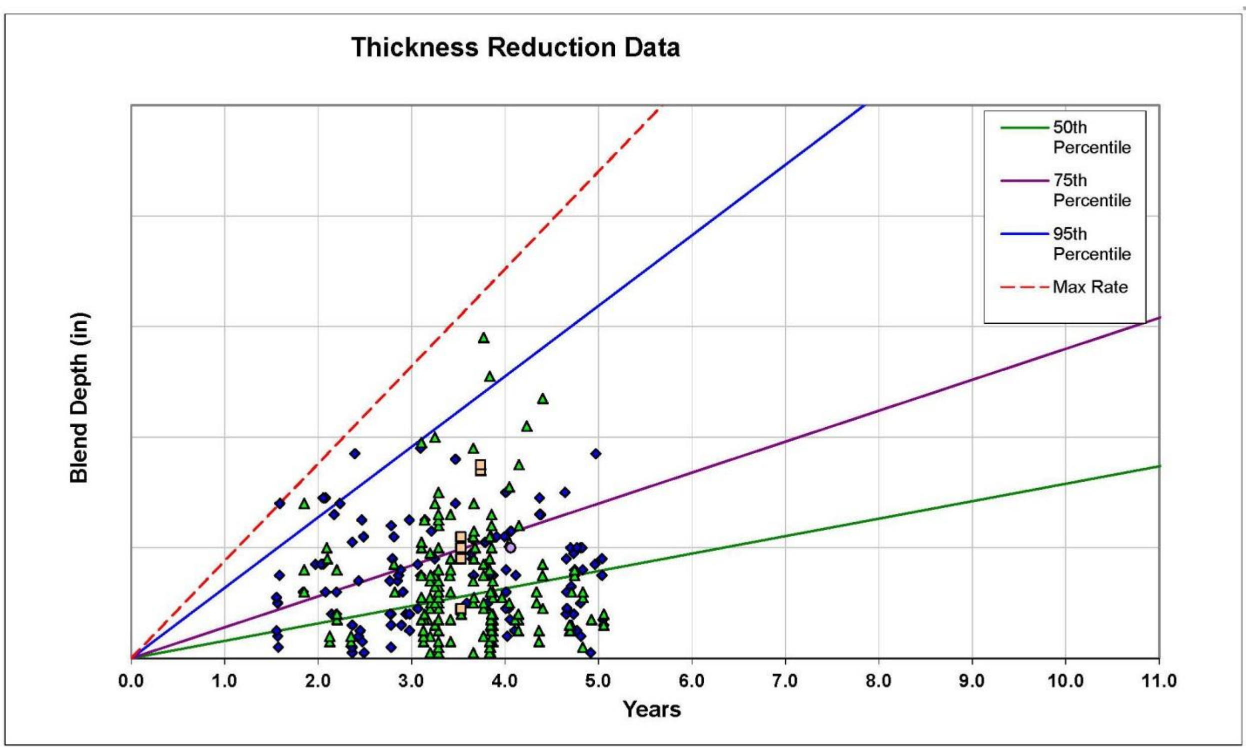


Figure 9.11-16. Structure Blend Depth Data From Depot

9.11.9. Stand-Up of the Initial C-22 ASIP

Brian Harper, Mercer Engineering Research Center (MERC); Kishan Goel, USN-NAVAIR; Scott Stringer, USAF-WR-ALC

The addition of the CV-22 to the USAF fleet presents unique engineering challenges to Air Force engineers, particularly in the area of Aircraft Structural Integrity Program (ASIP) support for the airframe. Naval Air Systems Command (NAVAIR) is the procuring authority for both the US Marine (MV-22) and the USAF (CV-22) variants of the V-22 Osprey. As a consequence, MIL-STD-1530 requirements were not explicitly included in the procurement specification. As a part of the transition to MIL-STD-1530, NAVAIR, USAF, and the Mercer Engineering Research Center (MERC) coordinated the stand up of the initial CV-22 Aircraft Structural Integrity Program (ASIP). This process involved the development and sustainment of ASIP metrics, development of a preliminary ASIP Master Plan, the addition of the CV-22 platform to the Aging Fleet Integrity & Reliability Management (AFIRM) fleet management tool, and the creation of an ASIP roadmap that identifies priorities and requirements toward compliance with MIL-STD-1530C.

The CV-22 ASIP program was evaluated using both the USAF ASIP Review Team development and sustainment templates. Several deficiencies were noted. Deficiencies in the development evaluation included durability and damage tolerance methodology, FEM correlation to full-scale fatigue test results, the lack of a plan to perform a Loads / Environment Spectra Survey (L/ESS) and omission of safety-of-flight structure from the CV-22 Individual Aircraft Tracking (IAT) program. Shortcomings from the sustainment evaluation included the lack of a Force Structural Maintenance Plan (FSMP) and a lack of substantiation of non-destructive inspection (NDI) intervals and methods. Recommended corrective actions included the following: ASIP Master Plan funding, DTA / Fail Safe review, analysis correlation to fatigue test results, fatigue test root cause investigation and corrective actions (Figure 9.11-17), FSMP development, inspection procedure review, use of VSLED data to perform L/ESS, expansion of IAT tracking points, comparison of fatigue-based methodology to DTA-based methods (Figure 9.11-18), crack history database development and BCA for design improvements vs. inspection / repair.

MERC drafted and delivered the initial CV-22 ASIP Master Plan. The plan details the state of the CV-22 ASIP program for the five elements of MIL-STD-1530C and provides a framework for tracking future development of the CV-22 ASIP program. MERC also extended the Aging Fleet Integrity & Reliability Management (AFIRM) tool for the USAF CV-22 fleet. This web-based fleet management tool provides information about the CV-22 fleet status, including possessed and assigned bases, NMC / PMC status, and sortie data.

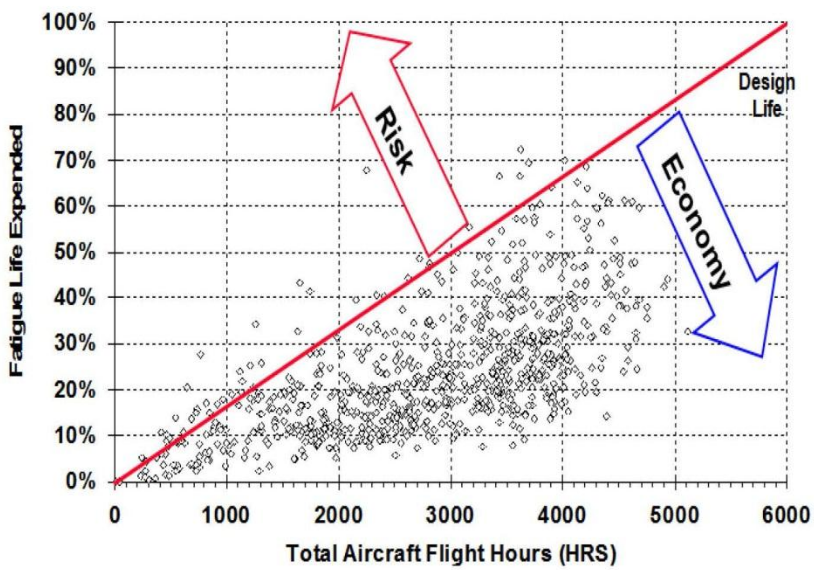
This work to leverage existing OEM and NAVAIR data and map it to the framework of the five elements of the ASIP requirements per MIL-STD-1530C has formed the basis for the CV-22 ASIP. Funding is imperative to remedy the identified deficiencies and continue the initiatives started during this initial task in order to ensure compliance with the USAF ASIP program and fleet health objectives over the operational lifetime of the CV-22.

- Teardown of fatigue test components
- Fractographic analysis
- Correlation to structural models
- Identification of additional components to add to SAFE / IAT program

Teardown / Inspection photo via National Institute for Aviation Research, Wichita State University

Figure 9.11-17. Fatigue Test Root Cause Investigation and Corrective Actions

- NAVIAR SAFE tracking is fatigue-based method
- USAF standards dictate equivalent flight hours based on DTA-based methods
- Goal of both methodologies is the same → track structural effects of actual aircraft usage
- Investigation of differences between two approaches may identify improvements to service lives



Source – NAVAIR SAFE 101: Introduction to USN Airframe Structural Life Management

Figure 9.11-18. Comparison of Fatigue-Based Methodology to DTA-Based Methods

9.11.10. Examination of Durability and Damage Tolerance Design Criteria

Dale Ball, Lockheed Martin Corporation

Variants of the F-35 Joint Strike Fighter have been designed according to both United States Air Force (USAF) and United States Navy (USN) requirements. For the F-35A Conventional Take-Off and Landing (CTOL) Aircraft, structural integrity design guidance is provided by MIL-STD-1530C. For the F-35B Short Take-Off and Vertical Landing (STOVL) and F-35C Carrier Variant (CV), this guidance is provided by NADC 87089 60. Specific design requirements are derived (tailored) from JSSG-2006, the Joint Services Specification Guide for Aircraft Structures. With the program nearing the completion of SDD, and with three full-scale durability tests well underway, have a unique opportunity exists to quantitatively compare the durability and damage tolerance (DaDT) design criteria imposed by the two services and to assess the effectivity of each in guarding against variations in initial material quality, design accuracy, and initial manufacturing quality.

In this technical activity a quantitative comparison of USN fatigue-crack-initiation-based and USAF fatigue-crack-growth-based DaDT design criteria is conducted, primarily through the use of remaining life diagrams (Figures 9.11-19 through 9.11-21). These diagrams are prepared for various geometry, material and usage scenarios and then used to assess the potential for premature structural failure due to the presence of unanticipated stresses or undetected flaws. The stresses considered are of the type that can arise when loads are miscalculated or stress concentrations are missed during design. The flaw sizes considered are based on typical initial quality data (corrosion pits, surface scratches, etc.). The technical activity concludes with a survey of F-35 structure to which both sets of criteria have been applied. The results indicate which regions of the airframe are sized by which criteria.

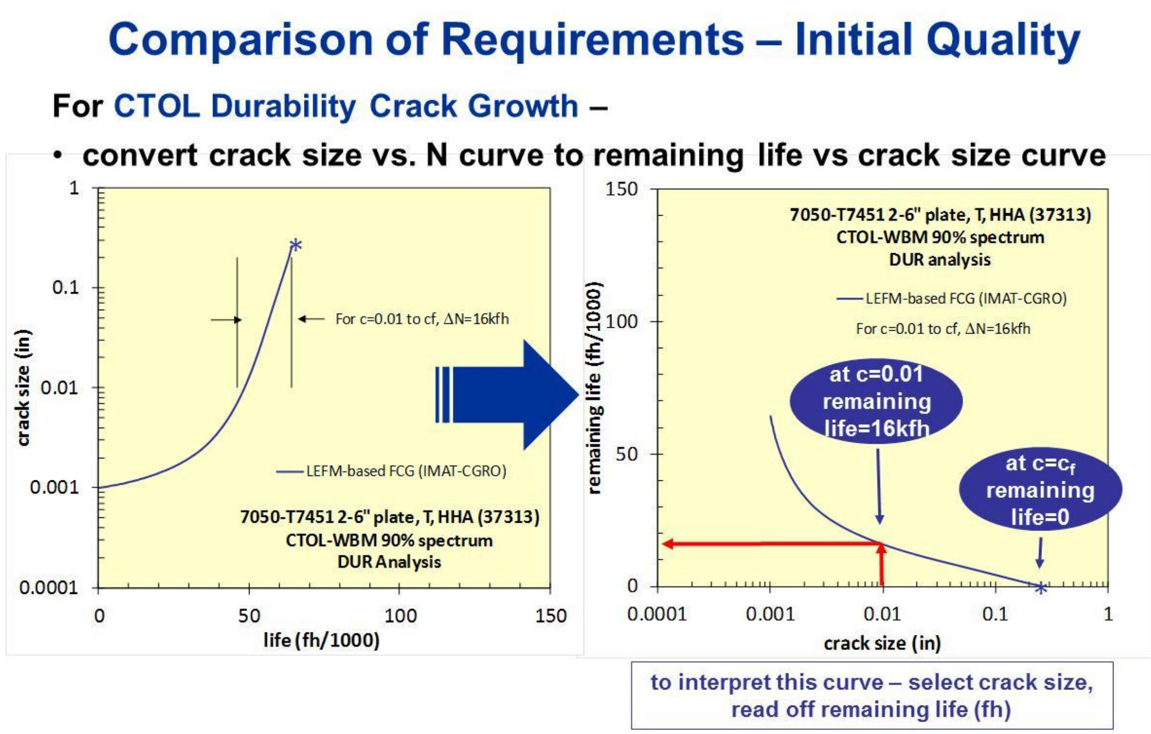


Figure 9.11-19. CTOL Durability Analysis

Comparison of Requirements – Initial Quality

For CTOL Damage Tolerance Crack Growth –

- **convert crack size vs. N curve to remaining life vs crack size curve**

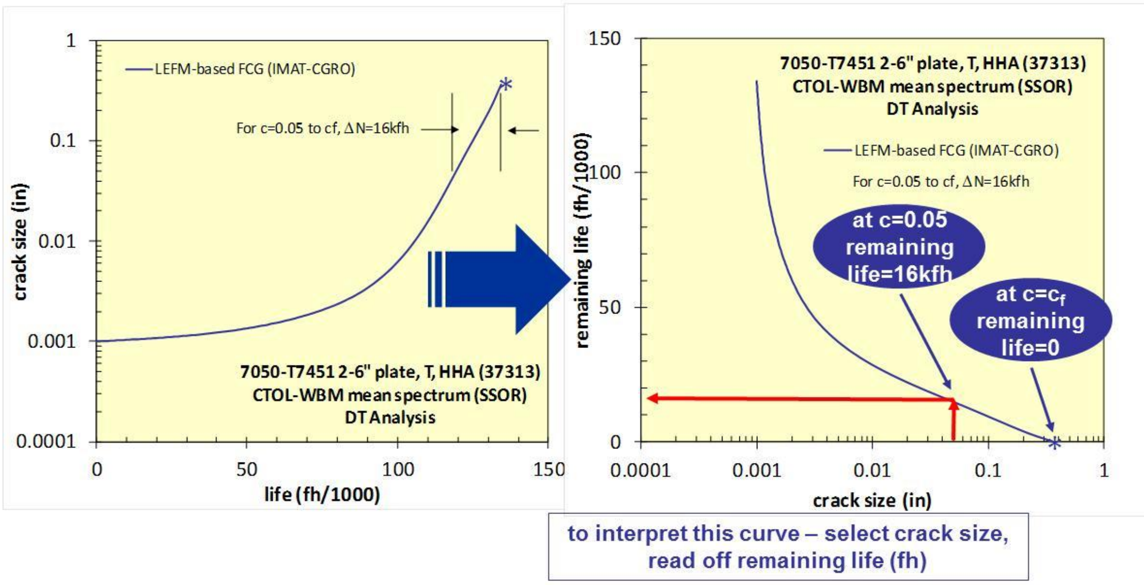


Figure 9.11-20. CTOL Damage Tolerance Analysis

Comparison of Requirements – Initial Quality

Comparison of DUR and DT analyses at same operating stress:

- **For CTOL WBM, $RUL_{DUR} < RUL_{DT}$ at all crack sizes**
- **Therefore – WBM loaded structure sized by DUR is better able to sustain anomalous starting crack size than that sized by DT (this is due to more severe spectrum)**

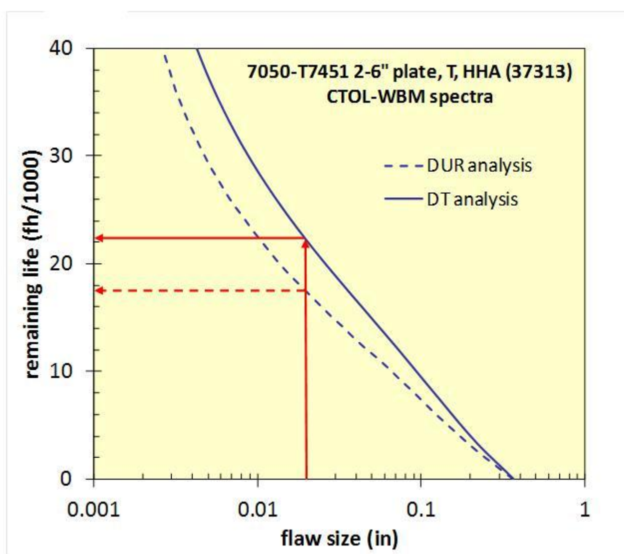


Figure 9.11-21. Comparison of CTOL Durability and Damage Tolerance Analyses

9.11.11. F-22 Roadshow: Benefits on Overall ASIP Execution

George Crosthwaite and John McClure, F-22 SPO

In 2008, the F-22 Systems Program Office (SPO) started visiting bases to conduct Aircraft Structural Integrity Program (ASIP) Road Shows because there was major incongruity with communication between the SPO and the field regarding structural issues and usage of the aircraft (Figure 9.11-22). The goal of these visits is to assist the field in understanding ASIP related processes and issues, communicating aircraft usage severity, and keeping them informed of future ASIP problems that could arise. This technical activity discusses the ASIP Road Show format and specific examples of ASIP Road Show impacts.

Figure 9.11-22. F-22 Aircraft

In order to communicate with the F-22 bases, the F-22 SPO has an annual face-to-face meeting with the Ops and Maintenance commanders at a given base to discuss the F-22 ASIP. Other members of those groups such as pilots and maintainers are often brought in to make sure the information is shared across all levels of the chain. There are some organizational challenges as seven AFBs operate the F-22 and they are shared across five different commands. Topics typically discussed include fleet usage compared to base usage (Figure 9.11-23), the purpose of ASIP, current ASIP projects, and F-22 force projection. The F-22 SPO also meets with field personnel, such as NDI and LO shops to pass along information about ongoing issues and to see if they have any issues of which the SPO was unaware.

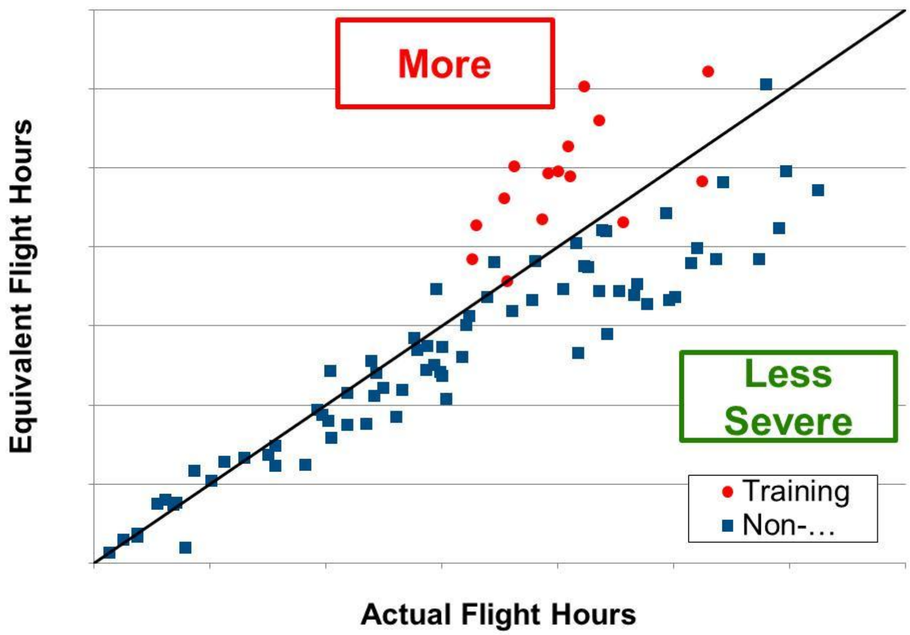


Figure 9.11-23. Usage

The ASIP Road Shows have also helped the SPO understand and improve various ASIP issues. An example of this is F-22 Individual Aircraft Tracking (IAT) data capture. The F-22 SPO base visits have allowed it to communicate the problem of low data capture rates with the field and understand various process issues that led to low data capture rates. When the F-22 SPO started the road shows, data capture rates were as low as 65% for production aircraft and have now been improved to over 90% (Figure 9.11-24). Base visits have also helped the F-22 SPO communicate upcoming issues with the field, such as NDI changes or corrosion findings/mitigations (Figure 9.11-25). The F-22 SPO has also been able to instruct the field on how to obtain various coating samples (Figure 9.11-26), which can be correlated to current environmental testing being performed. The list of impacts to the F-22 ASIP grows as each Road Show is conducted.

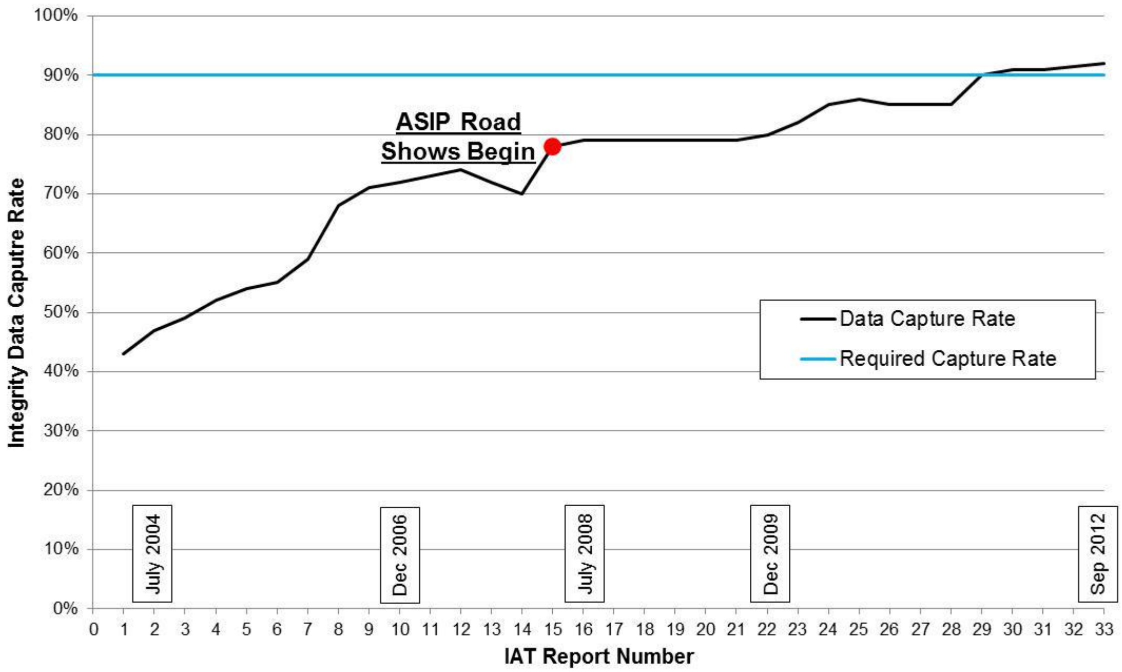


Figure 9.11-24. Integrity Data Collection Trend

- Ongoing problem with F-22
  - **Corrosion on the cockpit floor being the most recent find**
  - **Make sure they are pro-actively looking for and preventing corrosion**

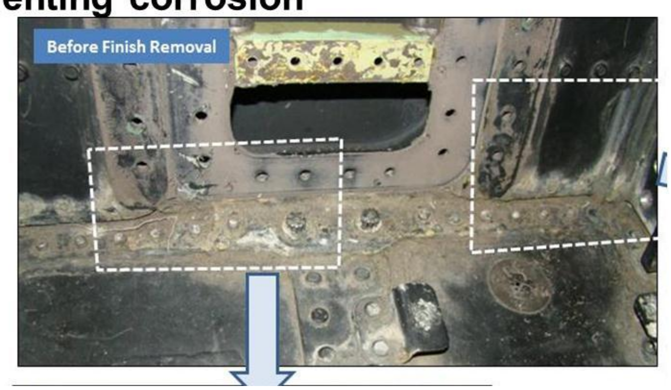


Figure 9.11-25. Corrosion

Figure 9.11-26. Coating Issues

9.11.12. Qualification of Advanced Aircraft Structural Sustainment Tools/Processes within the A-10 Aircraft Structural Integrity Program (ASIP) Environment – Perfect Point E-Drill

Scott Carlson, Southwest Research Institute and Mark Thomsen, USAF-00-ALC

As the United States Air Force (USAF) continues to extend the service life of its aircraft beyond their original design life, the need has arisen for new/advanced tools and processes to perform essential sustainment tasks with a reduced probability of damaging critical aircraft structure. Many of these tools and processes were neither available nor envisioned during original production. Therefore, drawing and technical data requirements do not exist allowing or prohibiting use of these tools and/or processes. As a result, evaluation and qualification, where applicable, of new processes is required to ensure that unintended detrimental conditions are not imparted in the aircraft structure.

One of these essential sustainment tasks is the removal of permanent fasteners. The A-10 Structures Branch at Hill AFB, UT contracted Southwest Research Institute (SwRI) to investigate a tool for use in the removal of all fasteners common to the lower forward skin of the A-10 Wing Center Panel (WCP). In response to the USAF request SwRI proposed the use of the Perfect Point E-Drill. The E-Drill uses Electrical Discharge Machining (EDM) to cut off the head or tail of the fastener, thus allowing the remainder of the fastener to be more easily removed (Figure 9.11-27). This process reduces the time required to remove an interference-fit fastener and has the potential to reduce and/or eliminate mechanical damage at the fastener hole that is commonly induced using conventional drilling procedures.

- Fastener Removal Tool
  - Using EDM to Cut the Fastener Head

Perfect Point's™ E-Drill™1

Removal of Fastener Head Using E-Drill™1

Figure 9.11-27. Perfect Point E-Drill

Recent experience with other high-energy shop tools suggested that a thorough qualification processes was warranted. The A-10 ASIP team leveraged this recent experience to develop a lean, optimized qualification test matrix whereby the effects of the tool might be quantified over the range of applicable tool settings, materials, forms, tempers, coatings, and thicknesses. At issue were the possible impacts to material strength, durability (Figure 9.11-28), and inspectability.

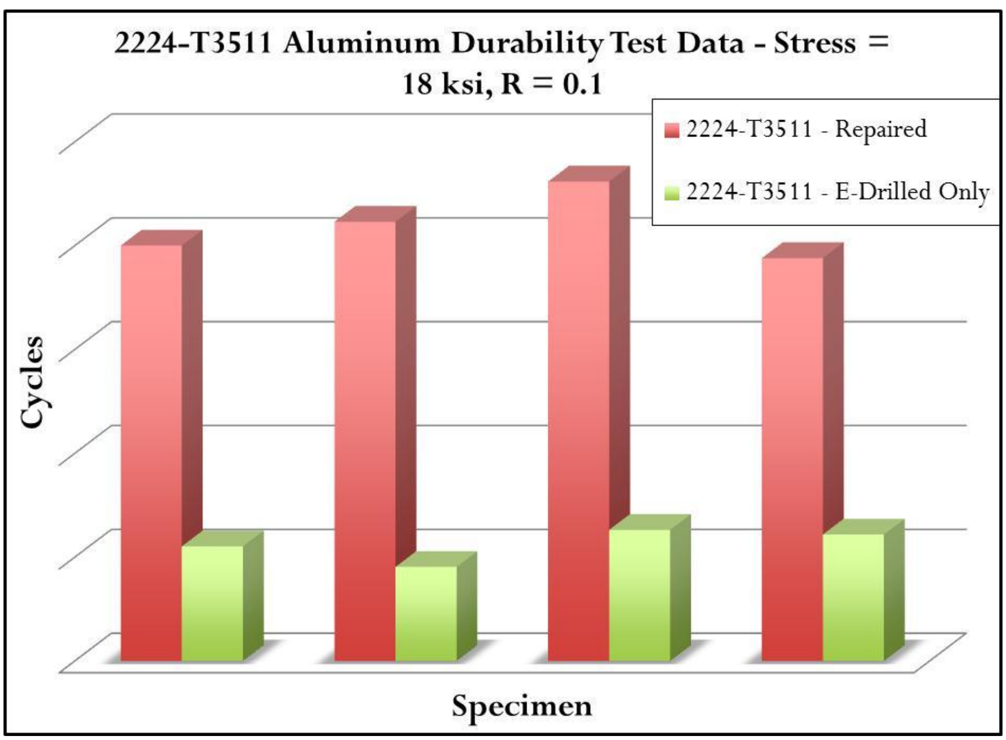


Figure 9.11-28. Durability Test Results

This technical activity outlines the A-10 ASIP/SwRI qualification methodology, plan, execution and experimental results for the E-Drill. The intent of this "case study" is to share lessons learned with aerospace structures and integrity communities. Key take aways include evaluation team qualifications, experimental and testing methodologies, metallurgical investigation, NDI and evaluation processes and the paramount role of concise communication within the team and with leadership/management.

9.11.13. Corrosion Impacts to the F-16 ASIP

Bryce Harris and Kimberli Jones, USAF F-16 SPO

Corrosion has had an impact to F-16 (Figure 9.11-29) structural integrity, particularly in recent years. The damage is not characterized as widespread, but rather is found to be acute degradation in specific components often as a result of water entrapment due to design or maintenance practices (Figures 9.11-30 through 9.11-32). Sealant application, drain valve placement and use, and geometric contours all affect the amount of moisture present, which is necessary for corrosion to occur. Mechanical system design and integrity has also been found to be a factor in structural corrosion degradation; the breakdown of the Environmental Control System duct insulation contributed to the condensation collection and some of the corrosion found in the cockpit (Figure 9.11-33). This corrosion impact to structural integrity has generally not been an alarming safety issue, but rather has impacted cost, availability, sustainability, and capability.

While fatigue cracking in the absence of corrosion effects is well understood, the interaction between corrosion and fatigue cracking is not always as well understood or quantified. Powerful engineering tools such as full-scale finite element models enable engineers to anticipate fatigue crack locations and associated component fatigue lives. Predictive tools of this nature as related to corrosion and interactions with fatigue are not generally employed within the realm of the Aircraft Structural Integrity Program (ASIP). Similarly, the existence of and reliability of such tools is not well known, or at least not widely utilized. Successful structural maintenance within the United States Air Force ASIP guidelines has relied on non-criticality and fail-safe capability of corroded components as well as conservative assumptions as to the crack-like behavior of corrosion pitting. If a critical scenario was to arise with significant corrosion damage in single load path structure at a critical detail, the current corrosion mitigation approach may not be considered adequate from an aircraft availability standpoint. This could warrant adoption of additional guidelines or engineering analysis procedures to characterize the corrosion/fatigue interaction. This technical activity will focus on the corrosion aspects of structural integrity, related fleet impacts, and ASIP corrosion mitigation methodology.

Figure 9.11-29. F-16 Aircraft

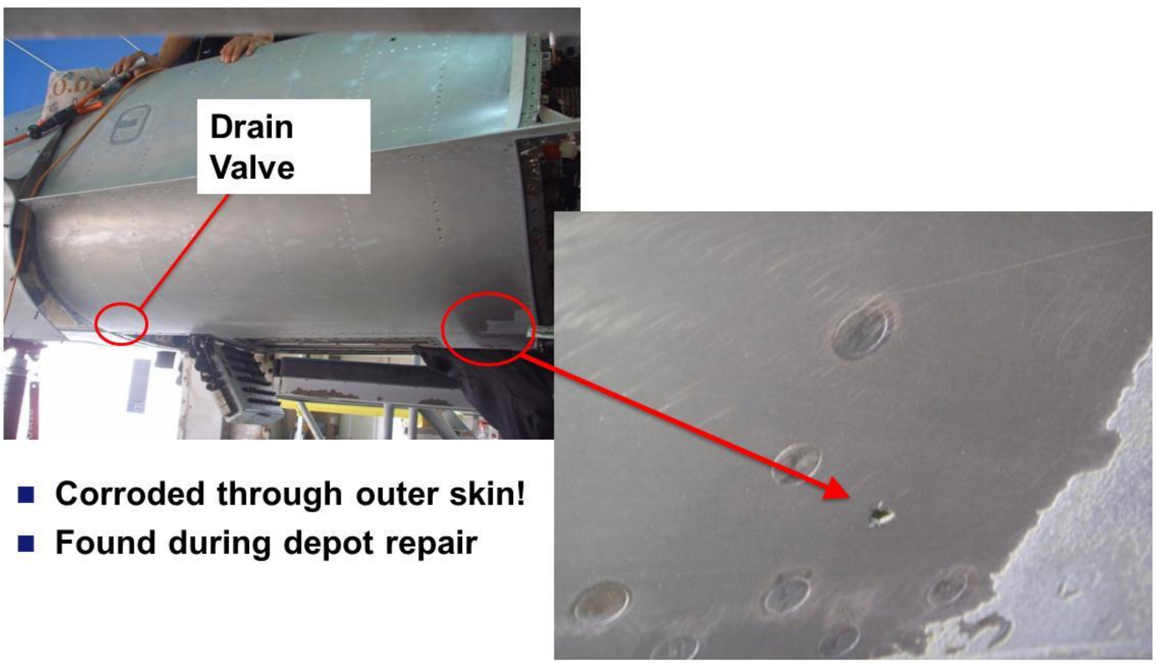


Figure 9.11-30. Forward Cockpit Corrosion Example

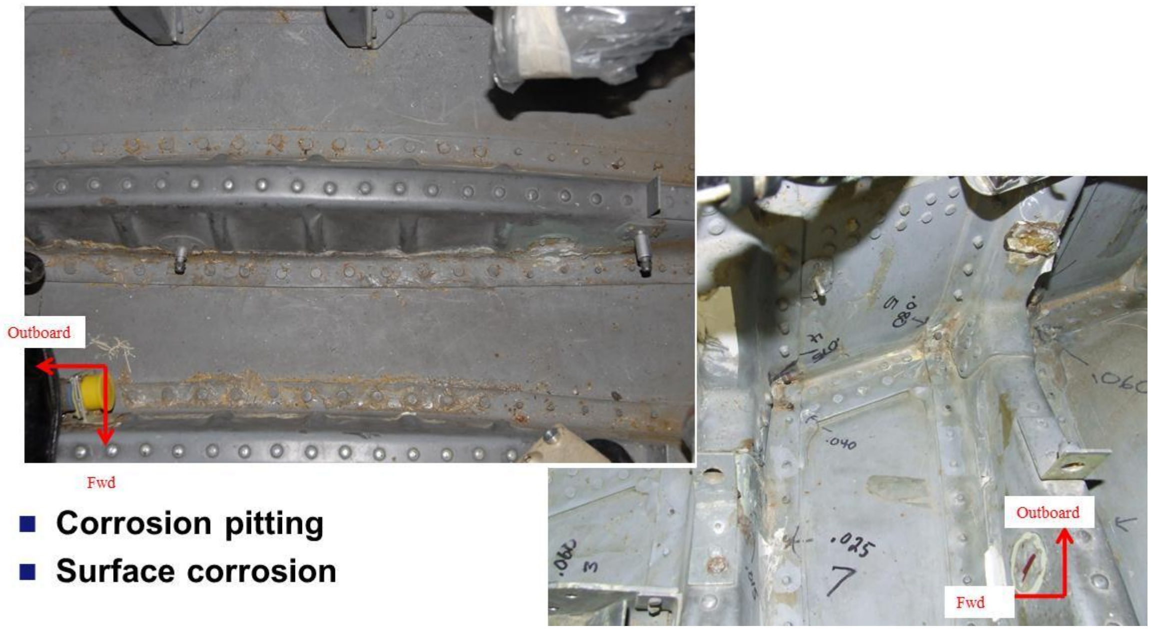


Figure 9.11-31. Aft Cockpit Corrosion Typical Case

Figure 9.11-32. Wing Aft Spar Corrosion

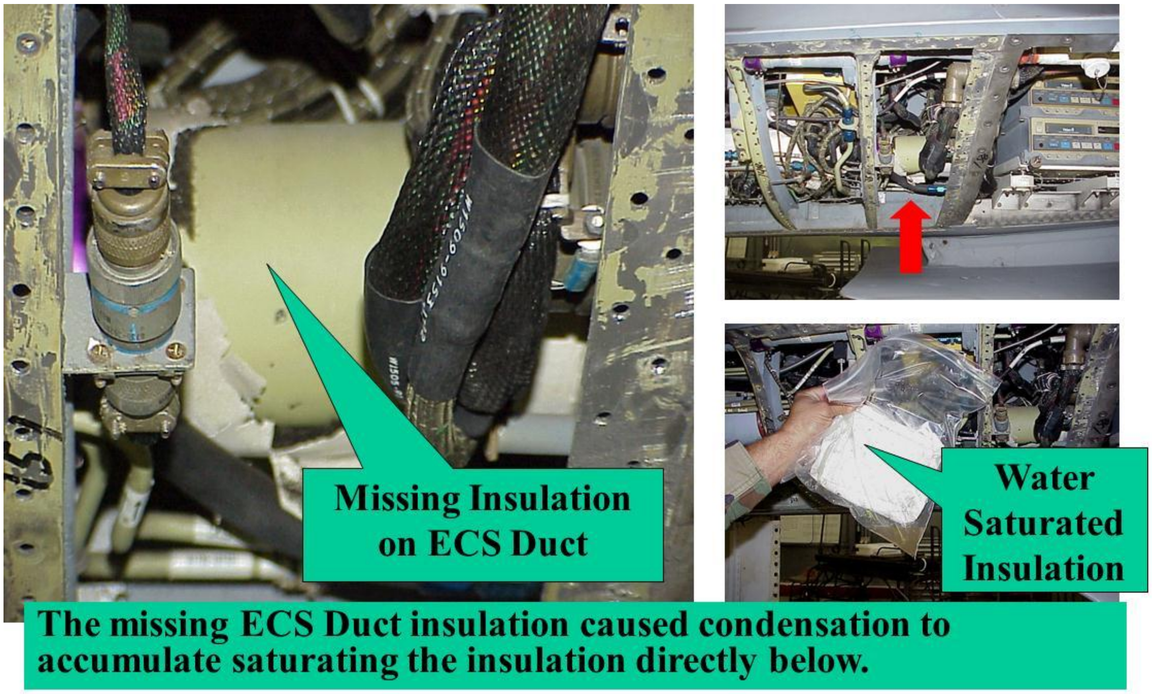


Figure 9.11-33. Cockpit Corrosion an Origin of Moisture

9.11.14. T-38 ASIP: Going the Extra Inch

David Wieland and Clint Thwing, Southwest Research Institute; Michael Blinn and Chad King, USAF-OO-ALC

The T-38 aircraft (Figure 9.11-34) was introduced into USAF service in the early 1960s. Based on the design paradigms of the times, the T-38 "Talon" was primarily developed to meet static strength requirements. During the early years, the Talon did have some fatigue tests and analyses performed, but these efforts were limited to components. The first damage tolerance analysis (DTA) of the entire T-38 aircraft took place in the late 1970s. A product of its time, the T-38's first DTA was based on relatively conservative assumptions and did not take advantage of continuing damage or load redistribution. Further, the T-38's DTA also relied heavily on available handbook solutions for geometry correction factors. Finally, the early analyses also made use of NDI detectable flaw sizes developed by an AFRL steering group led by Dr. Jack Lincoln.

Figure 9.11-34. T-38 Aircraft

During the latest DTA Update, the first nondestructive inspection (NDI) structures bulletin (SB) was released. Based on this new information, it was apparent that previous NDI detectable flaw size assumptions were much smaller than those allowed in the NDI SB. Further, the simplifying assumption of neglecting continuing damage and load redistribution for T-38 DTAs was not acceptable. Compounding the challenge, many newly identified fatigue critical locations (FCLs) could not be analyzed using available handbook solutions (Figures 9.11-35 and 9.11-36).

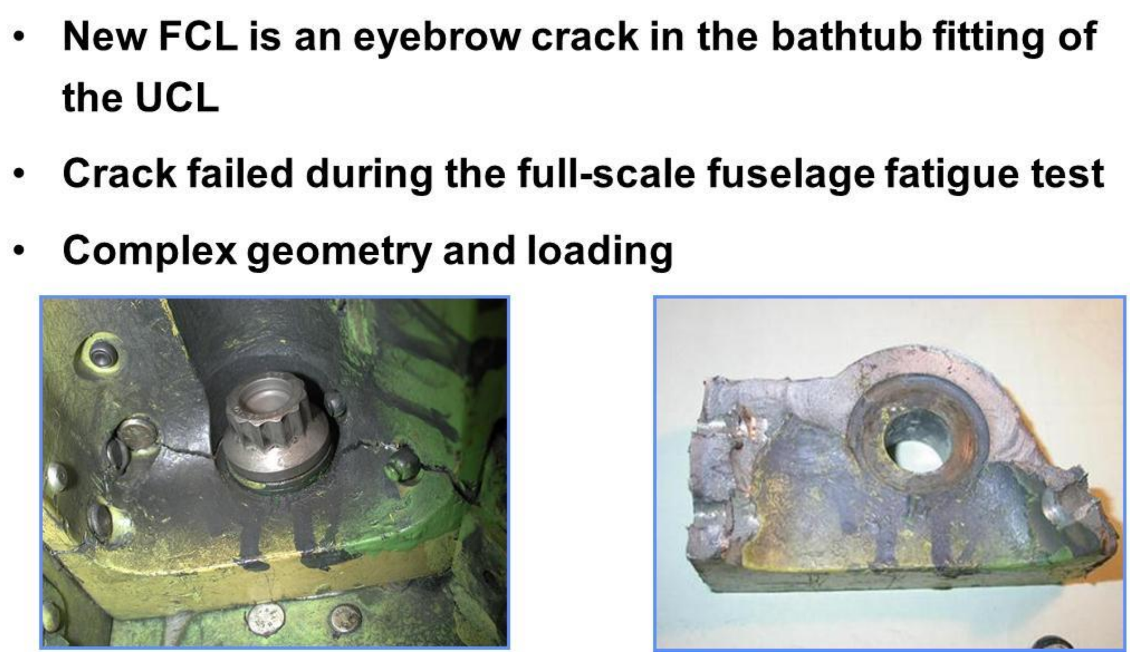


Figure 9.11-35. New FCL: Upper Cockpit Longeron (UCL)

- Modeled in StressCheck
- Loads from forward fuselage model
- Crack grown elliptically
- Stress intensities determined from StressCheck used as user defined betas in AFGROW
- Predicted durability life agreed well with test life

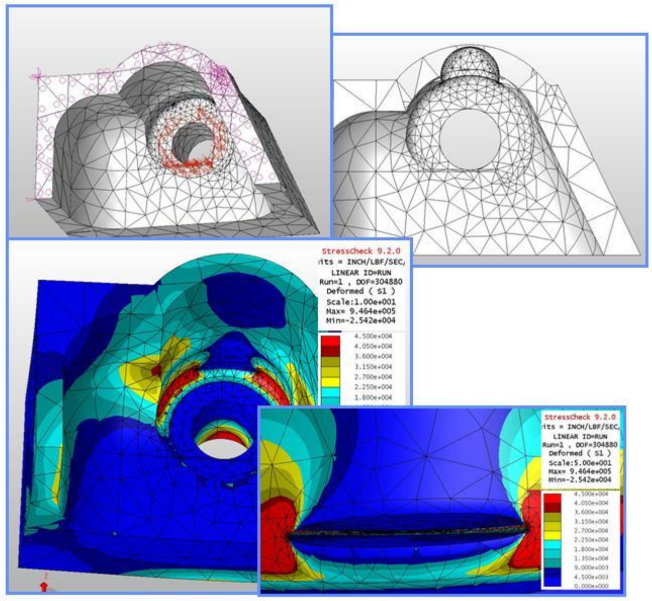


Figure 9.11-36. Upper Cockpit Longeron Modeling

This technical activity discusses how the T-38 ASIP team successfully went the extra mile (or crack length inches in this case) to improve the T-38 DTA and implement the resulting inspections. The continuing damage method used and how load redistribution was utilized is detailed (Figure 9.11-37). Development of geometry correction factors for unique geometry using tools such as StressCheck is also discussed. Finally, the new NDI procedures and tools developed that allow for the DTA inspections to be implemented are also highlighted.

- Critical flaw size increased
- Inspection interval increased by 50%
  - Allowed alignment of ASIP inspections to major Mx phase

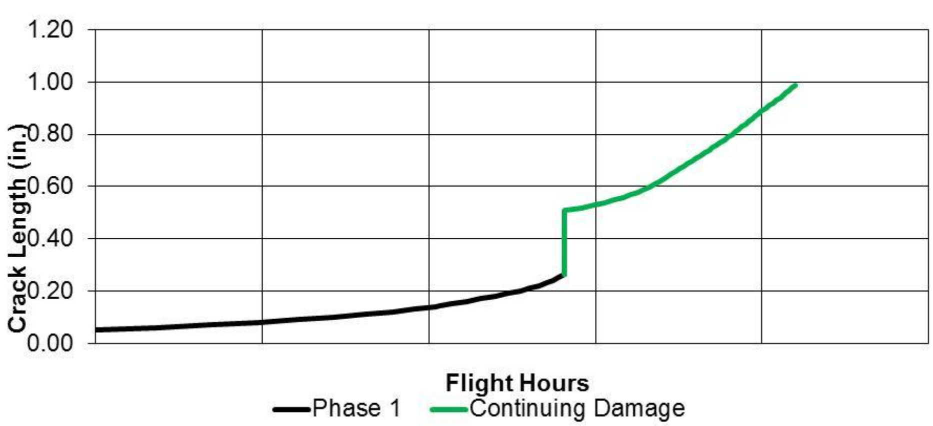


Figure 9.11-37. Continuing Damage Effects

9.11.15. C-5 Aft Fuselage Underfin Frame Structure Cracking Investigation

Thomas Burke, Lockheed Martin Corporation

During routine maintenance in June 2009, a crack was discovered on a C-5B at Travis AFB in the FS 2538 underfin frame in the Batman Fitting (Figures 9.11-38 and 9.11-39). A series of these fittings attach the vertical stabilizer to the aft fuselage. There are 11 on each aircraft, spanning the front to rear spars of the stabilizer. A TCTO was released to inspect the cracking area on the FS 2538 frame and similar areas on the remaining Batman Frames. A number of additional fitting cracks were found, all at FS 2538. The more significant cracks were repaired by a concept called the "Bat Phone" because of its appearance (Figure 9.11-40). The Bat Phone repairs are considered temporary. This repair concept is only viable for cracks found in the forward flange of the fitting where all cracks have been found to date. The fore-aft spacing between the FS 2538 and FS 2557 frames limits accessibility to the aft side of the frame, precluding the use of the phone repair concept. Analyses using both the airframe level FEM and a highly detailed, fastener level model predicted very short crack growth lives just as is seen in the fleet. The premature fleet cracking and the accompanying short life predictions in combination with the limited effectiveness of the Bat Phone repairs suggest the inspection/repair approach should be temporary. Therefore, the final corrective action for the FS 2538 Batman Fitting cracking problem was to design, fabricate, and install a more robust Batman Fitting.

The analysis also identified other hot spots where fatigue life predictions fall short at life limited locations. These dorsal island components (Figures 9.11-41 and 9.11-42) are highly critical, in some cases providing the single load path between the empennage and fuselage. This led to further investigation. First, the original fatigue test article records were reviewed for problems in the dorsal area. The test articles were re-inspected to verify that additional cracks hadn't been missed in the original inspections. Additional fatigue test article inspections were accomplished in the event that cracks were missed during inspections performed during the original testing. The review revealed cracking in the vicinity of the FS 2538 Batman Frame and dorsal island structure but not at the expected locations. Second, to determine the extent of the cracking problem, a "mini teardown" on several retired C-5 aircraft was performed and limited cracking was found. Due to the difference between this and the analysis, the fatigue test was revisited. The cracks found at the expected locations were small and not catastrophic in nature even after application of four lifetimes of testing. Due to this result, the fatigue load spectrum was resurrected and simulated on the detailed FEM. Sufficient correlation was observed. The critical dorsal island locations were then re-analyzed for the test spectrum. For all locations, the analysis predicted that these structures should have experienced complete failures early during the test. Therefore, an approach to develop a correlation factor was formulated to adjust inspection requirements based on this analysis. Applying these test derived correlation factors to the baseline analysis for each location of interest gives adjusted intervals; however, inspections are still required. Fleet inspections for the structure will commence after specific NDI procedures are developed. A repair kit has been developed to significantly improve damage tolerance characteristics on uncracked aircraft and still provide benefit on a cracked airplane.

- ***Eleven "Batman" fitting frames***
  - Machined aluminum die forgings
    - *C-5A – 7075-T6*
    - *C-5B – 7049-T73*
  - Batman fitting frames transfer all horizontal and vertical stabilizer loads to fuselage
  - FS 2538 is most forward frame, i.e., front spar joint, very critical structure.

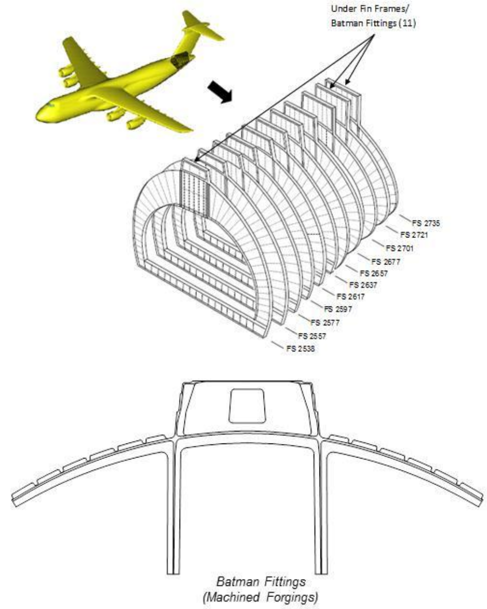
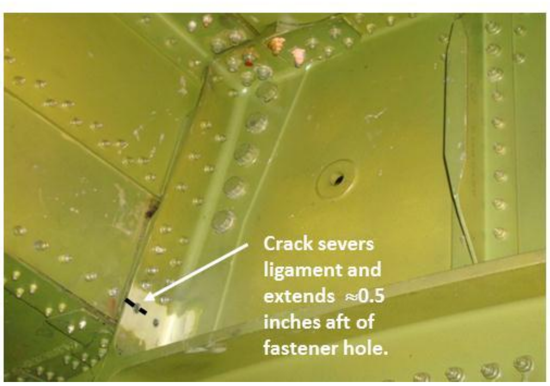


Figure 9.11-38. Batman Fitting Frames

- ***Numerous FS 2538 Batman Fittings found cracked in fleet***
  - Initial finding – found during routine maintenance.
  - Total of 15 Airplanes found cracked to date at critical location.
  - Cracking of various severity (short, long, horizontal, vertical).
  - Cracking is fatigue, not SCC (based on met lab report).



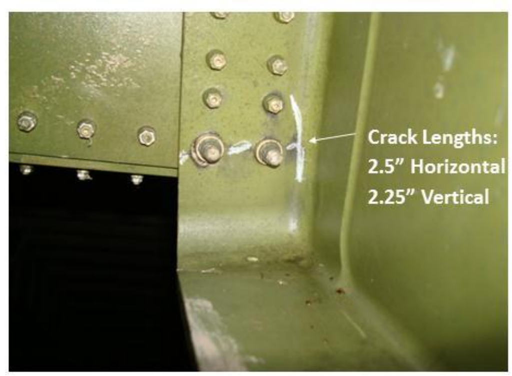


Figure 9.11-39. Batman Fitting Cracking

- Repairs, referred to as "Bat Phones" have been designed
  - *Mechanically fastened steel fitting (4130)*
  - *Bat Phone repairs considered temporary until batman fitting can be replaced.*
  - *Repair concept only viable for the cruciform section forward flange.*
  - *If rear flange cracked, ground, retire or severely restrict.*
    - » No rear flange cracks found in service aircraft to date.
- Cracking solution
  - *Batman Fitting replacement*
  - *Technical drawing package released*

Figure 9.11-40. Batman Fitting Repairs

- ***N9340 Detail FEM – Exploded View***

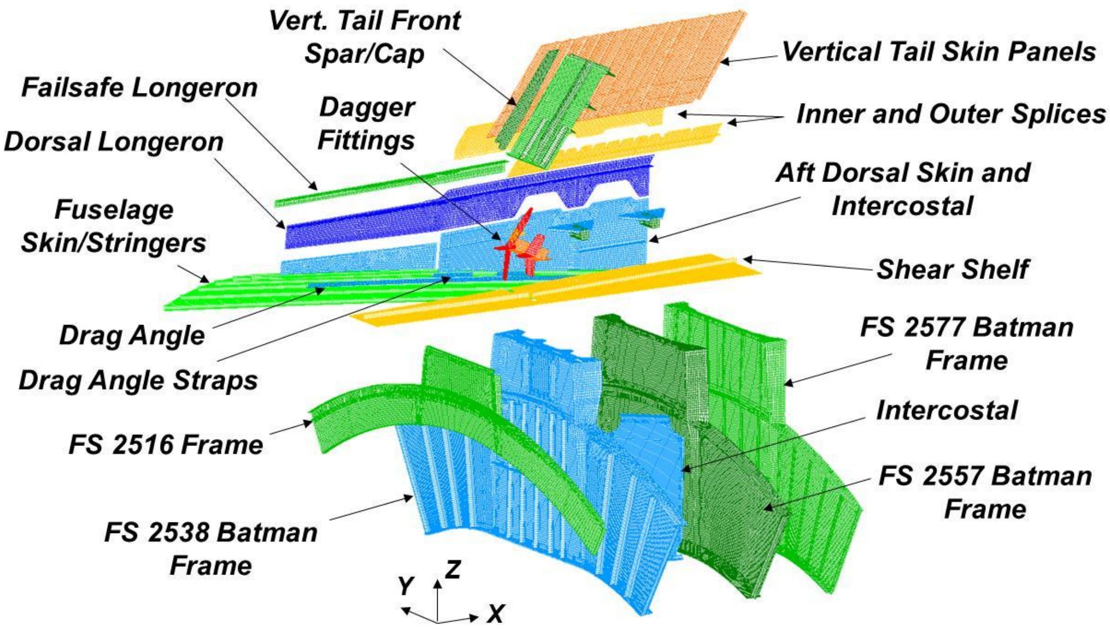


Figure 9.11-41. Batman Frame and Dorsal Island

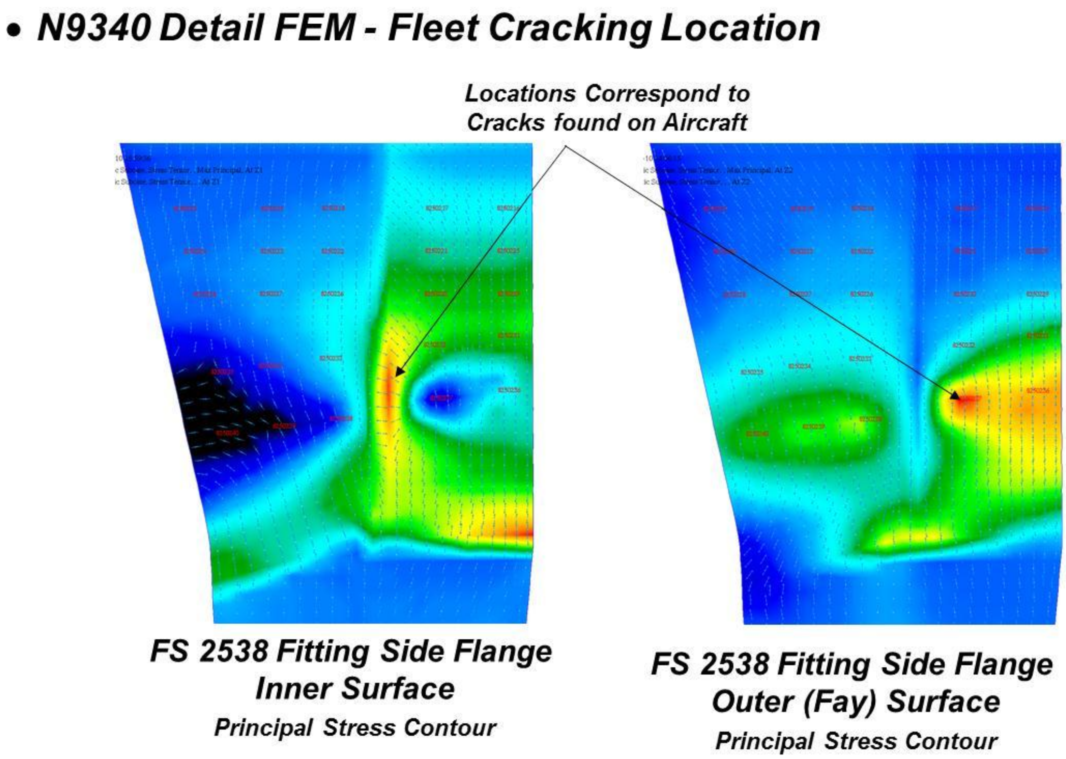


Figure 9.11-42. Batman Frame and Dorsal Island Cracking Location

9.11.16. Overview of the Full-Scale-Durability Tests on the F-35 Lightning II Program

Marguerite Christian, Lockheed Martin Corporation

The Aircraft Structural Integrity Program for the F-35 Lightning II (Figure 9.11-43) is unique in that it includes dedicated full-scale-static and durability test articles for each of the three variants included in the program: Conventional Take Off and Landing (CTOL), Short Take Off and Vertical Landing (STOVL), and Carrier Variant (CV) (Figure 9.11-44). These tests are a key component of the structural certification process and provide the data required to validate the structural analyses and to demonstrate the strength and stability of the airframe (Figure 9.11-45). The static and durability tests enable efficiencies through test consolidation and also through economies of scale. Investments made in the test fixtures and data acquisition systems coupled with efficient test protocols enable testing to progress rapidly and efficiently.

Durability testing of the airframe is conducted in dedicated fixtures at Lockheed Martin in Fort Worth, Texas and at BAE Systems in Brough, England (Figure 9.11-46). The Horizontal Tail (HT) and Vertical Tail (VT) tests are performed off aircraft; CTOL and STOVL have completed the required two lifetimes of testing (16000 test hours). The CV HT and VT tests, as well as the three full airframe tests are underway.

This technical activity provides an overview of the test methodologies, challenges faced and the results to date for F-35 durability test articles (Figure 9.11-47). The F-35 Static and Durability Test Programs, developed to satisfy the requirements of MIL-STD-1530C, continue to demonstrate the structural integrity of the F-35 airframe design and provide a model for the remaining variants as well as future aircraft programs.

Figure 9.11-43. F-35 Lightning II Aircraft

Figure 9.11-44. Tri-Variant Joint Strike Fighter (JSF)

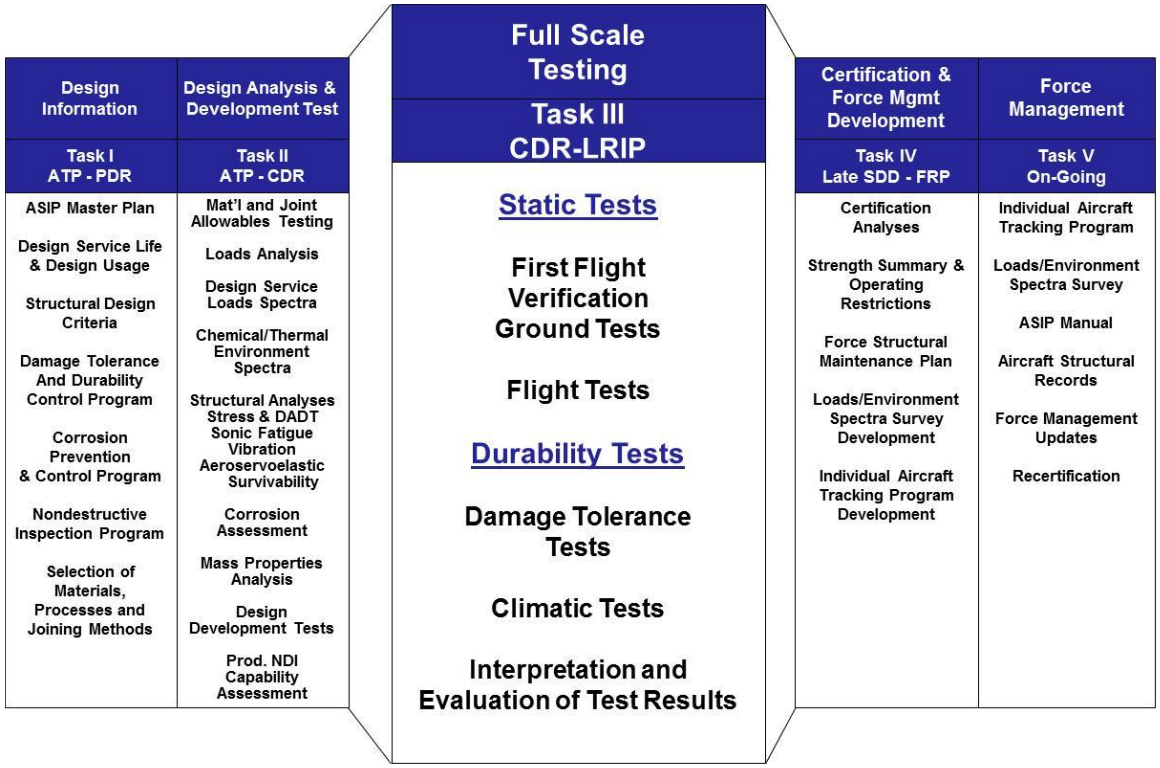


Figure 9.11-45. Relationship of F-35 Full-Scale Tests to ASIP

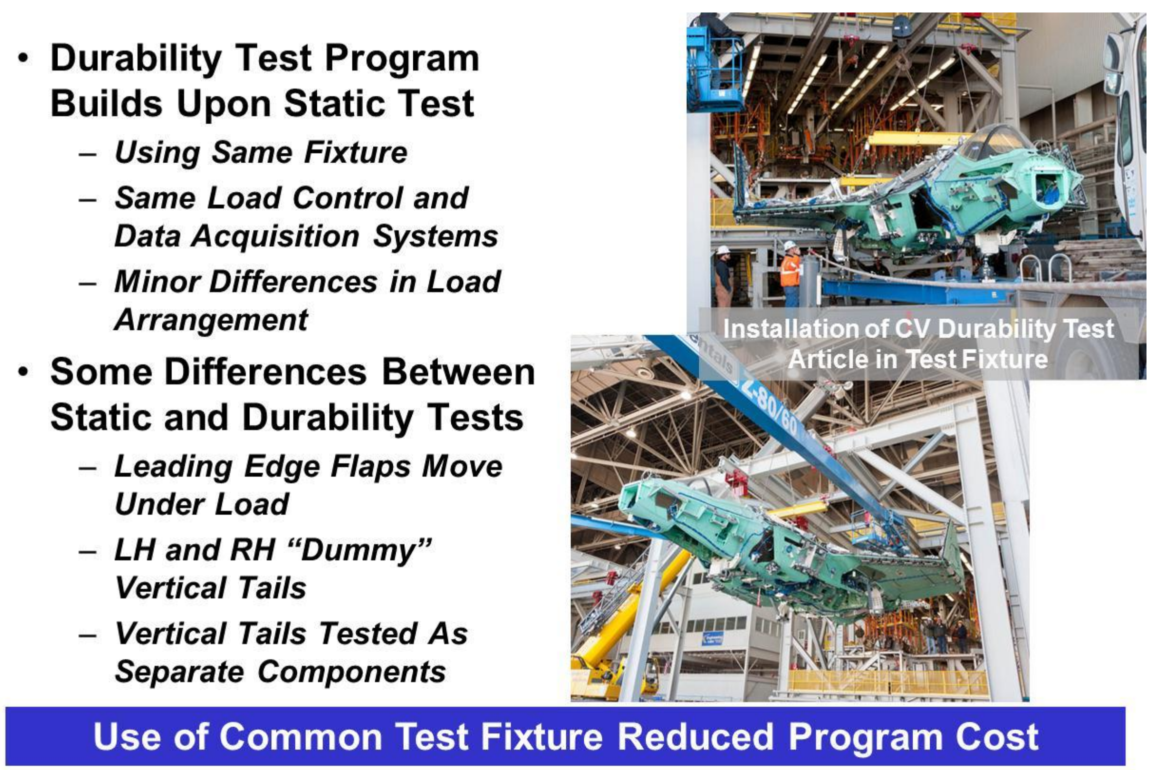


Figure 9.11-46. Full-Scale-Durability Test Development

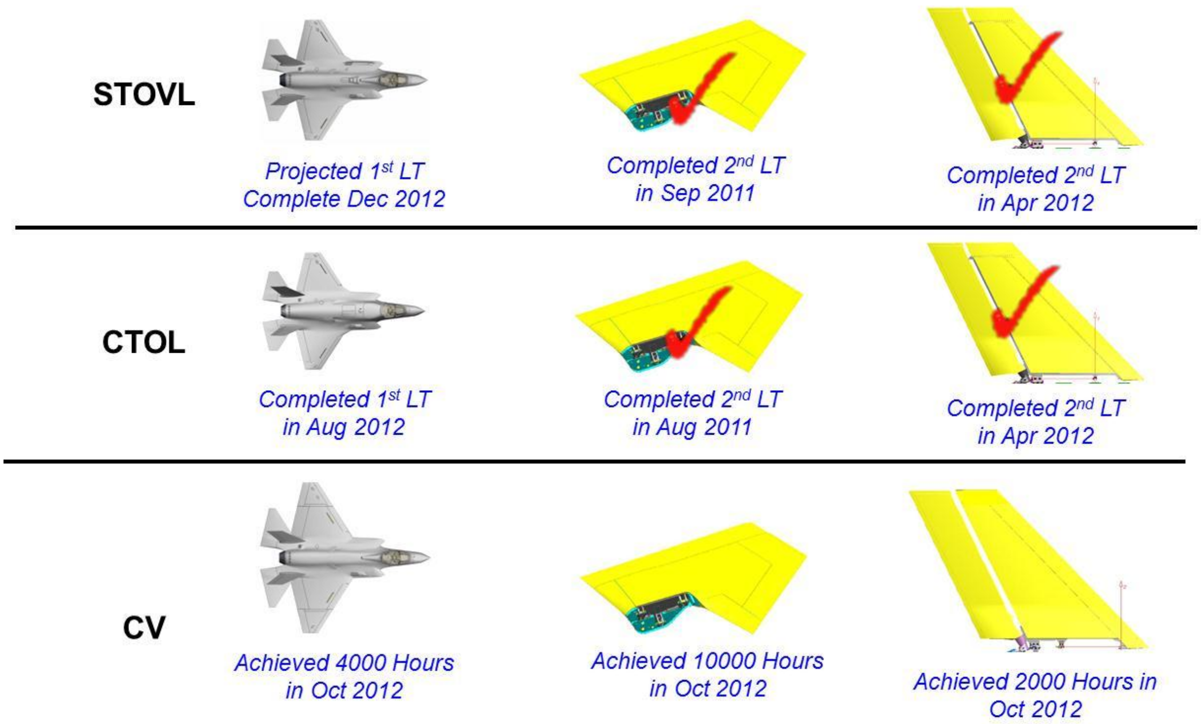


Figure 9.11-47. F-35 Durability Test Status

9.11.17. C-130 Force Structural Maintenance Plan Update

Darrin Fritz, USAF-WR-ALC; John Edwards and Jason Ward, Mercer Engineering Research Center (MERC)

This technical activity details the 2012 update of the C-130 Force Structural Maintenance Plan (FSMP), which provided review and revision of the fleet monitoring, tracking, and maintenance protocols. Mercer Engineering Research Center led an extensive effort to revise the inspections, events, and processes that form the backbone of the Automated Inspection, Repair, Corrosion and Aircraft Tracking (AIRCAT) (Figure 9.11-48), which serves as the C-130 web-based individual aircraft tracking program (IATP). The work included incorporation of severity factor variability methodology (Figure 9.11-49), redesign of IATP algorithms and reports, modification/creation of inspection procedures (Figure 9.11-50), detailed update of damage records, and corrections/expert analysis in support of the 2012 scheduled update to the AIRCAT system. The resultant FSMP is a static assessment of the 2012 C-130 fleet, a qualitative review of the underlying algorithms and data, and a web-based utility for real-time updates via the IATP reports.

FSMP updates to the Inspection, Corrosion, And Repair Reporting (ICARR) and AIRCAT systems ensure accurate and detailed data from inspection accomplishments and findings for fleet management, establish new techniques for analysis of field reported data, and incorporate all available legacy damage records into the ICARR and AIRCAT history tables. Updates to the Fracture Growth Tracking program improve inspection tracking and provide the Aircraft Structural Integrity Program (ASIP) Manager with a detailed view of projected inspection requirement dates against scheduled inspections.

Assessment of the DTA updates to the AIRCAT system led to several improvements and revisions. The FSMP update developed techniques and reports to assess and account for Severity Factor and Usage variance within the fleet. A detailed error analysis assessed the methods of estimating future usage and identified the optimum window for predicting usage. The FSMP effort also corrected gaps in the methodology for defining Fracture Zones and associated inspections within the AIRCAT system and established Risk Analysis-based tracking within the AIRCAT system using Air Force risk assessment guidelines for the C-130 rainbow fitting structure. The new Risk Method allows the ASIP Manager to monitor these zones as traditional DTA-based zones for forecasting inspections and tracking results.

The 2010-2012 FSMP updates the input, processing, and reporting of ASIP critical information. The result is a suite of improved C-130 ASIP tools to manage the structural health of the fleet and to more effectively and efficiently incorporate any future updates to the underlying analyses and requirements. This technical activity details the scope of the update and the methodologies developed for improving the C-130 FSMP and IATP.

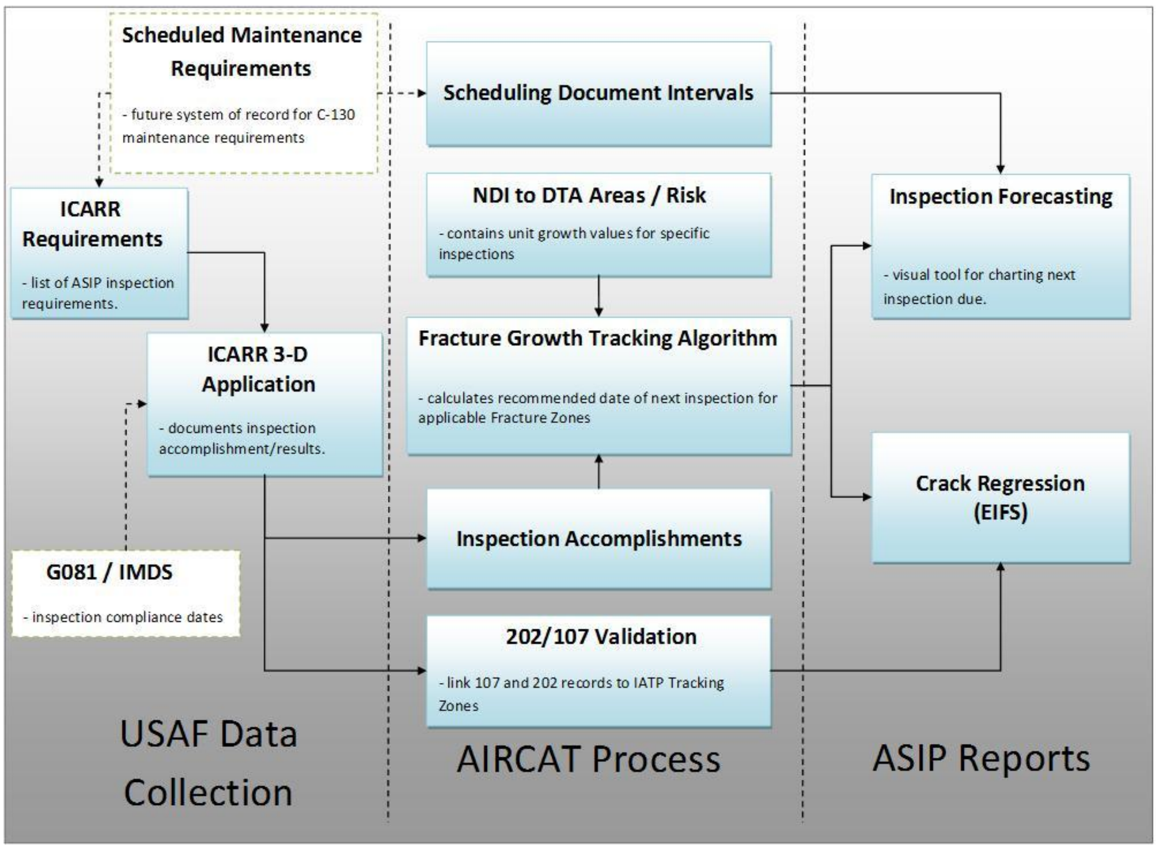
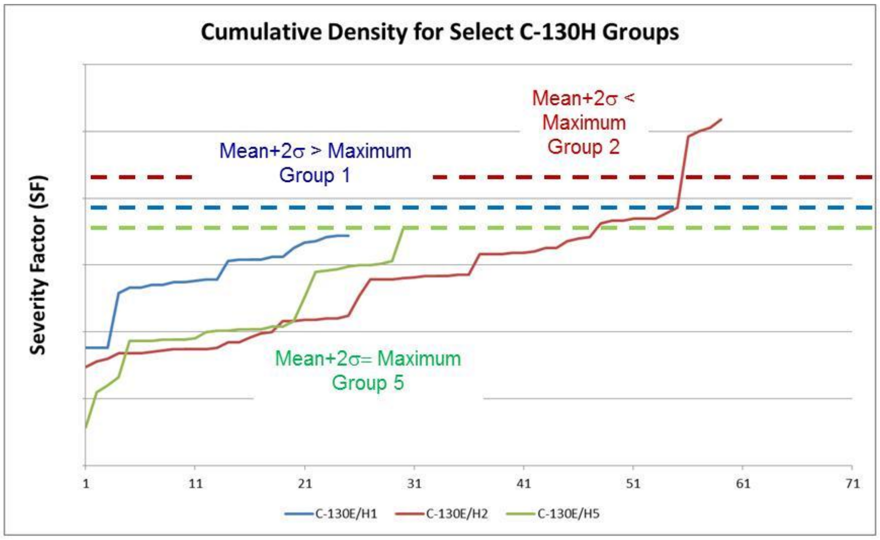
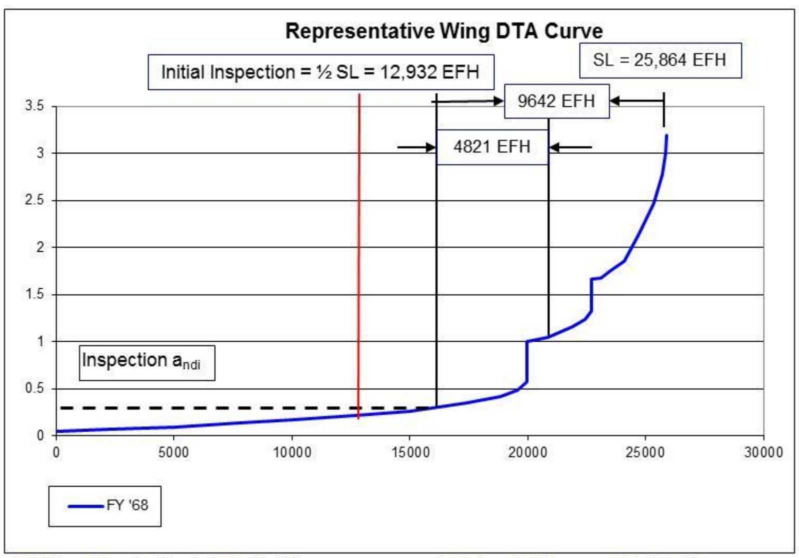


Figure 9.11-48. C-130 AIRCAT



- Dashed lines represent Mean+2σ SF for each Group (1,2, & 5)
- Note that the Mean+2σ SF exceeds the Maximum in Group 1 (blue)
- Mean+2σ excludes the outliers in Group 2 (red), seen at tail of curve

Figure 9.11-49. Representative Severity Factor Determination



EFH = Equivalent Flight Hours   AFH = Airframe Flight Hours

SF = Severity Factor = EFH/AFH   SL = Safety Limit

- For example above, a_{ndi} to SL = 9642 EFH
- ***Recurring Interval* = 9642/2 = *4821 EFH***

- DTA drives ASIP inspections
- Recurring interval is ½ the distance b/w detectable crack length (a_{ndi}) and the Safety Limit (SL)
- These recurring intervals must be mapped to Airframe Flight Hours (AFH) for execution
- Remember the 4821 EFH recurring interval from this example…

Figure 9.11-50. Inspection Scheduling Process

9.11.18. Alleviating Maintenance Burden: Converting F-16 Fleet Management from Slow Crack Growth to Fail-Safety

Stephanie McMillan, Lockheed Martin Corporation

As the F-16 fleet continues to age, maintenance inspections driven by slow-crack-growth methodology increase in frequency and in scope. As the number of inspections increase, concerns arise with regards to the number of cracks that may be missed, the damage that may be caused during the inspection procedure, and the effect of the length of downtime on mission planning. To mitigate these concerns, the United States Air Force (USAF) F-16 community is turning toward management by fail-safety. This methodology, however, comes with limitations. Aircraft managed primarily by fail-safety are managed for safety only, not economics. Also, by the USAF determined definition of fail-safety, the entire structure cannot be managed by fail-safety due to considerations of load-path redundancies and design damage sizes. Other difficulties present themselves in the form of retrofit actions and recordkeeping of such actions, inspectability issues, and the ability to determine crack arrestment for partial failure scenarios. Fail-safety becomes very beneficial, however, when invasive inspection procedures can be replaced with visual inspection criteria in critical areas (Figure 9.11-51). With careful consideration, the F-16 structure can be managed by a combination of slow crack growth and fail-safety methodologies to mitigate both inspection burden and economic burden. This technical activity outlines the definition and requirements of slow crack growth and fail-safety methodologies, benefits and limitations of both methodologies, examples of life and inspection calculations (Figure 9.11-52 and 9.11-53), and lessons learned from many recent fail-safety analyses performed for USAF in the past year.

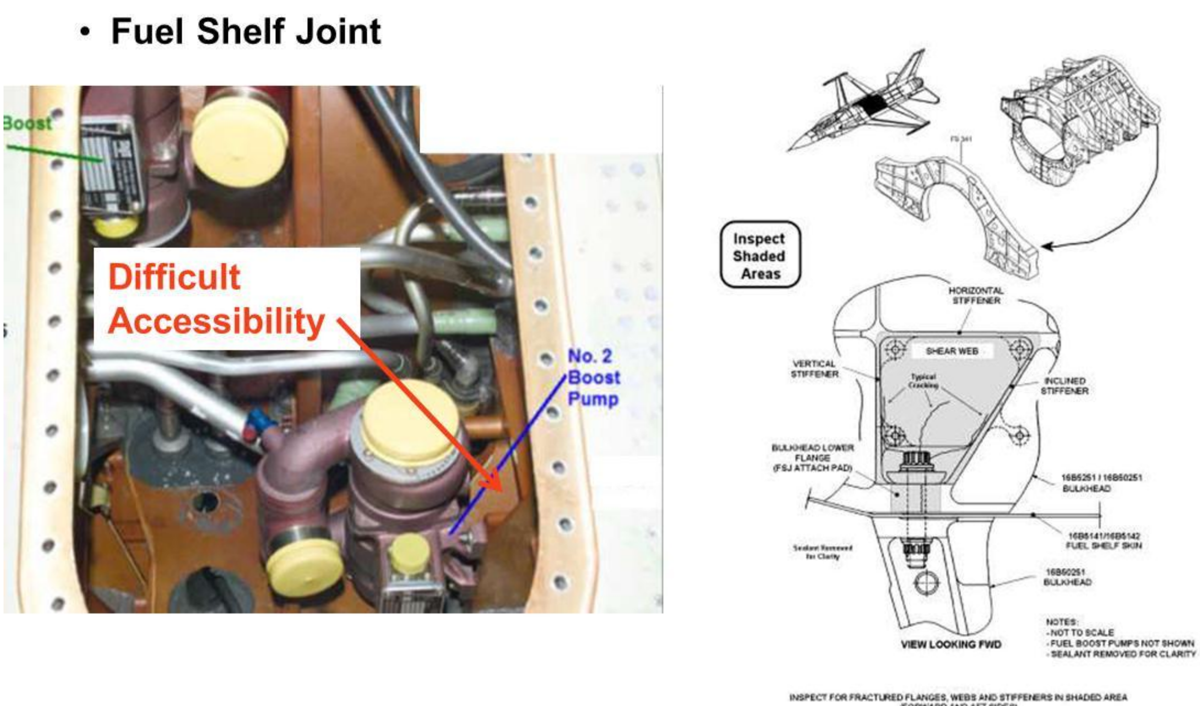


Figure 9.11-51. Example Location for Fail-Safety

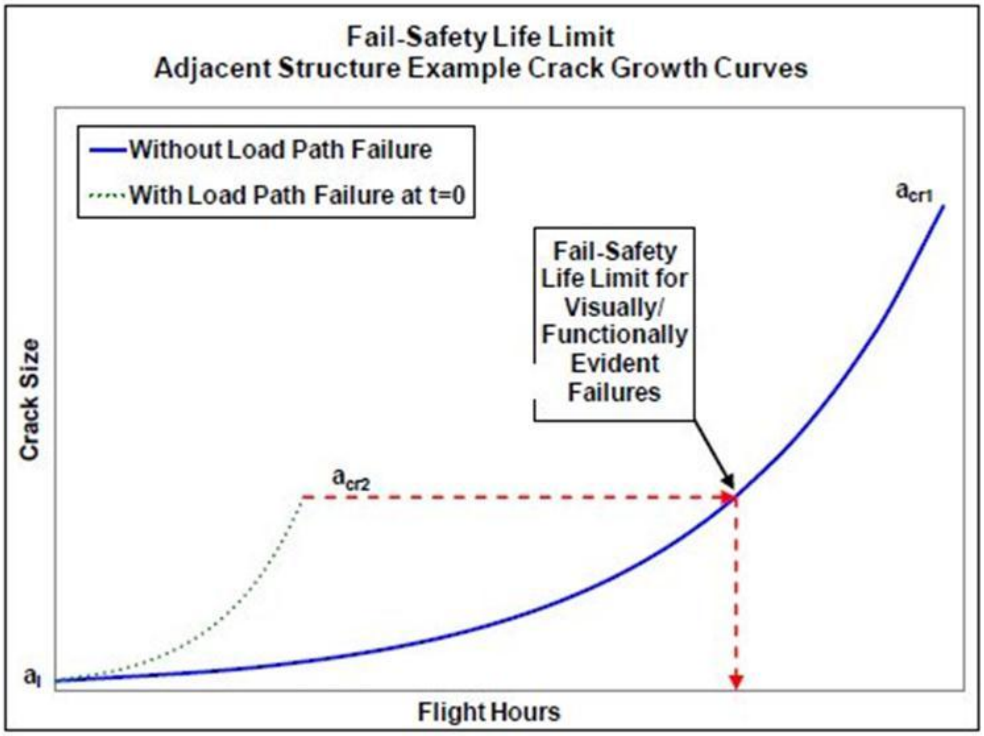


Figure 9.11-52. Fail-Safety Life Limit (FSLL) for Visually/Functionally Evident Failures

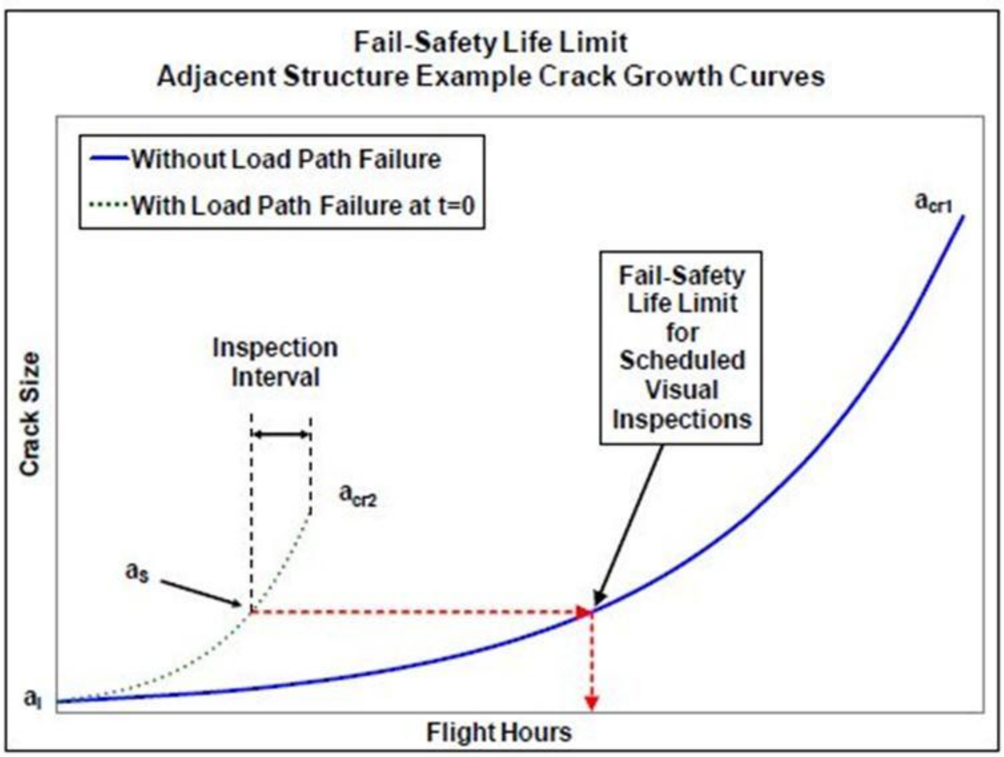


Figure 9.11-53. Fail-Safety Life Limit (FSLL) for Scheduled Visual Inspections

9.11.19. T-6 Texan II Control Stick Lever

Christopher Ifft, USAF Life Cycle Management Center

September 13, 2011, an in-flight failure occurred on aircraft Pt-244. The aft control stick lever right lobe was fractured after pulling out of a maneuver (Figures 9.11-54 and 9.11-55). This resulted in the student pilot losing aileron control, and the instructor pilot losing elevator control. Coordination between the student and the instructor pilot allowed for a safe landing of the aircraft. This generated an investigation into the failure of the control stick lever, an inspection of all control stick levers in the fleet and production, and an eventual replacement of all control stick levers with a more robust design. This technical activity covers the contributing factors of the control stick lever design that lead to: the in-flight failure, the steps that were taken to attempt to discover the loads that lead to the failure of PT-244's aft control stick lever, the Air Force Research Laboratory characterization of cracked control stick levers, the steps taken to identify all of the cracked control stick levers (Figure 9.11-56), the redesign of the control stick lever (Figure 9.11-57), and the testing to show the new control stick lever is a robust design (Figure 9.11-58). This technical activity reveals a vital reason why flight controls should not be made of castings and how ASIP standards should be applied to MECSIP.

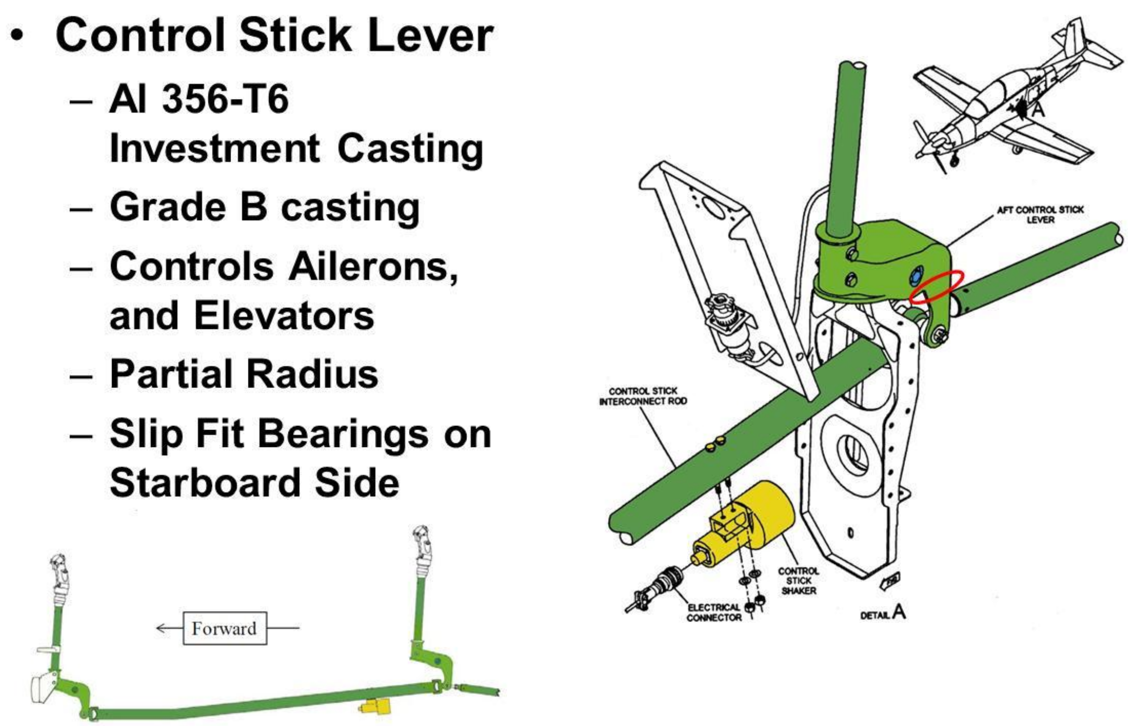


Figure 9.11-54. Control Stick Lever Background

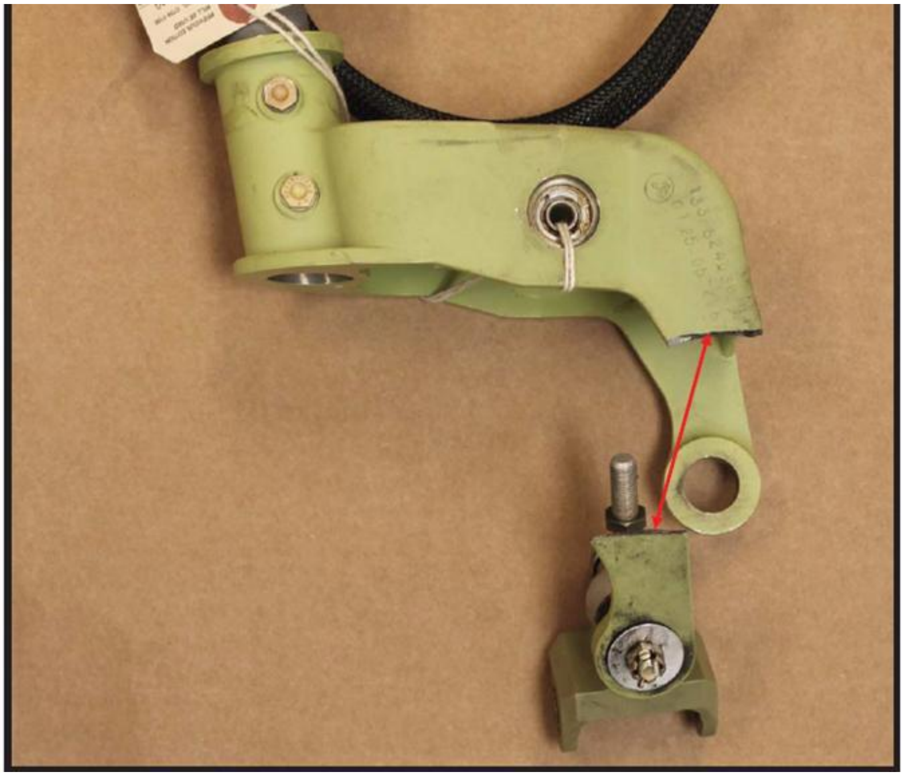
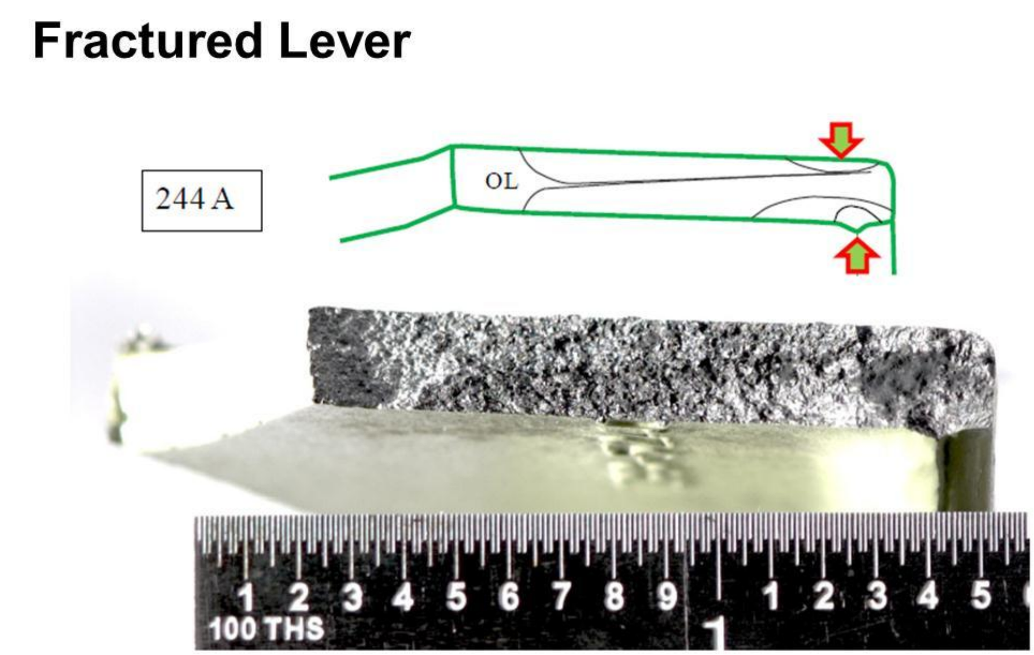


Figure 9.11-55. Incident Aircraft Control Stick Lever



OL= Overload

Green Arrows = Origination Points

Figure 9.11-56. AFRL Failure Analysis

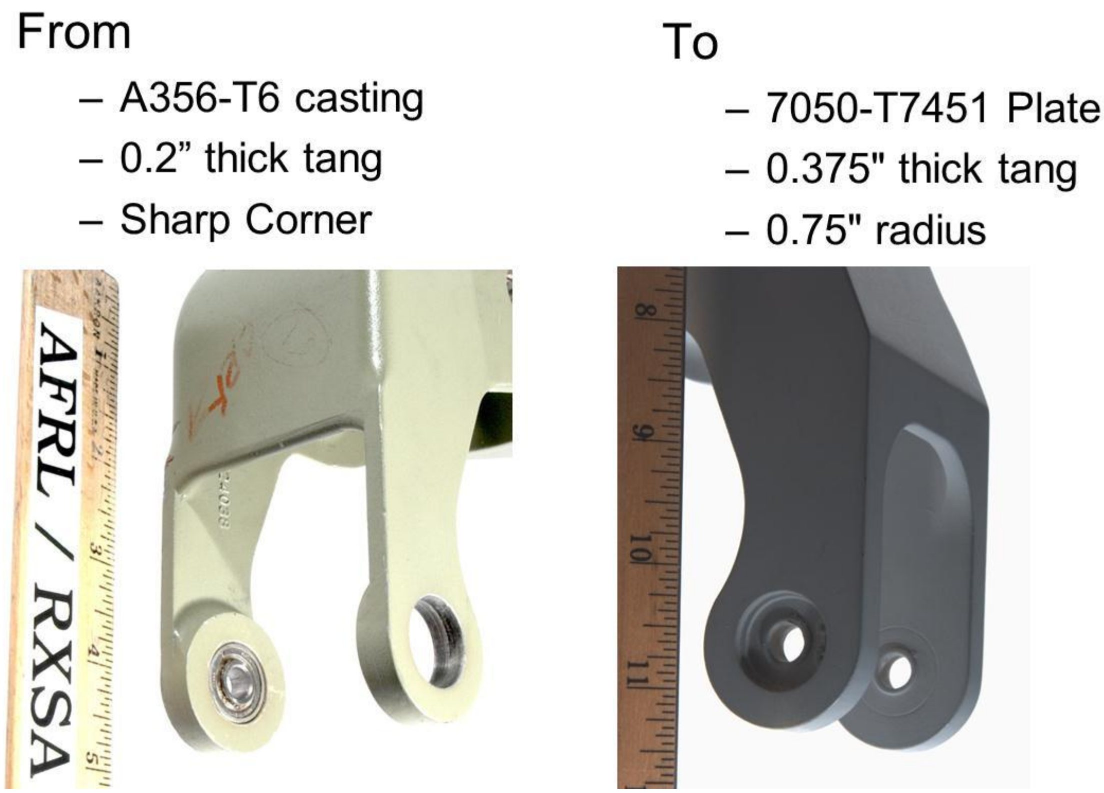
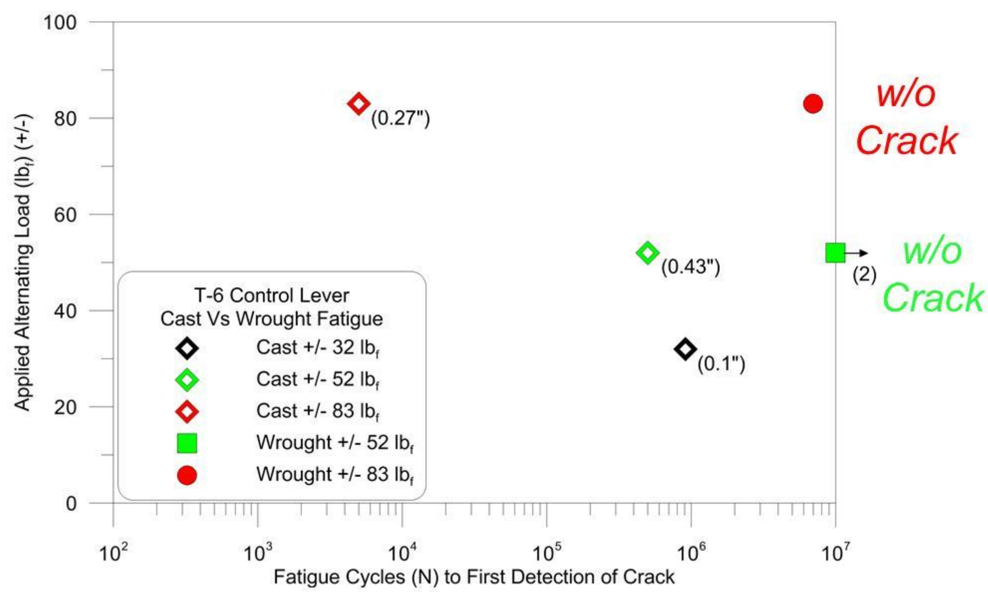


Figure 9.11-57. Design Changes



No cracks have been induced in wrought levers.

At +/- 83 lbf, cracking of the cast article was first noted at 5,000 cycles, one wrought lever has been cycled more than 7.9 Million times (> 1000X life).

Figure 9.11-58. Life Improvement

9.11.20 Advanced Drilling and Assembly Processes: One-Up Assembly

Antonio Rufin and Tanni Sisco, The Boeing Company – Commercial Airplanes

New materials, production rate pressures, a greater share of manufacturing and assembly responsibilities by global partners and suppliers, and the drive for higher manufacturing efficiencies are giving impulse to the development of new assembly technologies, a greater emphasis on process automation, and in some cases, a radical reassessment of production practices.

The Boeing 787 (Figure 9.11-59) program is an example of an airplane where many of these considerations have come together. With a large share of the structure consisting of advanced composite materials (Figure9.11-60) and many of its elements being produced as complete assemblies at different sites worldwide, production processes and the engineering behind them have had to adapt.

Figure 9.11-59. Boeing 787 Dreamliner

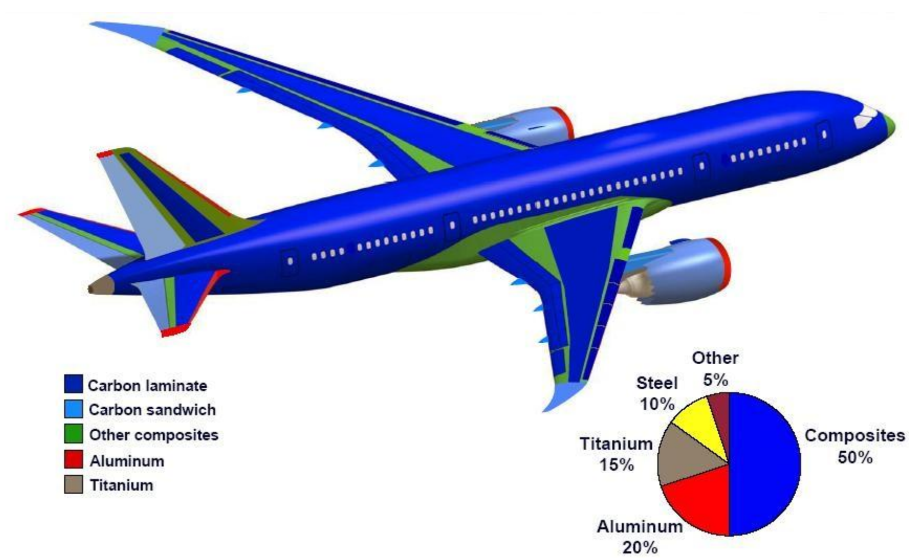


Figure 9.11-60. Boeing 787 Structural Materials

One of the greatest challenges in building the 787 is that build processes for certain areas of the airframe no longer permit adjoining elements to be separated and deburred after hole drilling. Additionally, elimination from the drilling process sequence of separate, clean, and deburr operations can increase production efficiency by reducing time-consuming manual labor steps and by enabling, or at least facilitating the introduction of automated assembly methods, making one-up assembly processes desirable from a production economics viewpoint. The downside is that burrs (in this context, rough hole edges and trapped foreign matter in the joint –not just simply the material projected out of the hole during drilling or reaming, Figures 9.11-61 and 9.11-62) represent a less-than-optimal hole quality and can have a deleterious effect on fatigue. This is an issue common to both all-metal and metal-composite joints. Further, indications are the impact on fatigue properties is highly sensitive to material, titanium for example being more sensitive than aluminum (Figure 9.11-63).

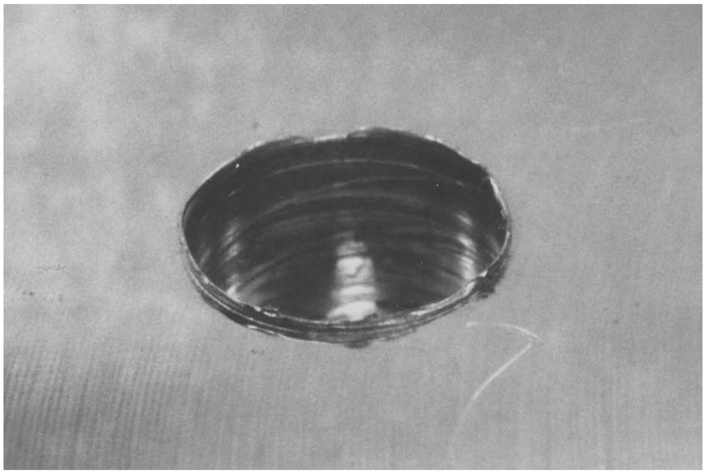


Figure 9.11-61. Hole Exit Burr in Titanium

Figure 9.11-62. Interface Burrs and Surface Contamination in Carbon Fiber-Reinforced Composite-Metal Joint

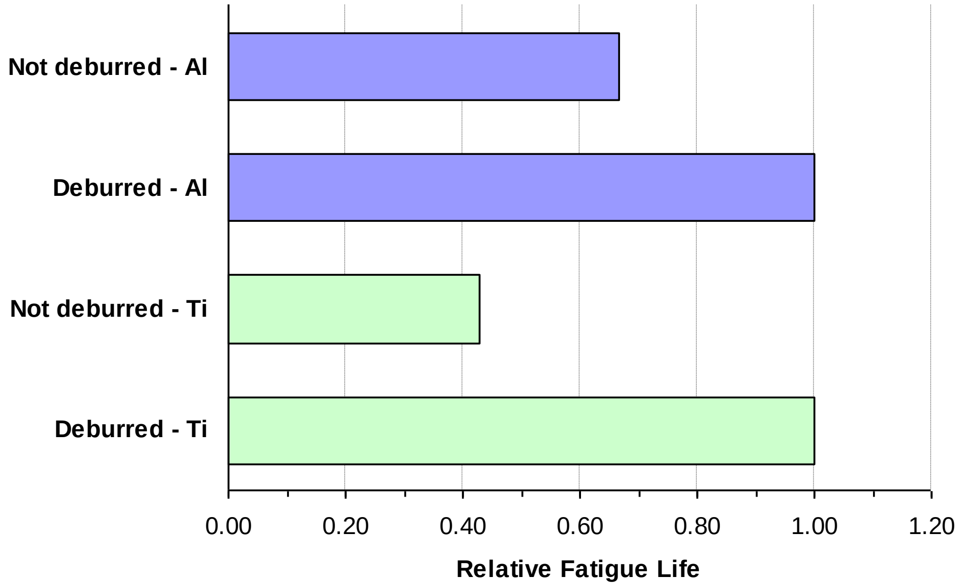


Figure 9.11-63. Typical Fatigue Performance, Non-Deburred High-Load Transfer Joints

The effect of this condition on fatigue performance has long been known, and the use at Boeing of one-up assembly processes is not entirely new, but the goal now is to extend their use in non-traditional areas of the structure, though obviously in a disciplined fashion and with an appropriate level of generic engineering analysis coverage. To make this possible, a candidate "one-up assembly" process must meet three basic criteria: (1) it must be stable, controlled, and appropriately documented, (2) where feasible, consideration needs to be given to the application of techniques capable of offsetting the potentially adverse effect of the one-up assembly condition of the joint on durability, and (3) the supporting engineering data and analysis have to be in place so that the impact of the process on structural integrity can be assessed.

Recent efforts at Boeing Commercial Airplanes on one-up assembly have greatly facilitated a widespread implementation of this technology on the 787 program with full engineering support in the form of generic design values for controlled one-up assembly processes, based on extensive testing. The focus is now gradually shifting to the extension to other airplane programs, both legacy and new.