A REVIEW OF RESEARCH
ON
AERONAUTICAL FATIGUE IN THE UNITED STATES

2009-2011

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9.1. INTRODUCTION

Leading government laboratories, universities and aerospace manufacturers were invited to contribute summaries of 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 – William J. Hughes Technical Center
- NASA – Johnson Space Center
- NASA – Langley Research Center
- USAF Academy – CAStLE
- USAF – AFMC/EN
- USAF – F-22 SPO
- USAF-OO-ALC
- USAF Research Laboratory – Air Vehicles Directorate
- USAF Research Laboratory – Materials and Manufacturing Directorate
- USAF-WR-ALC
- USN – NAVAIR

Academia
- Clarkson University
- Georgia Institute of Technology
- Mississippi State University
- University of Texas
- Wichita State University – NIAR

Industry
- Alcoa – Defense
- Alcoa – Technical Center
- Alion Science and Technology
- APES, Inc.
- Computational Mechanics, Inc.
- Computational Tools
- Fatigue Technology
- Jacobs ESCG
- KB Inspection Services
- Lambda Technologies
- Lockheed Martin Aero – Ft. Worth
- Mercer Engineering Research Center (MERC)
- Metal Improvement Company
- Mustard Seed Software
- Northrop Grumman Corporation
- Radiance Technologies, Inc.
- SAIC – Hill AFB
- Southwest Research Institute

- Technical Data Analysis, Inc.
- The Boeing Company – Defense, Space & Security
- The Boeing Company – F-22 Engineering
- The Boeing Company – Integrated Defense Systems
- The Boeing Company – Research and Technology
- Tom Brussat Engineering, LLC
- TRI/Austin
- Ultra Electronics
- UniWest

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.

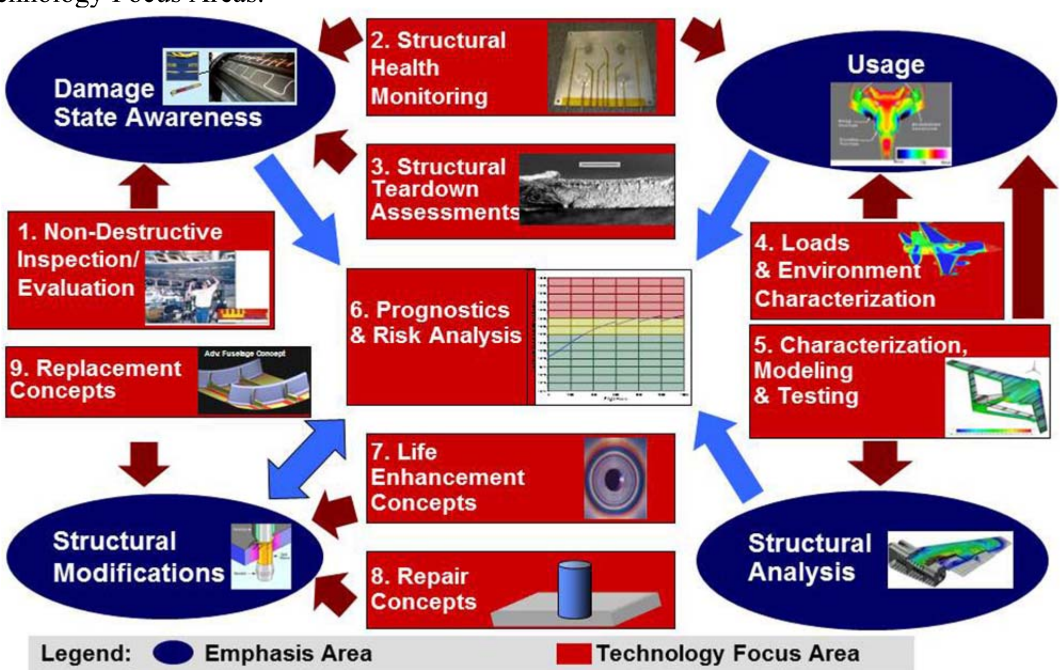


Figure 9.1-1. Condition Based Maintenance + Structural Integrity (CBM+SI)

9.2. NON-DESTRUCTIVE INSPECTION/EVALUATION

9.2.1. The United States Air Force Nondestructive Inspection Improvement Program

David Forsyth and Mark Keizer, TRI/Austin; Mark Gehlen and Carlos Pairazaman, UniWest; Jeff Guthrie, Michael Morgan and Darren Stamper, Alion Science and Technology; Ronald Kent, KB Inspection Services; and Damasco Carreon, USAF Research Laboratory – Materials and Manufacturing Directorate.

As part of the continual effort to improve the United States Air Force inspection capability, the Nondestructive Inspection Improvement Program (NDIIP) performed a third party review on Safety Of Flight (SOF) inspections on different airframes, to ensure that technical orders (T.O.s) accurately describe details necessary to effectively accomplish the inspections. The scope includes the initiation of organizational and work management strategies to improve the process and reduce human error factors that can, on rare occasions, cause large cracks to be missed. Ninety inspections were reviewed and the redlined T.O.s have been delivered to the Air Force. Another 90 SOF inspections are currently being reviewed in addition to 27 labor intensive inspections.

This effort describes a formal process to evaluate inspection procedures and the metrics used to quantify inspection procedure reliability and cost/time effectiveness. Alion Science and Technology and TRI/Austin personnel used the process to evaluate Air Force inspections at the three Air Logistics Centers, gathered inputs from Air Force inspectors, and modified technical orders for implementation by the Air Force.

During the inspection reviews, categories of similar inspections were identified. One of the issues that occurred on every airframe was eddy current inspection around raised head fasteners. Probe kits to improve the inspection around raised fasteners were developed, tested, and delivered to the USAF depots for use (Figures 9.2-1 and 9.2-2). A POD study to assess the real ability of these probes to increase the sensitivity for these inspections is also discussed (Figure 9.2-3).

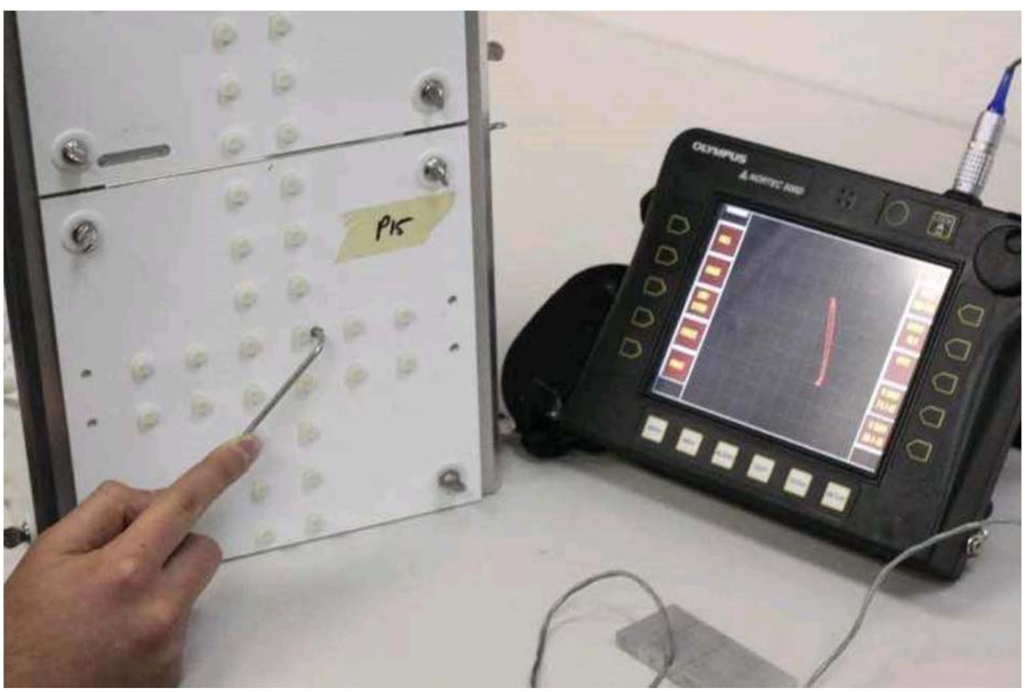


Figure 9.2-1. Existing USAF Specimens from Previous Pencil Probe QAPA

Figure 9.2-2. New Raised-Head-Fastener (RHF) Probes

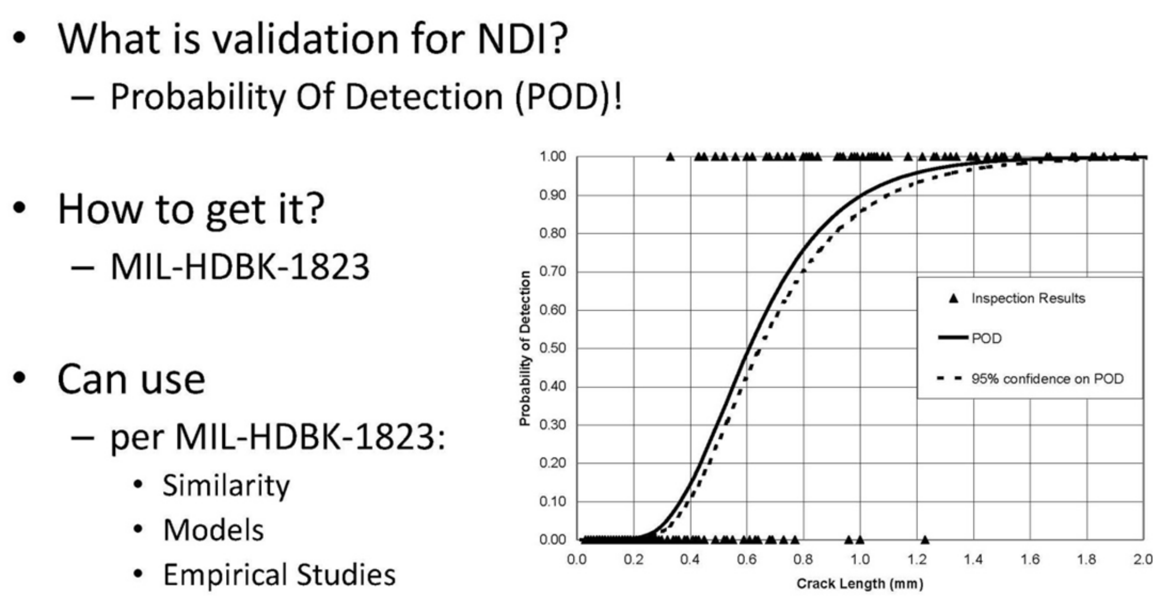


Figure 9.2-3. Validation via Probability of Detection (POD)

9.2.2. Inspection Sample Sizes: Going Beyond AFMCI 21-102

Zachary Whitman, Southwest Research Institute

Air Force Material Command Instruction (AFMCI) 21-102 provides guidance for Analytical Condition Inspection (ACI) programs. Recommended sample sizes are given for a prevalence level of 20% in the fleet at 90% confidence. Unfortunately, sample sizes are often different than those recommended in the instruction, both smaller and larger, which affects how many non-conformances will be found by the sample inspection. The sample sizes were likely established using either the binomial or, more appropriately, the hypergeometric distribution which considers sampling from a finite population without replacement. Fundamentally, inspection sampling is always done without replacement since one aircraft would not be inspected more than once; thus, one of the underlying assumptions of the binomial is violated. Both the binomial and hypergeometric distributions are based upon selection of the sample where all candidates have the same likelihood of non-conformance. However, the sampling recommendation in AFMCI 21-102 purposely weights the sample toward the ‘the most severe stratum’; acknowledging that the probability for each aircraft is not equal (Figure 9.2-4). These issues, as well as others, have led to a new methodology being used by the T-38 Aircraft Structural Integrity Program (ASIP).

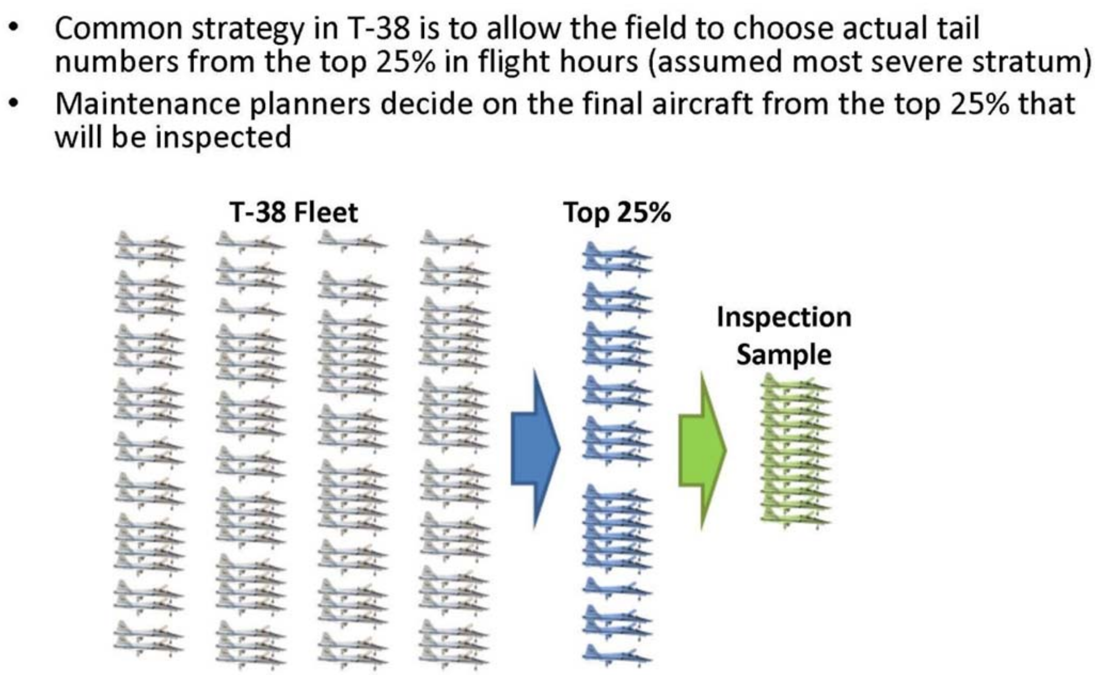


Figure 9.2-4. T-38 Application to Analytical Condition Inspection (ACI)

Fleet prevalence can easily be calculated over a wide range of sample sizes and inspection findings by using Monte Carlo Simulation (MCS). Different confidence intervals do not require re-analysis since the described technique evaluates the MCS data for rank percentiles corresponding to the desired confidence levels. In addition, different sampling strategies can be tested that are not possible using conventional statistical distributions and assumptions. Furthermore, if the failure mode is well described, such as with a Weibull model, then selective sampling can be considered. For example, in a recent T-38 challenge, an inspection was needed that would find a low likelihood-of-cracking prevalence (1%) at high confidence (95%). Since a previous Weibull analysis had been performed for this location a shape factor existed describing the failure mode. The equivalent hours on each aircraft was also known. A MCS was generated for different inspection scenarios to evaluate how many of the most at-risk aircraft needed to be inspected. The result was a sample size and strategy that would verify that the fleet was 99% crack free at 95% confidence after inspecting only a fraction of the entire fleet, far less than would have been needed had a binomial or hypergeometric distribution been assumed (Figure 9.2-5). This case, along with other recent applications, is reviewed and discussed.

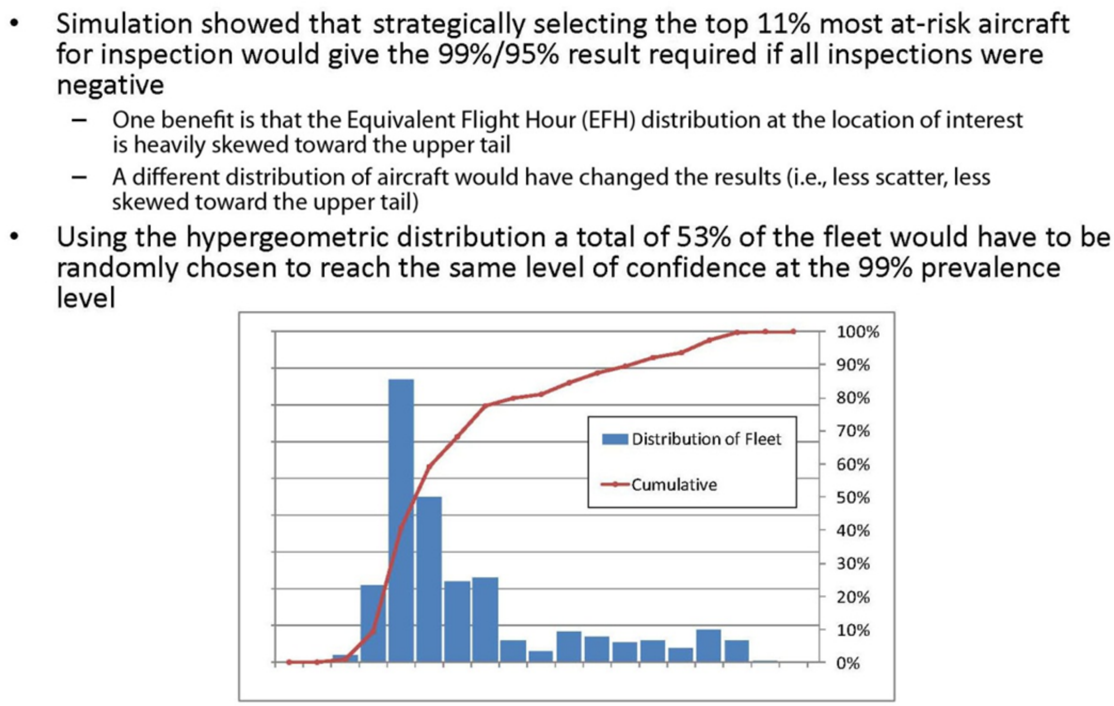


Figure 9.2-5. T-38 TCTO Sample Design

9.2.3. F-16 ASIP Impacts of NDI Capability Guidelines for USAF Structures

Bryce Harris and Kimberli Jones, USAF-OO-ALC; Tim Jeske, Lockheed Martin Aero – Ft. Worth; and Sean McIntyre, SAIC – Hill AFB.

Several USAF Structures Bulletins have been recently released to address significant concerns about USAF assets with the intent to mitigate risk associated with aging fleets. Overviews of the bulletins will be compared to F-16 inspections to explain the revised fail-safe approach and the requirements to ensure that safety-of-flight structure is adequately defined and inspected. The focus of this activity is on Structures Bulletin EN-SB-08-012, which defines the recommended NDI flaw size capabilities for computing the reinspection intervals for structures managed by USAF ASIP. Comparison data is provided for F-16C Block 25/30/32 aircraft that relates the NDI detection capabilities as a function of the various inspection methods for the Lockheed Martin F-16 legacy sizes, the updated 2006 recommended sizes, and the flaw sizes as defined by Service Bulletin EN-SB-08-012. This source data is used to evaluate the impact of the new guidelines on the resulting reinspection intervals and to determine sustainment costs due to more frequent inspections resulting from the published detection capabilities. The impact of the new guidelines on the resulting reinspection intervals for a bulkhead vertical stiffener bolt hole is presented in Figure 9.2-6.

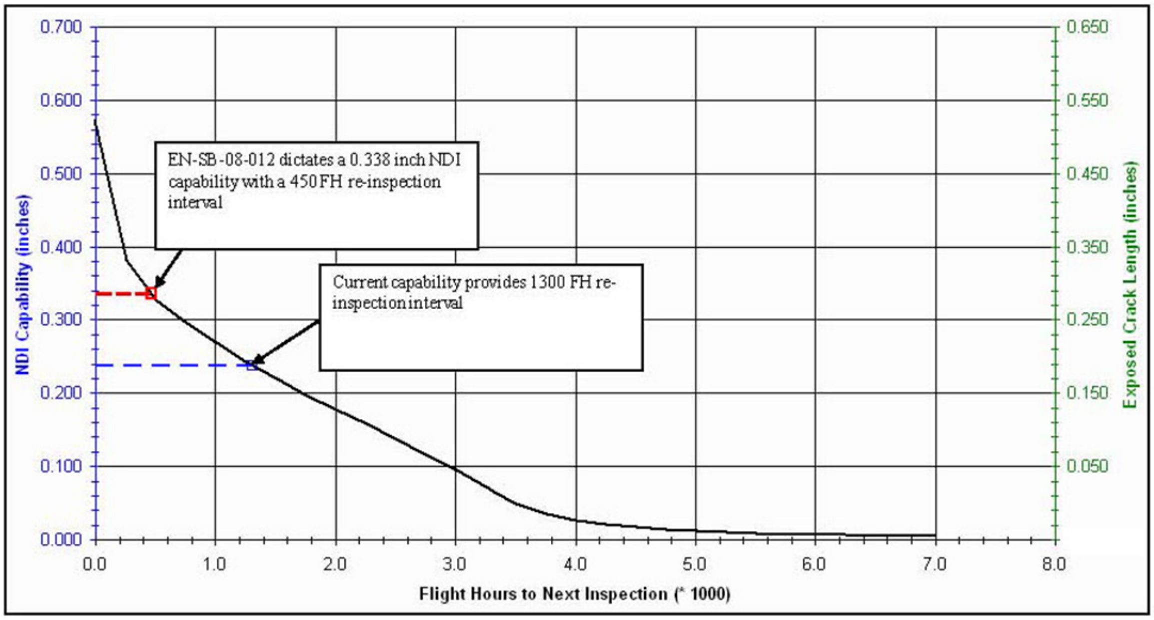


Figure 9.2-6. Bulkhead Vertical Stiffener Bolt Hole Example

When inspection requirements are changed, there are various areas, including cost and aircraft availability, that are affected and should be considered. Predicted cost data considers disassembly requirements, the magnitude of the inspection area, the NDI method, configuration, and procedures. These data are readily available in the current Force Structural Maintenance Plan, but may need updates due to more frequent inspections for non-fail-safe structure. The increased sustainment costs can be mitigated by improved NDI procedures or by managing using fail-safe criteria, which can create situations that are not easy to resolve. Strictly managing according to safety ignores the possible economic impacts. The economics may dictate that cracks are detected within repair limits or prior to the need for major modification requirements. Inspection areas requiring improved NDI procedures are investigated and developed to reduce the sustainment costs and to allow for economical management. Example cost and aircraft downtime data associated with new inspection requirements are presented to support development of maintenance policies and as source data for the cost effectiveness of improved NDI procedures.

9.2.4. Large Area Detection of Cracks Using Magnetoresistive Sensor Arrays

Donald D. Palmer, Jr., The Boeing Company – Research and Technology; Nancy L. Wood, The Boeing Company – Integrated Defense Systems; and Charles F. Buynak, USAF Research Laboratory – Materials and 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. A number of studies have shown that magnetoresistive sensor technology improves the ability to detect smaller flaws at greater depths compared to currently deployed eddy current capabilities (Figures 9.2-7 and 9.2-8). The U.S. Air Force has sponsored a number of research and development initiatives over the past several years directed at deployment of magnetoresistive sensor arrays into depot maintenance operations. The deployment strategy focused on sensor integration into large area scanning platforms currently used at the Air Logistics Centers. This activity focuses on development of (1) system enhancements necessary to accommodate MR sensor arrays, (2) image interpretation algorithms to aid in interpretation of MR sensor-generated data, and (3) an MR-based large area inspection process to address detection of cracks in thick and multi-layer wing structure. Future directions for MR sensor technology as they relate to structural life enhancement initiatives are also discussed.

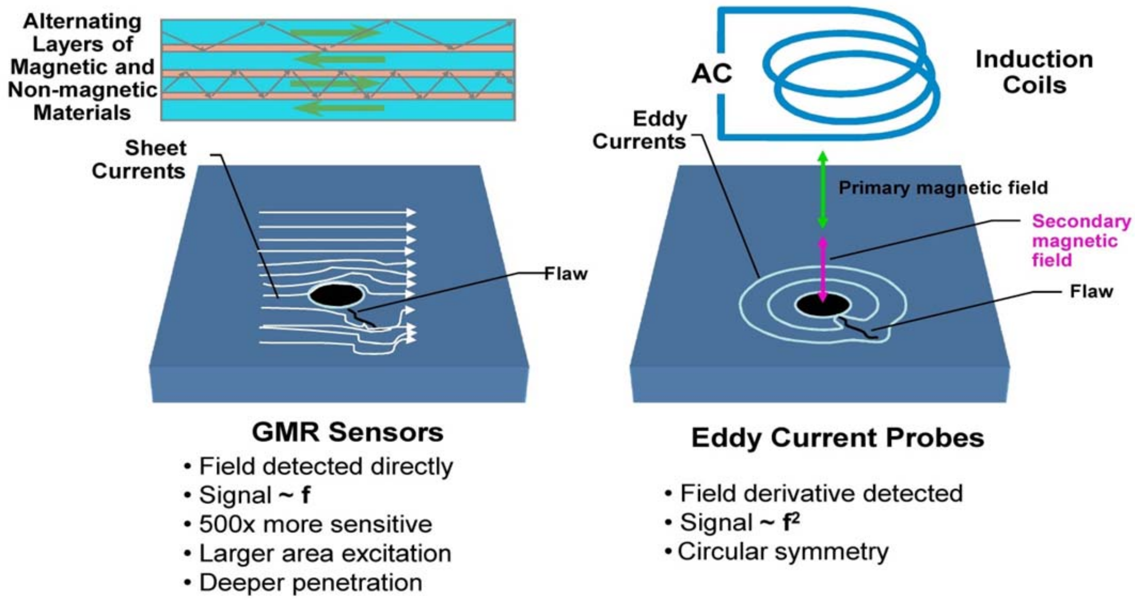


Figure 9.2-7. Magnetoresistive (MR) Sensors vs. Eddy Current Probes

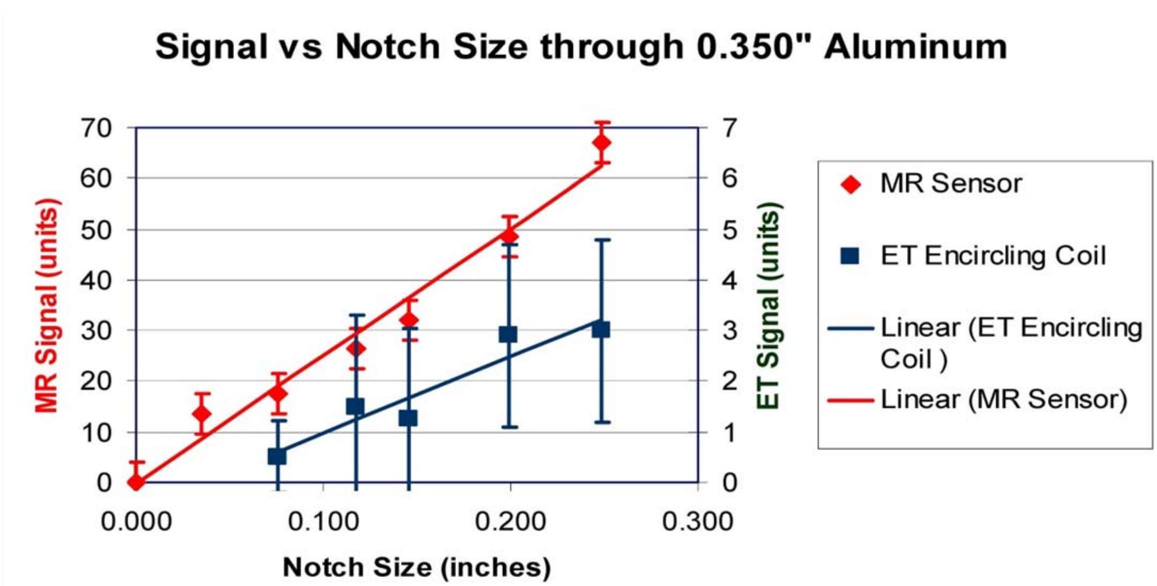


Figure 9.2-8. Quantitative Comparison Through Thick Aluminum

9.2.5. Computed Radiography vs. Conventional Film Radiography for Crack Detection

Kenneth J. LaCivita, USAF Research Laboratory – Materials and Manufacturing Directorate

Due to the increase in cost, decrease in availability, and hazards associated with waste disposal of materials and supplies for film, transition to digital means of x-ray data imaging are required for aerospace inspections requiring radiography. Numerous aircraft maintenance facilities are currently using x-ray Computed Radiography (CR) for the detection of water and foreign object damage (FOD), but are currently unable to use CR for crack detection due to the lack of validated inspection procedures using CR methods (Figures 9.2-9 and 9.2-10). A research program is currently underway to perform a validation study to examine the detection capability of CR for crack detection (Figure 9.2-11). Under this program, guidelines and procedures for conducting x-ray inspections using CR will be developed, and those processes validated using engineered crack specimens and actual aircraft components. This research activity will discuss results of the comparative study for both film based x-ray and CR for crack detection in realistic aerospace structures (Figure 9.2-12). A related topic, digital radiography for production acceptance of aerospace castings, will also be briefly summarized.

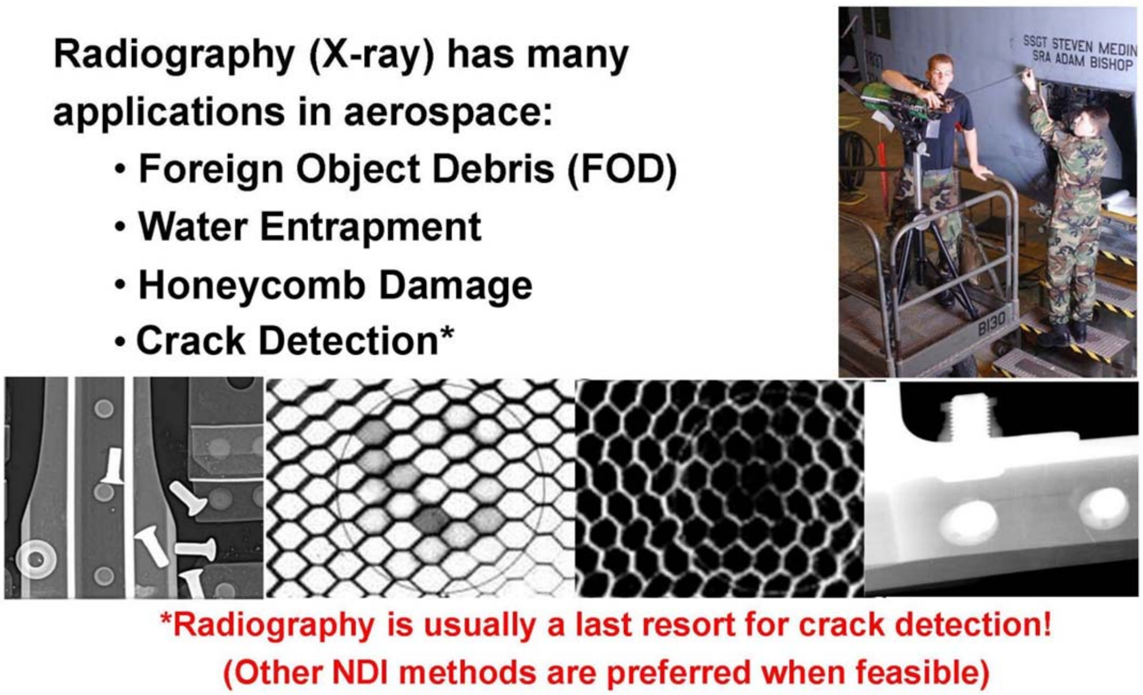
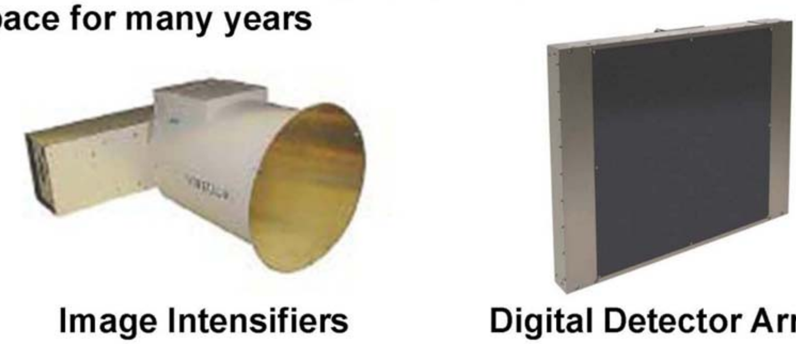


Figure 9.2-9. Radiography Aerospace Applications

- **Film radiography has been the primary modality used in the USAF**
- **Some forms of digital radiography (DR) have been used in aerospace for many years**



But for these DR technologies:

- **digital pixels are "large" (150 microns +)**
- **typically require:**
  - **"micro-focus" x-ray tubes and "geometric magnification"**

impractical for field use!

Figure 9.2-10. Digital Radiography Technologies

- POD analysis in work
  - May establish detection limits
  - Publish in Structures Bulletin?

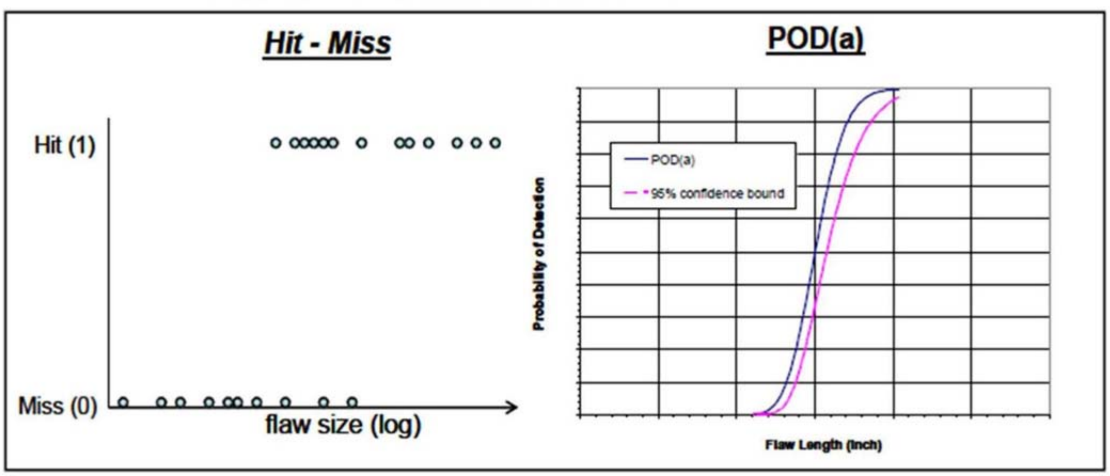


Figure 9.2-11. Computed Radiography POD

– Laboratory hit/miss analysis identified "film equivalent" systems

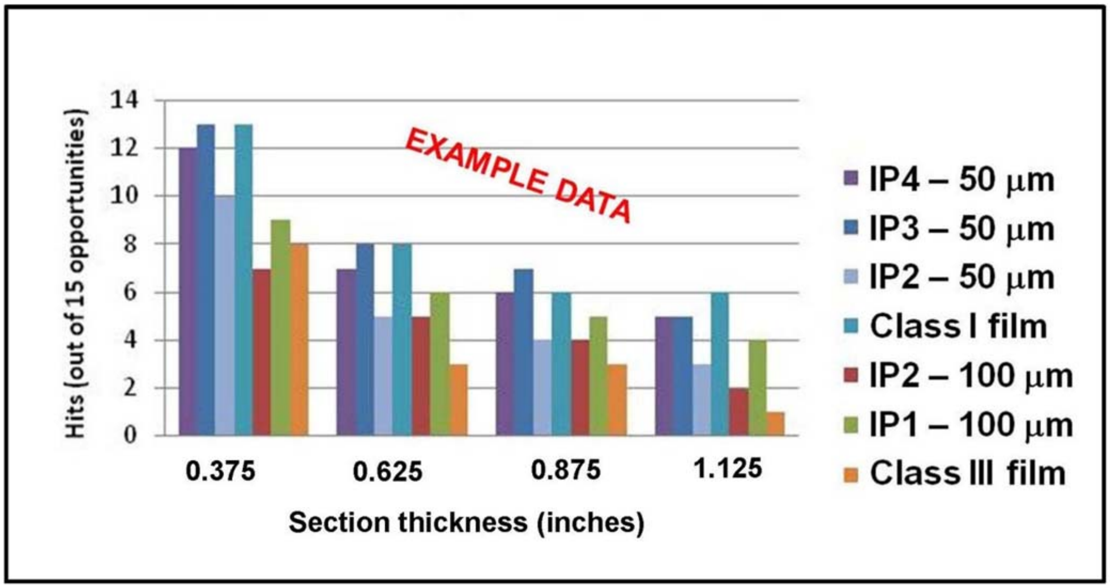


Figure 9.2-12. Hit-Miss Comparison of Crack Specimens

9.3. STRUCTURAL HEALTH MONITORING

9.3.1. CBM+ Viability for a Large Transport Aircraft

Dale Ball, Lockheed Martin Aero – Ft. Worth

The condition-based maintenance plus (CBM+) initiative was established by the US DoD in 2002 in an effort to address the dramatic increases in maintenance costs and reductions in mission capable rates brought about by increased usage rates of aging assets. CBM+ can be roughly defined as a force management philosophy that relies on state (condition) monitoring, along with state informed prognostic capability for the planning of maintenance / sustainment actions for individual assets. To realize effective CBM+ advanced sensing technology, providing both enhanced usage data as well as direct indications of structural health is necessary. This sensing technology is commonly referred to as Structural Health Monitoring (SHM). In 2008 the USAF recognized that in order to realize the significant potential of CBM+ to reduce operating costs and increase aircraft availability, without compromising the operational and flight safety assurances currently provided by ASIP, CBM+ would have to be implemented within the context of ASIP.

This technical effort will describe one strategy for the integration of CBM+ with ASIP (aka CBM+SI). The integration strategy touches each of the five pillars of ASIP and is articulated, primarily via a proposed modification to the ASIP standard, MIL-STD-1530C. The technical effort will also review the process by which a given structural maintenance issue may be evaluated as a candidate for transition to the new maintenance management paradigm. This process includes 1) identification of a candidate structural issue (one with sufficient maintenance / cost burden), 2) selection of an appropriate SHM sensor (or combination of sensors), 3) design and development of an SHM system architecture for new systems (or an integration plan for existing systems), 4) laboratory demonstration on realistic structure, 5) development of the concept of operations for the SHM system and the condition-based maintenance planning that it enables, and finally 6) development of the business case that demonstrates (or disproves) cost effectiveness. This process was developed and matured by working through a specific structural application, the aft crown region of the C-5A aircraft, a description of which will be included in the discussion (Figure 9.3-1).

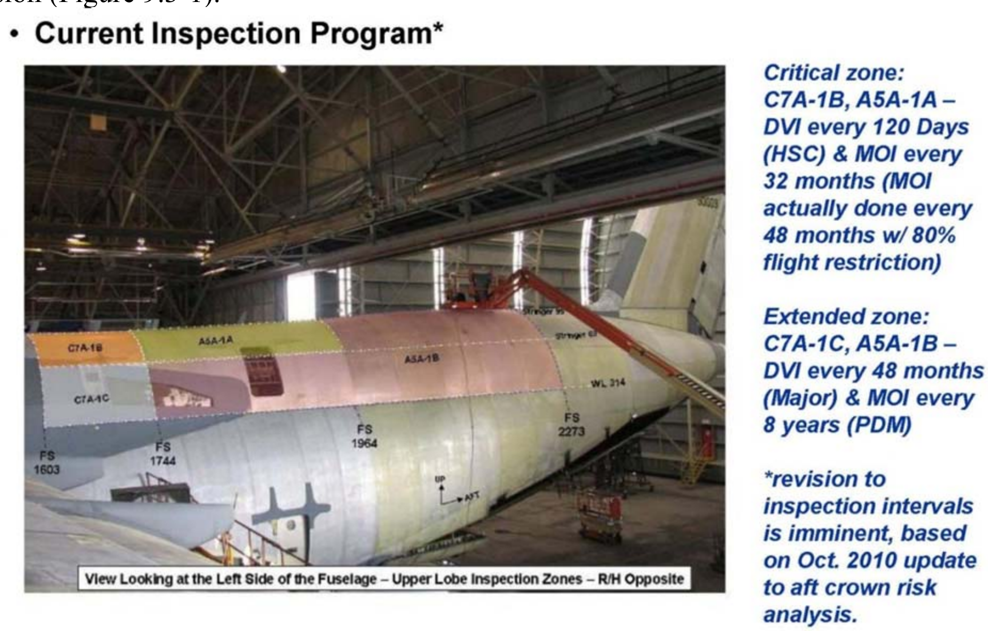


Figure 9.3-1. C-5A Aft Crown Structural Application

9.3.2. Structural Health Monitoring System Certification and Validation of Reliability

Eric Lingren, Gary Steffes, and Charles Buynak, USAF Research laboratory – Materials and Manufacturing Directorate; John Aldrin, Computational Tools; Enrique Medina, Radiance Technologies, Inc.; and Mark Derriso, USAF Research Laboratory – Air Vehicles Directorate

Structural health monitoring (SHM) systems may hold the potential as enablers for condition-based maintenance and prognostic strategies. However, before this potential can be realized, the lack of an accepted path to on-aircraft qualification must be addressed. First, it is necessary for SHM systems to satisfy existing flight certification requirements for on-board systems. Second, the SHM system capability and reliability, in terms of damage detection and characterization, must be quantified. The SHM system capability and reliability assessment includes quantifying false-positive potential and detection sensitivity variance caused by multiple factors including changes in operation environment, material, geometry, and system reliability over their expected useful life (Figure 9.3-2).

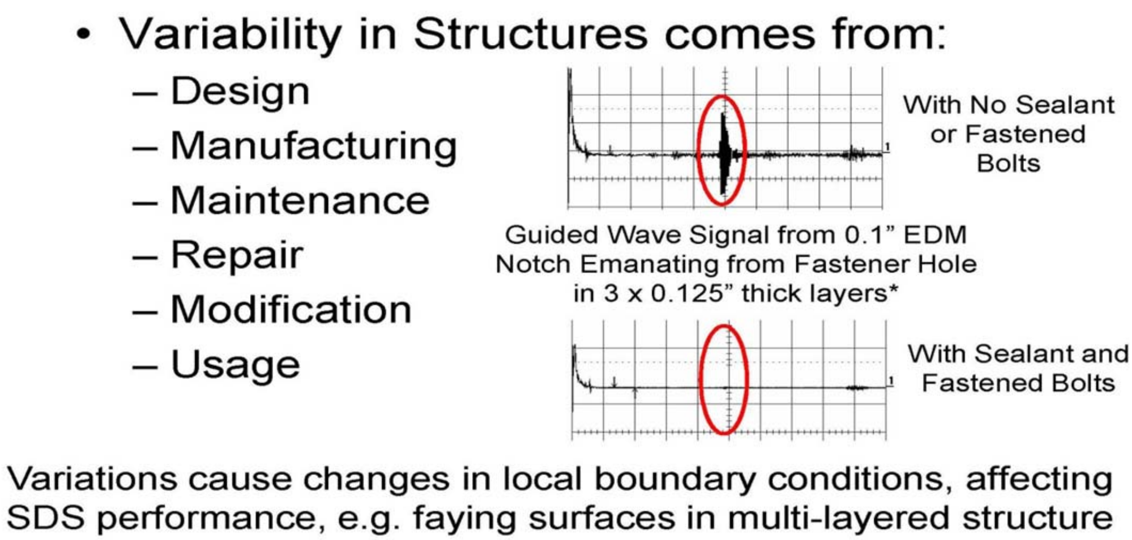


Figure 9.3-2. Detection Sensitivity Variance

This technical effort presents a framework for SHM system qualification, specifically for Aircraft Structural Integrity Program (ASIP) managed structures, building on existing (military and commercial) standards and handbooks for on-board system certification and nondestructive testing validation. The validation methodology incorporates statistical metrics of reliability for SHM systems used for damage detection, localization, and sizing. In addition, it includes model-assisted probabilistic reliability assessment protocols designed for characterizing SHM methods using empirical and simulated data including uncertainty analysis. A multi-scale approach to SHM reliability evaluation is presented that attempts to minimize the number of specimens, the length of time for testing, and the degree of full-scale testing required for obtaining statistically meaningful characterization results. The phased test plan includes coupon testing, laboratory testing on relevant structures and environmental conditions, a system level test, aircraft installation validation, and limited flight testing. The feasibility of applying this approach to typical sensing methods found in SHM systems is explored and additional challenges concerning modeling efforts and uncertainty propagation are addressed.

9.3.3. Hot Spot Monitoring of Aircraft Structures

Mark M. Derriso, USAF Research Laboratory – Air Vehicles Directorate

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. Automated inspection of these areas, 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 traditional non-destructive inspection methods will not work. This activity discusses the development of hot spot monitoring techniques for a metallic lug component (Figure 9.3-3) and a composite wing structure (Figure 9.3-4) using piezoelectric-generated elastic waves.

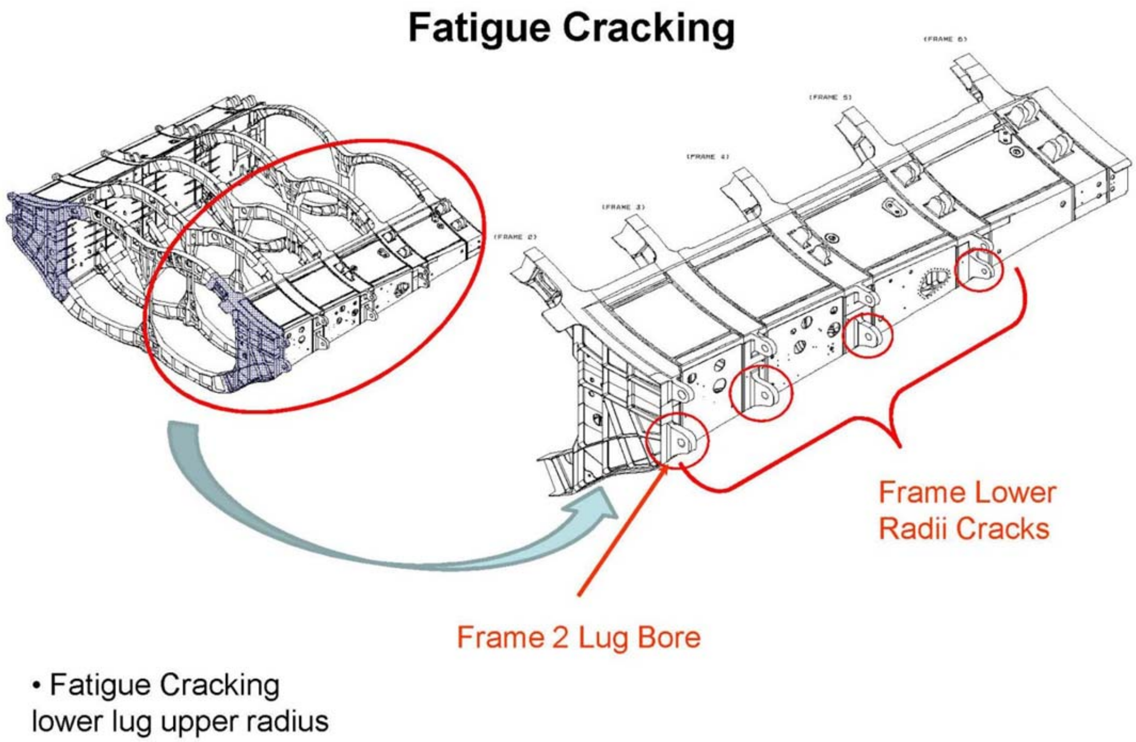


Figure 9.3-3. Metallic Lug Example

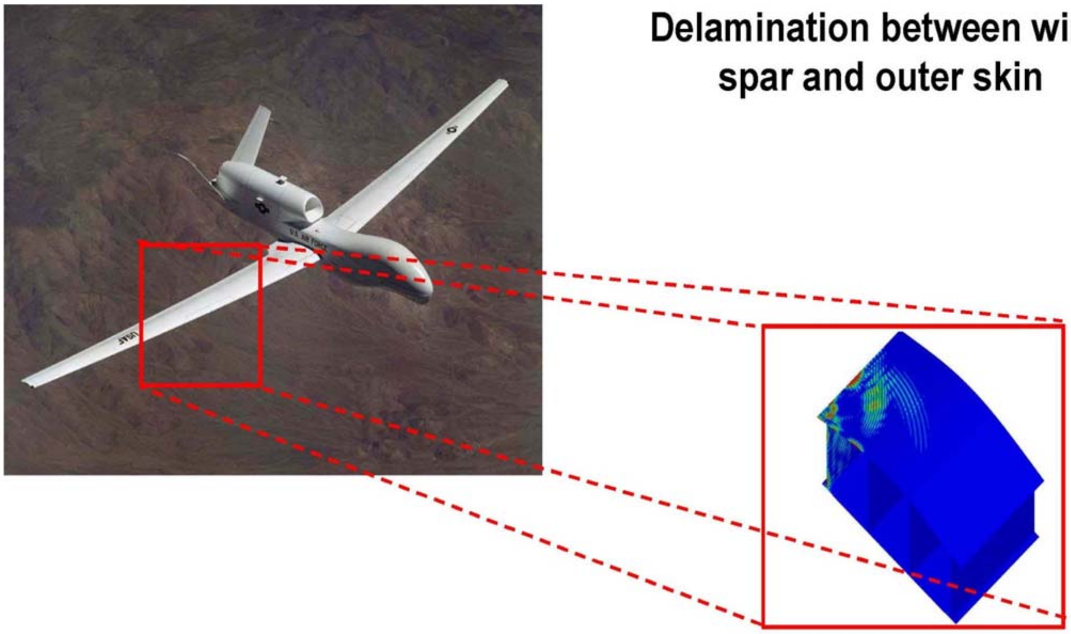


Figure 9.3-4. Composite Wing Box Example

Development of hot spot monitoring for the metallic lug component has followed a multi-step approach progressing from simple coupon tests to the full-scale component. Initial testing performed on titanium dogbone coupons was complicated by issues of sensor system robustness and the reliability of "truth" data, but showed the potential to detect thru-cracks of less than 0.10 inch. Subsequent testing, performed using titanium cantilever beam specimens, utilized packaged piezoelectric sensors to improve sensor robustness and fluorescent dye penetrant to improve the reliability of visual "truth" data. This testing showed promise, but additional testing is needed to further refine the technique as only limited data were available from the cantilever beam testing. Recent experiments include fatigue testing of lug subcomponents with a geometry and material properties very similar to the full-scale component. Five subcomponent tests have been performed with waveform data collected over a range of frequencies and visual crack lengths recorded after specific numbers of cycles. Preliminary work demonstrates that damage indices can be mapped to crack length for edge sensors. Further work is required to combine the readings of all the piezoelectric sensors into a single crack length estimate. Building on the results from all of the earlier testing, SHM system development is underway for a full-scale lug component to be fatigue tested under spectrum loading.

For the composite wing structure, initial studies have been performed on a short section of the wing which is fabricated from graphite/epoxy with the upper and lower skins bonded to spars. Preliminary modeling has been performed to understand wave propagation in the complex structure and experimental investigations have been performed to detect laser-induced disbond damage. Currently, a larger section of the composite wind is undergoing fully reversed, cyclic three-point bend loading. Laser-induced disbonds at various locations on the spar/skin bonds are expected to grow. Half of the upper and lower surfaces of the wing are being monitored during the cyclic loading using an ultrasonic tomography approach. Conventional non-destructive inspection methods have been used to provide initial confirmation of the disbonds and periodically to detect any growth in the disbond regions. Recent results from the tomographic imaging performed during testing, and a comparison with non-destructive inspection results, are discussed.

9.3.4. Investigation into the Applicability and Capability of NDI/SHM Techniques on Large Transport Category Fuselage Structure

Melinda Laubach, Anthony Alford, and Larry Braden, Wichita State University – NIAR; and David Forsyth, TRI/Austin

Numerous Nondestructive Inspection (NDI) and Structural Health Monitoring (SHM) techniques are actively marketed today. Many of the NDI techniques claim to be able to detect small damage while performing inspection on a large region of aircraft structure quickly. Several SHM techniques claim to be able to identify small damage and detect damage progression, while monitoring a large area.

In an effort to assess these claims on actual large transport category fuselage structure, personnel from the National Institute for Aviation Research tested four fuselage crown skin panels from retired aircraft with three different induced or naturally occurring damage scenarios subjected to uniaxial loading conditions. Since the test articles were acquired from retired military transport aircraft, the pre-test damage condition was determined with advanced NDI methods. A variety of traditional and emerging NDI techniques (Figures 9.3-5 and 9.3-6) were used to inspect the skin planes prior to test. An extensive structural teardown was performed after component failure, including fractographic analysis, to characterize all damage present in the test articles.

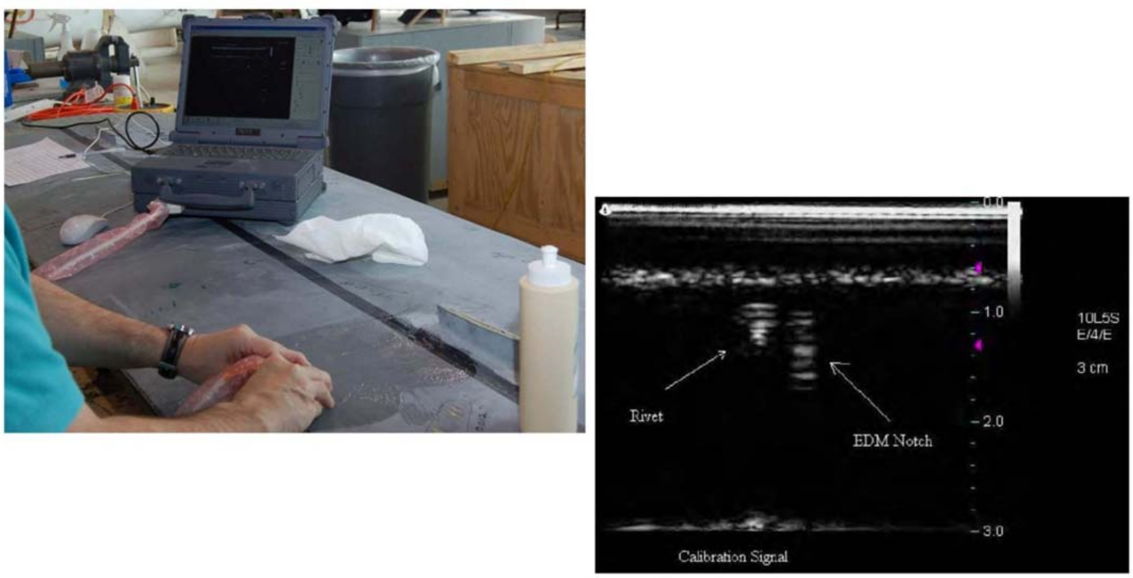


Figure 9.3-5. Examples of NDI – Array UT

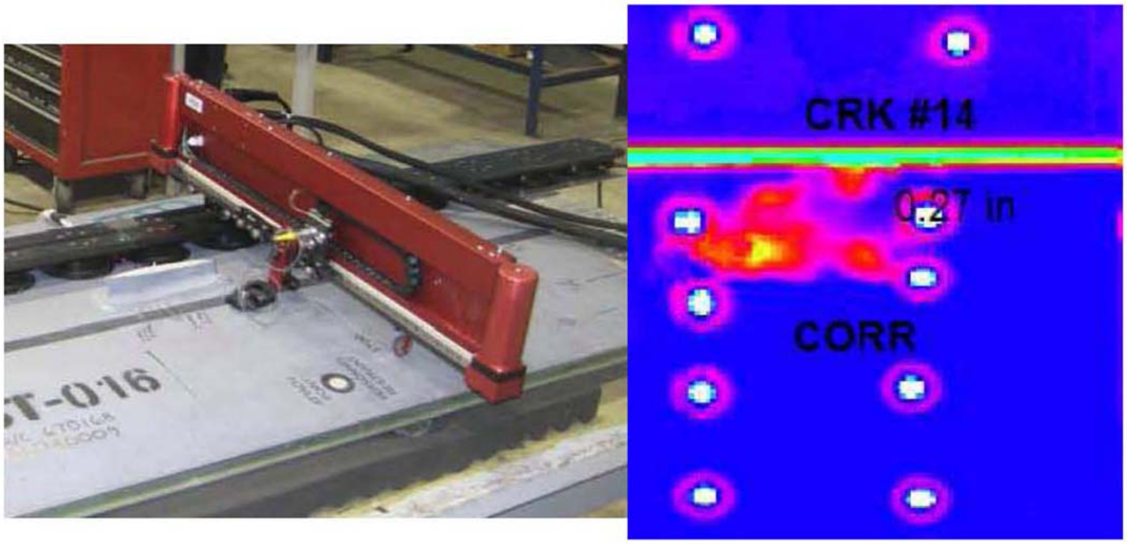


Figure 9.3-6. Examples of NDI – Semi-Automated Scanned ET

During the structural testing of the fuselage panels, fatigue cycles were applied to nucleate and propagate fatigue cracks from artificially induced or naturally occurring damage. Two SHM technologies (Figures 9.3-7 and 9.3-8) were installed on one of the panels tested to detect the onset of fatigue damage and track its progression. The results of the NDI/SHM technologies are discussed and compared with teardown findings.

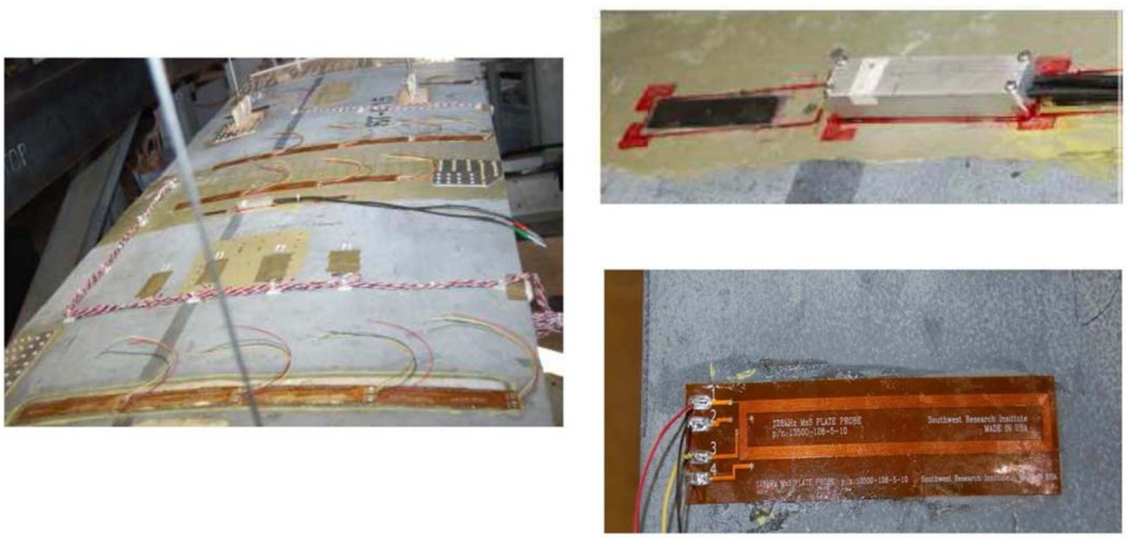


Figure 9.3-7. SHM Installations – Guided Wave UT

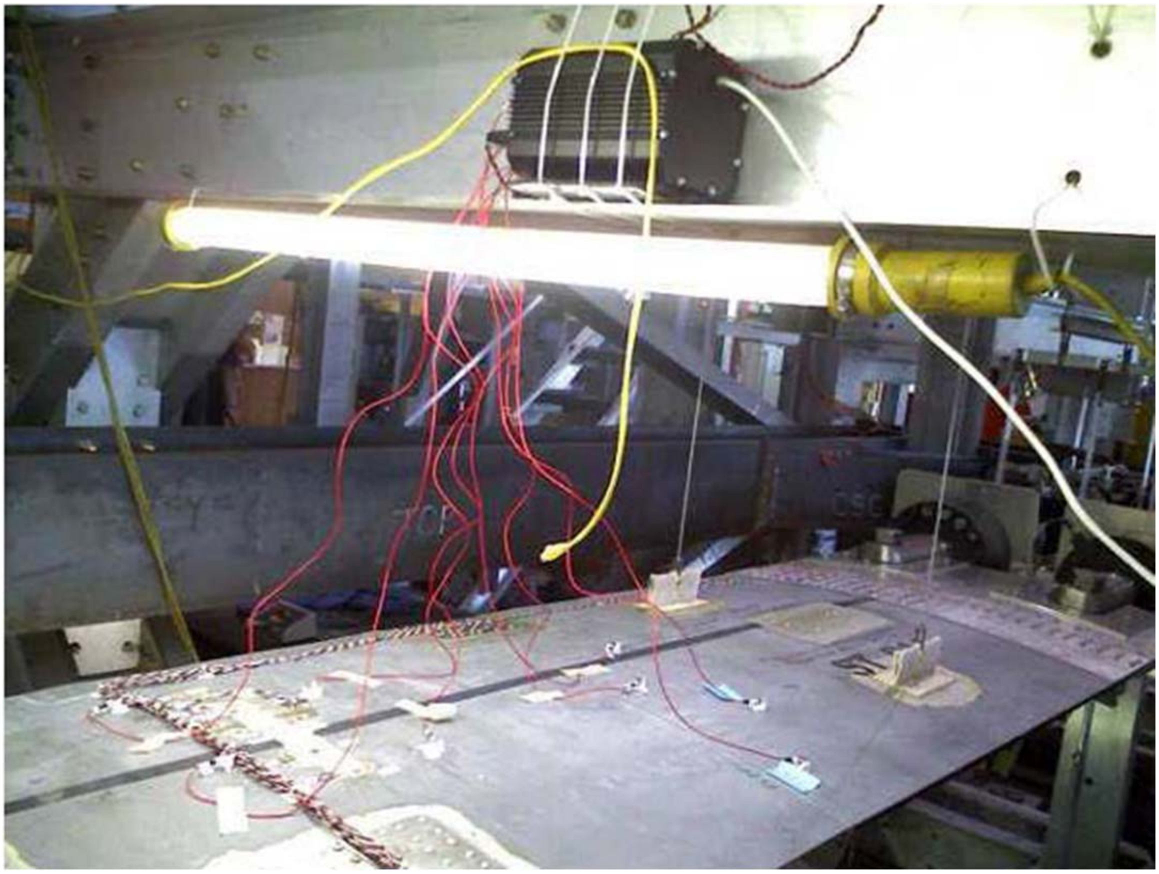


Figure 9.3-8. SHM Installations – Acoustic Emission

9.3.5. Structural Health Monitoring R&D Roadmap

John Bakuckas, FAA – William J. Hughes Technical Center

Efforts are underway to develop self-sufficient Structural Health Monitoring (SHM) systems using networks of integrated sensors for the continuous monitoring, inspection, and damage detection in aircraft structures to improve safety and reduce labor cost and human error. While ad hoc efforts to introduce SHM into routine aircraft maintenance practices are valuable in leading the way for more widespread SHM use, there is a significant need for an overarching plan that will guide near- and long-term activities and will uniformly and comprehensively support the evolution and adoption of SHM practices. The Federal Aviation Administration (FAA) is addressing these issues through the development of an SHM Research and Development (R&D) Roadmap. This plan will contain input from aircraft manufacturers, regulators, operators, and research organizations so that the full spectrum of issues, ranging from design to deployment, performance and certification is appropriately considered. It will be used to assess what regulatory guidance is needed to assure the safe incorporation of SHM through formal certification programs. The roadmap will be used to guide future FAA research in SHM.

A number of activities have been carried out to support the SHM R&D Roadmap including: 1) producing an SHM Technology Readiness database, 2) implementing an SHM Industry Survey, 3) constructing an SHM Sensor Database, and 4) completing a formal review of pertinent FAA, industry, and military documents to identify precedents and to better direct the FAA's response to SHM issues. An important element in developing the FAA SHM R&D Roadmap is a clear understanding of the current status of SHM technology and the pending regulatory issues facing the aviation industry to safely adopt SHM practices. To acquire such information, a comprehensive survey was implemented with the aviation industry to determine the technology maturation level of SHM, identify integration issues and prioritize the research and development needs associated with implementing SHM on aircraft. This survey was sent to persons involved in the operation, maintenance, inspection, design, construction, life extension, and regulation of aircraft. Over 450 people responded to the survey to provide industry information on SHM deployment and utilization, validation and certification, SHM standardization, sensor evolution and operation, cost-benefit analysis, and SHM system description. Overall, it was determined that there is a strong interest in SHM. Over 200 applications, covering all aircraft structural, engine, and systems areas, were identified. Industry's main concerns with implementing SHM on aircraft are achieving a positive cost-benefit and the time required to obtain approval for SHM usage. OEMs and airlines felt that research and development efforts should be focused on: global systems, sensor technology, system validation and integration, and regulatory guidance. In addition, they felt that standardization and guidelines are needed in validation, certification, and sensor design with aviation in mind. A Technology Readiness Database was also assembled using a compilation of pertinent information retracted from SHM and NDI conference proceedings, technical journal articles and industry information. The SHM Technology Readiness Database was compiled from over 3,000 papers from key SHM conferences and journals. It includes a listing of SHM sensor and sensor systems with their maturation ratings based on Technology Readiness Levels (TRL) widely used by military, NASA, and government agencies. Figure 9.3-9 shows the evolution of SHM Technology Readiness from TRL 2-3 (initial hardware configuration) in 2006 to TRL 5-6 (system prototype testing) in 2009. This helps establish an SHM technology advancement rate to predict that SHM will reach TRL 7-9 (certification and operation) in 3 to 5 years.

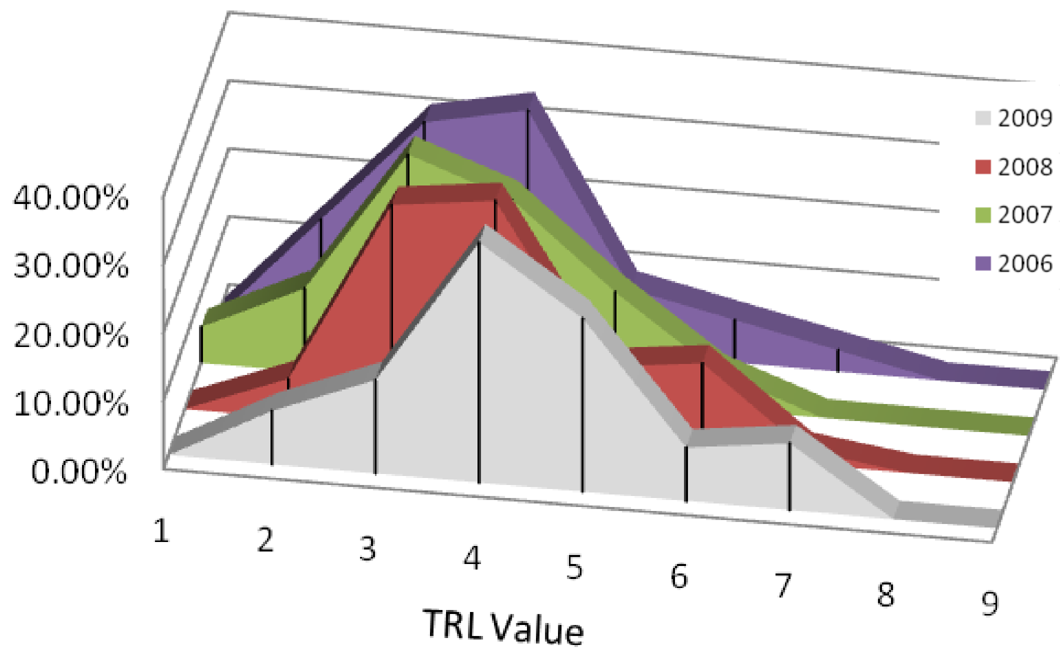


Figure 9.3-9. Evolution of Technology Readiness Levels Depicts the Rate of Progress in SHM

These efforts have provided a clear understanding of the current status of SHM technology and the pending regulatory issues facing the aviation industry to safely adopt SHM practices. These data, along with foundation information already gathered from the FAA and industry, is being used to produce the FAA SHM R&D Roadmap.

Points of Contact:

- Dennis Roach, Sandia National Laboratories – AANC, 505-844-6078
- Paul Swindell, FAA William J. Hughes Technical Center, 609-485-8973
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9.4. STRUCTURAL TEARDOWN ASSESSMENTS

9.4.1. F-15 Structural Disassembly and Analysis Support Project

Amanda Alpaugh, USAF-WR-ALC

The F-15 C/D airframe's design life goal was 8,000 flight hours. With average flight hours for the C/D model fleet of 7,100 hours and Congressional direction to extend the service life of the fleet to the year 2025, the WR-ALC/GRM (formerly the 830th Aircraft Sustainment Group) is now in the process of validating the sustainability of the C/D model airframe for an additional 15 years. In order to extend the aircraft service life, the F-15 System Program Office (SPO) is conducting structural teardowns on two F-15 C/D fuselages and six wings as well as a Full Scale Fatigue Test. The results will provide data points for determining the effects of current usage on the F-15 airframe. This analysis will be used to reevaluate the Force Structural Maintenance Plan (FSMP) and support continued structural assessment, sustainment, and mission readiness of the fleet.

In February of 2009, the F-15 D model fuselage was delivered to S & K Technology's teardown facility in Byron, GA. Over the next 15 months the aircraft was disassembled (Figure 9.4-1) and select parts underwent visual and non-destructive inspections (NDI). During NDI, 30,814 fastener holes were inspected by Bolt Hole Eddy Current and all surfaces of critical parts were inspected by Fluorescent Penetrant Inspection. A number of material defect or damage indications were identified and have been assessed to determine their structural significance (Figure 9.4-2). Failure analysis has been conducted by the Israel Air Force on a small subset of the crack like indications and the F-15 SPO plans to continue analyzing the remaining indications (Figure 9.4-3).

Figure 9.4-1. Documentation of Structural Disassembly

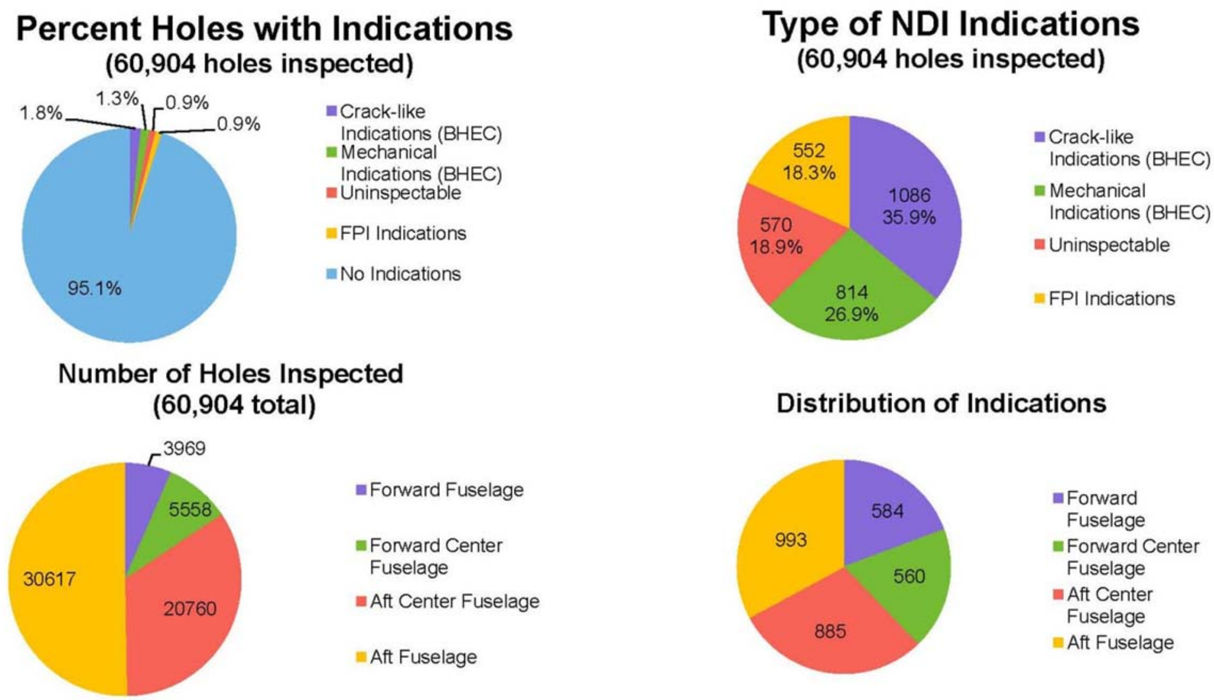

Figure 9.4-2. F-15 Aircraft NDI Results

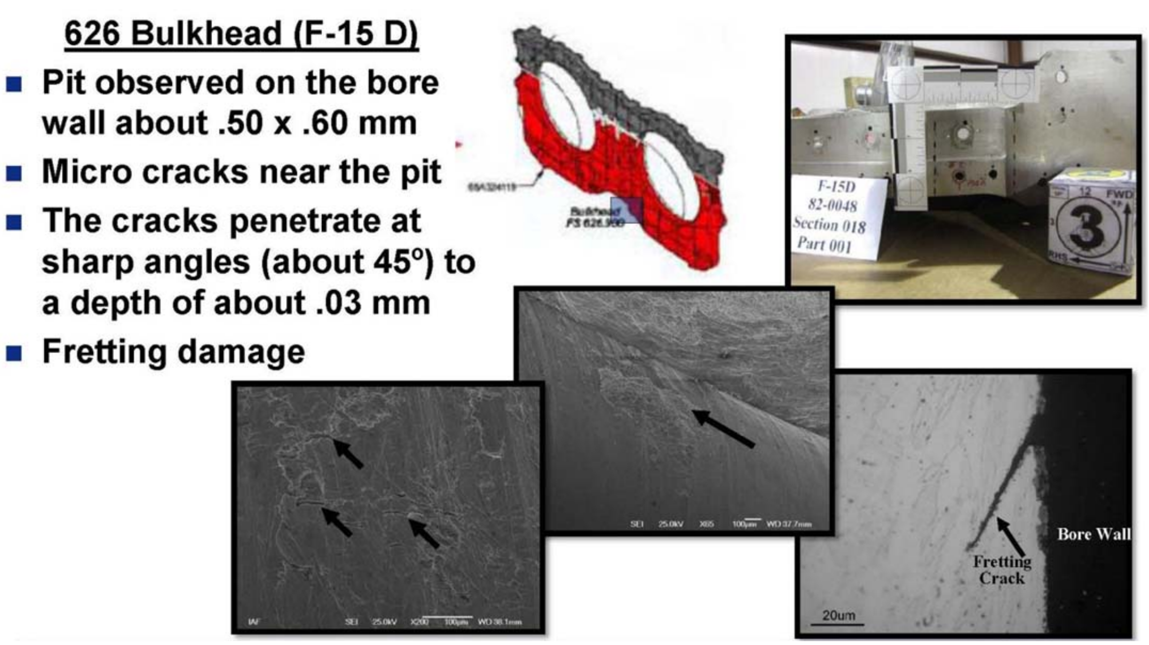

Figure 9.4-3. Fractography of F-15D 626 Bulkhead

This activity provides an overview of the methodologies and protocols used during the planning, disassembly, and inspection phases as well as a brief discussion of the F-15 Teardown Data Management System (TDMS) that is used to organize the data collected from teardown and allow collaboration between stakeholders. Along with the overview, the activity will include a detailed discussion of the results of the F-15 D teardown as well as the status of the F-15 C wing teardowns. The detailed discussion will also cover how the results of the teardown will be used to conduct further damage tolerance and structural integrity analysis which will help reassess the airframe's service life and support decisions regarding the F 15 fleet.

9.4.2. Teardown Projects

Cindy Klahn and Jesse Vickers, USAF Academy - CAStLE

B-1B Lancer. The Center for Aircraft Structural Life Extension at the United States Air Force Academy is currently developing a plan for a destructive-teardown-analysis program for the B-1B Aircraft. The development of the B-1B started with very uncertain beginnings in 1964. Development was halted and re-started twice before being fielded in 1986. While it was the first USAF aircraft requiring fracture-mechanics-design methodology, the interrupted development process did not include a full-scale fatigue test of the entire aircraft. Instead, only the major components of the B-1A version were fatigue tested. In addition, some of the structure redesigned after the component tests was not retested. As the USAF continues to fly the B-1B aircraft well past its test-demonstrated service life, there has arisen a need to conduct a durability testing program. This program consists of a full-scale fatigue test followed by a comprehensive teardown program. The results of this program will provide invaluable information for the long-term sustainment of the B-1fleet.

The full-scale fatigue test has just been awarded to validate the planned service life to 2040 and will consist of two separate tests: a fuselage test and a wing test. In parallel with the fatigue test, CAStLE is developing the plan for the subsequent teardown of the test article. The goal of this plan is to quantify and characterize potential structural problems before they occur in the fleet. It will also evaluate the durability of current repairs in the fleet. The teardown program will capture all of the findings in a database that will be used throughout the lifecycle of the B-1B.

KC-135 Stratotanker. The Center for Aircraft Structural Life Extension at the United States Air Force Academy, at the request of the KC-135 System Program Office, is managing a destructive-teardown-analysis program of three KC-135 aircraft. Teardown of the first aircraft, a KC-135R model, was completed in early 2011. The second aircraft, a KC-135E, began its teardown at the end of 2010. The third aircraft, another KC-135E, will be torn down in the future. The primary objectives of this teardown program are: to meet the ASIP requirements of MIL-STD-1530C, to determine the condition of the C/KC-135 fleet in order to assess its viability for continued operations, and to make fleet management recommendations. A very robust and thorough selection process was utilized to select the teardown aircraft and the areas of the aircraft to be examined. This resulted in nearly 400 teardown sections being defined, making this program one of the largest, most in-depth teardown programs ever conducted by the United States Air Force.

9.5. LOADS & ENVIRONMENT CHARACTERIZATION

9.5.1. The Effect of Age Dependent Spectra on Service Life Analyses

Travis Hawks, Joksan Holguin, Tim Jeske, and Matthew Edghill, Lockheed Martin Aero – Ft. Worth

The F-16 program has had aircraft in flight for over 30 years, collecting flight data for much of that time. During that period, the tasks, missions, and environments that the aircraft have flown varied significantly. To maintain the structural integrity of the airframe, the baseline loads spectra are ideally updated about every 5 years. In the earlier years of the F-16 program, the severity of the baseline usage had either increased or not decreased significantly. Operations in the past several years, however, have resulted in a reduced severity of baseline usages (Figure 9.5-1 and Table 9.5-1).

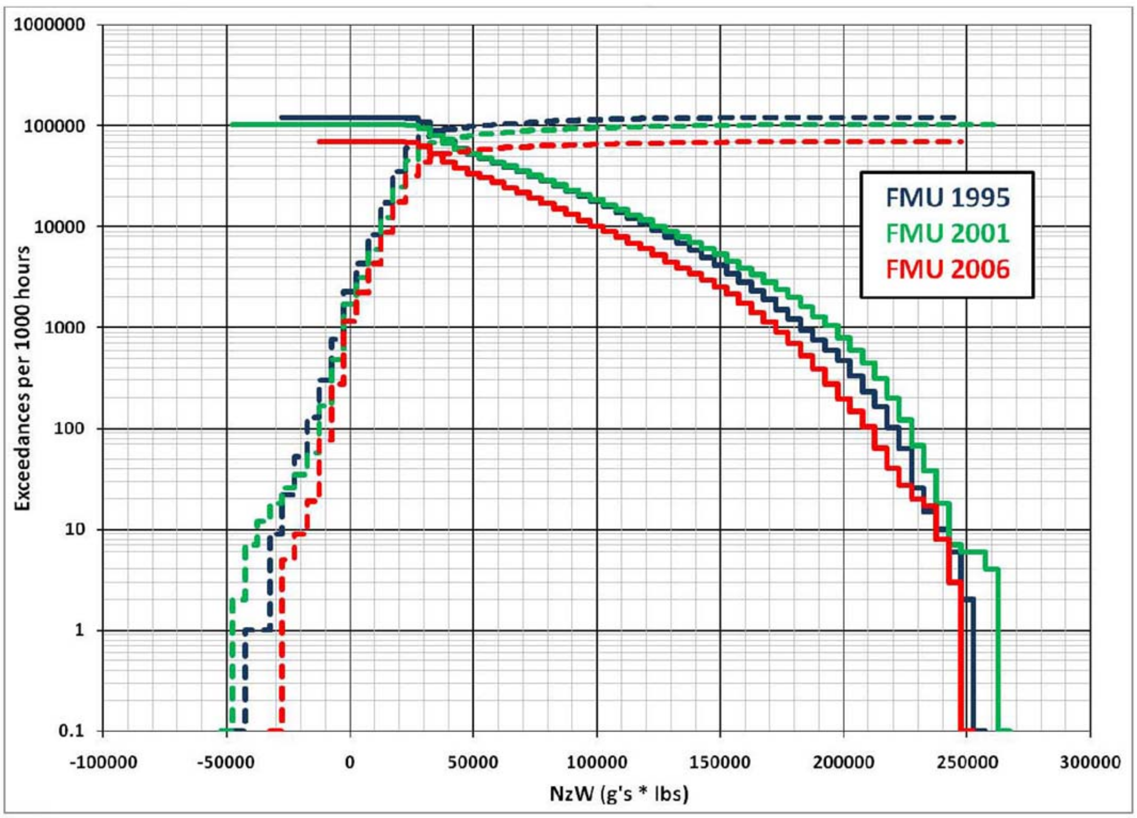

Figure 9.5-1. USAF Baseline Spectra

Table 9.5-1. Spectra for USAF Block 40 Aircraft

Spectrum	Average Flight Hours When Applicable
Design	-
FMU 1995	0 – 1,441
FMU 2001	1,441 – 3,294
FMU 2006	3,294 - ∞

The reduction in baseline usage severity has a significant impact when applied in the typical manner to DADT, FSMP, and IAT analyses. The accepted method for performing these tasks is to apply the most recent baseline usage from the time of aircraft delivery. When the baseline usages were increasing in severity, this could result in overly conservative service life predictions and lead to excessive costs and aircraft downtime. Now that usage severities appear to be on the decline, the analytical results could potentially lead to operating the F-16 fleet at a higher level of risk.

A study was initiated to examine the effect of the current procedure of using the latest baseline usage for all analyses. The usage history for USAF Block 40/42 F-16 aircraft was reviewed to determine the average flight hour span from which each historical baseline usage was gathered. A technique to use a composite baseline spectrum was devised. This composite spectrum was used to analyze several control points (Figure 9.5-2) on the airframe and the results were compared to those found with stand-alone baseline usages (Figure 9.5-3). This study evaluated if using a composite spectrum technique for a long-life and multi-usage aircraft is an appropriate tool for determining service life and associated maintenance intervals. The outcome of this study could potentially change the fleet management policies of the worldwide F-16 program.

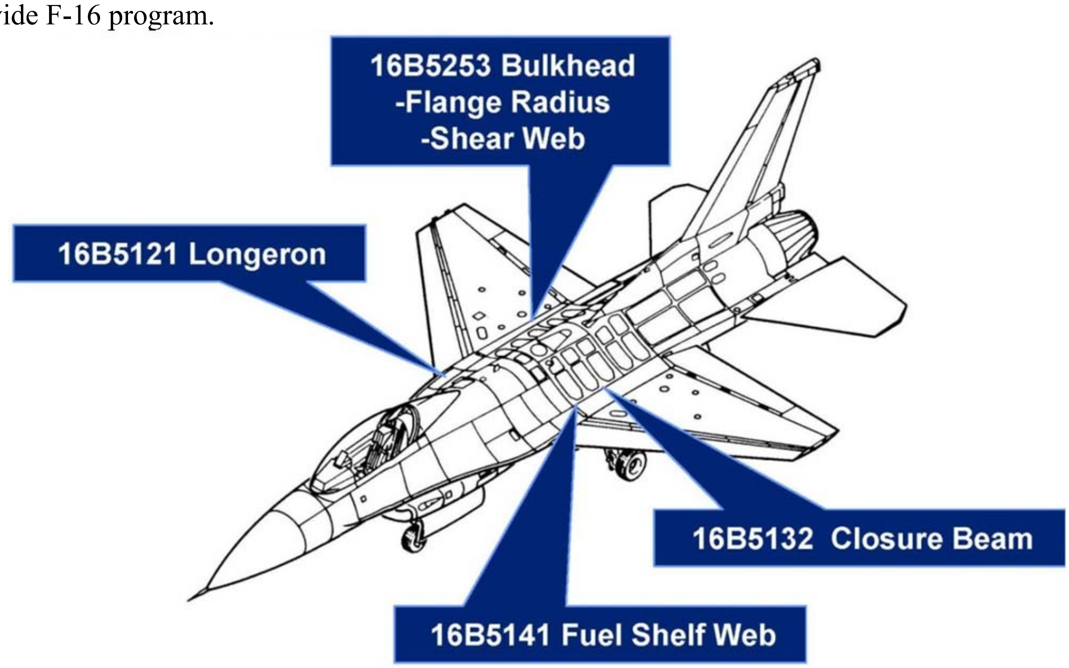

Figure 9.5-2. Selected Control Points

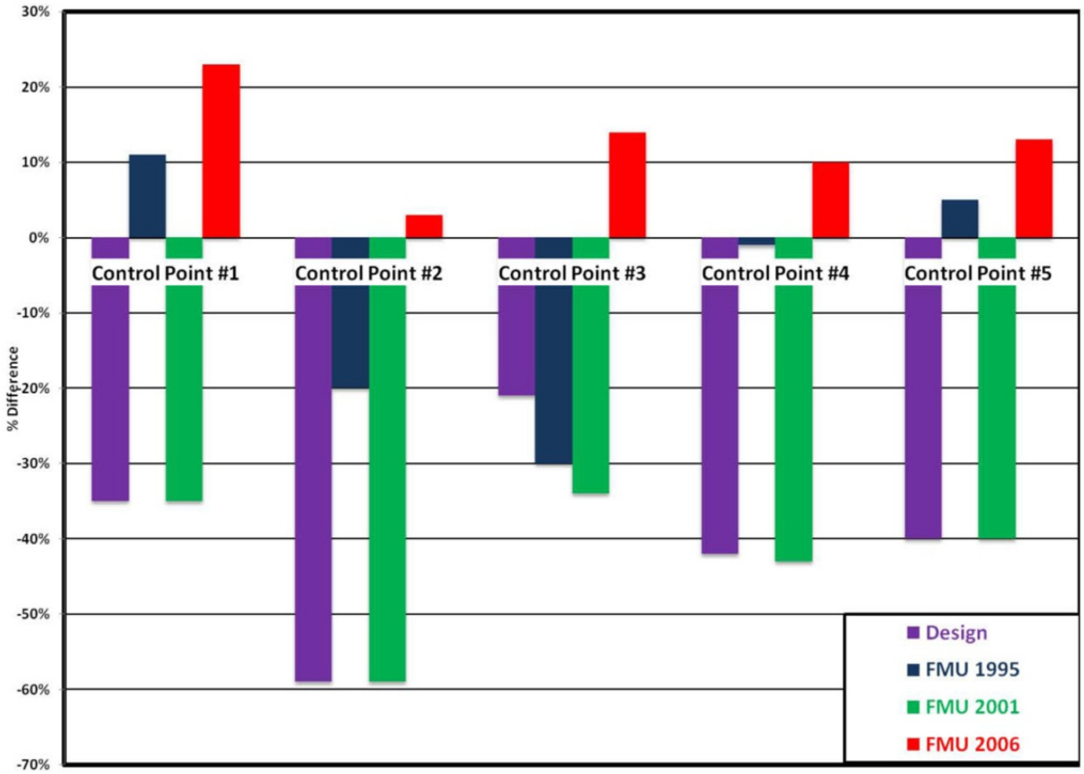

Figure 9.5-3. Percent Difference in Life for Individual Spectra vs. Composite Spectrum

9.5.2. T-38 ASIP: L/ESS is More

Michael P. Blinn and Jacob McReaken, USAF-OO-ALC

The T-38 "Talon" is an aging aircraft, now nearing 50 years of successful service with the USAF. As a legacy weapon system, fielded prior to the implementation of damage tolerance, the T-38 provides a unique challenge to the ASIP manager. In particular, many of the "standard" approaches to structural sustainment - such as incorporating flight loads data collection systems into the design of the structure - were not fully in place at the time the USAF took first delivery of the T-38 in March 1961. In summary, the T-38 ASIP has (historically) found itself in the position of catching-up with the latest methods and technologies that are more-or-less "standard" with the newer weapon systems.

In the late-1990s, the T-38 underwent a metamorphosis from an analog to a digital weapon system, as the avionics were enhanced to meet the needs of modern jet fighter pilot training to 2020. This avionics upgrade had the secondary benefit of providing a vehicle for the T-38 ASIP to collect flight loads data on a fleet-wide basis, through a relatively straight-forward software change, without the associated (and expensive) hardware change (Figure 9.5-4). Prior to the new avionics, flight loads data were collected on a biennial basis, with less than 5% of the fleet instrumented with data recorders for Loads/Environment Spectra Survey (L/ESS) and/or Individual Aircraft Tracking (IAT) needs. With the new avionics, the T-38 ASIP was now positioned to work towards meeting the needs of both the L/ESS and IAT for fleet management and safety of flight.

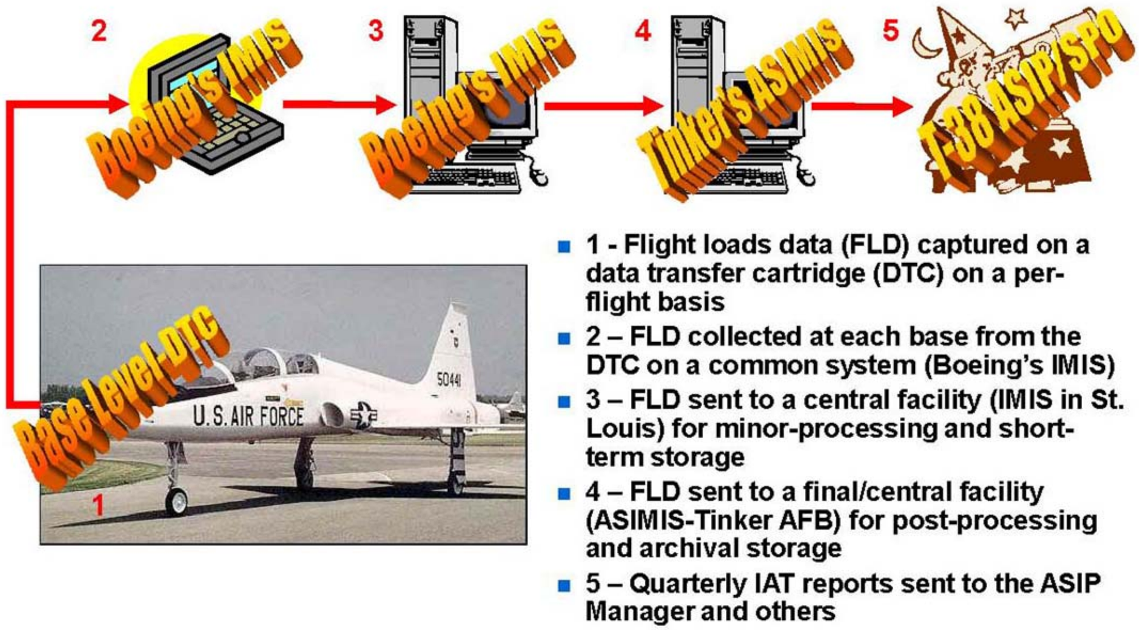

Figure 9.5-4. System Design T-38C L/ESS and IAT

This technical effort presents the results of a multi-year, multi-organization effort to field a meaningful and viable L/ESS and IAT program for the T-38 weapon system. Although the initial premise of a "simple" software change was the driver for this effort, numerous challenges were faced by the T-38 ASIP and the associated team to provide for a validated L/ESS and IAT program. In addition, the T-38 ASIP also faces the challenge of incorporating those legacy aircraft, without the digital avionics, into the latest T-38 L/ESS and IAT programs.

9.5.3. Service Loads Handbook Development

John Rustenburg, Dan Tipps, Don Skinn, and Todd Jones, University of Dayton Research Institute; and William Buckey and Hsing C. Yeh, USAF ASC

The Department of Defense Joint Service Specification Guide for Aircraft Structures (JSSG-2006) establishes the structural performance and verification requirements for an airframe. The Specification Guide includes references to an as yet to be developed ASC-Technical Report (ASC-TR-xxxx) that contains repeated load sources by aircraft type, mission type and mission segment utilizing the best historical data available. The University of Dayton Research Institute (UDRI) was tasked with the development of a service loads handbook to meet the requirements of the referenced technical report.

This technical effort presents the development of a multi-volume handbook of aircraft usage and service load statistics for use in the derivation of repeated load spectra to meet the requirements of the Joint Service Specification Guide. The technical effort describes the flight and ground phase definitions as well as data reduction procedures applied to recorded parameter data to derive aircraft usage, flight and ground service load statistics, and movable structures operational data. The technical effort shows how the data reduction procedures and data analysis methods developed at the University of Dayton Research Institute are being used successfully to develop a service loads document with the best historical data available. Examples of the peak counting technique and maneuver/gust separation technique employed are presented. Samples of various data formats used to describe the aircraft usage and operational load statistics for a refueling aircraft and a cargo aircraft are shown.

The handbook will consolidate aircraft usage and operational statistics for a variety of in-service aircraft in a single document and will serve as a practical guide towards establishing rational repeated loads criteria for military aircraft structure. The first volume of the handbook contains descriptions of the criteria and data editing methodology used in the derivation of operational usage and service loads statistics from measured data. The succeeding volumes will present load source data by aircraft category, such as refueling aircraft, cargo aircraft, bomber aircraft, fighter aircraft, attack aircraft, trainer aircraft, and special aircraft (Table 9.5-2).

Table 9.5- 2. Candidate Aircraft Breakdown by Category/Volume

AIRCRAFT CATEGORY	USAF AIRCRAFT	NAVY AIRCRAFT	HANDBOOK VOLUME
REFUELING	**KC-135**	**KC-130**	II
CARGO	C-5, **C-17**, C-130	C-2A	III
BOMBER	B-52, B-1B, B-2		IV
FIGHTER	F-4, F-15, F-16, F-22	F-14, F-18	V
ATTACK	A-10, AC-130, F-111, A-7	AV-8B, EA-6B	VI
TRAINER	T-1, T-6, T-37, T-38	T-2, T-34C, T-44A, T-45	VII
SPECIAL	TR-1		VIII

Note: Highlighted aircraft represent aircraft data processed to date.

9.5.4. F-35 Structural Prognostics and Health Management

Michael R. Woodward, J. C. McConnell and Robert J. Burt, Lockheed Martin Aero – Ft. Worth

The complexity of the F-35 program in terms of fleet size, basing scenarios, number of users, Fleet Management philosophies and the level of program concurrency presents a unique challenge in the development and implementation of a Fleet Management Program (Figures 9.5-5 and 9.5-6). The F-35 Structural Prognostics and Health Management Program (SPHM) draws heavily on legacy efforts in terms of technical approach, but due to program scope and computational improvements both on-board and off-board, incorporates differences in infrastructure, data management and data processing (Figure 9.5-7).

Figure 9.5-5. F-35 SPHM Effort Supports 3 Aircraft Types

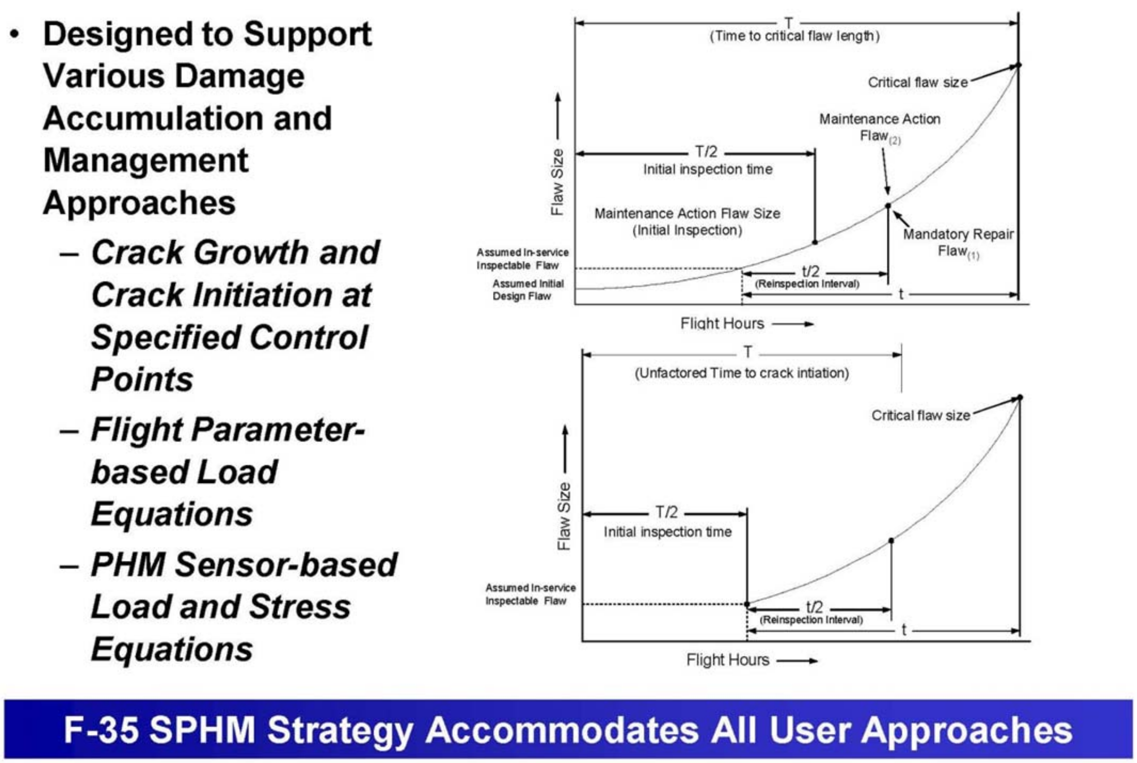

Figure 9.5-6. F-35 Structural Health Management Strategies

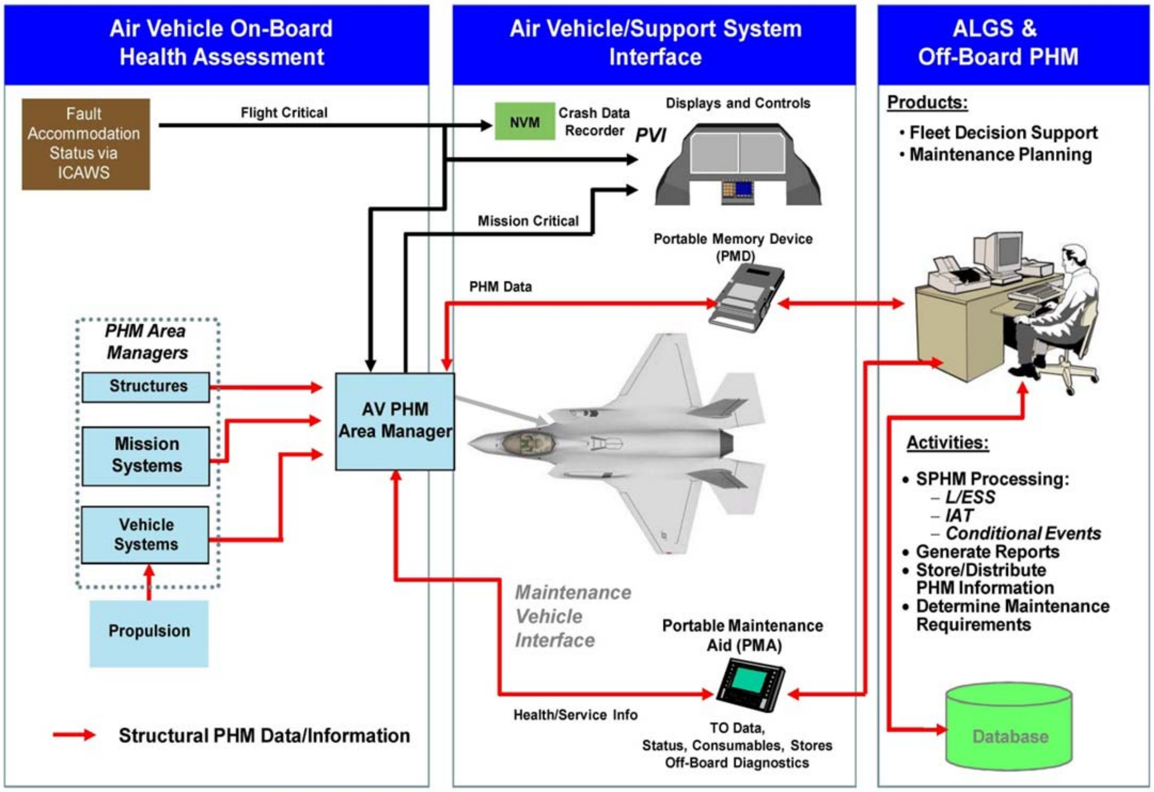

Figure 9.5-7. F-35 PHM Architecture

The F-35 SPHM effort consists of the following elements: Individual Aircraft Tracking (IAT), Loads/Environment Spectra Survey (L/ESS) and Conditional Event (CER/CEA/CEM). The IAT approach incorporates the use of both SPHM gages for direct measurement of key aircraft strains and subsequent calculation of damage and parametric algorithms for the calculation of damage at detailed control points. The L/ESS approach incorporates the use of the F-35 Design Loads Database for a direct calculation of L/ESS loads rather than parametric algorithms. The Conditional Event approach consists of the on-board detection and reporting of overload events (CER), and the off-board analysis (verification, refinement and mapping) (CEA) of the overload events to the appropriate maintenance tasks (CEM). The Design Loads Database is used for both on-board and off-board load calculations.

Due to the nature of the deployment of the F-35 Fleet, data processing will be performed in a distributed manner. Both the raw and processed data will be periodically sent to a central location for subject matter expert assessment and review. Although real time individual aircraft health data will be available at the unit level, periodic IAT / L/ESS reports will be generated containing an assessment of the data and recommendations to the Fleet Manager. As with legacy programs, periodic baseline usage and life updates will be performed.

The F-35 SPHM development effort is underway with an initial deployment in late 2010/early 2011 and a final SDD update incorporating Full-scale Ground and Flight Test findings in 2013-2014.

9.5.5. Methodologies for Sonic and High Cycle Fatigue Life Tracking on the F-22

Craig D. Hampson, William D. Anderson, and James J. Wentz, Lockheed Martin Aero – Ft. Worth

The F-22 usage definition for establishing baseline design and certification criteria for high cycle and sonic fatigue assessment of the air vehicle structure and subsystems is discussed. The usage is derived from two sets of mission profiles composed of eleven peace-time mission profiles and 3 combat mission profiles (Figure 9.5-8). These profiles were established early in the program and are the foundation of the derived vibration and acoustics design and certification environments defined in the F-22 Environmental Criteria and Acoustic Loads for Sonic Fatigue Design Documents. These data were used to establish margins on aircraft structure and in damage rate equations to determine control point damage rates in established segments of the flight envelope. The F-22 fleet usage is currently being tracked using the Integrity Data Analysis and Reporting System (IDARS) and data from this system is currently updated quarterly. The combination of the design levels and individual aircraft usage is fundamental in determining the useful life of the aircraft and setting inspection intervals. This technical effort presents the methodology used in the development of life equations for structure that has been analyzed to certification levels and times. The details of how the individual aircraft usage in various flight regimes can impact the life of the aircraft will be discussed as well as the steps used to accumulate damage in each segment, the method for filling missing usage data, and the process for extrapolating the data to determine the life of the aircraft. The use of probabilities of failure in a risk-based assessment is also discussed as well as the effect of uncertainties in the underlying data. Also discussed is a database and process that has been established that allows for rapid update of these data for each IDARS data update. Individual aircraft tracking improves the reliability of predicting sonic and high cycle fatigue life based on aircraft usage. The life management process includes updates of sonic and high cycle fatigue margins, damage rates, and probability of failure and associated risk of all F-22 structure subject to sonic and/or high cycle fatigue, all based on individual aircraft and fleet operational usage. This will provide an effective and reliable way to track all aircraft structure subjected to sonic or high cycle fatigue and a basis for making an informed decision on the potential need for inspections and or repairs as the aircraft age.

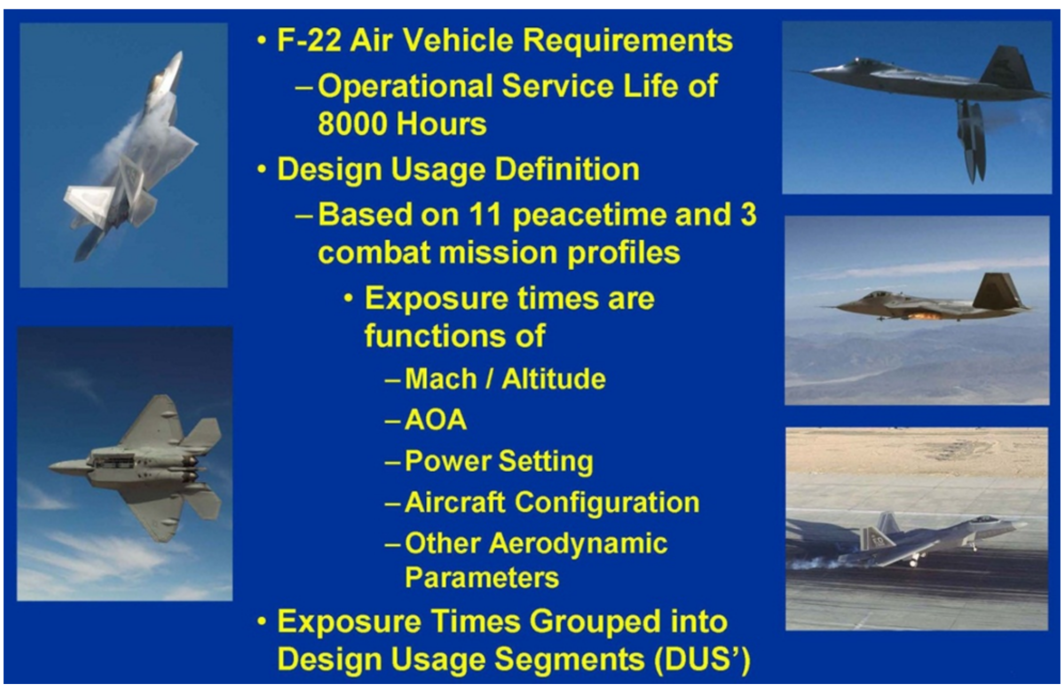

Figure 9.5-8. F-22 Design Usage

9.5.6. Impact of Flight Data Acquisition on Fleet Management Decisions

James Greer, USAF Academy – CAStLE

In the summer of 2005, a U.S. Coast Guard HC-130H aircraft was instrumented for the purposes of monitoring the loading and environmental conditions affecting the Center Wing Box (CWB) structure. The primary instrumentation consists of accelerometers (N_z) and strain gages (43 channels, uniaxial and rosette). Sensors for cabin and pressure altitude, temperature, and humidity were also installed. Other aircraft parameters, such as true airspeed, weight-on-wheels, ramp door position, and flap position are also collected by the monitoring system. Collecting aircraft parameters facilitates matching loads and environmental information to different phases of flight and flight conditions. A simple but novel method [1] of obtaining height above terrain was also developed for this program.

Prior to this effort, usage and flight severity data were primarily inferred through surveys of aircrew and fleet managers. However, with over 2,000 hours of flight data now in hand, survey data have been supplemented with actual usage data. These data are being used by the U.S. Coast Guard to make fleet management decisions for the HC-130H.

An in-depth review of the system was presented at the 2007 Aging Aircraft conference [2], in which the data were summarized. An even more detailed system description is now available in a limited distribution CAStLE report [3]. The flight data have raised the level of confidence in determining the Equivalent Baseline Hours (EBH) of the aircraft in the fleet. This has had the effect of eliminating some of the (understandable) conservatism built into the usage survey data. The actual flight data have shown that the operating environment is less severe than originally thought.

After rebaselining fleet severity factors, sorted by EBH "age," the youngest 16 of the Coast Guard's HC-130H aircraft will not need a new center wing box until 2013 vice 2011 [4]. While this is not a "miracle cure", these younger aircraft have had their lives extended nearly 20%. Obviously this gain considers structure as the life-limiting "system" of the HC-130H. There are other life-limiting systems and components as well (engines, wiring, hydraulics, avionics, etc.). However, the aircraft structure is arguably the most difficult and costly to affect in terms of life extension or replacement.

This program has highlighted the importance of collecting actual usage data from in-service aircraft. Of course, the results could have gone the other way: they might have shown that the general usage was more severe than believed. This would have led to earlier grounding of aircraft, but also to safety enhancement, possibly even to lives being saved. So there are no "wrong" answers—only more accurate ones—to be gained by collecting the data.

Point of Contact:

- James Greer, Center for Aircraft Structural Life Extension (CAStLE), (719) 333-3618

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[2] J. M. Greer, Jr., G. A. Shoales, P. J. Christiansen, R. A. Hartnett, and E. B. Sheppard, Jr., "HC-130H Center Wing Box Monitoring System Update," in Proceedings of the 10th Joint FAA/DoD/NASA Conference on Aging Aircraft, Palm Springs, CA, 2007.

[3] J. M. Greer, Jr. and G. A. Shoales, "HC-130H Center Wing Flight Data Acquisition: System Installation and First Year," Center for Aircraft Structural Life Extension (CAStLE), USAF Academy, CO, Technical Report USAFA-TR-2008-01, October 2007.

[4] J. M. Greer, Jr., G. A. Shoales, and R. A. Hartnett, "Impact of Flight Data Acquisition on Fleet Management Decisions," in *Proceedings of the 12th Joint FAA/DoD/NASA Conference on Aging Aircraft*, Kansas City, MO, 2009.

9.6. CHARACTERIZATION, MODELING & TESTING

9.6.1. Effect of Defect and Damage Tolerance Study for Bonded Composite Structures

Mostafa Pourmand, Northrop Grumman Corporation

Air platform criteria defines the requirements necessary to achieve structural integrity while minimizing the life-cycle cost through a series of disciplined, time phased tasks. While the usage of composite material for primary airframe structures increases (Figure 9.6-1), alternate approaches are being explored to assess structural integrity, long-term durability, and damage tolerance of these composite structural members.

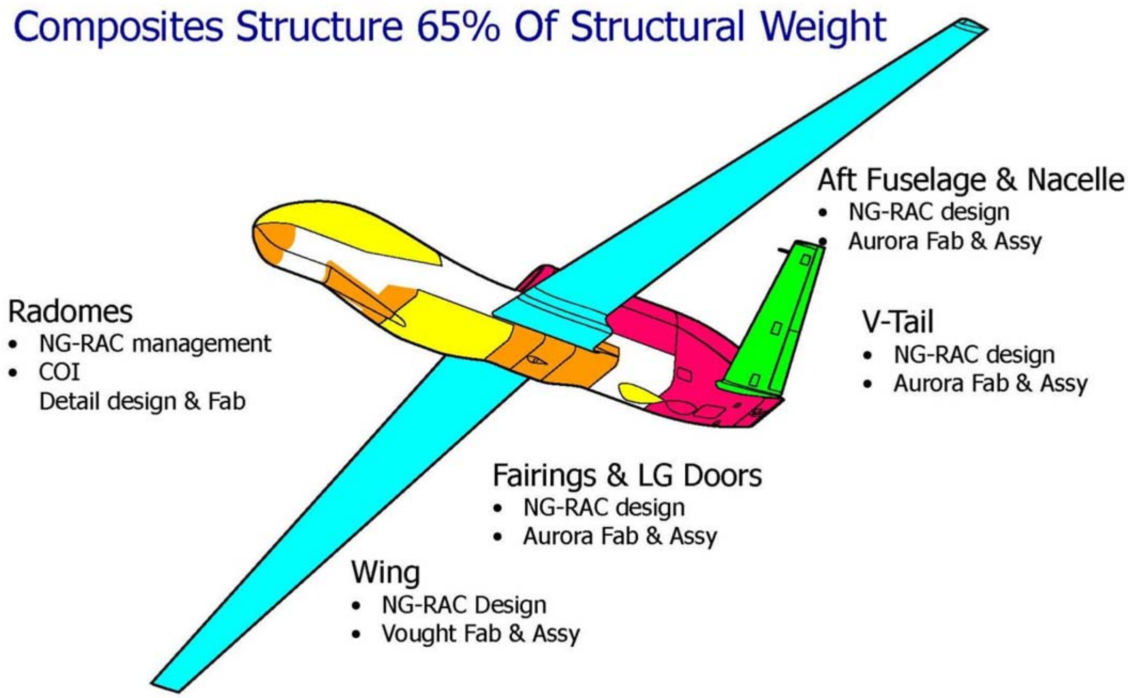


Figure 9.6-1. Global Hawk (RQ-4) Composites Structure

The ASIP Plan requires five distinctive tasks (Design Information, Design Analysis & Development Tests, Full-Scale Testing, Certification and Force Management Development and Execution). This must be followed step by step in order to provide acceptable low risk for a successful program while leading to certification.

Analytical methods are being developed for composite structures (Figure 9.6-2). The methods have to be validated by empirical data and test results. The conventional building block approach can be tedious and costly due to many variables within composite layups and processes (Figure 9.6-3). This technical effort provides a case study for alternate cost-effective damage tolerance and effect-of-defect study on all bonded composite structures. The life-cycle cost will drastically be reduced if the inspection intervals and repairs can be limited through substantiating empirical test data and flight tests.

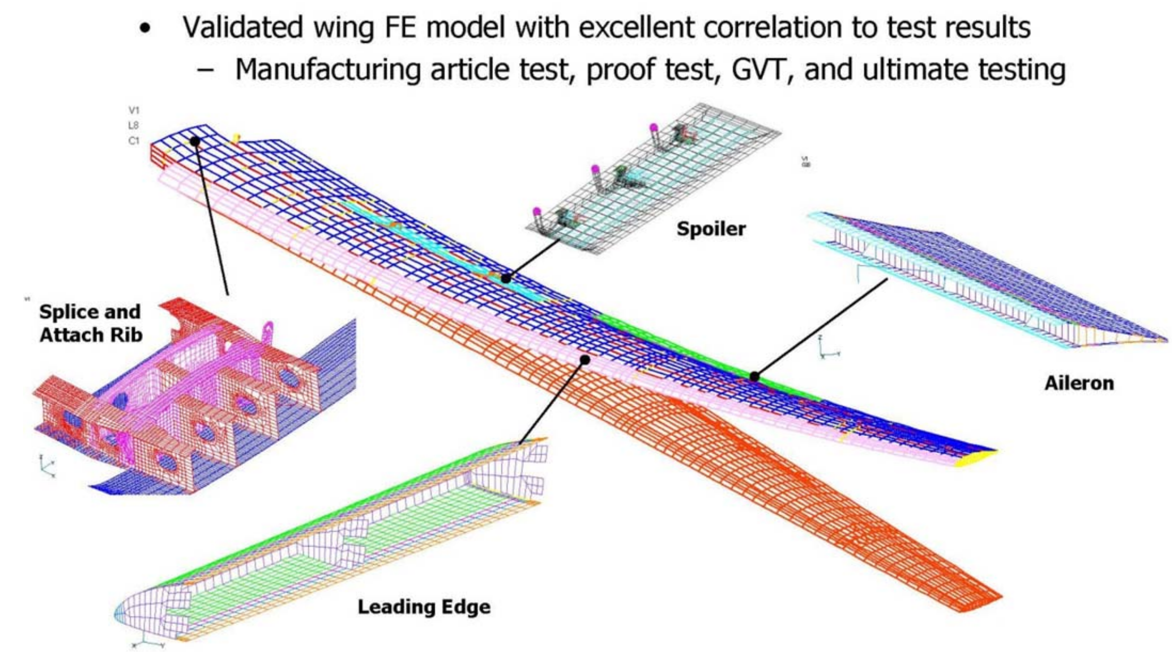


Figure 9.6-2. Global Hawk Wing Finite Element Models

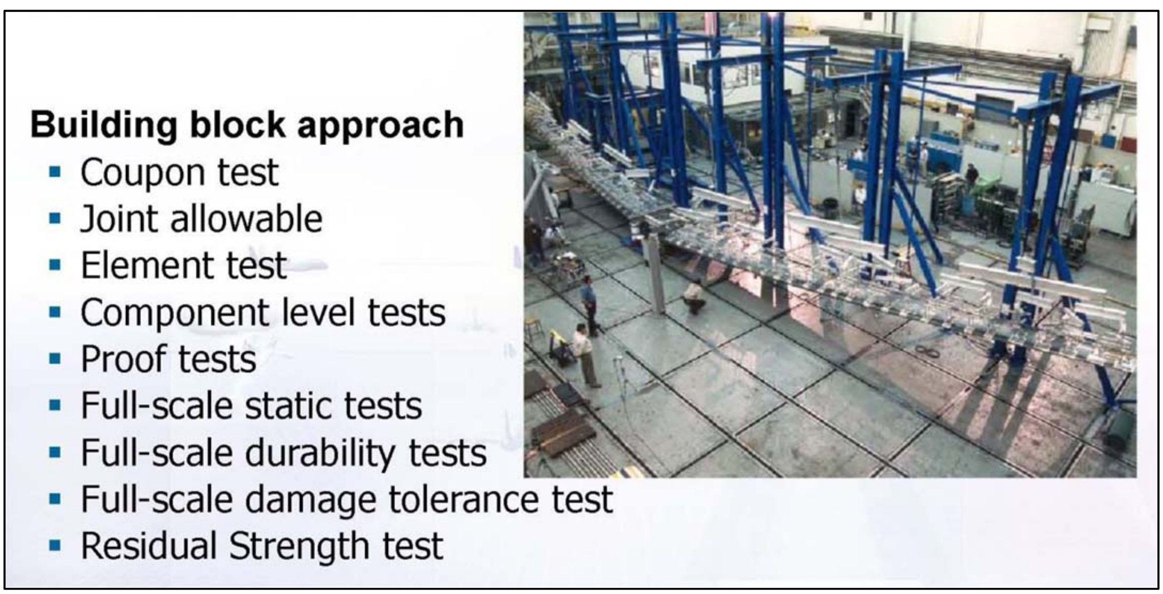


Figure 9.6-3. Global Hawk Integrated Structural Test Program

9.6.2. FEA & DTA Development for A-10 Fuselage Longeron Cracking

Hazen Sedgwick, Paul N. Clark, Robert Pilarczyk, and Gregory Stowe, USAF-OO-ALC

In May 2007 the A-10 fuselage fatigue test article experienced a catastrophic failure (Figure 9.6-4). The failure led to a fleet wide inspection of the fuselage at the failure location and other locations where cracking had occurred. Along with the inspection requirements, the cracking established the need for significant repairs. Repairs have been developed for the critical cracking locations. The repair set is temporary to restore full strength for each of the critical locations.

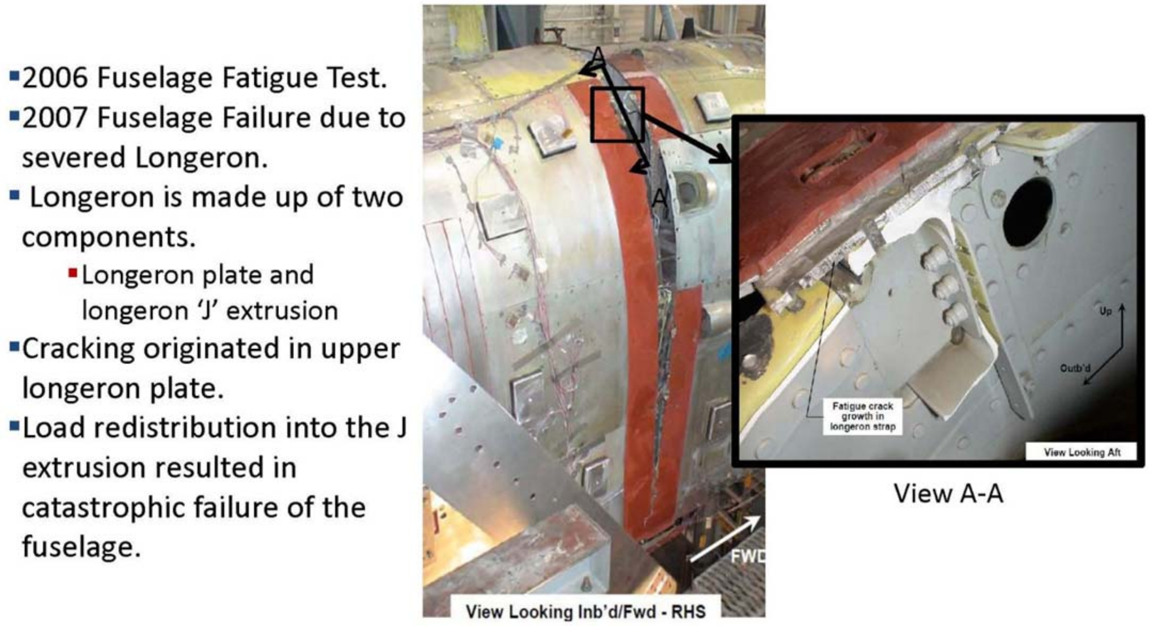


Figure 9.6-4. A-10 Fuselage Fatigue Test Failure

As a result of these inspections, a large crack was detected in the upper longeron of a fuselage at depot in July of 2009 (Figure 9.6-5). Upon further inspection, it was discovered that the crack had almost completely severed the upper longeron strap which is a critical component in the longeron assembly. Although the crack was not at the same location where failure occurred during the 2007 fatigue test, it was within an inch of the previous critical location. Upon finding the large crack, the life of the assembly came into question along with the temporary type repair. It was determined that more detailed Finite Element Analysis (FEA) was needed to support a confident Damage Tolerance Analysis (DTA). In order to achieve the information that was needed to perform DTA, multiple detailed FEA were completed. The first FEA was the baseline model which was used to correlate strain gauge data on the full-scale test. The next FEA was done to model the severed longeron strap that was found during a depot inspection. The next few models were completed in response to a variety of cracks discovered on multiple aircraft in the same upper longeron area. The DTAs were then developed using stresses, pin loads and beta factors from the various analyses (Figure 9.6-6).

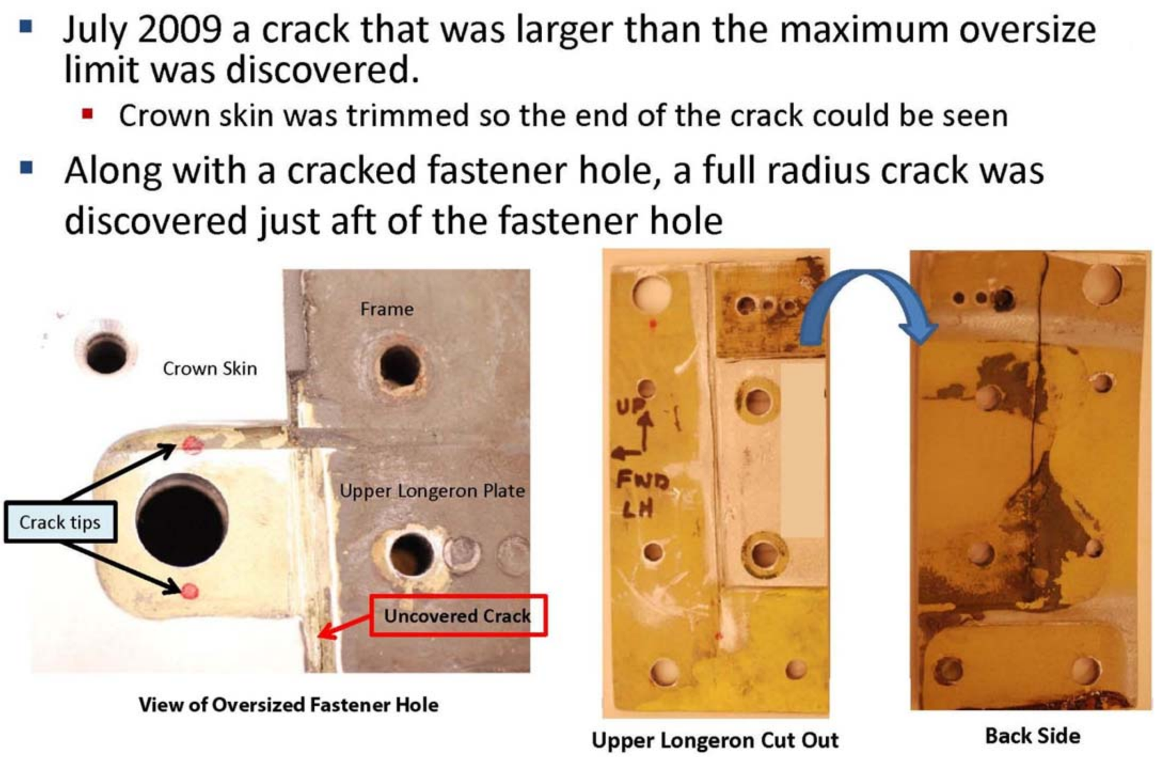


Figure 9.6-5. First Critical Finding at Depot

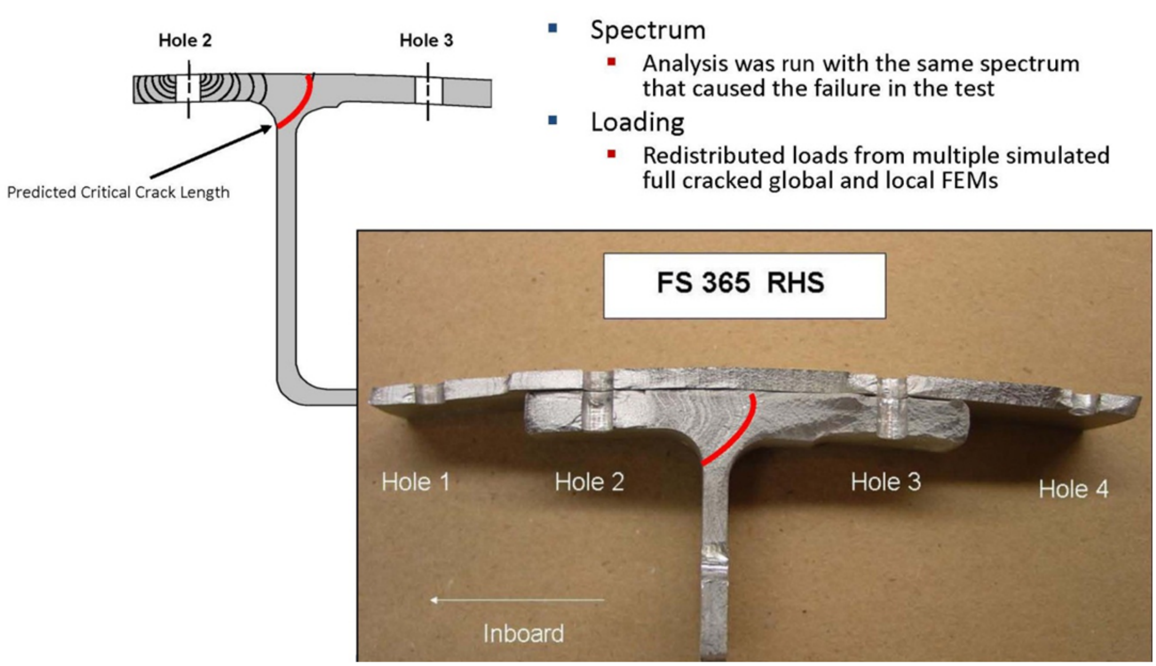


Figure 9.6-6. DTA Results Validation

This technical effort describes the techniques used to model different cracking configurations in the A-10 upper longeron strap. It also describes the DTA methods, including continuing damage techniques, used to determine recurring inspection intervals for the A-10 fleet.

9.6.3. Full-Scale Drop Testing of the F-35C Lightning II

Richard Chichester, Lockheed Martin Aero – Ft. Worth

High sink rate shipboard landings, soon to become routine for the F-35C (Figure 9.6-7), impose unique loading conditions on the airframe, landing gear, and installed equipment. Vertical sink speeds for carrier-based U.S. Navy fighter aircraft can exceed 25 feet per second, making the total landing energy to be absorbed an order of magnitude higher than that of a normal land-based landing. Shipboard landings produce highly transient landing gear loads that are transferred into the airframe through the gear trunnions and dragbrace, creating large dynamic loads on weapons, engines, and sensitive avionics equipment. Shipboard landing loads must be defined and verified during certification of a carrier-based aircraft. Methods for determining landing gear loads are fairly well understood, but many non-linear effects and complicated damping mechanisms introduce uncertainty into aircraft-level analytical dynamic response predictions that must be verified through testing. Determination of these effects and validation of shipboard landing loads through flight testing is both difficult and dangerous. Precise control of very high landing sink speeds and unusual touchdown attitudes challenges even the best pilot. The contribution of ship motion included in the definition of design landing parameters cannot be simulated during shore-based flight tests. To safely verify the loads and dynamic response of the F-35C during shipboard landings, a full-scale aircraft drop test was conducted (Figure 9.6-8). This technical effort describes the F-35C drop test objectives, setup, conditions, and examples of the results obtained. The role of the F-35C drop test in the overall F-35 Lightning II structural certification program is also summarized.

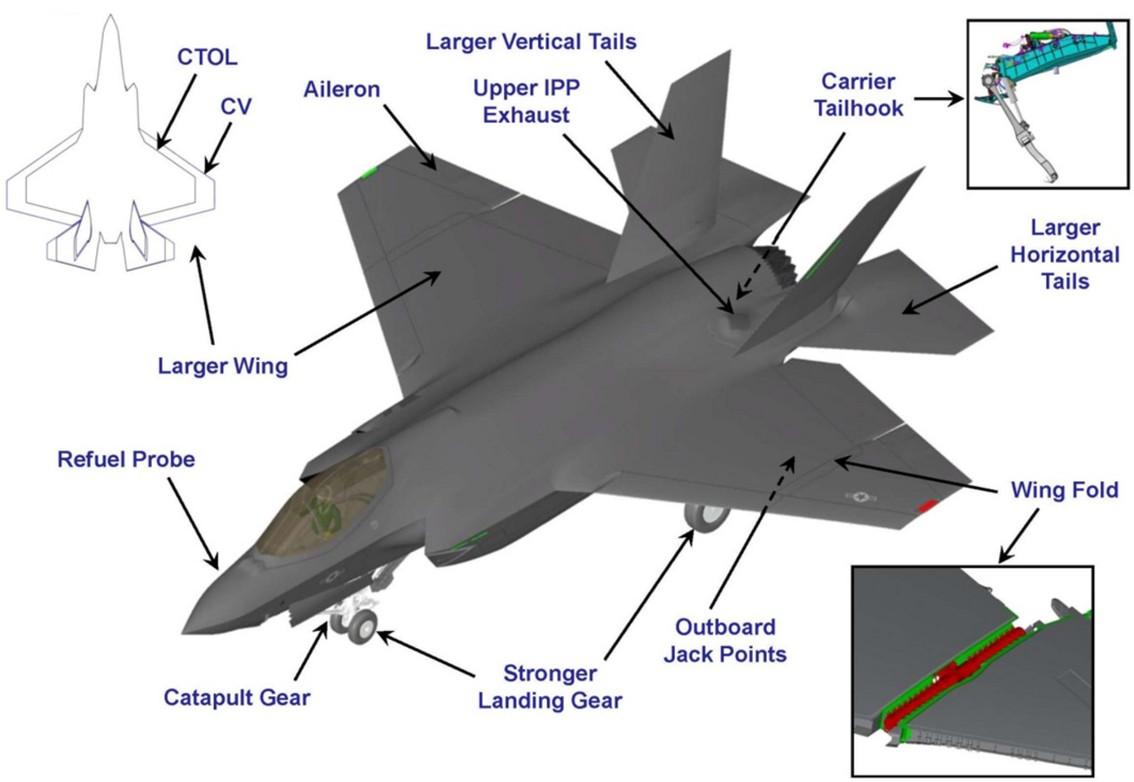


Figure 9.6-7. Unique F-35C Design Features

- **F-35C CG-1 Drop And Static Test Article**
- **Lift System Simulates 1g Of Aircraft Lift At Touchdown**
- **Engine, Avionics, and Systems Components Simulated With Dummy Masses For Correct Vehicle Mass Properties**
- **Water Used To Simulate Fuel And Control CG**
- **All Drops At Carrier Landing Design Gross Weight**
- **Testing Conducted At Vought Aircraft, Grand Prairie, TX**

Figure 9.6-8. F-35C Drop Test Article

9.6.4. Validation of Non-Linear Thermo-Mechanical Analysis for the B-2 Aft Deck

Greg Schoeppner and Jack Coate, USAF Research Laboratory – Material and Manufacturing Directorate; and Robert Tashiro, Northrop Grumman Corporation

Severe thermal loading of B-2 exhaust-washed aft decks has resulted in life-limiting cracks that jeopardize operational life cycle and structural integrity. Through analysis it was determined that localized buckling of the aft deck skin resulting from constrained thermal expansion is responsible for crack initiation and propagation. Although numerous repairs have been implemented to mitigate the effects of cracks on operational life cycle, the limited life of the decks dictates that replacement decks be built. Since full-scale durability testing of a new 3rd Generation Aft Deck (3GAD) design was not feasible prior to design approval, experimental validation of the analytical/numerical model predictions for the existing 2nd Generation Aft Deck (2GAD) was accomplished to provide confidence in the fidelity of the models (Figure 9.6-9). Predictions for the full-field temperature and displacement distributions for the 2GAD resulting from a series of engine ground run conditions were accomplished by Northrop Grumman. In an independent effort by the Air Force, the temperatures and displacements of the 2GAD were measured during engine ground run testing at Edwards AFB for the series of engine ground run conditions used for Northrop Grumman's predictions. A combined Forward Looking Infrared (FLIR) thermography camera and an ARAMIS digital image correlation system was used to measure the time dependent full-field temperature and displacement of the 2GAD for each of the engine ground run conditions (Figure 9.6-10). A comparison of the experimentally measured and analytically predicted temperatures and displacements showed an excellent correlation and confirmed that localized buckling of the skin does occur as predicted by the analytical models. Based on the excellent correlation of the predictions with the measure response for 2GAD, the risk associated with approving the design of 3GAD based solely on analytical/numerical predictions was deemed to be acceptable. The technical effort will focus on describing the analytical/numerical models used to predict the behavior of the aft deck, the experimental techniques, equipment, and set-up for the engine ground run test (Figures 9.6-11 and 9.6-12), and finally a comparison of the predicted and measured response of the aft deck.

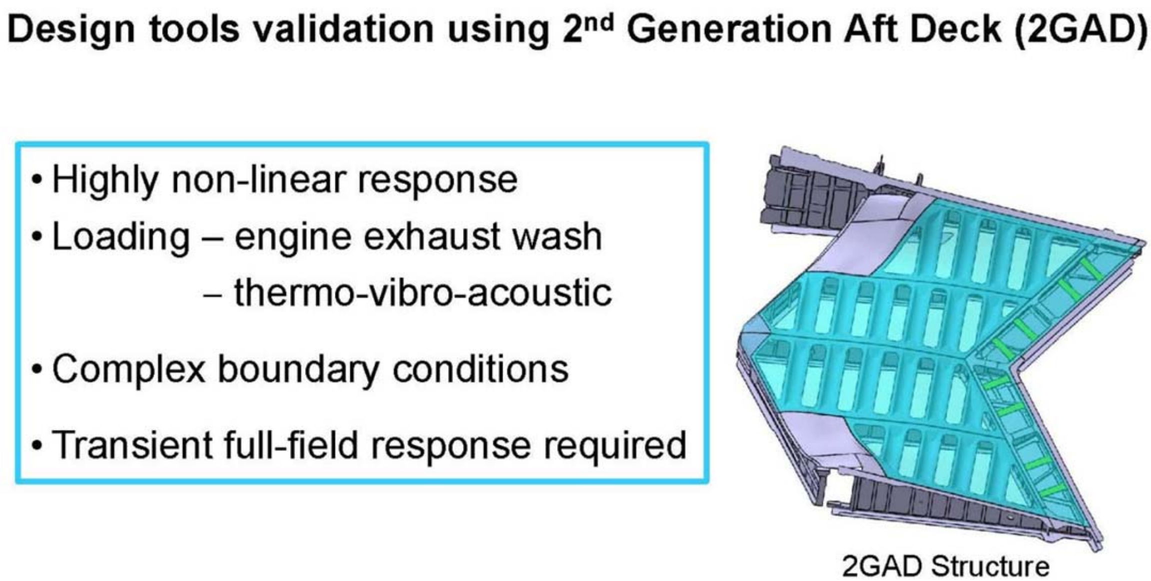


Figure 9.6-9. Design Challenges for 2nd Generation Aft Deck (2GAD)

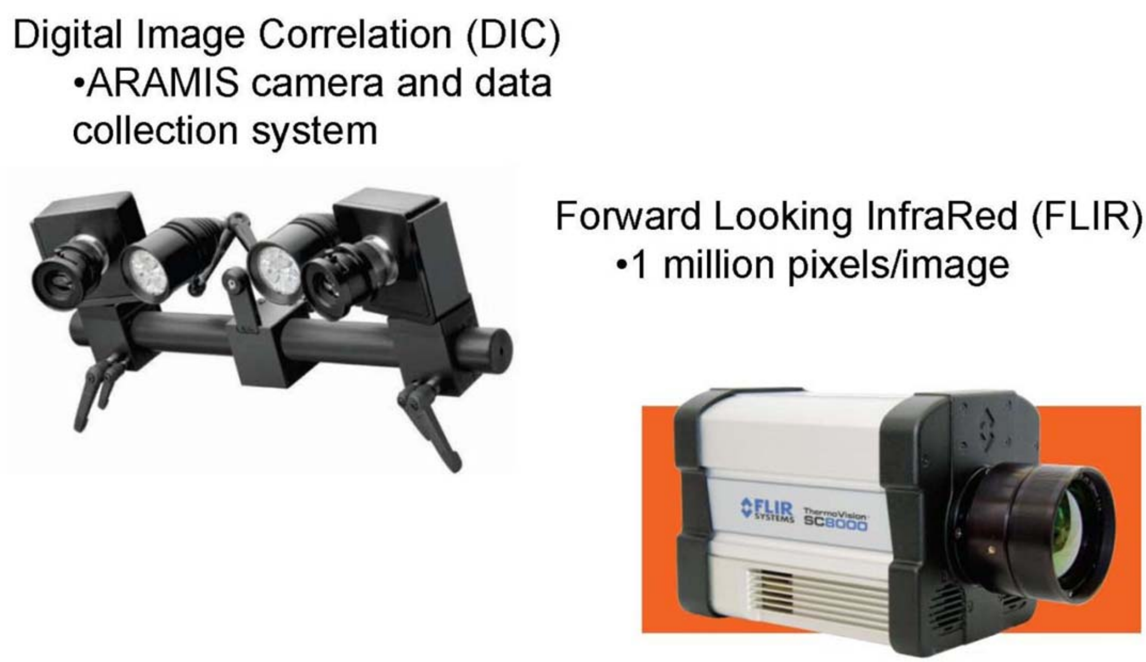


Figure 9.6-10. Combined DIC & FLIR Equipment

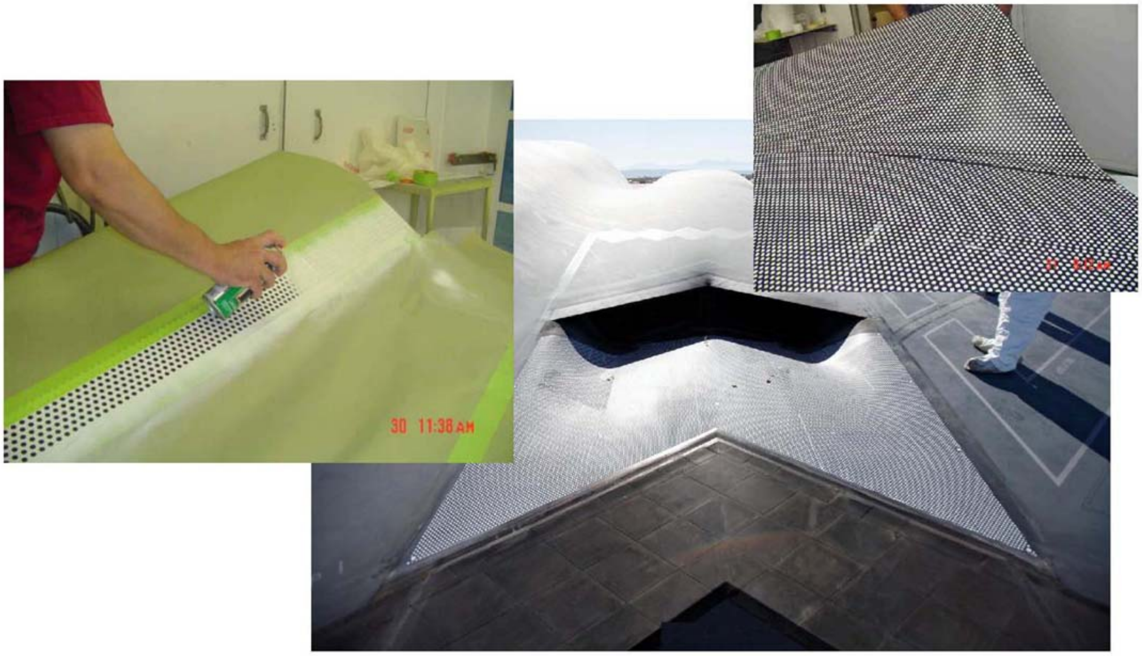


Figure 9.6-11. DIC Speckle Paint for Combined Ground Test

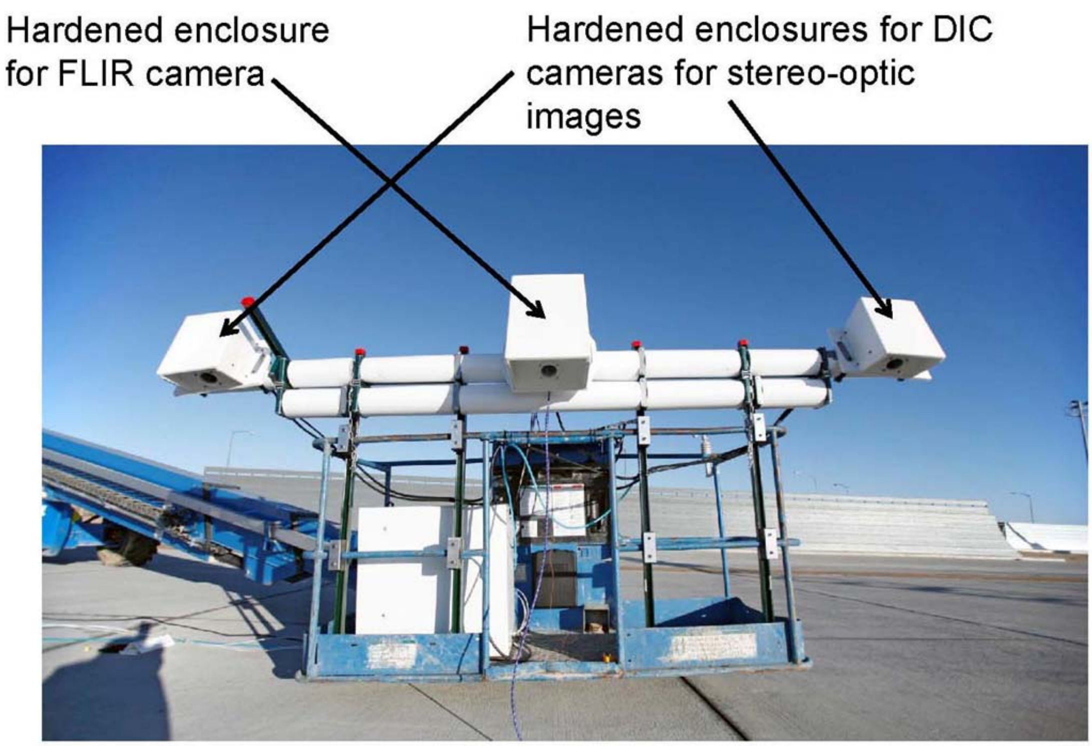


Figure 9.6-12. Hardened Camera Enclosure for Combined Ground Test

9.6.5. Large-Grain Effects on Fatigue Growth of Corner Cracks in Ti 6AL-4V BSTOA

Thomas R. Brussat, Tom Brussat Engineering, LLC; Timothy Blase and Paul Toivonen, Lockheed Martin Aero – Ft. Worth; and Ryan Carey and Andrew Makeev, Georgia Institute of Technology

Ti 6Al-4V BSTOA is used in safety critical aircraft structure because of the excellent da/dN properties associated with its unusually large-grain microstructure. Grain sizes typically range from .02 to .05 inch. However, for a .05-inch corner crack, the size of the grains relative to the crack size brings into question the validity of the linear elastic fracture mechanics (LEFM) theory used in damage tolerance analysis methods. This technical effort describes an extensive experimental and analytical effort to investigate and improve the accuracy of LEFM in fatigue crack growth analysis of corner cracks in Ti 6Al-4V BSTOA. Beam specimens with a one-inch square, diamond-shaped cross section are cycled in 4-point bending, so the crack propagates as a near-circular corner crack (Figure 9.6-13). Using highly polished surfaces and intense lighting, simultaneous macro-photographic digital images record the lengths of damage on both visible faces of the corner crack automatically (Figure 9.6-14). This permanent data record enables post-test recording of hundreds of crack measurements starting at lengths smaller than 0.02 inch. A matrix of 72 test results is presented and analyzed from two plate and two forging heat-treat lots of material and including 6 cyclic stress conditions and 3 replications of each combination of stress and material lot. Results consistently show accelerated growth rates for .05 inch corner cracks compared to the measured rates in compact tension specimens at the same cyclic stress intensity (Figure 9.6-15). Empirical beta factors are developed to adjust LEFM theory for damage-tolerance-sized corner cracks. Application to typical aircraft structural damage tolerance problems (such as corner cracks at fastener holes) is examined and discussed.

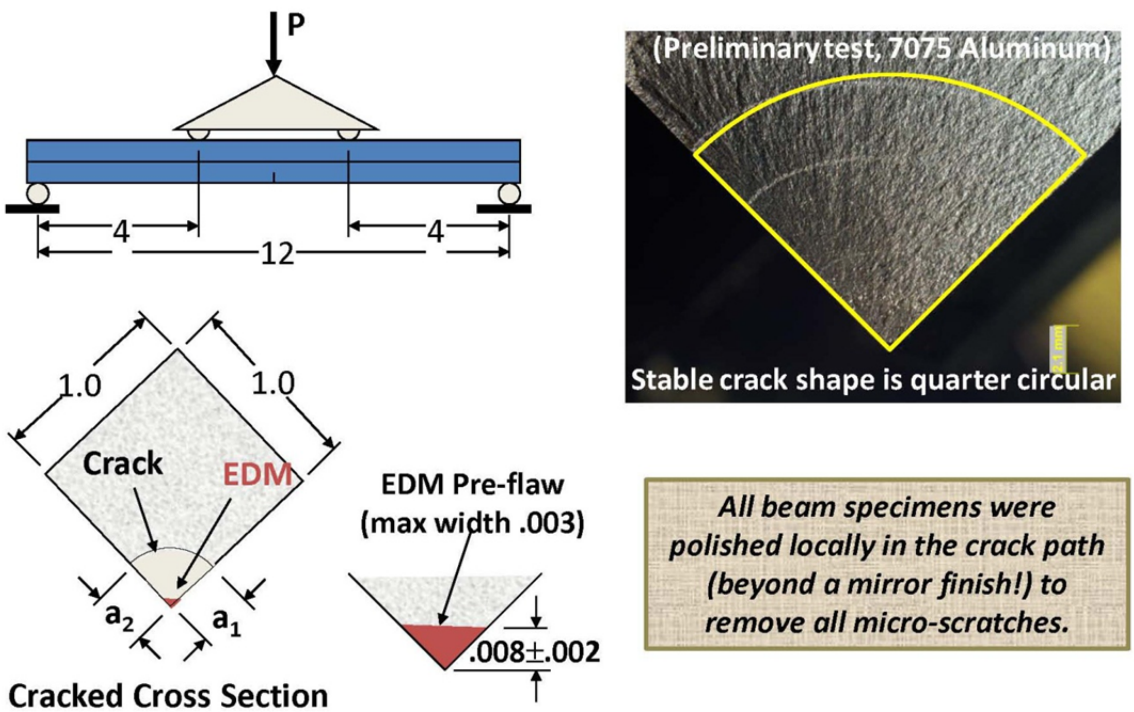


Figure 9.6-13. "Diamond Beam" Test Specimen

Pairs of cameras (front & back) are auto-coordinated with peak stress; record crack sizes

High-intensity lights impinge at shallow angle on finely-polished specimen surface

Low-friction supports do not restrict bending deflections of beam specimen

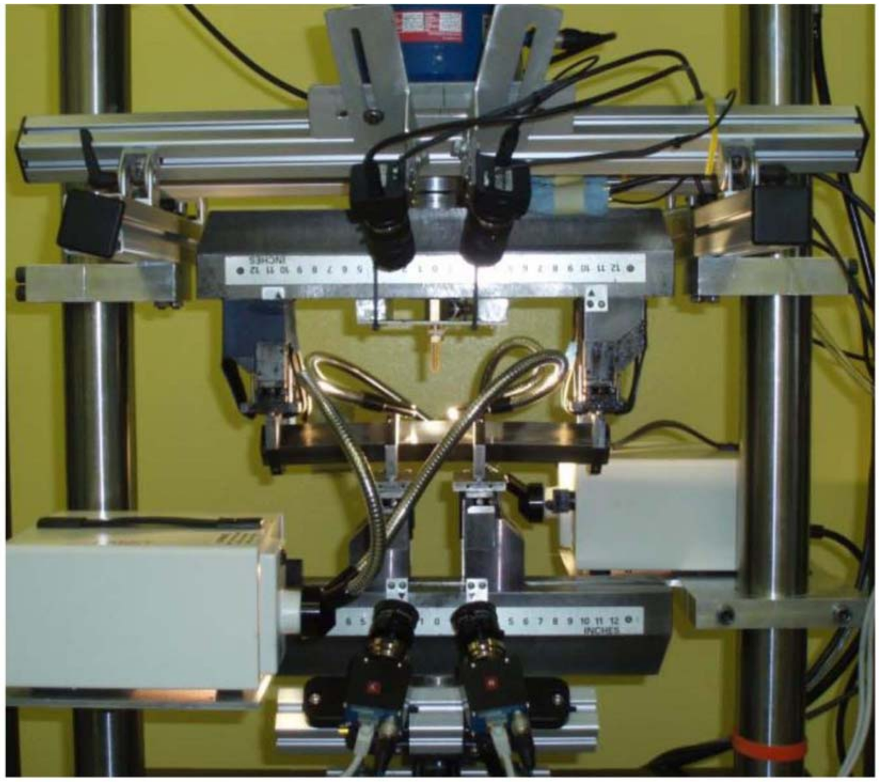


Figure 9.6-14. Test Methods: Test Set-up

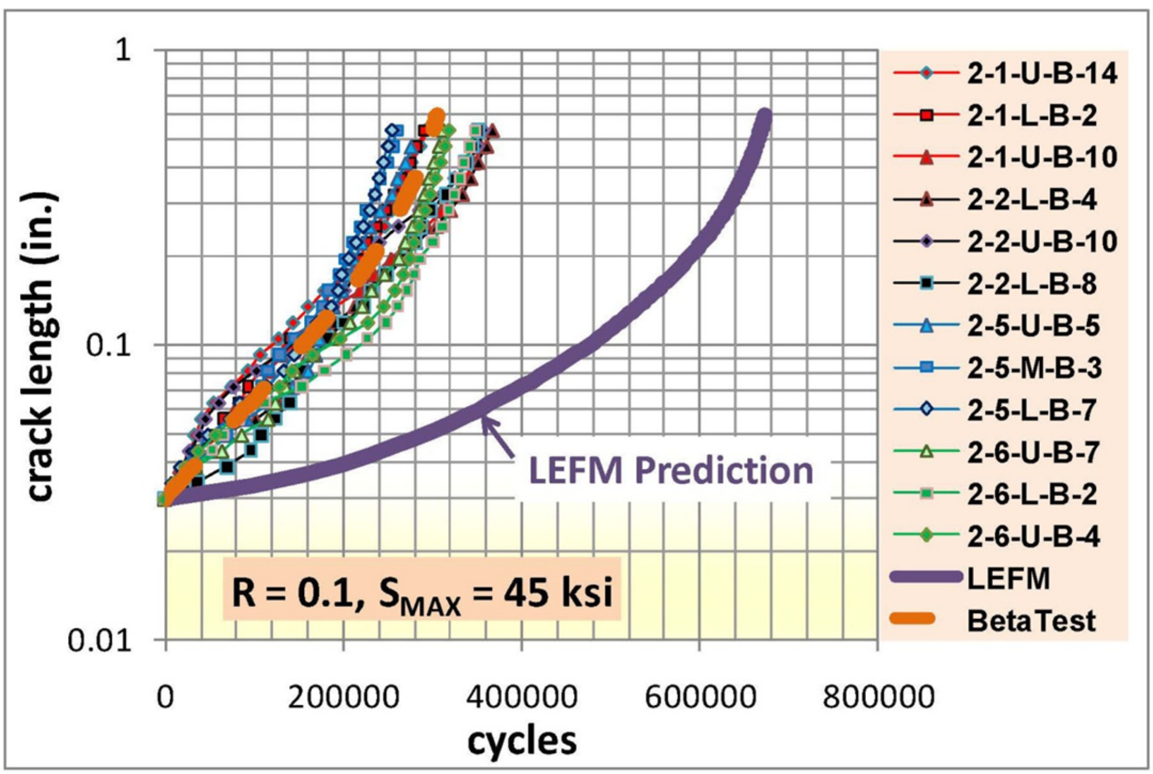


Figure 9.6-15. Test Results & LEFM Prediction at 45 ksi

9.6.6. Aircraft Level Finite Element Analysis Validation for the STOVL F-35 Lightning II

David M. McSwiggen and Robert J. Burt, Lockheed Martin Aero – Ft. Worth

Finite Element Analysis plays a pivotal role in the design and analysis of fifth generation fighter aircraft, such as the F-35 Lightning II Joint Strike Fighter. The F-35 Program is transitioning from design and development into full-scale ground and flight testing. The validation of vehicle-level finite element models using full-scale test results is an essential pre-requisite for successful execution of downstream Aircraft Structural Integrity Program tasks. Final airworthiness certification, development of force management data packages, and life tracking all depend on a validated internal loads model. These efforts ensure the structural integrity of the aircraft, leading to certification for flight in accordance with and in support of a rigorous and disciplined Aircraft Structural Integrity Program.

This technical effort provides an overview of the full-aircraft Finite Element Analysis Validation process for the Short Take Off and Vertical Landing (STOVL) variant of the F-35 Joint Strike Fighter (Figure 9.6-16), shows how the Finite Element Analysis validation (Figures 9.6-17 and 9.6-18) fits into the over all structural certification of the aircraft and provides a preliminary report on the STOVL F-35 finite element model performance as compared to measured full-scale test results for selected locations throughout the aircraft.

Figure 9.6-16. F-35 STOVL Variant

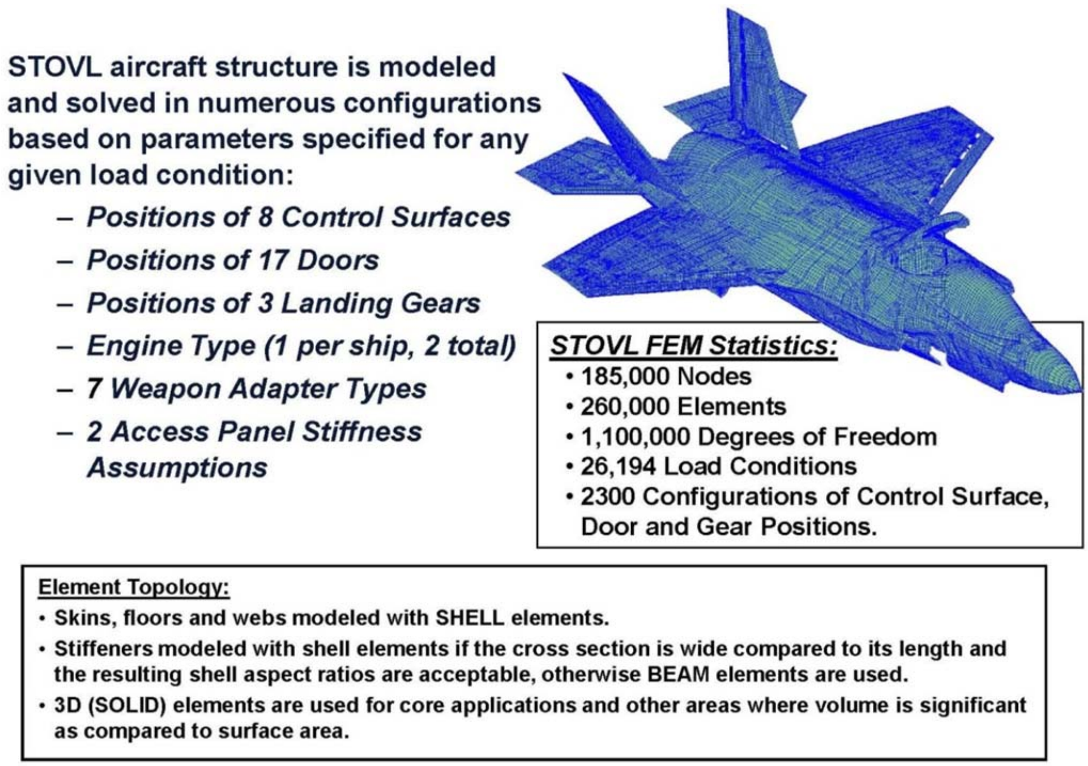


Figure 9.6-17. STOVL Aircraft Finite Element Model

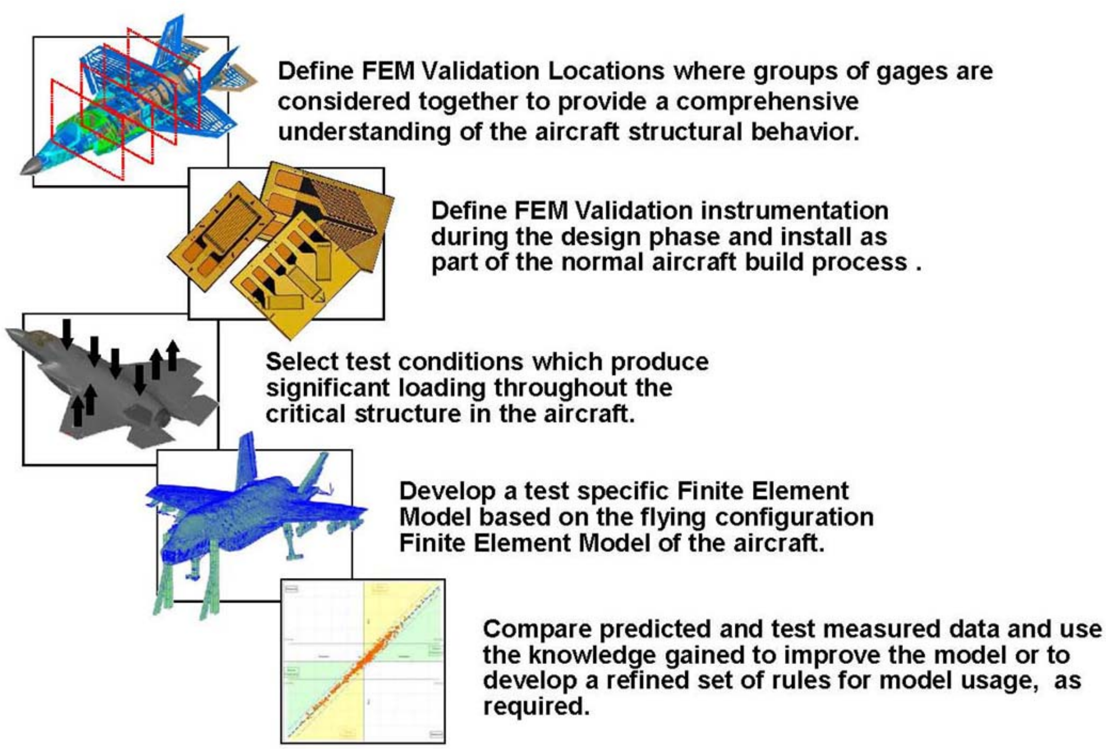


Figure 9.6-18. Aircraft FEM Validation Process

9.6.7. Development and Validation of an FE Model of the F-15C Aircraft

Robert McGinty and David Carnes, Mercer Engineering Research Center (MERC)

In November 2007, a US Air Force F-15C fighter jet experienced a structural failure in the forward fuselage that led to its complete separation from the aircraft during flight. The subsequent investigation revealed the failure was caused by cracks growing from a flaw in the upper longeron that was introduced during its fabrication nearly 30 years earlier. Although the mishap's cause was fully resolved, portions of the investigation were obliged to rely on dated structural analysis models that were slow and cumbersome to use. Updating the models to take maximum advantage of the latest in computational technologies could greatly improve their efficiency, versatility, and overall value in addressing structural analyses in the future. Shortly thereafter, the Air Force tasked Mercer Engineering Research Center (MERC) to develop a highly detailed finite element (FE) model of the F-15C aircraft to quickly address this need as well as to support a planned fatigue test of the airframe.

MERC has recently completed the development and validation of a highly refined FE model of the F-15C aircraft within a one-year time frame set forth by the Air Force. Every structural component in the aircraft, excluding individual fasteners and clips, is explicitly represented in the FE model by linear plate elements, five hundred thousand in all. The high level of detail in the model permits unparalleled insight into the structural response of the airframe to imposed loading conditions. This technical effort reviews the development and validation of the FE model with particular emphasis placed on the unique challenges of developing such a highly refined model and the benefits derived from the detailed structural insights it provides.

A major obstacle to the development of any FE model of the F-15C derives from the fact that it was designed prior to the adoption of solid modeling techniques. Only mechanical drawings were available to guide mesh development. Therefore, MERC chose to create a complete solid model (Figure 9.6-19) of the aircraft to provide the geometric foundation for the subsequent mesh generation (Figure 9.6-20). The efficiencies in meshing gained from the solid model's presence compensated for the time required to create it, permitting the near simultaneous development of both.

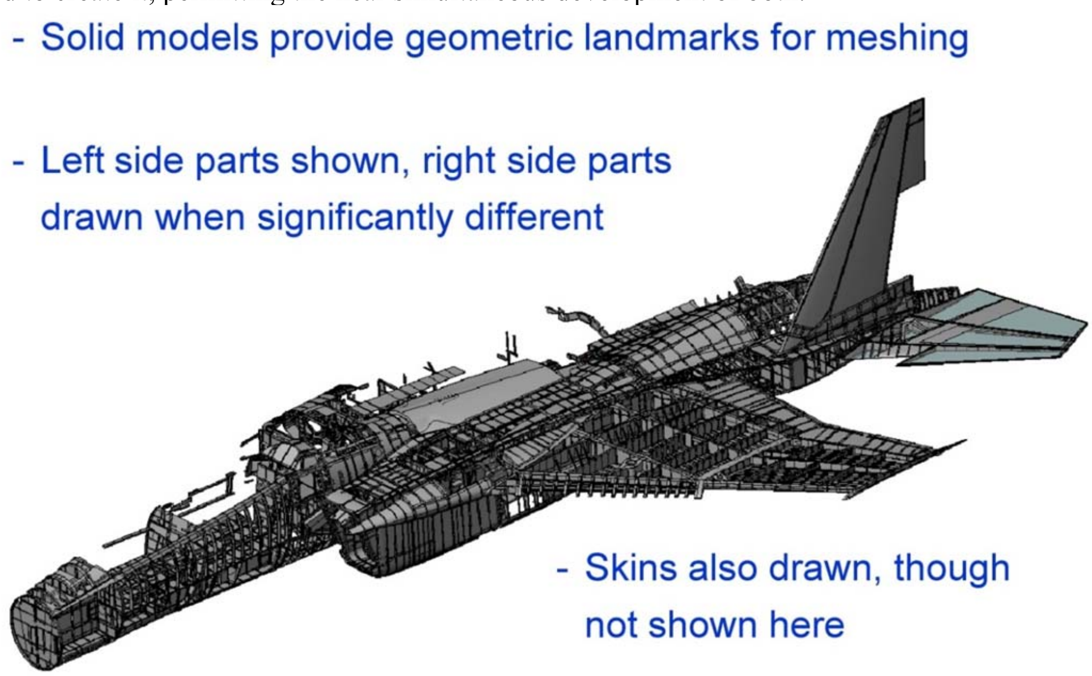


Figure 9.6-19. F-15 Solid Model

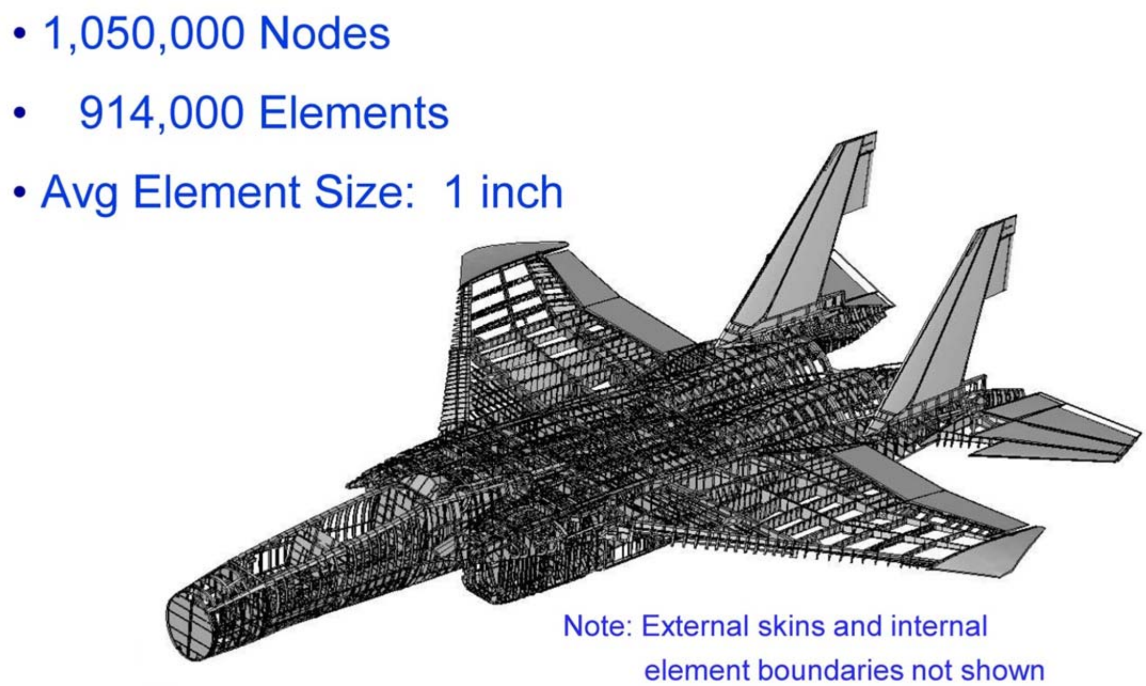


Figure 9.6-20. F-15 Finite Element Model

All components were meshed using linear plate elements placed at midplanes of the solid models. Short rigid elements oriented perpendicular to the plate elements were used to fasten parts together, permitting improved representations of the stack-ups of fastened parts. One dimensional beam and bar elements are not used in the model. This enhances direct visualization of the structure and is particularly valuable when displaying color contours of stress and strain states. Mesh development also adhered to minimum element size criteria with consideration for explicit transient FE analyses in mind. The model is therefore ideally suited for use in any future mishap analyses, such as the one that motivated its development three years ago. Finally, the model has been validated against ground-based strain gage surveys performed by the OEM (Figure 9.6-21).

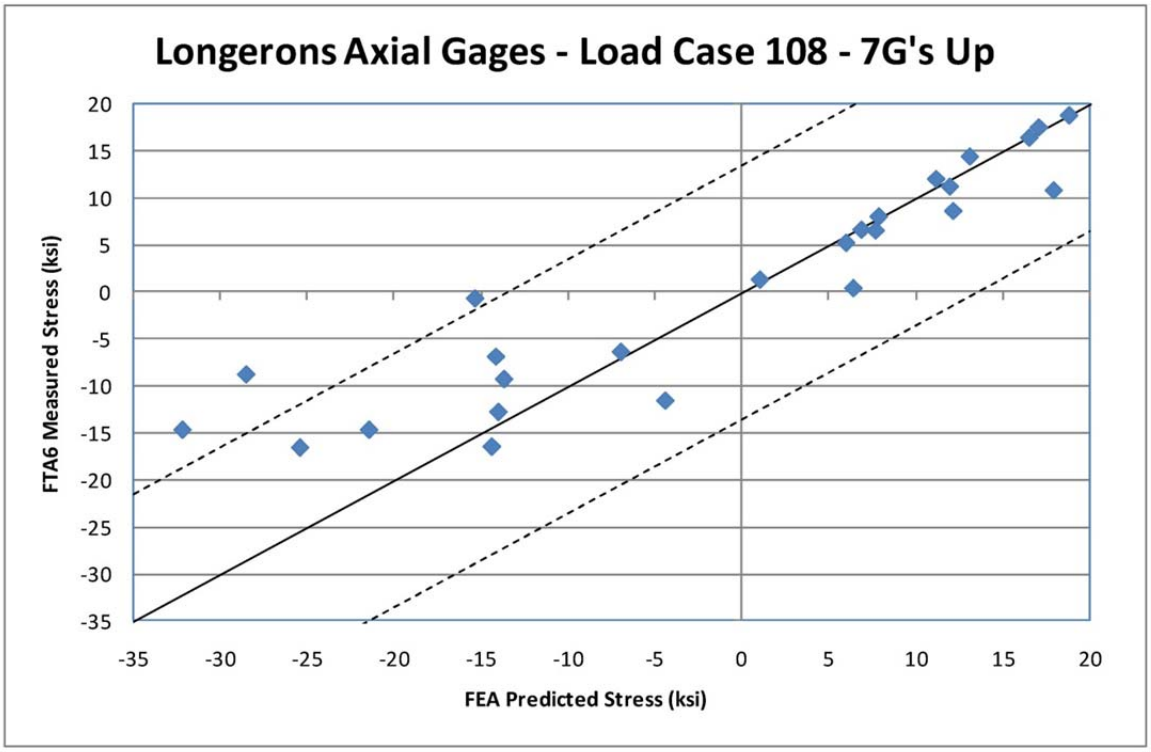


Figure 9.6-21. F-15 Fuselage Validation

9.6.8. Aircraft Level Dynamic Model Validation for the STOVL F-35 Lightning II

David A. Boyce and Robert Burt, Lockheed Martin Aero – Ft. Worth

Validation of the vehicle-level finite element models used for structural dynamic analysis is a key element of the F-35 Lightning II Joint Strike Fighter structural certification plan. These models are the fundamental aircraft stiffness and mass representations used for all F-35 flutter, aeroservoelastic, buffet, and dynamic response predictions. As the F-35 Program progresses from design and development into full-scale ground and flight testing, a disciplined approach is being taken to correlation of vehicle-level dynamic models with component and full-scale ground vibration tests (GVT), Figures 9.6-22 and 9.6-23. Initial results, including full-scale GVTs, show excellent correlation with predictions (Figure 9.6-24). As the F-35 family of aircraft begins flutter and buffet flight testing, these validated structural models form the basis for preflight envelope expansion predictions as well as for simulation of empennage dynamic fatigue tests to be conducted in the laboratory. This technical effort describes the development of the F-35 dynamic finite element models, the conduct of component and full-scale GVTs, correlation of the finite element models, results to date, and planned use of the models as the F-35 program moves forward.

Figure 9.6-22. Aircraft Ground Vibration Test (GVT)

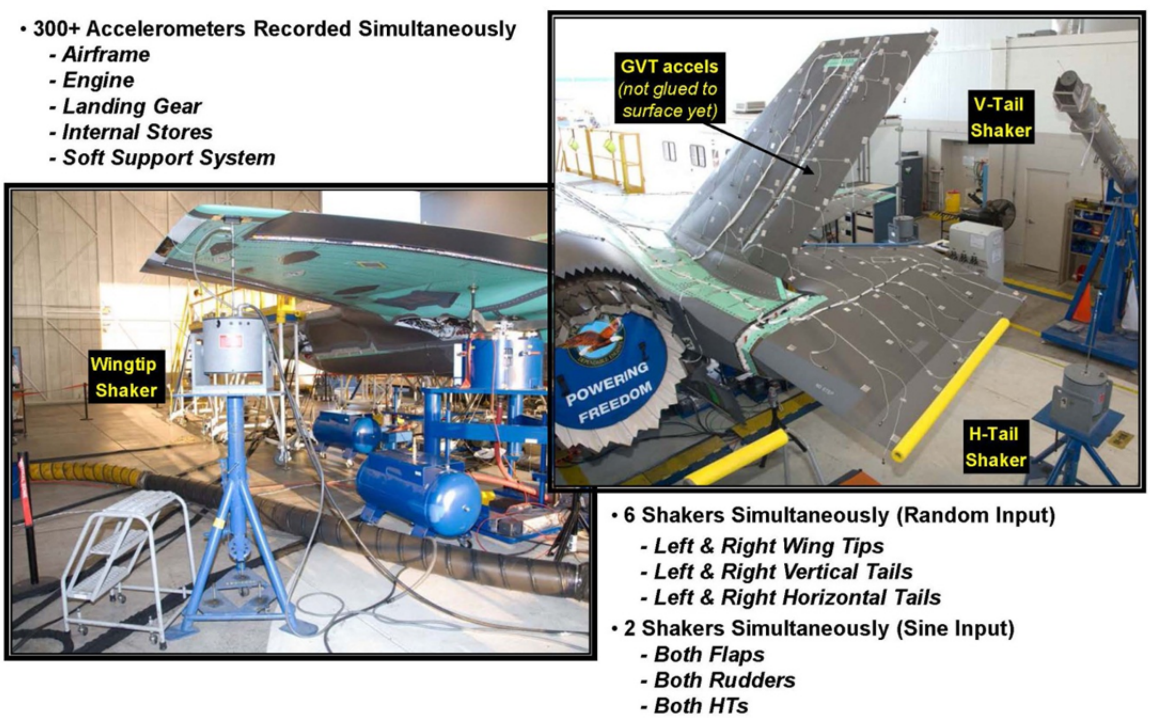


Figure 9.6-23. GVT Instrumentation & Shakers

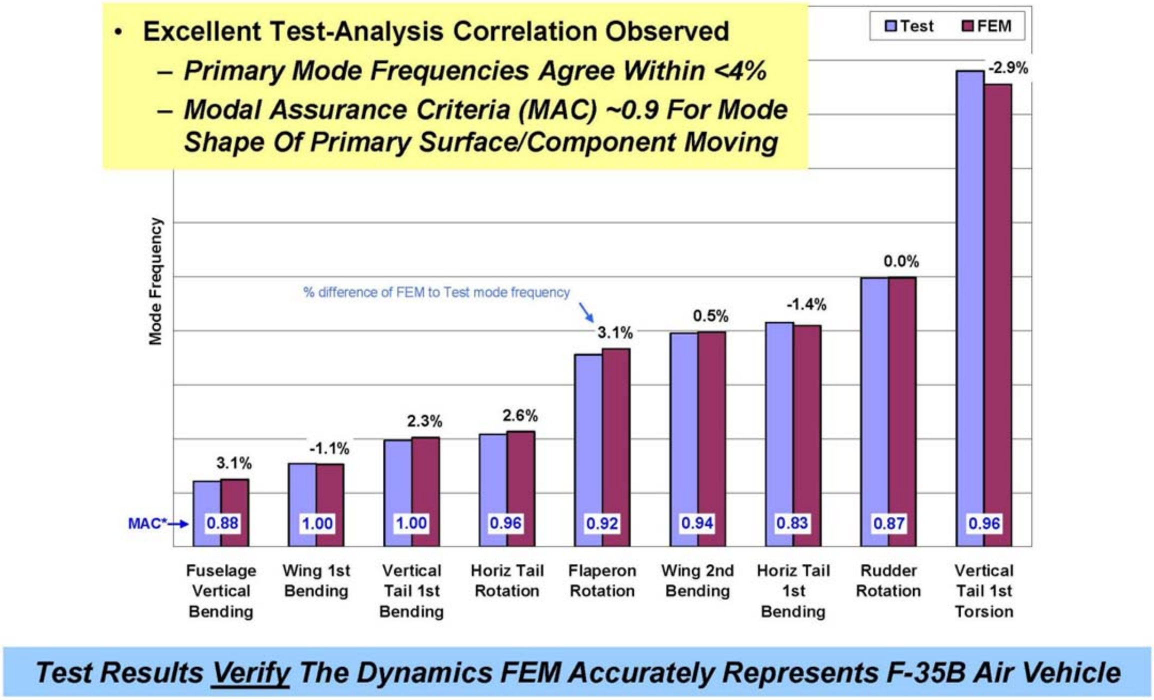


Figure 9.6-24. Clean Wing GVT-FEM Correlation

9.6.9. Stress Intensity Solutions for Continuing Damage

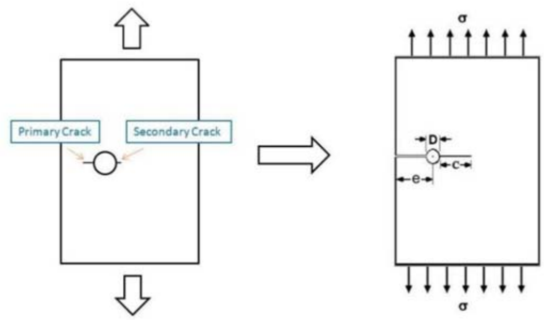
James A. Harter, USAF Research Laboratory – Air Vehicles Directorate

This technical effort presents and documents closed-form stress intensity factor solutions for through-the-thickness and corner cracks at an edge notch under tensile loading. The edge notch is representative of a continuing damage scenario in which a crack has grown from a hole through the near edge ligament. In order to continue a damage tolerant life prediction, a secondary crack is assumed to exist on the opposite side of the hole (Figure 9.6-25).

When an initial crack grows to a physical boundary (free edge, adjacent hole, etc.), standard LEFM methods can not be used to continue to predict crack growth in the structure unless a secondary crack is assumed to exist.

One of the most common continuing damage scenario is the case of a crack growing from a fastener hole to the near edge of a plate.

When the near ligament is severed, the resulting geometry is equivalent to a U-shaped notch.



Standard practice assumes an initial 0.05 corner crack (primary crack) on the near side of a hole and an 0.005 corner crack (secondary crack) on the opposite side of the hole.

Figure 9.6-25. Continuing Damage Scenario

Users in the field (ALCs) currently rely on simplified and/or case-by-case finite element models to determine stress intensity solutions, which are used to predict the life of structures with continuing damage requirements. The accuracy of simplified models is very difficult to assess, and detailed finite element models often require a great deal of time and expertise to develop. A verified, closed form solution would be of great help to ALC users.

The new solutions were based on results of an extensive finite element modeling effort using FRANC3D/NG & ABAQUS. The solutions cover hole edge distances from e/D = 0 to 49.5, and have been verified for e/D = 0 to 8. In addition, finite width effects are also included and have been verified for a number of plate geometries (Figure 9.6-26).

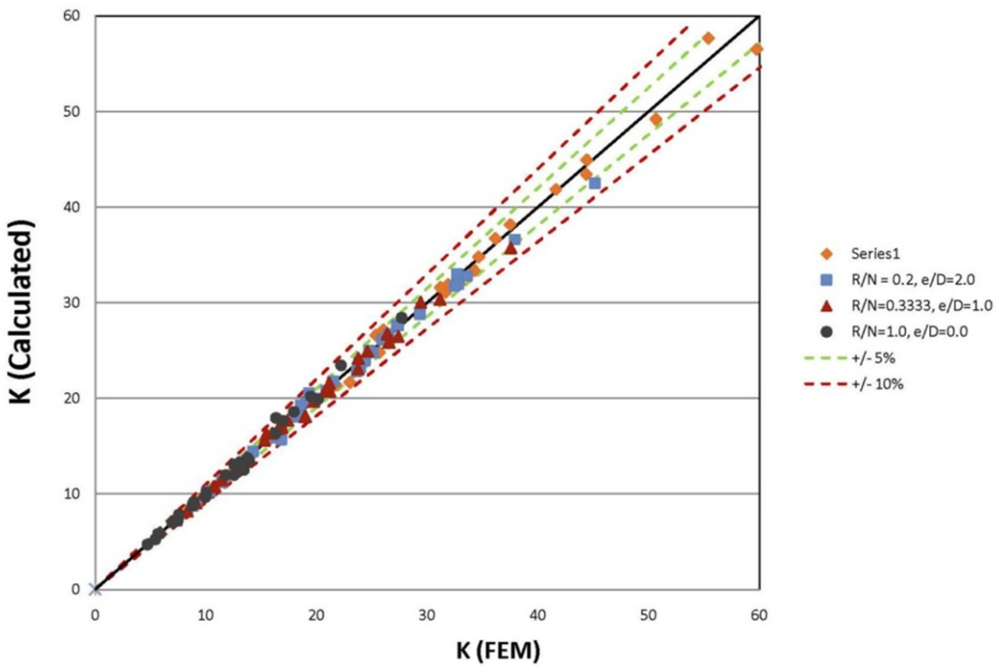


Figure 9.6-26. Corner Crack Notch Correlations

These solutions will provide the ALC users with a documented and verified solution for many common continuing damage problems.

9.6.10. Stress-Intensity Factor Equations for Very Deep Surface Cracks and Two-Symmetric Corner Cracks at a Circular Hole in a Plate

J. C. Newman, Jr. and Y. Yamada, Mississippi State University; M. Mear and H. Tran, University of Texas; and I. S. Raju, NASA-Langley Research Center

During the past decade, Fawaz and Andersson have generated some very accurate stress-intensity factor solutions for corner cracks at a circular hole subjected to remote tension and bending loads using a p-version finite-element analysis (FEA) code. These new results were previously compared with the Newman-Raju equations for remote tension, and the Zhao-Newman-Sutton equations for remote bending. The significant discrepancies occurred for very deep cracks [crack-depth-to-plate-thickness (a/t) ratios > 0.8]. This technical effort is part of an effort to analyze some of the other three-dimensional (3D) crack configurations, such as very deep surface cracks in a plate, using a three-dimensional boundary-element code, FADD3D, to see if the deep-crack discrepancy in the Newman-Raju equations occurs in the other 3D crack configurations. In addition, two-symmetric corner cracks in various width plates were also analyzed with the FADD3D code (Figure 9.6-27). This technical effort also presents some improved stress-intensity factor equations for a surface crack in a plate and for two symmetric corner cracks at a circular hole in a plate under remote tension and bending loads that cover a very wide range of a/t ratios, crack-depth-to-crack-length (a/c) ratios, and hole-radius-to-plate thickness (r/t) ratios. Comparisons are made between the Fawaz-Andersson FEA solutions and the FADD3D results for two-symmetric corner cracks at a circular hole, and the Raju-Newman FEA solutions and FADD3D results for surface cracks in a plate under tension and bending loads (Figure 9.6-28). Comparisons are also made on their influence on damage-tolerant crack-growth life predictions. These new equations can easily be incorporated into the AFGROW and NASGRO life-prediction codes by modifying the existing Newman-Raju equations.

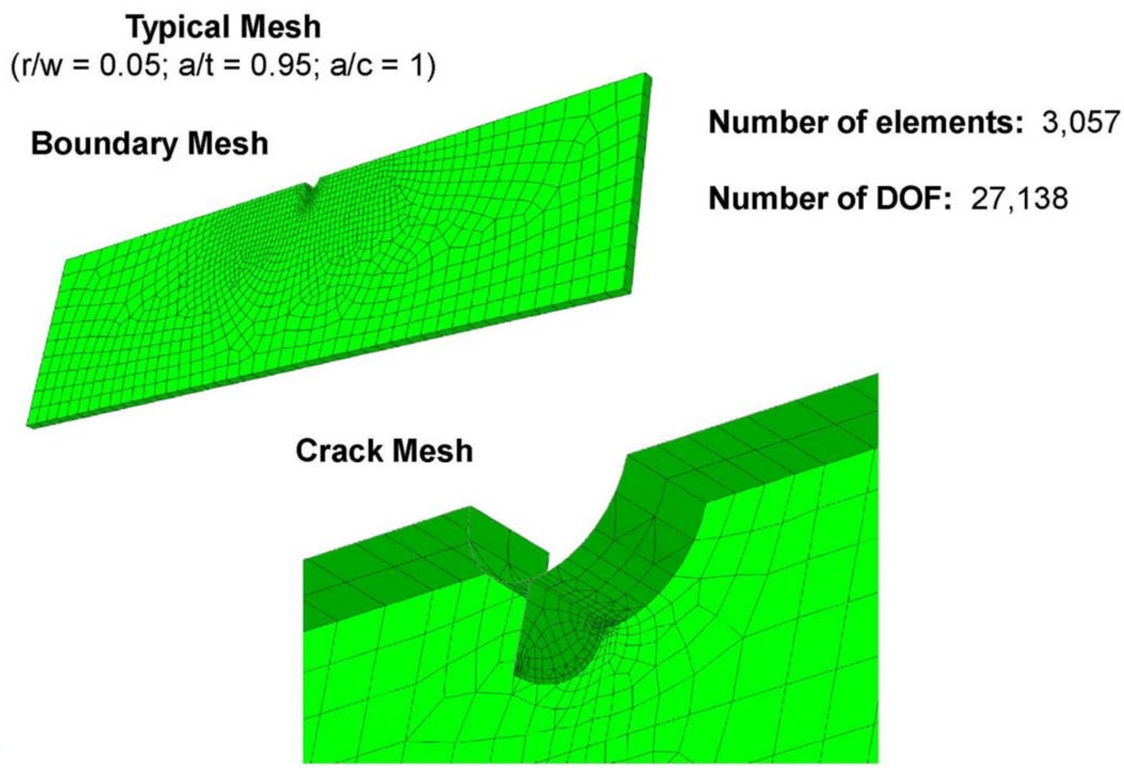


Figure 9.6-27. Typical FADD3D Mesh for Corner-Crack-at-a-Hole

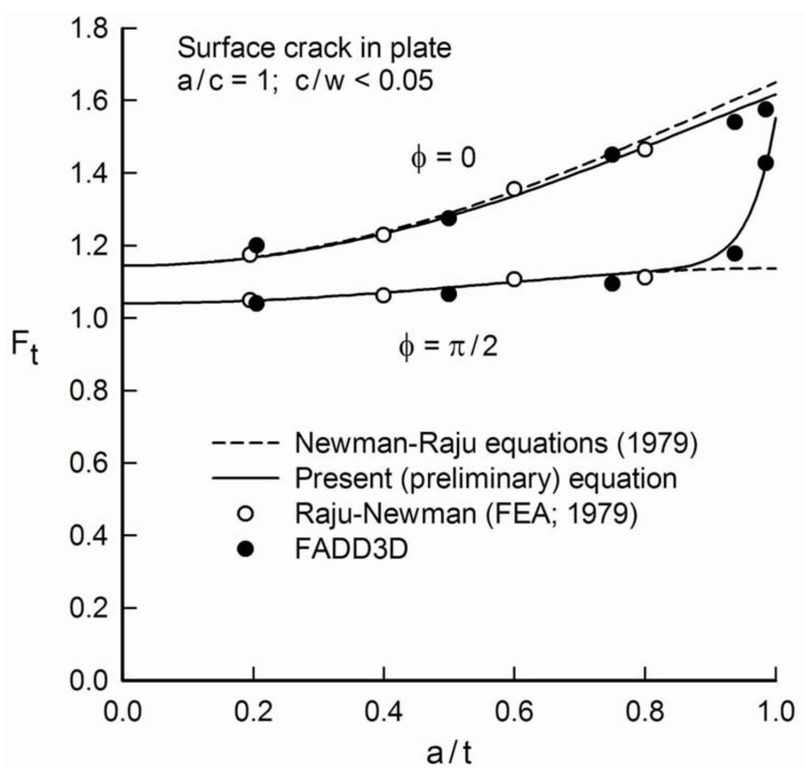


Figure 9.6-28. Surface Crack in Plate Under Remote Tension

9.6.11. Birdstrike Certification Tests of F-35 Canopy and Airframe Structure

Steve D. Owens, Eric O. Caldwell, and Mike R. Woodward, Lockheed Martin Aero – Ft. Worth

Airframe structure and canopy components of the F-35 Joint Strike Fighter are designed to sustain bird strike impacts defined by a probabilistic criteria without compromise to continued flight safety. This technical effort summarizes the probabilistic risk-based bird strike design requirements implemented for the F-35 airframe and canopy (Figure 9.6-29). Results of structural analyses and tests generated to certify compliance with these requirements are discussed (Figure 9.6-30). High speed video from bird strike tests of the F-35A Conventional Take-off and Landing (CTOL) canopy and F-35B Short Take-off Vertical Landing (STOVL) variant Lift Fan Inlet (LFI) door were developed (Figure 9.6-31). During testing the STOVL LFI door was struck using a real bird and ballistic gel used to simulate soft body impact response. Results from these test variations are compared.

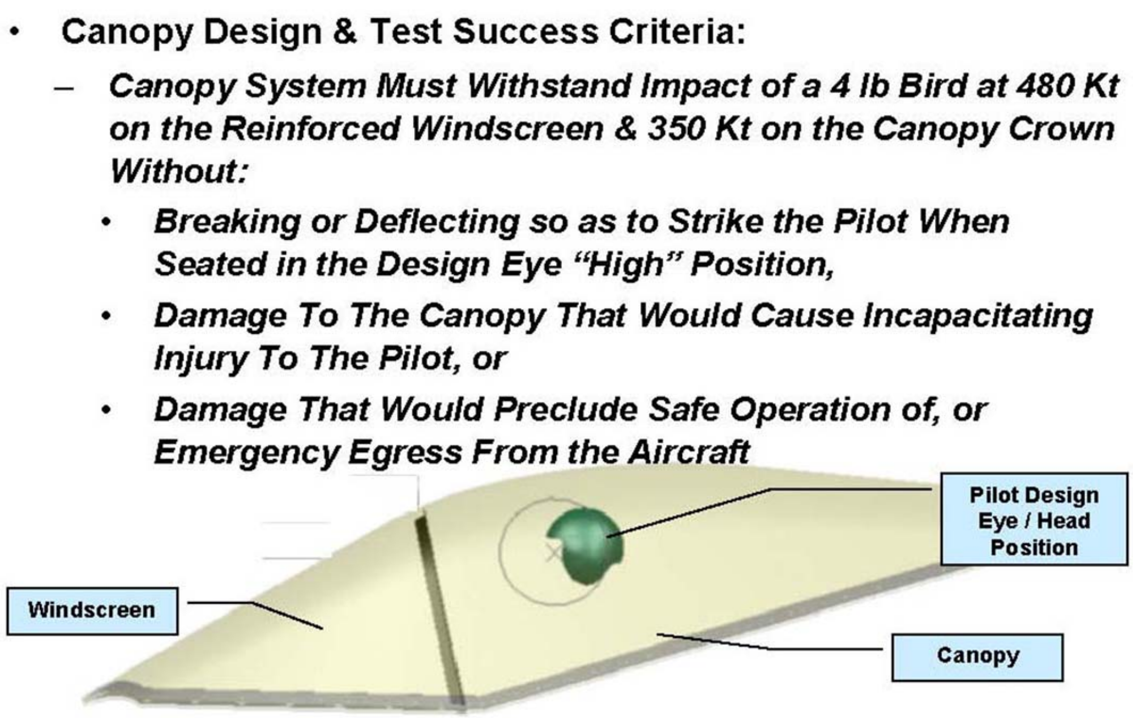


Figure 9.6-29. Canopy Bird Impact Design Criteria

Figure 9.6-30. Bird Impact Test Facility & Setup

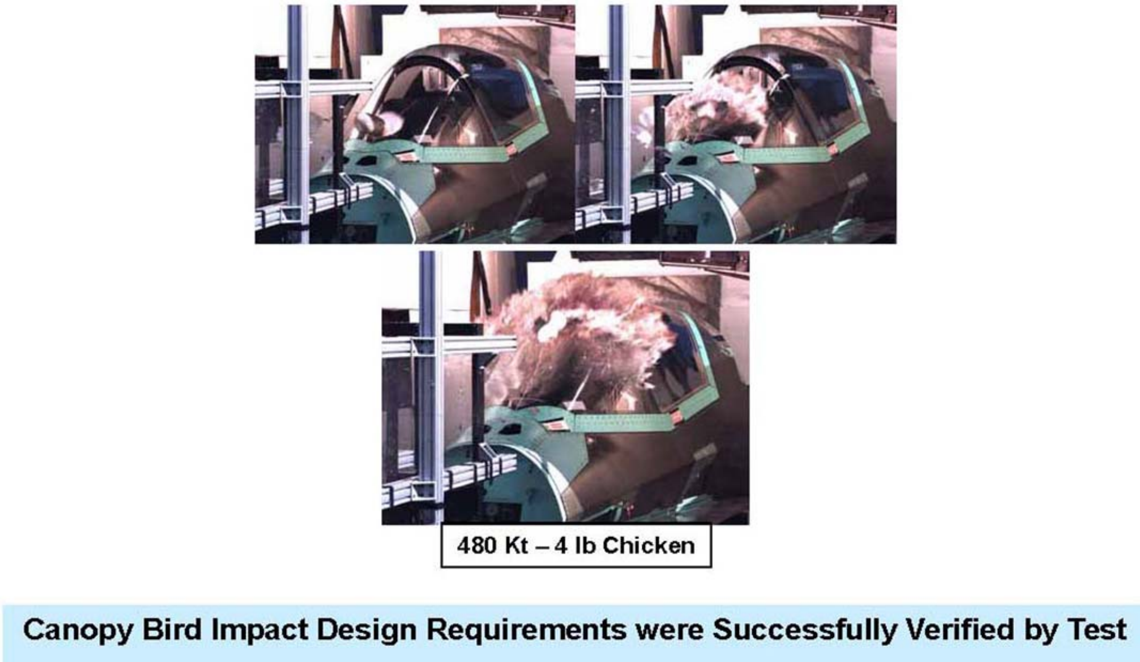


Figure 9.6-31. F-35 STOVL Windscreen Bird Impact Test

Statistical and qualitative analysis techniques were used to assess vulnerability of airframe structure to bird impact. Results of these assessments are described. An example of design changes made early in airframe design development is presented.

Bird impact tests of F-35 airframe and canopy components were successfully conducted as part of the F-35 structures building block test program that is fundamental to ASIP ‘Pillar Two’. These tests, coupled with supporting structures analyses, led to successful verification of F-35 canopy and airframe capabilities to sustain design criteria specified bird impacts without compromise to continued airworthiness.

9.6.12. Defect Assessment: Defect Scanner

Tom Curtin, Computational Mechanics, Inc.

Another major addition to the BEASY fracture analysis capability has been the BEASY Defect Scanner. This is a tool that can be used with either a BEASY model or an FE stress model (ABAQUS, ANSYS or NASTRAN) in order to identify ‘critical’ locations.

The tool creates a “map” of critical crack sizes (Figure 9.6-32) on a model using computed stress values. The critical crack size at any point is the crack size at that point where the SIF value reaches a defined SIF value. This value can be the threshold stress intensity factor in order to establish the minimum size at which cracks may start growing; it can be the critical SIF value in order to determine the size of crack that will cause part of the structure to fracture. The smallest critical crack sizes shown on the map enable users to clearly identify the areas of greatest concern for that component.

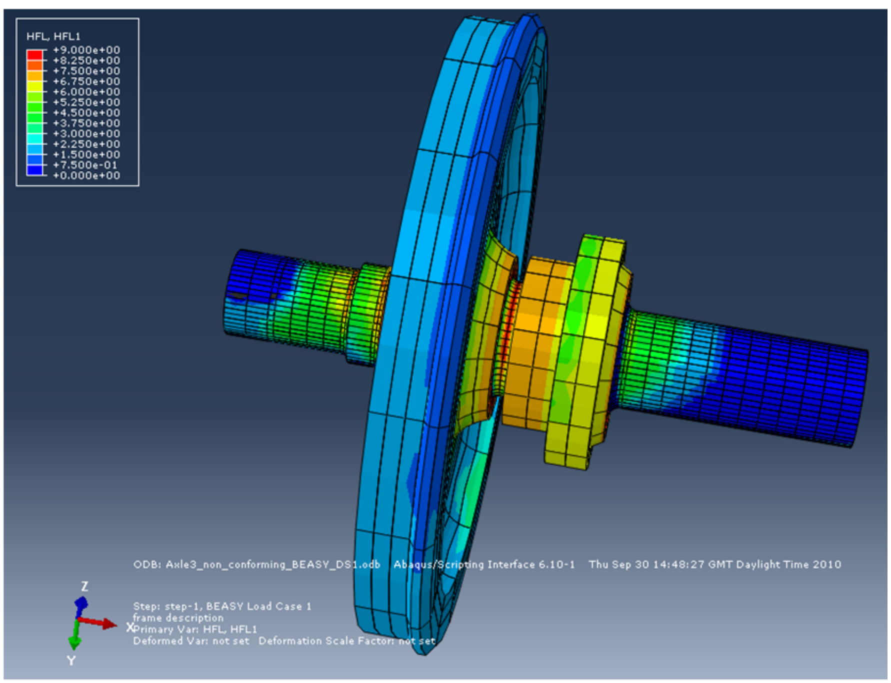


Figure 9.6-32. Contour Map of Critical Crack Sizes

In addition, different ‘regions’ of the model can be defined in order to identify how the critical crack sizes differ between defined parts of the structure. This process could then be integrated with inspection and approval processes (e.g., for incoming parts or reports from maintenance inspections) in order to refine and optimize the decision making process for material rejection/scrappage.

The locations of critical crack sizes identified with this tool can then be used, if required, as an input to a more detailed crack analysis using the BEASY Fracture products.

9.6.13. Simulation Based Corrosion Management of Aircraft Structures

Tom Curtin, Computational Mechanics, Inc.

BEASY is a major contributor to an international research project SICOM which aims to develop computer models capable of simulating corrosion and surface protection measures in aircraft structure (Figure 9.6-33).

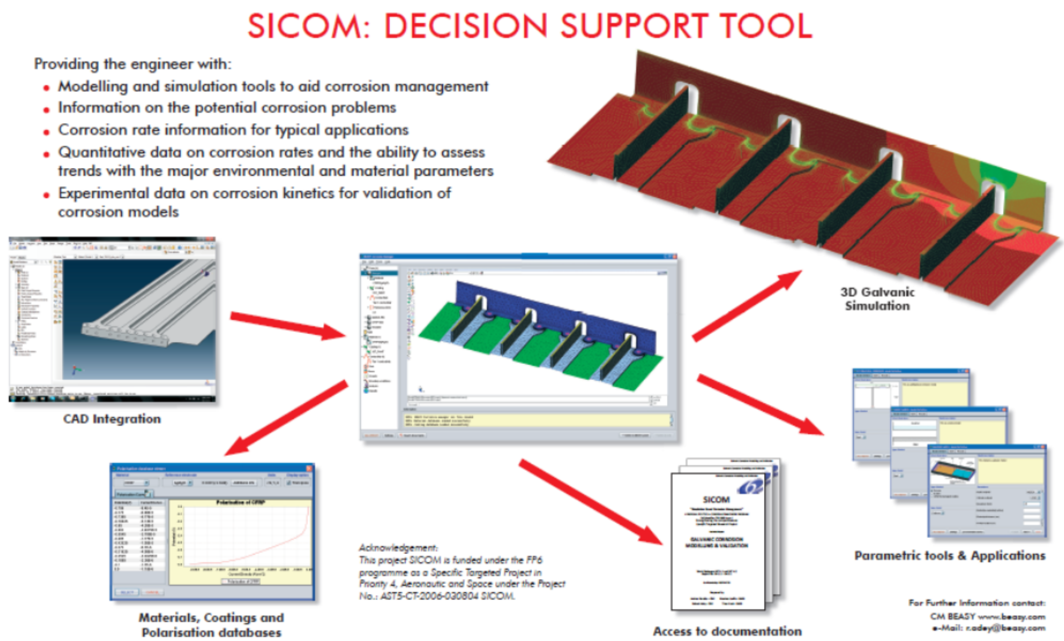


Figure 9.6-33. Decision Support Tool to Assess Surface Protection in Aircraft Structures

Corrosion modelling tools for prediction of corrosion occurrence and corrosion propagation will be a driver for new technical advances in the fields of corrosion maintenance, development of new materials, structural designs and surface protection systems. SICOM will provide models that can become an essential part of future predictive maintenance concepts to avoid unanticipated and unscheduled maintenance with high costs. Data from monitoring systems and non-destructive inspection can be used as model input. Model outputs will be utilized for the repair decision process or can supply structural integrity concepts and thereby fill the gap between monitoring or inspection and calculation of the structural impact of corrosion. Aircraft development costs will be reduced through savings on testing time and quantity.

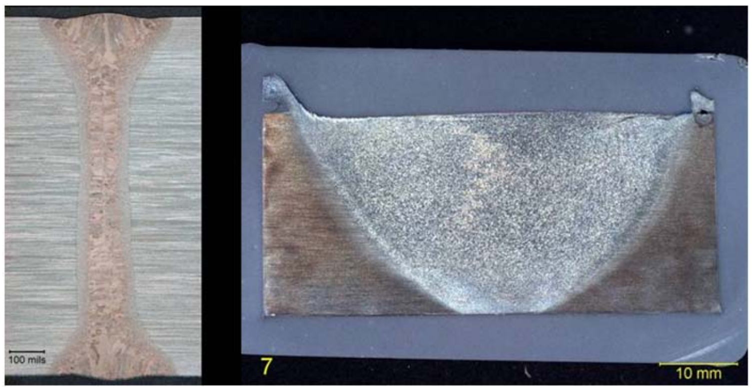
A major development is the Galvanic Corrosion Decision Support system which can be used to model and optimize surface protection systems used in aerospace structure.

9.6.14. Fatigue and Fracture Characterization of Thick Section Ti-6Al-4V Weldments

Paul D. Edwards, The Boeing Company – Research and Technology

Since 2005, The Boeing Company has been developing advanced joining technologies in the Ti-6Al-4V titanium alloy as an alternative to traditional processes in order to reduce the cost and weight of commercial airframe structures. One of the primary challenges associated with implementing such manufacturing processes is the development of reliable durability and damage tolerance design allowables. This will likely be a long and relatively expensive process, but gathering initial screening data is extremely important as it will serve as a starting point for designers to evaluate high value welded components.

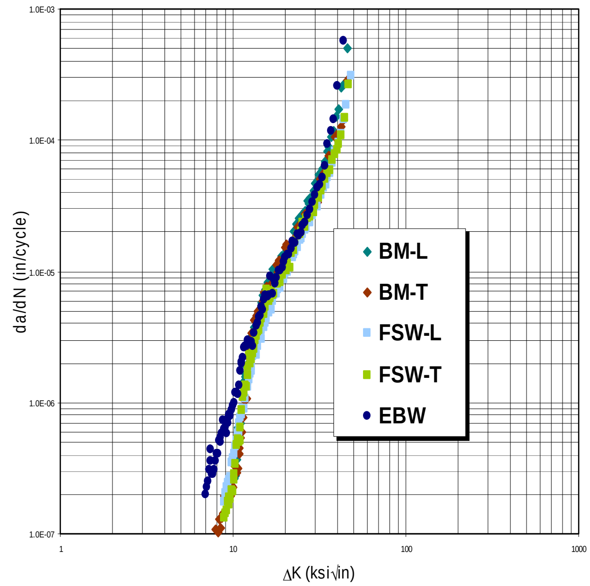
For many primary structural applications, such as side of body cords, longerons, crown frames, etc., relatively thick, approximately 1-in gage, welded pre-forms will be required, which will be finished machined to the net geometry. For this thickness, Electron Beam Welding is a well established process that could be utilized. However, Friction Stir Welding of Ti-6Al-4V has recently been developed and shown to be capable of welding such thicknesses. This solid state process could be capable of providing higher quality, more reliable joints. Thus, the focus of this study was to directly compare the fatigue, fracture and crack growth behavior of 1-in gage Ti-6Al-4V Electron Beam and Friction Stir Welded square groove butt joints, Figure 9.6-34.



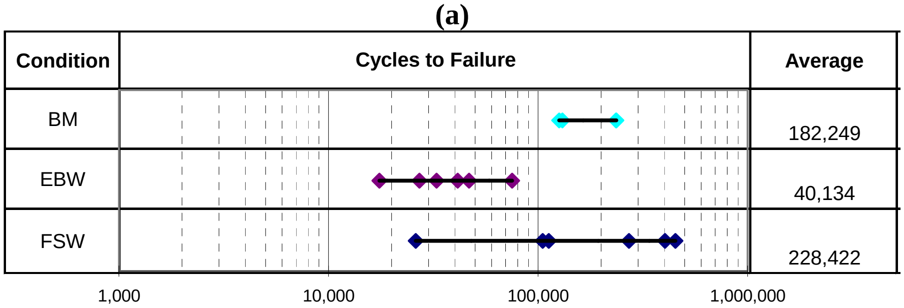
(a) (b)

Figure 9.6-34. 1-in Gage Ti-6Al-4V a) Electron Beam and b) Friction Stir Welded Ti-6Al-4V Square Groove Butt Joints

It was found that the Region II fatigue crack growth rates in both weld joint types were identical to the base metal, Figure 9.6-35 (a). The Friction Stir Welded joints were also identical to the base material in the threshold region, but the Electron Beam welds showed a lower threshold. With respect to fracture toughness, the Friction Stir Welded joints were superior, showing a 40% improvement compared to the base metal. Conversely, the Electron Beam welds resulted in an 11% reduction in fracture toughness relative to the base material. In high cycle fatigue, the Electron Beam Welds showed an inferior fatigue life compared to the base metal. Fracture surface examination showed that this drop in fatigue life was due to sub-surface porosity. The Friction Stir Welds were superior to the base metal in an average sense, but there was a higher degree of scatter, Figure 9.6-35 (b).



(a)



(b)

Figure 9.6-35. a) Fatigue Crack Growth and b) High Cycle Fatigue Behavior of Friction Stir and Electron Beam Welds Along with the Respective Base Material

NOTE: All fatigue tests were conducted on Kt = 1.0 specimens at a constant load of 90ksi under a load ratio of R = -0.2

Overall the fatigue testing was somewhat inconclusive since fatigue is difficult to characterize with such a small number of samples. However, based on these initial screening results, the Friction Stir Welding process is capable of producing a higher performance joint in fatigue, crack growth threshold behavior and fracture toughness compared to the Electron Beam Welds and in some cases even the base material. The challenge now is establishing general design allowables for these processes. Unfortunately, testing of coupons only truly represents coupons and these results can not necessarily be directly applied to complex structures. Thus, the next step in this development effort is to begin bridging the gap between coupon testing and structural certification utilizing numerical simulations and validating with experimental evaluations of welded structures.

Point of Contact:

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9.6.15. Reformulation of Fatigue: From Cycle-Based Approach to Time-Based Approach

Yongming Liu, Clarkson Unviersity

The fatigue damage accumulation process is a multi-scale phenomenon, which involves different spatial and temporal scales. Earlier studies of fatigue analysis can be traced back to about 150 years ago for the stress-life approach (i.e., S-N curve or Wholer curve), in which the cyclic stress range is correlated with fatigue life. 100 years later, Paris proposed that the fatigue damage can be analyzed by crack growth in which the crack growth rate per cycle is correlated with the stress intensity factor (SIF) range ΔK. Historically, fatigue is most commonly analyzed using this cycle-based formulation where the cyclic stress/SIF range and number of cycles are used to correlate and predict fatigue damage.

The cycle-based formulation caused many intrinsic difficulties in the classical fatigue theory. Many research topics are actually related to the definition of fatigue in cycles. They are discussed below. Cycle-counting requirement is one of the intrinsic difficulties in the fatigue analysis. A realistic time history of loading has to be transformed to a cycle history for the fatigue analysis. This transformation introduces additional uncertainties since not all information are transformed during the cycle counting. This issue becomes more complicated under general multiaxial loadings, where a well defined cycle does not exist even under the nonproportional constant amplitude loading. Stress ratio effect is another example of the intrinsic difficulties of the cycle-based approach. Many empirical and theoretical approaches have been proposed and are still ongoing to solve the stress ratio effects. If the cycle-based approach is used, cyclic range of mechanical driving force (e.g., stress, strain, and SIF) is not sufficient to describe the stress state of the material and another degree-of-freedom has to be used. This issue cannot be solved following the cycle-based fatigue formulation.

Recent in-situ scanning electron microscopy (SEM) testing [1] and a preliminary theoretical study [2] show the feasibility of a time-based formulation for fatigue crack growth analysis. The key idea is to formulate an instantaneous fatigue crack growth function within a loading cycle (i.e., da/dt) rather than the averaged crack growth rate per cycle (i.e., da/dN). In this time-based formulation, no cycle counting is required and the stress ratio effect disappears by definition. In-situ SEM testing shows a non-uniform crack growth and multiple crack growth mechanisms within a loading cycle for Al-7075-T6. A schematic plot is shown in Figure 9.6-36 for the multi-mechanism crack growth in the sub-cycle scale. During the initial loading, no crack growth is observed due to the crack closure mechanism. Next, a fast crack growth is observed due to the brittle-like crack growth behavior right after the crack opening. Following this, a slower crack growth rate is observed due to the crack blunting and plasticity development. Finally, if loading is increased again, microcrack development ahead of the major crack will accelerate the crack growth and eventually break the specimen. This multi-mechanism fatigue crack growth bahavior within one cyclic loading cannot be captured using the cycle-based approach, where the average crack growth per cycle is used.

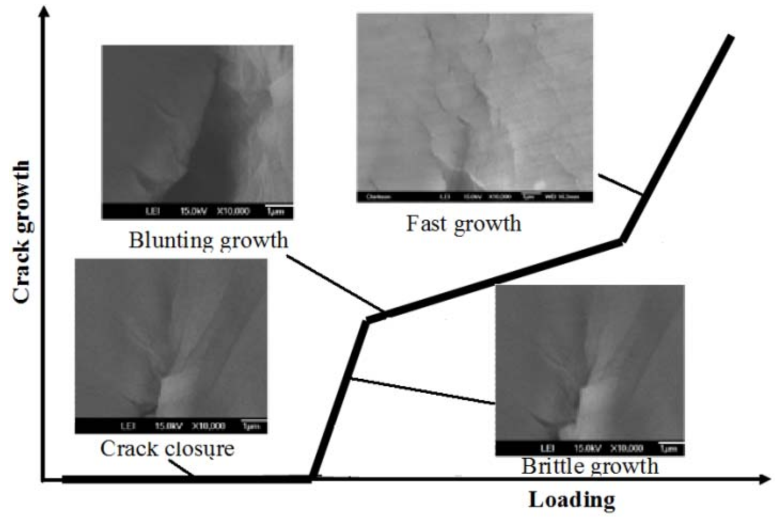

Figure 9.6-36. Schematic Illustration of Different Growth Mechanisms in One Loading

Imaging analysis is used to measure the crack growth kinetics within one loading cycle. An example is shown in Figure 9.6-37. It shows the direct observation of crack growth and CTOD variation behavior within one loading cycle.

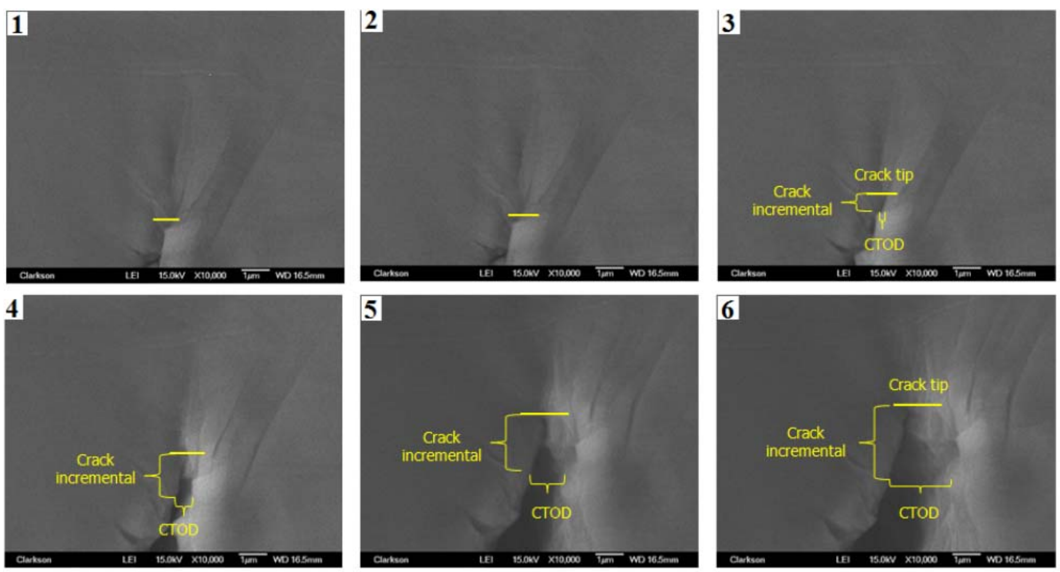

Figure 9.6-37. Illustration of Crack Increment and CTOD Variation Under Different Loading Levels

Detailed digital measurements for multiple specimens [1] revealed that a generalized time-based crack growth law can be developed as

$$\dot{a} = H\left(\dot{K}\right) \cdot H\left(K - K_{op}\right) \cdot \frac{f(K_{\max}, K)}{E\sigma_y}\dot{K} \tag{1}$$

where $\dot{a}$ is the instantaneous crack growth rate, $\dot{K}$ is the applied load changing rate, f is a generic kernel function describing the crack growth rate with respect to the crack tip opening displacement change, and f is a function of the applied maximum SIF ($K_{\max}$) and the current SIF level (K). A bilinear or a power law function is identified in [1] for Al-7075-T6. H is the Heaviside step function and

$H(x) = \begin{cases} 1, & \text{if } x > 0 \\ 0, & \text{if } x \le 0 \end{cases}$. K_{op} is the opening stress intensity factor level where the crack opens and begins to grow. The time-based formulation concept has been used in a preliminary theoretical study [2] to predict the crack length under different constant and variable-amplitude loadings and a satisfactory result has been observed.

Although the time-based formulation is still in its immature stage compared to many existing cycle-based methodologies, the author feels that the change of mathematical formulation of fatigue damage from the cycle-based approach to the time-based approach offers a new, alternative, and systematic way of examining many issues in fatigue.

Point of Contact:

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References:

[1] Zhang, W. and Y. Liu, Investigation of Incremental Fatigue Crack Growth Mechanisms Using In-situ SEM Testing, International Journal of Fatigue, In Press, doi:10.1016/j.ijfatigue.2011.03.004.

[2] Lu, Z. and Y. Liu, Small Time Scale Fatigue Crack Growth Analysis, International Journal of Fatigue, 32(8): p. 1306-1321.

9.6.16. Residual Stresses and Fatigue Growth Test Methods on 7050 Aluminum Alloy

John Bakuckas, FAA – William J. Hughes Technical Center

Residual stresses inherent in the material due to the forming and/or machining process greatly affect fatigue-crack-growth rates and fracture data determined from a laboratory test specimen. On any fatigue-crack-growth or fracture test, efforts should be made to assess the presence or absence of these residual stresses, so that the test data can be corrected for the presence of residual stresses or it can be determined that the test data was not affected.

Previous testing of 7050 aluminum alloy standard compact, C(T), specimens indicated that this material showed very large differences between the ASTM standard load-shedding method and the compression precracking (CP) test procedures in generating fatigue-crack-growth (FCG) rate data. Two CP test methods, compression precracking-constant amplitude (CPCA) and compression precracking-load reduction (CPLR), produced lower thresholds and faster FCG rates—constant-amplitude data with no (or minimal) load history effects. However, some issues were raised on whether residual stresses due to CP and/or from forming/machining that were present in the specimens caused these differences. In an effort to resolve these issues, an FCG test was conducted on a C(T) specimen using the “on-line” crack-compliance method to evaluate stress-intensity factors due to residual stresses. This test would give an opportunity to further evaluate the extent of the CP affected region and to determine whether residual stresses from forming and machining the specimens were present during the FCG tests.

Figure 9.6-38 shows the results of a compression precracking constant-amplitude (CPCA) and compression precracking constant-stress-intensity-factor (CPCK) test to (1) determine the extent of the residual stresses from CP loading on a C(T) specimen made of the 7050 alloy and (2) determine whether there was forming or machining residual stresses in the specimens. Fracture Technology Associates’ commercially available crack monitoring system and the crack-compliance method, which uses the upper slopes on differential load-displacement (or BFS) records, was used to calculate stress-intensity factors due to residual stresses. The first part of the test (indicated by blue data points) was CPCA loading and the second part (indicated by red data points) was CPCK loading. The acceptance of the CPCA and CPLR test procedures has been hampered by the fact that the presence or absence of residual stresses had not been confirmed. Figure 9.6-38 shows a schematic of the machined notch and the compressive plastic-zone size at the notch tip. The results show that the 3 plastic-zone criterion (vertical dashed line) is satisfactory; and that this material and the C(T) specimens did not have any significant residual stresses. Thus, the C(T) specimens did not have any significant forming and machining residual stresses and, thus, the differences shown were due to the test methods, as expected.

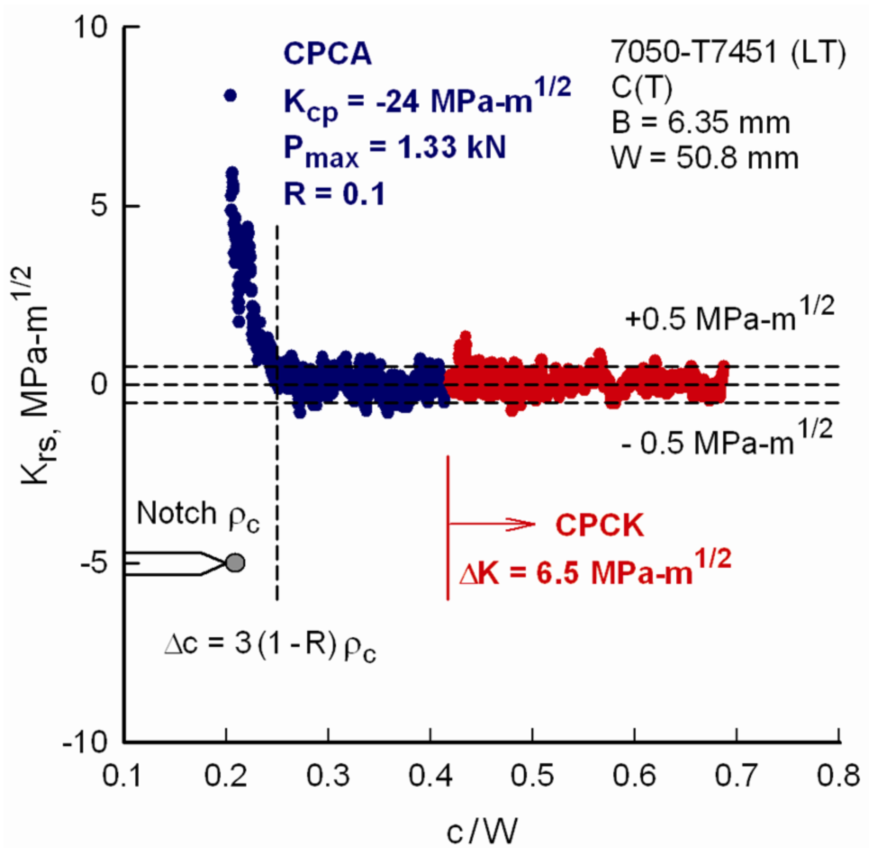

Figure 9.6-38. Residual Stress-Intensity Factors on a C(T) Specimen Made of 7050 Aluminum Alloy

Points of Contact:

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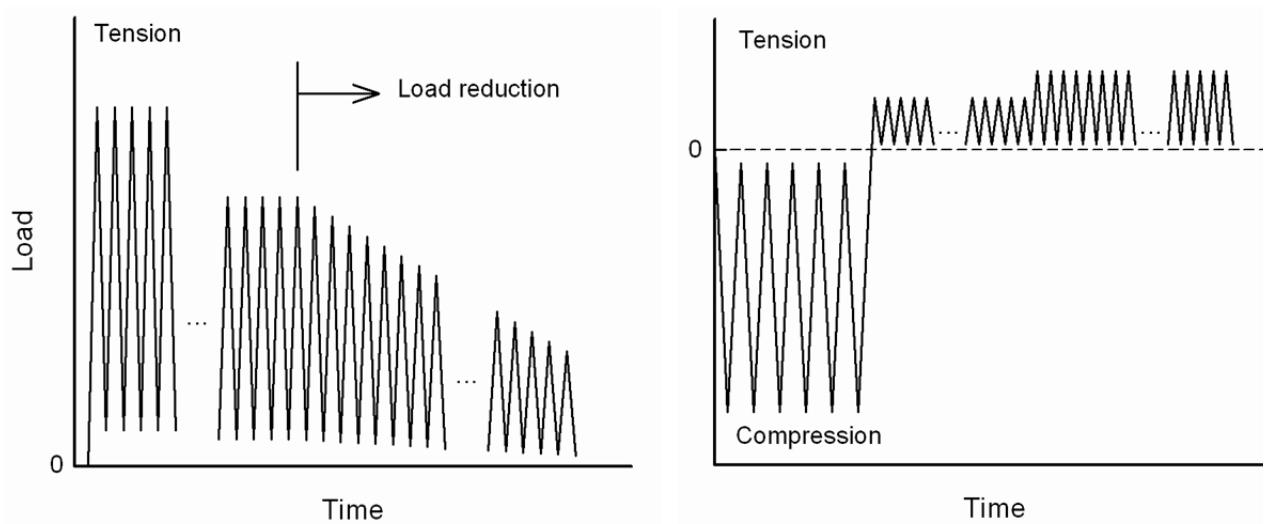
- Traci Stadtmueller, FAA William J Hughes Technical Center, 609-485-4768
- John Bakuckas, FAA William J Hughes Technical Center, 609-485-4784

9.6.17. Load-Reduction Threshold Testing Causes a Width Effect

James Newman, Jr., Mississippi State University

Accurate representation of fatigue-crack-growth thresholds is extremely important for many structural applications. Presently, in the United States, the threshold regime is experimentally defined by the ASTM E-647 load-reduction (LR) test procedure. Tests have shown a rise in the crack-closure levels as the threshold conditions are approached using the LR method. This behavior was attributed to plasticity-, roughness- and/or fretting-debris-induced crack-closure effects. Analyses have also shown a rise in the crack-closure level using strip-yield and finite-element models, which showed that the LR test method exhibited anomalies due to load-history effects.

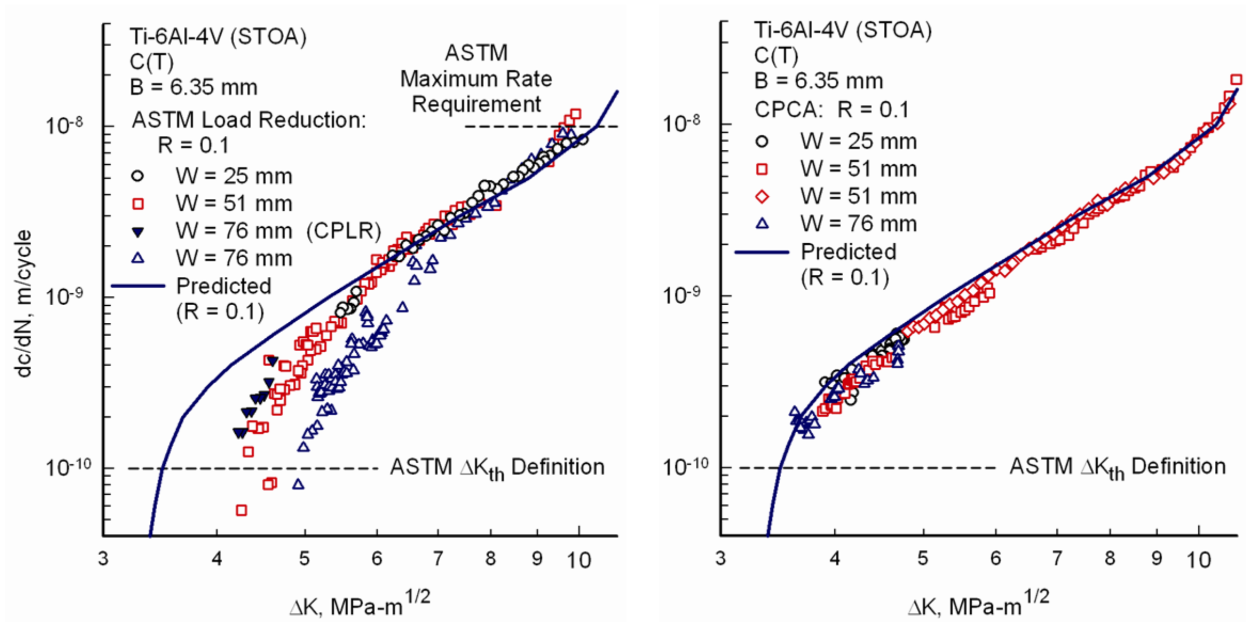
Fatigue-crack growth rates in the threshold and near-threshold regimes for a titanium alloy, Ti-6Al-4V (STOA), were determined using two test methods: (1) ASTM E-647 load-reduction procedure (Fig. 9.6-39 (a)), and (2) the compression precracking constant-amplitude (CPCA) test method (Figure 9.6-39 (b)). In the current load-reduction method, tensile loads are used to initiate a crack at a starter notch but loads are reduced before the load-reduction procedure is used. In contrast, the CPCA test initiates a crack at the starter notch by compression precracking (CP) and then conducts constant-amplitude (CA) loading. If the crack does not grow, then the applied loading is slight increased until the crack starts to grow and the CA loading is then held constant. Tests were conducted over a wide range of stress ratios (R = 0.1 to 0.7) on compact specimens made of three different widths (25- to 76-mm).



(a) ASTM Load-Reduction Sequence **(b) CPCA Loading Sequence**

Figure 9.6-39. Types of Loading Applied to Fatigue-Crack-Growth Specimens

Test data at R = 0.1 for the ASTM LR method are shown in Figure 9.6-40 (a). These data show a “fanning out” of data at the lower growth rates as a function of specimen width (W). These results reveal a very disturbing trend, but these data were very similar to those presented by Garr and Hresko on Inconel-718 (ASTM STP-1343), which showed a width effect on threshold behavior using the ASTM LR method. A CPLR (compression precracking followed by load reduction) test was conducted on another 76-mm wide specimen that produced much faster rates than the ASTM LR test on the same specimen width. This behavior was unexpected, since the maximum rate for the CPLR procedure was a factor-of-20 lower than the ASTM allowable rate requirement. But these results may indicate that this titanium alloy is very sensitive to load reduction; and caution must be used whenever LR procedures are used. A large amount of the near-threshold data on the 25-mm wide C(T) specimen was eliminated due to the excessive clip-gage force used to monitor crack growth. But the 51-mm test produced a lower threshold than the 76-mm specimen.



(a) ASTM Load-Reduction Method **(b) Compression Precracking Constant-Amplitude Method**

Figure 9.6-40. Crack-Growth Rate Data on Ti-6Al-4V STOA Titanium Alloy at R = 0.1 Using Two Test Methods

In contrast, data for the three specimen widths using the CPCA test method for R = 0.1 loading show drastically different behavior, as shown in Figure 9.6-40 (b), with no "fanning" nor specimen width dependency, as was noted with the E647 LR method. Data for the three specimen widths plotted directly on top of each other over the same range in crack-growth rates examined. The ΔK-rate curve is clearly independent of specimen width and crack length, and the rate is only as a function of the applied ΔK, the key assumption in the LEFM approach to life prediction.

In an effort to explain this behavior, the solid curve is a predicted curve based on the R = 0.7 data (as the ΔK_{eff}-rate curve) using the FASTRAN crack-closure model with a constraint factor (α) of 2. The solid curve in Figure 9.6-40 is the predicted behavior for R = 0.1 using the crack-closure model. At high rates, the predicted results were reasonable, but in the low- and mid-rate regions, the rates were over predicted by about 20 to 30%. Debris and/or roughness may be present in the threshold regime, which may explain the slight discrepancy.

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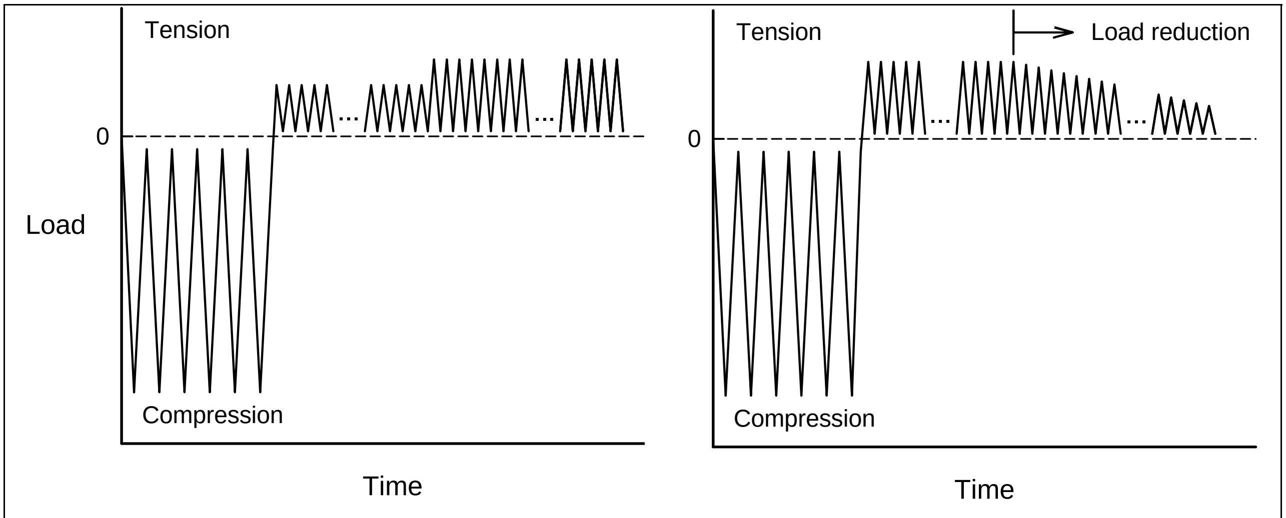
- Work sponsored by the U.S. Office of Naval Research, Dr. A. Vasudevan, Technical Monitor.

9.6.18. Compression Precracking Threshold Testing on 7050 Aluminum Alloy

James Newman, Jr., Mississippi State University

To generate fatigue-crack-growth-rate data in the near-threshold regime, without appreciable load-history effects, compression precracking (CP) methods, developed over the years by a number of investigators, such as Suresh, Pippan et. al. and Newman et. al., were used. Using CP threshold test methods, environmental effects, such as oxide and/or fretting-debris-induced closure, crack-surface roughness, and plasticity effects would *naturally* develop under constant-amplitude (CA) loading conditions. A crack grown under CP loading, as in Figure 9.6-41 (a), is fully open at the start of CA loading. The crack is growing partly because of tensile residual stresses induced by compressive yielding at the crack-starter notch. Currently, trial-and-error procedures are required to select the initial CA loading magnitudes near the unknown threshold value. If a tensile-load range is selected that would produce a stress-intensity factor range below threshold, then the crack may initially grow, but become a non-propagating crack; however, if the load is high enough, then the crack will grow. The crack must be grown *several* compressive plastic-zone sizes before the effects of the tensile residual stresses have decayed and the crack-opening stresses have stabilized under CA loading conditions. This method is called the CPCA threshold test method. A second method is CP, followed by CA loading, and then LR following current ASTM E-647 procedures, except that the initial stress-intensity-factor range and crack-growth rate at the start of LR test is much less than the maximum allowed in the current standard. This method is referred to as CPLR threshold testing and the loading is depicted in Figure 9.6-41 (b).



(a) CPCA Loading	(b) CPLR Loading

Figure 9.6-41. Types of Loading Applied to Fatigue-Crack-Growth Specimens

Figure 9.6-42 shows a comparison of test data generated at R = 0.1 using the ASTM load-reduction (LR) test method. These results show that the LR test method produced higher thresholds (ΔK_{th} at 1E-10 m/cycle) and lower rates than the CPLR test method. The results from NASA Langley Reseach Center (LaRC) also showed some variations for different test specimens in the near threshold regime. This may have been due to the possible influence of load-history effects. Four tests were conducted with 4 different initial ΔK_i values to start the LR tests. The two test results shown by the solid symbols violated the ASTM LR standard (initial rate greater than 1E-08 m/cycle). The two tests at the lowest ΔK_i values satisfied the standard, but produced slightly lower thresholds than the two tests with highest ΔK_i values.

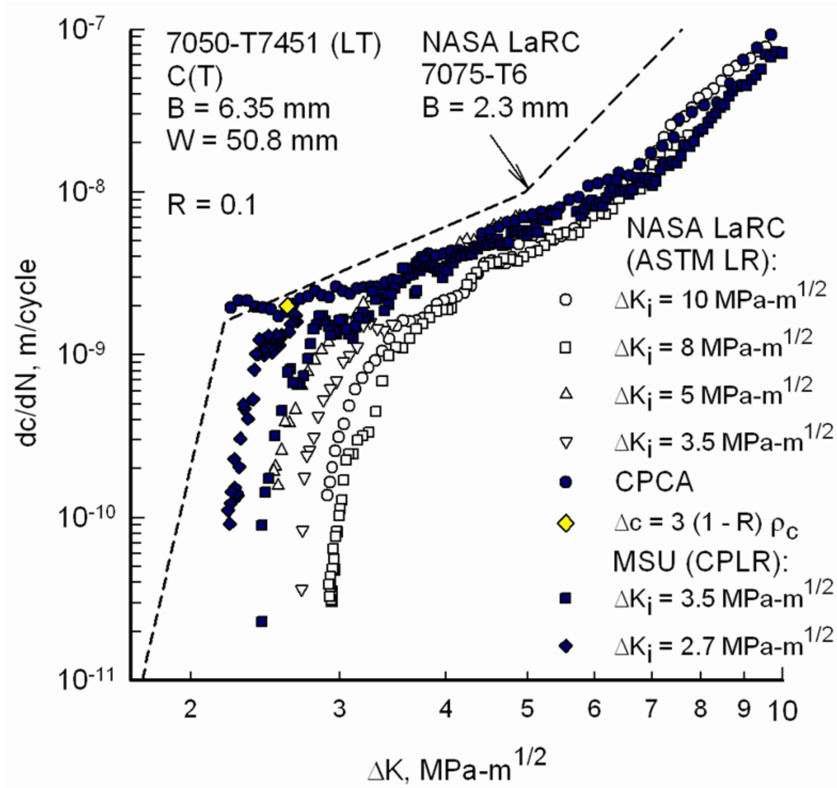


Figure 9.6-42. Crack-Growth Rate Data on 7050 Aluminum Alloy at R = 0.1

At MSU, two specimens were also tested at R = 0.1 using the CPLR test method. After CP loading, one test had a starting ΔK_i level near the lowest value from the LaRC tests. These results are shown as the solid square symbols in Figure 9.6-42. After reaching a very low rate in the CPLR test, a CA test was conducted to generate the upper portion of the curve. These data fell at slightly lower ΔK values than the LaRC tests at the same rate. A second CPLR test had a ΔK_i level of 2.7 MPa√m and these results are shown as solid diamonds. These results fell at even lower ΔK values than the previous CPLR test. It was very surprising that the very low starting ΔK_i levels would still have an effect on the near-threshold results during load reduction. NASA LaRC had also conducted a CPCA test at R = 0.1 on a specimen machined from the same block. The CPCA test results started on the high-R curve because the crack was fully open ($\Delta K = \Delta K_{eff}$), but as the cracks grew the crack-opening-load level rose and the data approached the R = 0.1 data. The solid diamond symbol shows where the crack extension in the CPCA test had reached the crack-growth criterion where the influence of compression pre-cracking tensile residual stresses would have diminished and the crack-opening loads would have stabilized. The lowest CPLR results matched well with the valid data from the CPCA tests. The dashed curve shows the results for a thin-sheet 7075-T6 aluminum alloy also tested at LaRC, which had a very similar shape at R = 0.1.

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- Work sponsored by the U.S. Federal Aviation Administration, Dr. John Bakuckas, Technical Monitor.

9.6.19. 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; and Venkataraman Shivakumar, Randall Christian, and Yajun Guo, Jacobs ESCG

The NASGRO® software for fracture mechanics and fatigue crack growth analysis continued to be actively developed and widely used during 2009 and 2010. 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, 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 began its fourth three-year cycle in 2010. The international participants currently include AgustaWestland, Airbus, Alcoa, Boeing, Bombardier Aerospace, Embraer, Hamilton Sundstrand, 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 Volvo Aero. In addition to Consortium members, 89 single-seat and 6 site NASGRO licenses were issued in 2009-2010 to users in 17 countries.

Two new production versions of NASGRO were released in 2009 and 2010. Version 6.0, released in March 2009, included three new stress-intensity factor (SIF) solutions (Figure 9.6-43). CC12 is a bivariant weight-function (WF) solution for a crack spanning a chamfer at the corner of a plate. EC04 is a bivariant WF solution for an offset embedded crack in a plate, and EC05 is a corresponding univariant WF solution for the same geometry. Existing univariant WF solutions for a single corner, surface, or through crack at an offset hole were extended to treat two symmetric cracks at a centered hole under symmetric stressing. New GUI functionality was added to display the specific toughness values at each crack tip and allow the user to change individual values as desired. The critical crack size module was enhanced to facilitate calculation of the threshold crack size—the largest crack size at which no crack tips will propagate under specified cyclic loading. The material properties module was enhanced to facilitate data processing from specimens with surface cracks or K-gradient load histories.

Version 6.1, released in August 2010, contained a wide variety of new features. Crack case TC16 (through crack in a curved stiffened panel) was enhanced to account for a two-bay crack with a broken stiffener, to allow the Swift bulge factor for 1-bay cracks, and to include the damping effect and non-zero biaxial load ratios for the Chen-Schijve model. A capability to input and use separate stress gradients for tension and compression loads was implemented for most WF SIF solutions. SIF compounding features were substantially enhanced to permit compounding for all through cracks, all corner cracks, and most surface cracks; multiple compounding tables can be specified and superimposed for each stress component (tension, bend, etc.); compounding factors are passed on to post-transition geometries; and all compounding factors are available in post-processing plotting. A new set of additional (optional) failure criteria including plastic limit load, failure assessment diagram (FAD) according to FITNET Options 1 and 3, and the Newman Two-Parameter Fracture Criterion, was developed and implemented for commonly used crack cases. Seven new sets of fatigue crack growth rate data were added to the material properties module. Many other new features and bug fixes were also included in both Versions 6.0 & 6.1.

Significant progress was achieved on the development of NASGRO 6.2, with Alpha release scheduled for early 2011, and Production release later in the year. New features completed include new SIF solutions (Figure 9.6-43) 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. Tabular fatigue crack growth rate data capabilities were completely revised and enhanced to add new interpolation, threshold and instability options; full GUI plotting; and increased capacity. An improved temperature interpolation scheme was also implemented. New solution algorithms with improved robustness and accuracy were implemented for inverse calculation modes (e.g., compute initial crack size given target life). 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. Fatigue crack growth data for seven aluminum alloys were added to the material properties module.

Southwest Research Institute has been conducting NASGRO training courses since 2006. During 2009 and 2010, SwRI trained 194 students in 12 courses, including 4 courses in San Antonio, Texas, and 8 courses at remote sites including NASA Centers, NAVAIR facilities, the ESA Technical Center in the Netherlands, and aerospace companies in two ICAF countries.

Further information about NASGRO is available at www.nasgro.swri.org.

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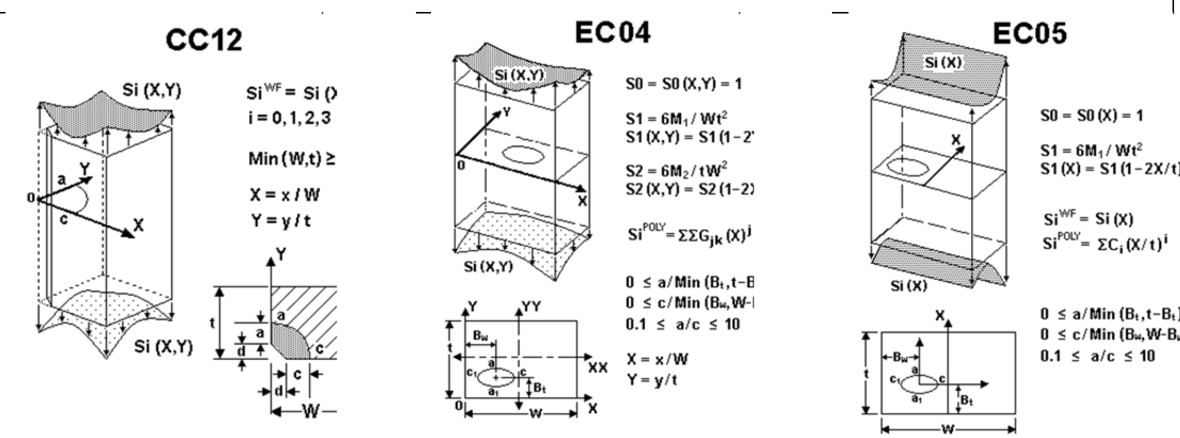
New Stress Intensity Factor Solutions in NASGRO 6.0



New Stress Intensity Factor Solutions in NASGRO 6.2

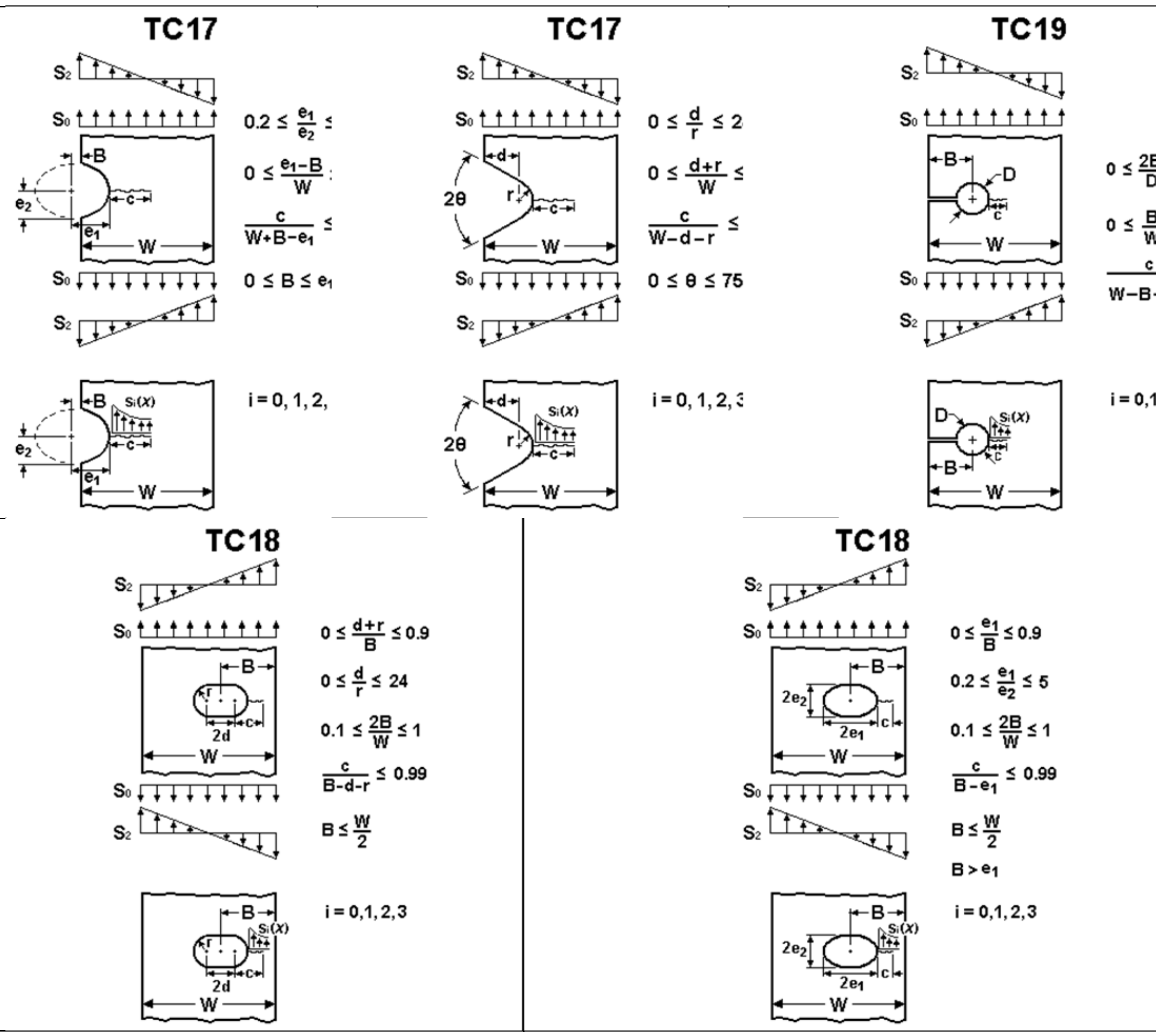


Figure 9.6-43. New Stress Intensity Factor Solutions in NASGRO Versions 6.0 and 6.2

9.6.20. Continued Development of the DARWIN Software for Probabilistic Damage Tolerance Analysis and Risk Assessment

Craig McClung, Michael Enright, Wuwei Liang, Yi-Der Lee, and Jonathan Moody, Southwest Research Institute; and Simeon Fitch, Mustard Seed Software

Aircraft gas turbine engine components may contain rare anomalies that can potentially lead to uncontained failure of the engine. Several Federal Aviation Administration (FAA) Advisory Circulars have been recently introduced to address life-limited engine parts (AC 33.70-1), including manufacturing-induced surface damage anomalies at machined holes (AC 33.70-2) and titanium rotors with inherent material anomalies (AC 33.14-1). The associated risk of fracture can be predicted using DARWIN®, an award-winning probabilistic fracture mechanics software code developed by Southwest Research Institute® under FAA funding. DARWIN can also be used for conventional deterministic damage tolerance analysis. The software uniquely links engineering fatigue crack growth analysis and probabilistic assessment with 2D and 3D finite element models through a powerful user-friendly graphical user interface (GUI). DARWIN has been under development since 1995 and has been licensed since 2000. Current commercial licensees include eight gas turbine engine companies and two government laboratories in seven countries.

Two new versions of DARWIN—7.0 and 7.1—were released during 2009 and 2010.

The DARWIN graphical user interface (GUI) significantly reduces the time required to assess the risk of fracture, but the accuracy of the assessment is still dependent on the skill and judgment of the analyst. A new algorithm was implemented in DARWIN 7.0 that automatically determines (without user input) the orientation, size, and stress input for a fracture model that will produce accurate life results, given only the finite-element model and the initial-crack location. Additional algorithms are planned for future versions to automate the generation of probabilistic fracture mechanics models.

DARWIN 7.0 includes several new stress-intensity factor (SIF) solutions for improved assessment of fracture risk. Figure 9.6-44 shows a new SIF solution for an edge through crack in a variable thickness plate. New univariant and bivariant weight function SIF solutions for an embedded crack in a plate are also included in DARWIN 7.0.

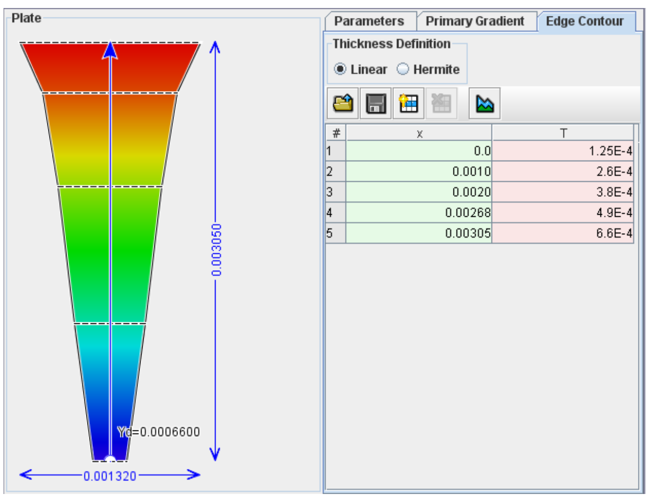


Figure 9.6-44. DARWIN 7.0 Includes a New SIF Solution for an Edge-Through Crack in a Variable-Thickness Plate

DARWIN 7.0 also includes several new features for assessment of risk associated with surface damage. As shown in Figure 9.6-45, the GUI was enhanced to provide visualization of blade slot surfaces in 3D finite-element models. Risk assessment of turned surfaces in 2D models was enhanced to include treatment of stress concentrations such as hole features. A new capability was implemented to allow the user to apply manufacturing process credits (defined in AC 33.70-2) to surface damage risk assessment results. DARWIN 7.0 also includes a new report form that provides the essential assessment data applicable for an FAA review.

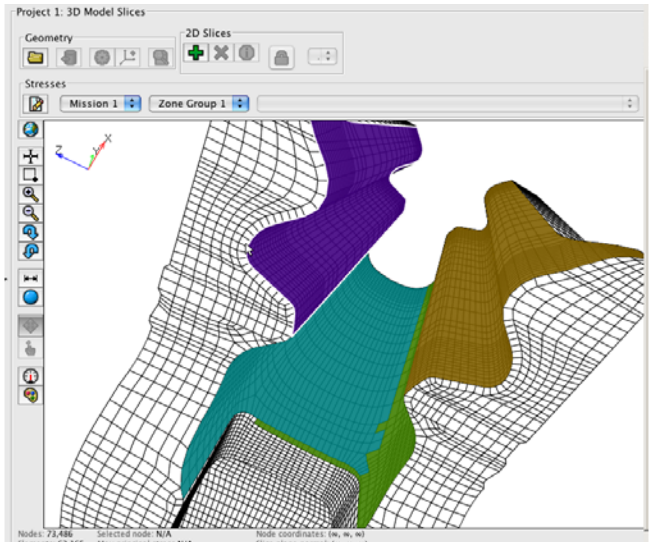
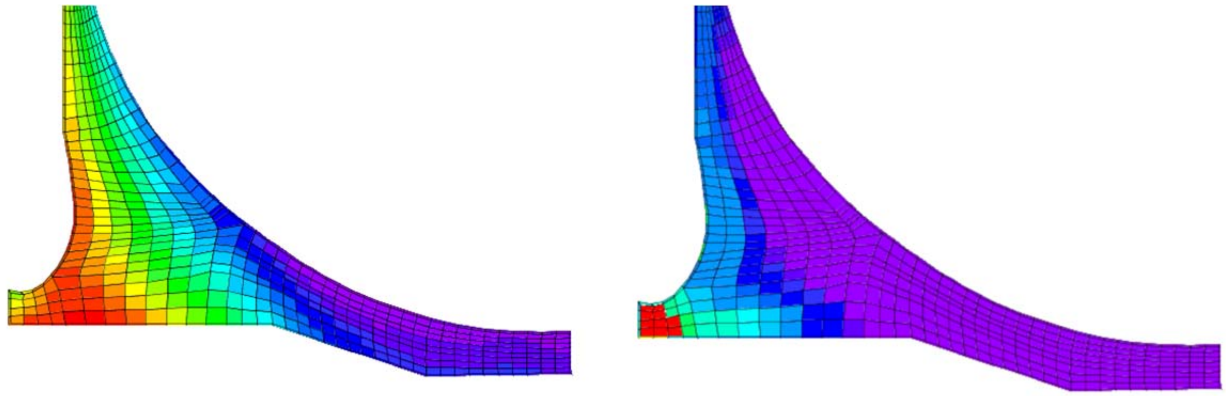


Figure 9.6-45. DARWIN was Recently Enhanced to Provide Visualization of Blade Slot Surfaces for Surface Damage Risk Assessment

A number of other general enhancements were provided in DARWIN 7.0, including bivariant shakedown, enhanced filtering of finite-element models, improved crack growth life interpolation, enhanced stress gradient search, and improved display of GUI warning messages.

A new capability for automatic generation of life contours was developed in DARWIN 7.1 for application to 2D finite-element models (Figure 9.6-46). When the user executes this option, an anomaly of one or more user-specified sizes is automatically placed at each of the nodes in the finite-element model. The automatic geometry model process (introduced in DARWIN 7.0) generates a fracture model at each anomaly location, and then the fatigue-crack-growth (FCG) lifetime to failure is computed for each model. The resulting family of calculated life results is displayed in the GUI using conventional contouring methods, as is often done for stresses. The figure demonstrates that the stress hot spots (red regions of high stress) do not correspond exactly with the life hot spots (regions of low FCG life), due to the additional geometry factors that influence FCG life.



(a) Hoop Stress Contour Plot for a 2D Impeller Geometry

(b) Corresponding DARWIN Life Contour Plot

Figure 9.6-46. DARWIN Version 7.1 Includes a Capability for Automatic Generation of Life Contours

A new HCF Threshold Check capability was introduced in Version 7.1 that allows the user to include the influences of vibratory (HCF) stresses in fatigue crack growth and fracture risk computations. Fracture is assumed to occur when the stress intensity factor associated with the HCF stress exceeds the HCF fatigue crack growth threshold value. The HCF threshold check is performed once per mission in conjunction with the application of the peak LCF stress in the mission.

In 1991, the Aerospace Industries Association (AIA) created the Continued Airworthiness Assessment Methodologies (CAAM) Committee to develop methods to resolve safety-related problems associated with engines and auxiliary power units installed on transport aircraft. The CAAM Committee developed FAA Advisory Circular 39-8 which describes a process for characterizing and assessing the risk associated with safety-related events. It defines events and specifies risk factors and per-flight target risk values associated with specific hazard levels. A new Fleet Assessment Tool was introduced in DARWIN 7.1 that can be used to assess the risk associated with safety-related events described in AC 39-8. As illustrated in Figure 9.6-47, the initial version can be used to predict risk factors and per-flight target risk values associated with an aircraft fleet. Future versions will provide additional capabilities such as the influence of corrective actions on overall fleet risk.

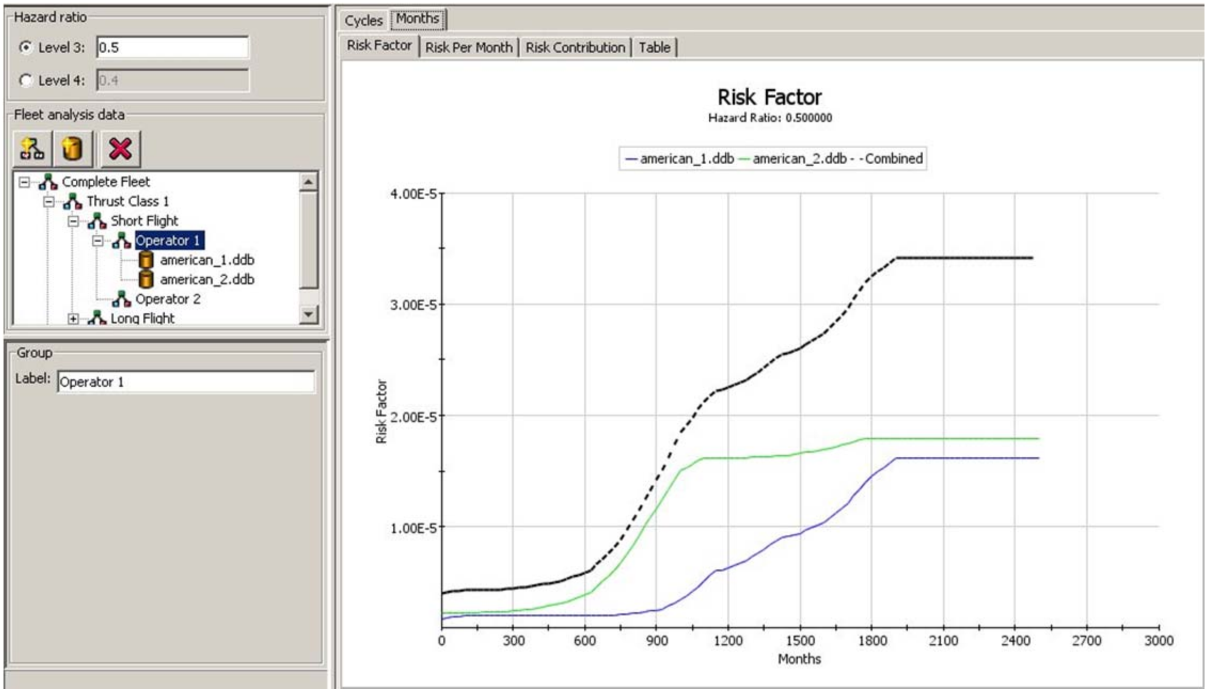


Figure 9.6-47. A New Fleet Assessment Tool was Introduced in DARWIN 7.1 that can be Used to Assess the Risk Associated with Safety-Related Events Described in AC 39-8

More information about DARWIN is available at www.darwin.swri.org.

Point of Contact:

- Michael Enright, Southwest Research Institute, Michael.enright@swri.org, 1-210-522-2033.

9.6.21. Evaluation of Remaining Life and the End-of-Life Residual Strength of the F/A-18 Inner-Wing Step-Lap Joint

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

The F/A-18 wing root structure consists of an AS4/3501-6 carbon/epoxy composite stepped-lap joint bonded with FM-300 film adhesive to a 6Al-4V titanium-splice fitting. 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 aircraft service and to evaluate the service life remaining based on the usage history. Spectrum loading representing fleet usage is used for cyclic testing to determine the remaining life of the step-lap joints. This effort also supports the life-extension efforts without additional large-scale testing to determine 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 8 decommissioned wing skins. 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 (TDFS and CDFS, respectively) that contained 6,000 spectrum fatigue hours per lifetime. Fatigue loads were enhanced by load severity factors ranging from 1.15 to 1.60. All runout specimens were evaluated for residual strength (Figure 9.6-48). Figure 9.6-49 summarizes the fatigue and residual strength of runout specimens. Both static and fatigue results indicate that the service history including the environmental exposure has not degraded the structural integrity of the bonded step-lap joint.

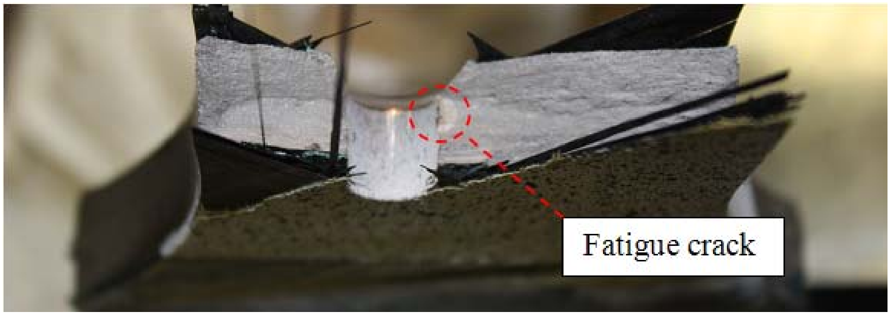


Figure 9.6-48. Fatigue Damage at Fastener Hole (Inspections After Residual Strength Test)

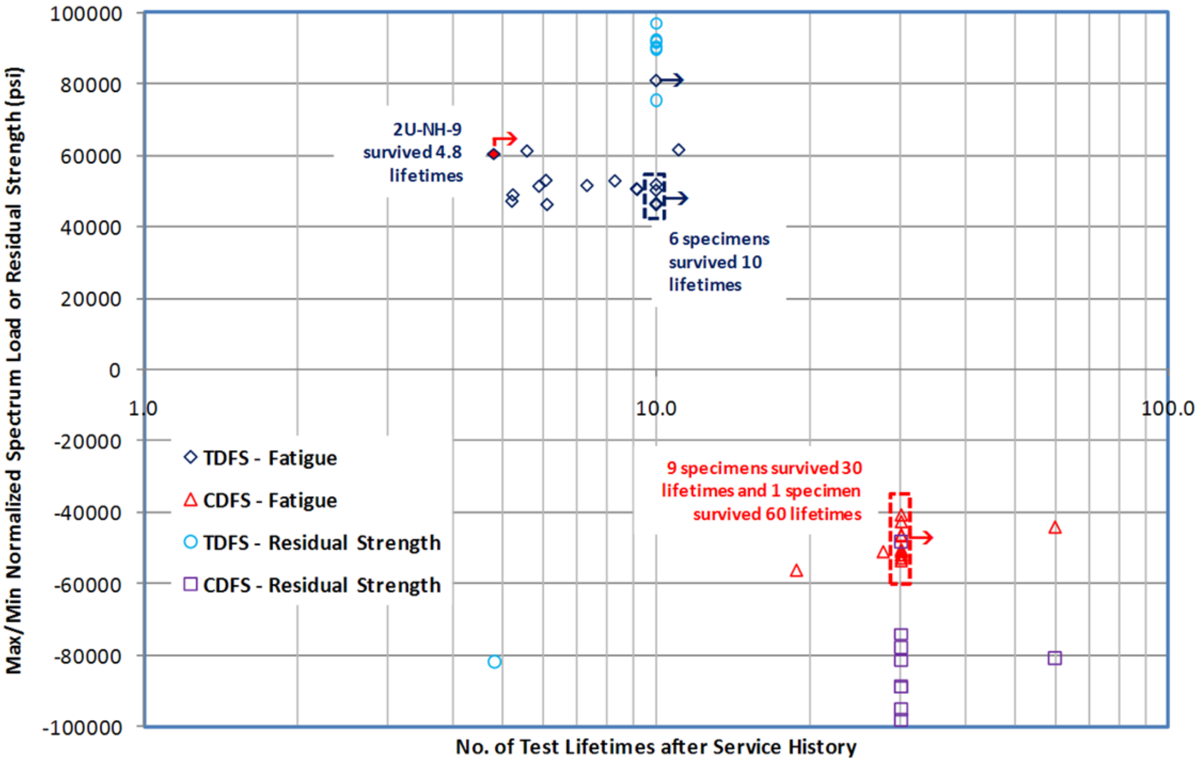


Figure 9.6-49. Progressive Damage Growth Initiated as Fatigue Failure in Titanium

Detailed inspections were carried out during fatigue testing for capturing the progressive failure (Figure 9.6-50) and following residual strength testing for failure analysis (Figure 9.6-48). Further, the effects of extreme exposure to salt-fog environment will be investigated using a similar specimen configuration as described in this technical effort.

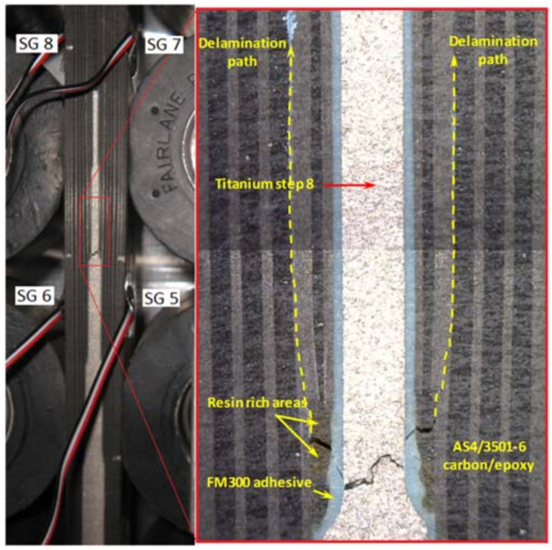


Figure 9.6-50. Progressive Damage Growth Initiated as Fatigue Failure in Titanium (Survived Approximately 5.6 Lifetimes with 1.45 Load Severity Factor)

9.6.22. The Effect of Chromate on Small-Scale Fatigue Crack Growth

Sarah Galyon Dorman, USAF Academy – CAStLE

Many aircraft service-related issues are caused by the presence of corrosion damage. Corrosion can cause damage in many ways, from causing pits to reducing the fatigue life of parts. Corrosion mitigation systems are used to limit corrosion damage on aircraft. Often for the United States Air Force (USAF), these systems include chromate containing coatings, typically chromate conversion coatings and primers. Chromate has environmental and personnel concerns that have caused the USAF to pursue newer, greener coatings. Chromate has been used for many years on military aircraft for corrosion prevention, but its effects on fatigue crack growth are not fully understood and are not considered in current damage prognosis programs. Chromate ions have shown beneficial effects on long fatigue crack (larger than 1 mm) propagation; if similar benefits are achieved by chromate primers on small-scale damage (less than 1 mm), unforeseen loss of fatigue life could occur when chromate-containing coatings are replaced with green coatings. The Center for Aircraft Structural Life Extension (CAStLE) at the USAF Academy has undertaken work to determine the effect of chromate coatings on small-scale fatigue damage. Quantifying the chromate effect at the small-damage scale is necessary to provide a baseline to compare the efficacy of new and environmentally safer coatings on fatigue life.

CAStLE is focusing on the effect of chromate primers on crack growth under the damage tolerant flaw size (less than 1 mm). An environmental-fatigue-test method was used to mimic actual aircraft conditions as accurately as possible while maintaining simplicity and rigor of laboratory experiments. All testing was completed on a legacy aluminum alloy (AA) and temper (7075-T651) that is still on some older aircraft today. This legacy alloy has known issues with corrosion and therefore presents a typical worst case scenario for corrosion mitigation. Tests are being conducted to compare the results from bare AA7075-T651 specimens and chromate primer coated AA7075-T651. Different environments, including dry/humid air and 0.06 M salt solution (NaCl) are being examined in this study. Figure 9.6-51 shows some results from initial tests. Note that the primered samples in pure water without the low level of chloride ions in the 0.06 M NaCl show a slower fatigue crack growth rate than the bare samples 0.06 M NaCl. It is possible that the slower crack growth rate is an effect of the chromate primer, assuming the corrosion fatigue is hydrogen driven and not chloride driven. More testing is needed to determine the chloride/hydrogen relationship as it relates to chromate-containing coatings.

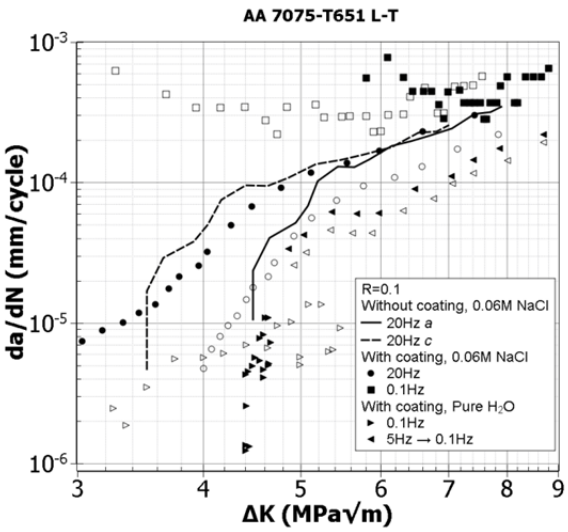


Figure 9.6-51. Crack-Growth-Rate Curves for Coated and Bare AA7075-T651 Samples

Another area of interest for the program has been with recent developments in the application of thin salt films to the surface of a sample. Full-immersion corrosion-fatigue tests, where the sample is completely submerged in an aqueous solution, have been used for many years in corrosion-fatigue testing. However, it is often suggested that the test method is not representative of aircraft service environments. Work has begun on the application of thin salt film layers to sample surfaces which are then hydrated during testing rather than soaking the samples in a solution. These tests have shown great promise towards understanding corrosion inhibitor reactions in more real-world aircraft environments. Work is beginning on how to apply the thin-film work to primered samples.

Point of Contact:

- Sarah Galyon Dorman, Center for Aircraft Structural Life Extension (CAStLE), 719-333-0244

9.6.23. Three-Dimensional Crack Growth Modeling in Mixed-Mode Fatigue of Titanium and Aluminum Alloys

Sarah Galyon Dorman, USAF Academy – CAStLE

In complex aircraft structure, crack growth rarely propagates in the idealized fashion assumed in durability and damage tolerance analyses (DADTA). Usually the applied loading is not perpendicular to the crack nucleating feature and subsequent crack propagation. This situation is known as mixed-mode crack growth or in more general terms, when through-the-thickness effects are considered, three-dimensional (3D) crack growth. Most DADTAs conducted assume mode I only; thus, engineering judgment is used to estimate the amount of error present in the idealized models. The Center for Aircraft Structural Life Extension (CAStLE) at the United States Air Force (USAF) Academy generated mixed-mode crack-growth and residual-strength data for aluminum alloy (AA) 2024-T351 and titanium alloy 6242 (Ti-6Al-2Mo-4Sn-2Zr). State-of-the-art stress analysis and life predictions tools were then used to predict the results, including fatigue life and crack trajectory.

Specifically, mixed-mode fatigue-crack-growth life and trajectory data were produced for 1.6 mm (0.073 inch) thick AA2024-T351 and 2.29 mm (0.090 inch) thick Ti-6242 and using Arcan specimens in an Arcan test fixture. The Arcan test fixture allows the Arcan sample to be rotated to produce different mixtures of mode I and mode II loading (with 0° being tension/compression (mode I) and 90° pure shear (mode II)). Arcan specimens were tested at angles of 0°, 30°, 60° and 75° under constant-initial-stress intensity loading followed by a *K*-increasing test at a stress ratio, *R*, of 0.1. Figure 9.6-52 shows the crack trajectories for the AA2024-T351 and Ti-6242 samples tested at 60º. While mechanical testing was being completed, the crack prediction model of the ARCAN specimen using FRANC3D/NG, a solid-modeling mesh generation and fracture mechanics code from Cornell University Fracture Group, was refined based on the results of the AA2024-T351 program. A parallel effort was also undertaken to develop an engineering model of mixed-mode crack growth where contributions to mode I and mode II growth were accounted for using K_I and K_{II} and the appropriate baseline-crack-growth data. For both the FRANC3D/NG and engineering model analysis, crack-growth-rate-data are required and were produced per ATSM E647 using 6-inch wide, Ti-6242 M(T) specimens for both material thicknesses under constant-amplitude loading and an *R* of 0.1.

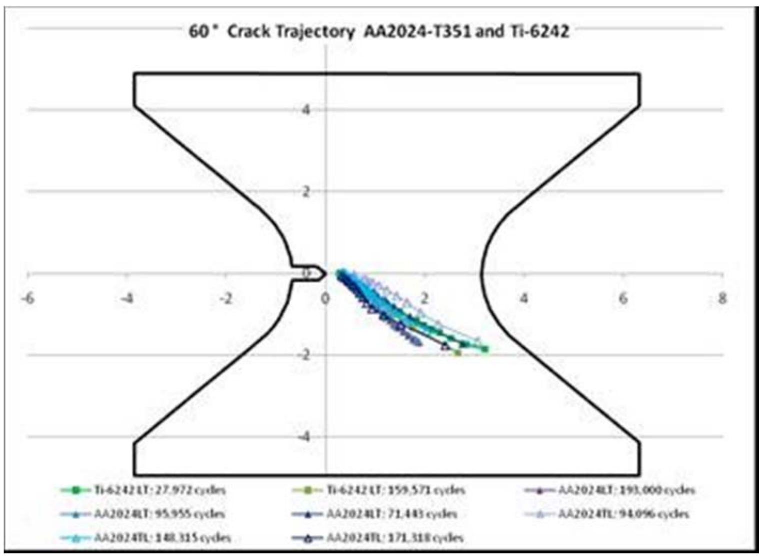


Figure 9.6-52. Mixed Mode Crack Trajectories for AA2024-T351 and Ti-6242 Arcan Samples Tested at 60

Some modeling work was completed using AA2024-T351 examining the effect of orthotropic toughness on mixed-mode crack trajectory modeling. The inclusion of orthotropic toughness caused the modeled crack path to curve towards the produced path at the start of the crack, shown in Figure 9.6-53. The ability to get the crack path to bend in the right direction suggests that further study into the orthotropic toughness could be the key to successful modeling of the mixed-mode crack growth in the Arcan specimen.

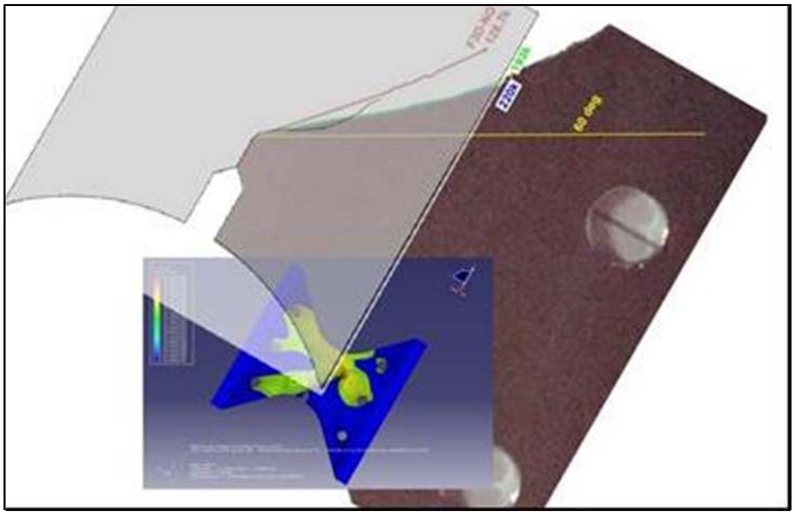


Figure 9.6-53. The Effect of Orthotropic Toughness (Black Line) on FRANC3D/NG Crack Growth Prediction Compared to AA2024-T531 Crack Growth Trajectories (Blue and Green Lines) at 60°

This program was also able to verify the existence of a side load on the test frame actuator produced by the 3-pin Arcan system. An analysis using a visual image correlation (VIC) system also showed the presence of a sideward displacement for this fixture condition.

The Arcan fixture system, including pins, was modeled to help remove some of the uncertainty in the crack growth predictions. Using the results of that model, the crack paths produced during testing were still not able to be replicated.

For the mechanical testing, most of the results were as expected; the mixed mode loading was detrimental to fatigue life, and trajectories were similar between the Ti-6242 and AA2024-T351 samples. The residual-strength trends were quite different between the two alloys, in that the increased mode mixity improved the peak load achieved in Ti-6242, but reduced the peak load in AA2024-T351. The reasons for this have to do with the different mechanisms of crack propagation in the two materials.

The final conclusions from the Program are as follows:

AA2024-T351 Modeling

1) The use of orthotropic toughness properties in modeling produced a crack path that initially turned towards the crack trajectory produced by mechanical testing.
2) However, as the crack grew the predicted path no longer matched the crack trajectory of the sample.
3) Further work is needed on orthotropic toughness to determine if it could accurately predict the physical crack paths.
4) The 3-pin Arcan specimen and fixture configuration causes a real and measureable side load on the test frame actuator.

Titanium Modeling

1) Using the available material property data, the crack paths observed during testing were unable to be modeled.

2) The VCCT SIF method was able to be verified using FRANC3D-NG and ABAQUS.
3) The Arcan specimen, fixture and pins were successfully modeled using MSC/Patran in an effort to determine the effect of the fixture on the crack trajectory.

Mechanical Testing

1) The increased mode mixity negatively affected the fatigue life of the Ti-6242 samples.
2) During the residual-strength testing, the increased mode mixity from 0° to 60° increased the peak load achieved by 25%.

Point of Contact:

- Sarah Galyon Dorman, Center for Aircraft Structural Life Extension (CAStLE), 719-333-0244

9.6.24. Experimental Characterization and Modeling of an Oxide-Oxide Ceramic Matrix Composite

Sandeep Shah, USAF Academy – CAStLE

The focus of this effort is to model the mechanical behavior of a CMC system, specifically Nextel™ 720 fiber-reinforced aluminosilicate matrix CMC from COI Ceramics Inc., San Diego, CA, and to validate the model with a parallel experimental effort.

Modeling. A repeating unit cell (RUC) model of the Nextel™ 720 fiber-reinforced CMC has been successfully formulated and run using NASA's MAC/GMC computer code. Development of the RUC followed closely the work of Bednarcyk[1]. Bednarcyk's development of the RUC for a plain weave geometry required the development of six subcell geometries (Figure 9.6-54).

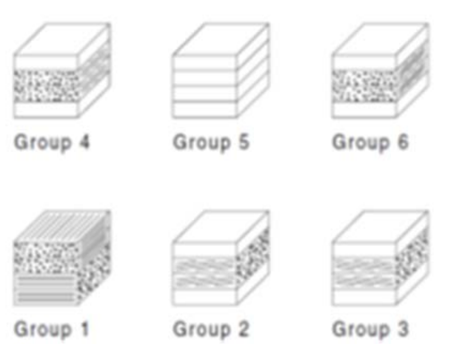


Figure 9.6-54. Six Subcells for Woven Composite

These six subcells were assembled into an arrangement consisting of 64 subcells to represent the smallest RUC of the plain weave. For the eight harness satin weave (8HSW) geometry of the current work, two additional subcells (now a total of eight) are required, and the RUC is much larger; when these eight subcells are arranged into the smallest unit cell of the 8HSW, it consists of 256 subcells.

Findings

Modeling. The very large RUC required to model the 8HSW severely taxed the capabilities of the current generation of PCs with 32-bit operating systems (OS). The plain weave model required MAC/GMC to operate on about 2,800 dependent variables, while the 8HSW CMC model requires the use of eight material models and the tracking of 10,791 dependent variables. The model has been run successfully now in "stand alone" mode and as a material properties input to ABAQUS commercial finite-element software. The modeling effort is currently awaiting experimental results for material properties inputs.

Experiments. We have set-up a servo hydraulic frame with a furnace and pull rod system which can go up to 1,100˚C. We have calibrated this set-up using a silicon-carbide fiber-based ceramic composite specimen. The test matrix for the specimens is based on the previous work by Ruggles-Wrenn[2], wherein we are studying the mechanical response of the CMC at room and elevated temperatures as a function of the frequency of loading cycles.

Point of Contact:

- Sandeep Shah, Center for Aircraft Structural Life Extension (CAStLE), 719-333-8496

References:

[1] Bednarcyk, Brett A., "Modeling Woven Polymer Matrix Composites With MAC/GMC," NASA Contractor Report NASA/CR-2000-210370, Ohio Aerospace Institute, Brook Park, OH, August 2000.

[2] Ruggles-Wrenn M. B., Siegert G. T. and Baek S. S., "Creep Behavior of Nextel 720/AluminaCeramic Composite with +/- 45˚Fiber Orientation at 1200˚C", *Comp. Sc. Tech.* **68** 1588-1595 (2008)

9.6.25. Residual Life/Strength Testing of Lap Joints From Aging Transport Aircraft

Sandeep Shah, USAF Academy – CAStLE

Aging and material degradation in 7XXX series aluminum alloys is a very common problem faced by aircraft maintenance depots around the world in maintaining legacy aircraft. Predicting the residual life of the aging legacy alloy structures is a challenging task. However, reliable residual life data can provide effective inspection schedules to depots and reduce aircraft downtime. This task focuses on evaluating 7XXX fuselage structure for residual life and residual strength. Recent efforts at CAStLE are concentrating on providing reliable residual life data for the fuselage lap joint. The objective of the proposed effort is to determine the residual life of one fuselage-lap-joint section under spectrum loading simulating ground–air–ground (GAG) cycles. Static residual strength is also being evaluated under this effort.

For these tests, specimens consisted of actual fuselage structure and were tested in servo-hydraulic frames with OEM-defined GAG spectrums. The specimens varied in the length of the lap joint, ranging from 14-20 inches. The specimens were tested in a uniaxial spectrum tensile loading configuration. The lap joints were analyzed not only for the residual life and strength, but also to inspect for any environmental damage inside the occluded regions of the lap joints. Of the 24 lap-joint sections tested so far, only two showed some signs of corrosion inside the lap joint, indicating the efficacy of the corrosion-prevention program. All the sections exceeded the goal GAG cycles for the residual life, and all static strength specimens exceeded the required minimum static strength.

The failed lap joint sections after the test revealed multi-site and multi-element damage. All damage was found to have been introduced by testing, rather than the result of previous service history; as-received lap joints from retired aircraft did not have any pre-existing damage. This was confirmed by placing a marker band cycle at 1/3 the full-scale-spectrum load before starting the tests. None of the cracks showed any presence of marker bands, confirming the absence of any previous damage. Electro-impedence Spectroscopy (EIS) was also used to evaluate coatings in the lap joints.

Point of Contact:

- Sandeep Shah, Center for Aircraft Structural Life Extension (CAStLE), 719-333-8496

9.6.26. Detailed Three-Dimensional Modeling of the C-130 Center Wing Box for Damage Tolerance Analyses

James Greer, USAF Academy – CAStLE

Reliable assessments of remaining life and residual strength in aircraft structure depend on accurate analyses. High fidelity tools, in concert with accurately characterized damage, will yield high-fidelity damage tolerance predictions. Under the sponsorship of the Materials and Manufacturing Directorate of the Air Force Research Laboratory and others, the Center for Aircraft Structural Life Extension (CAStLE) at the U.S. Air Force Academy has developed an extremely detailed finite element (FE) model of the C-130 center wing box using manufacturer part drawings. This FE model, which contains several billion degrees of freedom, will allow a user to insert damage at single or multiple locations in the model and make accurate life and residual strength determinations. Approximately 1,000 structural elements, including everything from wing spars to doublers and shims, are modeled in three dimensions with 20-noded hexahedral solid finite elements. Approximately 50,000 fasteners will be included in the model as well, and each fastener is also modeled using solid elements. Parts were modeled using SolidWorks™ software, and model meshing was accomplished using TrueGrid™ software from XYZ Scientific Applications, Inc. (Figure 9.6-55) In addition, XYZ developed a new hole-insertion tool which allows consistent meshing and hole insertion through multiple structural layers at each of the thousands of hole locations. An automated fastener generation tool was developed by CAStLE engineers to make, mesh, and insert the fasteners while identifying and cataloging all potential contact surfaces near fastener locations. In this way, an advanced *p*-version finite element code (STRIPE) can be used to accurately calculate stresses, strains, and stress intensities (including forces generated by contact) in and around damage locations. The model is currently nearing completion.

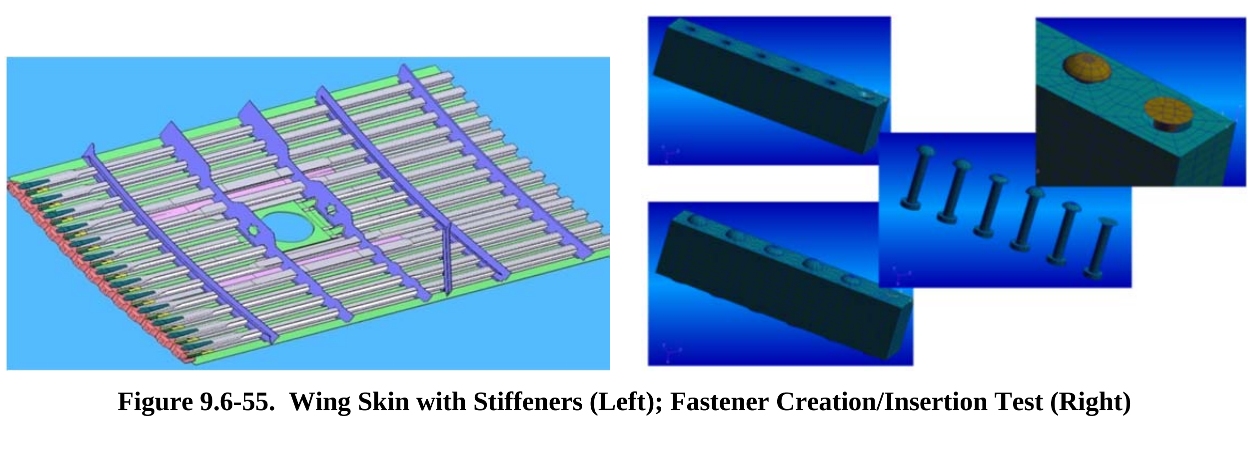


Figure 9.6-55. Wing Skin with Stiffeners (Left); Fastener Creation/Insertion Test (Right)

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- James Greer, Center for Aircraft Structural Life Extension (CAStLE), (719) 333-3618

9.6.27. Processing Ti-Gum Metal and Microstructure and Texture Analysis

Saravanan R. Arunachalam, USAF Academy – CAStLE

Processing of β-phase (bcc) titanium-gum metal with typical composition Ti-23Nb-0.7Ta-2Zr-1.2O and evolution of texture during cold rolling were investigated in conjunction with their microstructures. Furthermore, surface properties were measured using nano-hardness measurement using MTS SA2 nano-hardness tester. The titanium-gum metal was processed by the powder metallurgy route, then compacted and sinter forged in an inert, ultra-high purity argon atmosphere. The sinter-forged alloy was solutionized and cold rolled up to 80%. The microstructure and texture evolution was studied for 2%, 10%, 25%, 50% and 80% cold-rolled conditions. Both hardness and modulus were measured from the nano-indentation for the solutionized and cold-rolled specimens. Measured hardness showed negligible change with respect to percentage cold rolling whereas the modulus shows an increasing trend with the deformation. Modulus increased from 86.6 GPa for the solutionized condition to 99.9 GPa for the 80% cold-rolled condition, and this was attributed to surface properties rather than the bulk behavior. Texture evolution from the early stage of the deformation was captured by the electron-back-scattered-diffraction (EBSD) technique and represented in the form of both pole figures and orientation-density-function plots. The gum metal shows an ideal <100> || (100) for the 50% cold rolled condition and an ideal fiber texture <111>|| Z for the 80% deformed condition. From 2% to 25% deformed conditions, a combination of both these textures were observed. This titanium-gum metal was developed at the Center for Aircraft Structural Life Extension (CAStLE), US Air Force Academy.

Point of Contact:

- Saravanan R Arunachalam, US Air Force Academy, 719-333-0236

9.7. PROGNOSTICS & RISK ANALYSIS

9.7.1. Prognosis: Lessons Learned to Date

Paul Hoffman and Nam Phan, USN-NAVAIR; Stephen Engel, Elias Anagnostou, and John Madsen, Northrop Grumman Corporation: and Alan Ross, Ultra Electronics

Fleet management is confronted with a Herculean challenge; flying beyond originally established service limits, a.k.a. aging aircraft. Hence, new life assessment paradigms are being sought. One potential solution to the challenge is the Structural Integrity Prognosis System, SIPS, under development by DARPA, Northrop Grumman Corporation and NAVAIR. The life management system, SIPS, is a combination of analytical models for cumulative damage and an acoustic sensor system for global detection (Figure 9.7-1). The analytical models available include fundamental micromechanical models (Figure 9.7-2) as well as traditional strain-life and crack growth models. Previously, the acoustic sensor system proved durable and productive as a stand-alone airborne inspection but was not coupled with any reasoning tools. Now, the combination of analytical models, acoustic information and Bayesian updating has shown significant potential. The evaluation of the prognosis concept went from laboratory specimen testing (Figure 9.7-3) to EA-6B outer wing panel verification tests. Then as a proof-of-concept, one P-3 Orion was instrumented with the acoustic sensor package and the SIPS assessment of a single critical zone was routinely conducted as flights accumulated during a period exceeding one year. While the flying period for the proof of concept has ended, the teardown validation process has not been completed. Nevertheless, at this time all evidence foretells of a successful proof-of-concept. Consequently, under review is a proposal for a transition demonstration in which two P-3 aircraft are to be equipped with the third generation SIPS package to cover the two life-limiting zones. Then each aircraft will be monitored and assessed over a two-year period to evaluate if the aircraft can fly beyond originally established maintenance intervals, i.e., originally established service limits. This technical activity will highlight accomplishments and lessons learned to date.

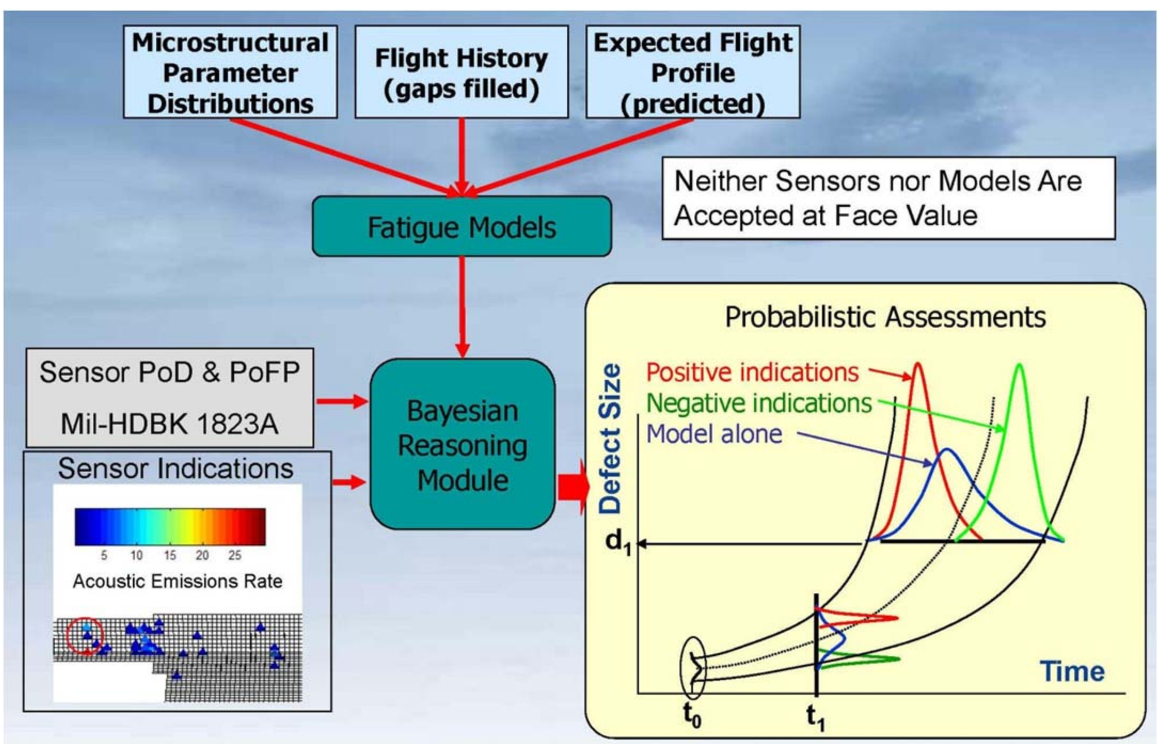


Figure 9.7-1. SIPS Probabilistic Assessments that Account for Sensor and Model Uncertainties

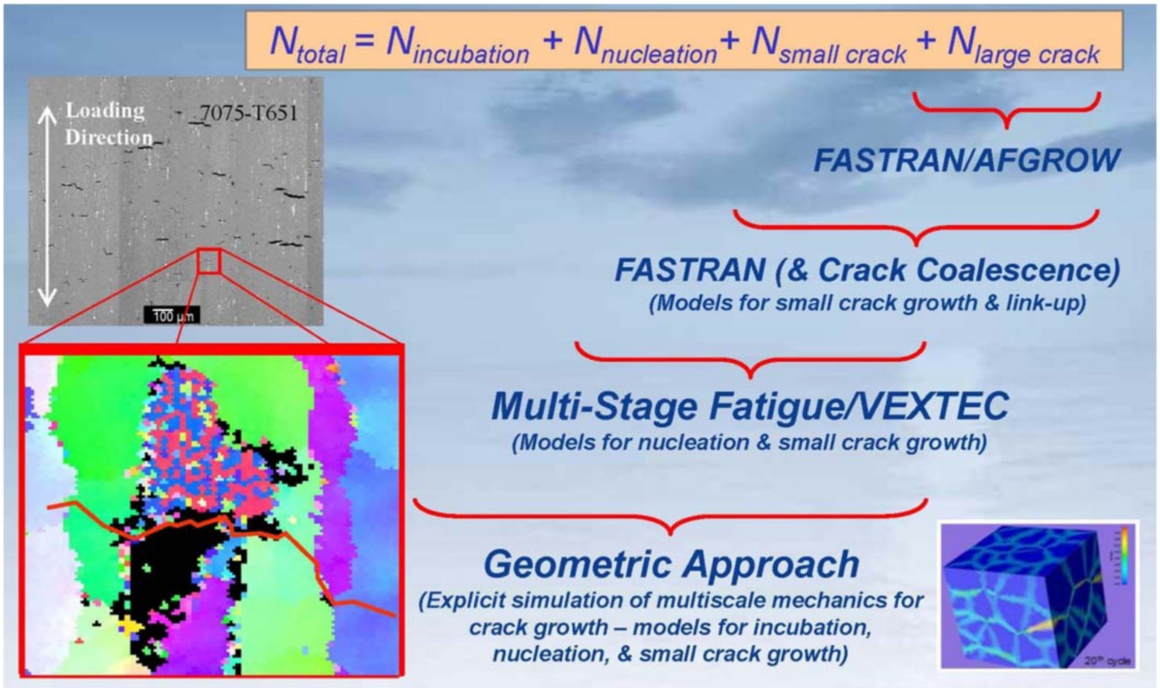


Figure 9.7-2. Microstructurally-Based Models for Fatigue Life Prognosis

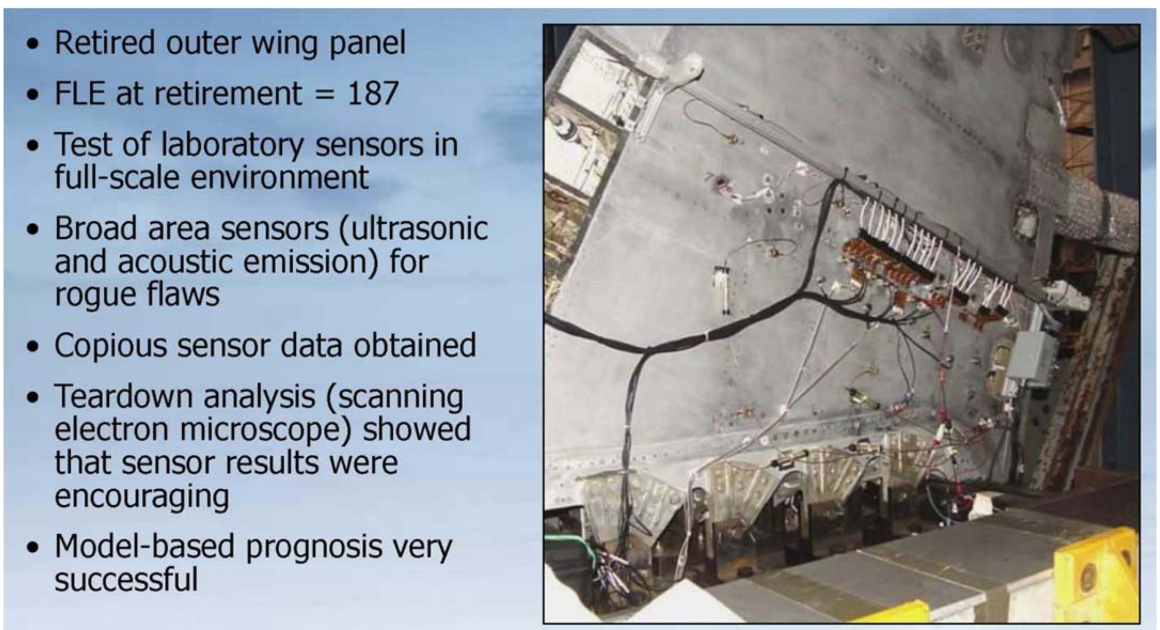


Figure 9.7-3. Model/Sensor/Prognosis Validation

9.7.2. Optimization of F-22 Scheduled Maintenance

J. E. Park and P. J. Caruso, Lockheed Martin Aero – Ft. Worth; and R. E. Bair and W. Garcia, USAF F-22 SPO

A probabilistic method is used to adjust the F-22 fracture critical structural maintenance requirements to improve aircraft availability (Figure 9.7-4). Scheduled maintenance is organized into unique groups called Planned Maintenance Packages (PMP) of 300 hour increments. The scheduled maintenance requirements are adjusted by the F-22 Individual Aircraft Tracking (IAT) program. A challenge to aircraft availability is created when a damage tolerance inspection requirement causes significant downtime for panel access or restoration. F-22 Structures Engineering developed an optimization procedure to increase aircraft availability and decrease unnecessary restoration of panels. The optimization method is a 2-parameter Weibull analysis used to compare the probability of failure for two possible scheduling requirements: 1) damage-tolerance-based interval, and 2) proposed-PMP-based interval. The optimization procedure identifies the inspection requirements for each control point (CP), and groups control points that require the same panel access together for their initial and recurring inspection intervals. Each group may include control points that require limited overfly of the IAT damage tolerance interval. A weighting function is applied to each control point risk-of-failure calculation to account for the number of potential crack initiation details and the reliability of the IAT equation. IAT overfly limit criteria is applied to preclude numerically possible, but unconservative inspection requirements. Every allowable combination of scheduled inspection requirement is evaluated for each F-22 air vehicle, and an optimized risk and cost inspection schedule is developed (Figure 9.7-5). An example of an optimal schedule of F-22 maintenance requirements is presented. Lessons learned to be applied to aircraft with similar maintenance cost issues is discussed.

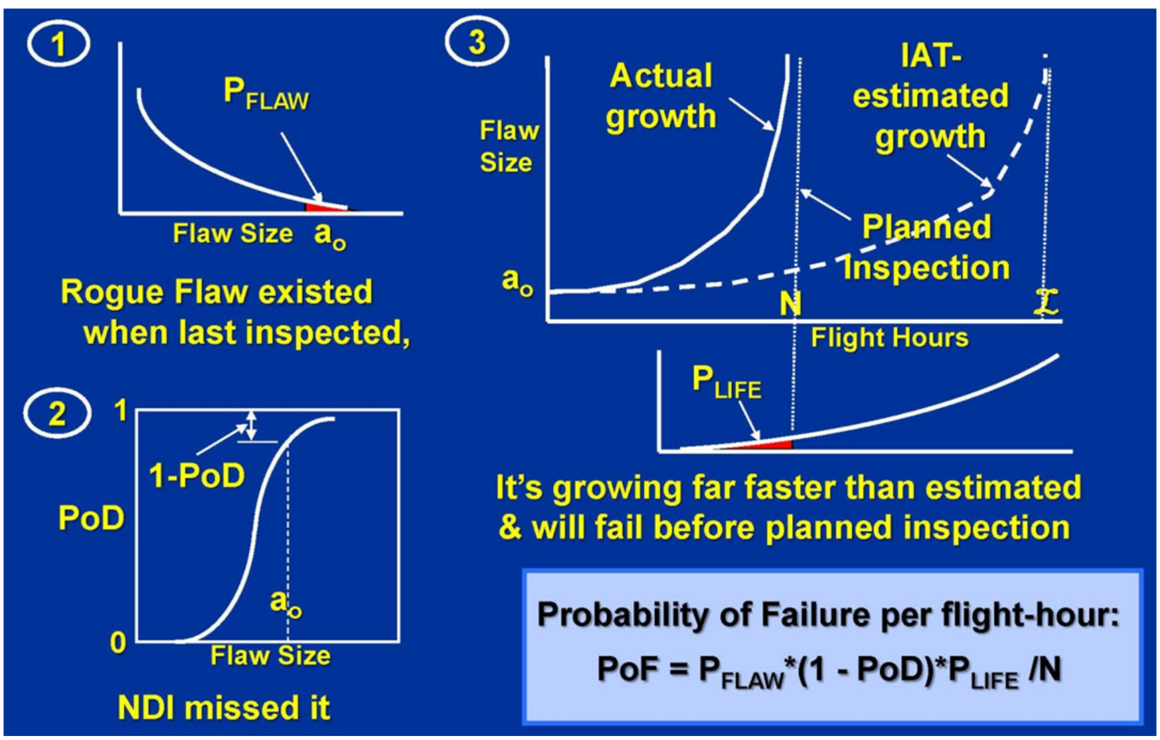


Figure 9.7-4. Three Types of Probability

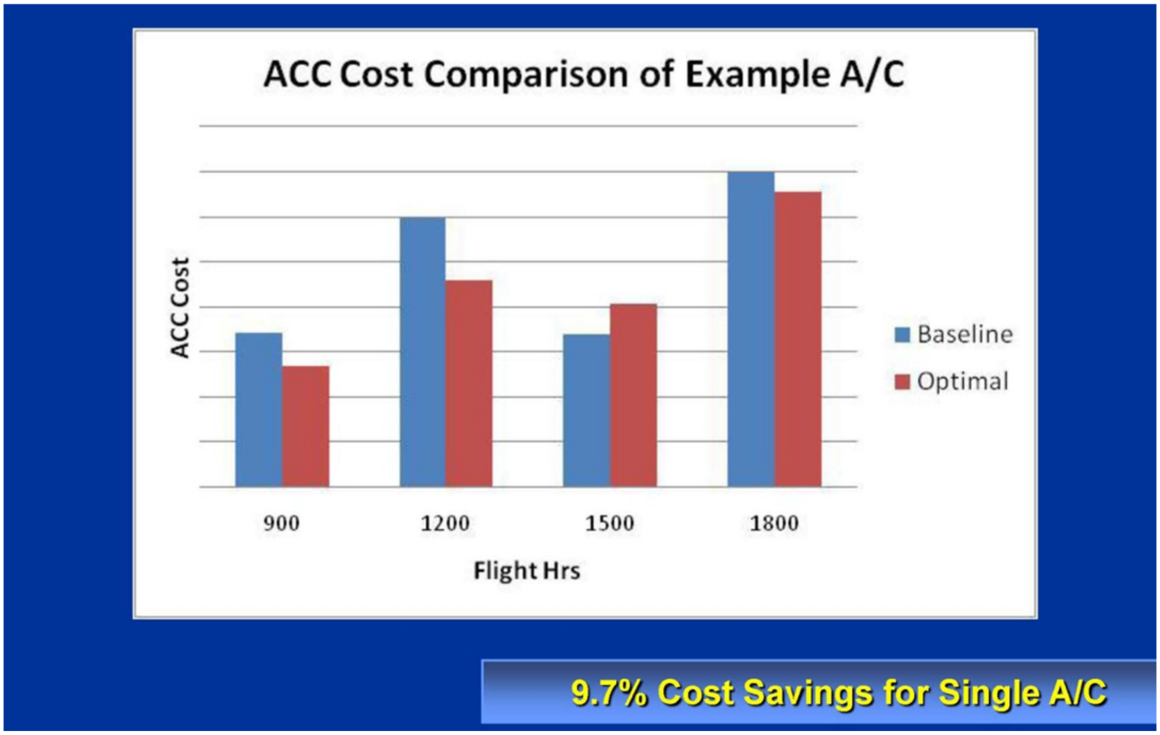


Figure 9.7-5. Sample Optimization Results

9.7.3. Application of Overflight Criterion to Fleet Management

Michael Blinn and Mark Thomsen, USAF-OO-ALC; and Zach Whitman, Southwest Research Institute

Recent challenges have highlighted the need to re-evaluate some of the T-38 Aircraft Structural Integrity Program's (ASIP) analyses; specifically, the estimates of damage and risk to the fuselage (Figure 9.7-6). The full-scale fuselage fatigue test established a number of new locations on components that were not identified previously. These locations occurred on structure that has been replaced and/or modified since the T-38 initial operational capability (IOC). Furthermore, these component locations have accrued significant flight time in the fleet prior to the discovery of the new damage locations.

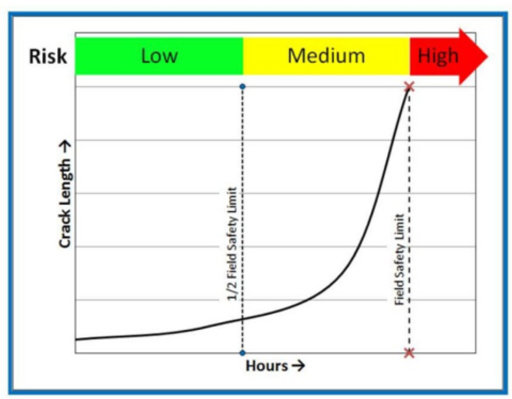
The different locations on each aircraft had reached a different point relative to the safety limit as established by the damage tolerance analysis (DTA). For some aircraft, this puts certain locations at an elevated risk with regard to an over-flight criterion. More detailed analyses, such as a PROF-type risk analysis and Weibull analysis of present risk of damage, have been developed to better quantify the actual fleet risk at these locations. Qualitatively, each analysis can present results in a high-medium-low, stoplight-type format. The actual unit basis for each analysis, however, is different.

Over-flight provides Yes/No results to gates established by the DTA. The risk is low if an aircraft hasn't overflown an inspection. The risk is medium if it has overflown the inspection but not the safety limit and high if the safety limit has been overflown (Figure 9.7-7). PROF can provide a probability of failure in terms of hazard rate or probability of failure which can be compared to MIL-STD-1530 limits to establish risk at Medium-Serious-High. The Weibull-type analysis is based on the findings from the full-scale test. The test is then compared to the fleet data through use of equivalent flight hours (EFH). The present or future risk can be calculated that establishes the expected number of cracks that would reasonably be expected to exist in the fleet. A dilemma is created when, on differing bases, the level of risk calculated by the different analysis methods are not the same. Further uncertainty is created when the predicted number of cracks is different than what is uncovered following an inspection. This technical effort discusses the various steps taken to resolve or explain the differences in the various analyses relative to actual fleet experience.

- 1950s – Designed Durable; Not Damage Tolerant
- 1960s
  - Initial Operational Capability (IOC) 1961
  - Set Several Women's Speed, Distance, and Altitude Records in 1961 (flown by J. Cochran)
  - Set Four Time-to-Climb Records in 1962 (flown by Maj. W. Daniel)
- 1970s
  - Last T-38 Delivery in 1972
  - Durability/Fatigue Requirements and Damage Tolerance Introduced as Program Progressed
  - Total of 1,187 T-38s Produced
- 2010
  - Trained Pilots in Several Different Countries
  - Over 75,000 Pilots
- T-38's Primary User Remains the USAF (AETC)
- ACC, AFMC, and GSC also Utilize the T-38
- Navy and NASA have T-38 fleets

Figure 9.7-6. T-38 History Overview

- Consider a Relatively Simple Over-Flight Criterion - Based on DTA Safety Limit
- For Initial Inspection Interval Determination, Crack Growth Begins at "Rogue Flaw"
- Risk Depends on Where an Individual Aircraft Falls Along the Curve
  - Low Risk - Zero to ½ Safety Limit
  - Medium Risk – ½ to Full Safety Limit
  - High Risk - Beyond Safety Limit
- Aircraft Hours for Comparison in Terms of EFH, Base-lined to DTA Usage
  - "Credit" Given for Prior History
  - Benefits Moderately or Benignly Flown Aircraft
  - Penalizes Severely Flown Aircraft Appropriately



Notional Chart as from ASC/EN's Aircraft Structural Integrity Team's (ASIT) 2010 T-38 ASIP Review at OO-ALC

Figure 9.7-7. Over-Flight Considerations – Damage Tolerance Analyses

9.7.4. UH-1N Tail Boom Attach Fitting DTA – A Risk-Based Approach

Steven Lamb, USAF-WR-ALC; and David S. Carnes, Gregory J. Todd, and Robert McGinty, Mercer Engineering Research Center (MERC)

The USAF fleet of UH-1N helicopters is approaching 35 years of service, with another 10 years remaining until full retirement. Recent development of an Aircraft Structural Integrity Program for the airframe has included an effort to establish equivalent flight hours (EFH) for critical structure in order to enhance fleet safety and to provide decision support for aircraft retirement. One area for which EFH is being established is the lower right tail boom attach fitting (Figures 9.7-8 and 9.7-9). Mercer Engineering Research Center (MERC) was tasked to perform a damage tolerance analysis (DTA) of the tail boom attach fitting in support of the EFH development task. The DTA yielded an unacceptably short inspection interval for the fitting. Subsequently, a risk-based approach based on guidelines from the Aircraft Structural Risk and Reliability Analysis Handbook (as presented at ASIP 2009) was undertaken.

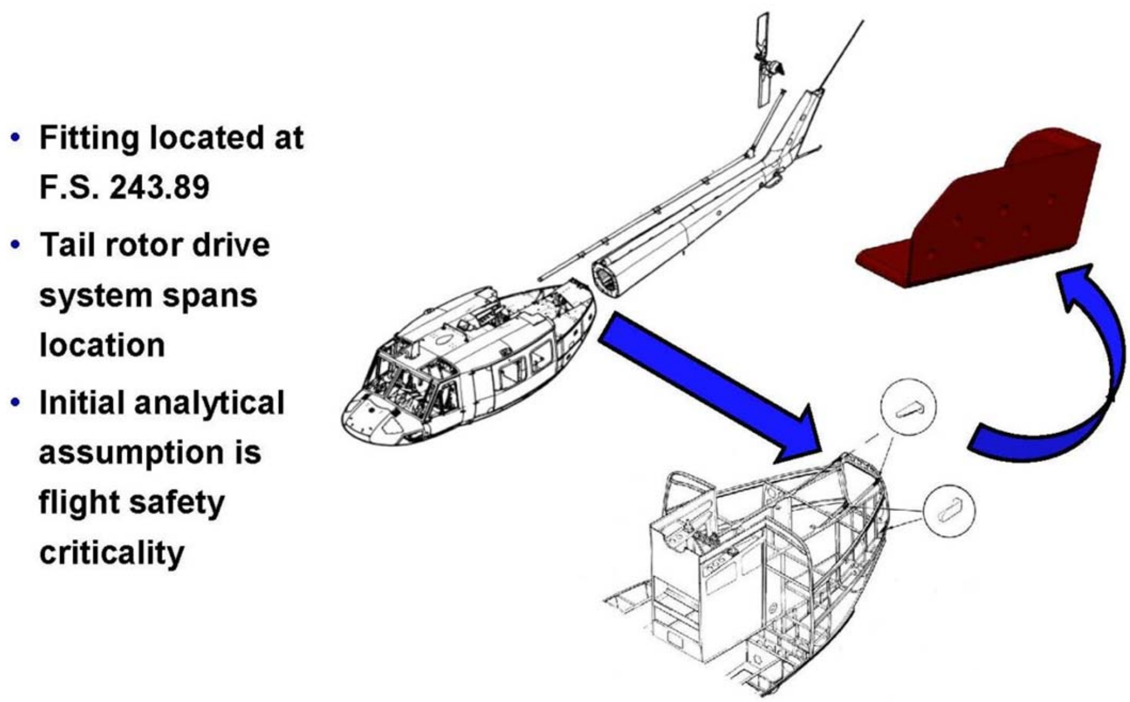


Figure 9.7-8. Analysis Location

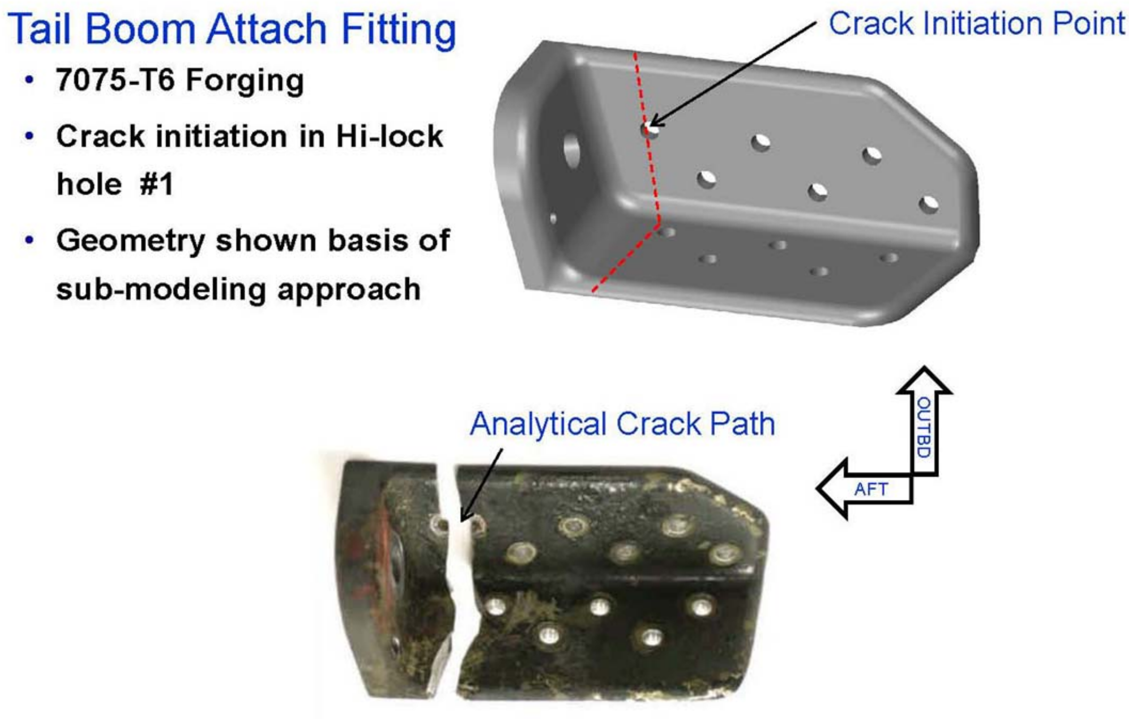
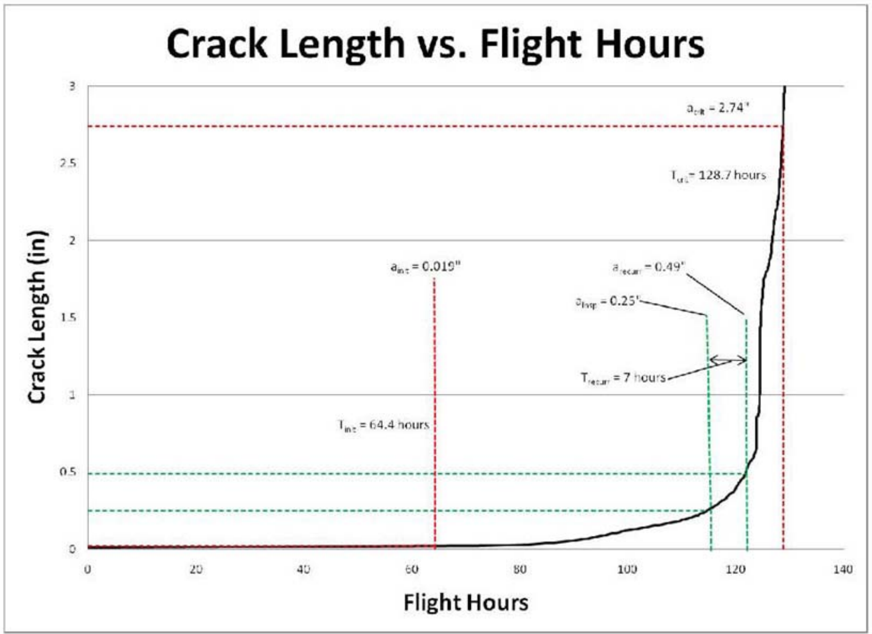


Figure 9.7-9. Tail Boom Attach Fitting Geometry

The DTA utilized a sub-modeling approach to solve for stress intensity factors (K) as a function of crack length. Flight load conditions were developed from a flight strain survey on the UH-1N. These loads were placed on the global UH-1N FEM and the resulting boundary stresses at the fitting were translated to a detailed FEM for K solutions. A load spectrum was developed based on the mission usage profile from the ASIP master plan. The rainflow counted flight load spectrum was used with AFGROW to predict the crack length versus time for the lower left tail boom attach fitting (Figure 9.7-10). AFGROW results indicated that the DTA for this component is highly sensitive to input parameters, particularly the initial flaw size. This high sensitivity and especially the exceptionally fast predicted crack growth rates led MERC to conclude that a probabilistic approach is better suited for this analysis.



- **AFGROW run with 0.01" initial flaw**
- **7075-T6 material data from NASGROW database**
- **7 hour recurring inspection is unacceptable (dye penetrant every 3.5 flights)**

Figure 9.7-10. AFGROW Crack Growth Results

The probabilistic DTA was performed using PROF (PRobability Of Fracture) software. Distributions were developed for equivalent initial flaw size (EIFS), fracture toughness, probability of detection, max stress per flight, crack length over time (obtained from AFGROW), repaired crack size, and fleet information (number of aircraft, average hours per flight, and number of locations of part). Results show that a new fitting has a 1 in 3.3E7 probability of failing during the next flight within the first 100 hours of use. For fittings with a 10,000-hour flight history, this increases to a 1 in 2.7E5 probability of failing. Per the preview of the Aircraft Structural Risk and Reliability Analysis Handbook, this is equivalent to a MIL-STD-882D hazard risk index of 8.

This technical effort presents the findings that drove the need for probabilistic methods, and the risks as quantified by the probabilistic approach. The effort has also identified areas where further details/accuracies are required (material behavior at high R-ratio loading, loads, load spectra). It has also identified areas where further investment will not further quantify the risks or clarify mitigation/management (improved K values, refined structural mesh). Conclusions and recommendations are also presented.

9.7.5. Using Retrodiction to Increase the Reliability of T-38 Wing Replacement Forecasts

Zachary Whitman and Hal Burnside, Southwest Research Institute

The T-38 wing has undergone multiple revisions to improve the fatigue life of the structure ever since the first Air Force delivery in 1961. The current version of the wing was procured and installed fleet-wide in the early 1980s and is now approaching 30 years old. The wing economic life estimate and the wing replacement forecasts are almost 15 years old (Figure 9.7-11). The wing economic life model is fixed; yet the T-38 mission severity has steadily increased and will likely continue to increase until retirement. The fleet size has continued to be reduced over the last 15 years, giving cause for concern regarding the accuracy of current and future wing replacement estimates.

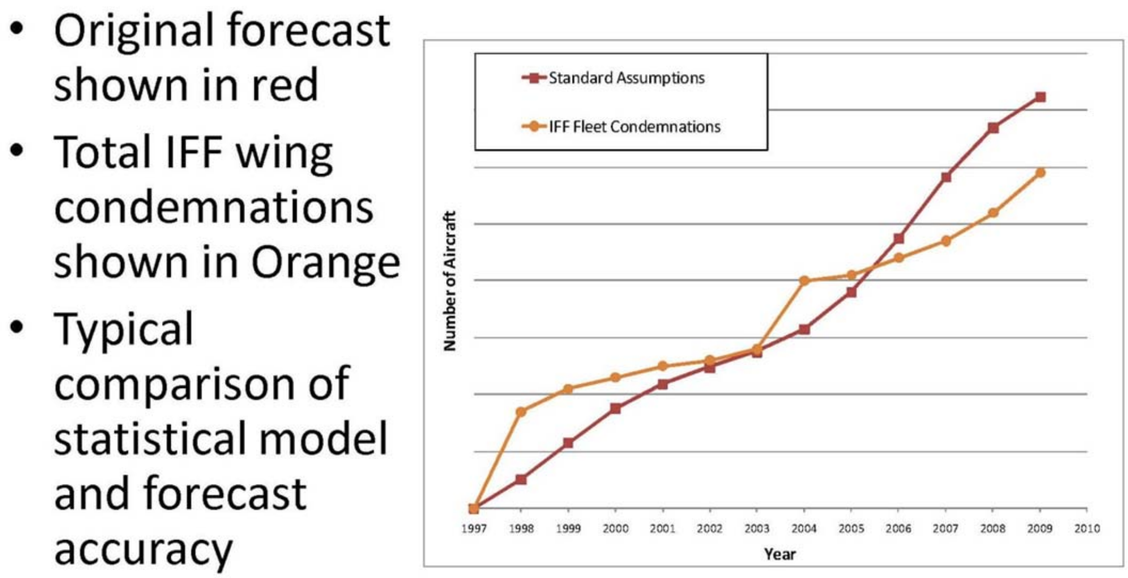


Figure 9.7-11. Wing Condemnation Forecasts

All forecasts are hindered by an unknown future. In the case of the T-38 wing, there are uncertainties in the assumed fleet size, wing hours in the fleet, and usage. To estimate the accuracy of the wing replacement forecasts, actual fleet assignments, hours flown, and usage severity are input into the original life model in a method known as retrodiction. A retrodictive model uses known events to explain the past; i.e., fleet flying rates and severity explain wing replacements through the wing-life model (Figure 9.7-12). In this technical activity, retrodiction is used to re-calculate the wing replacement forecast with knowledge of actual fleet history from the year of the forecast to the present day. Differences between the retrodiction and actual wing replacements can then be attributed to the inaccuracies of the wing economic life model, whereas differences between the original wing replacement forecast and the retrodiction are due to inaccuracies in the future fleet assumptions. Assumptions made regarding the future fleet are typically 'conservative' estimates based on the historical usage of the current fleet. This methodology is necessary for certain planning purposes, such as new wing acquisition. In the case of other planning activities such as scheduling, excess conservatism can be an impediment. History proves whether these assumptions were actually conservative.

- Known over the 12 year period that the actual average fleet flying rate only 210 hours per year
- Revised ‘forecast’ with corrected yearly flight hours would appear unconservative when compared to the total IFF wing Condemnations

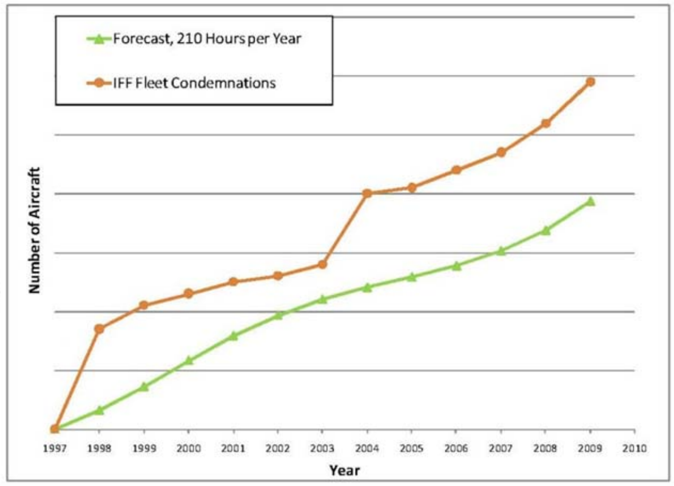


Figure 9.7-12. Retrodiction Analysis

This technical effort presents a comparison of the original predictions, retrodiction based on the original analysis, and the actual wing replacement history. Retrodiction provides the ability to separate the analysis error into two parts: variation between the statistical model and actual wing, and errors that arise from assumptions made about a future fleet. This feedback can then be utilized in forecasts for the next 15 years to either ‘tune’ the analysis or increase confidence in the projected wing replacements (Figure 9.7-13).

- Consider giving forecasts for a range of flight hours per year for later interpolation or use in ‘what-if’ scenarios

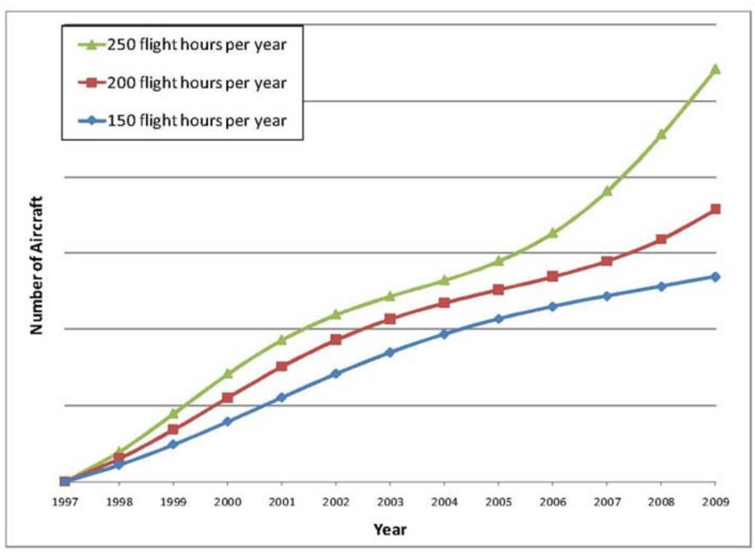


Figure 9.7-13. Forecasts for Range of Flight Hours Per Year

9.7.6. The Impact of Non-Destructive Evaluation in Determining the Optimal Maintenance Schedule Based on ASIP Risk Requirements

Tony Torng and Ko-Wei Liu, The Boeing Company

The aircraft industry around the globe is currently focusing on applying physical-model-based probabilistic analyses to assess the quantitative risk for the Aircraft Structural Integrity Program (ASIP). In particular, the USAF has established the risk/reliability-based requirements in MIL-STD-1530C. The advantages of Risk-Based Analysis (RBA) over deterministic approaches are well recognized, such as in reducing over–conservatism imposed by using safety factors, identifying optimal maintenance schedules, and reducing the overall life-cycle costs. To help determine the quantitative risk and the optimal maintenance schedules, computer codes such as the PRobability Of Fracture (PROF), a code developed by the University of Dayton Research Institute, and a Boeing Proprietary Risk-Based Design and Maintenance System (RBDMS) code can be applied. Critical input random variables for this analysis include the equivalent initial flaw size (EIFS), material fracture toughness, crack growth curve, geometry curve, flight load exceedance spectra, and non-destructive evaluation (NDE) probability of detection (POD) model inputs (Figure 9.7-14).

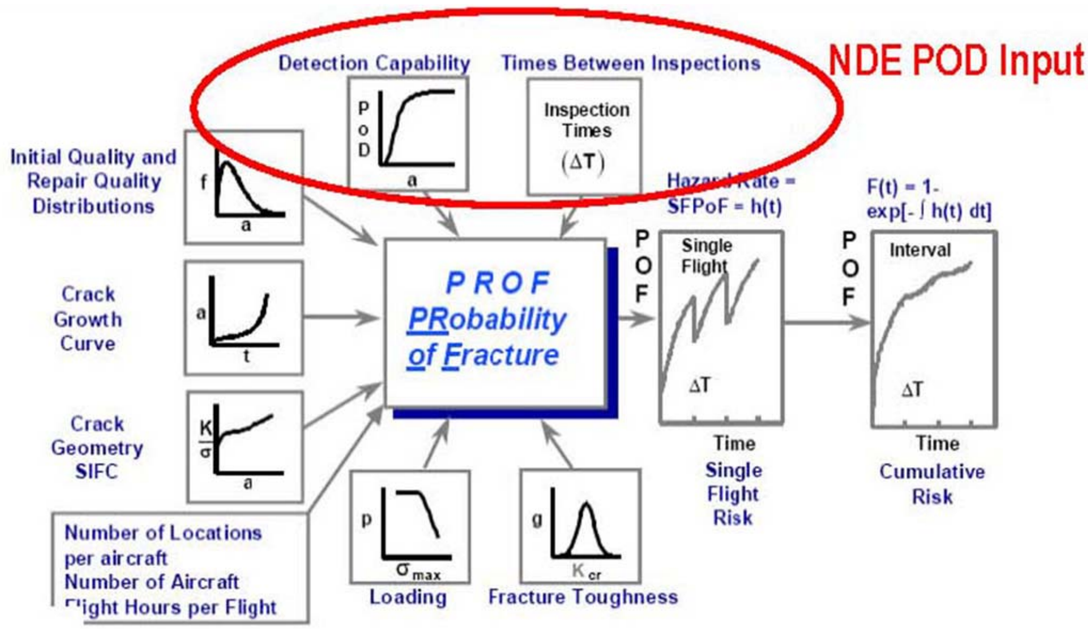


Figure 9.7-14. Risk Analysis Input Parameters and Single Flight Probability of Failure (SFPOF)

This objective of this technical effort is to investigate the impact of NDE POD in determining the optimal maintenance schedule based on the risk results calculated using the proposed risk-based codes (PROF and RBDMS). One aircraft example (Figure 9.7-15), which used a surface eddy current method to detect the crack, will be used to demonstrate the critical role of NDE POD in determining the optimal maintenance schedule. Several POD capabilities will be assumed and modeled to determine their corresponding optimal maintenance schedules. The impact of various POD models will be compared and used to compare with the inspection schedule assessed by the traditional deterministic safety factor approach (Figure 9.7-16 and Table 9.7-1). In addition, how to determine the missed crack distribution and its impact to the overall risk will be discussed. Finally, how to integrated NDE with Structural Health Monitoring (SHM) to reduce the overall risk, especially for those unexpected failures caused by the missed cracks, will be briefly discussed.

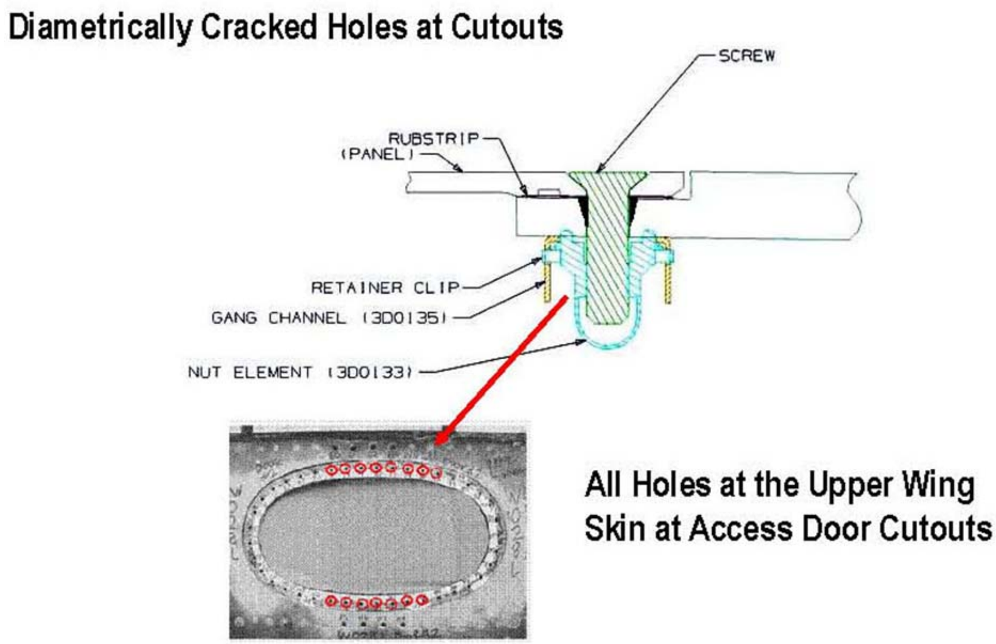


Figure 9.7-15. Demonstration Example: Upper Wing Skin at Access Door Cutouts

- Damage tolerance life (DT life) – crack growth life from a single 0.05” lead crack to critical crack length. In addition, a safety factor of 2 is included.
- Based on deterministic analysis, DT life was calculated equal to 89500 hours at critical crack length of 0.23”. Then, DT life was divided by a safety factor of 2 = 44700 hours to determine the time for inspection and repair.

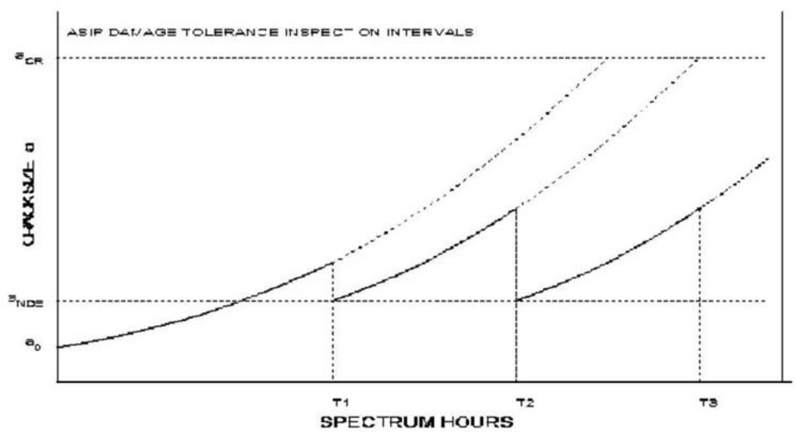


Figure 9.7-16. Damage Tolerance Life – Deterministic Approach Result

Table 9.7-1. Risk Assessment Results Summary Based on 1.E-7 Single Flight POF Criterion

Methods/NDE Inputs	1st Inspection Time and POD	2nd Inspection Time and POD
Deterministic DT Life Results = 89500 hours 2nd inspection time based on a_{NDE} = 0.05	44750 hours	44750 hours
NDE1 (90/95 POD = 0.025, 50/95 POD = 0.015, and 10/95 POD = 0.009)	82000 hours, % detected = 99.35%	73600 hours, % detected = 99.16%
NDE2 (90/95 POD = 0.05, 50/95 POD = 0.03, and 10/95 POD = 0.018)	82000 hours, % detected = 77.67%	23300 hours, % detected = 21%
NDE3 (90/95 POD = 0.075, 50/95 POD = 0.05, and 10/95 POD = 0.0333)	82000 hours, % detected = 26.19%	2280 hours, % detected = 20.46%

The Second Inspection Time Reduced Greatly When Using Different NDE Capability (From 73600, 23300, To 2280 Hours)

9.7.7. A Computer Simulation Evaluation of the P-3C Upper Wing Cover

Arnold E. Anderjaska and Douglas Algera, Technical Data Analysis, Inc.

The P-3C upper wing cover is unique and its evaluation for fatigue and corrosion has been difficult. The fatigue damage is primarily caused by ground tension loads but the structure is designed by the high compression flight loads. Separate, conventional fatigue, corrosion and residual strength evaluations have been made. To consider these combined aspects and to look at the problem from a different perspective, the subject SAIFE program was adapted to simulate the P-3C upper wing cover situation. The SAIFE is a computer program that performs a simulation risk analysis of the life of a fleet of aircraft structures accounting for classical fatigue cracking and also cracking caused by corrosion, production defects and service damage. It is a dynamic simulation that predicts service defects and inspects for them. It tightens inspections if problems are found and loosens inspections if problems are not found. All found defects are corrected by repair or modification depending on relative cost.

The simulation covers the seven span-wise plank splice areas that cracked in the P-3C Full Scale Fatigue Test (FSFT) and one area that cracked in service. The assumed wing failure mode was based on the P3V FSFT, which experienced massive failures, and it includes the cracked panel shedding load to adjacent panels. The crack initiation sizes and the average time to initiation were based on the P-3C FSFT. The average crack growth from initiation to wing failure was generated using the stress spectra based on the P-3C Individual Fatigue Tracking (IAT) results and its Finite Element Model (Figure 9.7-17). The corrosion, production defect and service damage rates were based on a study of 10 years of civil transport usage.

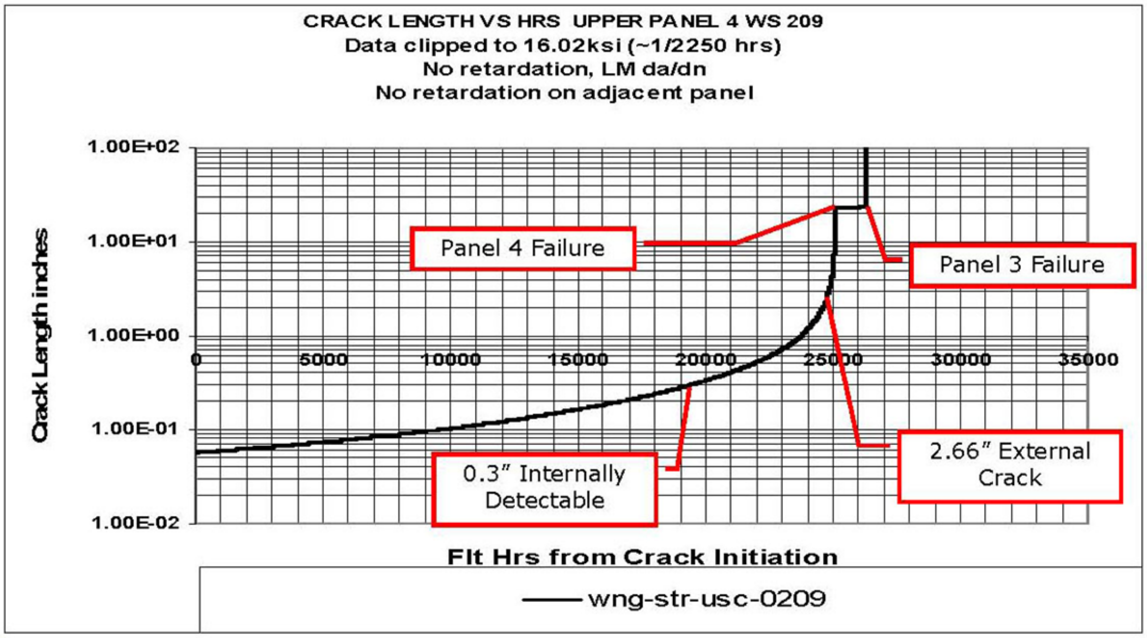


Figure 9.7-17. Details of Crack Growth Progression

Three cases were run to the maximum service life: 1) with the past P-3C inspection program, 2) with the past P-3C inspection program assuming detection by fuel leakage, and 3) with additional rigorous internal inspection. The results indicated that for the P-3C upper wing cover, large cracks or wing failures could occur unless additional rigorous internal inspections were made. It indicated that even when the average time to crack initiation plus the growth to failure is double the service life, such large cracks or failures could occur. This was the result of scatter in loads, fatigue crack initiation and growth, corrosion, production defect and service damage occurrences, their interaction, and realistic detection probabilities. This illustrates why aircraft designed and tested to their design life, especially when their life or usage is more serve, experience significant cracking in their old age.

9.7.8. Continuing Correlated Risk Analysis of the C-5A Fuselage Aft Crown

Jeffrey Johnson and Ed Ingram, Lockheed Martin Aero – Ft. Worth; Thomas Brussat, Tom Brussat Engineering LLC; and David Wilkinson, USAF-WR-ALC

Throughout 40 years of service the C-5A aft fuselage upper crown area has demonstrated remarkable damage tolerance (Figure 9.7-18). Fleet-wide over its long history, over 1,300 cracks have been discovered and repaired in the brittle, stress-corrosion-susceptible 7079-T6 skin material, yet the aft crown has never experienced a catastrophic skin failure, nor any significant cracking in the 7075-T6 stiffeners. To confidently project the safety record of this area into the future, a risk analysis is needed that correlates with the historical and technical data. This technical effort presents the status of an updated risk analysis of the aft crown region which incorporates the following:

- Up-to-date service cracking data that provide over 1,300 crack-size data points for estimation of the EIFS distribution and the POD for periodic visual inspections (Figure 9.7-19)
- A meticulous quantitative characterization of these service cracks to account for crack orientation and the variation in stress with crack location
- Residual strength test results from large stiffened panels extracted from retired aircraft
- New stringer-level FEM results for the aft crown region to correlate with these test results
- SCC data for aged panel material from retired aircraft

- Assessment of the corrosive weather environments where the C-5A fleet has been based

- **Late 1960s design emphasized high strength and minimum weight**
- **Skin material is 7079-T6 clad one side, 0.050 inch thick**
  - **Brittle with relatively low toughness**
  - **Susceptible to Stress Corrosion Cracking (SCC)**
- **Critical crack length at limit load is less than stringer spacing**

Figure 9.7-18. C-5A Aft Crown Region

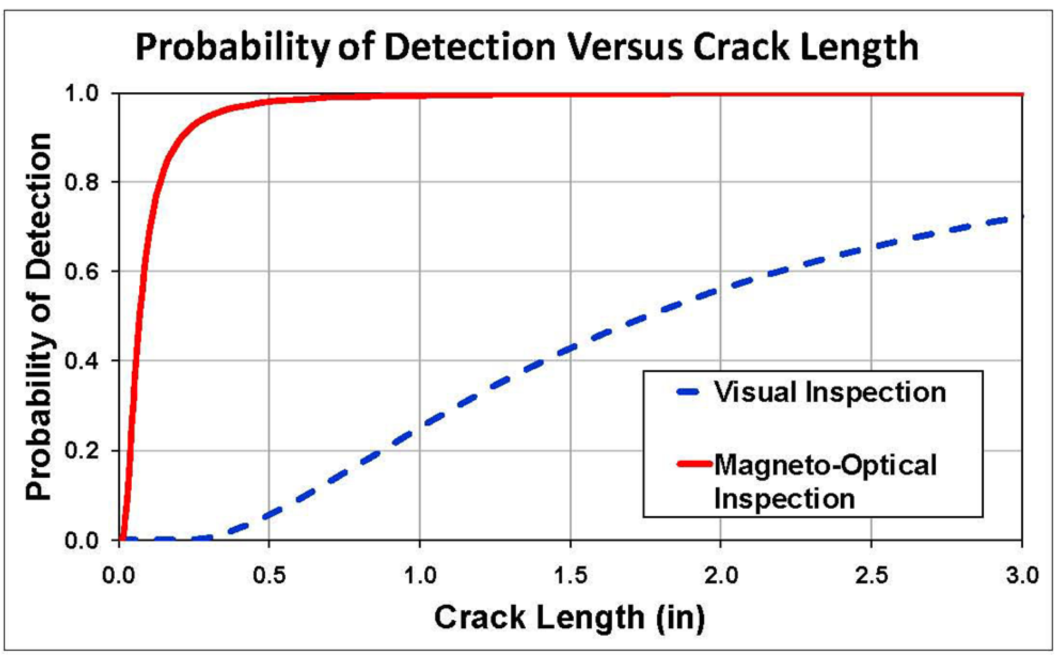
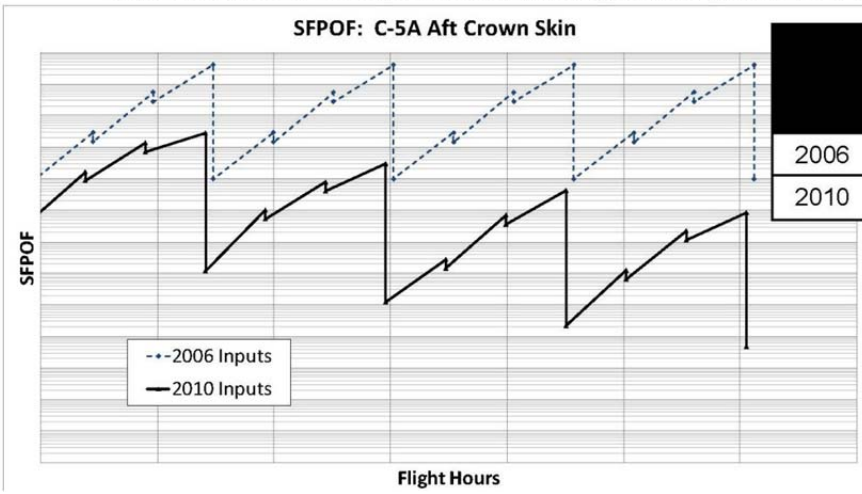


Figure 9.7-19. POD for Visual and Magneto-Optical Inspection Methods

The estimated single flight probability of failure (SFPOF) for continued structural integrity of the C-5A crown is presented (Figure 9.7-20). Assumptions and methods are used that correlate with the above data and that back-estimate a SFPOF of the C-5A aft crown that is consistent with 40 years of safe operation.

- **Effect of changes in crack growth curve and residual strength on SFPOF**
  - **Same inspection types and intervals to compare CGC and RS changes**
  - **Recommended inspection intervals for C-5A fleet were increased to take benefit of SFPOF reduction**
  - **Will continue to compile crack findings and update risk at future date**



	MOI Interval (Months)	Visual Interval (Months)
2006	32	4
2010	48	16

Figure 9.7-20. Single Flight Probability of Fracture (SFPOF)

9.8. LIFE ENHANCEMENT CONCEPTS

9.8.1. F-22 Laser Shock Peening Depot Transition and Risk Reduction

Ken MacGillivray, Robert Bair and Wirt Garcia, USAF-F22 SPO; Brent Dane, Metal Improvement Company; and Morgan Osborne, The Boeing Company

Laser Shock Peening (LSP) is a developing technology intended to enhance fatigue life on metallic structure, with current applications primarily in engine fan blades. However, the USAF F-22 program is using Laser Shock Peening to enhance fatigue characteristics on the wing attachment lugs of operational aircraft (Figure 9.8-1). This is the first application of LSP for the Air Force on thick titanium structure and the first application of LSP on operational aircraft in depot. As such, the F-22 program has completed an extensive technology transition and risk reduction program.

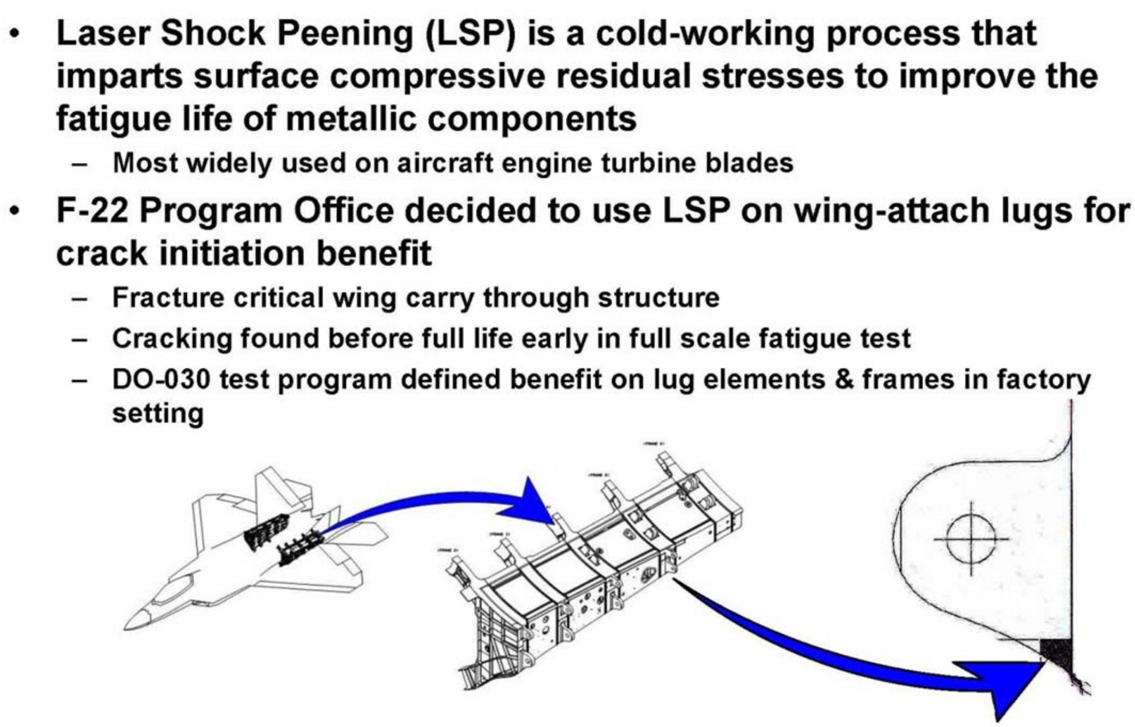


Figure 9.8-1. Laser Shock Peening (LSP) Background

This program has included the development of necessary hardware to allow for full laser peening of eight of the aircraft's wing lugs in a depot environment. Numerous process controls and quality control measures have been established to ensure the consistent application of exact residual stress requirements. A rigorous safety program has been established to ensure both safety of the aircraft structure and personnel (Figure 9.8-2). The F-22 program has also conducted testing to reduce risk in phenomena occurring as a result of the LSP process that are not well understood. This technical effort highlights the achievability of this application of LSP and will present the results of this transition/risk reduction program as well as the F-22 Program's plan forward to implement this new technology in depot.

Figure 9.8-2. Risk Reduction: Ensuring Safety at Depot

9.8.2. Mitigation of Fatigue and Cracking Damage in F-16 Wing Pylon Cutout Through Low Plasticity Burnishing (LPB)

Narayanan Jayaraman, Russell Lascelles, and Paul S. Prevey, Lambda Technologies; Selen Minarecioglu and Carlos Cordava, Lockheed Martin Aero – Ft. Worth

The F-16 wing lower skin pylon cutouts are reportedly prone to fatigue cracking in the wing lower skin at wing weapon pylon cutouts resulting in a fatigue debit and reduced damage tolerance life (Figure 9.8-3). Inspection and maintenance costs adversely impact readiness and increase the total cost of ownership and operation. A program was developed to establish the LPB process conditions and optimize the engineering to improve damage tolerance life and enhance fatigue performance of the parts.

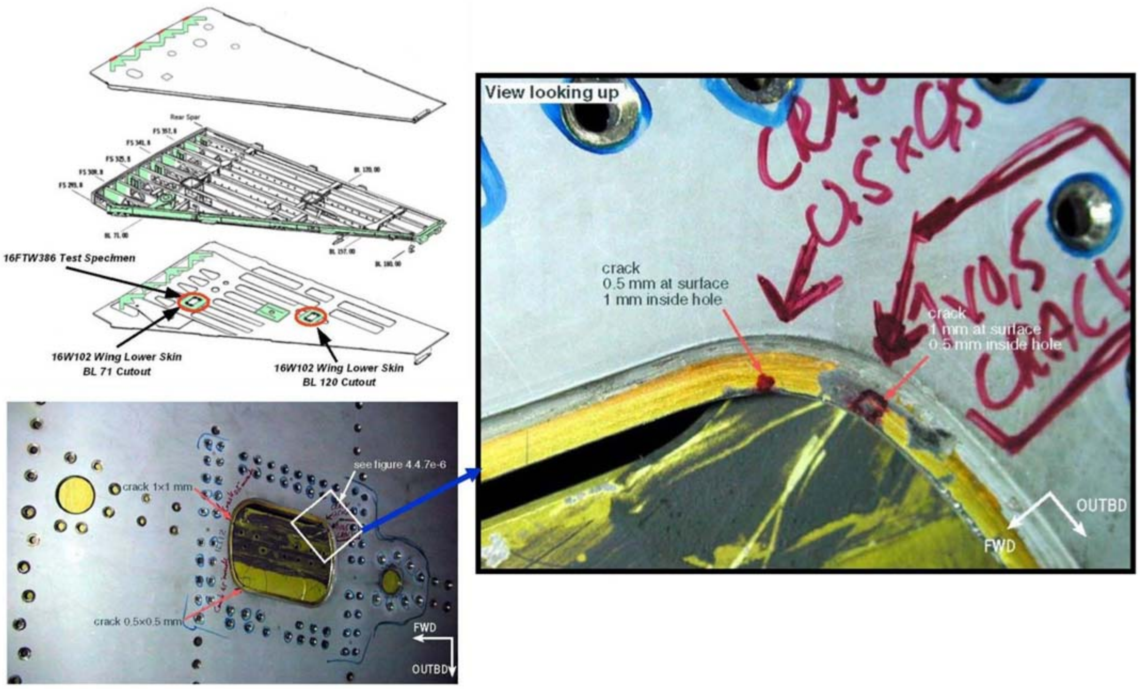


Figure 9.8-3. Fatigue Crack Locations in Wing Lower Skin Pylon Cutouts

An integrated total solutions approach was used, consisting of the following steps: (1) use of design tools including Lambda's Fatigue Design Diagram (FDD) method and linear elastic fracture mechanics (LEFM) based crack growth analysis, to determine the compressive residual stress field required to achieve an acceptable level of damage tolerance, (2) modification of LPB tools and fixtures to process the component, (3) design of LPB process parameters and the corresponding robotic/pressure codes, (4) measurement of residual stresses (RS) in LPB processed parts, and (5) fatigue testing of actual wing lower skin mockups to verify the effectiveness of the LPB process.

Durability testing was conducted with the load spectrum provided by LM Aero on two baseline specimens and on three specimens enhanced by LPB. The spectrum load block was repeated until specimen failure. One block of test spectrum represents 500 simulated flight hours. Damage tolerance testing was done on two specimens enhanced with LPB. Electric Discharge Machining (EDM) notches were cut into the cutout lip before testing. EDM provides a highly reproducible flaw with residual tension and cracks in the recast layer at the bottom of the notch. The specimens were enhanced using LPB, and 0.017 in. deep EDM notches were introduced. The specimens were then subjected to constant amplitude loading to grow the EDM notches to the required initial damage tolerance flaw size. Upon completion of the constant amplitude cycles, the specimens were subjected to the same spectrum that was used during the durability testing.

All the samples (without and with prior simulated damage) in the LPB processed conditions show no harm from such processing, and subsequently showed equal or better than the un-notched baseline specimen fatigue performance, even in the presence of simulated damage (Figure 9.8-4).

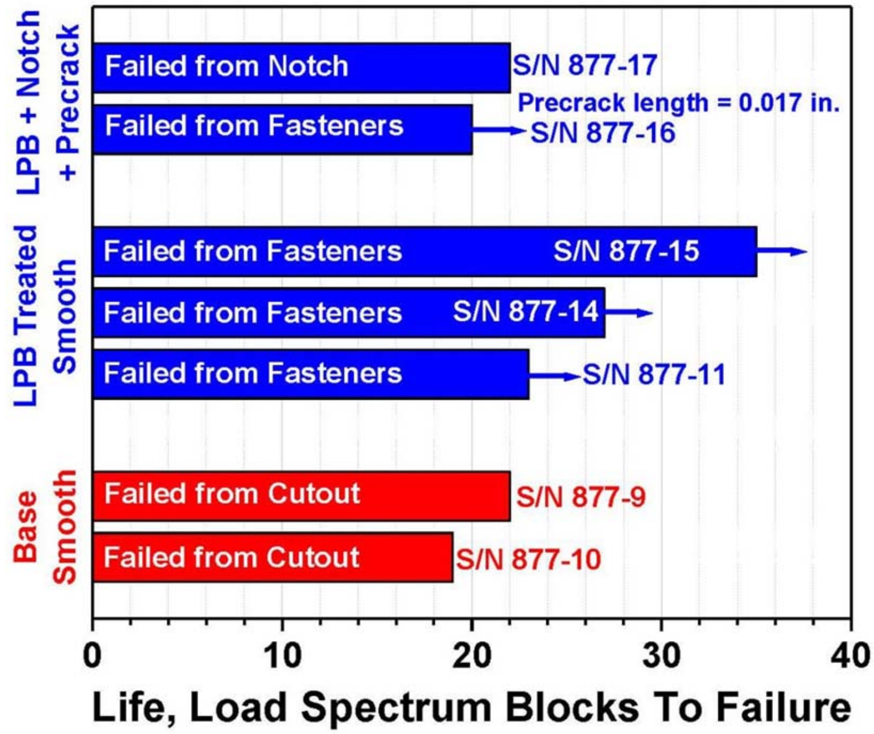


Figure 9.8-4. Spectrum Load Fatigue Test Results

The robustness of the LPB process and production processing bounds were established by finding a range of processing conditions within which RS distributions remain essentially constant, fatigue performance of processed parts show complete mitigation of damage, while the part distortion is kept within manufacturing tolerances. The tool and the LPB processing parameters are production ready. The next objective is to move the solution to the field, depot or third-party facility.

9.8.3. Hole Repair Solutions 101 – Structural Terminating Repair Solutions Using Cold Expansion Methods

Len Reid, Fatigue Technology

Cold expansion methods such as split-sleeve-cold expansion of holes, high interference-fit-expanded bushings and rivetless nut plates have played a major role in effecting terminating structural repairs; or at least greatly extending the fatigue life and inspection intervals, on military and commercial aircraft for many years. A number of these repair technologies are already employed on the F-16, A-10, C-130, P-3, KC-135, JSTARS, Navy E-6 and a number of helicopter platforms. However, many maintenance depots and operating units are not aware of the various options to repair damaged or discrepant holes that are available to provide long term or terminating repair solutions. The purpose of this technical effort is to review various hole repair options and show a number of applications where the cold-expanded bushing, nut plate, blind-threaded insert and panel-repair methods have been successfully incorporated to rapidly repair military aircraft in either depot or operational facilities.

Cold expansion methods/processes induce a residual compressive stress around a hole to shield it from the effect of cyclic loads and greatly improve the damage tolerance of fastened joints, lugs or other structural applications (Figure 9.8-5). ForceMate bushings install bushings at high interference fit to mitigate fretting and fatigue damage in attaching lugs but can also be an effective method to resize damaged and discrepant fastener or other holes in structure during repair (Figure 9.8-6). The ForceTec rivetless nut plate is also effectively an expanded bushing that can repair riveted nut plate installations where the fastener holes have become worn and /or may have initiated fatigue cracks that could compromise structural integrity or demand repeat inspections of the locations. If left unchecked, these cracks or repair scenarios could lead to more extensive repairs or ultimately major structural failure.

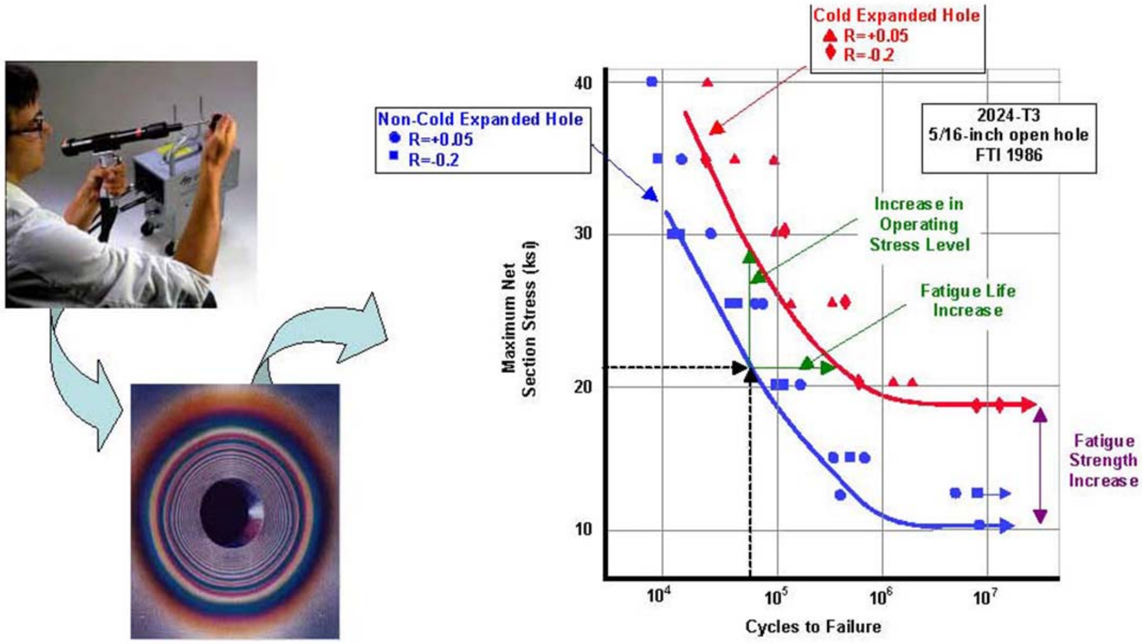


Figure 9.8-5. Fatigue Life and Fatigue Strength Improvements for Cold-Expanded Holes

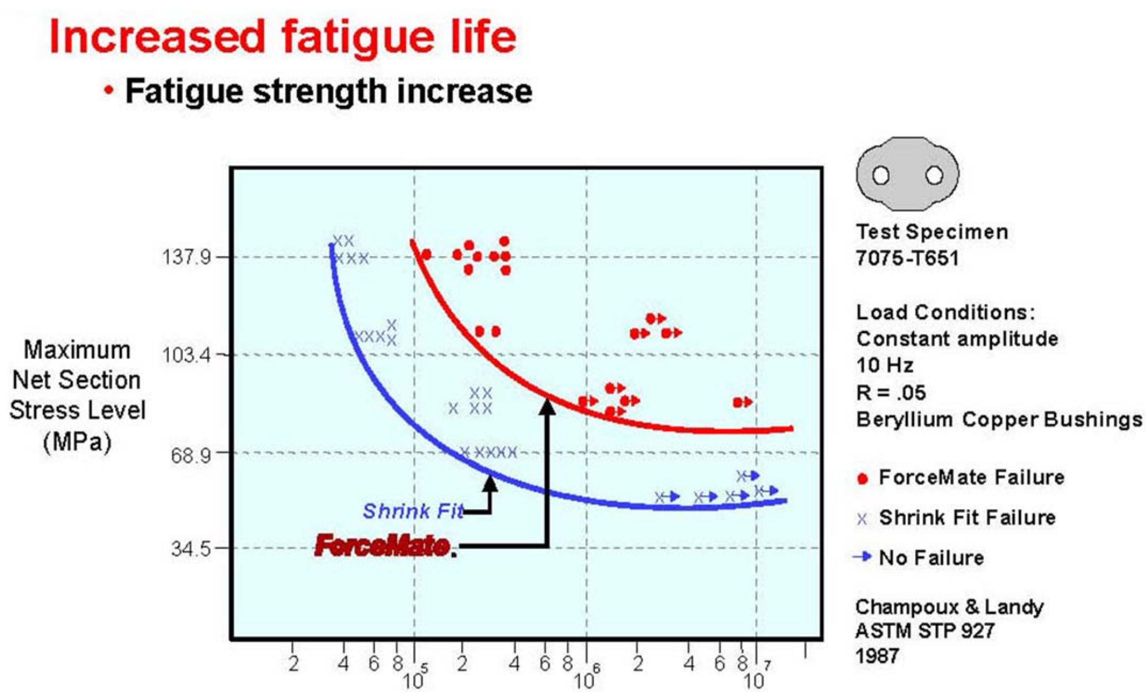


Figure 9.8-6. Expanded Bushing Method Benefits

This technical effort will review these structural life enhancement solutions as well as other solutions to replace structural fasteners such as Jo-Bolts or maybe damaged load bearing fasteners in a "one off" repair scenario. Examples of where they are being used on a number of military platforms to incorporate effective time saving repairs that enhance durability and damage tolerance and restore structural integrity will be given. The effectiveness of the repairs and their use as preventative structural enhancements have led to major cost savings and long term benefits in extending inspection intervals or providing terminating repair actions. In many applications they can be used to restore an aircraft to operational service rapidly to ensure optimum utilization of the asset.

9.8.4. Warthog Stamina: Enhancements to the New A-10 Wing

Paul N. Clark, Robert Pilarczyk, and Hazen Sedgwick, USAF-OO-ALC; and Scott Fields, The Boeing Company

In 2003, a decision was made to pursue a new wing for the A-10 weapon system. The original A-10 wing was designed with a service life of 6,000 hours. Shortly after production began, several improvements were implemented to extend the service life to 8,000 hours. With vigorous inspections and maintenance the average current operating time on the A-10s is approximately 9,500 hours with the high time aircraft exceeding 13,000 hours. A tradeoff study was performed comparing the business cases for continued inspection and maintenance versus procurement of a new wing and the new wing business case came out on top. The goal of the Wing Replacement Program was to make some localized design changes to known structural issues with thick-skin A-10 wing to enhance the service life of the new wings to 10,000 hours without scheduled structural inspections.

A brief review of the business cases will lead into the problem statement introduction along with a clear statement of the objective. Thirteen fatigue critical locations (FCLs) on the A-10 thick-skin wing configuration will be identified as the targets for the enhanced service life (Figure 9.8-7). Several other design changes that were implemented will be discussed including, but not limited to: component consolidation, manufacturing alternatives, material improvements, material substitutions, and surface treatments. A summary of the FCL life improvements and accompanying analysis will round off the technical effort (Figure 9.8-8).

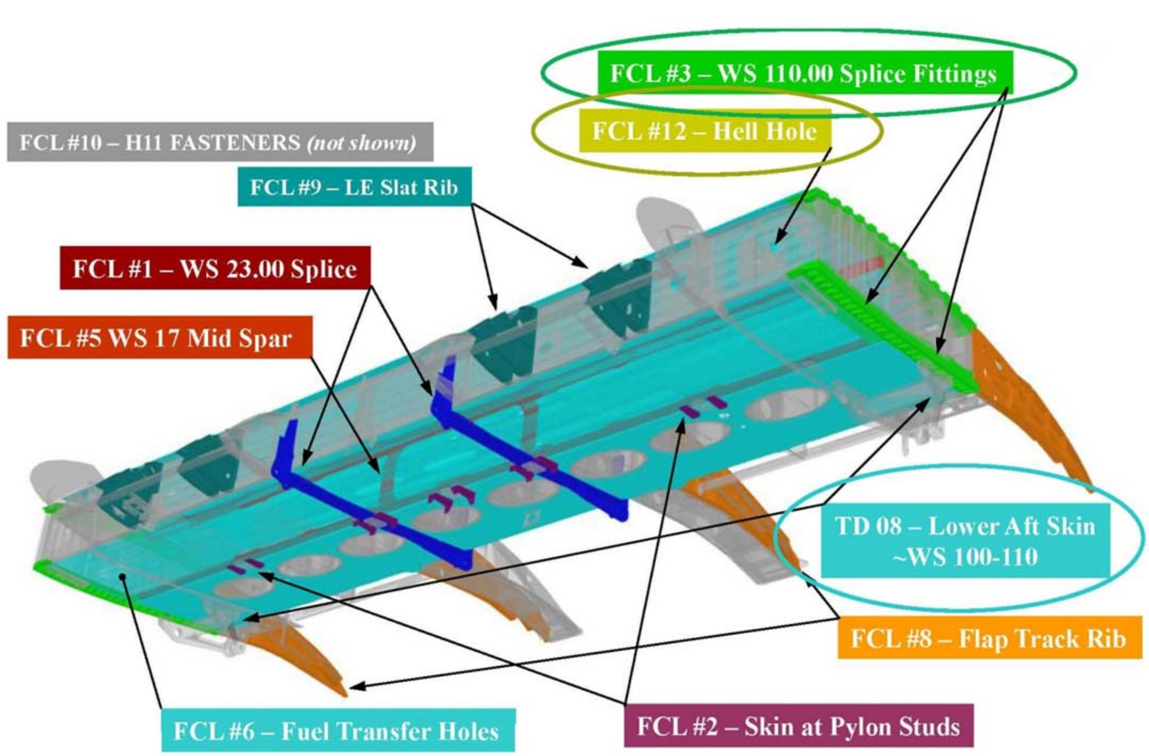


Figure 9.8-7. Fatigue Critical Locations of Center Wing Panel

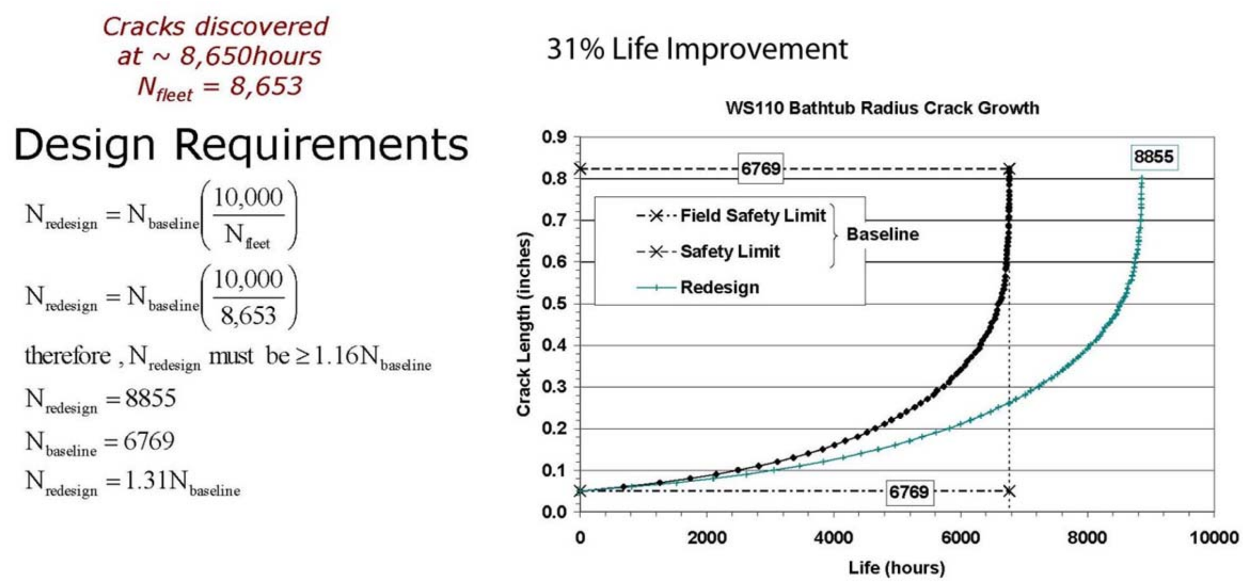


Figure 9.8-8. Life Improvements of FCL#3

9.8.5. Validation and Verification of Cold Worked Holes

William Ranson and Nam D. Phan, Direct Measurements Inc.

It is a widely known fact that residual stresses are one of the major contributing factors which would potentially impact the fatigue life of structural components. Residual stresses can occur due to either manufacturing, fabrication processes or intentionally engineered into structures in attempts to improve fatigue life. Equally important is that designers understand how to account for the potential effects of residual stresses on the life of structures or products. Cold working of fastener holes in aircraft structures is one method of fatigue life extension. This technical effort presents a method of residual stress measurement and documentation of the large number of fasteners in the wing sections.

An optical method has been developed to measure directly the hole diameters before and after cold working. The split sleeve method developed by Fatigue Technology Incorporated (FTI) specifies the percent of cold work by the allowable diameters pre and post cold work which is a measure of the retained expansion which can be used as a rapid and inexpensive qualitative method to obtain cold work information. Since the retained expansion depends on the degree of plastic deformation produced by the process, it becomes a very good approximate measure of the magnitude of residual stress. A test was designed to determine the optical sensor accuracy compared to an optical measuring microscope with wire EDM machined holes. The wire EDM produces a clean and precise rim around the hole while still producing a cut to the accuracy of 0.0002-inch. The measuring microscope has an accuracy of 1μm. At a magnification of 37.9x diameter readings are dependent upon the section of the hole that is measured (Figure 9.8-9). With a smaller deviation found using the measuring microscope, the more defined data can be correlated to the optical sensor device (Figure 9.8-10). The optical sensor compared to the measuring microscope has an accuracy of 0.0002-inch which is sufficient for a field measurement of retained expansion.

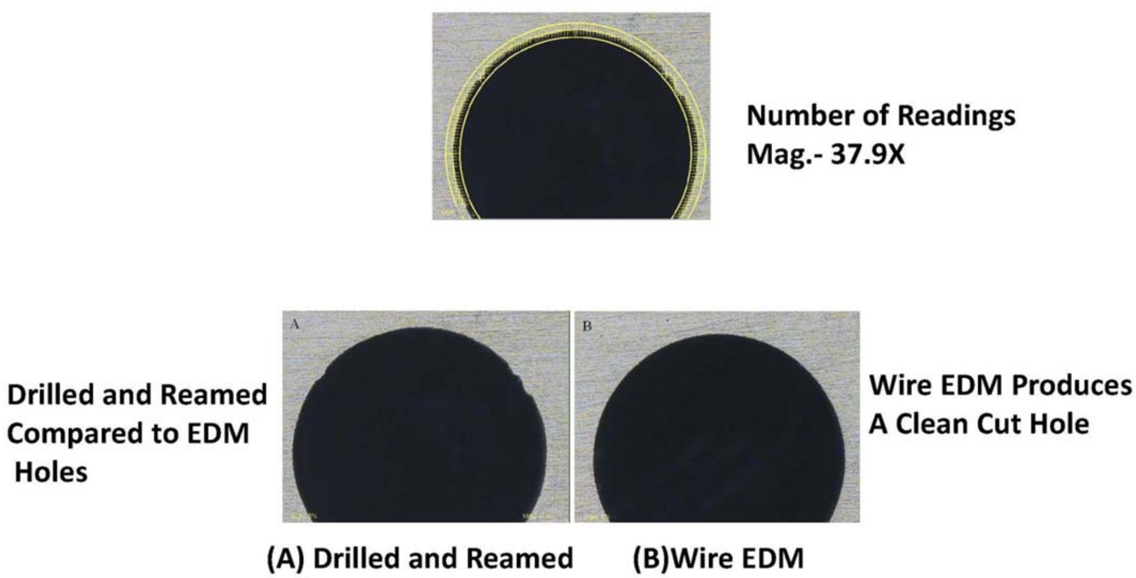


Figure 9.8-9. Smart Scope Zip 250 Microscope Images

Figure 9.8-10. Hand-Held Computer and Sensor

Documentation is obtained by establishing a database of fastener locations to be cold worked. As each fastener location is cold worked, the database automatically generates, as part of the administrative function, a date and time stamp along with the artisan and administrator login. Maximum and minimum limits of allowable percent of expansion and pre-cold-work holes are part of the database. Pre and post-cold-work diameters are recorded and compared to the allowable limits and the retained expansion records a GO/NOGO at that location. Three repeat measurements are allowed for a NOGO and once a GO is established that fastener location can no longer be accessed. The database also displays the complete record of cold-worked and non-cold-worked holes.

A comparison of the pre and post-expansion-diameters measurements between the measuring microscope and optical sensor are presented. As can be expected, some differences occur between the measurements which are a direct result of the asymmetric deformation around the hole.

9.8.6. Incorporating Laser Peening Residual Stress into a Holistic Life Assessment Approach

Craig L. Brooks, Scott Prost-Domasky, Thomas Mills, and Kyle Honeycutt, APES, Inc.

While the benefits of laser peening to aircraft structural integrity have been known for decades, reductions to aircraft maintenance costs have not been fully realized. Currently, certification of life improvements from laser peening is accomplished almost exclusively by testing, primarily because robust analytical methods do not yet exist. To take full advantage of laser peening benefits, DoD accepted design and maintenance practices such as those in the USAF Aircraft Structural Integrity Program (ASIP) demand a comprehensive approach to design, manufacture, and life cycle maintenance of their air vehicles' structural elements. This technical effort describes some results of an experimental and analytical modeling program that brings together three enabling technologies for taking full advantage of laser peening benefits: advanced life assessment methods, residual stress measurements (Figure 9.8-11), and non-destructive inspection techniques (Figure 9.8-12). The holistic life assessment approach described in this technical effort pursues the potential role of relaxation of residual stress fields to successfully predict not only the cycles required for coupon fracture but the intermediate crack length versus cycles "path" as well. Model performance has been measured against fractographic marker band data (Figure 9.8-13). Four methods for measuring residual stresses in laser peened coupons were evaluated, including an NDI tool that is already being used in the field.

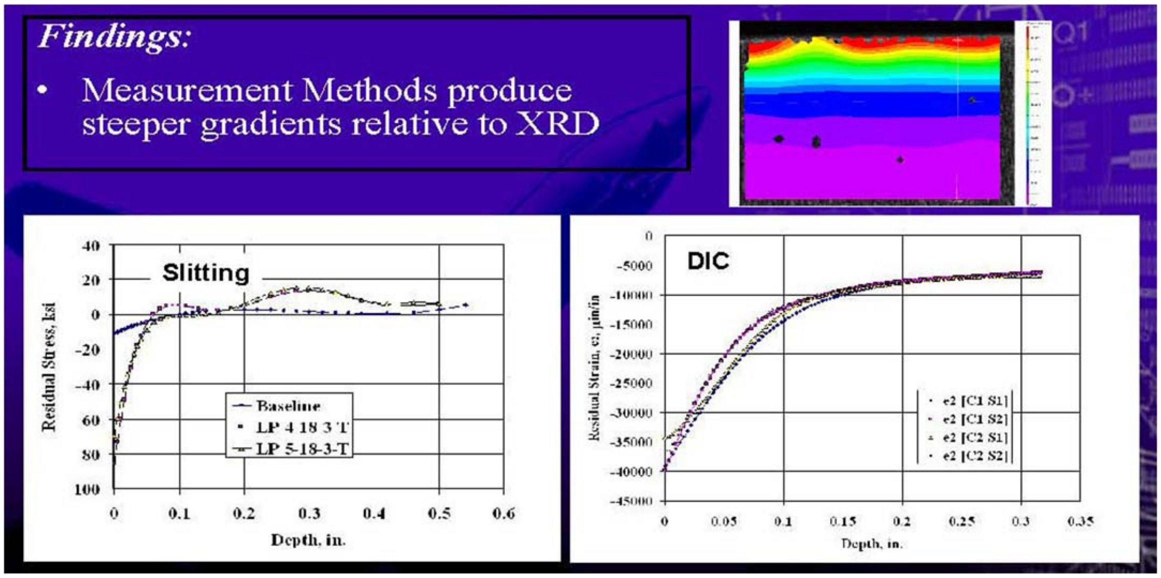


Figure 9.8-11. Slitting Method & Digital Image Correlation

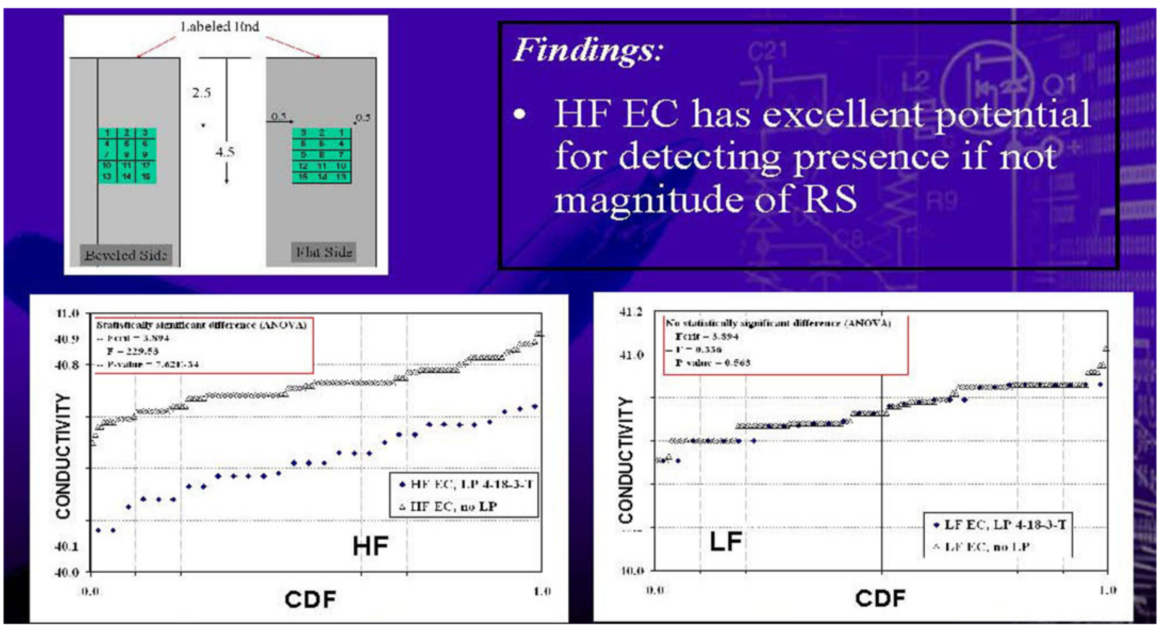


Figure 9.8-12. Eddy Current Results

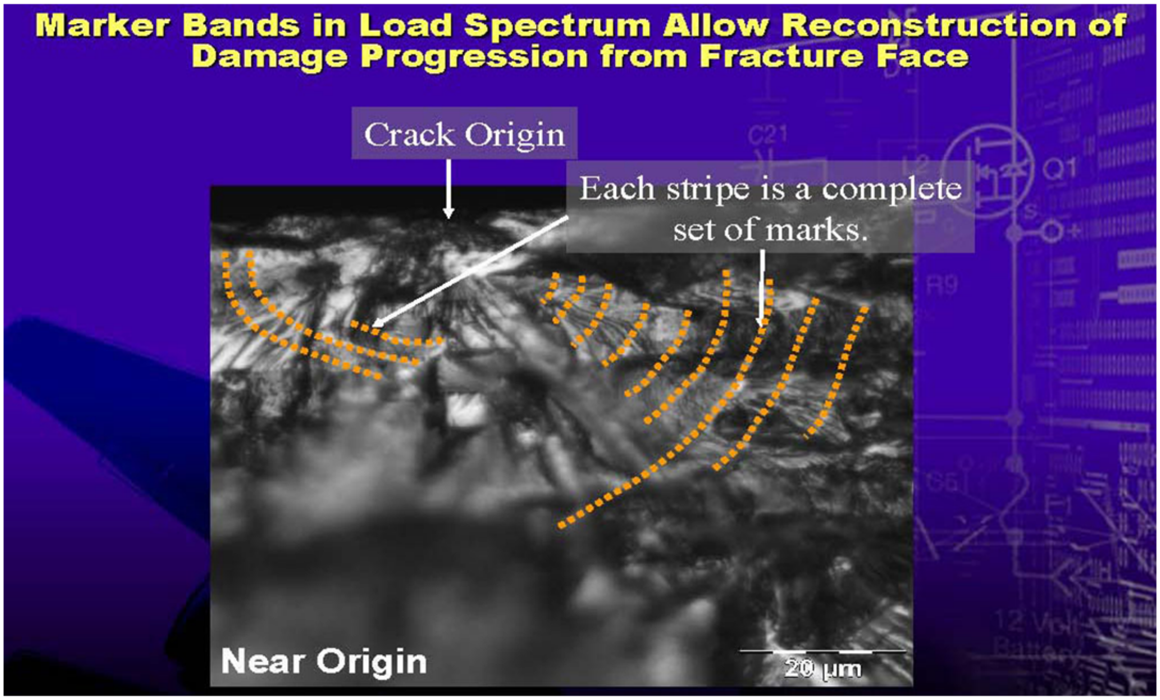


Figure 9.8-13. Fractographic Marker Bands

9.8.7. Application of Surface Residual Stresses for Durability and Damage Tolerance Improvements in F-22 Wing Attachment Lugs

Robert Bair & Wirt Garcia, USAF-F22 SPO; Jeffrey Bunch and Robert Weiss, The Boeing Company – F-22 Engineering

The F-22 program is evaluating life improvement methods to improve both the fatigue and damage tolerance lives of the wing attach lug radii. Full-scale-fatigue-test results of these areas have resulted in onerous, time intensive inspections of these non fail-safe critical locations. The F-22 program has pursued development of surface residual stress treatments for these locations as part of the F-22 Structures Retrofit Program to improve the durability and damage tolerance lives of these and other durability test life shortfall locations.

Engineered residual stresses have long been utilized to enhance the fatigue properties of metallic components. The F-22 program has conducted an investigation to establish a calculable life benefit of engineered residual stresses for applications to titanium structural components. Both glass-bead peening and laser-shock peening were selected for this investigation. Recent work has focused on both developing laser-shock peening for complex titanium hardware as well as performing sub-component testing to define the fatigue improvements available from both peening methods.

A scale-up method based on the ASIP building block approach was used in the laser-shock peening-development work (Figure 9.8-14). Residual stresses were measured on small test blocks, representative geometry blocks, subcomponent test articles, and full-scale test articles. The criteria for selection, residual stress data, as well as the chosen laser shock peening parameters will be discussed. Both laser-shock peening as well as glass-bead peening were applied to subcomponent test articles and cycled through a typical wing up-bending flight spectrum. Durability and damage tolerance benefits from both laser-shock and glass-bead peening have been calculated and will be discussed (Figure 9.8-15).

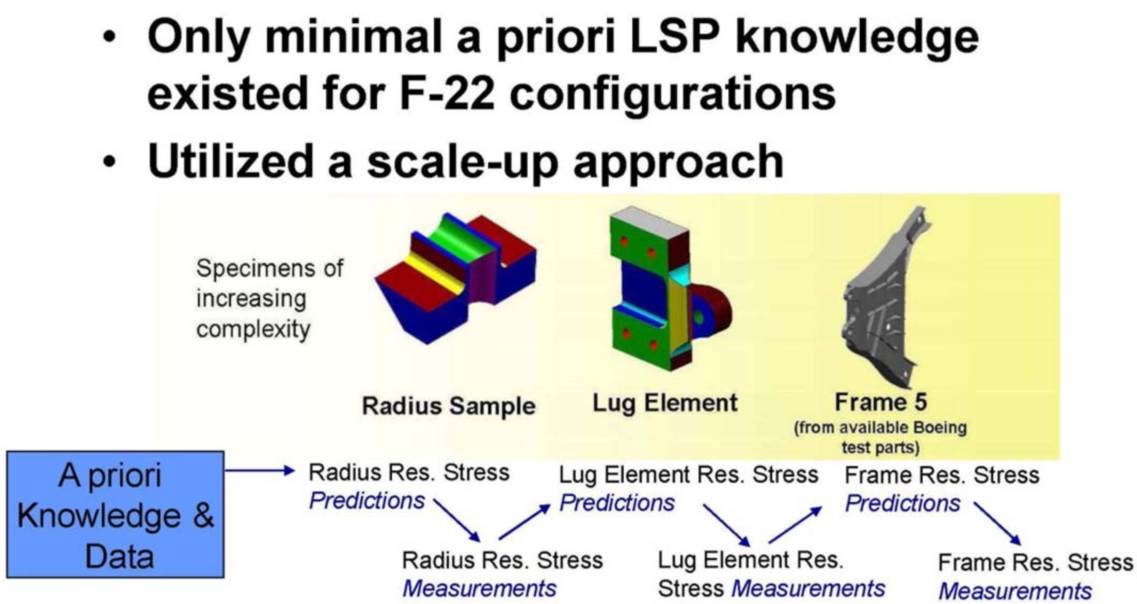


Figure 9.8-14. Experimental Layout

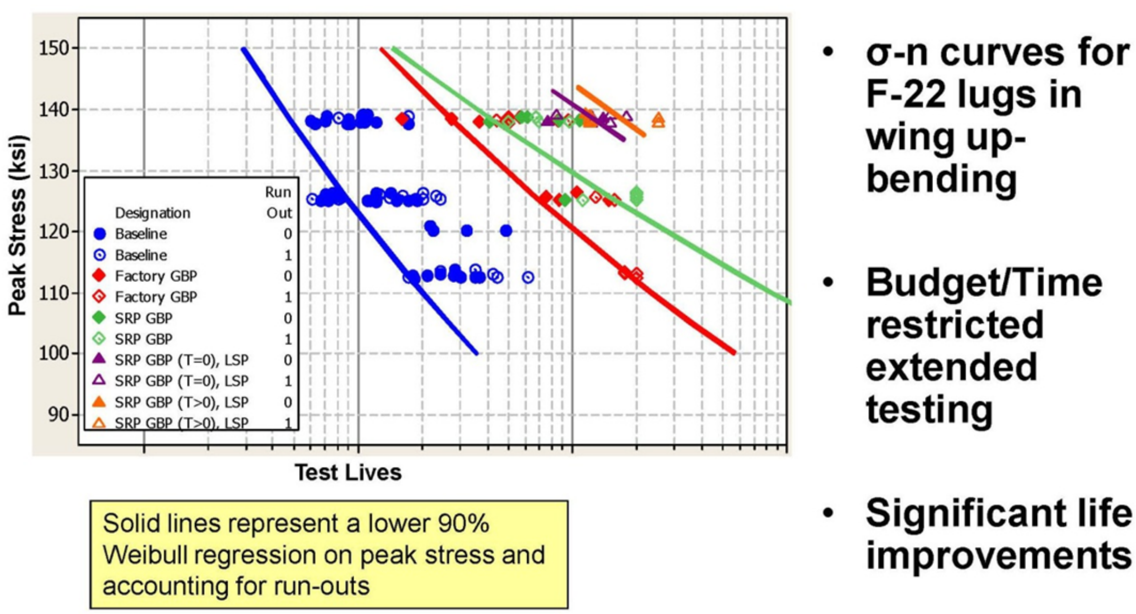


Figure 9.8-15. Stress-Life Summary

This technical effort will discuss the F-22 programs evaluation of both glass-bead peening and laser-shock peening. Test design, methodology and results will be presented to demonstrate the quantitative differences between the methods. Additionally, residual stress profiles will be presented to show the mechanics behind each process and the advantages and disadvantages of each method. Lastly, the technical effort will present the F-22 life benefit results based on completed lug element testing and some lug full-scale testing.

9.8.8. Robotic LPB Treatment of the P-3 Propeller Bore

N. Jayaraman, Lambda Technologies

The P-3 Orion turboprop aircraft has been in active service for almost 50 years. These versatile planes are used for surveillance, combat, cargo and research duties. The P-3 was developed to operate in the harshest environments and has seen action in Vietnam, Iraq, Afghanistan and Somalia. With a new focus on cost savings, the U.S. military has begun extending the required service life of its airborne fleet. By developing new technologies and maintenance methods, they hope to increase the lifespan of critical planes, such as the P-3, by 20 or more years. Each year that a model can see continuous service saves millions of tax dollars.

The P-3 is commonly used in marine environments and it needs to be resistant to corrosion pitting and stress corrosion cracking (SCC) to ensure proper operation. Any cracks caused by corrosion can propagate due to high-cycle fatigue during the basic operation of the aircraft. The aluminum propeller bore is vulnerable to this damage and is subject to a rigorous inspection and repair cycle to prevent it. The previous maintenance practice for fitting new bushings to the propellers was to use heavy shot peening to induce a layer of protective compression, followed by reaming and re-machining operations to restore the bore finish and propeller geometry. The P-3 propellers were originally designed for unlimited service life, however the loss of material during machining required the propellers to be scrapped after just three maintenance cycles.

To offset costs and viably extend the life of the P-3, NAVAIR, the Navy's aviation branch, began looking at other alternatives to shot peening that would reduce the scrap rate of propellers. They chose Low Plasticity Burnishing (LPB®). Developed by Lambda Technologies in Cincinnati, LPB induces a very deep, stable layer of compressive residual stress in the surface of a component. This layer of protection makes the piece dramatically more resistant to damage and can exponentially increase its fatigue life. The process works by rolling a high-hardness ball across the surface of the work piece to create beneficial compression. It leaves a mirror-like surface finish that facilitates inspection and doesn't require additional machining. Parts can be treated during manufacturing or after they have been in service and no alteration of the component's material or design is required.

The LPB processing is performed using basic CNC machines or robots, allowing for quick and easy integration into manufacturing processes. Computer control also guarantees repeatability and process regulation. The closed-loop pressure system for LPB adjusts in real time and exceeds six-sigma quality requirements. Each part is tracked by serial number, and SPC information is immediately available to quality control teams.

Complete turnkey robotic systems were developed by Lambda and installed at the PPI facility, along with two others at Cherry Point Marine Airbase and Warner Robbins Air Force Base. These three systems currently service the entire P-3 fleet. Figure 9.8-16 shows the basic robotic setup.

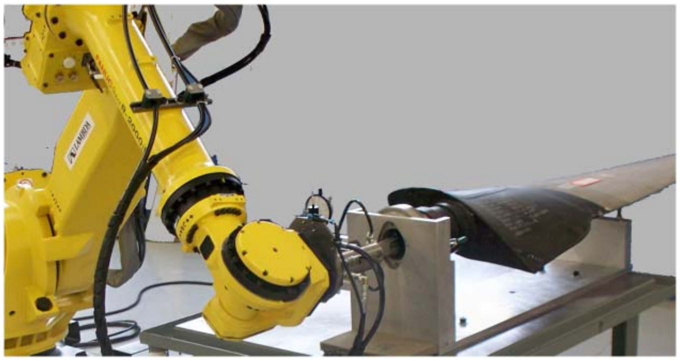


Figure 9.8-16. Robotic Processing of P-3 Propeller

The new LPB method didn't just replace the shot peening process. The compressive layer was deeper and more uniform with LPB. In aluminum applications, like the P-3 propeller bore, Lambda is able to design a protective stress field that is deeper than the deepest corrosion pits. Because the material is held in compression to a depth greater than any pits can reach, crack propagation from pits is eliminated. The fatigue life of the propeller bore is dramatically extended as the process also mitigates cracking from high-cycle fatigue. LPB only generates 3 to 4% cold work in the surface of the part, compared to almost 50% that can be caused by shot peening. The low cold-worked surface left by LPB is less chemically active than a shot-peened piece, providing another barrier against further corrosion damage.

By implementing LPB, PPI was able to improve the level of protection for the propeller bore and eliminate the reaming and machining steps, saving $1,000 per blade processed. Figure 9.8-17 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 $35,000 per propeller, millions can be saved in cost avoidance after just a few years of implementation.

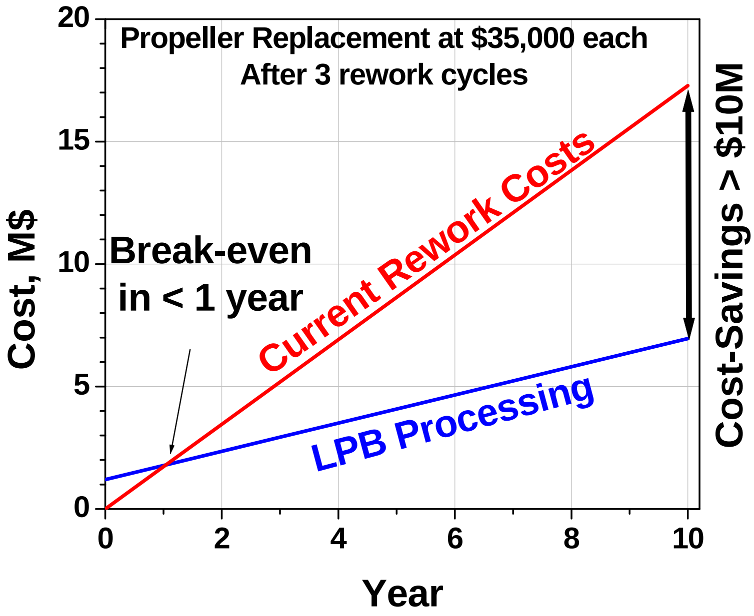


Figure 9.8-17. Estimated Maintenance Cost Savings with LPB

Point of Contact:

- Dr. N. Jayaraman, Director of Materials Research, Lambda Technologies Phone: 513-561-0883, www.LambdaTechs.com

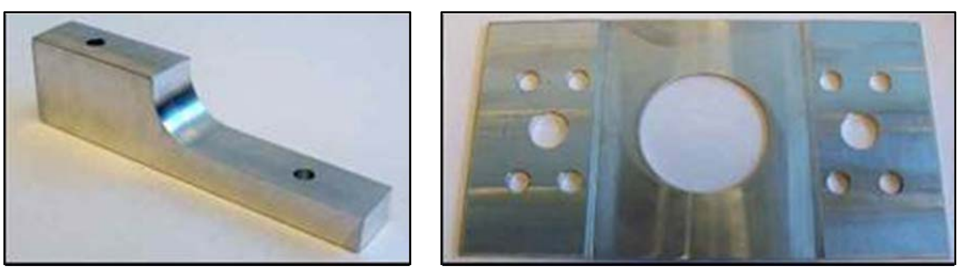
9.8.9. Life Extension of the P-3C Orion Floor Beams with LPB

N. Layaraman, Lambda Technologies

The US Navy and its foreign military partners are extending the service life of the legacy P-3C Orion aircraft. Fleet aircraft originally designed for 7,500 flight hours are currently averaging approximately 24,000. As part of the P-3 Service Life Assessment Program (SLAP), a fuselage full-scale fatigue test was conducted. Mechanical failure of the FS515 and similar surrounding underfloor beams occurred at roughly 20,000 maximum pressure cycles due to fatigue damage at stress concentrations from machined cutouts used to route control, hydraulic and electronic lines through the underfloor of the aircraft. Failure during flight could be catastrophic and cause loss of life and aircraft. In addition, inspection and maintenance costs adversely impact safety and readiness, and significantly increase the total cost of ownership and operation. A means of extending the fatigue life of the floor beams is required.

Low Plasticity Burnishing (LPB) has been accepted by the FAA for repair and alteration of civilian aircraft engines and structures. LPB has also been successfully implemented in several military engine and structural applications including robotic processing of the P-3C propeller bore at three MRO facilities. LPB can be applied to the floor beams on the aircraft during routine overhaul. This program was developed to establish the LPB process conditions for the floor beam and optimize the production engineering to mitigate damage, improve damage tolerance, and enhance fatigue performance. The goal of this effort was to demonstrate the LPB process and to document the improved fatigue performance on feature specimens of the P-3C FS515 floor beam. Successive phases of this program will adapt tool design and LPB processing parameters to actual components for production readiness for service at depot maintenance or a third-party facility.

Fatigue testing of feature specimens was performed to demonstrate the effectiveness of the LPB solution to achieve the required performance. Specimens were designed to simulate the critical failure areas of the P-3 floor beam. Figure 9.8-18 shows the two different geometries tested.



(Cracks 2-3) **(Cracks 4-5)**

Figure 9.8-18. Feature Specimen Designs

LPB provides fatigue life improvement corresponding to approximately one million flight hours per the NAVAIR 85th PMP for the P-3C as demonstrated in laboratory HCF testing. HCF test results are shown for both feature specimens in Figures 9.8-19 and 9.8-20.

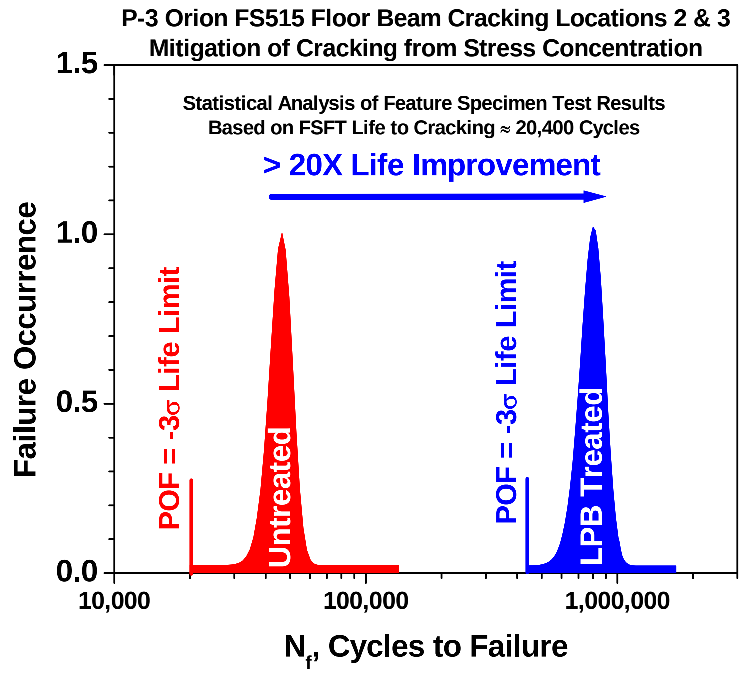


Figure 9.8-19. Fatigue Life Increase in Areas 2 & 3 Demonstrated by HCF Testing

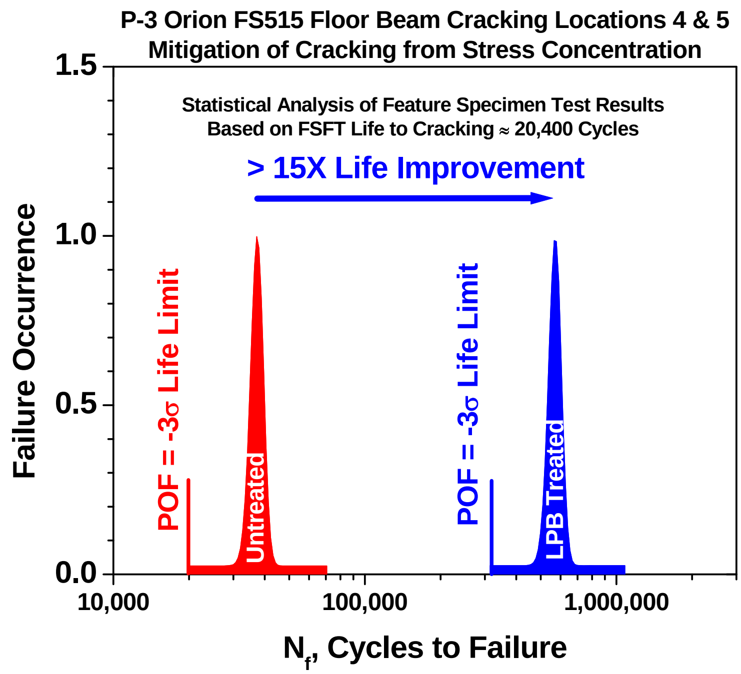


Figure 9.8-20. Fatigue Life Increase in Areas 4 & 5 Demonstrated by HCF Testing

Lifing analysis also predicts an increase by a factor of four in crack growth time to detectable size with the residual compressive stress provided by LPB. Combined, these results present the key benefits provided by LPB processing. Safe service life of the aircraft is not limited by the fatigue life of the FS515 and similar floor beams. In addition, the inspection interval of these floor beams can be increased, lowering the cost of operation and increasing aircraft time on wing and fleet readiness.

Point of Contact:

- Dr. N. Jayaraman, Director of Materials Research, Lambda Technologies Phone: 513-561-0883, www.LambdaTechs.com

9.8.10. Robotic LPB Application for MRO of a Main Landing Gear

N. Jayaraman, Lambda Technologies

The main landing gear of certain commercial and military aircraft are prone to stress corrosion cracking (SCC) and fatigue. Airlines spend millions of dollars each year on inspection and repair to ensure safety and remain compliant with airworthiness directives. In some cases, landing gear may require replacement to solve the problem, adding a costly unscheduled maintenance cycle.

300M HSLA steel is often used in landing gear because of its high strength and high fracture toughness. However, 300M steel is highly susceptible to corrosion fatigue and SCC, which can lead to catastrophic consequences for aircraft landing gear. Shot peening and plating of the landing gear are used to suppress corrosion fatigue and SCC with limited success. A method that will produce deeper compression in critical regions of landing gear will provide a dramatic improvement in foreign object damage (FOD) tolerance, corrosion fatigue strength and SCC susceptibility.

Lambda Technologies, located in Cincinnati, has developed a solution to cracking in landing gear that is cost-effective, reliable and permanent. Low Plasticity Burnishing (LPB®) was created as a mechanical means to combat FOD, fatigue, fretting-induced cracking, and SCC. LPB induces a deep, stable layer of residual compression in the surface of a part, mitigating crack initiation and growth.

Tension in metallic components induces crack formation. By producing a deep layer of high-magnitude compression, LPB deters SCC crack initiation and fatigue crack propagation, from undetectable micro cracking up to visible damage from FOD or corrosion. Alternative solutions involve changing the alloy to a tougher one or completely redesigning the component. With LPB, there is no need to alter the material or design.

The fatigue performance of LPB processed 300M steel test samples was compared to those in a shot peened or baseline condition. Samples were tested with active corrosion from a 3.5% salt solution, FOD simulated by a semi-circular EDM notch, and both corrosion and damage together. HCF tests were performed under constant amplitude loading on a Sonntag SF-1U fatigue machine at ambient temperature (~72F) in a four-point bending mode. LPB treatment dramatically improved the HCF and corrosion fatigue performance with and without a simulated defect. Figure 9.8-21 shows the results for all three conditions.

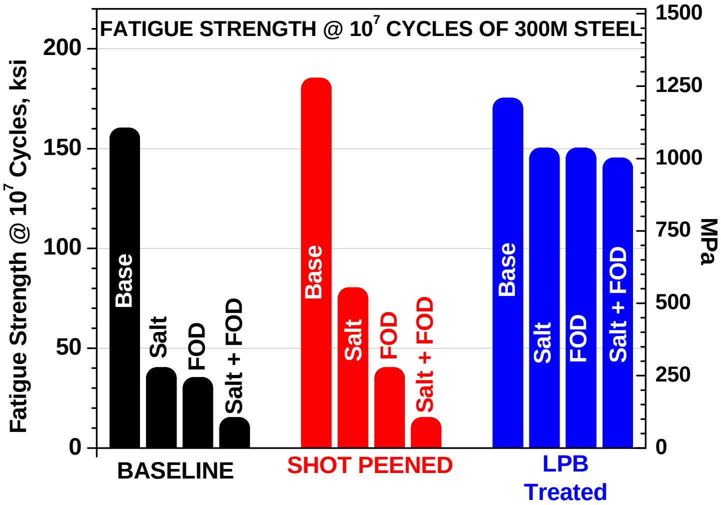


Figure 9.8-21. Summary of Fatigue Results

LPB also reduced the surface stress well below the SCC threshold for 300M steel, even under high tensile applied loads, effectively suppressing the SCC failure mechanism. Both untreated and LPB treated specimens were SCC tested at 1,033, 1,137 and 1,240 MPa (150, 165 and 180 ksi) static stress in alternate immersion in a neutral 3.5% NaCl solution. The load was monitored and the time to failure recorded. SCC testing of LPB treated landing gear sections was terminated after 1,500 hrs without failure, compared to failure in as little as 13 hours without LPB treatment. Figure 9.8-22 shows the SCC test results for various static loads.

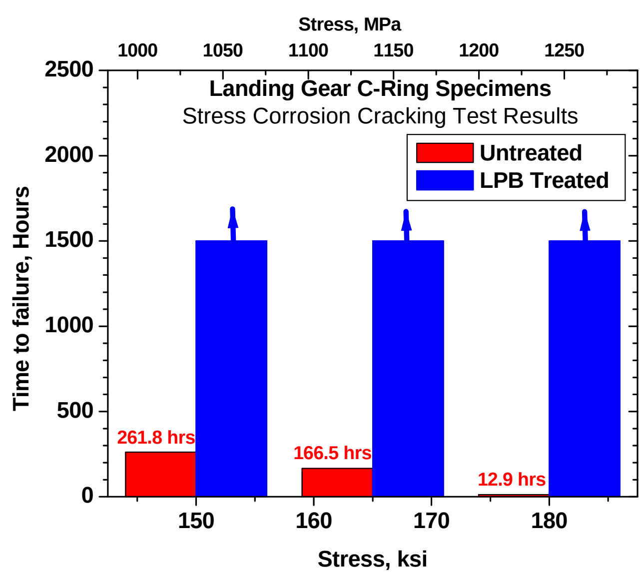


Figure 9.8-22. Summary of SCC Test Results

The patented LPB process offers an innovative approach to a common problem. Unlike traditional surface treatments such as shot peening and deep rolling, LPB provides protection with very little cold work. This makes the compressive layer very stable even when the landing gear sees mechanical overload and leaves a mirror-like surface finish that is less chemically active. The process mitigates cracking in a single maintenance cycle and never requires reapplication. The LPB processing is performed using basic CNC machines like mills, lathes and robots, making integration into production easy. Robotic LPB systems allow for treatment in the field and pieces like landing gear can be processed in the hangar without removal, allowing planes to spend more time in service.

Point of Contact:

- Dr. N. Jayaraman, Director of Materials Research, Lambda Technologies Phone: 513-561-0883, www.LambdaTechs.com

9.8.11. Fatigue Life Extension Through Cold Expansion of a Longeron Fastener Hole

James Greer, Center for Aircraft Structural Life Extension (CAStLE)

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 a USAF office in charge of the Aircraft Structural Integrity Program (ASIP) for a USAF aircraft. The objective of the work was to determine the effects of cold-expansion (CX) at a fastener hole in the dorsal longeron of the aircraft. The specimen configuration is shown in Figure 9.8-23. A corner crack was inserted at the 12 O'clock position of the fastener hole.

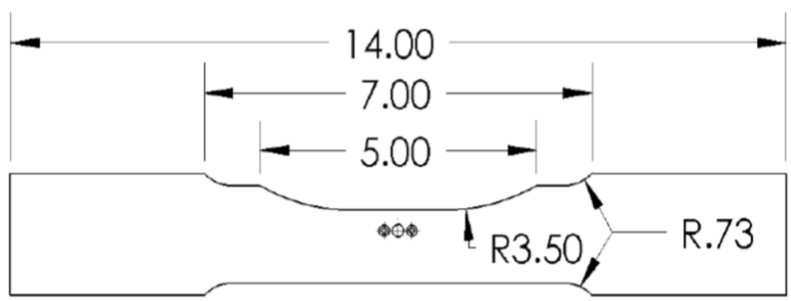


Figure 9.8-23. Test Specimen with Fastener Hole; Satellites Holes are for Installation of a Nut Plate

Cold expansion of the fastener hole significantly retarded hole-bore crack growth regardless of whether the flaw was pre-existing or introduced after CX. All of the bore-cracked CX specimens were tested to a minimum of 3,000 equivalent flight hours (EFH) of spectrum loading, and none showed significant crack growth. Two other CX specimens were tested to 10,000 EFH and 14,000 EFH with no significant crack growth. The only CX specimen taken to complete ligament failure withstood approximately 19,200 EFH of spectrum loading prior to ligament failure. The average life of a non-CX, bore-cracked specimen was about 900 EFH.

The stresses in the loading direction are depicted in the finite element model of Figure 9.8-24.

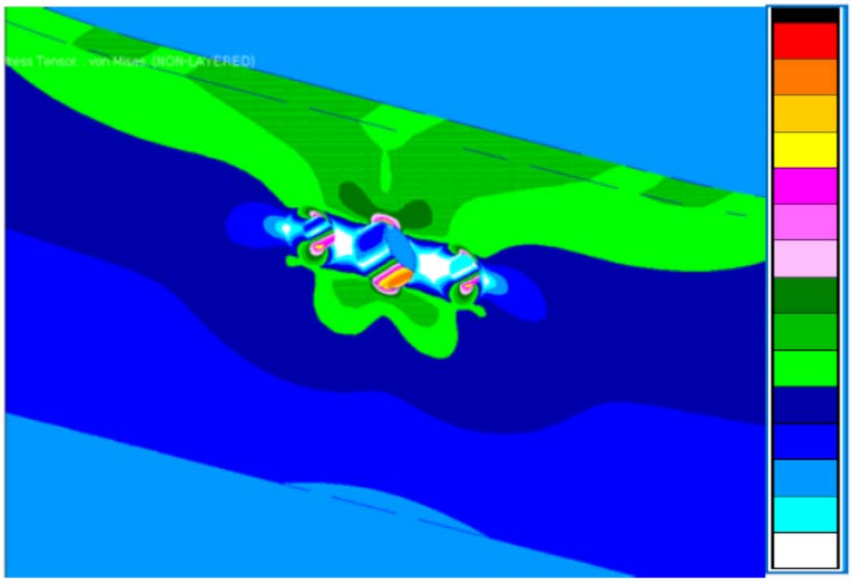


Figure 9.8-24. Finite Element Model of the Test Specimen Under Load

The inability to grow cracks from CX holes frustrated attempts to generate beta and shut-off overload ratio (SOLR) corrections to use in AFGROW to model crack growth from these holes. However, beta corrections were successfully developed for this fastener 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 damage tolerance analysis (DTA) at this location, although ignoring SOLR was uniformly conservative (as expected) in terms of crack life.

The residual tensile stresses created at the edge of the longeron by CX were cited as a potential concern early in the program. Hole CX appeared to reduce ligament life by about 25% in the presence of an 0.050 x 0.050in free-edge corner crack, but in none of the bore-crack (only) test specimens did a natural crack ever nucleate at the free edge.

Point of Contact:

- James Greer, Center for Aircraft Structural Life Extension (CAStLE), (719) 333-3618

9.9. REPAIR CONCEPTS

9.9.1. Positive Pressure Bonding of B-1 Dorsal Longeron Repair Doubler

Soo Oh, Chi Chan and Thompson Nguyen, The Boeing Company – Defense, Space & Security

The B-1 Dorsal Longeron Repair Program effort entails the repair of cracks found on B-1 aircraft where the pair of forward dorsal longerons splices to the aft dorsal longeron (Figure 9.9-1). Analysis of the dorsal longeron cracks determined that a permanent repair of the longerons was needed to preclude more extensive cracks and replacement of the longerons. The B-1 Engineering Team developed and tested dorsal longeron repair design and a prototype repair was subsequently developed and incorporated on an operational B-1 aircraft.

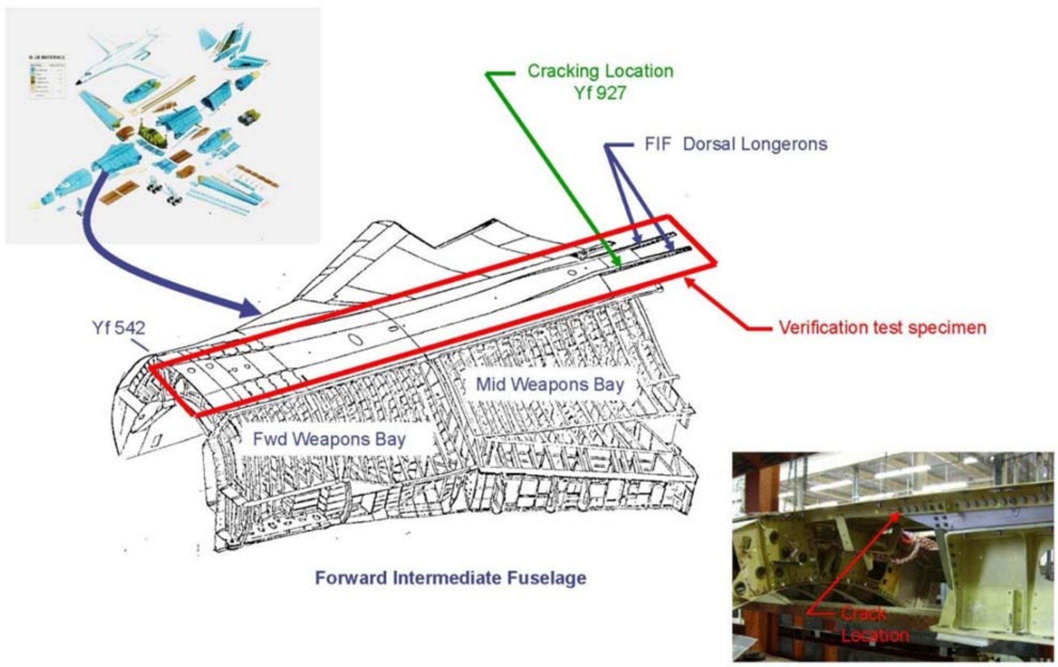


Figure 9.9-1. B-1 Dorsal Longeron Cracking Location

The initially developed dorsal longeron repair procedure includes the bonding of a doubler using a vacuum bag to apply pressure to achieve a structural bond (Figure 9.9-2). The large periphery of the repair doubler presents difficulties in achieving a consistent vacuum pressure throughout the adhesive cure cycle. Further, with the vacuum bag bonding process, the vacuum pressure is limited to 20 in.- Hg to minimize porosity in the bondline. Vacuum pressure is required to be maintained at an optimum level of 19.0 ± 1.0 in-Hg throughout the 8-hour bonding process. Pressure must be high enough to ensure proper mating of an 1/4” thick steel doubler to the longeron contour surface; however, it also must be low enough to avoid adhesive boiling or high porosity in the bondline that is very detrimental to the bond strength. With the vacuum bagging method, the repair team has often encountered non-conformances due to vacuum pressure loss which requires engineering analysis, disposition and customer concurrence prior to acceptance or rework. In cases of severe vacuum pressure loss, bonded doublers had to be removed and replaced.

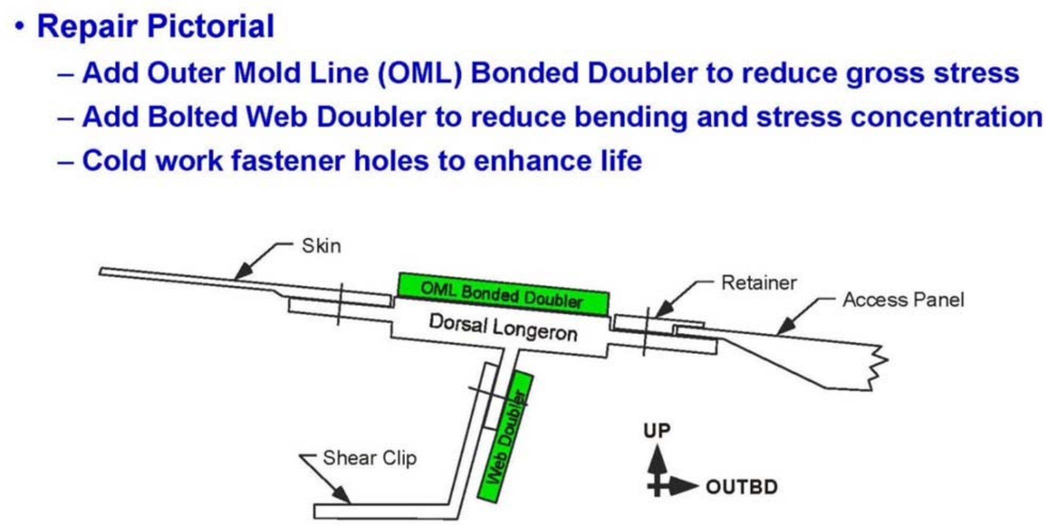


Figure 9.9-2. Typical Longeron Section with Repair Doublers

An alternative bonding method using positive pressure was developed to provide a more consistent and higher bondline pressure, improved reliability and repeatability, reduced labor (Figure 9.9-3), and increased bond strength than the vacuum bonding process. This method, utilizing an inflatable air bladder to apply pressure on the doubler during bonding, employs direct positive pressure versus vacuum induced pressure. Higher pressure can be applied to ensure good contact between the doubler and longeron contour surface without fear of adhesive boiling or high porosity due to high vacuum pressure (Figure 9.9-4). Bond strength is also improved. This method is more reliable and repeatable because applying pressure no longer relies on a perfectly sealed vacuum bag and airtight structure, eliminating the bonding issues associated with vacuum pressure loss.

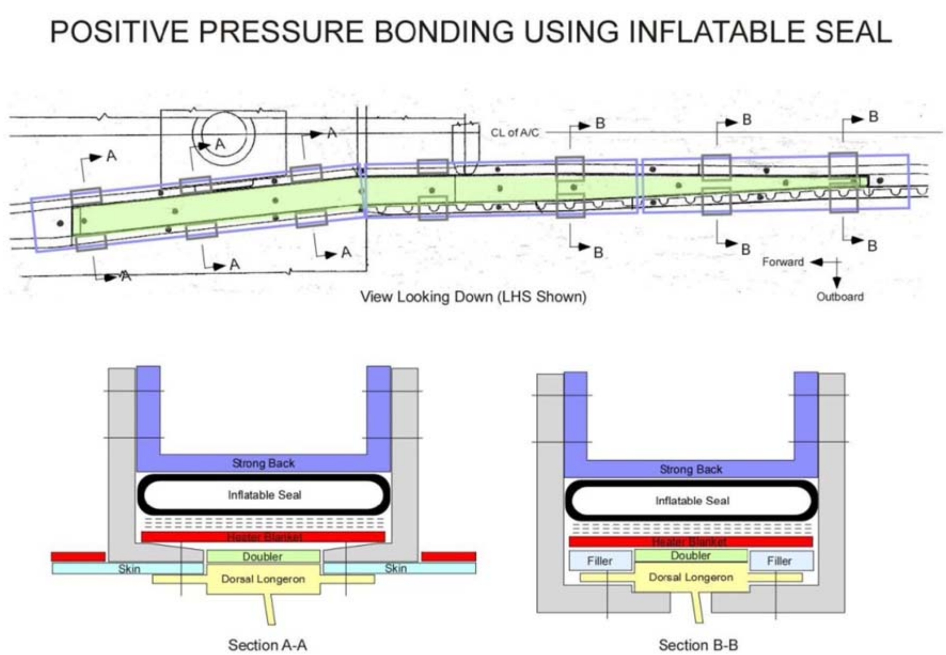


Figure 9.9-3. Conceptual Design of Improved Process

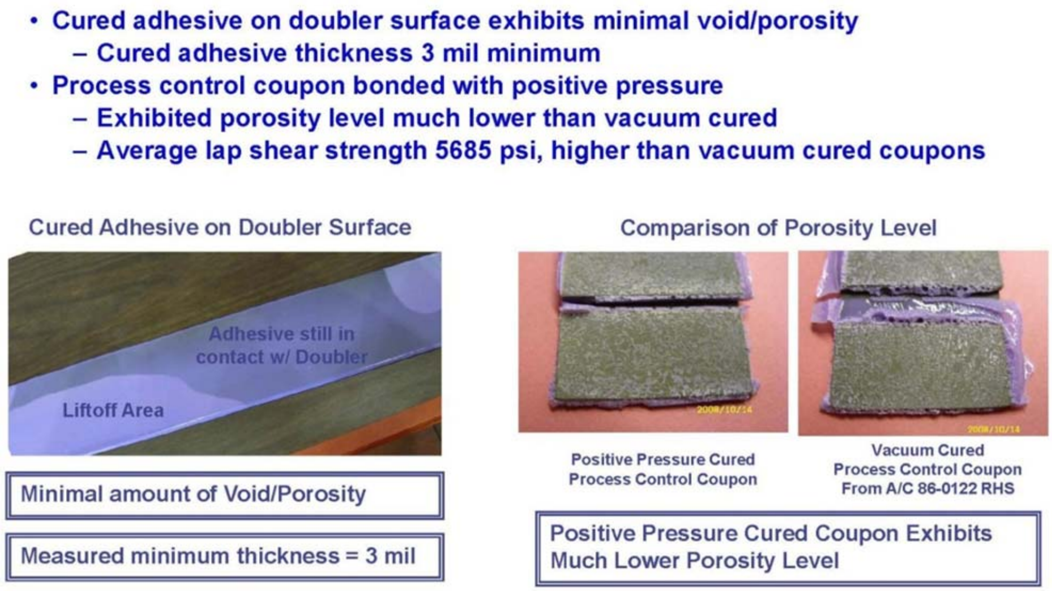


Figure 9.9-4. Improved Porosity Level for Positive-Pressure-Cured Coupons

This technical effort will present a repair development for the cracked B-1 dorsal longeron with an emphasis on positive pressure bonding tool and process development.

9.9.2. Ensuring the Airworthiness of Aging Fleets – ABDR and Extended Life Requirements

Randal Heller, Southwest Research Institute and Mark Thomsen, USAF-OO-ALC

Recent failures in the commercial and military-retired fleets suggest opportunities for improvement in structural lifecycle management. Both the 2005 Chalk's Ocean Airways and 2002 C-130A Air Tanker crashes were the result of structural failures originating at structural repairs. This technical effort seeks to investigate some of the unique challenges to structural integrity stemming from extended-life operation as well as mission/usage evolution.

Repairs designed prior to the advent of modern damage tolerance methods may be installed on our oldest aircraft. Additionally, repairs may have been designed with an original service goal in mind, and may no longer be adequate for extended-life operations. Oftentimes, documentation for legacy repairs is nonexistent due to BRAC-related base realignments, or the fact that ABDR-type repairs may have been installed in the AOR without full documentation (Figure 9.9-5). What is the responsibility of the ASIP engineer in these circumstances? Are aircraft inducted into the depots being given a "receiving inspection" to identify potentially inadequate repairs made to fatigue-critical structure?

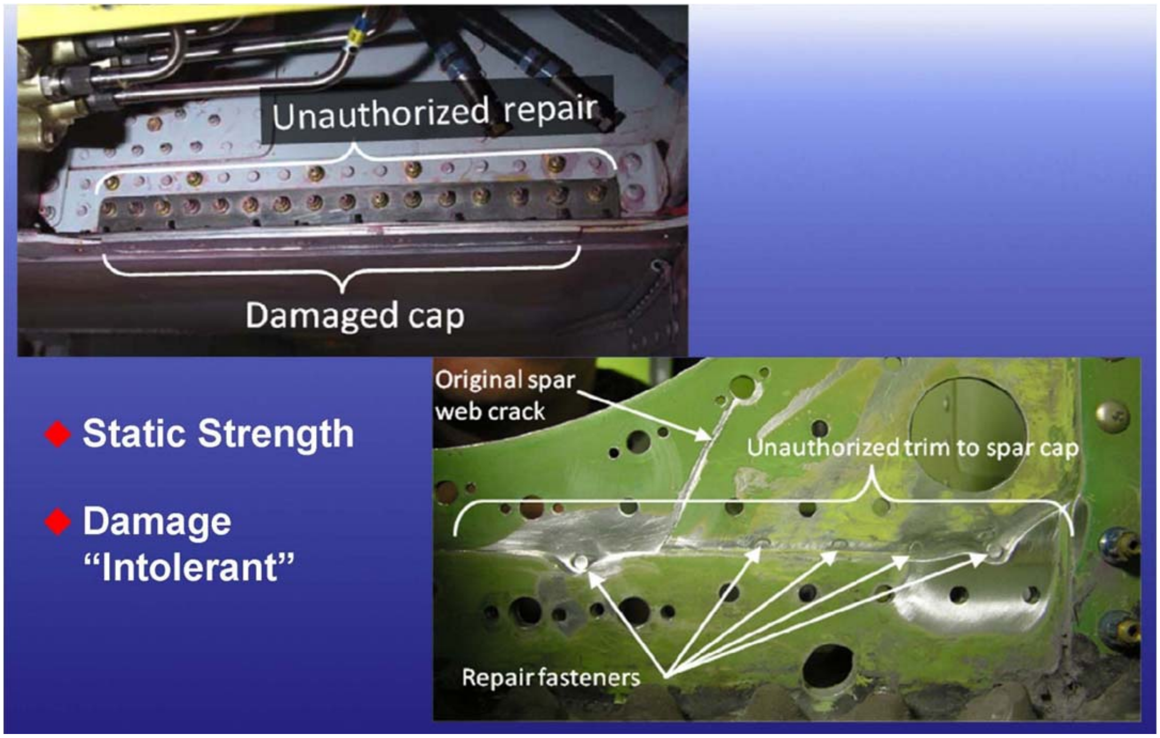


Figure 9.9-5. Challenge – Undocumented Repairs

The ASIP engineer is faced with many challenges pertaining to existing repairs: undocumented rework, damage "intolerant" approved repairs, and repairs designed to a previous life requirement or spectrum. Air Force structural engineers don't always have adequate training in damage tolerance principles as it pertains to structural repair design.

The challenges of continued airworthiness are only compounded when aircraft leave the USAF inventory to FMS or other-governmental-agency fleets. Frequently, no formal lines of communication exist to communicate evolving structural health data generated from fleet inspections or SLEP testing programs. In some cases, data are communicated at an annual Weapon System Review or through a TCG, but is the right information being communicated to the right individuals? Would safety be enhanced through the adoption of a notification system similar to the Airworthiness Directive process used by the FAA? Several steps can be taken to reduce risk and enhance safety: increase training for military engineers in damage tolerance principles, provide continuity on weapon systems through the civilian and contractor workforce, thoroughly review existing T.O. repair manuals for damage "intolerant" repairs, identify existing repairs to fatigue critical structure by ACI or TCTO, enhance communication with all operators/stakeholders, coordinate a consistent approach for funding and implementation through ASC/EN or other responsible organization, and measure the progress toward these goals at the annual ASIP reviews.

9.9.3. Bonded Repair of a Commercial Airframe Curved Fuselage Panel: Design/Analysis, Installation and Validation Test

Russell Keller, The Boeing Company – Research & Technology

The use of bonded repairs on military aircraft has demonstrated several advantages in their application over conventional bolted repairs. Bonded repairs eliminate additional stress concentrations because conventional bolted repairs require fastener holes for attachment thereby introducing stress risers.

In cases where composite patches are used, bonded repairs have the additional advantage of allowing the plies to be tailored in the direction of the desired stiffness for the applied loads and stress fields at the repair site. The objectives of a recent research effort were (i) to demonstrate the capabilities of the Composite Repair of Aircraft Structures (CRAS) software developed by Boeing under USAF funding to quickly and accurately design/analyze bonded repairs for fuselage structures (Figure 9.9-6) (ii) to demonstrate the application of bonded repair to skin crack damages, and (iii) to validate a bonded repair on a commercial fuselage curved panel under bi-axial loading. This technical effort will show the results of the design and analysis of composite and metallic bonded patches on a Boeing 727 curved fuselage panel (Figures 9.9-7 and 9.9-8) and their validation using the test results (Figure 9.9-9) recently performed at the FAA Technical Center FASTER facility.

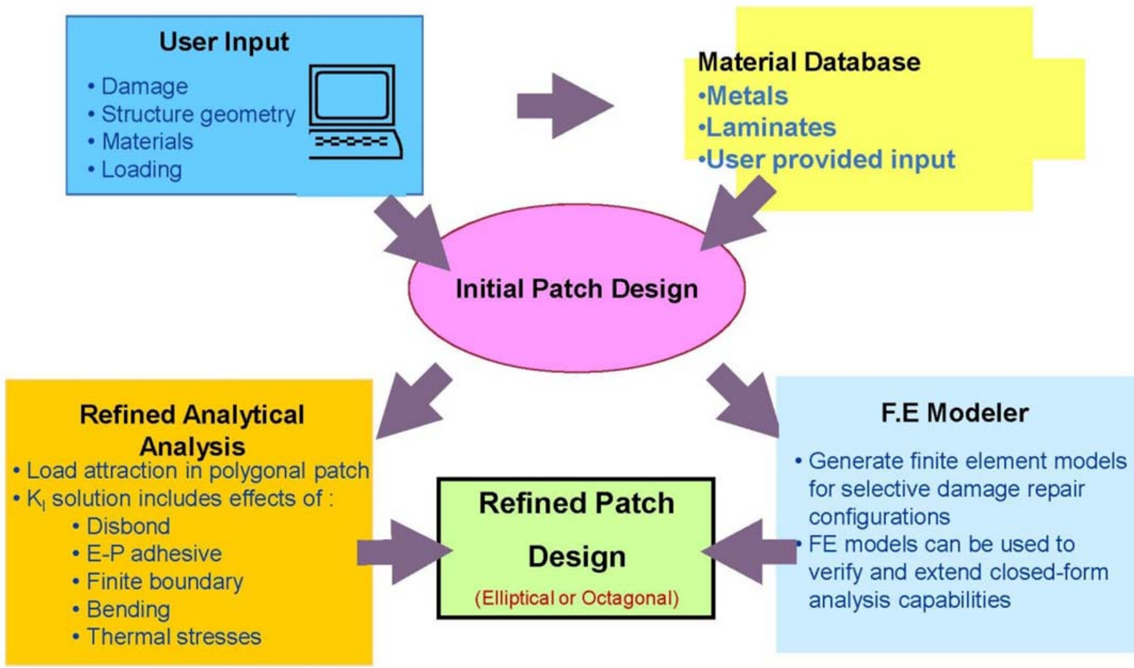


Figure 9.9-6. Composite Repair of Aircraft Structures (CRAS)

Panel Length	~125 inch
Panel Width	~73 inch
Panel Radius	~74.1 inch
Number of Frames	6
Number of Stringers	6
Frame Spacing	20 inch
Stringer Spacing	9.5 inch
Skin Thickness	0.040 inch

Figure 9.9-7. Test Article

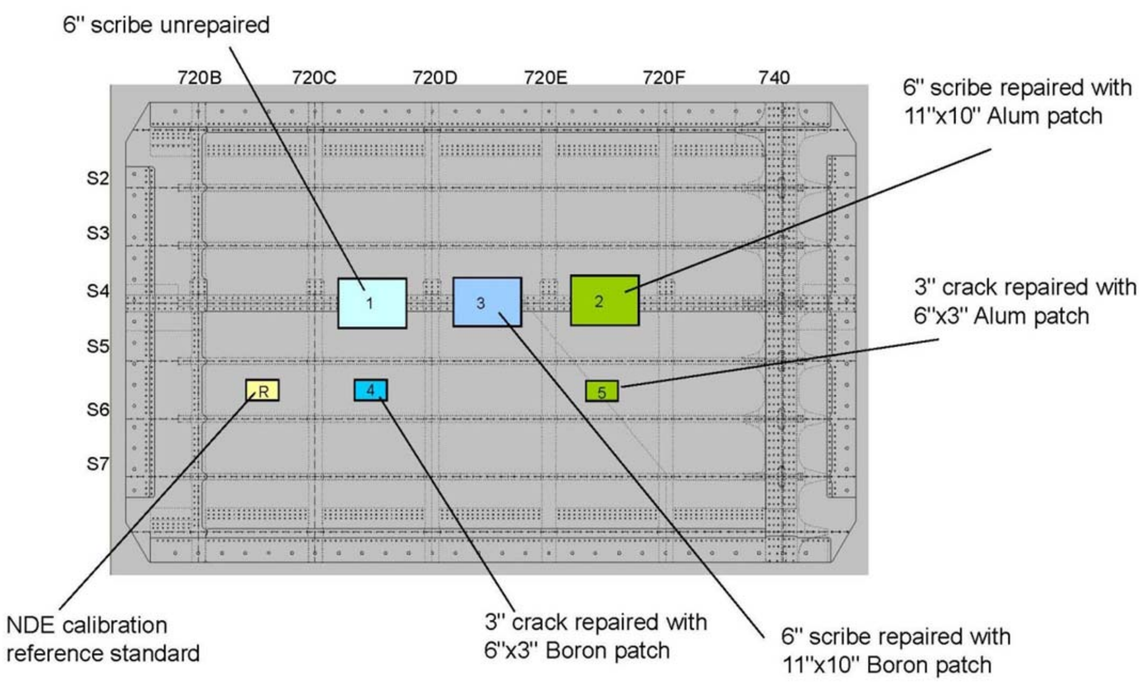


Figure 9.9-8. Test Article Damage/Repair Sites

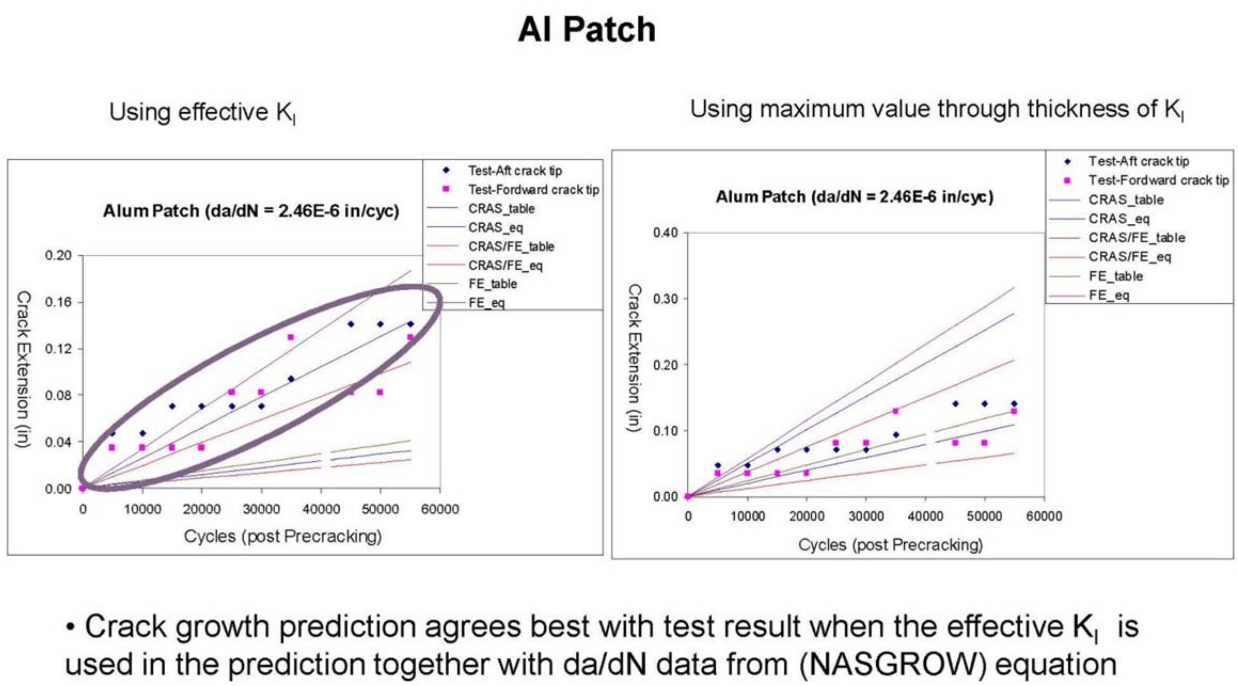


Figure 9.9-9. Test-Analysis Correlation of Crack Growth Curve

9.9.4. Fatigue Damage Tolerance of Adhesively Bonded Repairs

John Bakuckas, FAA – William J. Hughes Technical Center

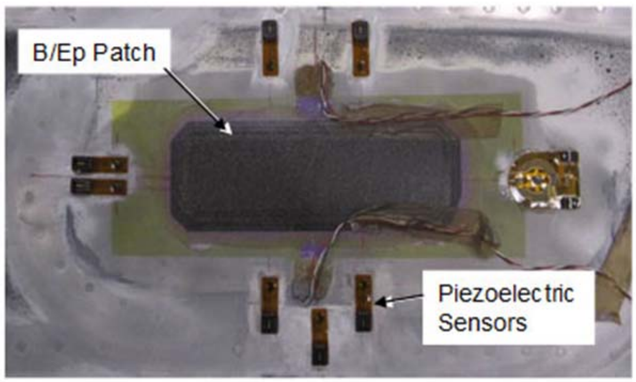
Adhesive bonding technology, using composite and metallic patches, offers an efficient and cost-effective approach to airplane structural repairs. Compared to conventional, mechanically fastened metallic repairs, bonded repairs have no stress concentrations due to holes, are less damaging to the parent material since no drilling or machining are required, and are more aerodynamically and structurally efficient. The application of bonded repairs has been studied primarily in the military sector where durability and damage tolerance aspects have been demonstrated. However, several technical challenges need to be addressed before bonded repair technology is generally accepted and implemented in both military and commercial primary structural applications. Currently, credit is typically not provided in certification programs of bonded repairs for slowing crack growth or restoring residual strength. Of primary concern is the ability to predict the fatigue behavior and ensuring the durability of bonded repairs.

The Federal Aviation Administration (FAA) and The Boeing Company have teamed in an effort to study the fatigue and residual strength performance of bonded repairs using boron/epoxy (B/Ep) and aluminum patches. The FAA Full-Scale Aircraft Structure Test Evaluation and Research (FASTER) facility is being used to conduct structural tests of various damage/repair scenarios on several metallic fuselage panels. The program objectives are to characterize the fatigue performance of bonded repairs under simulated service load (SL) conditions and to determine if the repair patches meet strength, deformation, and damage tolerance requirements in residual strength tests. The major focus areas are tools for evaluating and monitoring the bond-line integrity over the life of the part.

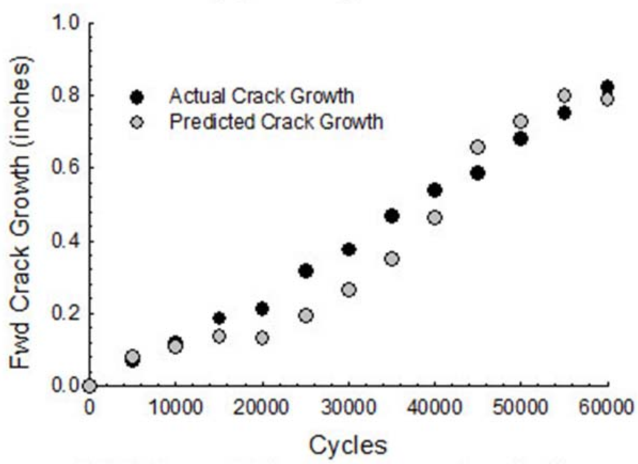
This program follows several phases to study different damage scenarios and corresponding repair configurations. On November 10, 2009, the first phase of structural testing test was completed.

These initial results revealed that properly designed and installed bonded repairs are durable and effective over long periods of fatigue and exceed typical design service goals of transport category airplanes. In addition, bonded repairs can successfully contain large damage under severe loads in excess of ultimate load requirements.

A more recent study focused on detection methods that may be used to assess bond integrity using commercially available nondestructive inspection (NDI) methods and a prototype structural heath monitoring (SHM) system. For this, repair patches were made intentionally deficient and contained imperfections to allow damage growth in order to assess the abilities of analytical methods and monitoring systems. The prototype SHM system was used to detect crack length at each of the repair locations as shown in Figure 9.9-10 (a). In general, results shown in Figure 9.9-10 (b) reveal that crack growth was readily detected in the patches and could be reconstructed by the SHM system.



(a) SHM System



(b) Fatigue Behavior Reconstructed by SHM System

Figure 9.9-10. Structural Health Monitoring (SHM) to Detect Crack Growth

Fatigue performance was significantly reduced for the repairs having a weak bond in which there was considerable strain redistribution and rapid crack growth. Representative results of the full-field hoop strain in the vicinity of a weak bond repair measured after several fatigue cycles using a digital image correlation system are shown in Figure 9.9-11. The patch boundary and initial defect are indicated in the figure. The white regions are areas where data could not be processed because of interference from strain gage wires. Strain increased in the panel skin along the edges of the repair patches and decreased slightly in the repair patch region. Along the notch centerline, strains were slightly in tension at the beginning (0 cycles) of the fatigue test and then increased substantially during the test, as indicated by the red fringe patterns after 20,000 cycles. Details of the hoop strain variation in the vicinity of the patch along three vertical sections are shown in the plots. As shown, after 20,000 cycles the strain gradient was quite high in a narrow band of approximately ±0.5 inch around the crack (section 1); over this short distance, the strain ranged from 2,000 to 12,000 με. This would suggest load redistribution in the skin from perhaps local disbonding in the crack region. However, there were no indications of disbond growth using the commercially available NDI methods. The condition and quality of the bonds will be further investigated using more advanced NDI and teardown evaluations.

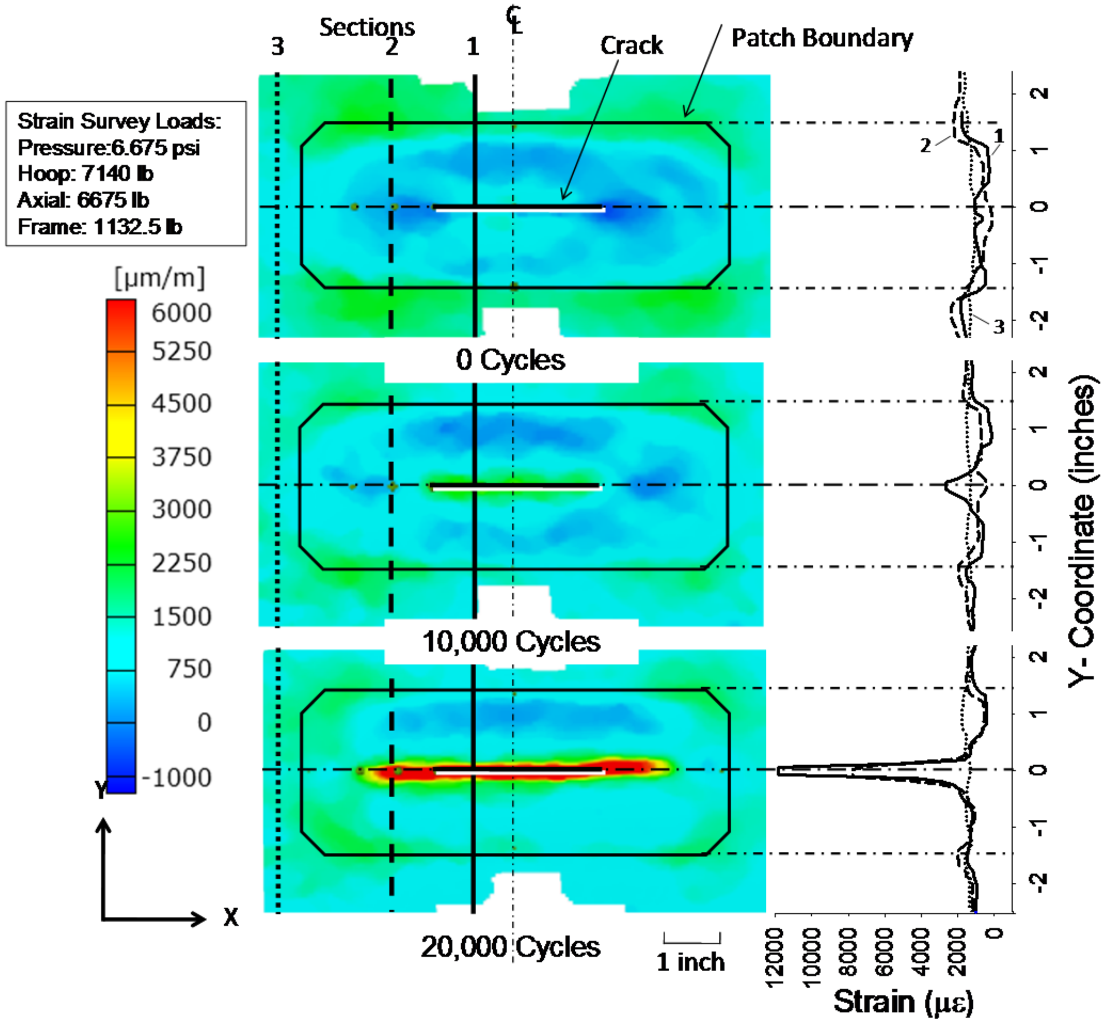


Figure 9.9-11. Strain Redistribution in a Weak Bond Repair

Points of Contact:

- John Bakuckas, FAA William J. Hughes Technical Center, 609-485-4784
- Ian Won, FAA Transport Airplane Directorate, 425-227-2145
- Bud Westerman, The Boeing Company, 206-662-3867

9.9.5. Survey of Repairs, Alterations, and Modifications for Widespread Fatigue Damage

John Bakuckas, FAA – William J. Hughes Technical Center

Fatigue of aircraft structure has long been recognized as a significant threat to the continued airworthiness of airplanes. This is because even small fatigue cracks can significantly reduce the strength of airplane structure. A phenomenon referred to as widespread fatigue damage (WFD) is identified as a severe degraded condition that threatens continued airworthiness of airplanes. All airplanes will reach this degraded WFD condition if cycled long enough. A major concern of WFD is that fatigue cracks are initially so small that they cannot be reliably detected with existing inspection methods and can lead to sudden catastrophic structural failure.

To address this safety concern, the Federal Aviation Administration (FAA) issued a Notice of Proposed Rulemaking (NPRM) on WFD in April 2006. The WFD NPRM proposed that for certain transport category airplanes, design approval holders establish the period of time for which it is demonstrated that the maintenance program is sufficient to preclude WFD, in baseline airplane structure, as well as in most repairs, alterations, and modifications (RAM). Comments to the NPRM suggested that inclusion of RAM in WFD assessments should be deferred until additional information is gathered. The FAA concurred that WFD assessments be focused on baseline structure only. However, the FAA is taking a proactive role and is assessing the need for addressing RAM for WFD through an effort with the Airworthiness Assurance NDI Validation Center (AANC).

The goal of this RAM research project is to provide data to better understand the risks that RAM may pose for WFD. Surveys were conducted on retired airplanes at aircraft salvage locations and on in-service airplanes at the operator's heavy maintenance locations to determine the quality and condition of RAM, as shown for example in Figure 9.9-12. An airplane sample plan was developed to target the number and models of airplanes that would represent the in-service U.S. registered transport category fleet. A total of 154 transport airplanes were surveyed, representing a statistical sample of the population of the U.S. fleet. Of the airplanes surveyed, a total of 2,584 RAM were identified and their details (e.g., quality condition, size, and location) were recorded in a searchable database. FAA engineers working with the database (235,144 entries) to access the RAM have made the following observations:

- U.S. Fleet has 5,014 aircraft with an average age of 16 years old
- Most airplane models have similar rates of RAM accumulation
- Approximately 50% of the RAM were type B and C
- Less than 0.6% of the RAM were of poor quality repairs
- Approximately 15% of the RAM were considered large (greater than 500 in^2)
- Approximately 31% of the RAM were repairs due to damage likely from blunt impact to the fuselage

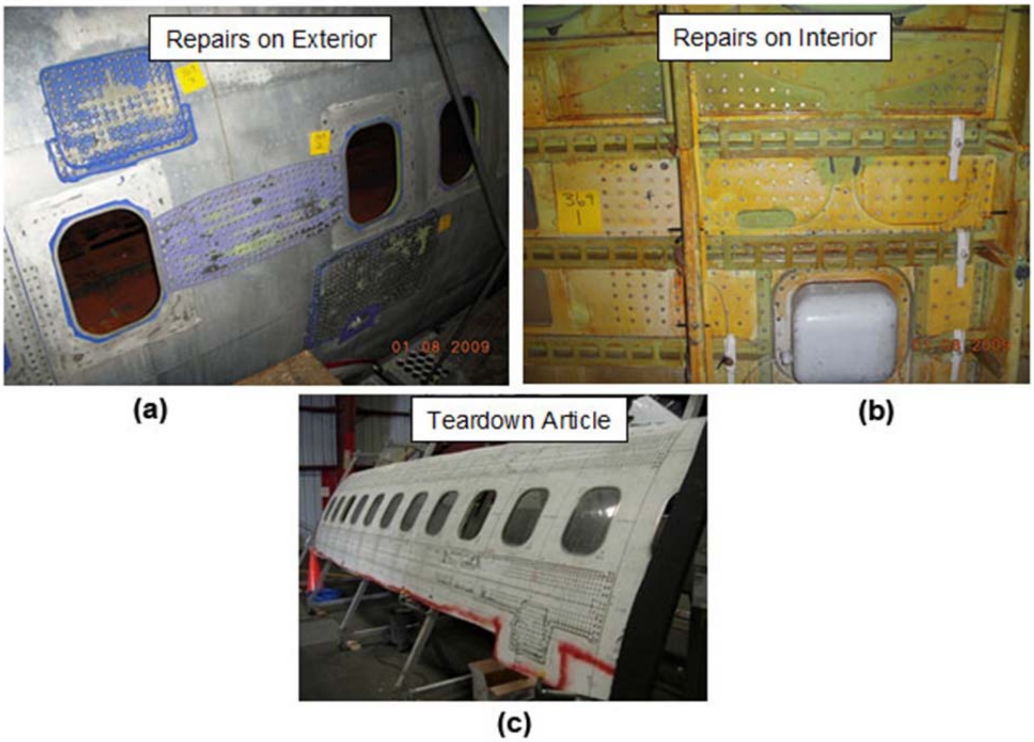


Figure 9.9-12. Examples of Repairs: (a) Exterior Surface of In-Service Airplane; (b) Interior Surface of In-Service Airplane; (c) Teardown Article for Damage Characterization

From the survey and database assessment, 22 RAM considered large (greater than 500 in^2) and susceptible to possible WFD have been identified for further teardown inspections. Over the first half of fiscal 2011, AANC will complete teardown inspections on collect samples. Once the teardown activities are completed, the FAA will evaluate the data and estimate the WFD risk in the U.S. domestic transport category fleet. Then, a determination will be made whether additional rulemaking is necessary.

Points of Contact:

- Mike Bode, Sandia National Laboratories– AANC, 505-843-8722
- Walt Sippel, FAA Transport Airplane Directorate, 425-227-2774
- Doug Ostgaard, FAA Transport Airplane Directorate, 425-227-2253
- John Bakuckas, FAA William J. Hughes Technical Center, 609-485-4784

9.10. REPLACEMENT CONCEPTS

9.10.1. A Finite-Element-Based Stress Intensity Solution for Cracks in Fiber Metal Laminates

Doug Miller and Hank Phelps, Lockheed Martin Aero – Ft. Worth

Fiber Metal Laminate Materials (FMLs) have been shown in the literature to provide improved damage tolerance and residual strength behavior when compared to conventional aluminum materials (Figure 9.10-1). The most mature FML in production application today is GLARE, which consists of alternating layers of 2024-T3 sheet and Glass Fiber/Epoxy unidirectional tape (Figure 9.10-2). When a fatigue crack initiates in the aluminum layers of the laminate, the high tensile strength Glass Fiber/Epoxy layers remain intact as a controlled delamination occurs at the interface, effectively "bridging" the crack and reducing the crack tip stress intensity. This reduction in stress intensity produces a longer crack growth life and increases residual strength. While many researchers have investigated this crack bridging mechanism, it is the authors' opinion that the analytical model that has most completely described the physics of the problem to date is the solution presented by Alderliesten. However, the process used by Alderliesten to apply the solution to a crack growth analysis precludes the use of standard crack growth analysis engines such as AFGROW. The purpose of this work is to describe the development of a finite-element-based FML stress intensity solution which reproduces the physics of Alderliesten's method, and to show how crack growth analysis predictions generated using AFGROW compare to test data.

Figure 9.10-1. Evolution of Fiber Metal Laminates

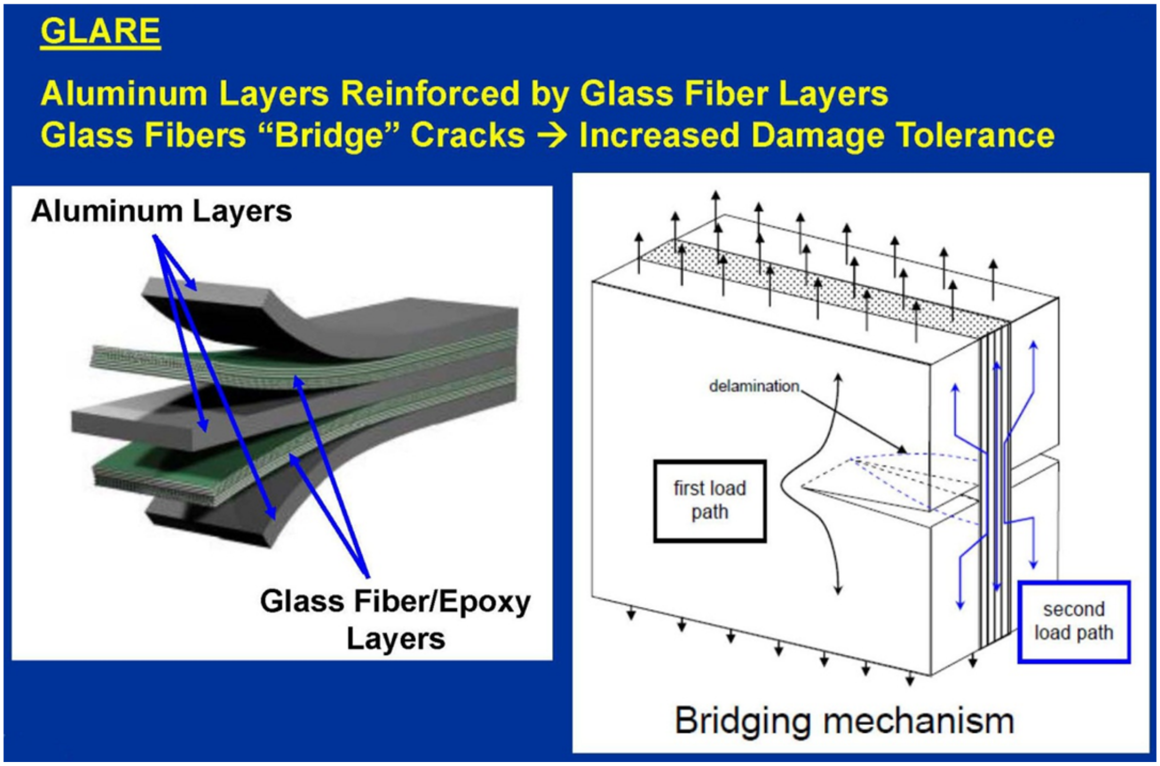


Figure 9.10-2. Background of GLARE

According to Alderliesten's method, the aluminum layer crack tip stress intensity is related to the stress in the fiber layers bridging the crack. This bridging stress is a function of the fiber layer extensional and shear stiffness, and the interface delamination shape and size. Where Alderliesten uses this information to relate the bridging stress to crack opening displacement in discrete strips along the crack flanks, the current method uses the same information to compute finite element spring stiffnesses along the crack flanks. The finite element analysis is performed using shell elements in NASTRAN to represent the summed aluminum layers, with the bridging fiber layer stiffness and delamination shape/size represented entirely by the spring elements. The crack tip stress intensity in the aluminum layers is computed using the Virtual Crack Closure Technique (VCCT). In order to perform a crack growth prediction using AFGROW, the stress intensity solution must be determined in advance. Where Alderliesten grows the delamination and the crack at the same time, the current method makes a-priori assumptions regarding the delamination shape and size as a function of crack length.

The current method compares well to crack growth test data for center-cracked panels and cracks from holes, and also correlates well to observed delamination shapes. The results of this work show that the increased damage tolerance life and residual strength characteristics can be predicted using standard crack growth analysis engines, which will make it easier in the future to certify structures containing FML materials (Figure 9.10-3).

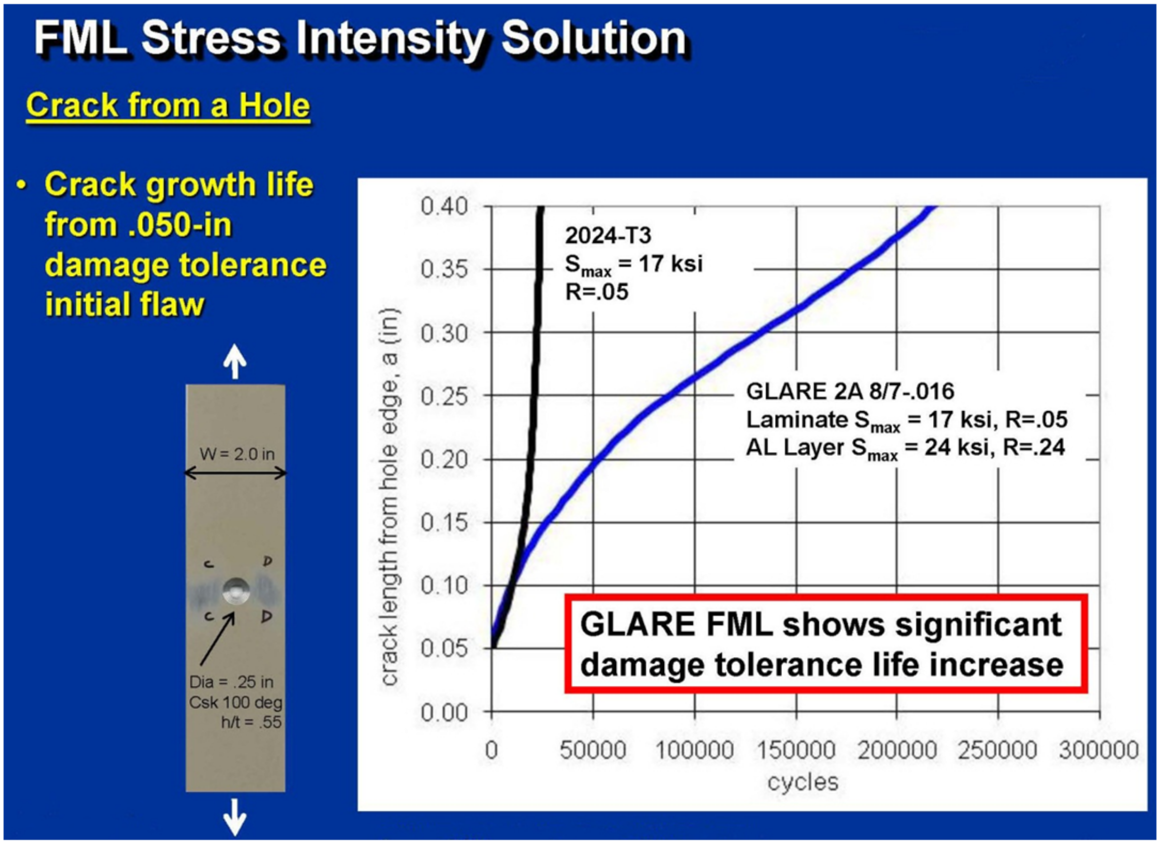


Figure 9.10-3. Damage Tolerance Life Increase of GLARE

9.10.2. Impact Characteristics of GLARE Fiber Metal Laminates

Hank Phelps, Jason Action, Doug Miller, and Joe Leonard, Lockheed Martin Aero – Ft. Worth

Lockheed Martin Aeronautics is currently investigating Fiber Metal Laminate (FML) materials under the USAF Advanced Hybrid Structures for C-130 Life Enhancement (HyLife) Program. The specific application under consideration is an outer wing lower skin, which is susceptible to external impacts from sources such as runway debris. As part of this program, testing was performed to verify the ability of FMLs to absorb impacts, a characteristic that has been widely publicized in the literature (Figure 9.10-4). The FML materials tested were of the GLARE family of laminates, GLARE 2A 8/7-.016 and GLARE 2A 4/3-.016, which consist of 2024-T3 aluminum with S2 Glass/Epoxy interleaf layers.

Panels of these configurations were impacted at various energy levels and the resulting dent sizes and depths were measured. The panels were subsequently ultrasonically inspected to evaluate the extent of internal damage. The extent of damage was larger than anticipated, based on the available literature for FML materials. One panel was subsequently cross sectioned to evaluate the damage present. The remaining panels were instrumented and subjected to compression testing.

This technical effort will cover the results of the impact testing, post impact damage assessment (Figures 9.10-5 and 9.10-6) and the subsequent panel compression tests. Comparisons will also be made to available impact data for graphite/epoxy laminates.

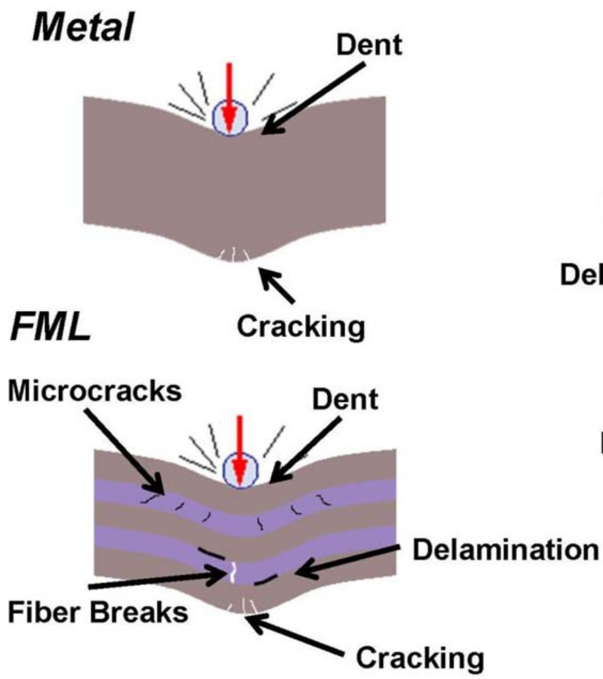


Figure 9.10-4. Impact Damage Characteristics

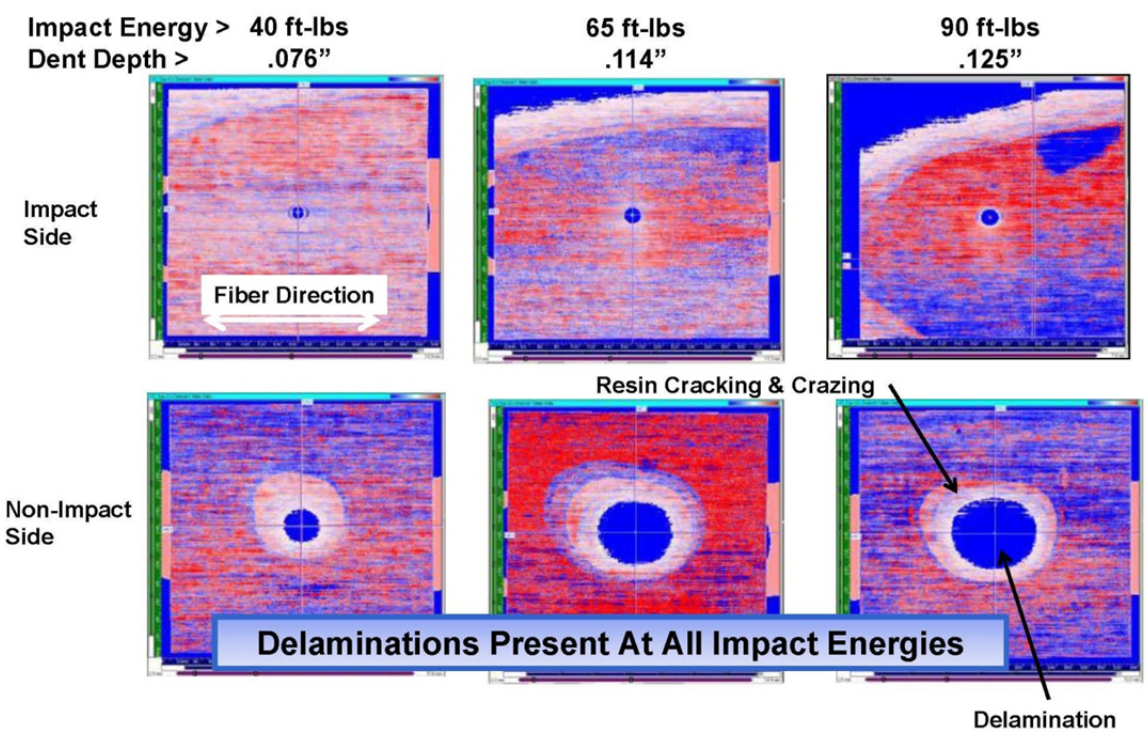


Figure 9.10-5. Post Impact NDI (GLARE 2A 8/7-.016)

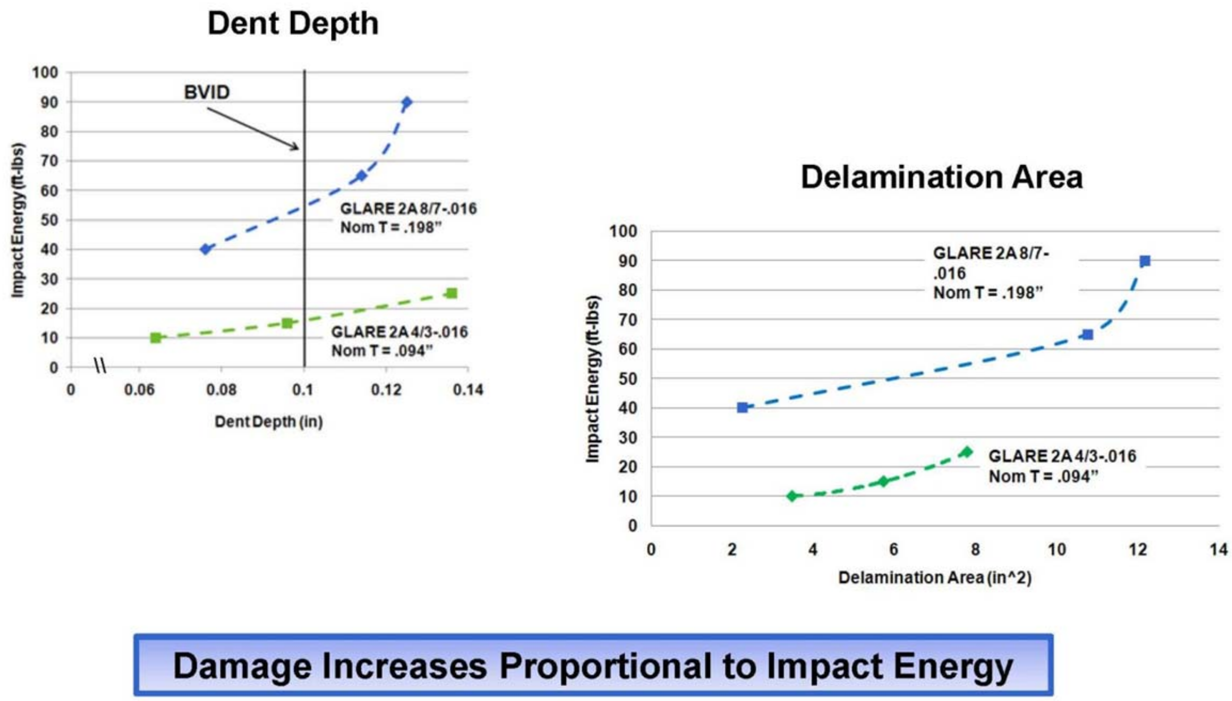


Figure 9.10-6. Damage vs. Impact Energy

9.10.3. C-130 Center Wing Rainbow Fitting Spare Redesign

Frank McElwain, Lockheed Martin Aero – Ft. Worth and Gerry Ringe, Mercer Engineering Research Center (MERC)

A USAF-funded program to redesign and test new (spare) configurations of fittings in the C-130 wing joint has successfully concluded (Figure 9.10-7). In 2006, a USAF Wing Service Life Independent Review Team (IRT) concluded that the center wing lower rainbow fittings experience onset of widespread fatigue damage at approximately 20,000 Equivalent Baseline Hours (EBH). The team identified the wing joint ("rainbow") fittings as the biggest threat to early onset of widespread fatigue damage. LM Aero has recommended specific inspections. The subject rainbow fittings are used on C-130E through C-130J aircraft. In order to reduce the inspection burden, in the USAF funded a redesign/test effort for the spare C-130 rainbow fitting in 2006 (Figure 9.10-8). This technical effort will discuss the goals that were established for the redesign program, the analytical methods used to determine new designs, and the component test program. During 2009, the redesigned rainbow fittings were successfully tested in a back-to-back three node skin/stringer panel configuration.

Figure 9.10-7. C-130 Center Wing Box

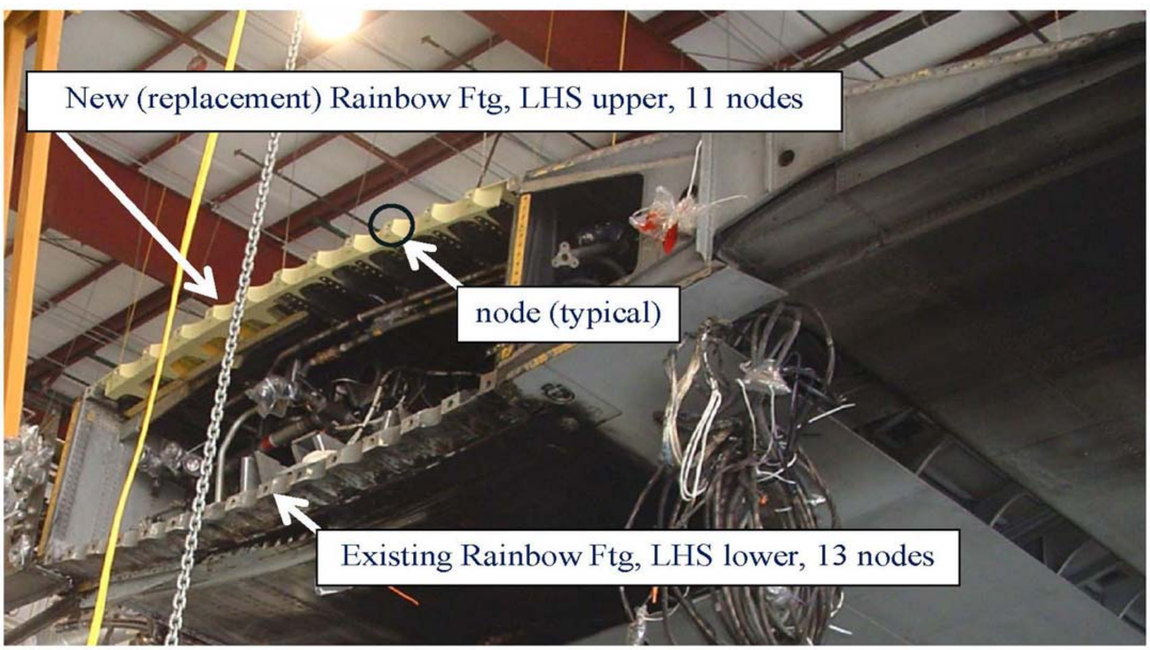


Figure 9.10-8. C-130 Rainbow Fitting Redesign

Team members (USAF, LM Aeronautics, and Mercer Engineering Research Center) worked in close coordination to successfully develop reconfigured spare versions of the upper and lower C-130 center wing rainbow fittings that exceed the program goals for static strength, durability, and inspection intervals. These findings are supported by analytical and test data that will be presented.

9.10.4. An Integrated Approach to Manage the Impact of Bulk Residual Stress on the Design-Build-Sustain Process for Primary Aircraft Structure

M. A. James, J. D. Watton and R. J. Bucci, Alcoa-Technical Center; and D. L. Ball, Lockheed Martin Aero – Ft. Worth

Integrated product development teams and computational methods have received significant attention in recent years as a means to accelerate new material insertion and reduce cost in the product development cycle (Figure 9.10-9). When combined, they form the new field of Integrated Computational Materials Engineering (ICME), which is held up as an enabler for the community with benefits ranging from material design through component design, manufacturing, and even sustainment. Recent trade studies indicate the benefit potential for sustainment is substantial; however, the up-front computational aspects of material and design integration are essential to capturing the cost saving and lifing benefits downstream.

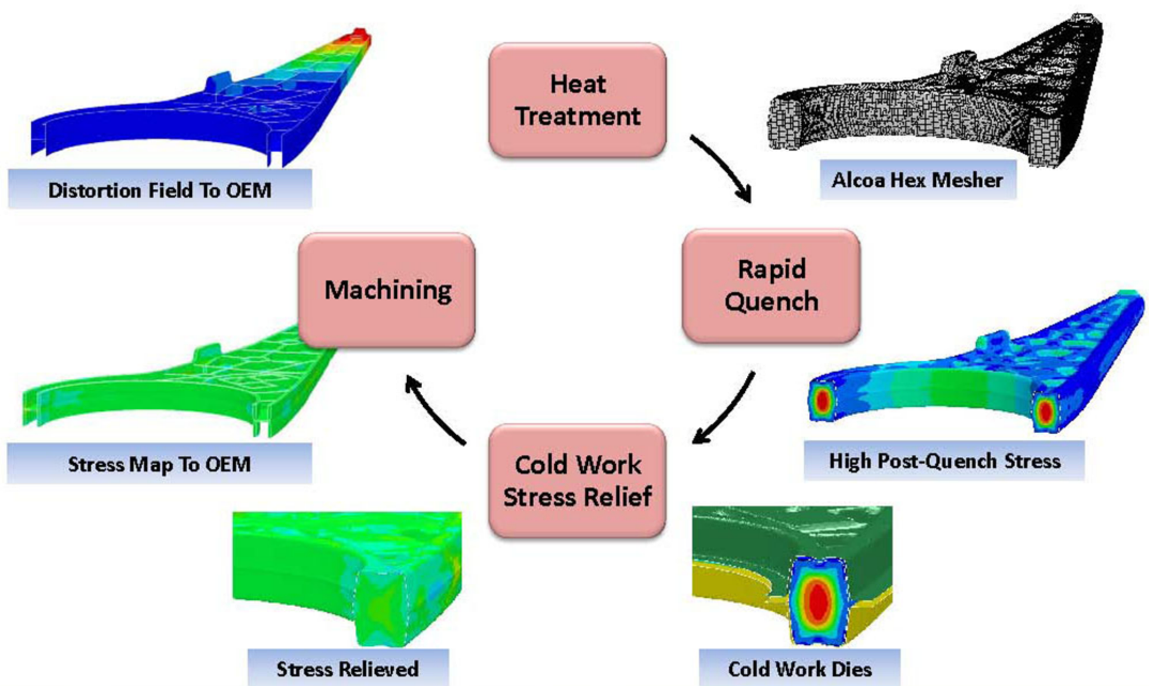


Figure 9.10-9. Foundation to Overall Modeling Capability

Alcoa and Lockheed have made substantial progress towards validating their respective visions of an ICME approach to manage bulk residual stresses and their consequence in large unitized structures, such as bulkheads and wing spars machined from large aluminum die-forgings. Recently, they have teamed to combine Alcoa's material and computational knowhow with the design and sustainment capabilities of Lockheed. The complementary capabilities and integrated approach enable a cradle-to-grave approach to managing the impact of bulk residual stress on primary aircraft structure.

The purposes of this technical effort are twofold: 1) to review residual stress measurement data on F-35 Joint Strike Fighter spar and bulkhead parts, and 2) to show recent progress on integrating Alcoa’s residual stress prediction capabilities in forgings with Lockheed’s production design/analysis capabilities for large single piece parts. Towards the first purpose, the technical effort summarizes over five years of Alcoa effort expended towards measuring residual stresses in simulations of F-35 production machined parts to demonstrate that Alcoa’s Signature Stress Relief cold work technologies indeed do reduce residual stress to single digits for the vast majority of the material volume (Figure 9.10-10). However, residual stress measurements are not possible at every location in a test article. As a result, prediction capabilities are necessary to provide residual stress estimates at widely varying locations throughout the structure. The technical effort demonstrates that significant progress has also been made in recent years towards validating quench and cold-work models for bulk residual stress prediction in host forgings and demonstrates Alcoa's capability to predict residual stress profiles in final machined parts. Towards the second purpose, the technical effort summarizes recent work to validate the use of structural zoning concepts to idealize Alcoa's residual stress predictions and to account for residual stress in production design processes. Finally, preliminary trade study results will be presented to show the impact of residual stress on the design/lifing process.

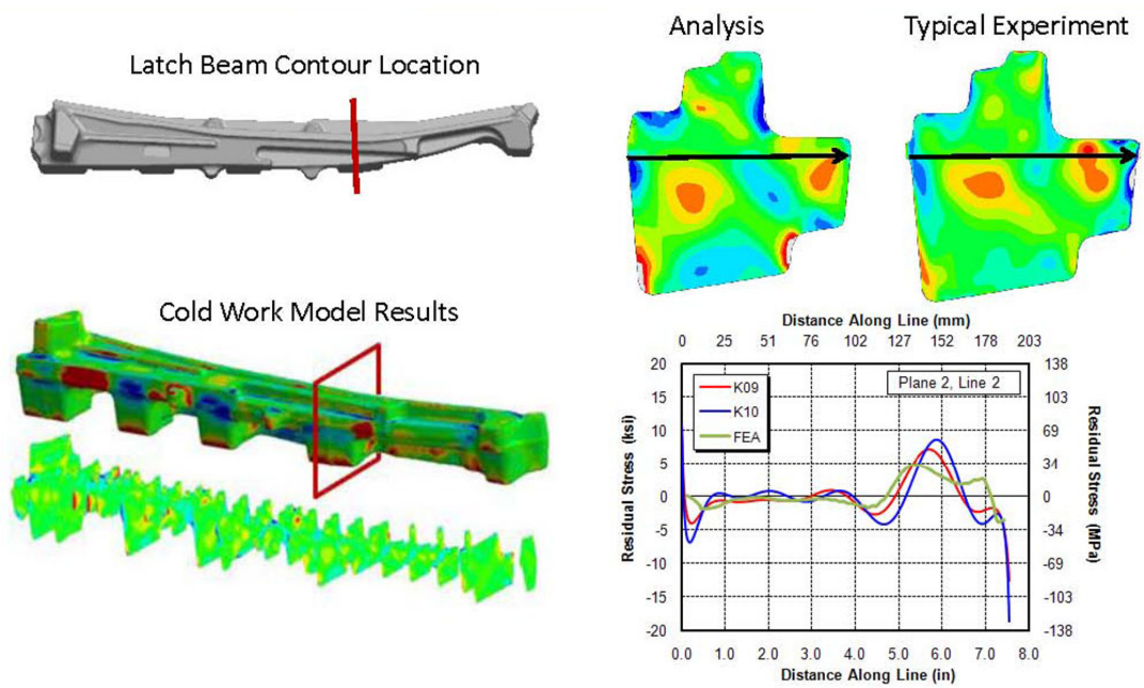


Figure 9.10-10. Cold Work Validation

Thus, the technical effort summarizes work in four areas essential to implementing residual stress management for aerospace applications of large aluminum die forgings: bulk residual stress measurement and modeling for the machined part, residual stress idealization in preparation for life modeling, and trade studies to quantify the importance of residual stress for sustainment.

9.10.5. Fiber Metal Laminate Development Programs

Henry Phelps and Dave Chellman, Lockheed Martin Aero – Ft. Worth; Frank DiCocco and Robert Bucci, Alcoa Defense; and Dave Heck, The Boeing Company

Fiber metal laminates (FML) have been used in Europe to save weight and meet the service requirements of the A380 jumbo liner and are planned to be used in select fatigue prone areas on the A400M transport (Figures 9.10-11 and 9.10-12). Domestically, the materials are still in the developmental stage, though several recent programs have explored the potential of this class of materials including the Metals Affordability Initiative (MAI) evaluating Advanced Hybrid Structures and recent AFRL programs to develop analysis preliminary design methods for FML structures. In addition, there have been associated independent research programs by the major OEMs and material producers including Lockheed Martin, Boeing, Northrup Grumman and Alcoa. The primary advantage of FML materials such as GLARE and CentrAl are the excellent damage tolerance characteristics of the materials (Figure 9.10-13). Engineered delamination and bridging by the glass fibers act to effectively reduce the stress intensity at the crack tip in the aluminum layers during loading. Current fabrication techniques are similar to composite layup and allow details and local build-ups to be incorporated during layup, minimizing the machining time and material loss associated with conventional metallic hogouts. This technical effort will discuss benefits and disadvantages of FML materials, the major thrusts of recent domestic programs and the technology gaps they are addressing in terms of development and certification under ASIP principles.

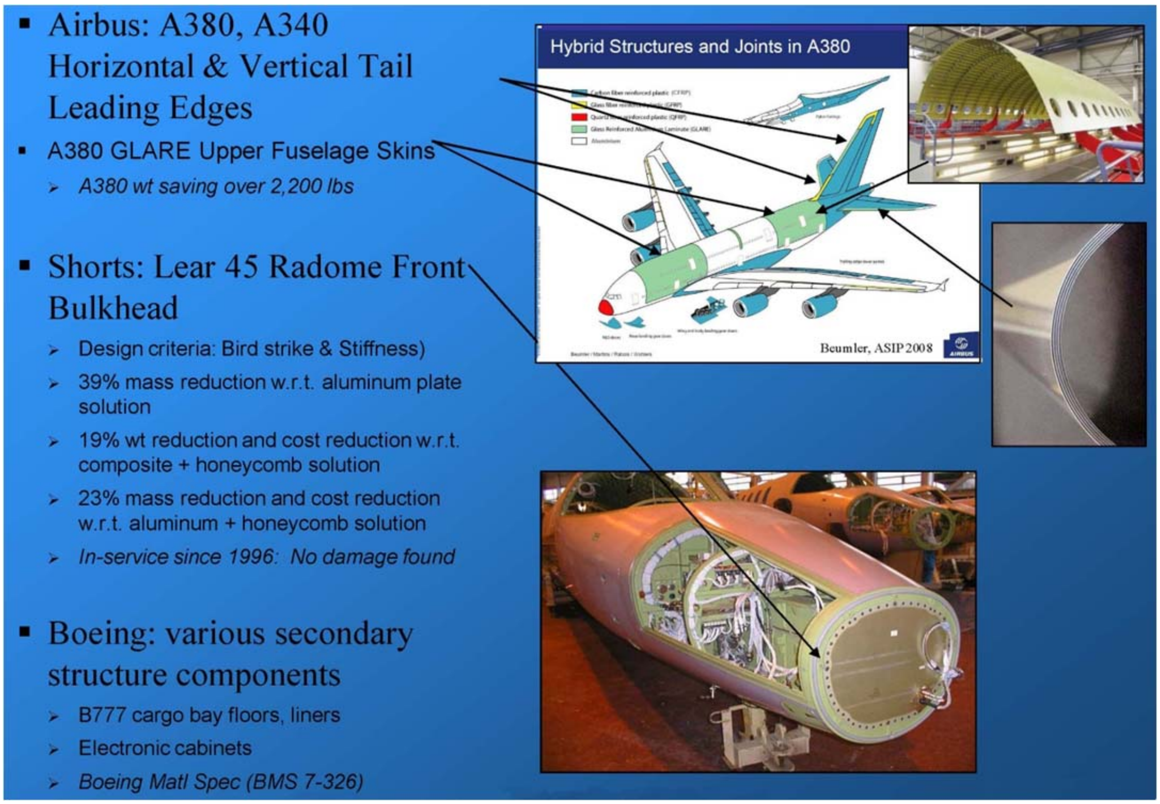


Figure 9.10-11. Fiber Metal Laminate Production Applications

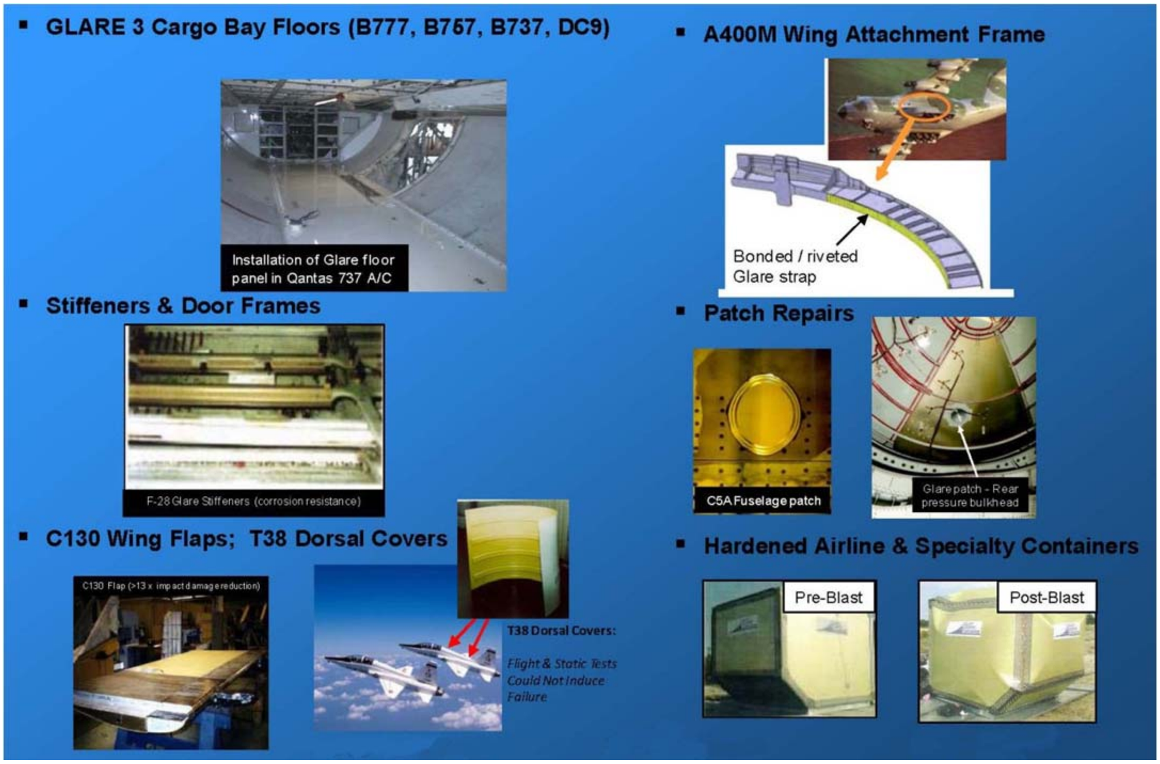


Figure 9.10-12. Additional Fiber Metal Laminate Applications

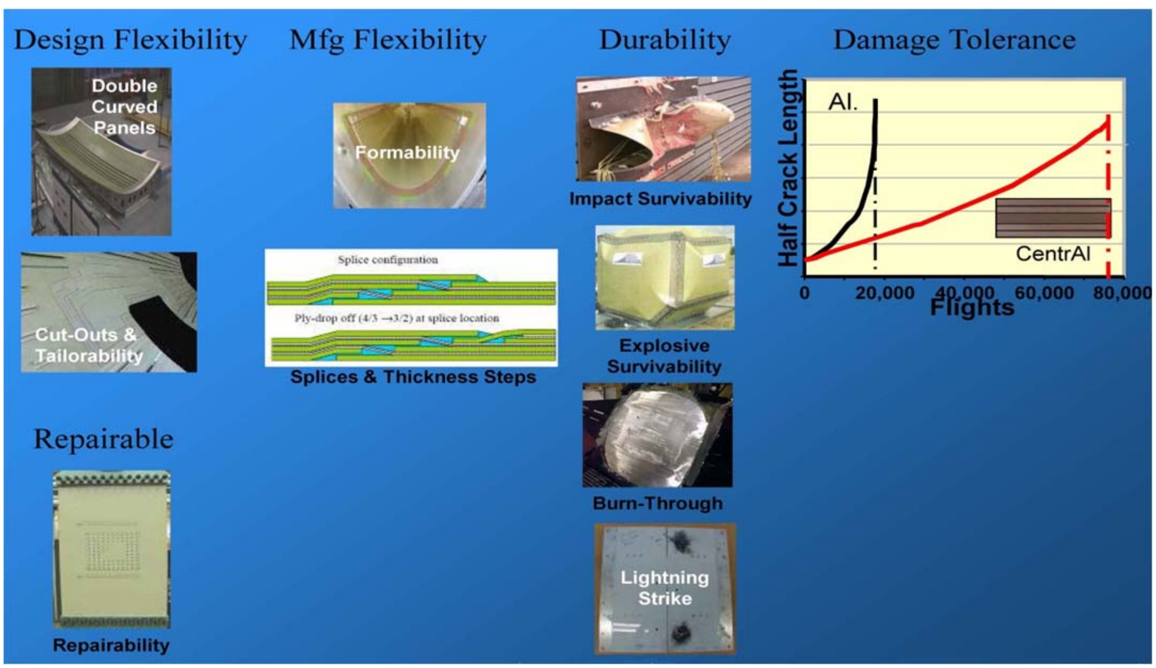


Figure 9.10-13. Fiber Metal Laminate Capabilities

9.10.6. Advanced Hybrid Structures (AHS) Life Enhancement and Replacement Concepts

Edwin E. Forster, USAF Research Laboratory – Air Vehicles Directorate

Designs that reduce life-cycle costs and minimize structural weight are desirable for future military aircraft; however, risk must be controlled to keep aging aircraft structures in service (Figures 9.10-14 and 9.10-15). The Air Force Research Laboratory, Air Vehicles Directorate supports investigation of Advanced Hybrid Structures (AHS) to provide a partial solution for some sustainment issues. Material systems that show improved fatigue resistance, excellent impact resistance and damage tolerance, as well as resistance to corrosion and lightning strike are considered potential candidates. Fiber Metal Laminate (FML) materials in the form of GLARE (GLAss REinforced aluminum) and CentrAl (Center reinforced Aluminum) have been suggested for structures that will enhance aircraft sustainment by reducing the frequency of inspection and increasing useful structural life.

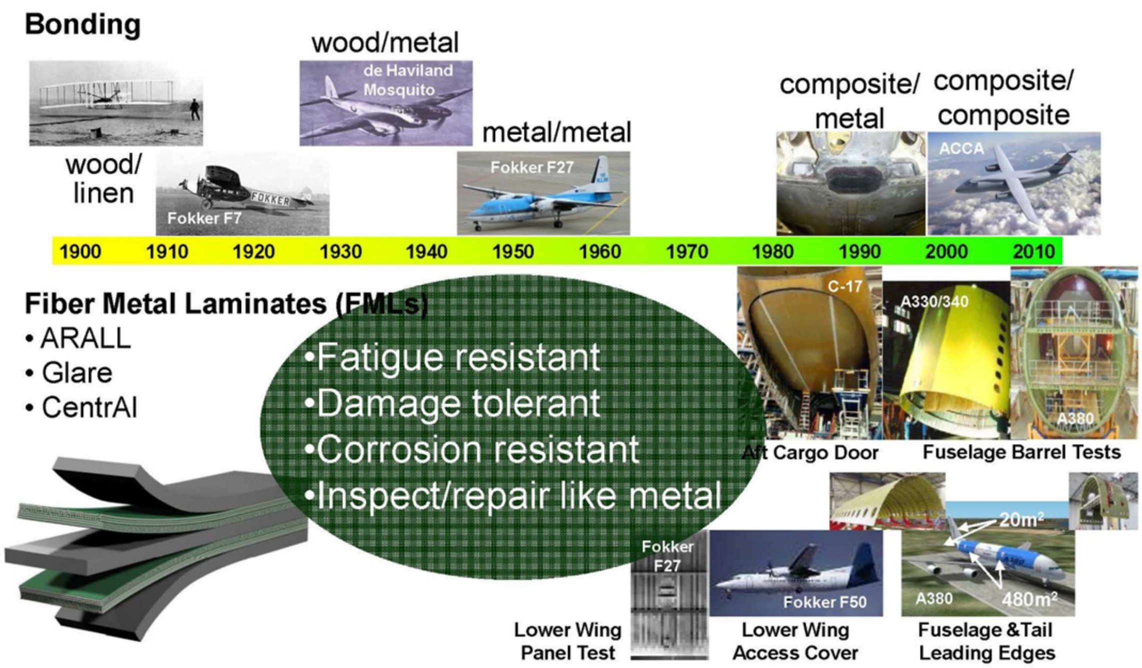


Figure 9.10-14. Some Historical Background

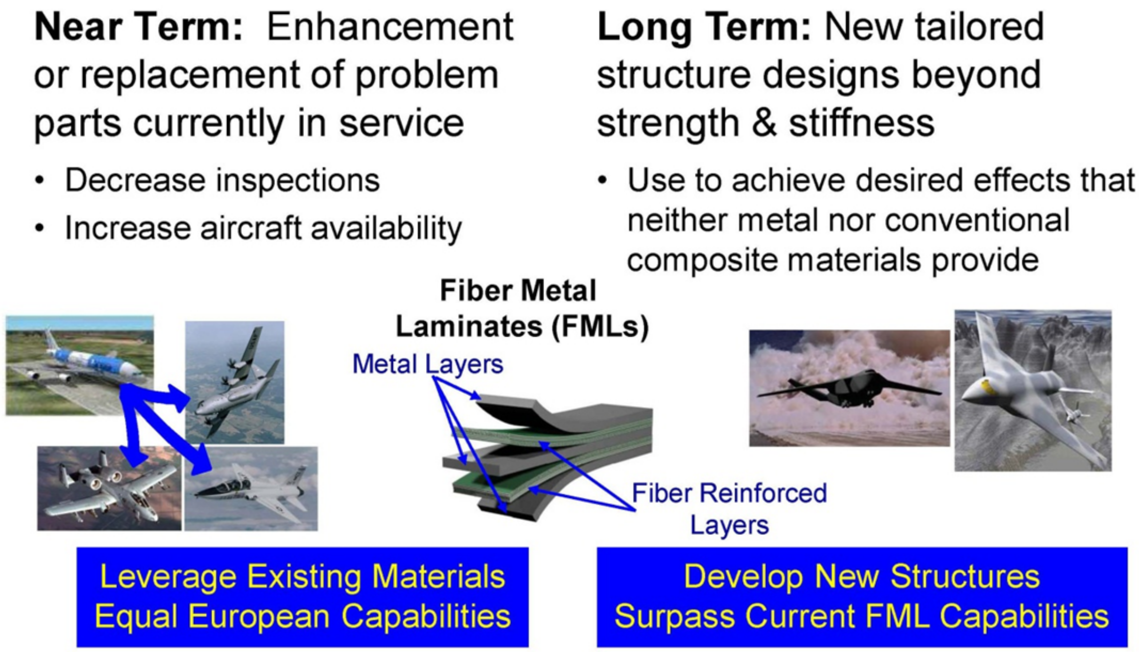


Figure 9.10-15. Near-Term and Long-Term Visions

Hybrid materials are a specific form of composite, that leverage the best characteristics of the constituent materials to provide a capability of the whole that is greater than the sum of the parts. In the case of FMLs such as GLARE, these materials have lower elastic modulus than monolithic aluminum metal and corresponding lower density, yet are capable of higher dynamic stress levels due to the slow crack growth characteristic of the material. Therefore, a structure composed of FML materials may provide an overall lighter weight component than a monolithic aluminum design. The complexities involved in the design of such aircraft structures will require design optimization of the structural application to meet all aircraft performance requirements. The resultant design may report benefits over the monolithic aluminum baseline design, which may include weight savings or sustainment benefits such as reduced inspection frequency. This technical effort will explore the capability of Advanced Hybrid Structures (AHS) for life enhancement of a metallic component and structural component replacement. This effort will utilize preliminary design tools to evaluate the performance and capabilities of AHS life enhancement and replacement concepts, in particular comparing with a baseline monolithic aluminum design. The overall effort is intended to shed light on the effectiveness of AHS for sustainment applications, and attempt to dispel some of the myths associated with these materials.

9.11. OVERVIEWS

9.11.1. Moving Forward with ASIP on the F-15

Jeff McFarland, The Boeing Company and Dave Currie, USAF-WR-ALC

While on training maneuvers in Nov. 2007, an F-15C model aircraft was lost due to a structural failure in the forward fuselage. In addition to a resolution of the structural issue, the investigation determined that the F-15 ASIP program had been allowed to lapse into a maintaining mode due to expenditure considerations. Since that aircraft loss, however, various tasks have been initiated to bring the F-15 ASIP program to a current state of readiness. This technical effort has been developed to help highlight initiatives undertaken to revitalize the F-15 ASIP program over the past three years.

Under recent ASIP activities, the F-15 program has addressed updated finite element models (Figure 9.11-1), completed a review of the forward fuselage critical locations, developed a new usage spectrum for the F-15C/D model, refined analysis and criticality for thirty of the most critical fatigue locations on the F-15 C/D and E models, incorporated risk analysis as a tool in making decisions, and initiated effort to conduct a set of full-scale-fatigue-test programs (Figure 9.11-2). These tasks will be reviewed and discussed within the technical effort.

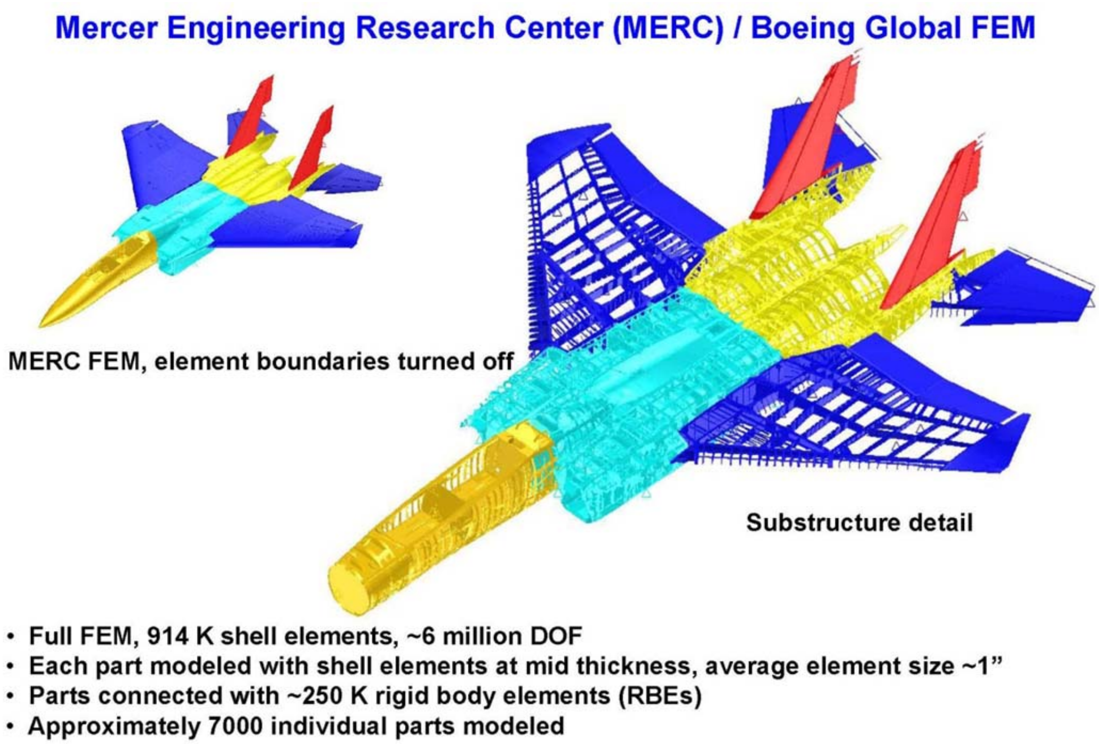


Figure 9.11-1. Finite Element Models

Figure 9.11-2. Teardown Test Articles

In conclusion, while the mishap almost resulted in a tragic incident, the remaining F-15 fleet has greatly benefitted from the revitalization of the ASIP program. This is significant since the F-15 fleet must remain viable until at least the year 2035. Many of these initiatives set the groundwork for keeping the fleet safe and benefit from reduced maintenance man-hour requirements

9.11.2. An B-1 ASIP Overview – Tough Issues, Real Solutions, Promising Future

Rodney Harberson and John Morgan, USAF-OC-ALC; and Randal Edwards, The Boeing Company

During its nearly 25-year history, the primary role of the B-1 has transitioned from a strategic bomber to a versatile conventional weapons delivery system. The B-1 ASIP program has played a vital role in supporting this transition, assessing the aging airframe structure, and maintaining the B-1's function as a key war fighter of the United States Air Force (USAF).

Some fatigue issues have arisen as the B-1 nears and surpasses its original design life goal (Figure 9.11-3). Scheduled inspections, and in some cases routine maintenance, have discovered cracks or the onset of fatigue cracking. Structural Life Extension Programs (SLEPs) and repairs have been implemented in a timely and cost effective approach. Conventional crack growth analysis, statistical evaluations, and risk assessments have been used extensively in support of both the repair effort and to assure flight safety prior to repair implementation. The risk analysis has also been incorporated into recurring ASIP tasks, such as the Force Structural Maintenance Plan (FSMP), to assure flight safety and enhance the traditional deterministically-derived inspection intervals.

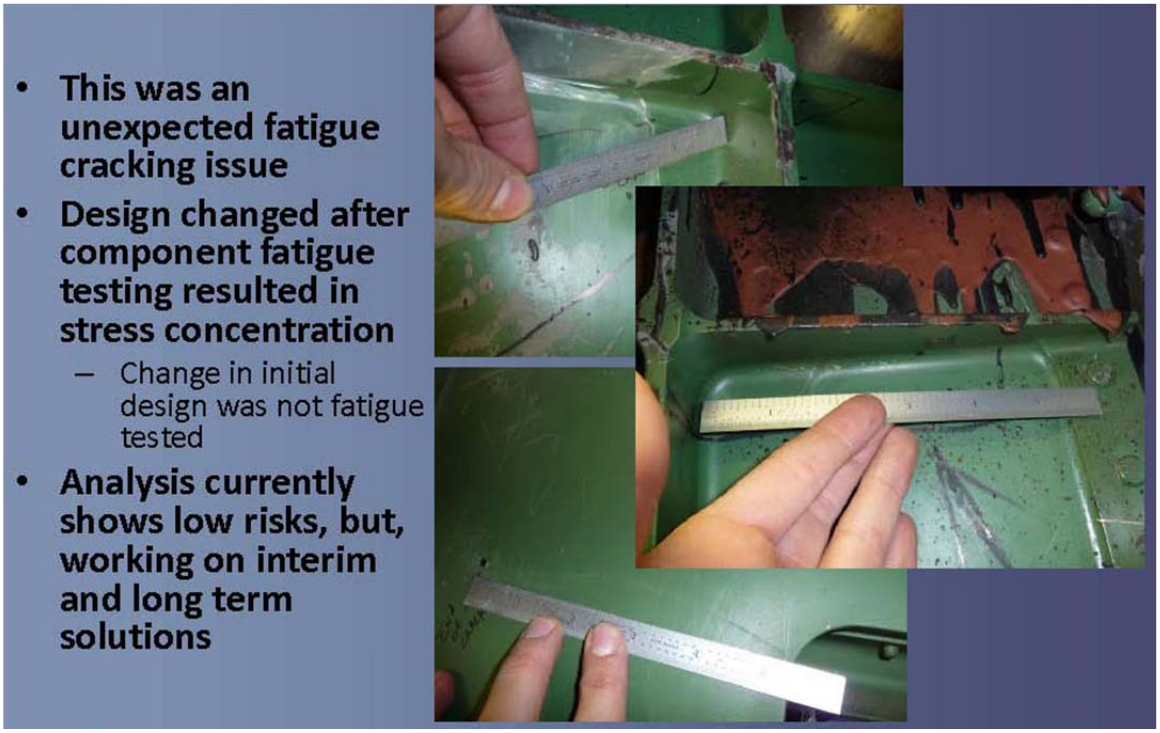


Figure 9.11-3. Wing-Carry-Through Cracks

No full-scale-fatigue test has been performed on the B-1 (Figure 9.11-4). Early in the B-1 program, large scale components were deemed sufficient to validate the structure. However, these components did not include all of the airframe structure and were only tested using a load spectrum based on initial design usage assumptions (Figures 9.11-5 and 9.11-6). In support of the ASIP program, tasking has begun to perform a full-scale-fatigue test of the B-1 airframe using the current usage spectrum. A retired airframe has been chosen and prepared for shipping to a test facility. The testing will help to establish and validate the service life of the B-1 structure as well as uncover other potential fatigue issues so that they can be addressed in a timely, safe and economic manner.

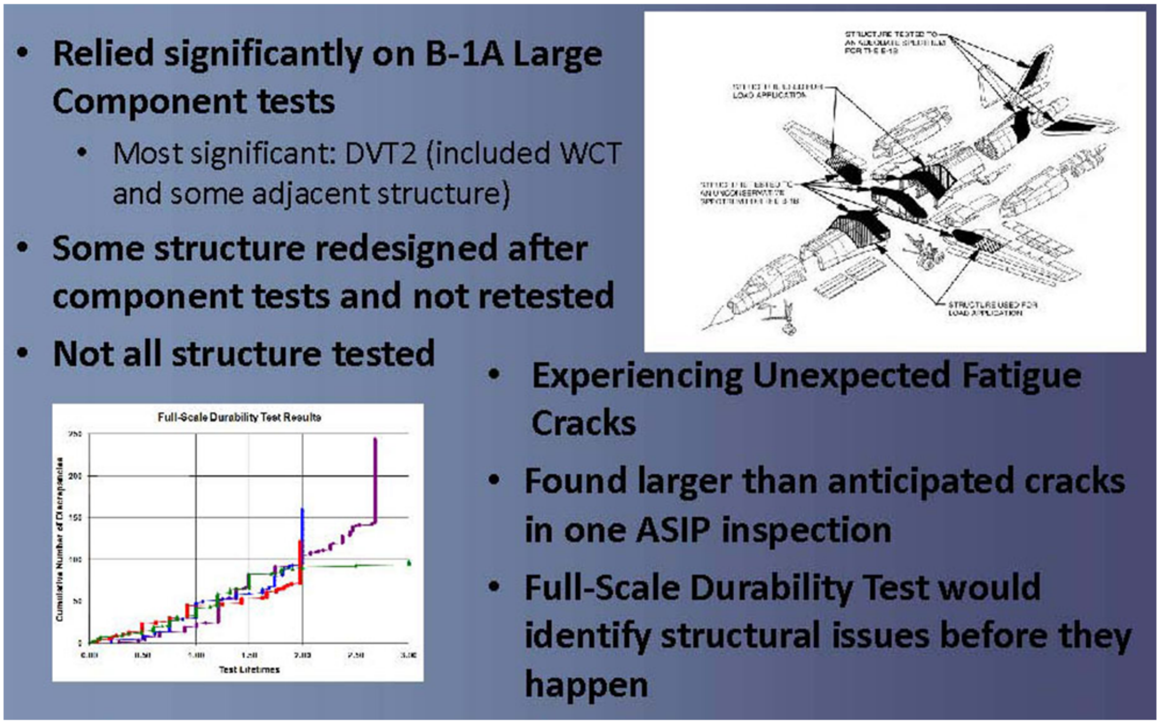


Figure 9.11-4. No Full-Scale-Fatigue Test

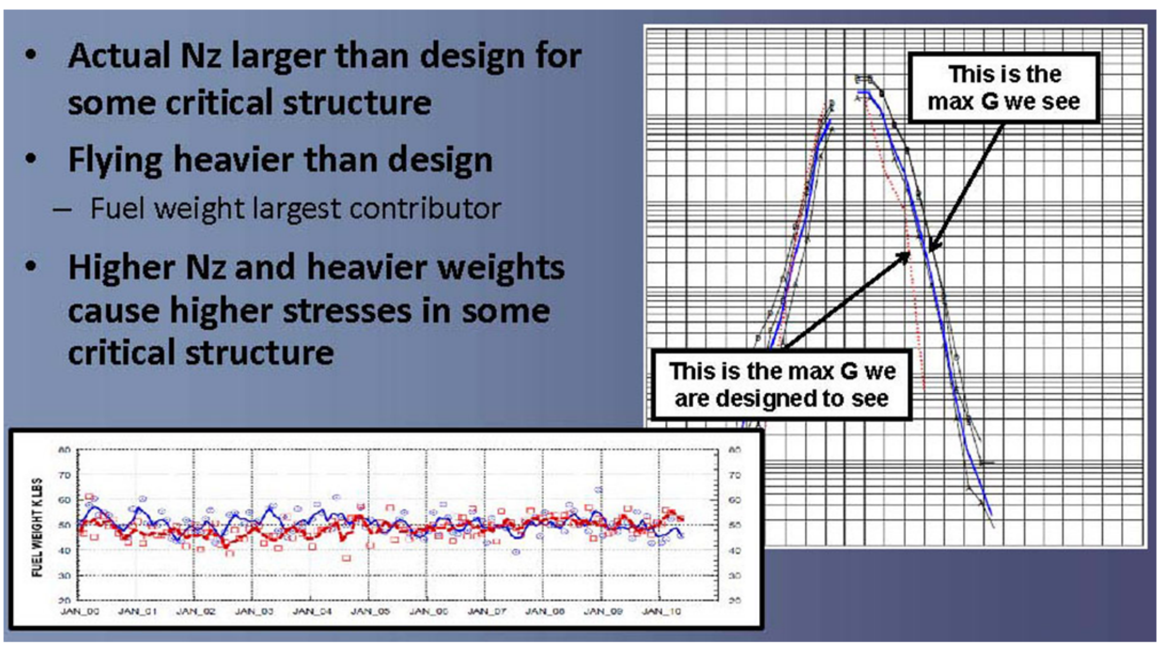


Figure 9.11-5. Design Usage Does Not Match Actual Usage

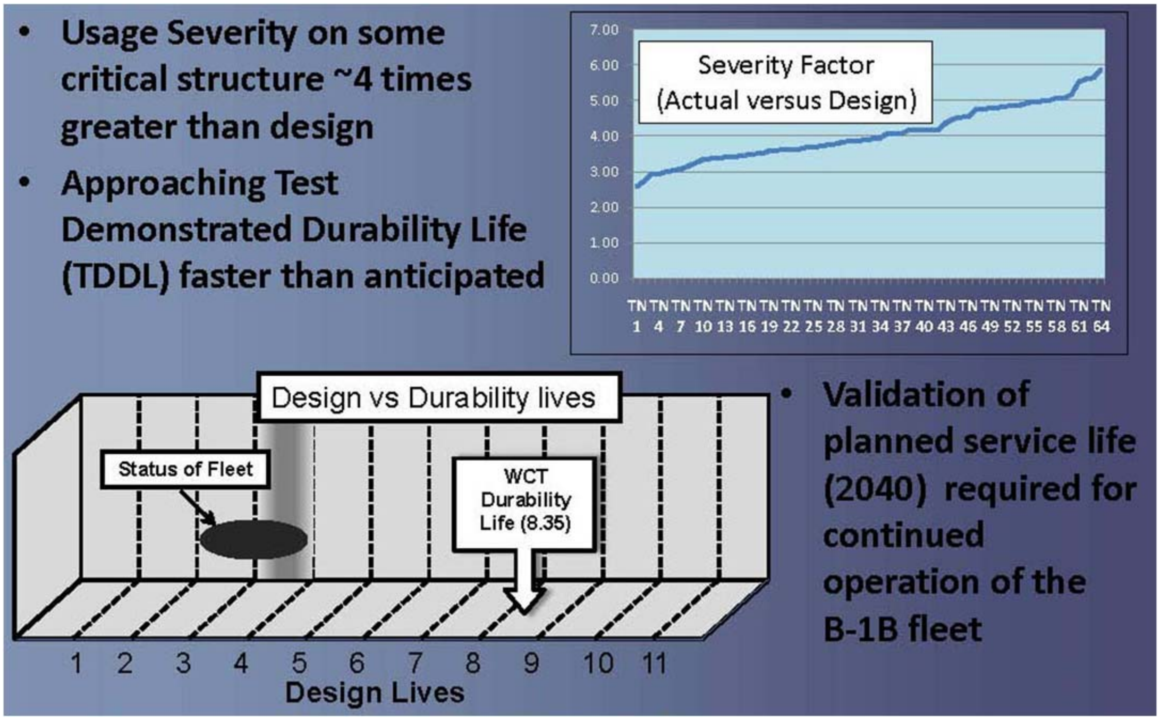


Figure 9.11-6. Effects of the Difference in Usage

9.11.3. The F-16 Sustainment ASIP: Impacts of Revised USAF Damage Tolerance Requirements for Fail-Safe Metallic Structure

Tim C. Jeske, Matthew Edghill, Lynette Limer, and Kevin Welch, Lockheed Martin Aero – Ft. Worth and Bryce Harris, USAF-OO-ALC

In September of 2008, USAF ASC/EN released Air Force Structures Service Bulletin EN-SB-08-001 which revised the damage tolerance requirements for fail-safe metallic structures. One of the primary objectives of this bulletin is to identify those safety-of-flight (SOF) locations which have inherent fail-safe capability. Following this release, LM Aero's F-16 Structural Integrity Team assessed the impacts of the bulletin to structure currently managed by fail-safe criteria. As background to this, in 2005 LM Aero was tasked under D.O. 0283 to perform an extensive fail-safe analysis and overall risk assessment of the USAF F-16 A/B fleet. One of the most critical structural areas managed by fail-safe criteria on early F-16 A/B airframes is the fuel shelf joint (FSJ) area on the Upper FS 341 carry-through bulkhead (Figure 9.11-7). The FSJ assessment resulted in decisive force management actions on these aircraft. This risk assessment included evaluating the risk of using fail-safe inspection intervals to maintain flight safety, providing risk-based retirement dates using fracture-based "line-in-the-sand" criteria, and presenting the risk assessment results in hazard-matrix format. This assessment was completed in 2006 and had substantial impacts to the fleet. The residual life analysis at that time was performed using the MIL-A-83444 philosophy.

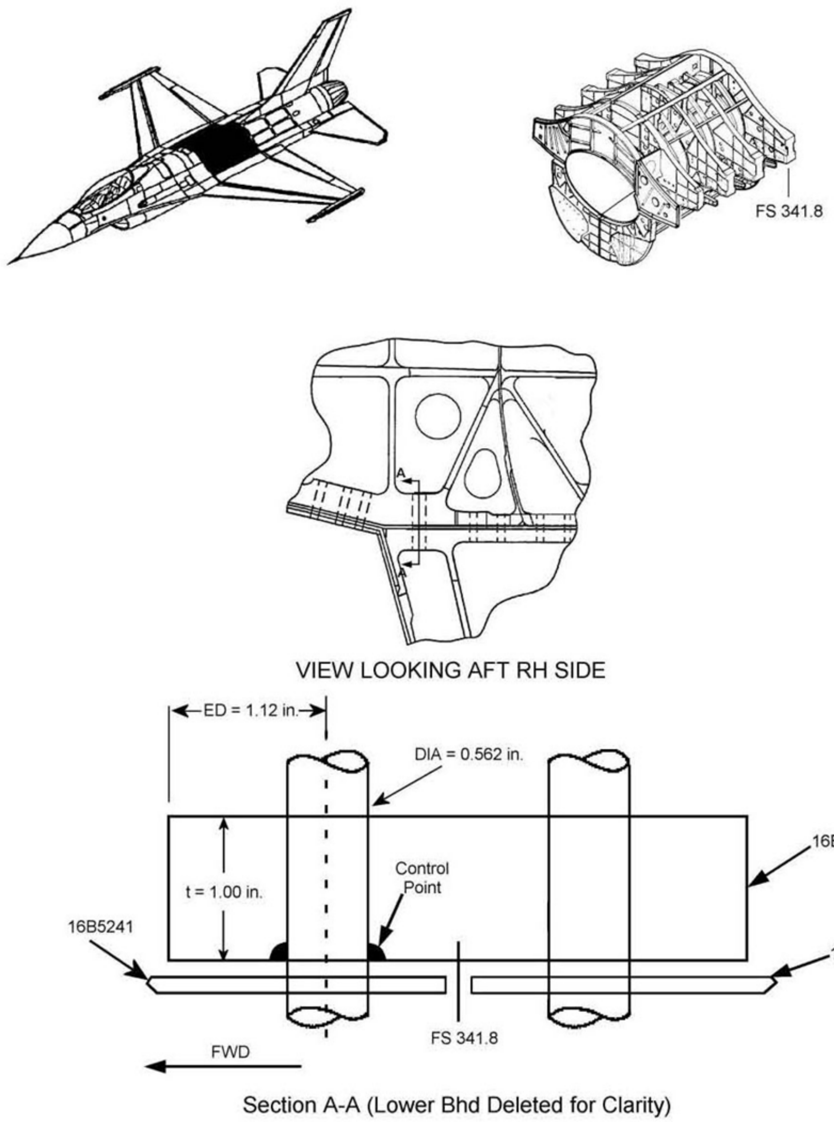


Figure 9.11-7. Fuel-Shelf-Joint Location

To highlight the differences between and assess the impacts of the new fail-safe criteria, this technical effort will compare the analysis results and associated management recommendations based on the legacy MIL-A-83444 approach and results/recommendations based on the recently released EN-SB-08-001 revised approach. As part of the F-16 Force Management Update (FMU), the fail-safe analysis of the FSJ has been updated. These updates include incorporation of updated finite element analyses, implementation of a new integrated metallic durability and damage tolerance analysis tool set, and updated fleet usage based on recent CSFDR data. A comparison of results (Table 9.11-1) and follow-on maintenance recommendations based on the following four scenarios will be presented:

- Original crack growth models using previous fail-safe approach (D.O. 0283)
- Original crack growth models using revised approach per EN-SB-08-001
- Updated crack growth models using previous fail-safe approach
- Updated crack growth models using revised approach per EN-SB-08-001

Table 9.11-1. Fail-Safe Assessment Comparisons

USAF Block 10/15								
USAF F-16A/B Block 15 FMU 2004 Baseline Usage, 16SL-04-83 Rev. B								
Primary	16B5251 Upper Bulkhead at FS 341.8, Fuel Shelf Joint Outboard Bolt Hole at BL 26 (Pre-ECP 1910 Retrofit Configuration)				Damage Tolerance Life: 3,500 FH (IMAT)			
Secondary	Analysis Tool	MIL-A-83444		EN-SB-08-001				
		Residual Life (hours)	Fail-Safe Capability Life (hours)	Intact Durability Life (hours)	Severed Durability Life (hours)	Desired Maint. Interval (hours)	Fail-Safety Life Limit (hours)	
16B5262 Lower Bulkhead at FS 357.8, RHS Lower Flange Bolt Hole No. 5 with Strap (ECP 1910 Retrofit Configuration)	IMAT	2,800	**6,300**	9,500	3,200	300	**8,300**	

Important factors that must be fulfilled to meet the requirements for classifying as fail-safe structure, the fail-safety life limit, and other relevant considerations in performing fail-safe analyses will be discussed.

9.11.4. Effective ASIP Despite Evolving POI: An F-16 FS 341 Bulkhead Cracking NDI Case Study

Kimberli Jones and Bryce Harris, USAF-OO-ALC

Aircraft maintenance requirements are performed regularly through standard pre- or post-flight checks and during more rigorous phase-based nondestructive inspections or programmed depot maintenance. Engineers and equipment specialists levy these requirements, complete with methods and procedures for proper completion. Engineering assumes that the intent of these requirements will be carried out as specified in the appropriate technical-data source, but this is not always the case for various reasons.

The F-16 Aircraft Structural Integrity Program has experience with instances involving cracking identified in an electrical connector hole on the lower FS 341 bulkhead (Figure 9.11-8). This case study continues to evolve over time and has been on-going for almost three years. A number of jets in the fleet have been identified with cracking, and analysis shows that most are expected to crack in the same location during the life of the aircraft. A repair has been implemented to address the cracking issue, but due to repair life limits based on crack size, a crack must be identified as early as possible in order to extend the life of the bulkhead (Figure 9.11-9). A supplement to the technical order was sent out to field units with a new inspection procedure requiring removal of an electrical connector to gain access to the crack origin. F-16 engineering assumed that subsequent inspections would be performed using this method, and smaller cracks would be identified in the future. The curious observation about the reported cracks was that they were very base dependent; some bases would find many aircraft with these cracks, while other bases never reported any cracking, or only large cracks, in the suspect region. Was the connector being removed by every maintainer every time? After questions arose from field units, F-16 ASIP realized there was an issue with understanding the new inspection procedure, as well as a lack of awareness of the change in requirements. As a consequence of the inspection confusion, predictions as to which aircraft had cracks were based on inspection schedules and actual crack finding trends rather than on crack growth and aircraft flight severity.

- Most block 40/42 and 50/52 aircraft are expected to have this cracking at some point in their service life

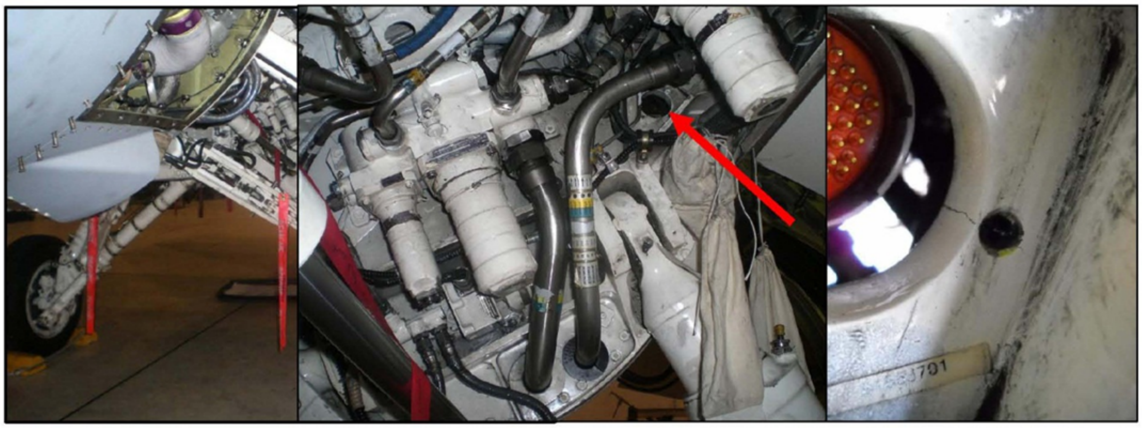


Figure 9.11-8. Cracking at Lower FS 341 Bulkhead Cannon Plug Satellite Hole

- Goal is to find cracks "small" so repair applies long-term, ideally until aircraft retirement
- Cracks considered to be mostly an economic concern
  - Repair much cheaper than bulkhead replacement

Figure 9.11-9. Doubler Repair

Issues with the 341 bulkhead inspection are not limited to field units; a few aircraft undergoing depot repair had inspections performed that did not match the unit's results. These inspections, both pre- and post-repair, affected repair lifing and potentially the safety of the aircraft. Additional visual inspections are levied on aircraft post-repair to ensure structural integrity of the bulkhead in the event of crack growth through the repaired bulkhead stiffener.

Probability of Detection (POD) and Inspection (POI) were not major concerns when bulkhead inspections and repairs were fielded, but the history involved in this case study has reiterated their importance. Fortunately, the F-16 has fail-safety in this area; otherwise, this situation would have been handled very differently. The technical effort will detail the history of the 341 bulkhead cracking and repair, along with lessons learned to avoid similar issues in the future.

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

Marguerite E. Christian, Lockheed Martin Aero – Ft. Worth

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) (Figure 9.11-10). 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 9.11-11). The test programs also include provisions to accommodate certification of future internal and external stores loadouts up to the capability of the aircraft structure. 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. The first two static tests of the STOVL and CTOL designs are complete, both achieving unprecedented test rates. The first durability test on the CTOL design began in May 2010, and the second durability test (STOVL) started in Aug 2010. The third and final static test for the CV design began in the 4th quarter of 2010. The stand-alone static tests of the horizontal tails for all three variants are now complete and the durability tests for the horizontal and vertical tails are in progress (Figure 9.11-12). This technical effort provides an overview of the load conditions and test spectra as well as the results to date for each static and durability test program. Test efficiencies resulting from the scale and organization of the test programs and lessons learned from the test approaches employed are described.

Figure 9.11-10. Tri-Variant Joint Strike Fighter (JSF)

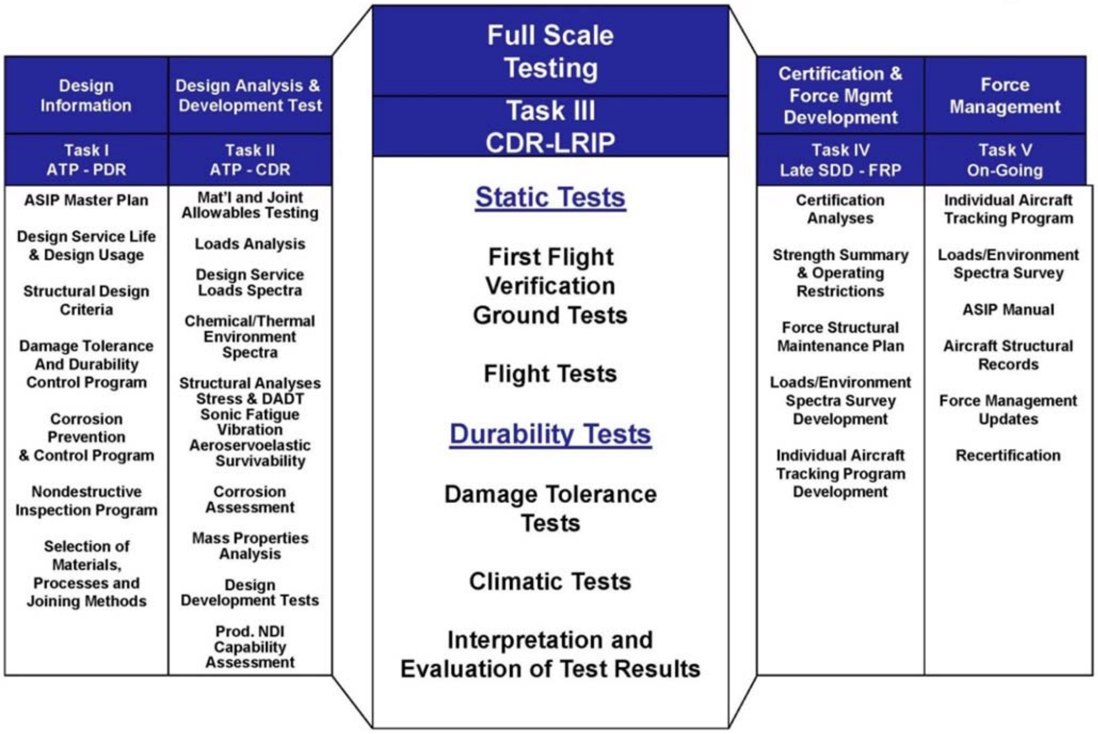


Figure 9.11-11. F-35 Full-Scale Tests and How They Relate to ASIP

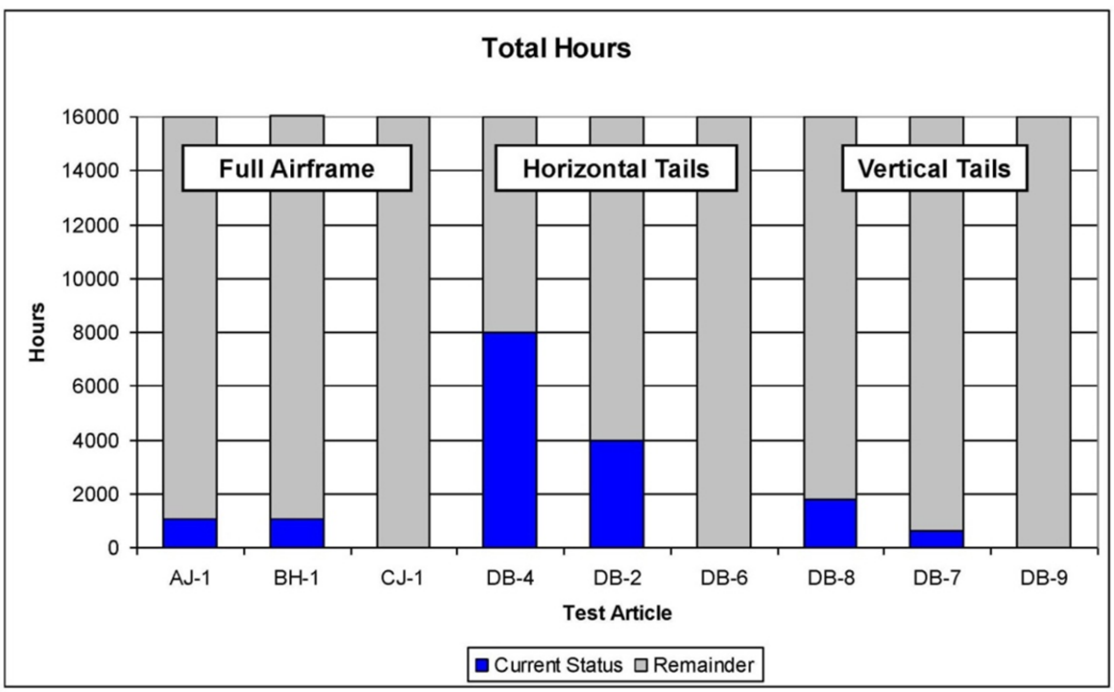


Figure 9.11-12. Progress Summary for F-35 Durability Test Articles

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.

9.11.6. Certifying the F-15C Beyond 2025

Paul A. Reid, The Boeing Company and Joseph D. Lane, USAF-WR-ALC

A key element in Task III of the Aircraft Structural Integrity Program (ASIP) includes performing full-scale-fatigue tests, as necessary, to establish the certified fatigue life for a given aircraft platform. This information plays a vital role in fleet management regarding how long a fleet of aircraft can safely be flown. The current F-15C airworthiness certificate, defined as half the fatigue life demonstrated during a full-scale-fatigue test, is based on a test conducted 16 years ago that consisted of only the wings and center fuselage. This established the current certified safety limit for the F-15C. Currently, the United the States Air Force Air Combat Command is planning to operate the F-15C beyond 2025. Based on current and projected fleet usage, Boeing and the Warner-Robins Air Logistics Center (WR-ALC) has determined that to meet this objective, the certified safety limit will have to be increased. Following the ASIP process and guidance from Task III, Boeing is conducting a full-scale-fatigue test of an F-15C model aircraft, configured with the latest structural enhancements, to establish a new certified safety limit with the objective of meeting the required safety limit.

The purpose of this technical effort will be to outline the laboratory test approach, fatigue methodology, and post-test process to correlate analytical predictions with test results. The major elements presented include details of the laboratory function, namely, test set-up, cycle rate optimization, and real-time damage monitoring. Similarly, the major tasks performed by engineering support are also discussed. These include determination of test loads, spectrum generation based on current and projected fleet usage, preventative repairs to maximize the airframe endurance, use of health monitoring systems (Figures 9.11-13 and 9.11-14), finite element analysis correlation, post-test disassembly and teardown, and a process to validate fatigue life predictions with test results.

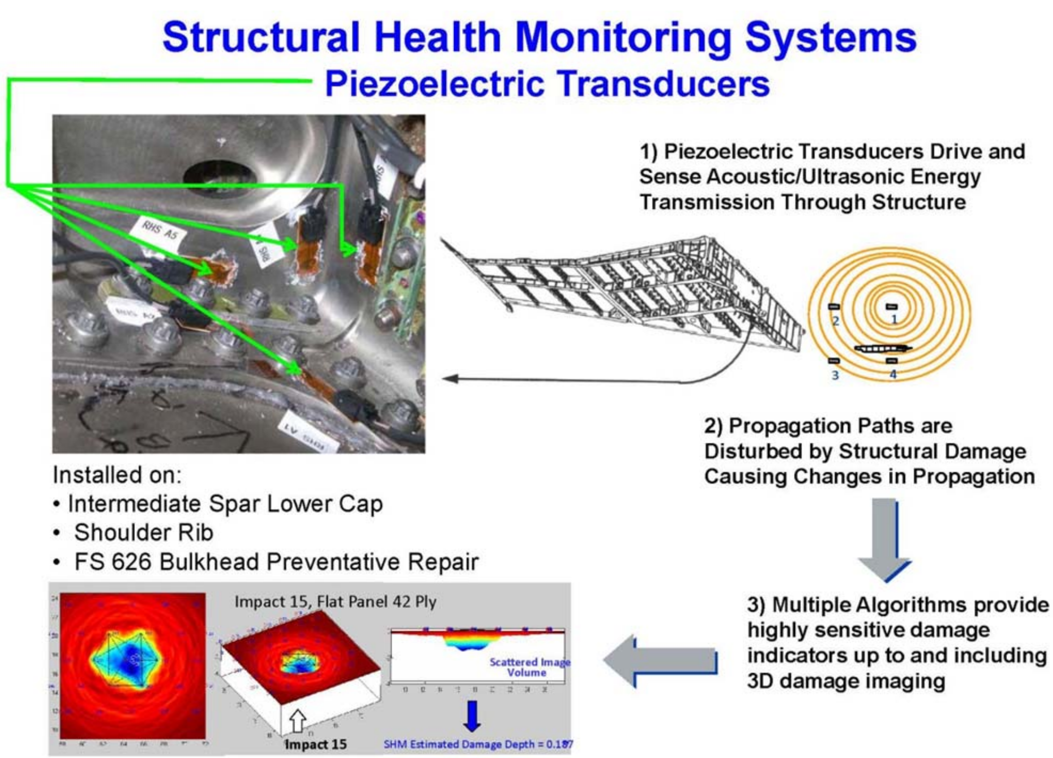


Figure 9.11-13. Piezoelectric Transducers

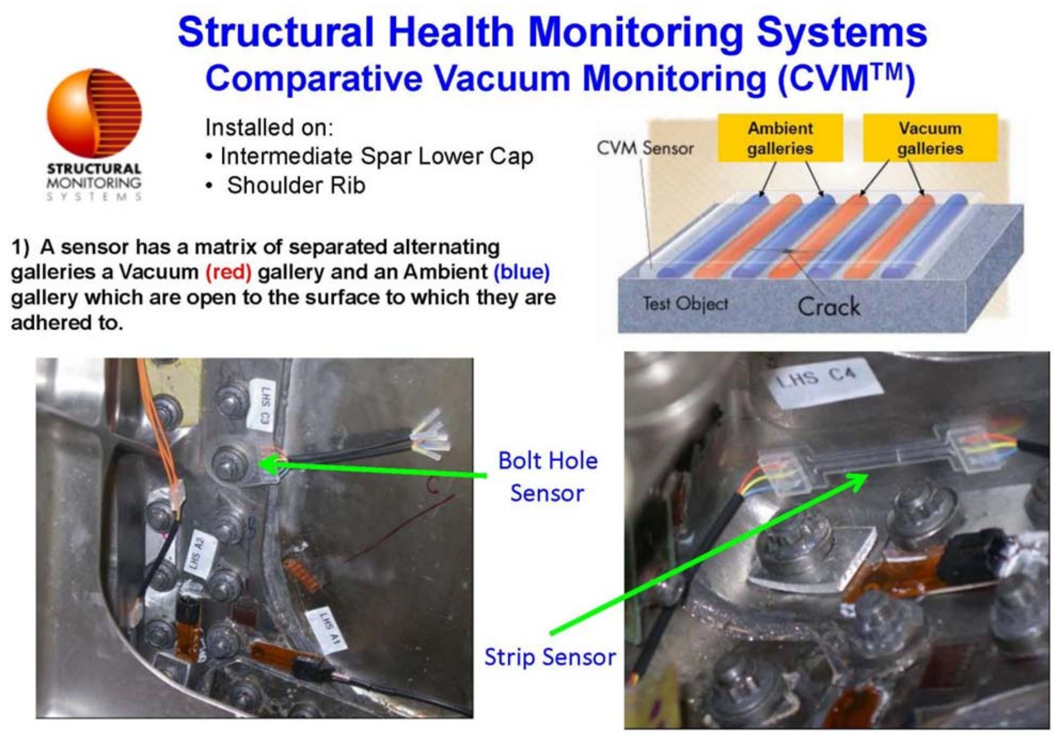


Figure 9.11-14. Comparative Vacuum Monitoring

In conclusion, to meet current USAF operational force requirements, the existing F-15C certified safety limit must be increased. Based on the ASIP process and guidance, Boeing and the WR-ALC, have put forth an approach to maximize the F-15C certified safety limit. The significance of this effort, as presented, is that these tasks are essential towards meeting USAF safety limit requirements while maintaining an economically viable airframe beyond 2025.

9.11.7. KC-135 Individual Aircraft Tracking Program (IATP) Issues

Jeff Wilterdink, USAF-OO-ALC; John Bailey, USAF 645 AESG; Toby Ortstadt, The Boeing Company - Wichita

Between 2004 and 2010, the KC-135 Program Office and the Boeing Aircraft Company conducted a comprehensive update to the IATP that features new DADTA baselines, MDS-specific growth factors, a PC-based operating system, and a web-accessible portal. The updated IATP commenced initial operation in the spring of 2010 and immediately grew analytical flaws much faster than before (Figure 9.11-15): one detail yielded severity factors of 6.6 against the full-scale-test spectrum. The ASIP Team initiated a comprehensive root cause investigation to verify all steps in the IATP processing and calculation. The six-step study examined usage inputs (Figure 9.11-16), growth factor tables, external loads, software execution, data entry, and the DADTA results. This technical effort will present the results of the study and recommendations for programs that plan to update their IATP.

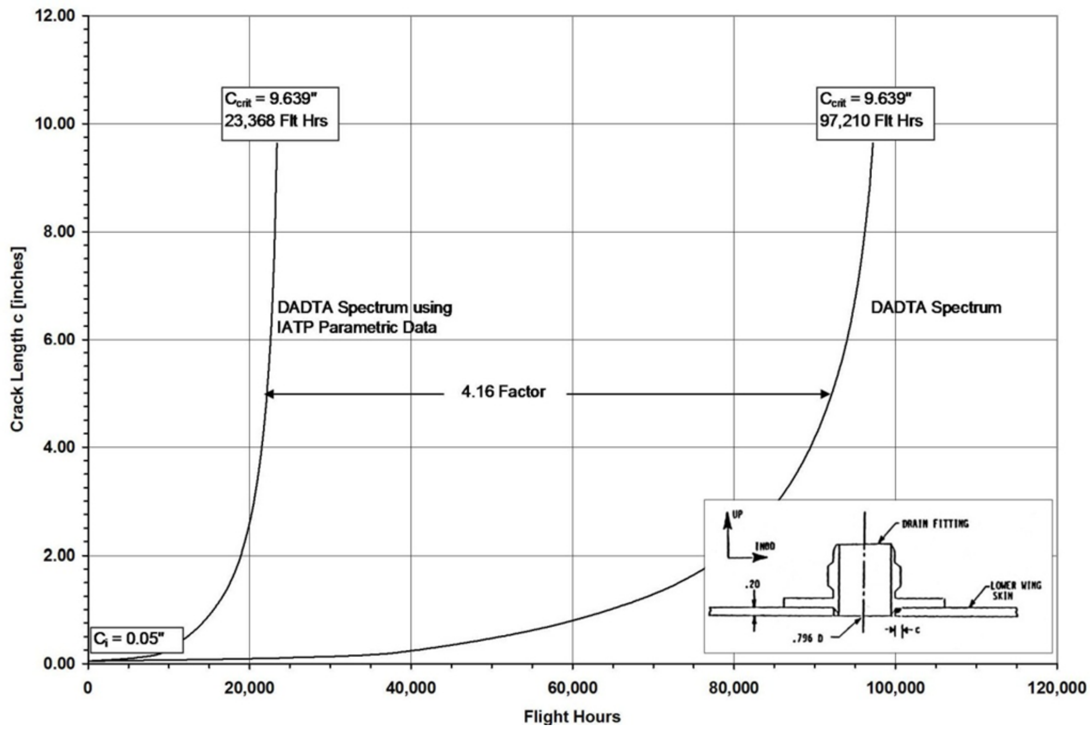


Figure 9.11-15. Crack Growth Curves From IAIP vs. DADTA

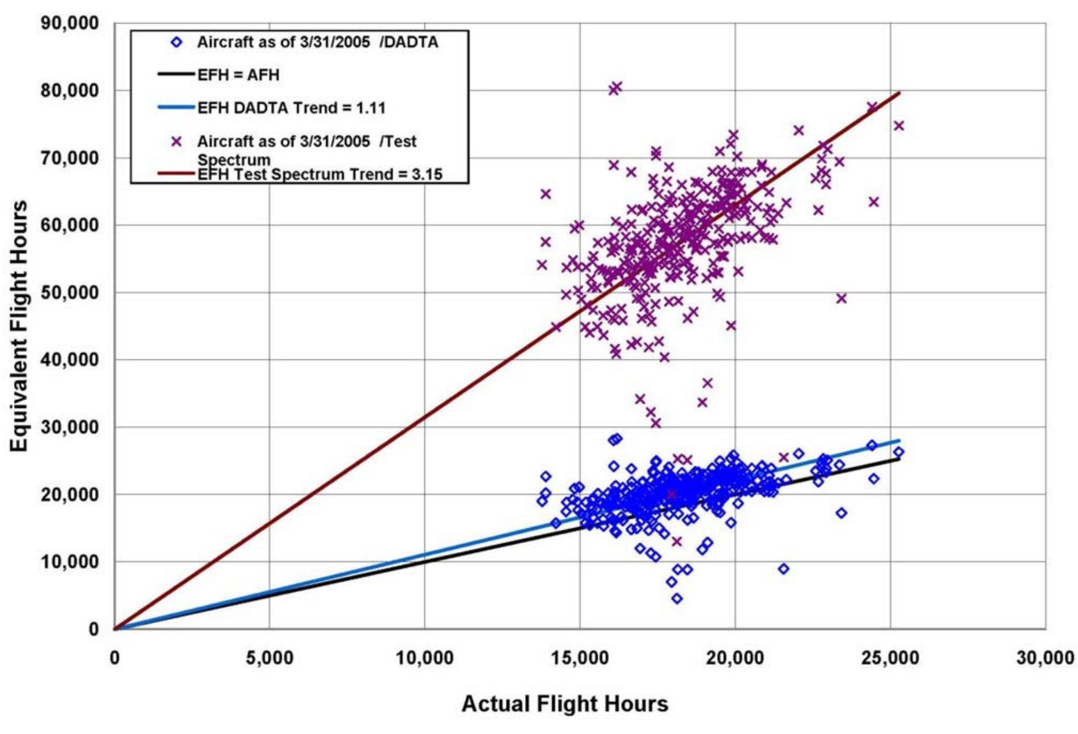


Figure 9.11-16. Usage Comparison

9.11.8. Is ASIP Still Alive? – (The A-10 Lower Wing Skin Cracking Issue)

Robert Pilarczyk, Scott Carlson and Gregory Stowe, USAF-OO-ALC

Is the United States Air Force's Aircraft Structural Integrity Program (ASIP) able to continue to monitor and maintain the high level of structural integrity needed to ensure safety of flight for our aging aircraft fleet? This question continues to be asked by those outside of the ASIP community and even U.S. Air Force leadership. Over the last year the A-10 ASIP team has helped to put a definitive answer of YES to this question.

In July of 2008 during an extended inspection of an A-10 Thunderbolt II thick-skinned Wing Center Panel (WCP), a significant fatigue crack was found in the lower aft skin at the rear spar cap near the Wing Outer Panel (WOP) attachment fitting. From this initial finding and after inspecting several other aircraft and seeing similar damage, the A-10 System Program Office (SPO) issued several Time Compliance Technical Orders (TCTO) to assess fleet-wide damage. From these TCTO inspections (Figure 9.11-17) it was determined that the A-10 fleet had a significant fatigue cracking issue in the lower wing skin. During this crisis the A-10 ASIP analysis team was called upon to develop the baseline crack growth behavior for the multiple wing configurations and the subsequent temporary and long-term repairs to fix the damaged locations. This technical effort will provide an overview of the methodology used by the A-10 ASIP analysis team to develop the critical structural repairs needed to ensure structural flight safety.

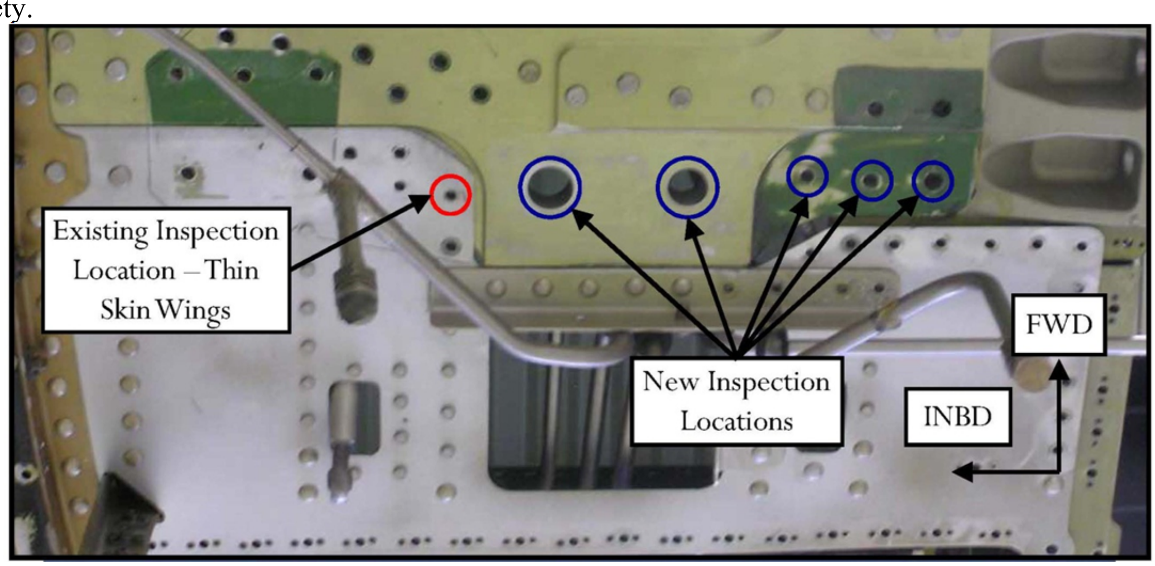


Figure 9.11-17. New Inspections Implemented

In order to design and deploy these repairs for the lower aft skin, the A-10 ASIP team developed over 20 global Finite Element Models (FEMs) for both the thick and thin-skin wing configurations. Each of these FEMs were essential in understanding the stresses in the major structural components and the load being transferred to each of the fasteners in the area. From these models a more refined local FEM was produced of just the lower aft skin in which simulated fatigue cracks were imbedded in the model and stress intensities were calculated at over 30 crack tip configurations. These stress intensities were then used to calculate "User Defined Beta Corrections" for integration into AFGROW fatigue crack growth predictions. The crack growth predictions were validated through failure analysis (Figures 9.11-18 and 9.11-19) of sections of the lower aft skin extracted from inspected aircraft.

- A Crack of Approximately 1.9 inches was Removed from Aircraft

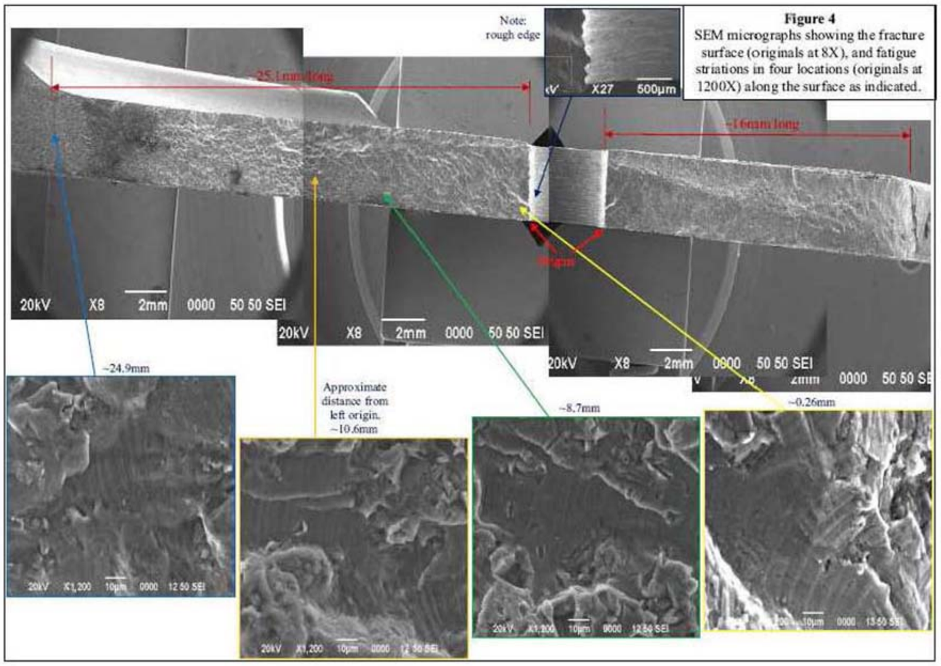


Figure 9.11-18. Failure Analysis – Detailed Fractorgraphy

- Multi-Site Nucleation
- Fretting
- Corrosion
- Crevice Environment

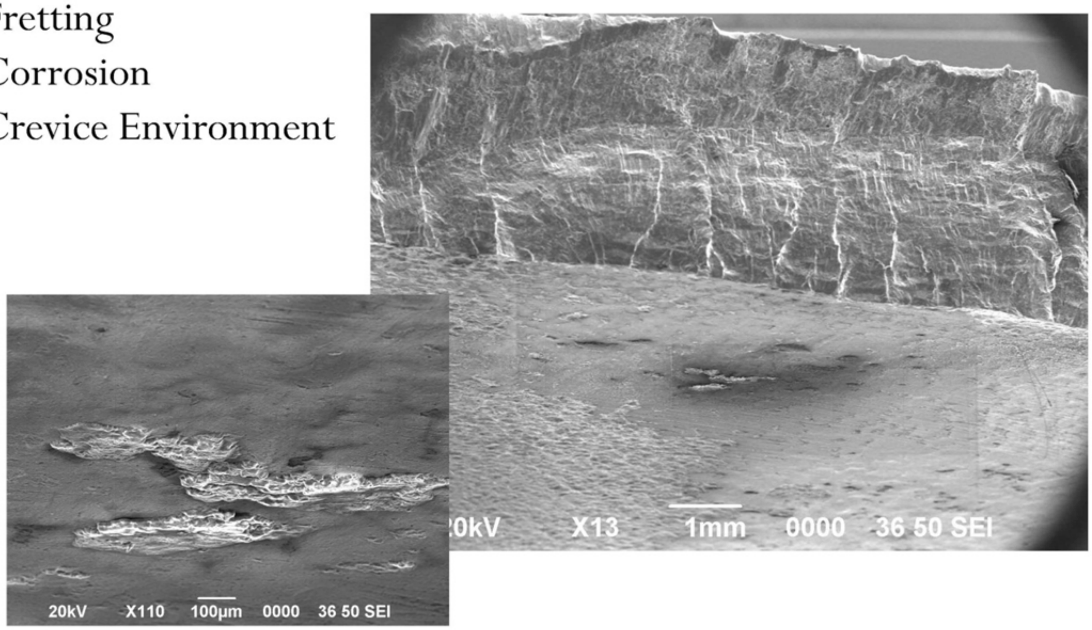


Figure 9.11-19. Failure Analysis – Radius Cracks

The ASIP program for each weapon system continues to evolve and develop over time. Through the use of the ASIP program, the A-10 was able to find this critical damage, model its impact to structural safety, analyze the risk to fleet safety, design a repair for the damage and get back to the fight with minimal downtime and zero impact to sortie rates in the AOR. Is the USAF's ASIP alive? Yes, and thriving! Through the effective use of ASIP it will be possible to ensure the safety of our weapon systems while extending their service lives to meet the needs of the warfighter.

9.11.9. Overview of the CTOL and STOVL Full-Scale Static Tests on F-35 Lightning II Program

Marguerite E. Christian, Lockheed Martin Aero – Ft. Worth

The Aircraft Structural Integrity Program for the F-35 Lightning II is unique in that it includes a dedicated full-scale static test 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). 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 test programs also include provisions to accommodate certification of future weapons systems up to the capability of the aircraft structure. The three tests enable efficiencies through test consolidation and also through economies of scale. Investments were made in the test fixtures and data acquisition systems which, when coupled with efficient test protocols, enable testing to progress rapidly and efficiently.

The first of these tests, the static test of the STOVL design, is nearing completion. The pace of STOVL testing achieved to date is unprecedented. The second static test being performed on the CTOL design began in the third quarter of 2009 (Figure 9.11-20). This technical effort outlines the number and type of load conditions tested (Figures 9.11-21 and 9.11-22) as well as the results to date for each static test program. Test efficiencies resulting from the investment in the test fixtures and data acquisition systems, the test protocol and the tools developed for efficiently handling the large quantity of data produced will be described.

AG-1 (CTOL)		BG-1 (STOVL)
159	Load Rams	164
3335	Strain and Deflection Channels	3711
163	Test Runs	211

Figure 9.11-20. Test Requirements Comparison of CTOL and STOVL

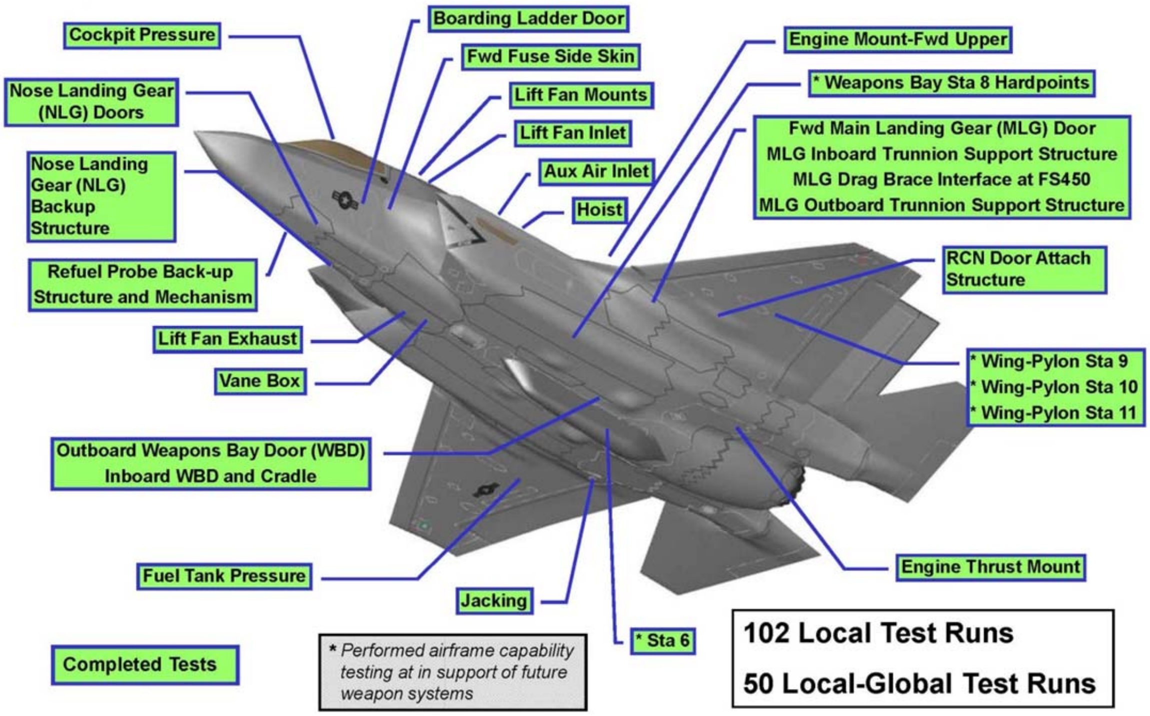


Figure 9.11-21. STOVL Local Static Tests

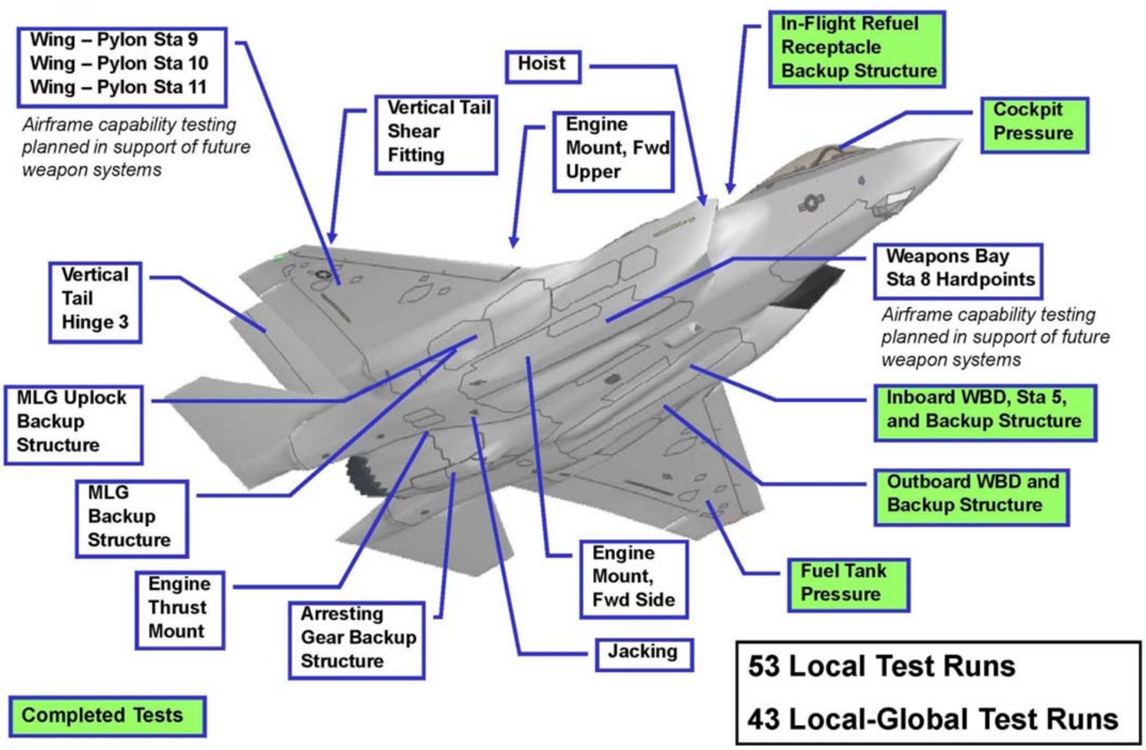


Figure 9.11-22. CTOL Local Static Tests

The F-35 Static Test Program, developed to satisfy the requirements of MIL-STD-1530C, has demonstrated the structural integrity of the F-35 STOVL design efficiently and provides a model for the remaining variants as well as future aircraft programs.

9.11.10. Development of a Full-Scale Life Extension Fatigue Test Program for the A-10

Mark Thomsen, USAF-OO-ALC and Sebastian Grasso, Northrop Grumman Corporation

During its 30-year operational history, the usage of the A-10 has changed dramatically. Mission type, mission mix, severity and frequency of flight operations have all continually evolved reflecting the updated warfighting doctrines of the United States Air Force (USAF).

Originally designed using durability and then damage tolerance methodologies, the A-10 has undergone three major spectrum updates, with the fourth reflecting Post-Desert-Storm usage. This fourth spectrum was completed in 1999 and has been used for damage tolerance analysis and for the recently completed wing-only fatigue test. This spectrum was also used on the fuselage/empennage fatigue test.

The wing-only fatigue test, completed in 2004, identified the incremental life improvement from SLEP modifications, and provided the hard data needed to justify the decision to procure new wings for the A-10. In 2007, the production contract for new wings was awarded, with a first article scheduled to be on dock in late 2010.

Based on the test history of the fuselage and empennage during weapon system development and the value of the recent full-scale wing-only fatigue testing, the life extension fuselage and empennage test program was developed (Figure 9.11-23). This testing is helping to assess the ability of the remainder of the airframe to meet the new service requirement. This testing is providing significant insight into the airframe's structural integrity by identifying fatigue susceptible locations early, allowing for timely development and implementation of structural repairs and modifications (Figures 9.11-24 and 9.11-25).

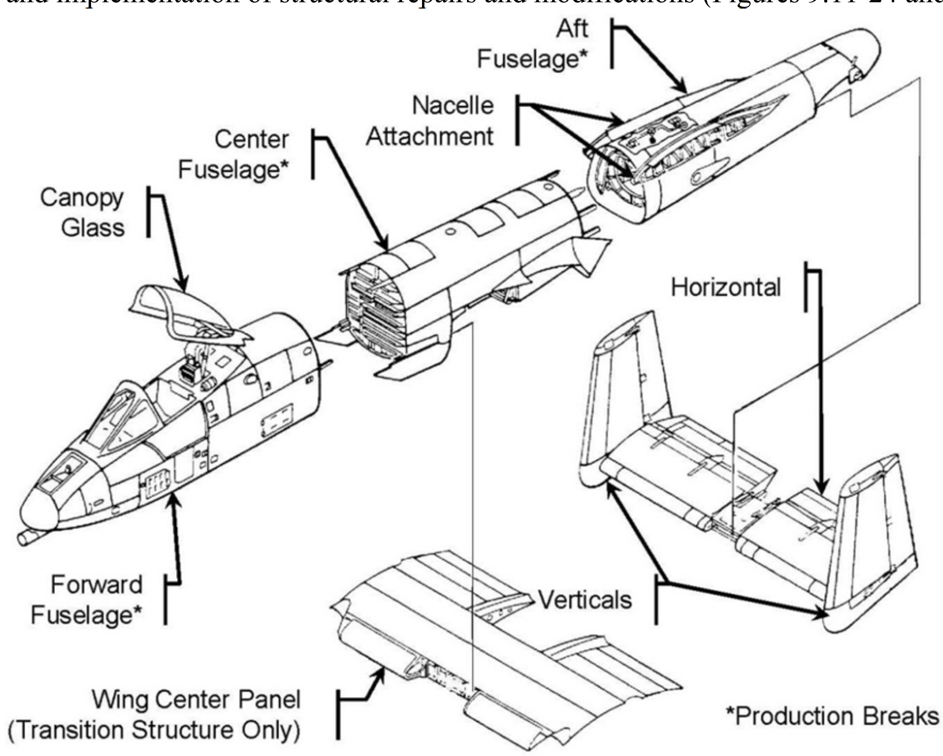


Figure 9.11-23. Life Extension Test Article

Figure 9.11-24. Catastrophic Failure

Figure 9.11-25. Repair

Recent fleet inspection findings have closely mirrored test results. Accelerated life testing has given a two-year head start in development of permanent repairs and modifications, many of which have already been proven out on the test article.

The Aircraft Structural Integrity Program (ASIP) plan calls for updated analysis and testing to support aircraft operations through the updated service requirement. A representative flight spectrum is crucial for fleet management. It allows more accurate inspection intervals, helps ensure structural integrity and significant savings can be realized in maintenance costs due to a combination of early detection and timely repairs.

To support the updated ASIP, a new data recorder has been installed and is collecting data to continue monitoring the aircraft usage. So far, this new data recorder has uncovered numerous significant changes in usage compared to previous spectra. Most significant is the addition of maneuver loading during cockpit pressurization. Improved targeting capability and revised threat avoidance philosophies have resulted in more severe maneuvers being executed at increased altitudes.

New flight data has also been used to validate the interaction of gunfire and maneuvers during flight for test spectra development. The gun fire rate has also changed since the A-10 entered service. In addition, changes in tactics have shown that the gun is not used as frequently, with greater use of stand-off weapons at higher altitudes. Even so, the original testing was in excess of what the fleet has demonstrated over the past 20 years. Additional testing was conducted in order to account for projections of usage into the future based on more recent gun usage profiles and to assess the durability of the structure in the presence of maintenance-induced damage.

This technical effort will document the challenges of ensuring structural integrity for an aging aircraft that has experienced significant changes in usage throughout its service history. Additionally, the specific program and test realities will be presented as they impact and influence the overall goal of the test.

9.11.11. The F-16 Sustainment ASIP: A 20-Year Retrospective

Kevin Welch, Lockheed Martin Aero – Ft. Worth

In June 2010, the F-16 sustainment ASIP hosted it's 20th program review. During those 20 years, the F-16 sustainment ASIP has evolved from a small, uncertain effort into a large international well-organized effort currently serving twenty-two operating agencies world wide (Figure 9.11-26). This F-16 sustainment ASIP has been widely recognized as a superior effort. This program has enjoyed significant programmatic successes and technical achievements.

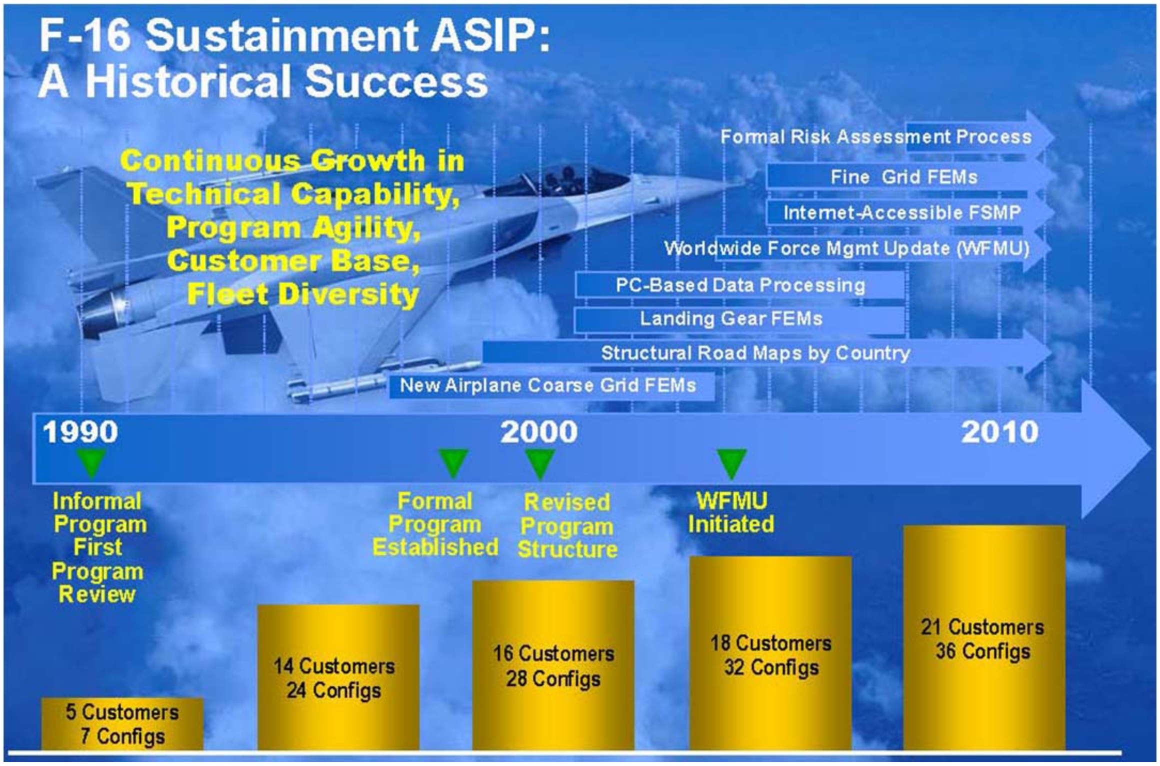


Figure 9.11-26. Historical Success of F-16 Sustainment ASIP

This technical effort will provide a review of program successes and technical achievements of the past 20 years and identify trends for the future. The program successes show how the program has evolved over time to respond to the needs of the worldwide F-16 operators, created an environment conducive to technical achievement, and provided a flexible programmatic model easily adaptive to continually changing customer needs. The technical effort will focus on significant technical achievements which are key elements of the technical infrastructure needed to maintain long term structural integrity of the worldwide F-16 fleet. These achievements have contributed to advancing the state-of-the-art and will provide enduring value to the F-16 operators.

This technical effort will also show how the programmatic successes and technical achievements operate in a feedback loop:

- Programmatic success provides an environment conducive to technical achievement.
- A stable program framework provides the capability to implement long-term plans.
- Long-term plans facilitate commitment of skilled personnel and resources.
- The resulting technical achievements encourage greater operator participation and detailed involvement.
- This active participation helps to continually define and refine the programmatic framework which encourages continued technical achievement.
- Technical achievements provide a foundation for further technical achievement.
- Continued technical success results in quality products and services which motive program participants to maintain and evolve the program framework.

The continued feedback loop ensures the program is responsive to both the common and unique needs of the individual operators and that the program provides the highest quality services and results for an affordable cost.

9.11.12. Nonconforming Titanium – USAF Response to the Threat of Substandard Material

Thomas Fischer, USAF-AFMC/EN

Nonconforming titanium (titanium billet that has been improperly sold as fully-processed plate or bar products) was discovered in the USAF's parts inventory in 2004 (Figure 9.11-27). Since that time, through efforts of the USAF, the US Department of Justice, and various other governmental agencies, the prevalence of nonconforming titanium and the various means by which it has entered DoD, NASA, and US industry supply chains has been investigated. The potential effects of nonconforming titanium include, in the worst case, unanticipated and unpredictable premature failure of safety-of-flight components and critical safety items. The USAF initiated a comprehensive effort to understand the severity of the problem and to mitigate the risk that it poses to the structural integrity of USAF weapons systems in September of 2009. Led by the Air Force Materiel Command's Titanium Task Force, this effort is comprised of multiple approaches that involve the engineering, maintenance, logistical, and legal communities. In addition, the USAF has partnered with other DoD services, and with DCMA, DLA, DoJ, FAA, and NASA organizations. This technical effort will review the progress made during the last year in the areas of bounding the problem, conducting testing, interacting with industry (airframe manufacturers as well as material suppliers), communicating internally and externally, and coordinating actions with other affected organizations. Results from mechanical testing of Ti-6-4 and Ti-6-6-2 plate and billet product forms will be presented (Figure 9.11-28), and the ramifications of these test results to aircraft structural integrity and operational risk assessments will be discussed. Mid-range and long-range plans for mitigating the risk of nonconforming parts that are either installed, in stock, or have the potential for being fabricated in the future will also be outlined.

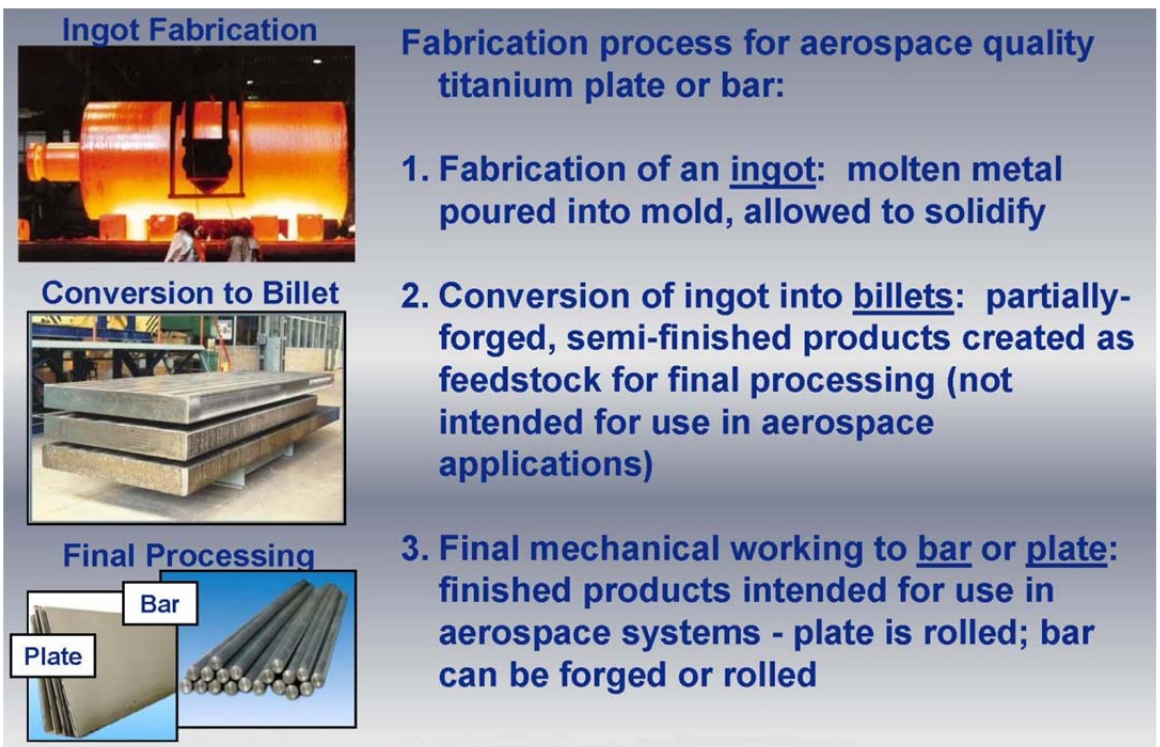


Figure 9.11-27. Titanium Finished Products

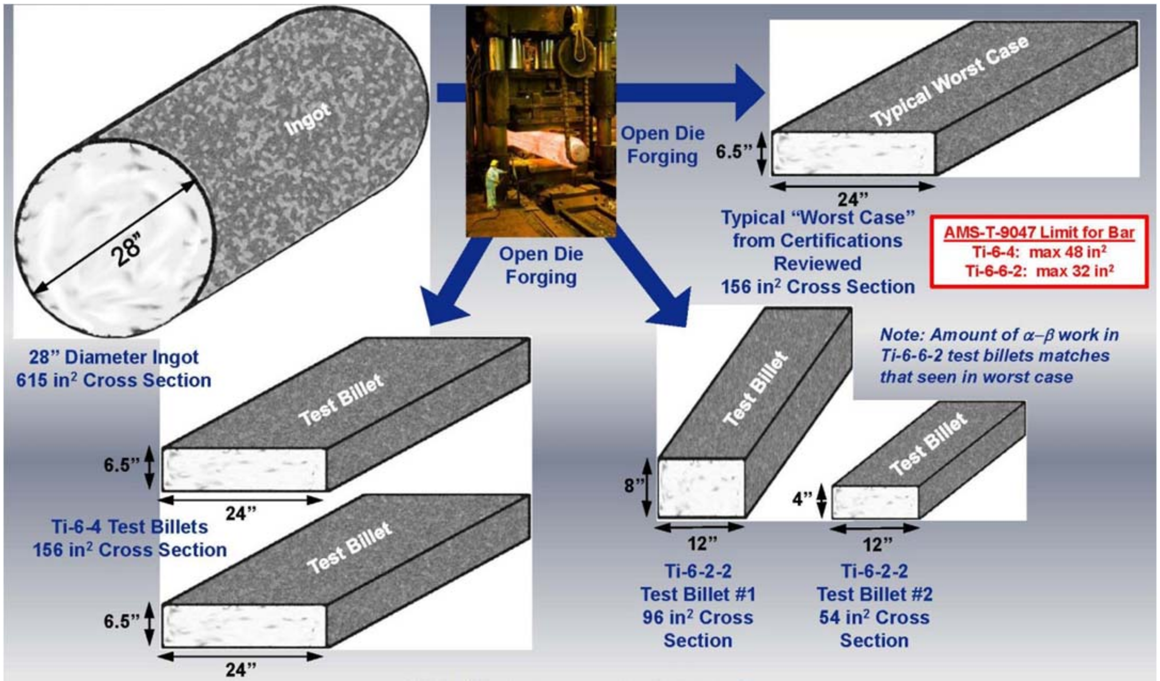


Figure 9.11-28. Test Material Selection

9.11.13. F/A-18A-D Service Life Assessment Program (SLAP)/Service Life Extension Program (SLEP)

Rigo Perez, The Boeing Company

The F/A-18A-D Service Life Assessment Program (SLAP) started in 2001 with a focus on airframe components affected primarily by landings, catapult launches, arrestments, and landing gear retract cycles (Figure 9.11-29). A second more general SLAP phase covering structure affected by ground and flight loads began in 2005 and concluded in 2008. A target safe life of 10,000 SLAP Flight Hours (FH) was eventually defined for SLAP/SLEP (Service Life Extension Program).

Figure 9.11-29. F/A-18A/C

Both F/A-18A-D SLAP phases began with fleet usage analysis. The master event spectra were developed to represent a 90th percentile level of fatigue damage in the fleet. Significant ground and flight loads analysis efforts were conducted. Global finite element models were developed to represent primary structure for the entire airframe [1]. This finite element task included correlations with strain gage measurements made during the full-scale tests.

Selection criteria were defined to identify "hot spots" or locations for fatigue life assessment. Using analysis software developed during SLAP, the fatigue spectra for the "hot spots" were generated and fatigue lives were then computed [1]. The fatigue analyses were supported by existing full-scale-fatigue-test data. Additional full-scale-fatigue tests were not required because the F/A-18A-D program had already completed extensive tests. These tests had simulated several design lifetimes and demonstrated significant airframe capability. However, SLAP did include teardowns and inspection of high-time retired jets to assess damage accrued during actual operation [1, 2].

As discussed in Reference [3], the fatigue life assessment indicated that many of the locations analyzed would need maintenance actions. This led to a multi-phase SLEP effort, which started in 2008 and is in work at the time of this writing. Some of the early SLEP tasks include definition of notional repair concepts, finite element model updates, spectrum software upgrades and fail safe analysis. A detailed criticality assessment of the hot spots was conducted to help manage and prioritize further SLEP actions.

References

[1] "F/A-18A-D Service Life Extension Program," NAVY SLAP Team, Tactical Aircraft Strength Branch, Presentation, 6/28/2007.

[2] R. Perez, C. Rose, T. Fellner, "A Database for Documenting Fatigue Test and Fleet Teardown Findings," 2005 ASIP Conference, Memphis, Tennessee, 11/29/2005 – 12/1/2005.

[3] D. Fulghum, "Hornets Aging Quickly," Aviation Week and Space Technology, 3/31/2008.

[4] R. Perez, "SLAP/SLEP Fundamentals," Aircraft Airworthiness & Sustainment 2010 Conference, Austin, TX, May 13, 2010.

9.11.14. F/A-18E/F Service Life Assessment Program (SLAP)

Rigo Perez, The Boeing Company

Phase A of the F/A-18E/F airframe Service Life Assessment Program (SLAP) started in 2008 and is in work at the time of this writing (Figure 9.11-30). A target safe life of 12,000 Flight Hours (FH) was defined. This phase includes the following technical tasks: (1) Fleet usage analysis and development of master event spectra representative of a 90th percentile aircraft in terms of fatigue damage; (2) Global finite element models to represent primary structure for the entire airframe. This finite element task includes correlations with strain gage measurements made during the original full scale tests; (3) Fatigue spectrum generation software; (4) Ground and flight loads analysis; and (5) Selection of fatigue hot spots.

Figure 9.11-30. F/A-18F

The phase B planned at the time of this writing will include spectrum generation and fatigue life analysis of the hot spots selected. The fatigue analyses will be supported by correlation to existing full-scale-fatigue-test data. This correlation will make use of test results obtained as part of the structural certification of the aircraft (Figure 9.11-31). The major fatigue tests were [1]:

- FT50: Entire airframe; Simulated three design lifetimes
- FT76: Block II forward fuselage; Simulated two design lifetimes
- FT77: Redesigned wing; Simulated three design lifetimes, with some areas tested to four lifetimes

Figure 9.11-31. F/A-18E Full-Scale Fatigue Test

The findings have been documented in a database for use during the SLAP program [2]. This integration of SLAP with existing programs has proven very valuable.

References

[1] T. N. Callihan, "F/A-18E/F Full Scale Structural Fatigue Testing," International Committee on Aeronautical Fatigue, Hamburg, Germany, 6/8/2005.

[2] R. Perez, C. Rose, T. Fellner, "A Database for Documenting Fatigue Test and Fleet Teardown Findings," 2005 ASIP Conference, Memphis, Tennessee, 11/29/2005 – 12/1/2005.

[3] R. Perez, "SLAP/SLEP Fundamentals," Aircraft Airworthiness & Sustainment 2010 Conference, Austin, TX, May 13, 2010.

9.11.15. F-16 Block 50 Full-Scale-Durability Testing

Bryce Harris and Kimberli Jones, USAF-OO-ALC

A portion of the United States Air Force (USAF) F-16 fleet is approaching their Certified Service Life of 8,000 equivalent flight hours (EFH). This aircraft fleet is especially critical to the USAF. In order to certify the aircraft beyond the current certified service life, a full-scale-durability test is required.

On some of the fleet, no full-scale-durability test was originally performed. The basis of the certified life limit is relevant data from the Block 30 full-scale-fatigue test, shown in Figure 9.11-32, and additional block- specific component testing and analysis. Finite-element models are continually updated and refined via the F-16 Aircraft Structural Integrity Program. An extensive database of field inspection results is available for fleet analysis needs and increases the accuracy of F-16 crack-growth projections.

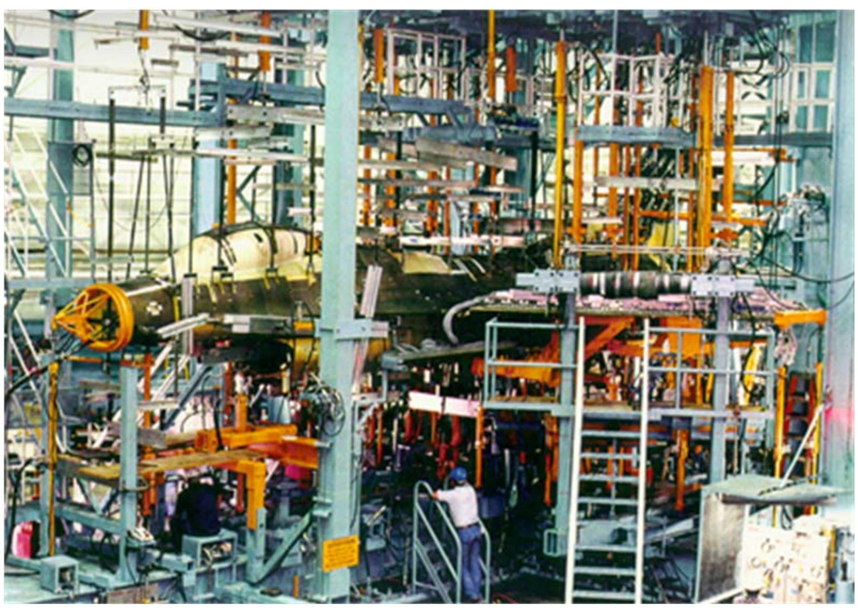


Figure 9.11-32. Block 30 Full-Scale-Durability-Test Fixture

A full-scale-durability test has been planned to certify applicable F-16 aircraft to a life limit beyond 8,000 EFH. The test provides data necessary to identify areas requiring structural modification, correlate structural analysis, and design modifications that meet extended-service-life requirements. A block 50 airframe, with previous in-service usage, has been acquired as the test aircraft.

Testing is expected to begin in late 2014, with an in-depth teardown of the test airframe to follow. F-16 modifications to allow for service-life extension will be developed and implemented as required, both during and after the durability test.

Point of Contact:

- Mr. Bryce Harris and Dr. Kimberli Jones, USAF F-16 ASIP, OO-ALC/GHBEX, 801-777-9381 and 801-777-3887

9.11.16. Durability and Damage Tolerance Testing of Starship Forward Wing with Large Damages

Waruna Senevirathe and John Tomblin, University of Wichita – NIAR; and Curtis Davies, FAA – William J. Hughes Technical Center

A methodology synthesizing the life factor, load enhancement factor, and damage in composites is proposed to determine the fatigue life of a damage-tolerant composite airframe. This methodology further extends the current practice during damage-tolerance certification to focus on the most critical damage locations of the structure and interpret the structural and loads details into the most representative repeated load testing in element level to gain information on the residual strength, fatigue sensitivity, inspection methods and inspection intervals during full-scale test substantiation. The proposed methodology was validated with several full-scale test examples of the Beechcraft Starship forward wings with large impact damages on the front and aft spars (Figure 9.11-33). Full-scale test articles were named as ST001 through ST006. The Beechcraft Starship forward wing was designed with a significant amount of conservatism. Thus, the Beechcraft design limit and ultimate loads (BDLL and BDUL, respectively) were adjusted for the purpose of this research following the three static tests ST001 through ST003 using a conversion factor. These redefined limit and ultimate loads are referred to as NIAR research limit and ultimate loads (NRLL and NRUL, respectively).

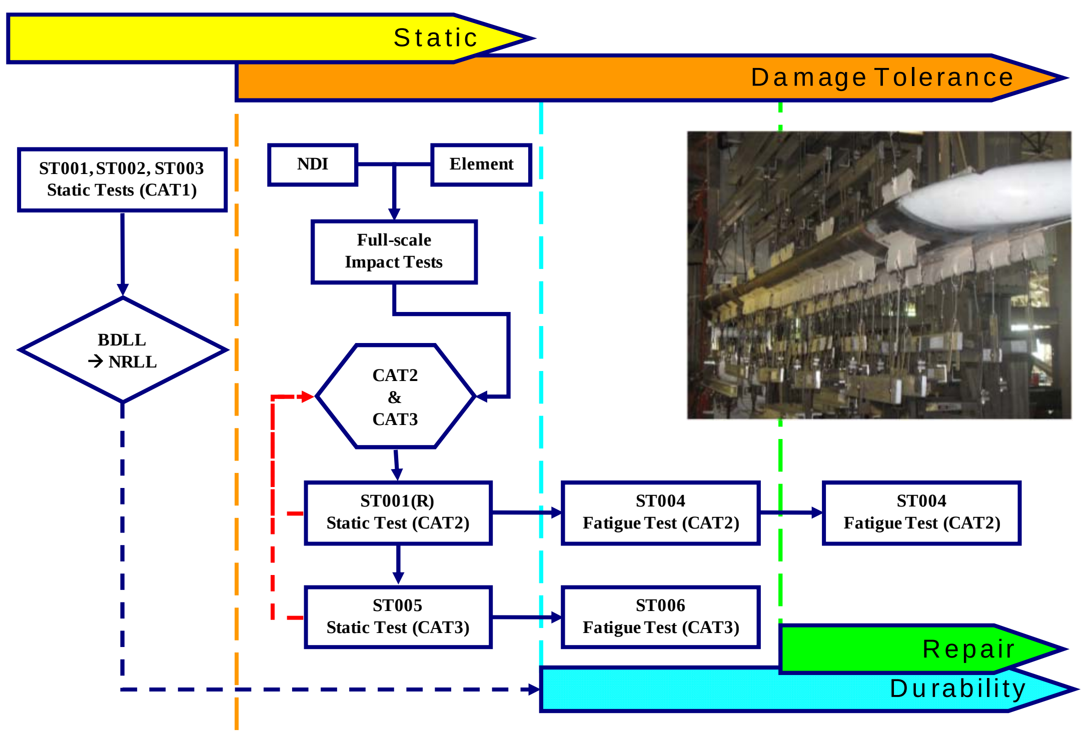


Figure 9.11-33. Outline of the Full-Scale DaDT Test Program

The front spar of the forward-wing structure is the primary load path and a large impact damage that results in a decrease of the residual strength to its limit-load was considered as a category 3 (CAT3) damage. A large impact damage that was on the aft spar was considered as a category 2 (CAT2) damage and its contribution to the final failure of the structure was secondary. Several element-level tests were conducted to determine the impact parameters for inflicting damage on full-scale structures. Strategic placement of strain gages around the damage and near critical areas provided real-time feedback during damage tolerance tests. The strain data provide information similar to a built-in health monitoring system and provide details in real time to assess the state of the damage, i.e., propagation or not, and any global effects on the structure due to possible damage growth.

In order to prevent unintentional failure of a damaged article during the durability and damage tolerance (DaDT) testing, especially when investigating extremely improbable high-energy impact threats that reduce the residual strength of a composite structure to limit load, rigorous inspection intervals are required. The probability of failure of the damaged structure with the enhanced spectrum loads can be evaluated using the proposed cumulative fatigue reliability model (CFR), which was validated through a full-scale test demonstration of a damaged article at the critical load path. Information from this model can be used also to allot economical and reliable inspection intervals during service based on a target reliability and a critical damage threshold.

A full-scale DaDT test conducted with CAT2 on the aft spar using the updated LEFs based on the design details of the Starship forward wing structure demonstrated the repeated life requirements according to the proposed load-life-damage hybrid approach, and the post-DaDT residual strength requirements. The Starship forward-wing DaDT test article with CAT3 on the front spar demonstrated the capability of the cumulative fatigue reliability model to predict the damage growth in terms of reliability and the capability of the model to determine the inspection levels. Although it is not a one-to-one correlation for the damage propagation or its size, the cumulative fatigue reliability model highlighted load segments that resulted in gradual progression of local damage, such as possible matrix cracks, and the global impact of high loads that resulted in evident damage growth.