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AIR VEHICLE INTEGRATION AND TECHNOLOGY RESEARCH (AVIATR)

Delivery Order 0002: Condition-Based Maintenance Plus Structural Integrity (CBM+SI) Strategy Development

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Lockheed Martin Corporation

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Final Report

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14. ABSTRACT
Military aircraft platforms are currently confronted by two pressing issues: increasing maintenance costs and decreasing availability. Nearly 30% of Air Force platform Operations & Support (O&S) costs are for structure inspection and repair. Condition-Based Maintenance plus Structural Integrity (CBM+SI) offers an opportunity to achieve significant platform availability increases and maintenance hour per flight hour decreases.

A promising solution to both of these issues is the application and integration of CBM+ and Structural Health Monitoring (SHM) technologies and processes into the Aircraft Structural Integrity Program (ASIP) activities. Lockheed has developed an approach to integrate CBM+SI into a new ASIP framework and demonstrate the potential improvements in aircraft availability, total cost of ownership and maintenance man-hours per flight hour.

In order to spur the adoption of CBM+SI into the Aircraft Structural Integrity Program of each platform, the benefits to the USAF from employing CBM+SI must be clearly established. The benefits with the most impact for the Air Force are, in order of importance, increased aircraft availability, reduction in total cost of ownership, and reduction in maintenance hours per flight hour. Widespread adoption of CBM+SI into the Aircraft Structural Integrity Programs, and a strong technical pull developed for the enabling technologies; will occur once a solid business case is made for CBM+SI.

Specific Objective: Develop and demonstrate a CBM+SI strategy for at least one structural application on a United States Air Force weapons platform. As part of this demonstration, the benefits to the Air Force as result of employing this CBM+SI strategy shall be determined.

15. SUBJECT TERMS
Condition-Based Maintenance Plus (CBM+), Structural Health Monitoring (SHM), Aircraft Structural Integrity Program (ASIP), Prognostics

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1. INTRODUCTION

Nearly 30% of Air Force platform Operations & Support (O&S) costs are for structure inspection and repair, CBM+ Structural Integrity (SI) offers an opportunity to achieve significant platform availability increases and maintenance hour per flight hour decreases.

The challenges to achieving a CBM+ maintenance concept for structures includes precursor prognostic time, prognostics false alarms and non-detects, airframe certification using CBM+ versus scheduled inspections, sensor integrity, sensor network complexity, and data processing infrastructure constraints.

2. BACKGROUND

Military aircraft platforms are currently confronted by two pressing issues: increasing maintenance costs and decreasing availability (Figure 1). A promising solution to both of these issues is the application and integration of CBM+ and Structural Health Monitoring (SHM) technologies and processes into the Aircraft Structural Integrity Program (ASIP) activities. Lockheed has developed an approach to integrate CBM+SI into a new ASIP framework and demonstrate the potential improvements in aircraft availability, total cost of ownership and maintenance man-hours per flight hour.

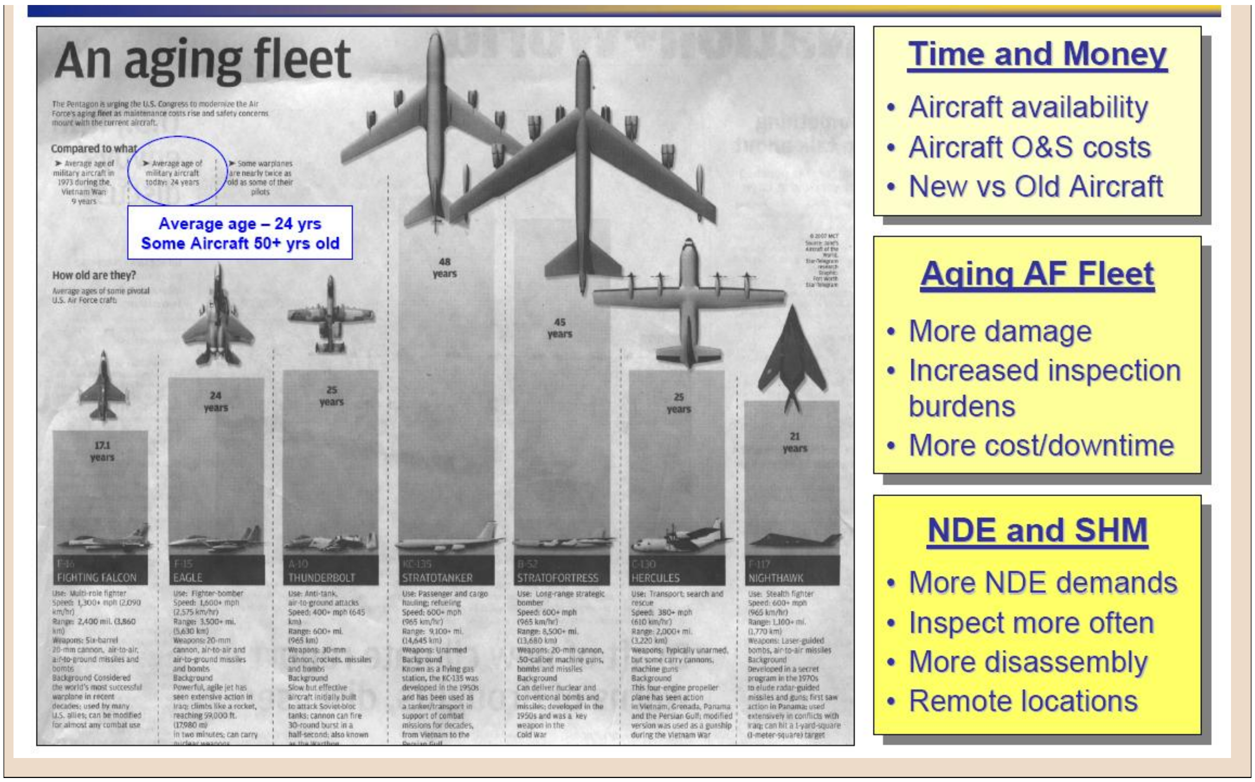


Figure 1. Aging Fleet is Driving Increases in O&S Costs and Decreases Availability

2.1 CBM+SI Customer Perspective

This section addresses and describes the customer perspective on CBM+ and its application to Structural Integrity as reflected in the CBM+SI Task Order solicitation.

2.1.1 CBM+ = Reduced Maintenance Costs and Increased Availability

Condition-Based Maintenance plus Prognosis (CBM+) has been proposed as an alternative to current maintenance practices that would both reduce maintenance costs and increase availability.

> *"Condition-Based Maintenance (CBM) can be defined as a set of maintenance processes and capabilities derived from real-time assessment of weapon system condition obtained from embedded sensors and/or external tests and measurements using portable equipment. The goal of CBM is to perform maintenance only upon evidence of need.*

CBM+ expands on these basic concepts, encompassing other technologies, processes, and procedures that enable improved maintenance and logistics practices. These future and existing technologies, processes, and procedures will be addressed during the capabilities planning, acquisition, sustainment, and disposal of a weapon system."

From the Air Force Logistics Management Agency (AFLMA) report, USAF Condition-Based Maintenance Plus (CBM+) Initiative (AFLMA Report LM200301800, Sep 2003)

Structural Integrity is added to the moniker in order to emphasize the application of CBM+ to airframe structure within the context of the Aircraft Structural Integrity Program.

2.1.2 CBM+ Supports Total Life Cycle Systems Management (TLCSM)

"CBM+ is the primary reliability driver in the Total Life Cycle Systems Management (TLCSM) supportability strategy of the Department of Defense. In concert with the other TLCSM enablers, …, CBM+ strives to optimize key performance measures of materiel readiness - materiel availability, materiel reliability, mean downtime, and ownership costs."

From The Materiel Readiness and Maintenance Policy web page.

However, few of the examples of CBM+ provided at this web site, in the AFLMA report and in the literature address structural issues in the context of CBM+. Most of the examples demonstrate CBM+ applied to engines, mechanical systems, and electronics. Thus, the data needed to demonstrate the cost effectiveness of CBM+ for airframe structures is lacking.

2.1.3 CBM+SI Objectives and Tasks

The primary objective of the CBM+SI Task Order is to develop and demonstrate CBM+SI for airframe structures. In order to spur the adoption of CBM+SI into the Aircraft Structural Integrity Programs of each platform, the benefits to the USAF from employing CBM+SI must be clearly established. The benefits with the most impact for the Air Force are, in order of importance, increased aircraft availability, reduction in total cost of ownership, and reduction in maintenance hours per flight hour. Widespread adoption of CBM+SI into the Aircraft Structural Integrity Programs, and a strong technical pull developed for the enabling technologies, will occur once a solid business case is made for CBM+SI.

Specific Objective:

Develop and demonstrate a CBM+SI strategy for at least one structural application on a United States Air Force weapons platform. As part of this demonstration, the benefits to the Air Force as result of employing this CBM+SI strategy shall be determined.

2.1.4 Tasks

The tasks identified to satisfy the stated objectives are:

- Develop and demonstrate CBM+SI strategies for structural applications on a USAF weapons platform to include:
    - Developing integrated, predictive maintenance approaches, which minimize unscheduled repairs, eliminate unnecessary maintenance, and employ the most cost-effective maintenance health management approaches.
    - Determining an optimum mix of maintenance technologies (e.g. diagnostics and prognostics)
    - Identifying the optimum opportunity to perform required maintenance, thereby increasing the number of assets in operational status
    - Providing real-time maintenance information and accurate technical data to technicians and logisticians that will expedite repair and support processes and return equipment to operational status
- Determine the benefits to the USAF as a result of these applications of CBM+SI in terms of:
    - Total cost of ownership
    - Aircraft availability rates
    - Maintenance hours per flight hour

2.1.5 Statement of Objectives (SOO) Summary

1. Develop one or more business cases demonstrating the benefits of CBM+SI over the current USAF maintenance practices.

2. Develop a framework for CBM+SI application to a USAF fleet.

2.2 What is CBM +

CBM+ has a broad scope: It is built upon the concept of Condition-Based Maintenance, but is enhanced by reliability analysis.

The Air Force (AF) slightly modified the CBM+ definition to clearly communicate that CBM+ is integrated throughout the life of the AF weapon systems.

Ten enabling technologies and concepts constitute the initial AF baseline for achieving the Department of Defense (DoD) vision for CBM+ implementation. These technologies are:

- Prognostics
- Diagnostics
- Portable Maintenance Aids
- Interactive Electronic Technical Manuals
- Interactive Training
- Data Analysis
- Integrated Information Systems
- Automatic Identification Technology

While all of these technologies could support the effective application of CBM+ for structures, Lockheed views the key enabling capability is the ability to know the current material condition of the structure and the additional ability to estimate the future material condition (Diagnostics – Current Material Condition and Prognostics – Future Material Condition).

Knowledge of the current state is best provided by the use of structural health monitoring that provides direct assessment of the current state. Usage monitoring, while not as effective as a direct assessment of the state of the structure, can when coupled with reliable models can also support implementation of CBM+SI.

The assertion that the ability to directly determine the current material condition of the structure is the most effective approach to predict the future state and to successfully implement CBM+SI is the basis for Lockheed's approach to the CBM+SI Task Order.

Lockheed understands that for platforms that do not capture usage data for each tail number, expanding the number of tails capturing this data or expanding the range of usage data is a viable approach for achieving the CBM+SI objectives. Enhance usage data coupled with advanced models and analysis can be effective in increasing inspection intervals and maintaining an acceptable risk. But aircraft usage data is still an indirect method for determining the current material condition and predicting the future material condition.

For the current material condition the usage data approach is still conservative, since variations in individual aircraft and the confidence in analytical tools result in some degree of uncertainty, resulting in inspections that may be more frequent than necessary.

For the future material condition the usage data approach is even more conservative, since the degree of uncertainty in the current material condition must be part of the uncertainty in the future condition as well and the variations in individual aircraft and the confidence in predictive analytical tools result in a larger degree of uncertainty, resulting in inspections that may be more frequent than necessary.

Traditional Non Destructive Inspection (NDI) coupled with visual inspection provides a high confidence assessment of the current material condition which supports predictions of future material condition that can be less conservative since there is a higher confidence in the current condition. One problem with NDI is that in many cases time consuming and special skills and equipment are required. Additionally some disassembly of the aircraft is required to provide access. For these activities are costly and impact aircraft availability.

The application of embedded Structural Health Monitoring (SHM) technology can augment and in some cases replace NDI. By having an SHM system embedded in the aircraft it is possible to have a high confidence assessment of the current material condition and data to support high confidence predictions of the future material conditions. Since the system is imbedded and automated the assessment of the current condition can be very frequent or even continuously. The ability to provide frequent assessments of the current material conditions address the problem associated with usage data and modelling of changes in utilization/usage that may occur between assessments, as well as other problems associated with traditional approaches.

For the above reasons Lockheed considers SHM Technologies a key enabler for the application of CBM+ to structures. We acknowledge that while SHM is necessary it is not in and of its self sufficient for a viable CBM+SI solution. SHM technology must be integrated into a system that relies on the application of the following;

- Portable Maintenance Aids
- Interactive Electronic Technical Manuals
- Interactive Training
- Data Analysis
- Integrated Information Systems
- Automatic Identification Technology

The degree to which each of the above are applicable will depend on the specific CBM+SI application and the target platform maintenance and operation environment, but for a comprehensive CBM+SI concept applied to a weapon system all will be needed and integrated into an effective element of the weapon system support concept.

2.3 Approach

The Lockheed CBM+SI Phase I Program Plan consisted of 10 major tasks (Figure 2).

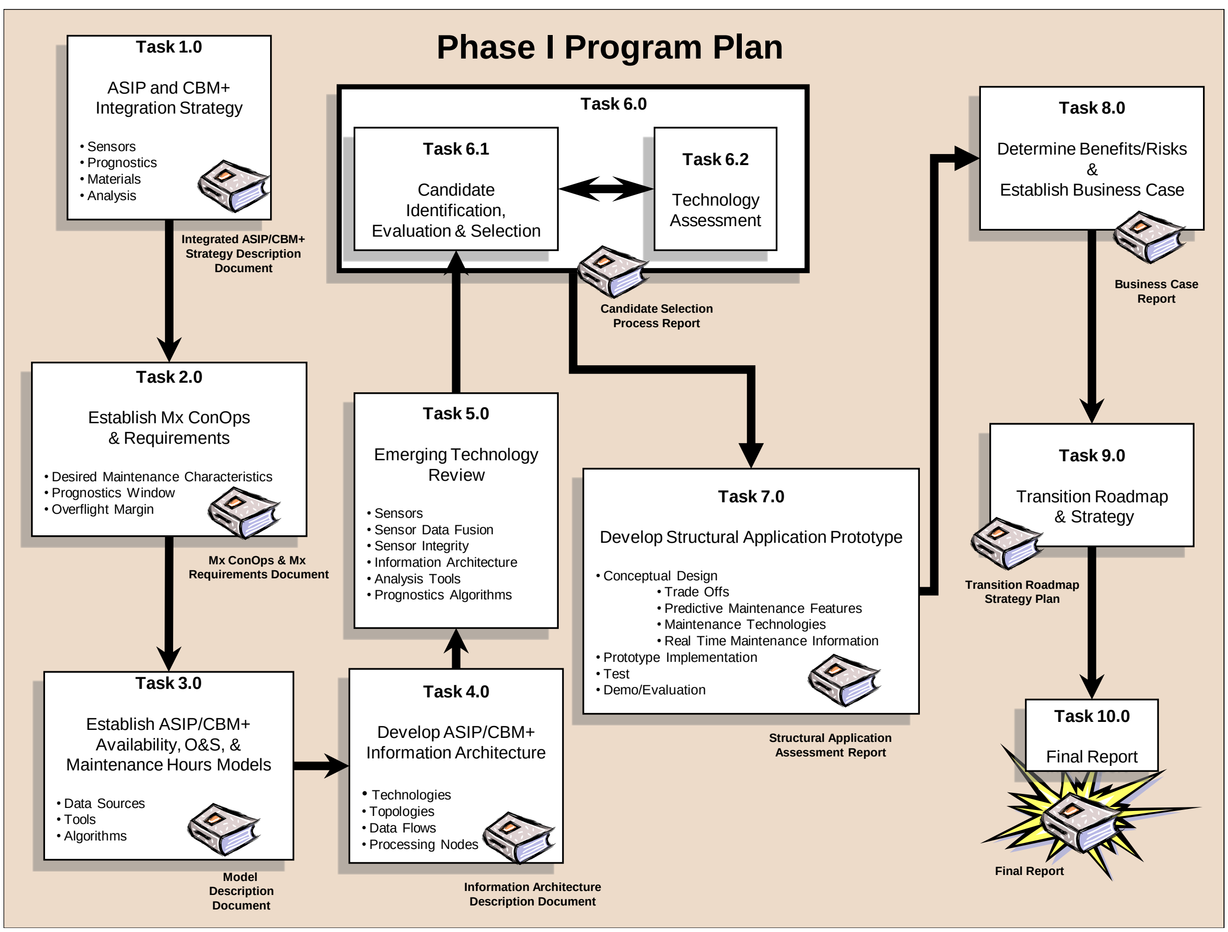


Figure 2. Program Plan

Even though Figure 2 depicts a serial process, many tasks were re-visited as information from later tasks became available. The following paragraphs will briefly discuss each task. More detailed task descriptions and results will be provided later.

2.3.1 Task 1 - ASIP and CBM+ Integration Strategy

Task 1 involved the development of a conceptual, modified ASIP, for both existing and new platforms, that explicitly addresses CBM+ and the development and validation of the technologies required to support it (SHM, prognostics, etc.). For example, for ASIP Task I, SHM technologies were assessed. Other notable additions are the requirement for a production SHM sensor technology capability assessment in ASIP task II and the requirement for production SHM validation in ASIP task III. The following Figure 3 identifies the modified and new elements recommended to support integration of CBM+ into the existing ASIP process.

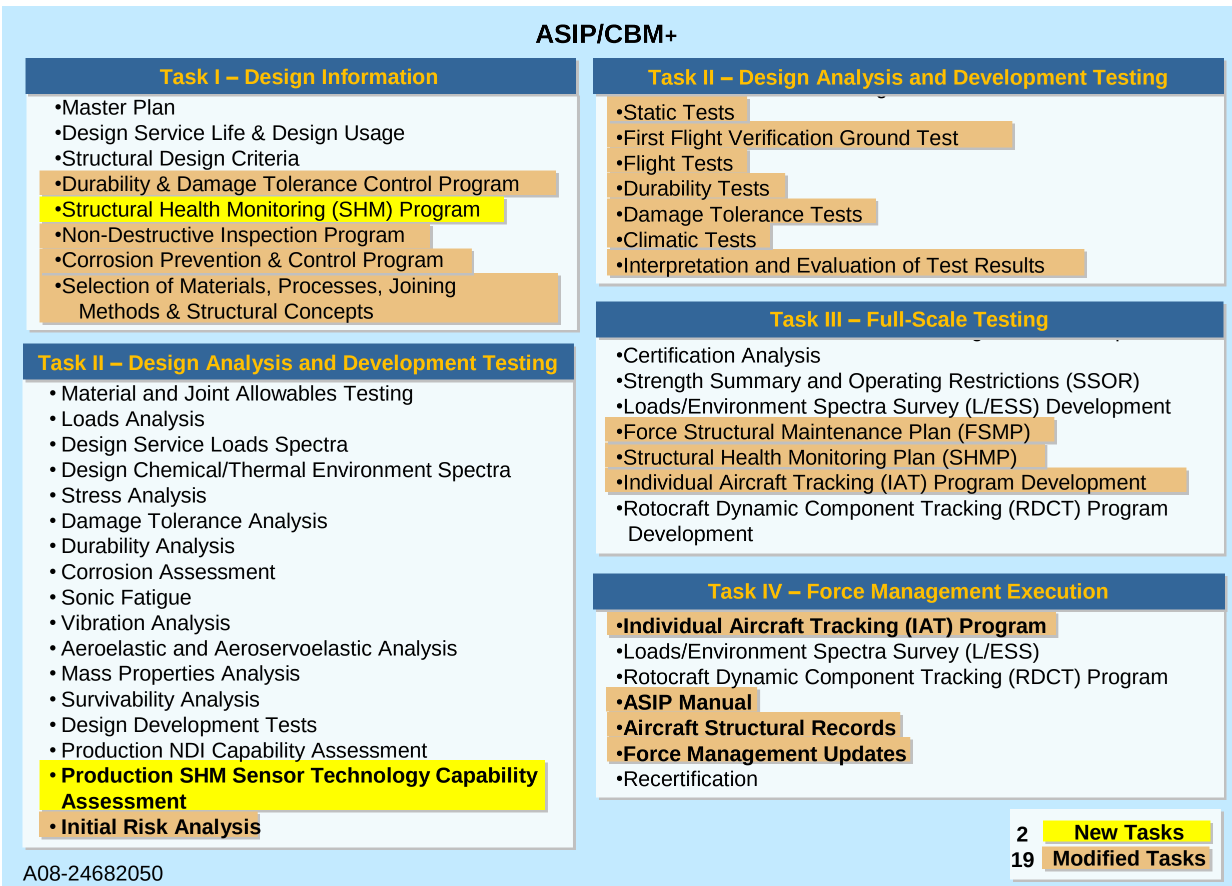


Figure 3. Modified and New ASIP Tasks

This task will be addressed in more detail in section 3.1.

2.3.2 Task 2 - Establish Maintenance ConOps and Requirements

This task identified and documents desired maintenance characteristics and preferred Maintenance Concepts of Operations consistent with the ASIP/CBM+ strategy identified in Task 1.0. This task addressed scheduled verses unscheduled maintenance, Organizational Level verses

Depot Level maintenance actions, Non Destructive Inspection (NDI) activities and identified potential new maintenance skills and training requirements.

This task will be addressed in more detail in section 3.2.

2.3.3 Task 3 - Establish ASIP/CBM+ Availability, O&S, and Maintenance Man Hours Models

This task assessed current legacy aircraft metrics and models to support identification and development of key metrics and methods for evaluating candidate platforms for structural applications to be selected in Task 5.0, and established the baselines and methods for assessing and quantifying benefits of specific structural improvements.
There is a wide array of potentially suitable metrics available in use by the various services and by specific programs within each service. The three major categories of metrics considered are Availability, Mission Capability and Maintenance Cost. Of these three Availability and Maintenance Cost are the most significant and common. Additionally both are applicable when assessing implementation of CBM+ at the Organizational Level and at the Depot level.

This task will be addressed in more detail in section 3.3.

2.3.4 Task 4 - Develop ASIP/CBM+ Information Architecture

This task defines information architectural features and requirements necessary to support CBM+SI in general and the application prototype in particular. This effort includes the identification of technologies, communication/network topologies, data flows and processing node requirements and over all Information Architecture requirements.

This task will be addressed in more detail in section 3.4.

2.3.5 Task 5 - Emerging Technology Review

In this task, we investigated both existing and emerging technologies for sensors, sensor data fusion, sensor integrity, information architecture, and analysis tools as well as prognostics algorithms. This included technologies such as Meandering Winding Magnetometer (MWM) sensor arrays from JENTEK Sensors Inc. and SMART LAYER from Acellent. Each technology was evaluated in terms of maturity level capacity to support the ASIP/CBM+ strategy.

Five sensor technology vendors were invited to Lockheed Martin Aeronautics Companies facility in Marietta Georgia to review structural health monitoring capabilities and concerns. They were:

> Impact Technologies based on their significant experience in system / architecture design and data fusion,
>
> TRI Austin because of their active ultra-sonic damage detection technology,
>
> Acellent Technologies, Inc. because of their passive and active piezoelectric ultra-sonic damage detection technology,
>
> JENTEK Sensors, Inc. for their MWM array damage detection technology, and

Goodrich for their Comparative Vacuum Monitoring (CVM)

We also invited Professors Jennifer and Thomas Michaels of Georgia Tech in to visit and discuss their work and our project.

This task will be addressed in more detail in section 3.5.

2.3.6 Task 6 - Candidate Identification, Evaluation and Selection

This task involved the selection of both the structural application and the SHM technologies to be used for the CBM+SI demonstration.

The structural application candidate needed to be one for which the inspection and maintenance burden is currently high, and for which that burden could be significantly reduced with implementation of SHM and CBM+. Estimates of the projected maintenance relief for each candidate took into account the overall maintenance planning for the weapon system (required scheduled maintenance performed concurrently may dilute or eliminate net the benefit).

The CBM+SI demonstration candidates were drawn from Lockheed Martin's family of platforms and included:

(1) C-130 Center Wing Box (CWB) Rainbow Fitting,
(2) C-5 Fuselage Aft Crown,
(3) F-16 wing skin/wing attach fitting joints, and
(4) F-22 side-of-body lugs.

SHM technologies and systems which were specifically applicable to the identified structural application candidate(s) were assessed, and associated risks as well as potential benefits were be evaluated.

General criteria for the selection of applicable technology included:

Availability of sensor hardware,
Validation of performance,
Long-term durability & robustness,
Cost,
Footprint,
Impact on structure,
Data processing and storage requirements, and
Power requirements

This task will be addressed in more detail in section 3.6.

2.3.7 Task 7 - Develop Structural Application Prototype

Lockheed demonstrated a CBM+SI strategy for the selected structural element and SHM technology selected in Task 6, on the C-5A weapons platform.

Structural items removed from a C-5A fuselage in Lockheed's bone yard were used for the prototype SHM system demonstration.

This task will be addressed in more detail in section 3.7.

2.3.8 Task 8 Determine Benefits, Risks and Establish the Business Case

After evaluation of the prototype was completed a business case was developed. The benefits in terms of (1) increase in force readiness, (2) decrease in cost of ownership, and (3) reduction in maintenance hours were identified and related to costs associated with implementation. Additionally, risks were identified and assessed.

This task will be addressed in more detail in section 3.8.

2.3.9 Task 9 Transition Road Map and Strategy

This task involved developing a roadmap for transition of the demonstrated CBM+SI strategy to fleet-wide application.

This task will be addressed in more detail in section 3.9.

3. TASKS

3.1 Task 1 - ASIP and CBM+ Integration Strategy

As discussed above, due to the seemingly uncontrolled escalation of O&S costs, the US Department of Defense (DoD) has been investing heavily in development of the technologies, processes and procedures necessary to implement CBM+ policy for military departments and defense agencies [3.1]. The DoD has issued an instruction for the incorporation of CBM+ into contracts to commercial maintenance operations for weapon systems, equipment, and materiel throughout all life-cycle phases [3.2]. At the same time, however, the maintenance philosophy (for USAF aircraft) is defined largely in the USAF Aircraft Structural Integrity Program (ASIP) standard, MIL-STD-1530C [3.3]. This standard defines the design, test, and operational requirements necessary to achieve and maintain the structural integrity of USAF aircraft. The ASIP in its current form seeks to manage cost and schedule risk through a series of disciplined, time-phased tasks. The objective of the current program was to develop a strategy for the integration of CBM+ concepts and policies with the USAF ASIP in a way that allows applicability for both new and existing (legacy) platforms. This strategy was articulated in two ways. First, a proposed modification to the ASIP standard was developed. Second, a systems engineering approach was used to define a CBM+ implementation process flowchart.

3.1.1 Modification to ASIP Standard

Task 1 involved the development of a conceptual, modified aircraft structural integrity program (ASIP), which is applicable for both existing and new platforms, and that explicitly addresses CBM+ and the development and validation of the technologies required to support it (SHM, prognostics, etc.). The modified plan is presented in the form of a proposed revision to the current USAF ASIP standard; MIL-STD-1530C. A brief summary of the modified plan is given here, and the complete document is provided as attachment A. (In the attachment, modifications to the standard are shown in blue.) As shown in Figure 4, the revision affects all five “pillars” of ASIP; there are 21 modified subtasks and 2 new subtasks.

At the highest level, the development of a strategy for the integration of CBM+ with ASIP consists of matching an NDI and/or an SHM technology to a structural sustainment issue, developing prognostic capability for that issue, establishing a concept of operations for the installation, operation and maintenance of the inspection system and establishing the rules for maintenance action decision making. Note that multiple strategies can exist.

Task I	Task II	Task III	Task IV	Task V
Design Information	Design Analyses & Development Tests	Full-Scale Testing	Certification & Force Management Development	Force Management Execution
5.1.1 ASIP Master Plan	5.2.1 Materials & Joint Allowables	5.3.1 Static Tests	5.4.1 Certification Analysis	5.5.1 Individual Aircraft Tracking Program
5.1.2 Design Service Life & Design Usage	5.2.2 Loads Analysis	5.3.2 First Flight Verification Ground Tests	5.4.2 Strength Summary & Operating Restrictions	5.5.2 Rotorcraft Dynamic Component Tracking Program
5.1.3 Structural Design Criteria	5.2.3 Design Service Loads Spectra	5.3.3 Flight Tests	5.4.3 Force Structural Maintenance Plan	5.5.3 Loads / Environment Spectra Survey
5.1.4 Durability & Damage Tolerance Control Program	5.2.4 Design Chemical, Thermal & Environment Spectra	5.3.4 Durability Tests	5.4.4 Loads / Environment Spectra Survey Development	5.5.4 ASIP Master Manual
5.1.5 Corrosion Prevention & Control Program	5.2.5 Stress Analysis	5.3.5 Damage Tolerance Tests	5.4.5 Individual Aircraft Tracking Program Development	5.5.5 Aircraft Structural Reports
5.1.6 Non-Destructive Inspection Program	5.2.6 Damage Tolerance Analysis	5.3.6 Climatic Tests	5.4.6 Rotorcraft Dynamic Component Tracking Program Development	5.5.6 Force Management Updates
5.1.7 Structural Health Monitoring Program	5.2.7 Durability Analysis	5.3.7 Interpretation & Evaluation of Test Results		5.5.7 Recertification
5.1.8 Materials, Processes, Joining Methods, & Structural Concepts Selection	5.2.8 Corrosion Assessment			
	5.2.9 Sonic Fatigue Analysis			
	5.2.10 Vibration Analysis			
	5.2.11 Aeroelastic & Aeroservoelastic Analysis			
	5.2.12 Mass Properties Analysis			
	5.2.13 Survivability Analysis			
	5.2.14 Design Development Tests			
	5.2.15 Production NDI Capability Assessment			
	5.2.16 Production SHM Capability Assessment			
	5.2.17 Initial Risk Analysis			

modified task

new task

Figure 4. Modified and New ASIP Tasks

A modified Force Structural Maintenance Plan (FSMP) is the mechanism for the implementation of each proposed CBM+SI strategy. It will address the fundamental change in maintenance planning that can be achieved through the integration of SHM-informed diagnostics and near real-time prognostics. The revised FSMP will attempt to optimize maintenance planning so as to minimize unscheduled repairs and eliminate unnecessary inspections, thus increasing the number of assets in operational status.

3.1.1.1 ASIP TASK I

For the development of new aircraft, planning for CBM+ must begin during the design information phase (ASIP TASK I) of the program. If CBM+ is being implemented for a legacy system, then there are elements of ASIP TASK I that must be re-executed. First and foremost is the development and / or modification of the ASIP master plan (5.1.1). This plan specifies the technical approach and scheduling for the tasks required to design, develop, certify and operate the aircraft. It must be developed and/or modified to address the development, certification and utilization of any technologies required to enable new or modified force management practices. For example, if SHM enabled CBM+ is going to be used for a particular structural application on a legacy aircraft, then the ASIP master plan for that aircraft must be updated to reflect the planning for the development of the required SHM technologies (sensor reliability and durability), SHM system validation, and systems integration, architecture and operation concepts.

The implementation of CBM+ may require modification of the Durability and Damage Tolerance (DaDT) control program (5.1.4). One of the major purposes of this program is to specify the criteria for the classification of parts (part criticality). The introduction of condition monitoring may permit changes in part classification, and the conditions under which this would be allowed would have to be specified in the DaDT control program. Any changes in fracture control policy brought about by the use of CBM+ would have to be defined here. For example if SHM were to be employed for a specific structural component, then the SHM reliability and false call rates used for the damage tolerance analysis would be stated here. These data would be based on validated SHM reliability/capability for each structural application as a function of time/usage. This is analogous to the specification of initial flaw sizes for damage tolerance analysis based demonstrated NDI capability.

Corrosion detection is an area in which condition monitoring can be highly effective. This is especially true in applications for which usage of embedded sensors can eliminate the necessity for very expensive and time consuming disassembly and reassembly of structures (for example, structures with Low Observable coatings). The Corrosion Prevention and Control Program (5.1.5 of MIL-STD-1530C) must be modified as required to address the development, validation and usage of corrosion sensors.

If the maintenance planning for all or any part of the aircraft structure will be informed by SHM, then a SHM Program (5.1.7) shall be established for the aircraft structure. This program shall identify and define all of the tasks necessary for the development, test, qualification, and deployment of a SHM system. System qualification shall include demonstration of capability (in terms of Probability of Detection (POD) and false call rates for specified damage types) as well as demonstration of durability (in terms of sensor life and maintenance requirements). The purpose of the SHM system will be to ensure compliance with the durability and damage tolerance requirements and to enable both condition-based and inspection-based force structural maintenance planning. The SHM Program will address all phases of the aircraft program (design, engineering development, production, and in-service operation). The program shall establish a Structural Health Monitoring Requirements Review Board (SHMRRB) responsible for oversight and execution of the program. The SHMRRB shall be formed early in the design phase to review and assess material selections, structural design concepts for compatibility with SHM sensor capability and requirements and SHM system architecture and requirements. The SHMRRB shall also be responsible for review and approval of SHM sensor selection and capability assumptions implemented in the Force Structural Maintenance Plan. The board's decisions will be subject to USAF approval.

In some cases, CBM+ may be facilitated by judicious selection of materials, processing, joining and structural concepts (5.1.8). For example, if SHM is going to be used for a particular structural application, the material may tailored to maximize the sensor effectiveness (for example, modify selection criteria to include magnetic permeability) or the structural configuration may be modified to accommodate sensors and/or supporting wiring.

3.1.1.2 ASIP TASK II

For both new and legacy aircraft, the technologies required to support CBM+ implementation must be validated at the building block level during the development testing phase (ASIP TASK II). If the maintenance planning for all or any part of the aircraft structure will be informed by SHM, then the capability of the SHM system shall be established (5.2.16) to mitigate the risk of missing defects. Special emphasis shall be given to systems used to monitor fracture- and mission-critical parts. Capability / reliability demonstration of production SHM processes shall be addressed within the SHM Program. Both shall be demonstrated as a function of time / usage. Capability / reliability shall be quantified in terms of metrics (POD, false call rates, sensor failure rates, etc.) suitable for formal risk analyses. As determined practical and feasible by the SHMRRB, component level SHM system capability tests shall be conducted to mitigate the risk that full-scale testing does not yield measurable fatigue, impact, corrosion or other damage in monitored regions. In addition, component level durability tests shall be conducted in order to establish sensor and system durability.

The initial risk analysis (5.2.17) shall be modified as required to address the reliability (POD) and false call characteristics of each condition monitoring technology employed in the CBM+ concept. All significant variables impacting risk shall be included in the risk analysis. Examples of such variables include: EIFS distribution, load spectra, chemical and thermal environment, material properties, the NDI probability of detection (POD) and the SHM sensor and system probability of detection (POD).

3.1.1.3 ASIP TASK III

Full system validation of any technologies employed to enable CBM+ will occur during the ground and flight tests of ASIP TASK III. For new aircraft, this can occur during the initial static (5.3.1), flight (5.3.3) and durability tests (5.3.4, 5.3.5 and 5.3.6). For legacy aircraft, this may occur during updated (late block) flight and/or ground tests. For CBM+ concepts that are enabled by enhanced NDI or active SHM systems that are operated on the ground only and that do not require integration with any of the aircraft systems (power, data bus, etc.), it will not be necessary to flight qualify those systems. On the other hand, systems that will be operated in flight will require flight qualification (5.3.3). For new programs, such qualification will take place on the flight test aircraft; for legacy programs a test aircraft will be identified from the operational fleet. These tests shall include dynamic response, flutter, and aeroacoustic and vibration tests, as well as a flight and ground loads survey. These tests will serve to flight qualify any SHM equipment which is a part of the production configuration of the aircraft.

One of the major elements of ASIP TASK III is the full-scale durability test. The objectives of this test include 1) the demonstration that the economic life of the test article is equal to or greater than the design service life by the specified margin, 2) the identification of critical areas of the aircraft structure not previously identified by analysis or component testing, 3) the provision of a basis for special inspection and modification requirements for force aircraft and, 4) the demonstration of the capability of the SHM system to operate successfully under design loading / environmental conditions for the design service life. If the maintenance planning for all or any part of the aircraft structure will be informed by SHM, then a SHM system demonstration / validation shall be conducted as an integral part of the full-scale aircraft structure durability test. The objectives of the program shall be to: 1) demonstrate that the SHM system can detect damage at stages that are early enough to prevent catastrophic failure and permit remedial maintenance actions, 2) demonstrate that false positive rates remain below acceptable levels, 3) demonstrate that all non-field serviceable SHM equipment remains functional for the design service life of the aircraft and that serviceable equipment (connectors, energy sources, etc.) remain functional for a period deemed acceptable by the SHMRRB. In the event that the full scale durability test does not yield measurable damage or critical flaws, the SHM capability demonstration may rely on component level durability tests results as defined in ASIP TASK II (5.2.16).

3.1.1.4 ASIP TASK IV

CBM+ integration has a significant impact on ASIP TASK IV for both new and legacy aircraft. The force structural maintenance plan (FSMP) will be the primary mechanism for the implementation of CBM+SI. The modified FSMP will describe the inspection-based (recurring) structural maintenance program and the condition-based maintenance program, as well as the interaction between the two and the responsibilities of each (i.e., periodic, minor and major inspections, program depot maintenance (PDM), CBM scheduling, the CPCP, etc.). It will attempt to optimize maintenance planning so as to minimize unscheduled repairs and eliminate unnecessary inspections, thus increasing the number of assets in operational status.

The FSMP defines when, where, how, and the estimated costs of inspections and modifications. It also describes the inspection-based (recurring) structural maintenance program (i.e., periodic, minor and major inspections, program depot maintenance (PDM), the CPCP, etc.). If the maintenance planning for all or any part of the aircraft structure will be condition-based (i.e. CBM+), then the operation/scheduling of the structural health monitoring system shall be described. Furthermore, when both inspection-based and condition-based maintenance are used to manage the structure, the interaction between the two and the responsibilities of each shall be described. It is intended that the FSMP will be used to establish budgetary planning, force structure planning, and maintenance planning. The initial FSMP will generally be based on the design loads/environment spectrum and shall be updated when the data from the Loads/Environment Spectra Survey (L/ESS) becomes available and a new baseline operational spectrum is developed. Additional updates will be required when any of the following occur: 1) there are significant changes in operational usage, 2) unanticipated damage is discovered by the structural health monitoring system during operational usage, 3) unanticipated damage is discovered during scheduled inspections, 4) unanticipated damage is discovered during surveillance sampling inspections conducted using the Analytical Condition Inspection (ACI) Program, 5) unanticipated damage is discovered during structural teardown inspection programs, and/or 6) unanticipated damage is discovered during normal operational maintenance of the aircraft.

Note that both NDI and SHM are enablers for CBM+. Figure 5 illustrates the relationship between these technologies and indicates that when they are coupled with prognostics they form CBM+, which in turn is implemented via the FSMP.

Implicit in damage-tolerant structural designs are inspection requirements intended to ensure damage never reaches the sizes that can cause catastrophic failures. Inspections are required initially and at the repeat intervals determined by damage tolerance analysis. Such inspections shall continue to the estimated time, with the appropriate scatter factor, of the onset of Widespread Fatigue Damage (WFD). A significant benefit that could be affected by the usage of SHM could come in the form of reduced risk for the undetected onset of WFD, Multi-Element Damage (MED) specifically. SHM could be used not only to detect damage in multiple, functionally related, structural elements, but it could also be used to detect the load redistribution that accompanies such damage.

The aircraft SHM system is intended to ensure that damage in monitored areas is discovered early enough to allow scheduling of maintenance actions and that it never reaches the sizes that can cause catastrophic failures. Development of the SHM system shall consider material, geometry, accessibility, sensor POD and the resulting system level POD. The initial development of, as well as any subsequent modification to the SHM system will require approval by the SHMSRRB. There are two modes for SHM system operation: 1) continuous, in-flight monitoring, 2) on-ground operation at regular intervals or after specified flight events. It is anticipated that the vast majority of SHM systems will be operated on-ground, which means that the highest possible frequency of operation would be once per flight. Less frequent intervals of operation shall be determined as follows.

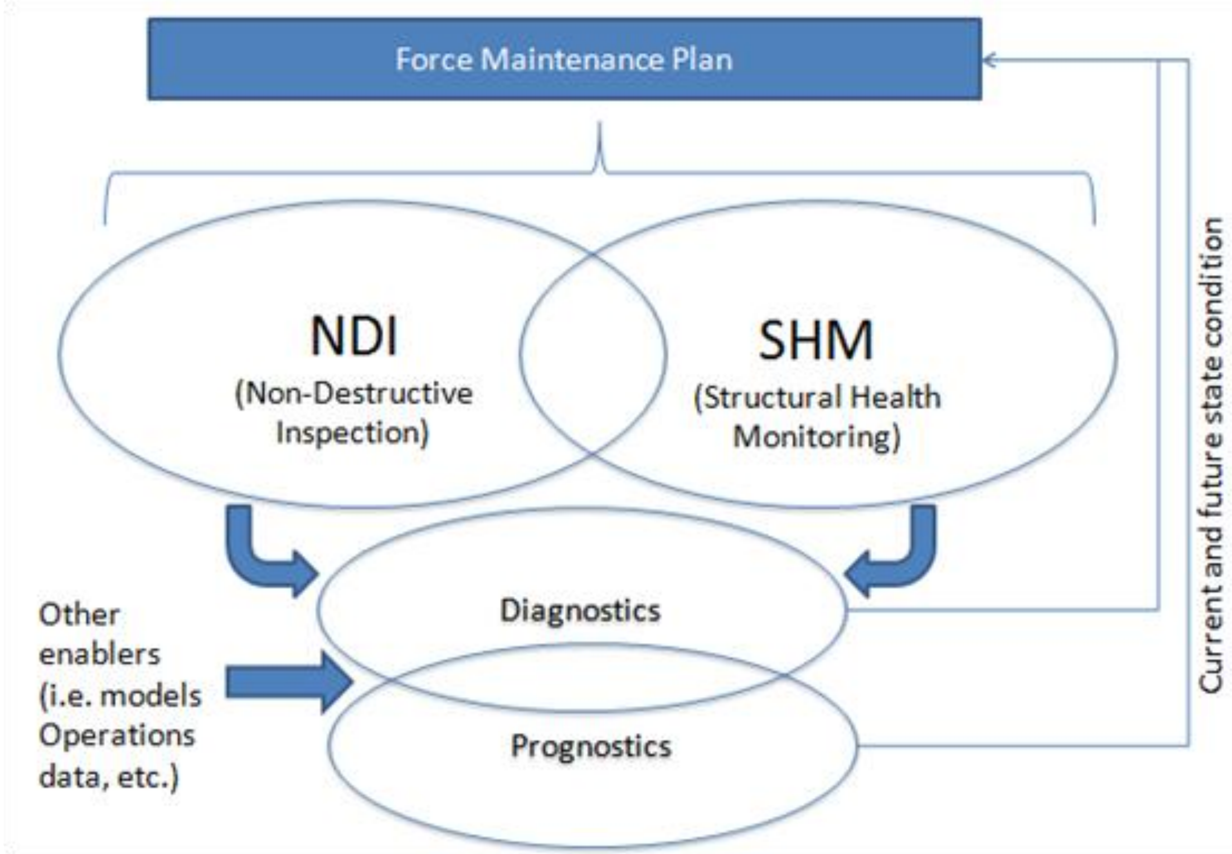


Figure 5. Relationship between SHM and other components of CBM+

SHM system operation intervals:

The criteria for establishing the frequency of SHM system operation shall be as follows:

a. The SHM system operation interval for fail-safe design concepts shall be less than or equal to one-half the life as determined by either: 1) fatigue analyses and tests with an appropriate scatter factor, or 2) slow damage growth analyses and tests assuming an appropriate initial flaw size. The required initial flaw size shall be based on the demonstrated flaw size detection capability of the SHM system. (The data obtained from the design development testing described in 5.2.16 shall be used for verification of the flaw size detection capability.)

b. The SHM system operation interval for slow damage growth design concepts shall be less than or equal to one-half the life from the assumed maximum probable initial flaw size to the critical flaw size. The required initial flaw size shall be based on the demonstrated flaw size detection capability of the SHM system. (The data obtained from the design development testing described in 5.2.16 shall be used for verification of the flaw size detection capability.)

c. The risk analysis of 5.2.16 and 5.5.6.3 should be used to determine if a reduction in the operation intervals are required to control the safety risk to an acceptable level or to reduce economic or availability consequences associated with damage repair.
SHM system operation for anomalous events:

> If the frequency of SHM system operation is less than once per flight, then a procedure shall be established whereby an anomalous usage event will trigger an SHM interrogation after the current flight. A list of anomalous usage events (overload, hard landing, ballistic damage, etc.) shall be developed and continuously updated.

The Individual Aircraft Tracking IAT program will be modified as required for both legacy and new programs. The IAT program is based on actual usage data and is used to adjust maintenance intervals on an individual aircraft ("by tail number") basis. All force aircraft shall have systems that record sufficient usage parameters that can be used to determine the damage growth rates throughout the aircraft structure. The total IAT control point set will include both the inspection-based maintenance (IBM) and the Condition-Based Maintenance (CBM) control point sets. Tracking analysis methods shall be developed which adjust the inspection and modification times based on the measured structural condition (damage state) for CBM control points, and on the actual measured usage of the individual aircraft for all (CBM and IBM) control points. The systems shall have sufficient capacity and reliability to achieve a 90-percent minimum valid data capture rate of all flight data throughout the service life of the aircraft.

3.1.1.5 ASIP TASK V

Finally, the implementation of CBM+ will affect virtually every task in the force management execution phase (ASIP TASK V) of both legacy and new programs. Force management is conducted by executing the FSMP. The maintenance schedule directed by the FSMP shall be adjusted for each aircraft by data received from the IAT Program or by the RDCT system. The FSMP shall be updated periodically to ensure it accurately and efficiently protects against structural failures. Updates to the FSMP shall be based on evaluations of changes in operational usage, major modifications, as well as aircraft inspection and/or structural health monitoring findings documented within the structural maintenance database. These evaluations may be based on analysis and/or testing (up to and including a possible additional full-scale static and/or durability test). Any changes to the force management strategy shall be documented in the ASIP Master Plan.

The IAT Program shall be used to adjust the inspection, modification, overhaul, and replacement times based on the measured condition (damage state) for CBM control points and on the actual, measured usage of the individual aircraft for all (CBM and IBM) control points.

Durability and Damage Tolerance Analysis (DADTA) and IAT Program updates:

> The IAT data, L/ESS data, and the aircraft structural records shall be used to determine when Durability and Damage Tolerance Analysis (DADTA) and IAT

Program updates should be conducted. Variations in the average usage from the analysis baseline and usage variation extremes from the analysis baseline shall be considered when the need for an update is determined. In addition, an update to the DADTA and IAT Program shall be conducted when aircraft damage findings (both from inspections and from structural health monitoring) indicate the accuracy of the analyses is less than expected.

Corrosion assessment updates:

The occurrences of corrosion shall be evaluated periodically with regard to the effectiveness of the preventive procedures (e.g., frequency of wash cycles, coatings, corrosion prevention compounds, etc.) used and, if possible, corrosion findings shall be correlated to the aircraft basing and the results of Task II and Task III environmental testing. The results of these evaluations and any observed trends will be used to develop improved maintenance procedures and adjust the corrosion inspection requirements in the FSMP. The corrosion assessment updates may also be used to define candidate structural applications for the introduction or expansion of a structural health monitoring system. In the event that a structural health monitoring system is introduced, it shal be qualified in accordance with ASIP TASKS I and III.

Structural Health Monitoring (SHM) updates:

The occurrences of damage and the effectiveness of structural health monitoring to manage them shall be evaluated periodically. The results of these evaluations and any observed trends will be used to define candidate structural applications for the introduction or expansion of a structural health monitoring system (for both hot spot and wide area applications), and to develop improved maintenance procedures and adjust the inspection requirements in the FSMP. In the event that a structural health monitoring system is introduced, it shall be qualified in accordance with ASIP TASKS I, II and III.

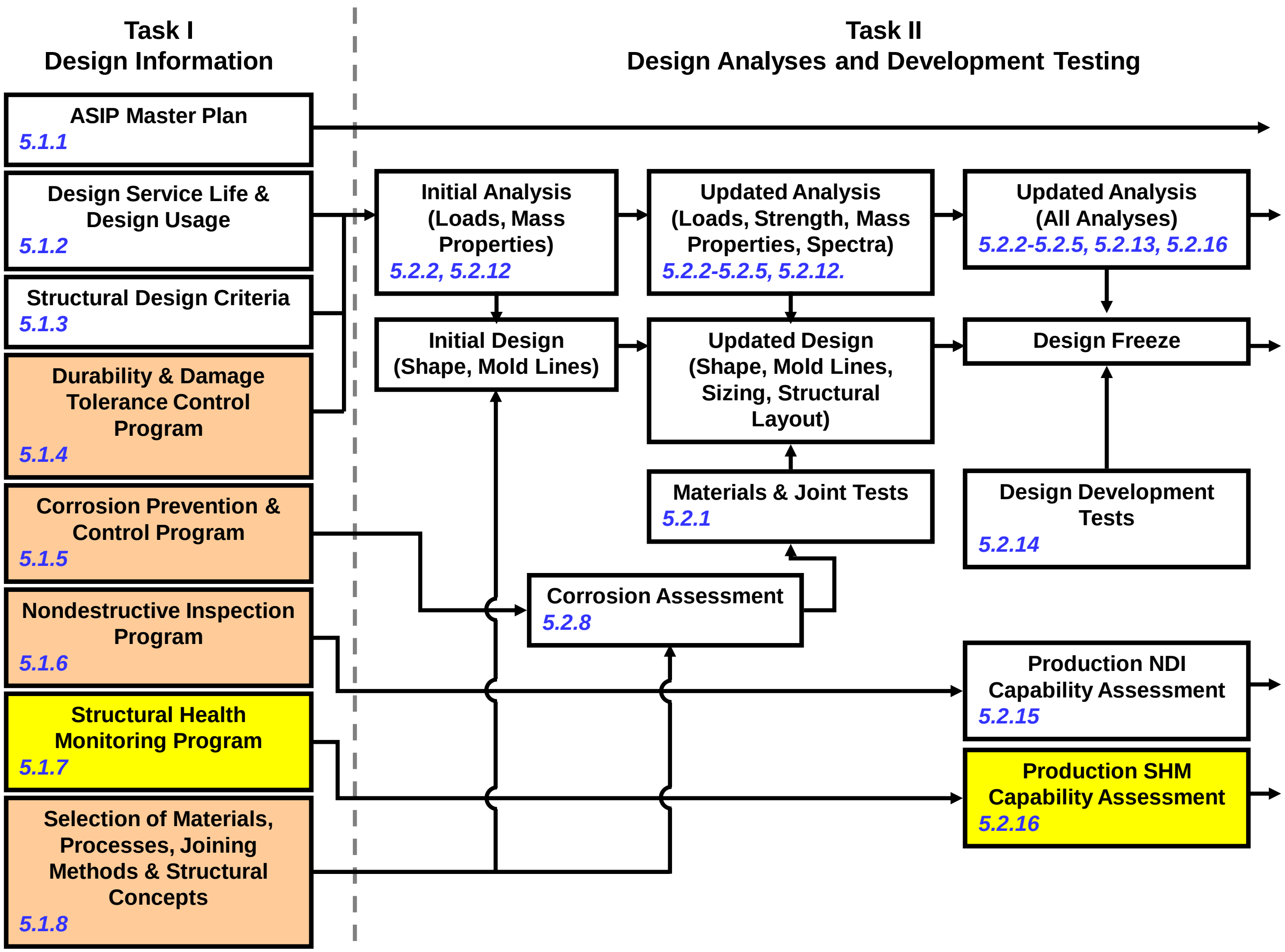


Figure 6. Recommended modifications and additions to ASIP Tasks I and II

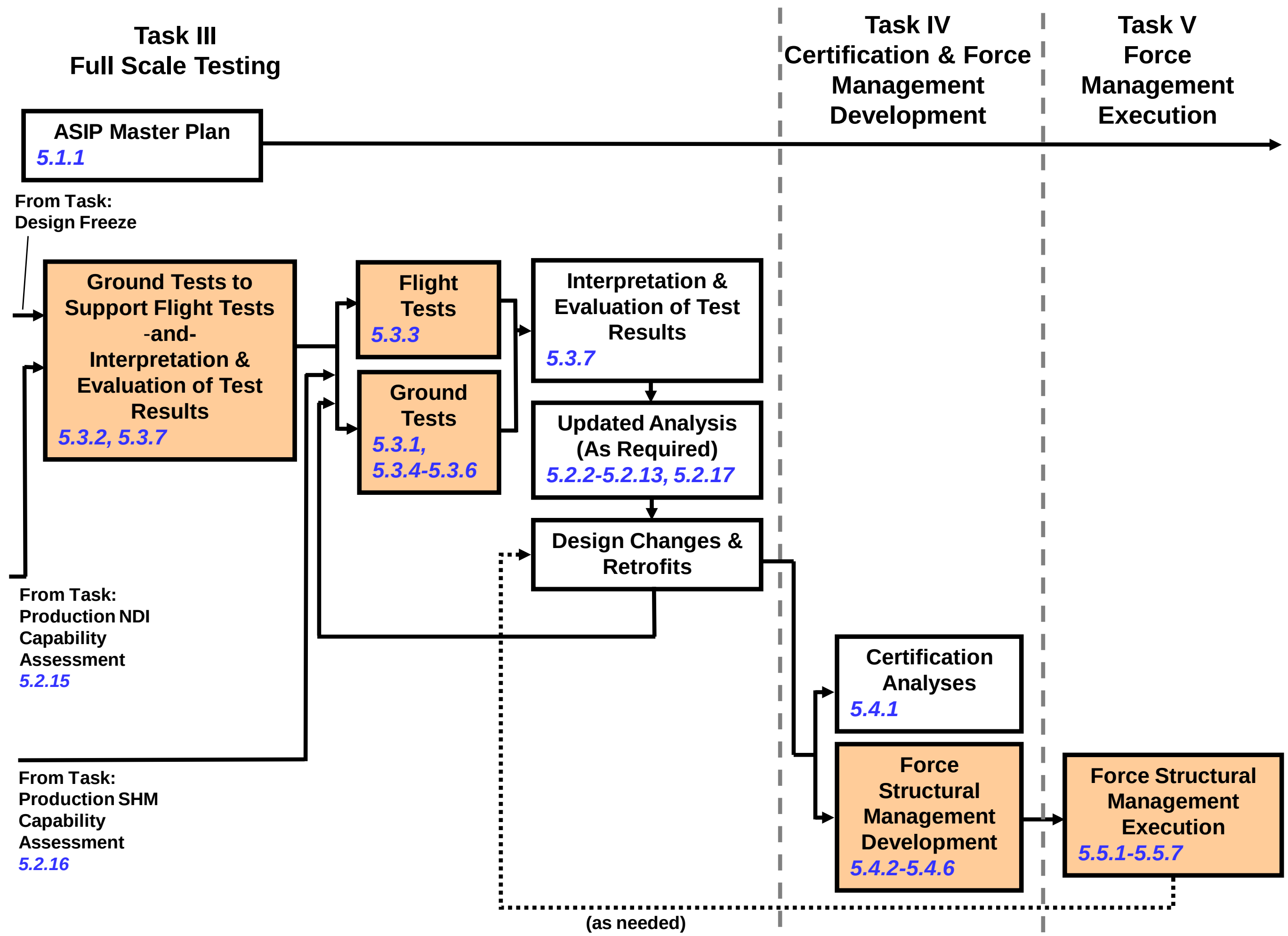


Figure 7. Recommended modifications and additions to ASIP Tasks III, IV and VCBM+ Implementation Process

The implementation of CBM+ for the management of an operating structure requires the design, development, test and integration of a wide range of technologies. These technologies range from sensor development and SHM system reliability determination to data fusion and data management, to physics-based prognostic model development, to accurate cost modeling for both current and projected operations and maintenance practices. In fact, the implementation is best accomplished using a systems engineering approach in which requirements are clearly defined and products are evaluated against those requirements. In the current program, the systems engineering framework proposed by Derriso and Haugse [3.4] for SHM system development was embedded within a larger framework for CBM+ implementation by adding a CBM+ requirements phase and by formally including cost as the final criterion by which designs are accepted or rejected.

As shown in Figure 8, the first task that must be accomplished is the clear definition of a potential structural application. This requires a thorough understanding of the structural sustainment issue, both from an engineering perspective (i.e. structural configuration, materials, loads, etc.) and from an operational perspective (i.e. current maintenance practice, intervals, repairs etc.)

The second requirement that must be met at the outset is the accurate definition of the baseline costs for the candidate application. These costs must include all recurring and non-recurring costs for the current inspection, maintenance and repair protocols (i.e. the cost of not implementing CBM+).

Next, the concept of operations for the CBM+ implementation must be defined. This step essentially sets the requirements for CBM+ for the candidate application. The CBM+ ConOps will prescribe the type(s), fusion, transmission and storage of structural condition data. It will specify the manner in which SHM system installation, maintenance and operation will be carried out, and it will establish the rules for maintenance action decision making.

Given the CBM+ ConOps, it will then be possible to define SHM system requirements. These requirements may or may not be based on a specific sensor technology. They will, however, address the presence (or absence) of existing IVHM infrastructure, available power, footprint (volume) limitations, data bus, data storage requirements, etc.

At this point, it will be possible to design an SHM system. This will include matching a sensor technology to the structural sustainment issue and generally designing within the SHM system requirements. When the design is complete, the cost of the CBM+ implementation will be estimated. This will include the SHM system development, installation and maintenance costs, as well as any materiel and maintenance man-hour costs and possibly asset non-availability penalties. At the same time, the system reliability will be quantified, both by analysis (MAPOD) and by test. This in turn will be used to calculate the single flight probability of failure (SFPoF).

The calculated Single Flight Probability of Fracture (SFPoF) will be compared against the flight safety requirements for the application in question. If the SFPoF falls below the requirement, then the CBM+ implementation costs will be compared against the baseline, and if the cost ratio is favorable and if the breakeven period is acceptable, then the management of the structure can be transitioned to CBM+. If the SFPoF is too high, then the decision must be made as to whether or the SHM system design can be revised within the existing constraints. This decision is based both on technical feasibility and cost constraints; each time the SHM system is re-designed and evaluated, the cost of doing so is added to the CBM+ implementation costs. If the SHM system does not meet the structural safety requirements, and if the decision is made that iteration of the design is not technically feasible and/or cost effective, then the decision must be made as to whether a viable system might be possible with a different CBM+ ConOps. If that decision is positive, then the entire process is repeated from the CBM+ ConOps development stage.

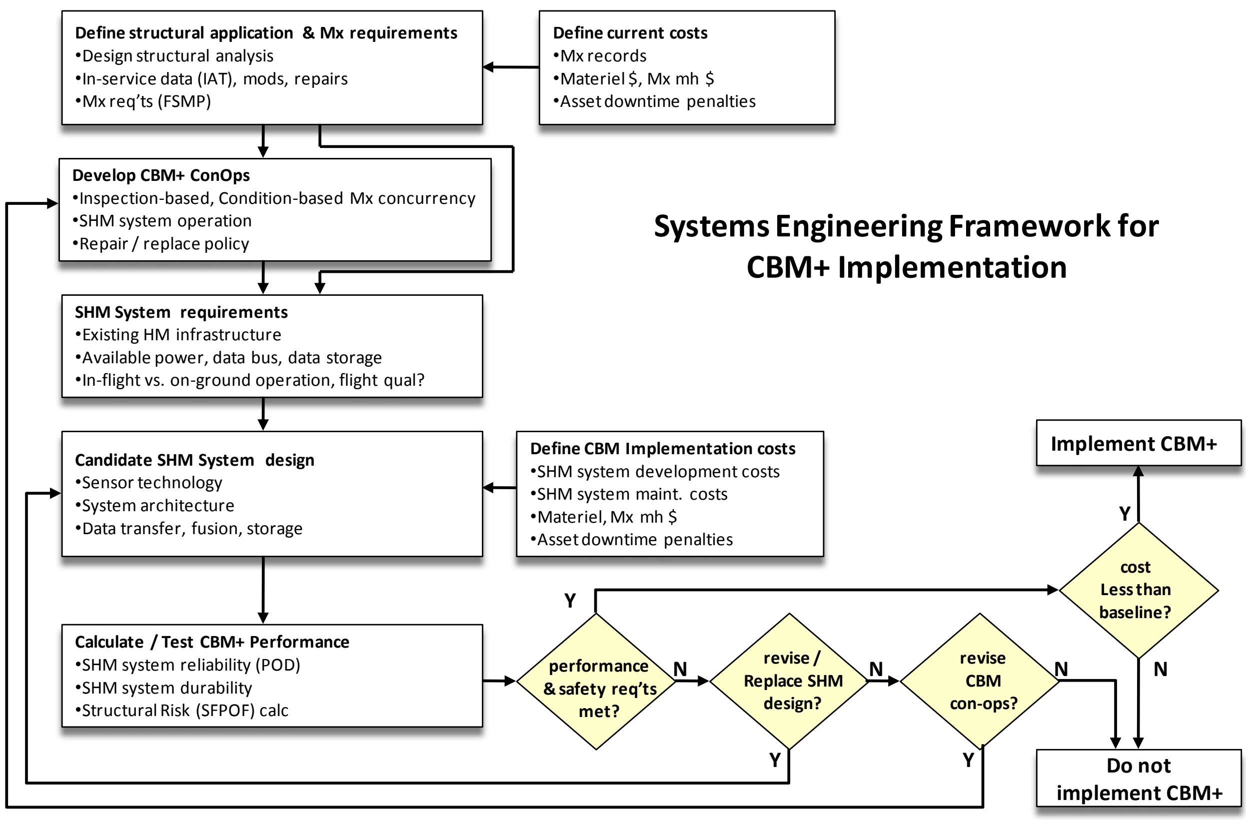


Figure 8. Systems Engineering Framework for CBM+ Implementation

3.2 Task 2 - Establish Mx ConOps and Requirements

This task identifies and documents desired maintenance characteristics and preferred Maintenance Concepts of Operations consistent with the ASIP/CBM+ strategy identified in Task 1.0. This task includes scheduled verses unscheduled maintenance, Organizational Level verses Depot Level maintenance actions, Non Destructive Inspection (NDI) activities and will identify potential new maintenance skills and training requirements.

3.2.1 General CBM+SI Mx ConOps and Requirements

3.2.1.1 Desired Maintenance Characteristics

The ultimate objectives of CBM+SI are to increase aircraft availability and reduce support costs. System maintenance characteristics consistent with those objectives are at a high level reductions in maintenance frequency, reductions in the duration (span) of maintenance actions, and reductions in resources needed to perform maintenance.

3.2.1.2 Preferred Concept of Operations

For effective CBM+SI there is no one preferred Concept of Operation. The type of platform (e.g. fighter, transport), the operational mission, current maintenance concepts, as well as specifics associated with the structural element of interest all are factors that must be considered. However since the objective of CBM+SI is to reduce the overall maintenance burden associated with structural inspection activities, a model concept is to introduce cost effective sensing and automation on the platform and in the support system that will result in the reduction or elimination of scheduled inspections, particularly at the Organizational Level.

3.2.1.3 Scheduled verses Unscheduled Maintenance

CBM+ at a high level is an approach that allows maintenance to become more proactive and less reactive. For aircraft systems, this involves not only having enhanced knowledge of the current health of system components, but also knowledge of the component's Remaining Useful Life (RUL). Knowing the RUL of a component gives maintenance and logistics the ability to anticipate needed maintenance and plan for the maintenance action. The planning that may occur can include ordering parts and materials before the aircraft component has actually failed, scheduling the maintenance to be performed concurrent with other planned maintenance activities, collecting all of the needed tools and equipment , and ensuring that the needed personnel will be available to support the maintenance activity.

In contrast to the proactive approach described previously, the reactive operate-to-failure concept, which is the current concept used for most aircraft components, results in high intensity ad hoc maintenance activities that are performed to support the operational tempo of the aircraft, which for fighter aircraft is intense. The operational objective is to generate sorties. When an aircraft returns from a mission the only advanced identification of aircraft status is radio communication with the crew prior to landing. If there are no discrepancies, then upon landing the aircraft can be serviced and can fly another sortie. If there are system discrepancies, then following a maintenance debrief the maintenance can be started. The needed equipment and personnel must be gathered up and dispatched to the aircraft. Delays due to the quantity of available equipment and needed personnel often result in increases in to overall maintenance time. After troubleshooting, parts will be ordered introducing further delays in restoring the aircraft back to an operational status. This reactive maintenance scenario is far less effective than the proactive scenario and results in reduced aircraft availability and potentially higher maintenance costs.

For CBM+SI, due to the relatively long time between structural failures as compared to many other system components, a proactive approach has benefits but it is does not directly affect sortie generation rate. Application of CBM+SI can, however, reduce the inspection burden at the organizational level and support implementation of High Velocity Maintenance (HVM) by providing data to assess the current material condition to support compressed Programmed Depot Maintenance spans.
HVM is a maintenance strategy and approach that moves aircraft through a depot faster by increasing the man-hours per day. Reducing the inspection burdens at the Organizational Level and reducing the Programmed Depot Maintenance (PDM) span both contribute to increased aircraft availability.

3.2.1.4 Organizational verses Depot Actions

Ideally a CBM+SI concept should strive to eliminate scheduled inspections at the organizational level. CBM+SI fully implement on an aircraft would provide status to the maintainer of the health of the structure and indicate specific areas where damage has been detected. The maintainer can then confirm the reported damage by performing a visual inspection of the area and performing repairs as required. Note that the inspection is an "On-Condition" task, not a scheduled task. The CBM+SI system could be triggered automatically or manually depending on the specific application. In either case the CBM+SI system could capture data representing structural health data that would be of use to the depot, in addition to data needed by the organizational level. This data for the depot could identify defects that are below the size that would trigger an organizational level, but significant to depot. This information would allow the depot to estimate and plan for structural repairs that will be required during the next PDM.

3.2.1.5 Non-Destructive Inspection (NDI) Activities

Ideally a CBM+SI concept should strive to eliminate the need for traditional NDI activities and equipment at the organizational level. When the F-22 Program established a two level maintenance concept, the Intermediate Level was effectively eliminated. For fighter aircraft prior to the F-22 (e.g. F-15, F-16), the Intermediate level maintenance infrastructure was large, costly and resulted in a large logistics footprint during deployment. The reality was that the intermediate level capability was not completely eliminated; it was effectively embedded into the F-22 aircraft. Rather than using a federated architecture for avionics and electrical subsystem, the F-22 implemented an integrated architecture consisting of line replaceable modules rather than boxes. The integrated avionics architecture made the two level maintenance concept possible.

With the application of SHM supporting CBM+SI, the NDI equipment/capability can be effectively moved into the aircraft just as the system test and fault detection and isolation capabilities for Avionics were moved onto the aircraft. Having SHM embedded into the aircraft will eliminate, or reduce, the need for inspections to determine the state of the structure, the embedded SHM capability will detect, localize and report structural problems just as the embedded fault detection and isolation capability does for Avionics. Unlike at the organizational level CBM+SI and SHM will not impact, at least for some time, the need for and use of NDI at the Depot level.

3.2.1.6 New Maintenance Skills and Training Requirements

The application of CBM+SI may require some new maintenance skills and training requirements at both the organizational and depot levels. The equipment on and off of the aircraft (e.g. sensors, data multiplexers and signal processing) will require maintenance due to inherent failures and induced damage. This will require training and skills specific to the applied technology. Also some theory of operation training will be required.

3.2.2 C5A Aft Crown Specific CBM+SI Mx ConOps and Requirements

3.2.2.1 Desired Maintenance Characteristics

For the C-5A Aft Crown CBM+SI application as with the general case, the objectives are increased aircraft availability and reduce support costs. For the C-5A Aft Crown application the specific objectives are: 1) eliminate the visual inspection that is part of the 120 day Home Station Check (HSC), and 2) eliminate the 32 month Magnetic Optical Inspection (MOI) Inspection.

To provide a cost effective alternative to the baseline approach, the CBM+SI approach must allow for inspection of the Aft Crown to be performed in 1 or 2 hours. The data recording and processing must be performed so that the status of both the monitoring system and the Aft Crown is available to the maintainer in 5 minutes. Interrogation of the monitoring system shall not require any disassembly of the aircraft and shall be executed from the passenger compartment.

3.2.2.2 Preferred Concept of Operations

For the C-5A Aft Crown the preferred concept of operation is to have the SHM sensors, wiring and connectors installed without any active electronics as part of the aircraft installation. The maintainer, at some interval, will go to the aircraft carrying a battery powered portable electronics unit (e.g., a laptop). The maintainer will connect the electronics unit to one test connector in the passenger compartment and execute the tests. The test results will be stored and any discrepancies will be presented to the maintainer. The information presented to the maintainer will identify any structural areas that require a follow-up visual inspection or anomalies with the sensor system requiring attention.

The data captured will be stored and managed by aircraft tail number and integrated into the Aging Fleet Integrity & Reliability Management (AFIRM) program which integrates the proven philosophies of the Aircraft Structural Integrity Program (ASIP) and the Functional Systems Integrity Program (FSIP), and strives to increase fleet reliability, safety and mission readiness, while also reducing costs. The AFIRM website consolidates all structures, systems, mission capability, and flight data into an easy to navigate and intuitive online application.

Sensor system elements that have failed or have been damaged will be repaired by replacing individual sensor elements or sensor array elements. If a single sensor element has failed the overall performance of the system may only be slightly degraded. In that case repair can be deferred until later.

The depot will be responsible for the maintenance of the SHM system in terms of repair of any deferred defects from the organizational level and overall system checkout and repair. Any damage to the SHM System as a result of structural or other maintenance will require repair by the depot.

While the depot assumes the burden of the SHM System maintenance, it benefits from the enhanced knowledge of the Aft Crown health based on the data collected and the assessments performed at 120 day intervals. This information can be used to manage the risks associated with the Aft Crown and to support improved planning in support of High Velocity Maintenance.

3.2.2.3 Scheduled verses Unscheduled Maintenance

For the C5A Aft Crown application the baseline scheduled 120 day and 32 month inspection tasks will be replaced by a SHM assessment task that will be performed, ideally, no more frequently than every 120 days. As mentioned previously this assessment task shall take no longer than 2 hours.

3.2.2.4 Organizational verses Depot Actions

Organizational level maintenance will be responsible for the periodic assessments, structural and SHM System repairs that are not deferred to depot. They are also responsible for ensuring that the assessment data is posted to Affirm, as well as the data associated with any structural or SHM system repairs. The 120 day Visual Inspection is effectively replaced by the SHM System facilitated inspection. The benefits of the SHM task over the visual inspection are:

1) The SHM facilitated task can be performed from inside the passenger compartment where as the visual inspection requires the maintainer to get on top of the C-5 aircraft which involves safety precautions and equipment (work stands) and a safety spotter. While doing the visual inspection, the maintainer will be exposed to the elements unless the aircraft is in a hanger.
2) The SHM system provides consistent repeatable objective results whereas the visual inspection is subject to variation due to the manual subjective nature of visual inspections.
3) The SHM system provides for automated data collection, whereas during the visual inspection the maintainer must manually record the inspection results.

The difference at the depot is that the depot is now responsible for the maintenance of the SHM system; this is somewhat if not entirely offset by the ability to have knowledge of the structural heath before the aircraft is inducted for PDM. The current inspection tasks remain relatively unchanged.

3.2.2.5 Non-Destructive Inspection (NDI) Activities

Other than eliminating the 32 month MOI there is no impact on current NDI activities.

3.2.2.6 New Maintenance Skills and Training Requirements

The new skills and training required are associated with the theory of operation, operation and repair of the embedded SHM Sensors and the portable electronics equipment.

3.3 Task 3 - Establish Availability, O&S, & Maintenance Man Hours Models

This assessed current legacy aircraft metrics and models to support identification and development of key metrics and methods for evaluating candidate platforms for structural applications to be selected in Task 5.0, and established the baselines and methods for assessing and quantifying the benefits of specific structural improvements.
There is a wide array of potentially suitable metrics available in use by the various services and by specific programs within each service. The three major categories of metrics considered were Availability, Mission Capability and Maintenance Cost. Of these three Availability and Maintenance Cost are the most significant and common. Additionally both are applicable when assessing implementation of CBM+ at the Organizational Level and at the Depot level.

3.3.1 Work Performed Prior to Aft Crown Selection

During the early work on this task while we were looking at the F-16, F-22, C-130J, F-35, and C-5 as potential platforms, it became apparent that due to differences in materials (composite verses metal), point program life cycle, customers and security concerns and access to data, it would require far more time and resources than available within the CBM+SI CRAD to develop a quantitative assessment approach to evaluate candidates across platforms.

3.3.2 C5A and Aft Crown Availability, O&S, & Maintenance Man Hours Models

Once we selected the C-5 as a platform, the data sources we used to initially assess the Aft Crown were primarily AFIRM and GO 81. GO-81 is the central common source of all unclassified maintenance data for Mobility airlift aircraft; It accumulates, validates, processes, stores, and makes accessible to Air Force and AMC managers the data necessary to keep AMC assigned and gained aircraft combat-ready. Worldwide logistics users connect to G081.

As for tools, we used AFIRM, in house developed ad hoc tools, as well taking a look at an LM Aero tool under development, OARCA (Operational Availability, Reliability, Cycle Time, Affordability)

Discount Rate	3%		
# of A/C	61		
MOI Labor Rate	$ 60.00		
HSC Labor Rate	$ 60.00		
Man Hours for MOI	128	Add duration in days of downtime	2
Man Hours for HSC	24	Add duration in days of downtime	3

	Option Description	NRE	Recurring Modification Cost	MOI Inspections per Year per A/C	HSC Inspections per Year per A/C	10 Year NPV	20 Year NPV	30 Year NPV	10 Year IRR	20 Year IRR	30 Year IRR
Option 1	Vendor XYZ offering for federated, battery operated system that can be remotely interrogated.	$ 200,000.00	$ 271,839.00	0.00	0.00	$0.00	$13,680,910.12	$23,860,792.09	3.00%	11.11%	12.41%
Option 2	Vendor ABC offering for fully integrated, A/C powered system that can be remotely interrogated.	$ 200,000.00	$ 50,000.00	0.06	0.50	($871,109.40)	$1,107,716.13	$2,580,148.16	-3.27%	6.59%	8.50%
Option 3	Vendor DEF offering for fully integrated, A/C powered system that can be controlled via the MFCD.	$ 200,000.00	$ 25,000.00	0.13	0.75	$160,102.04	$1,804,553.27	$3,028,179.43	4.87%	12.49%	13.64%
Option 4	Vendor GHI offering for federated, battery operated system that uses whiz-bang sensors.	$ 200,000.00	$ 12,500.00	0.25	1.00	$188,974.38	$1,304,063.13	$2,133,793.89	6.42%	13.66%	14.69%
	Include cost of downtime with a toggle to remove.										

Figure 9. LM Aero Cost Model – Future Projections – Currently Base-lined and in Refinement

LM Aero Cost Model – Future Projections (Figure 9) – Initially only addressed maintenance cost. The cost of unavailability, training and logistics costs associated with establishing and maintaining MOI capability were not included.

C-5 Aft Skin Stress Corrosion Crack Monitoring Cost Model

Option Information: Baseline C-5 as is today with required inspections due to aft crown cracking.

Baseline Costs

	2009	2010	2011	2012	2013	2014	2015
Number of MOI Inspections	0	16	16	16	16	16	16
Number of HSC Inspections	0	183	183	183	183	183	183
MOI Cost	$ -	$ 122,880.00	$ 122,880.00	$ 122,880.00	$ 122,880.00	$ 122,880.00	$ 122,880.00
HSC Cost	$ -	$ 263,520.00	$ 263,520.00	$ 263,520.00	$ 263,520.00	$ 263,520.00	$ 263,520.00
Total Cost	$ -	$ 386,400.00	$ 386,400.00	$ 386,400.00	$ 386,400.00	$ 386,400.00	$ 386,400.00

* Start MOI Inspections in 2010

Figure 10. Baseline MOI and HSC Cost Projections

Figure 10 illustrates the initial projected baseline Home Station Check and MOI Inspection costs. This projection was updated and refined as part of the Business Case Development.

The objective of the early C-5 cost and availability assessments was to confirm that the current C-5A Aft Crown maintenance concept imposed sufficient costs, unavailability and operational impacts to justify consideration as a viable CBM+SI candidate.

As it turned out when we further developed the data for the business case, the potential improvement in Availability that we initially assumed was not realistic. Due to the concurrent non-structural inspection and servicing activities performed as part of the 120 day Home Station Check the overall duration could only be reduced by an hour or two. The Aft Crown was the longest activity at an estimated 8 hours, there was another activity (Avionics Op Check) that was estimated at 5 hours and several others at 3 hours. Since the SHM System assessment, taking an estimated 1 hour, replaces the visual inspection the net improvement in availability was marginal. This topic will be discussed further in section 3.8 as part of the Business Case.

3.4 Task 4 - Develop ASIP/CBM+SI Information Architecture

This task investigated existing information architectures to establish a baseline understanding of the context that a CBM+SI implementation must exist within. Early candidate platforms included the C-5 and the C-130J, as well as the F-16 and F-22. Since technical descriptions for the C-5 and C-130J were more readily available and not as constrained by Program and Security limitations on public release, we investigated the Information Architecture for the C-5 (C-5A/B, C-5 AMP and C-5 M - RERP) and the C-130J.

Both the C-5 M (RERP) and the C-130J have modern and rich Information Architectures with significant processing, storage and interfaces to provide access to aircraft parametric data. This being the case, implementation of CBM+SI capabilities for these platforms is potentially less costly since they both already possess inherent sensing, processing and data management capabilities.

The first activity we undertook was to investigate the C-5 and C-130J Structural Health Monitoring (SHM) Architectures. Specifically, the SHM functions within the C-130J, C-5A/B, C-5 AMP and C-5 RERP were addressed.

Each aircraft variation incorporates a diagnostic subsystem that includes an SHM function. The LM Aero design intent is to integrate avionic and diagnostic (including SHM) functionality as much as practical in order to reduce hardware components while assuring safe, timely, accurate and reliable performance.

3.4.1 C-130J – Integrated Diagnostics System (IDS)

The C-130J IDS records in-flight information to support individual aircraft maintenance and fleet management for the C-130J aircraft. The data recorded includes information to perform an SHM assessment for each individual aircraft. Post-flight processing (i.e., maintenance and SHM) is saved to a removable solid state media device. Ground Maintenance System (GMS) is hosted on the Portable Maintenance Aid (PMA), allows post-flight debriefs and recording of aircraft corrective actions and tests. One SHM post-flight activity is the scheduling of inspection of airframe structural members based on calculated usage information. SHM converts recorded airframe acceleration to stress at ten (10) strategic airframe points (SHM Zones). Data is used to calculate the primary structure fatigue life expended. C-130J SHM is based on “g-tracking” methodology

3.4.2 C-5A/B Galaxy – Malfunction Analysis, Detection and Recording System (MADARS) II

Designed in the 1960’s, the C-5A was the first military airlifter to incorporate an onboard diagnostic system. During C-5B production (1980 timeframe), the C-5A MADARS was upgraded to the MADARS II. A selected number of flight parameters are monitored and recorded on removable media. Selected data is extracted by ground processing systems (i.e., GO-81) and is used to determine operational trends, airplane performance, and structural loading history. The C-5A/B SHM is known as Loads Environment Spectra Survey (LESS). The SHM consists of a combination of hardware and software components tightly integrated into the MADARS architecture. LESS equipped aircraft utilize strain gages that are interfaced to the Signal Conditioner/Monitor (SC/M).

3.4.3 C-5 AMP - Malfunction Analysis, Detection and Recording System (MADARS) III

The C-5 Avionics Modernization Program (AMP) is a C-5A and C-5B aircraft that is equipped with a completely new avionics suite. MADARS upgrade was also performed, as a separate effort. Known as MADARS III, all C-5 AMP aircraft have MADARS III. Original function of MADARS is retained including SHM. The SHM process requires the acquisition of select aircraft parameters and sensors on a subset of C-5 aircraft. This subset sample is then used to project fleet-wide useful life. There are no differences between the C-5A/B and C-5 AMP SHM premises. All information discussed for the C-5A/B SHM applies to the C-5 AMP.

3.4.4 C-5 RERP - Embedded Diagnostic Subsystem (EDS)

RERP SHM is backwards compatible with the already existing SHM post-flight processing schema. The EDS assures the continued acquisition of SHM data that has been accumulated since the aircraft first entered service. EDS does, however, significantly improve the resolution and accuracy of the collected SHM parameters. EDS is a completely digital product and is less susceptible to the problems experienced by the C-5A/B and C-5 AMP analog MADARS.

Currently, only the C-5B aircraft are scheduled for the RERP upgrade. That means there will be a total of six (6) aircraft with the LESS SHM function. The data acquisition methods for all of the legacy LESS parameters have changed. The analog Signal Acquisition Remotes (SARs) are replaced with a state-of-the-art Remote Interface Unit (RIU). In addition to the strain gages, several other aircraft parameters are recorded for post-flight SHM analysis. The original C-5A/B SHM premise is retained in RERP. SHM parameters, data collection frequency and recording are the same between aircraft variations

3.4.5 C-5A Off Aircraft Information Architecture

The baseline C-5A Off Aircraft Information Architecture, as with virtually all legacy aircraft, is limited in that its main government data systems are G081 and Integrity Data System. The C-5 also has the AFIRM Data System described below.

3.4.5.1 C-5 Aging Fleet Integrity and Reliability Management Program (AFIRM)

Lockheed Martin engineering support to the C-5 aircraft structural integrity and functional systems tasks of the Warner Robins Air Logistics Center (WR-ALC) is provided through the C-5 Aging Fleet Integrity and Reliability Management Program (AFIRM). Numerous studies, analyses, databases, technology enhancements, etc. are required to ensure a sound technical basis for successful fleet management. The C-5 AFIRM website acts as an online resource for the wealth of information described above.

The data that populates the AFIRM website comes from several different sources which are identified below:

AFKS - Air Force Knowledge Services

CAMS-FA/G081 - Core Automated Maintenance System for Mobility Forces/G081

D043 - Air Force D043 Master Item Identification.

REMIS USAF - Reliability and Maintainability Information System.

The Aging Fleet Integrity & Reliability Management (AFIRM) program integrates the proven philosophies of the Aircraft Structural Integrity Program (ASIP) and the Functional Systems Integrity Program (FSIP), and strives to increase fleet reliability, safety and mission readiness, while also reducing costs. Simply put, AFIRM = ASIP + FSIP. The goal of the AFIRM website is to consolidate all structures, systems, mission capability, and flight data into an easy to navigate and intuitive online application. It should be noted that the website is continually evolving to meet the current and future needs of the C-5 ASIP and FSIP Managers. It will be under constant expansion with new programs and will also receive periodic updates to keep the appearance and functionality as fresh and efficient as possible.

The particular value of AFRIM relative to the CBM+SI Application on the Aft Crown is that there is already a process and capability in place in AFIRM to manage and track the structural status of the Aft Crown. The current implementation supports manual entry of inspection and repair data for each C-5A, but the basic elements are in place to support an automated upload of data from the Aft Crown SHM System.

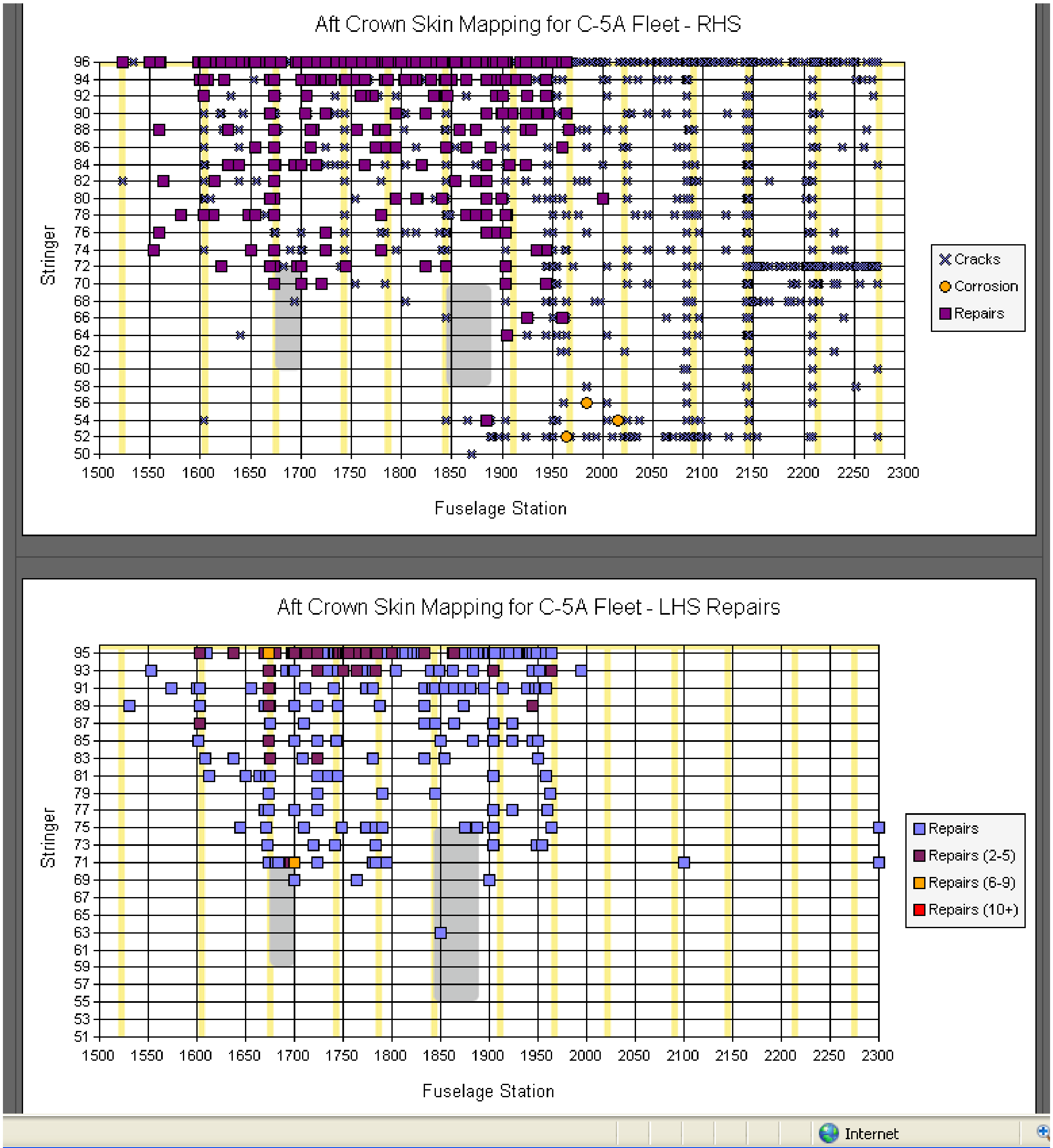


Figure 11. AFIRM Aft Crown Skin Mapping

3.5 Task 5 - Emerging Technology Review

In this task, we investigated both existing and emerging technologies for sensors, sensor data fusion, sensor integrity, information architecture, analysis tools and prognostics algorithms. This included technologies such as MWM sensor arrays from JENTEK Sensors Inc. and SMART LAYER from Acellent. Each technology was evaluated in terms of maturity level to support the ASIP/CBM+ strategy.

Four sensor technology vendors were invited to LM Aero to review structural health monitoring capabilities and issues. They were: Impact Technologies based on their significant experience in system and architecture design, and data fusion; TRI Austin, because of their active ultra-sonic damage detection technology; Acellent Technologies, Inc., because of their passive and active piezoelectric ultra-sonic damage detection technology; and JENTEK Sensors, Inc. for their MWM array damage detection technology. We also invited Professors Jennifer and Thomas Michaels of Georgia Tech to discuss their work and our project.

Later in the project we also invited Goodrich in to discuss their Comparative Vacuum Monitoring technology.

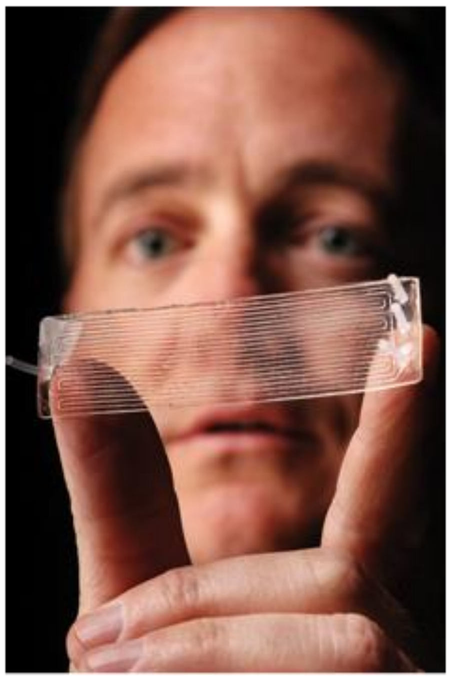


Figure 12. Comparative Vacuum Monitoring (CVM)

Comparative Vacuum Monitoring (CVM) is a thin, self-adhesive rubber patch, ranging from dime- to credit-card-sized, that detects cracks in the underlying material. The rubber is laser-etched with rows of tiny, interconnected channels or galleries, to which air pressure is applied. Any propagating crack under the sensor breaches the galleries and the resulting change in pressure is monitored.

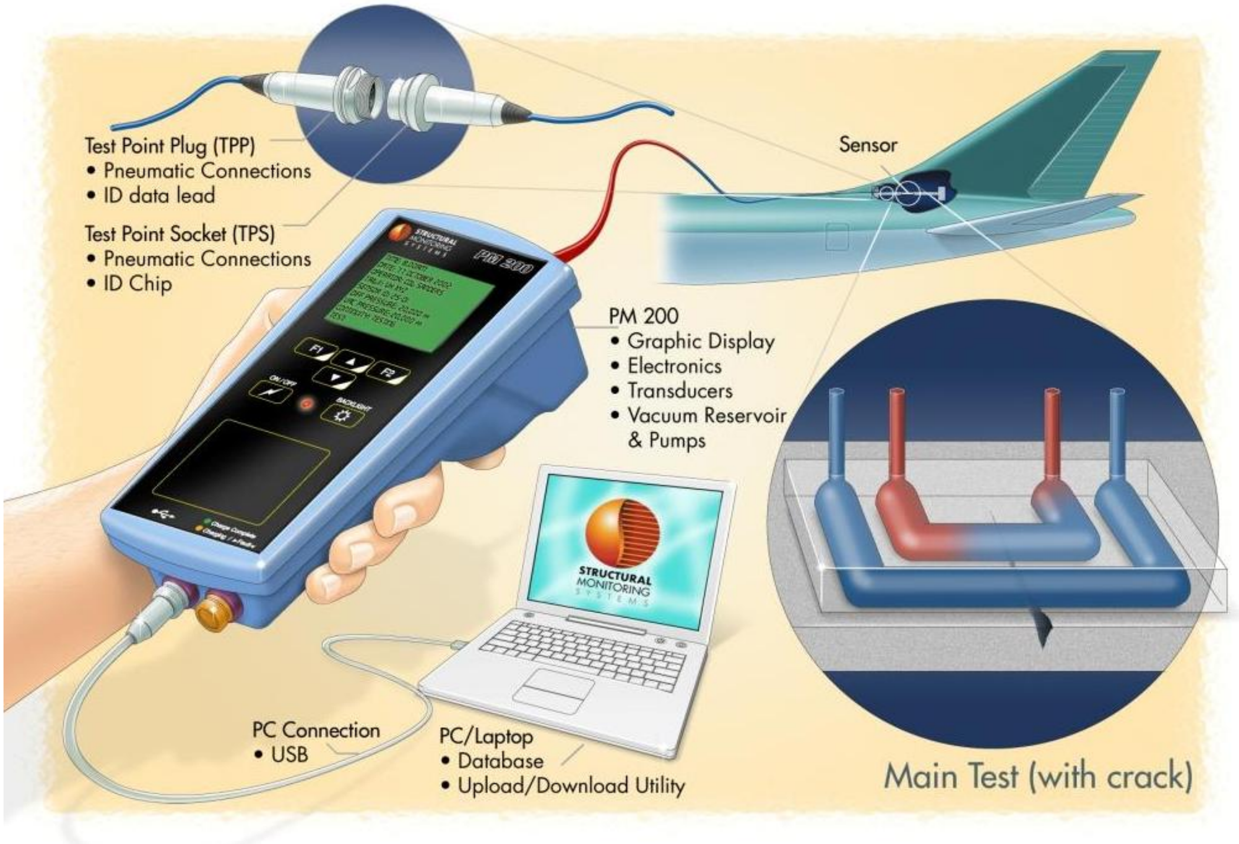


Figure 13. Comparative Vacuum Monitoring (CVM) System

After assessing the various technologies available in the context of the C-5 Aft Crown, we decided on Impact Technologies, JENTEK and Acellent to be involved in our Structural Application Prototype.

We initially planned on having Goodrich develop a white paper on a notional application of the CVM technology for the C-5A Aft Crown. The approach we discussed was installing a series of lateral CVM strips on the exterior of the Aft Crown one between each stringer. This installed system is completely passive and, with the portable test unit, presented an interesting approach for detecting through cracks. Unfortunately, the contractual relationship between Goodrich and the CVM technology supplier ended before we could pursue the white paper formally.

3.6 Task 6 - Candidate Selection

This task involved the selection of both the structural application and the SHM technologies to be used for the baseline CBM+ demonstration.

The structural application candidate needed to be one for which the inspection and maintenance burden is currently high, and for which that burden could be significantly reduced with implementation of SH monitoring and CBM+. Estimation of the projected maintenance relief for each candidate took in to account the overall maintenance planning for the weapon system (required scheduled maintenance in adjacent structure may dilute the benefit). The CBM+SI demonstration candidates were drawn from Lockheed Martin's family of platforms and include (but are not limited to) the following; (1) C-130 CWB Rainbow Fitting, (2) C-5 Fuselage Aft Crown, (3) F-16 wing skin/wing attach fitting joints, and (4) F-22 side-of-body lugs.

SHM technologies and systems which were specifically applicable to the identified structural application candidate(s) were assessed, and associated risks as well as potential benefits were be evaluated. General criteria for the selection of applicable technology included: availability (schedule) of sensor hardware, validation of performance, long-term durability & robustness, cost, footprint, impact on structure, data processing and storage requirements, and power requirements.

The first assessment was a qualitative assessment of each candidate platform as to its suitability for the application of CBM+. This assessment considered things like current program life cycle stage, security concerns, as well as high level technical considerations such as weight, volume and information architecture constraints.

3.6.1 LM Aero Candidate Platforms considered for Application Prototype Development

Before selecting a specific structural problem to address we needed to identify a suitable target platform. We looked at the F16, F-22, F35, C-130, and C5. We considered several technical and non-technical characteristics of each program and aircraft. The following summarizes the results of this assessment.

> **F-16** - Rejected due to constraints (weight, volume, access, power) associated with small aircraft. Difficult business case due to multiple customers.
>
> **F-22** - Rejected for same reasons as F16 (except multiple customers), plus current program focus is Structural Retrofits, Corrosion Prevention (not detection), Establishing program Depot Infrastructure, and keeping production line open and now shutting down the production line. Additionally F-22 is a Special Access Program SAR Program and as such security constraints would limit access to data.
>
> **F-35** - Rejected for same reasons as F-16, plus current focus is on executing development, test and production plan. Additionally a significant investment in PHM and SHM has already been made.
>
> **C-130** - Rejected - Due to the number of operators (customers) business case seems too complicated for scope of the CBM+SI Phase I effort
>
> **C-5** - Judged as best overall opportunity (Big aircraft, single customer, several structural candidates, Program and ALC interested in the project, etc)

3.6.2 C-5 Structural Application Prototype Candidates

The four C-5 Structural Items considered were:

C-5A End Fitting

Contour Box Beam Fitting (FS 484)

C-5B Horizontal Tie Box Fitting

C-5A Aft Crown

3.6.2.1 C-5A End Fitting

3.6.2.2 Description

The C-5A end fittings are located in the chine area of the C-5 and located on every frame the entire length of the cargo floor, except near the main landing gear. These fittings act as the splice member for each frame between the side frame in the cargo bay, the lower lobe and the cargo floor structure. See Figures 14 and 15.

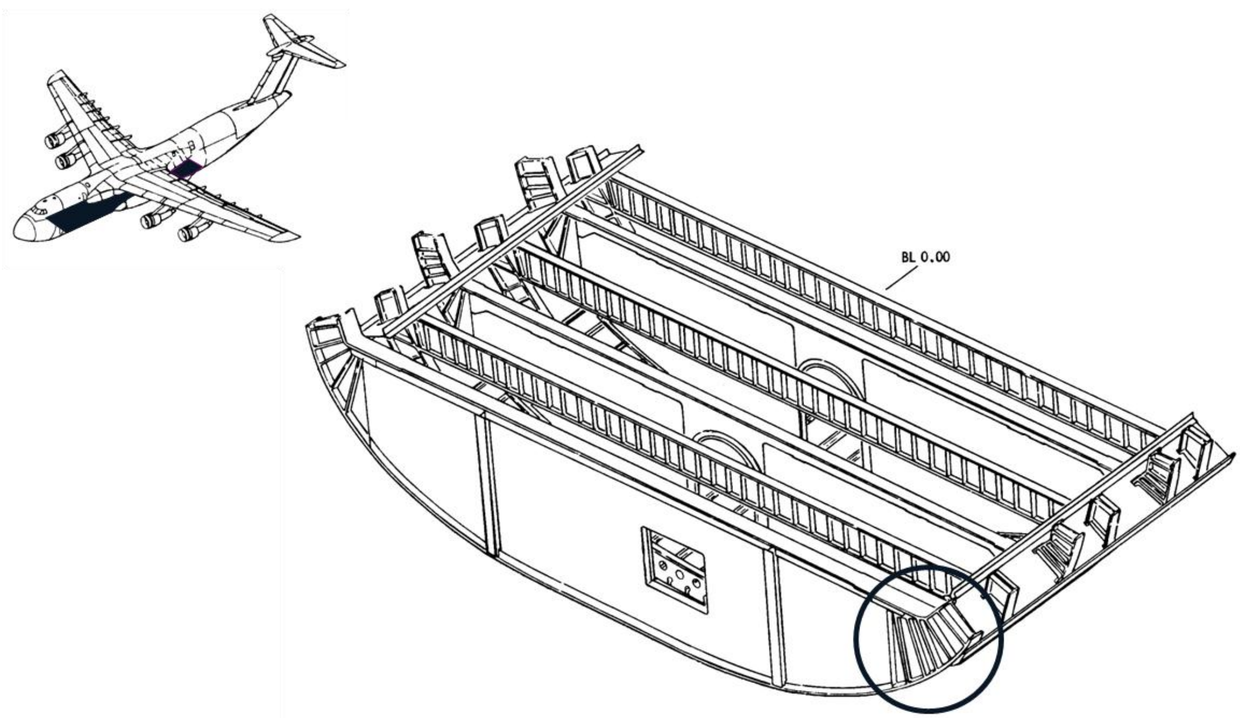


Figure 14. C-5A End Fitting Location

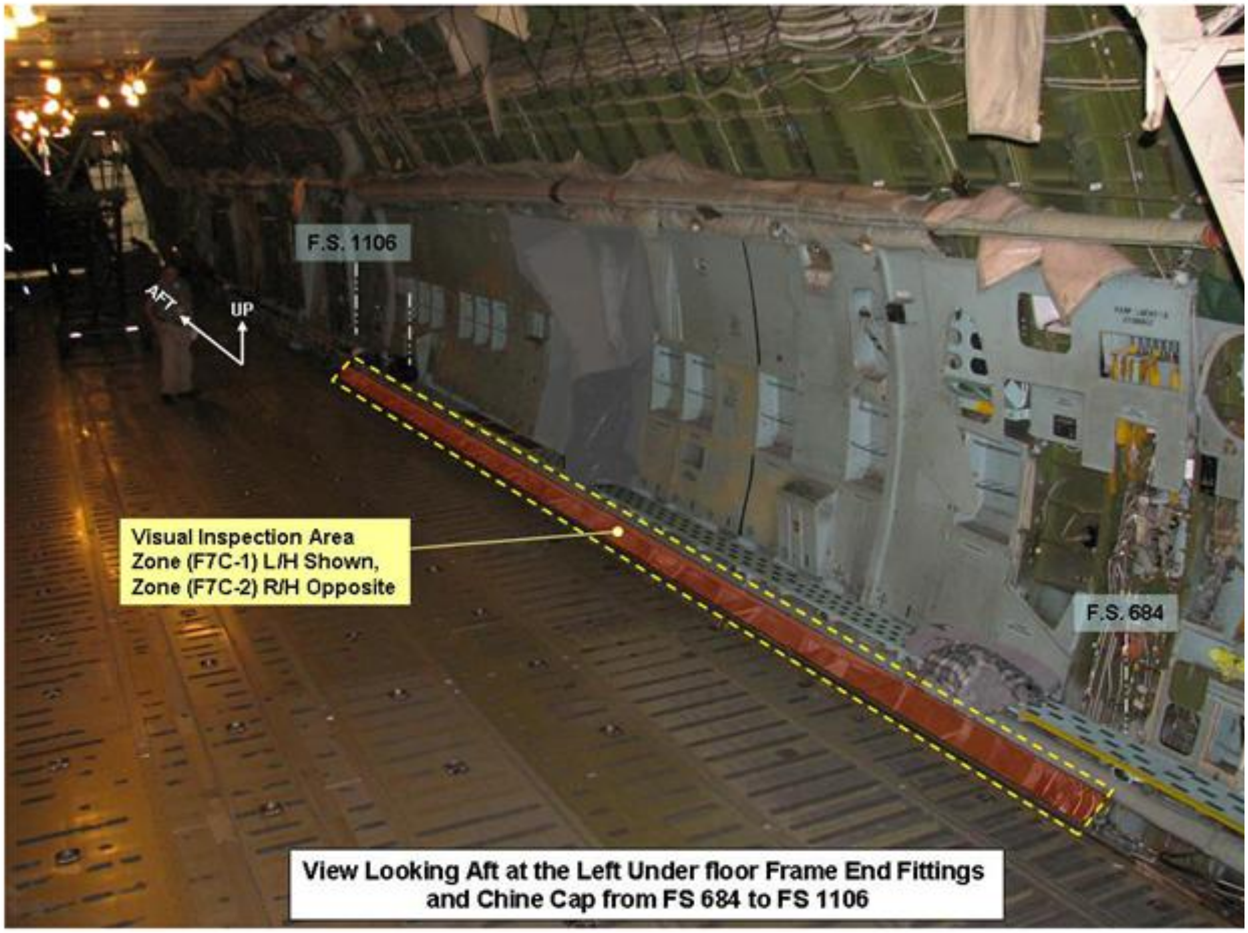


Figure 15. Pictorial View

3.6.2.3 Problem

These fittings are 7075-T6 forgings and are highly susceptible to stress corrosion cracking. Numerous stress corrosion cracks have been found on C-5A aircraft. The 7075-T6 fittings were changed to 7049-T73 to correct the problem on the C-5B aircraft. Repairs have been developed, analysed, and incorporated into Technical Order (TO) 1C-5A-3. Replacement procedures for fittings have been developed.

3.6.2.4 Reason Not Selected

Currently, this problem is being managed by inspection, repair and replacement. Fittings at Fuselage Station (FS) 524 and FS 1964 are difficult to repair and are being replaced on several aircraft. Fittings at other fuselage stations with damage beyond the TO 1C-5A-3 limits are also being replaced. WR-ALC has no fittings stocked and this is a long lead time item (9 months). Aircraft are leaving PDM restricted and being brought back when parts become available. This item was not selected since it is considered a supply issue, not a safety issue.

3.6.2.5 Contour Box Beam Fitting

3.6.2.6 Description

The C-5A FS 484 contour box fittings are located at WL 314 and are located on the most forward full frame on the aircraft. These fittings act as the splice member for the FS 484 frame upper to lower lobe, the contour box beam, which runs fore-aft, the canted frame, and the WL 314 floor beam. Figure 16 depicts the location of the contour box beam fittings, but does not show all of the additional structural elements that offer alternative load paths in this fail-safe joint.

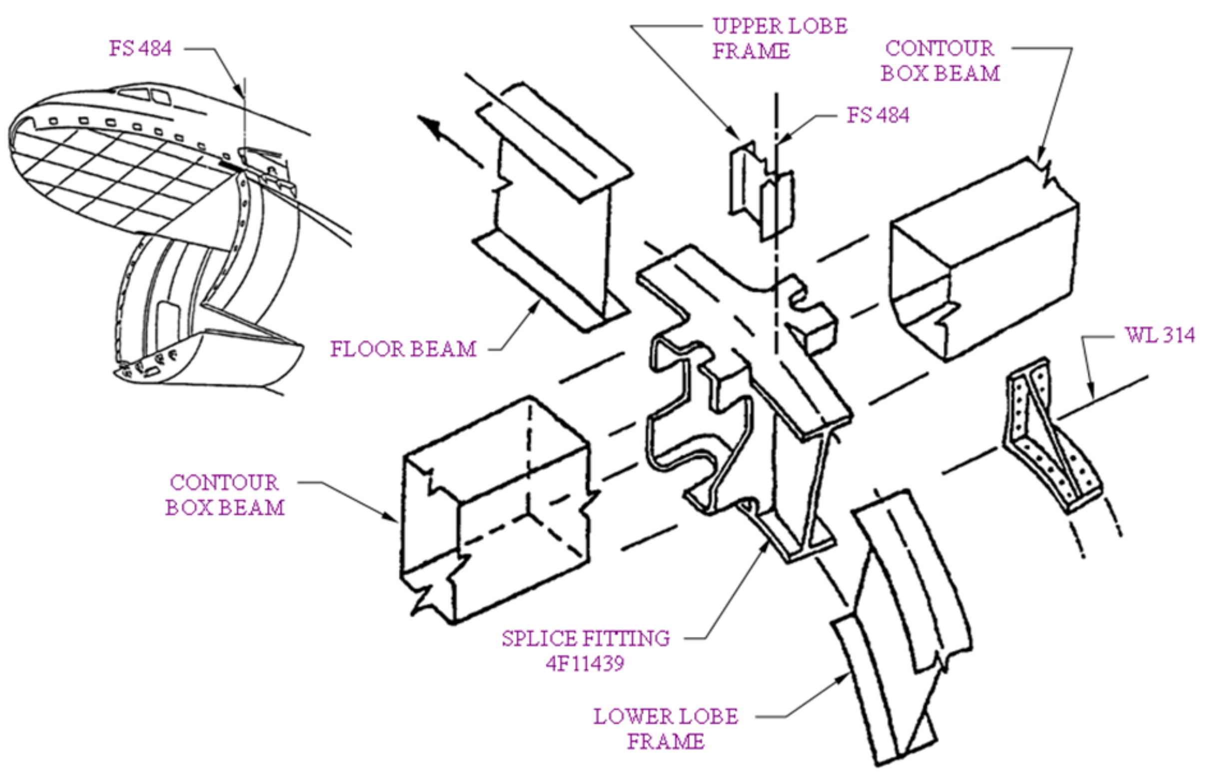


Figure 16. FS 484 Contour Box Beam Fitting

3.6.2.7 Problem

These fittings are 7075-T6 forgings and are highly susceptible to stress corrosion cracking. See Figure 17 for the type of damage that requires fitting replacement. The first fitting damage was reported in 1978. Various types of damage have been found on several aircraft. Repairs were developed for some known problems but no feasible repair exists for the type of damage shown in Figure 17.

Figure 17. Damaged Contour Box Beam Fitting

The fittings must be replaced. The primary loading of the fitting is due to fuselage pressurization. No damage has been reported on C-5B which had a material change to 7049-T73. The Air Force currently performs X-ray and eddy current surface every 16 Months and every 4 years. Attempts to find better NDI methods have been made but to date proven unsuccessful. A Contour Box Beam Fitting replacement program has been proposed but remains unfunded.

3.6.2.7.1 Reason Not Selected

This item was not selected since it is currently successfully being managed through inspect and replace.

3.6.2.8 C-5B Horizontal Tie Box Fitting

3.6.2.8.1 Description

The C-5 horizontal stabilizer rear tie box fittings carry the left and right stabilizer aft spar loads at the centerline of the aircraft, and provide the support for the pivot pin lugs at the vertical stabilizer attachment. Figure 18 shows the tie box and surrounding horizontal stabilizer structure.

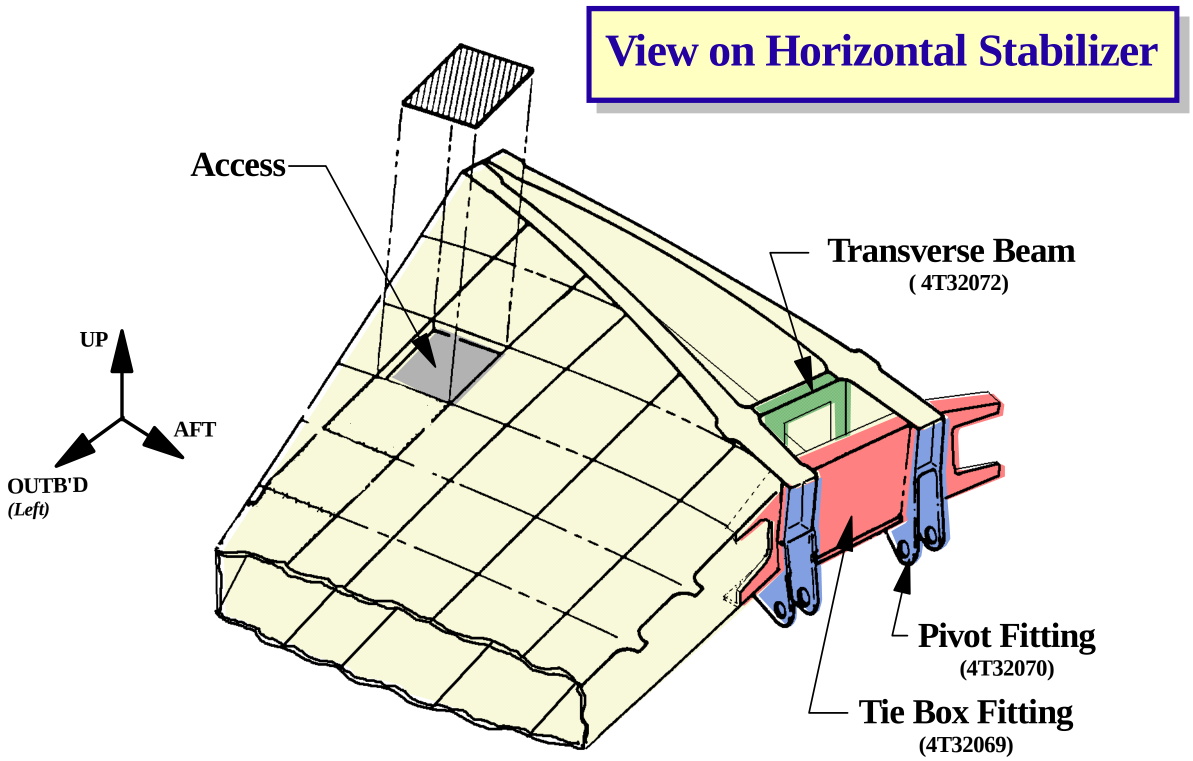


Figure 18. Horizontal Stabilizer Structure

The C-5A and C-5B tie box fittings were fabricated from 7075-T6 aluminum hand forgings (See Figure 19).

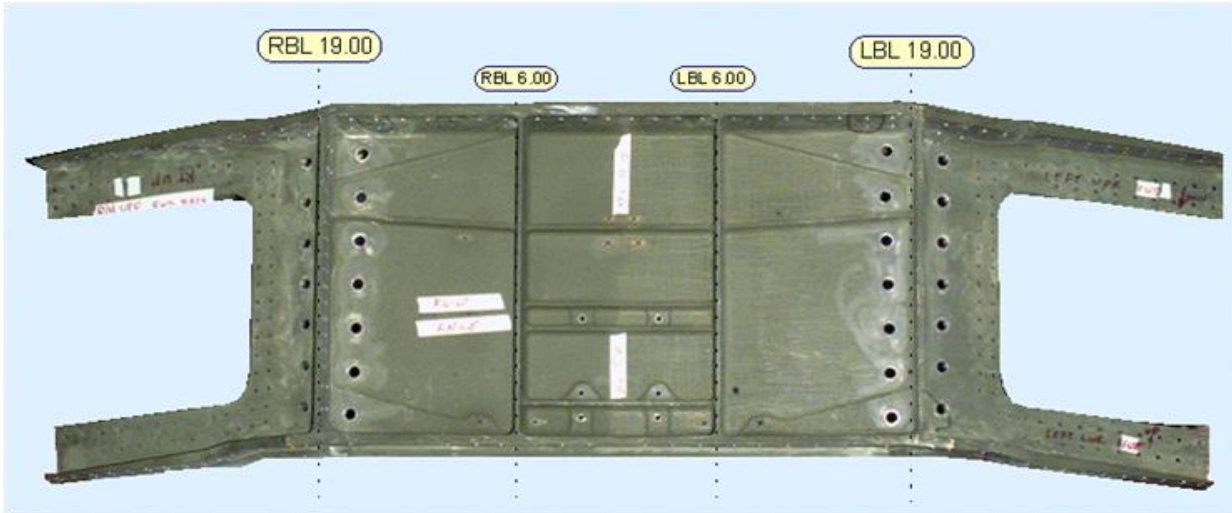


Figure 19. Tie Box Fitting

3.6.2.8.2 Problem

Stress Corrosion Cracking (SCC) was first observed in a tie box on aircraft 69-0008 (LAC 0039) during Program Depot Maintenance (PDM) in February 1998. The C-5B Tie Box Fittings are currently being replaced at PDM and several aircraft will have fittings replaced during other maintenance activities.

3.6.2.8.3 Reason Not Selected

Although this item is a significant safety issue, it was not chosen since all of these fittings will be replaced by the end of FY2012 which was deemed too soon to gain any benefit from sensor technology.

3.6.2.9 C-5A Aft Crown

3.6.2.9.1 Description

The C-5A aft fuselage skin was fabricated from 7079-T6 sheet, clad one side. The Upper Lobe has a radius of approximately 85” (compound curvature) in the region of interest. As stated previously, the skin is 7079-T6 sheet; clad one side with a varying thickness of 0.050” to 0.071” (varies according to stability requirements for compression combined with shear). The most critical region for cracking is 0.050” thick. The stringers are ‘Z’ and ‘J’ 7075-T6 Extrusion with a typical spacing = 7” to 8”. The frames are 7075-T6 rolled/stretch-formed with a spacing = 20”.

See Figure 20 for the aft crown skin region.

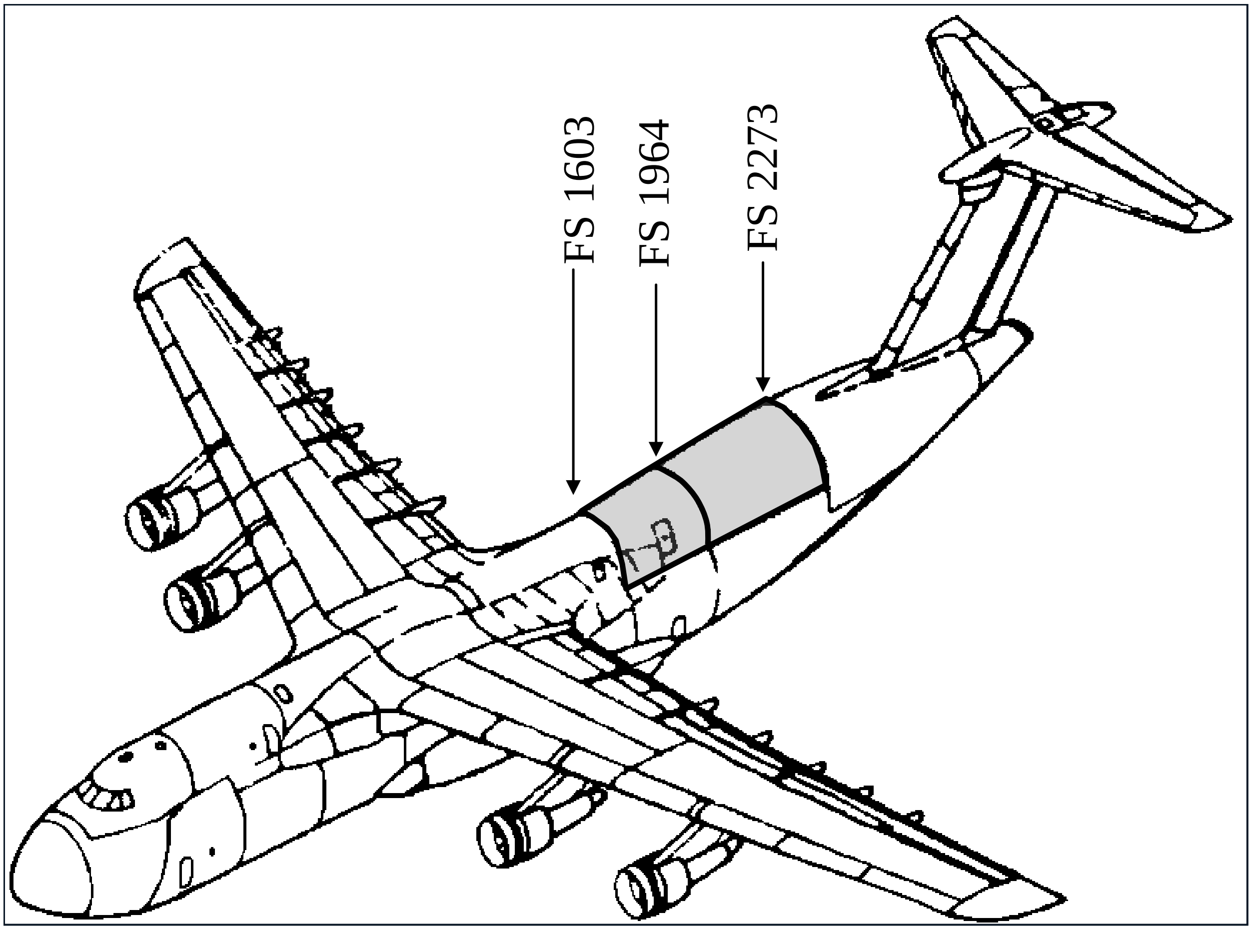


Figure 20. Aft Crown Skin Region

3.6.2.9.2 Problem

This is a well known example of a material with high static strength, low toughness, and high susceptibility to Stress Corrosion Cracking (SCC) damage. The increasing frequency of this type of damage peculiar to the C-5A has elevated the level of concern for safety of flight and has become an economic burden to repair damage when it is detected. Some damage found is illustrated in (Figure 21),

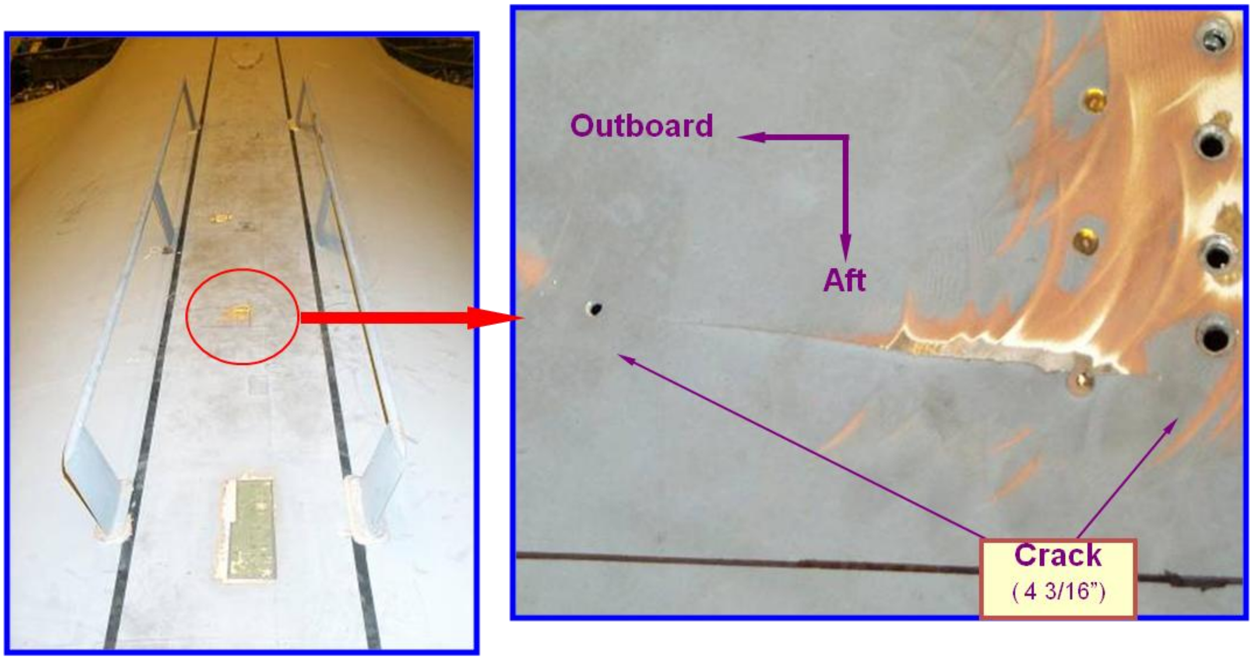


Figure 21. Crack Found in Nov 2002, A/C 67-0170

These problems prompted recommendations of urgent visual inspections at HSC intervals and better NDI procedures, including the Magneto Optic Imaging (MOI) technique.

The MOI uses a combination of an innovative eddy current induction method to induce magnetic fields in defects and magneto-optics to form images of the magnetic fields associated with the defects. These real time field images closely resemble the defects themselves. The MOI is able
to image through paint and other surface coverings in real time and displays results as visual images on a heads-up display and/or an ordinary TV monitor. The instrument is hand-held, portable, requires minimal training, and greatly increases the speed and reliability of inspection. Results may be videotaped, printed using a video printer or captured digitally. (Figure 22)

Figure 22. MOI in Use on Commercial Aircraft

Analysis concluded that the crown skin could not withstand the classical fail-safe criteria of a 'two-bay plus stringer' failure and recommended updating the Force Structural Maintenance Plan (FSMP). Fail-safety refers to the ability of a secondary load path member(s) to carry additional loads after damage or failure of the primary load path member for a specified period of time. Various repairs and modifications, including the possible addition of external longitudinal fail-safe straps, were investigated to mitigate the risk associated with the crown skin problem.

This cracking has been observed on A-Model Skins Only. See Figure 22 – for a typical damage map and Figures 23 and 24 for typical damage. The B-models skins are thicker and made from a different alloy. Cracking has not been observed in the B-models.

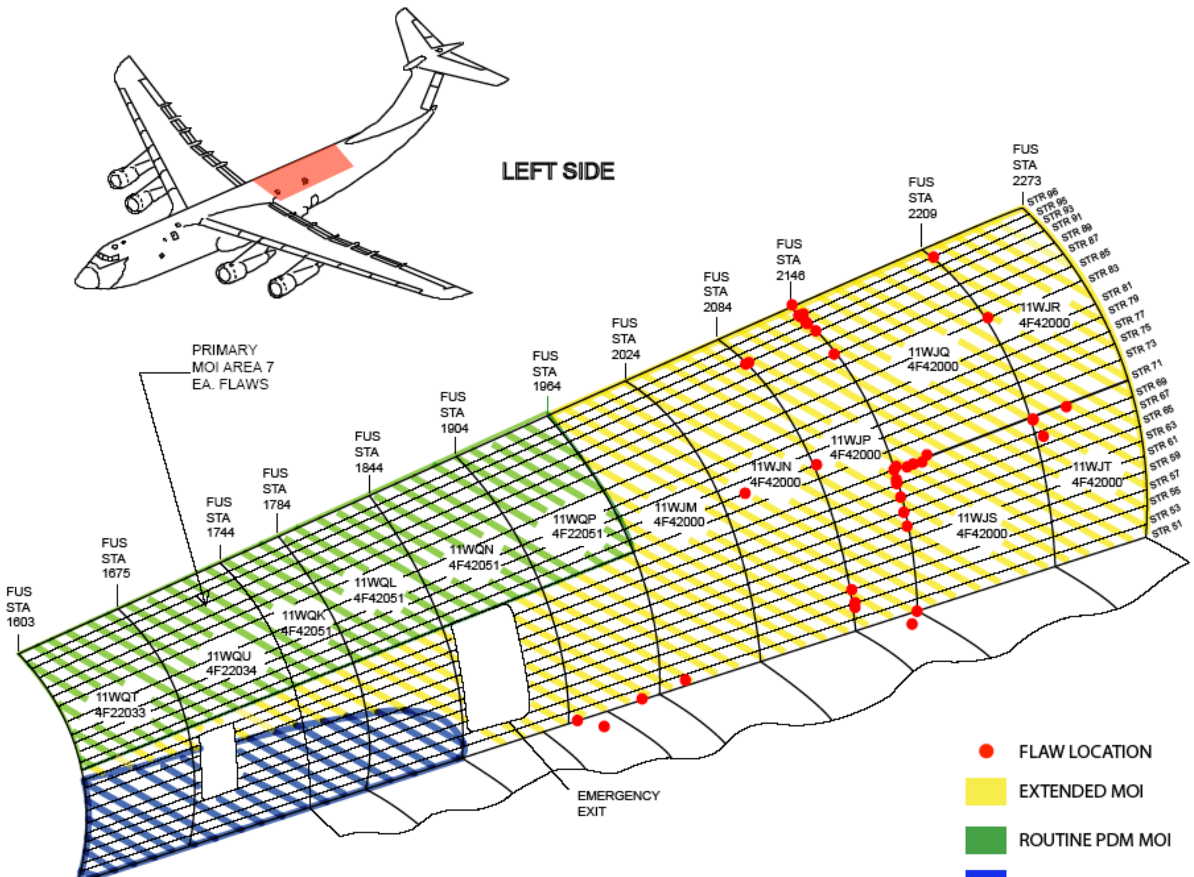


Figure 23 Damage Map from A/C 70-0453 Left Side

Figure 24. Cracking Found on A/C 70-0453

Figure 25. Cracking on A/C 68-0222 (Including Cross-Section)

3.6.2.9.3 Reason Selected

Conservative analysis methods caused by the absence of a suitable growth model and the high scatter, have led to a burdensome inspection program for the Air Force. Since the MOI that is performed in this area is such a labor intensive inspection, the C-5A Aft Crown Skin was seen as a perfect candidate for sensor technology to aid in the potential reduction of maintenance costs.

Currently, a Visual Inspection is accomplished at the HSC interval (120 days) and an MOI is accomplished in the critical zone at a Major interval (48 Months). See Figure 26.

C7A-1B, A5A-1A – DVI every 120 Days (HSC) & MOI every 32 months (MOI to be done every 48 months)

C7A-1C, A5A-1B – DVI every 48 months (Major) & MOI every 8 years (PDM)

Figure 26. Current Inspection Program Details

3.7 Task 7 - Develop Structural Application Prototype

With support from Acellent, Impact and JENTEK we developed a prototype application to evaluate the effectiveness of each sensing technology and to collect information needed to develop a business case supporting the development and installation of a production system based on the prototype concept.

3.7.1 Structural Application Prototype for C5 Aft Crown Implementation and Evaluation

3.7.1.1 Test Articles

We removed Aft Crown sections from the X997 test article in the Lockheed Martin boneyard in Marietta. A section was removed from X997, cut in half along BL 0 and set up in a lab in building B4 in Marietta.

Figure 27. Side View of X997 with section to be removed in red

Figure 28. Top View of X997 with section to be removed in red

Figure 29. Interior View of X997

Figure 30. Test Articles in Lab

3.7.1.2 Test Plan

3.7.1.2.1 Purpose

The objectives of the test program were to demonstrate the installation and operation of two damage sensor systems on a realistic structural component, and to evaluate the ability of each to detect both pre-existing (natural) and new (artificially induced) flaws. Each installation and demo took take place in a laboratory environment. The two sensors systems evaluated were the Acellent SMART Layer sensor and the JENTEK surface mounted Meandering Winding Magnetometer (MWM) Array sensor.

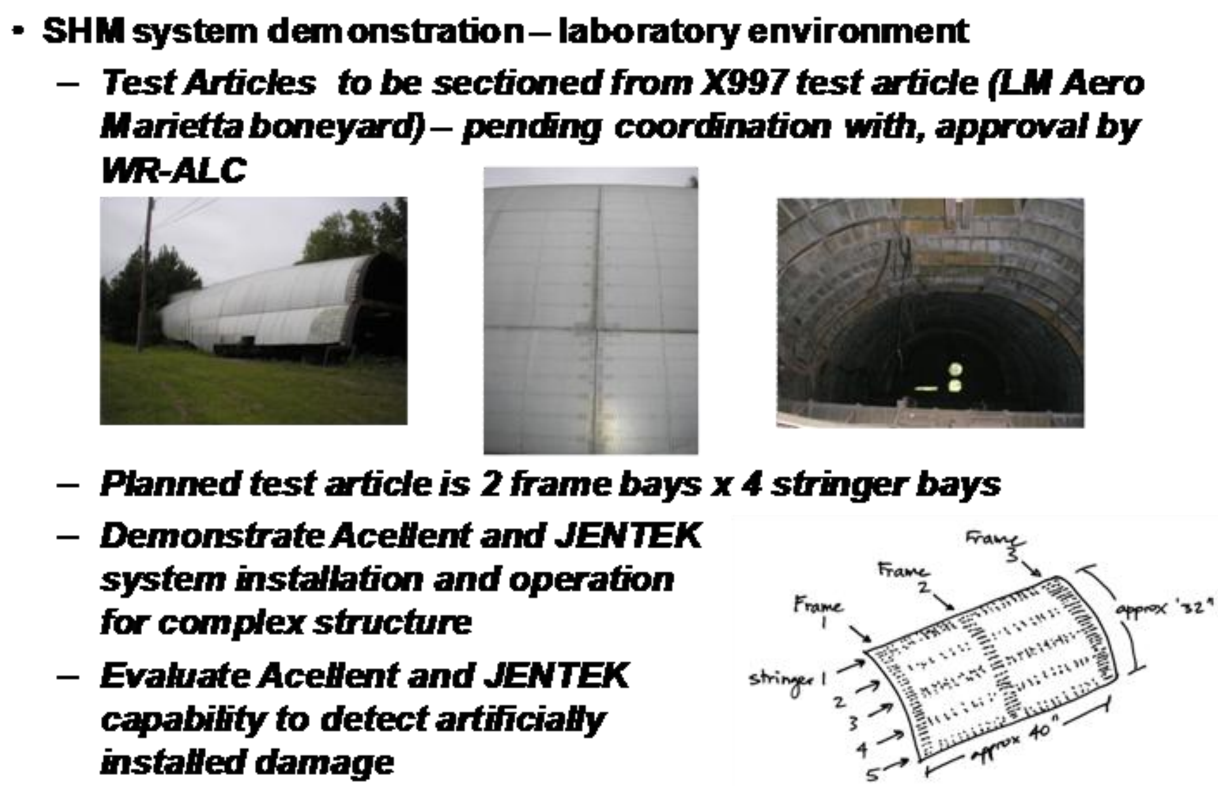
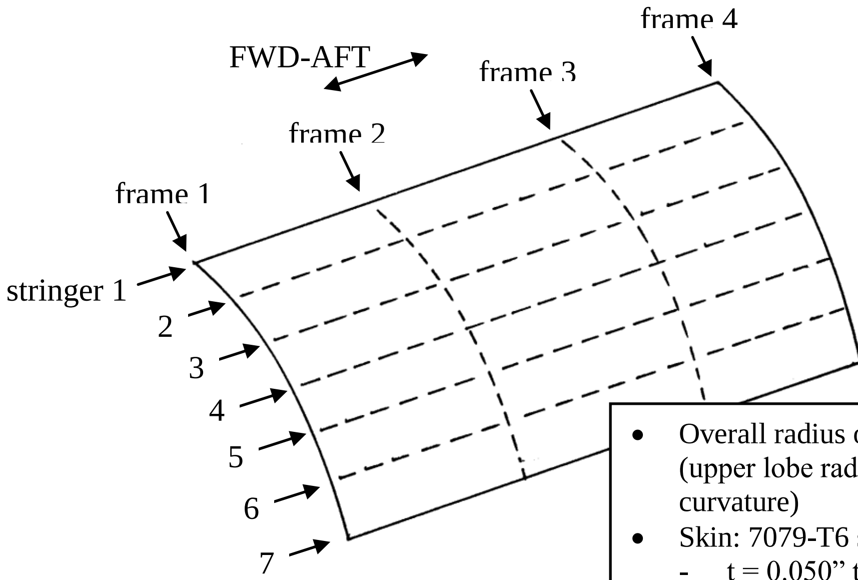


Figure 31. SHM Demo Overview

This program did not entail either static or cyclic loading of the test articles. The program assessed the ability of each of the two sensor systems to detect damage in complex structure in a laboratory setting.



- Overall radius of frame / skin assembly (upper lobe radius) ≈ 85” (compound curvature)
- Skin: 7079-T6 sheet, clad one side
  - t = 0.050” to 0.071” (varies according to stability requirements for compression + shear)
- Stringers: 7075-T6 Extrusion
  - ‘Z’ and ‘J’ cross-section
  - Spacing = 7” to 8”
  - Two rows of x” (material) fasteners w countersunk heads
- Frames: 7075-T6 rolled/stretch-formed
  - “J” cross-section
  - Spacing = 20”
  - Three rows of x” (material) fasteners w countersunk heads

Figure 32. C-5A Aft Crown Test Article

3.7.2 Acellent SMART LAYER Evaluation

3.7.2.1 Introduction

This section describes the test planning, installation, testing, demonstration and evaluation of Acellent’s SMART Layer Structural Health Monitoring (SHM) system on realistic structural components.

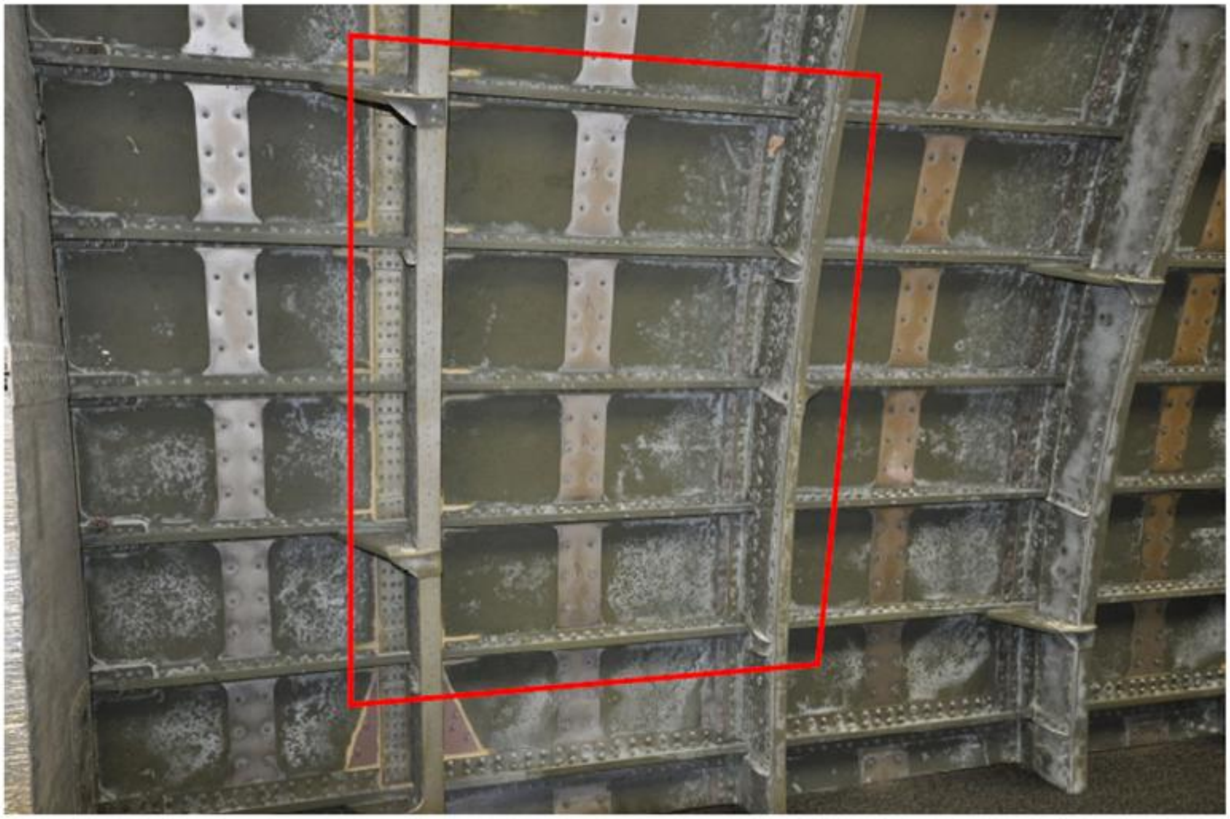


Figure 33. Area to be Monitored

For the demonstration, sensor layers were designed and manufactured to monitor four bays of the C-5 Aft Crown test specimen (Figure 33). The installation and demonstration took place in a laboratory environment at Lockheed Martin in Marietta, GA. A Dremel tool was used to cut artificial cracks.

These flaws were intended to simulate through-the-thickness cracks in the skin.

3.7.2.2 SMART Layer Design

There are several factors to consider when designing the SMART Layer. These include (1) the critical damage size that must be detected, (2) the propagation distance of the ultrasound signal in the structure, and (3) the location of all sensor “keep out” areas. For this demonstration, the target minimum damage size to detect was 0.5”. From previous experience on thin aluminum panels (< 0.2”), it was expected that the ultrasound signal could propagate across multiple bays (> 20”). Regarding “keep out” areas, a titanium strap runs in the circumferential direction in the center of each bay. The strap is mechanically fastened to the skin, but is not bonded. Because of this, the titanium strap area is considered a “keep out” zone for the piezoelectric elements, although the circuit layer can still cover the strap area.

A rectangular ring-shaped SMART Layer was designed to fit in the bays (Figure 34). The ring was overpopulated with piezoelectric elements to allow for a trade study related to sensor density to be conducted.

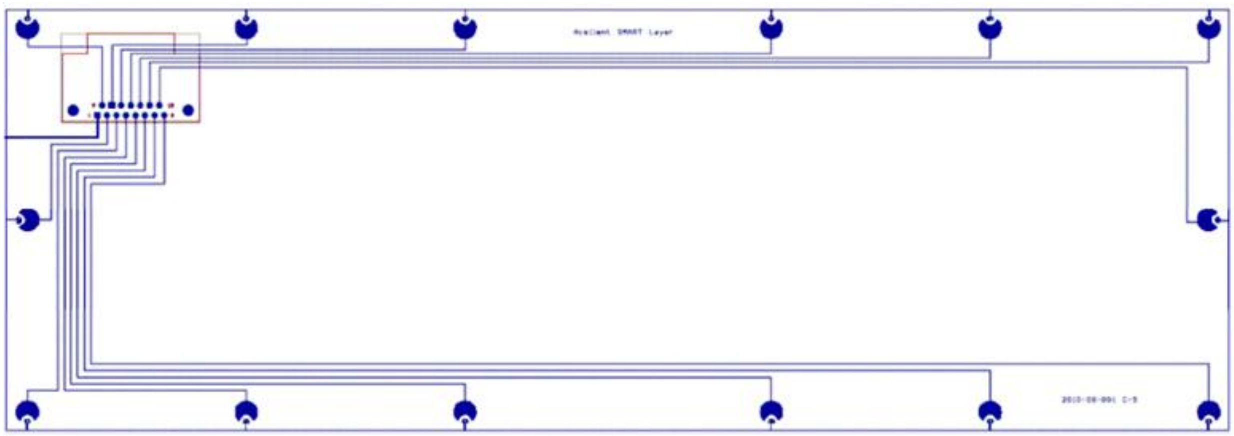


Figure 34. Ring-shaped SMART Layer Design

In all, four ring-shaped SMART Layers and two strips were manufactured to be installed in the bays as shown in Figure 35.

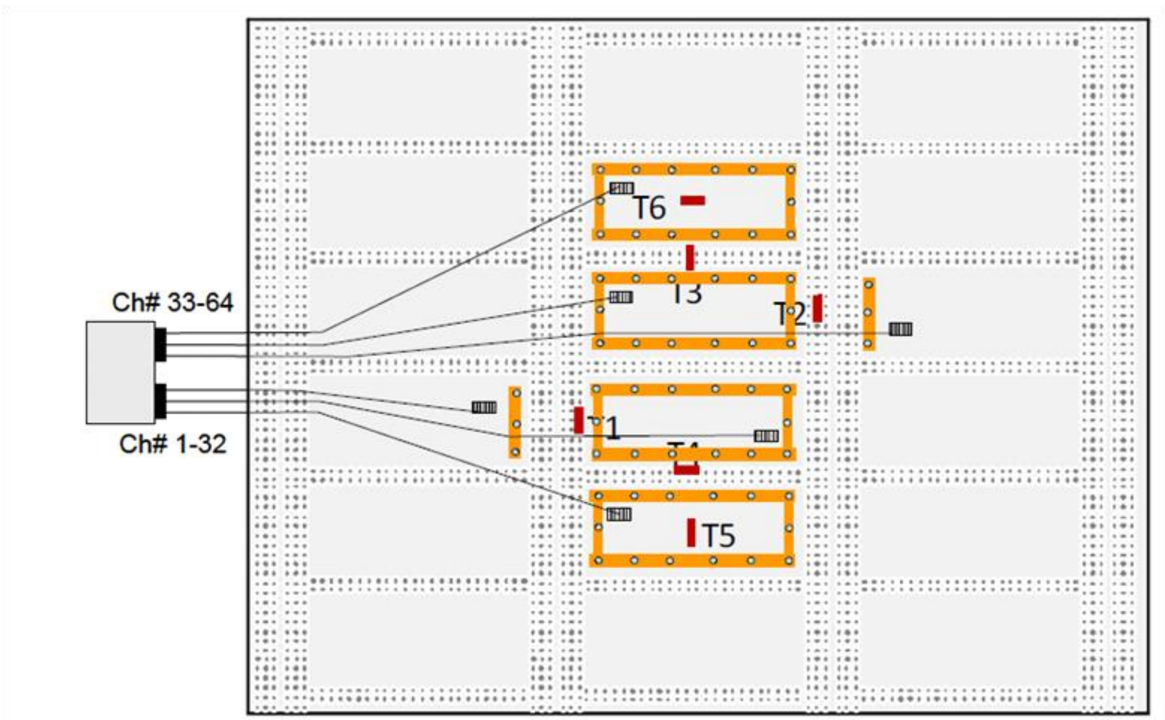


Figure 35. Four ring-shaped SMART Layers and Two Strips for Installation.

3.7.2.3 Sensor Installation

Before installing the sensor layers, the surface of the structure was first cleaned using a solvent. Then the surface was slightly roughened with sandpaper, and wiped down to clean the dust away.

A two-part epoxy adhesive (EA 9394) was used to bond the SMART Layers on the surface of the structure. The rigid interface between the piezoelectric elements and the structure provide the mechanical coupling needed to transmit strain between the actuator/sensor and the structure. Heat lamps were used to heat the structure to accelerate the curing of the epoxy. But the structure was fairly large and acted as a big heat sink. Therefore, we let the epoxy cure overnight before starting the tests the next morning. The installed SMART Layers are shown in Figure 36.

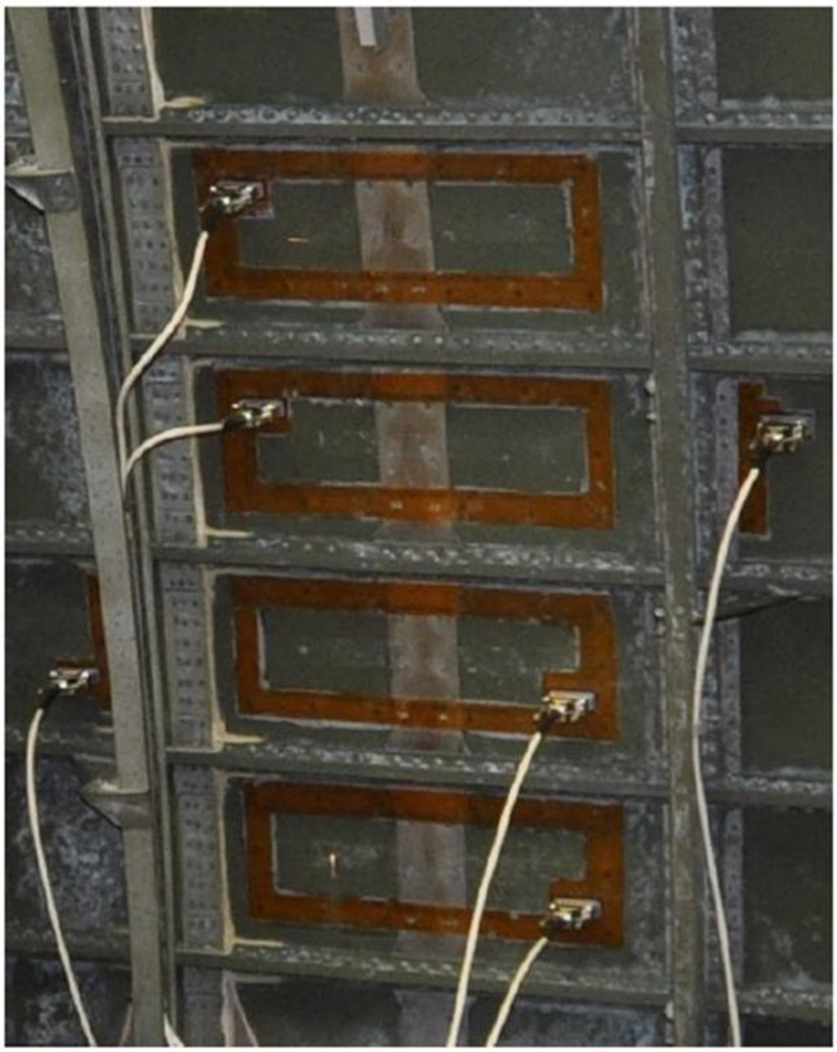


Figure 36. Installed SMART Layers

Figure 37. Close up View of Sensors

3.7.2.4 Software Configuration and System Calibration

Software was configured to send/receive signal data from the actuator-sensor paths shown in Figure 38. The white dots represent the piezoelectric sensors embedded on the strips and the red lines represent acoustic signal paths between each sensor.

The actuator input that is used to excite the piezoelectric elements is a 5-peak modulated sine wave burst (Figure 39). Signals were generated at 250 kHz and 350 kHz for each path, and a typical sensor response is shown in Figure 40.

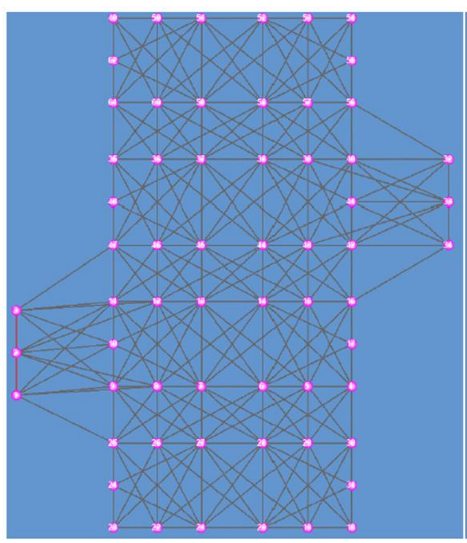


Figure 38. Actuator-Sensor Paths.

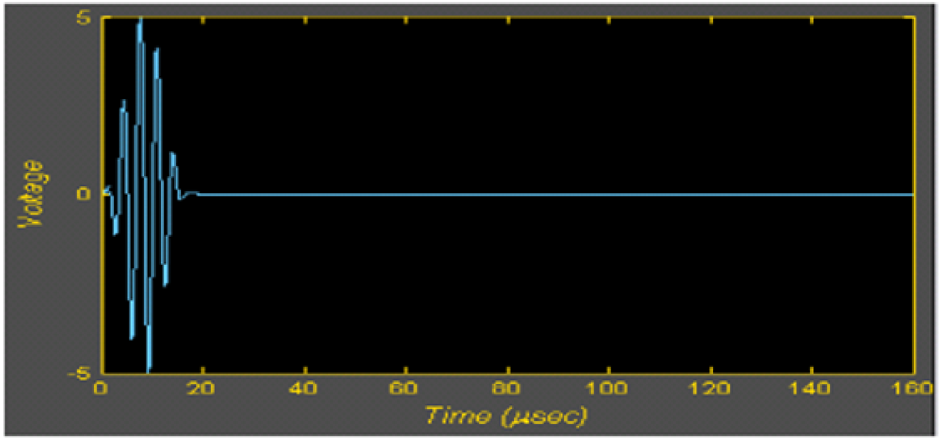


Figure 39. 5-peak modulated sine wave burst.

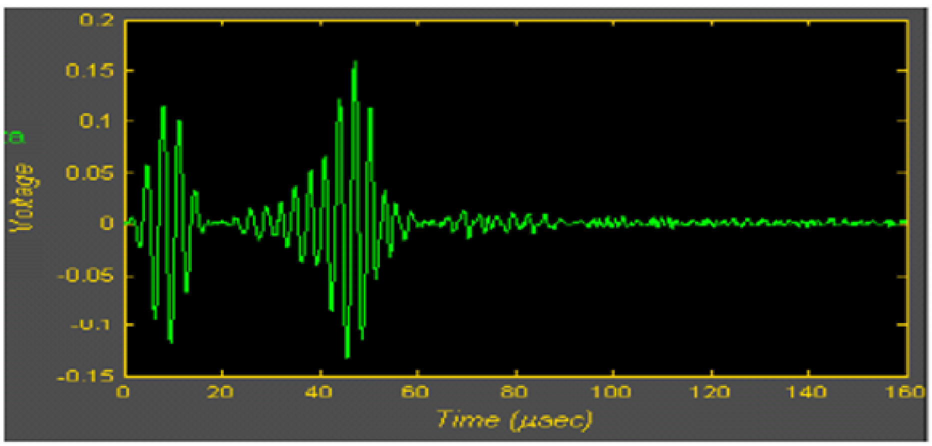


Figure 40. Sensor Response from a 250 kHz Actuation Signal

Prior to inducing damage, baseline data was collected at several temperatures from all actuator-sensor paths to calibrate the system for temperature effects.

3.7.2.5 Test Procedures and Results

Artificial cracks in the form of small notches were introduced using a Dremel tool. The notches were cut through the skin in different locations and orientations. Figure 41 shows the flaw types and orientations. For example, types T1 and T2 are circumferential cracks between fasteners under the frames, T3 is a circumferential crack between fasteners under a stringer, T4 is an axial crack under a stringer, T5 is a circumferential crack within a bay, and T6 is an axial crack within a bay. During the testing, 12 artificial cracks of the types T1 through T6 were introduced at various locations.

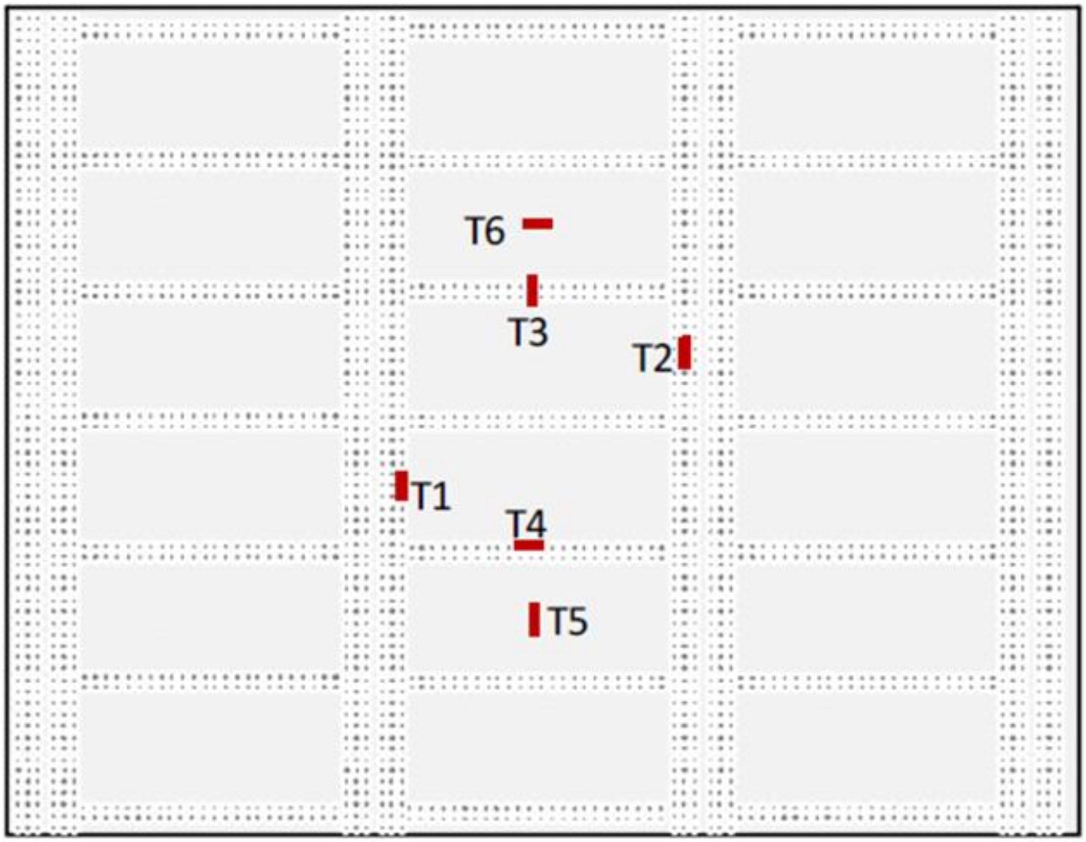


Figure 41. Artificial crack types T1 through T6.

After each flaw was induced, data was collected from the sensor network, and the diagnostic process was invoked to demonstrate the damage detection capabilities. The methodologies used in the diagnostic process are based on comparing the current sensor responses to the previously recorded baseline sensor responses from the undamaged structure. The differences between the two sets of signals are what contain the information about any existing damage or other anomalies. In this study, two variations of the diagnostic process are being used. The first method determines the total signal energy in each actuator-sensor path and generates a 2-D diagnostic image. This method, referred to as Direct Path Image, provides a quick visual representation of the location of structural changes and was used during the demonstration.

The second method used utilizes the wave velocities of the s0 and a0 Lamb wave modes in each actuator-sensor path to extract the reflections in each signal to generate a diagnostic image. This technique, referred to as Reflection-Based Analysis (RBA), produces more accurate visual representations of damage for single cracks, but has difficulties focusing when there are multiple damages located near each other because of the multiple reflections.

s0 Mode - The zero-order symmetrical mode (designated s0) travels at the "plate velocity" in the low-frequency regime where it is properly called the "extensional mode". In this regime the plate stretches in the direction of propagation and contracts correspondingly in the thickness direction. As the frequency increases and the wavelength becomes comparable with the plate thickness, curving of the plate starts to have a significant influence on its effective stiffness. The phase velocity drops smoothly while the group velocity drops somewhat precipitously towards a minimum. At higher frequencies yet, both the phase velocity and the group velocity converge towards the Rayleigh wave velocity.

a0 Mode - The zero-order anti-symmetric mode (designated a0) is highly dispersive in the low frequency regime where it is properly called the "flexural mode". For very low frequencies (very thin plates) the phase and group velocities are both proportional to the square root of the frequency; the group velocity is twice the phase velocity. This simple relationship is a consequence of the stiffness/thickness relationship for thin plates in bending. At higher frequencies where the wavelength is no longer much greater than the plate thickness, these relationships break down. The phase velocity rises less and less quickly and converges towards the Rayleigh wave velocity in the high frequency limit. The group velocity passes through a maximum, a little faster than the shear wave velocity, when the wavelength is approximately equal to the plate thickness. It then converges, from above; to the Rayleigh wave velocity in the high frequency limit.

Prior to inducing damage, a diagnostic image was generated showing no damage (all blue) as can be seen in Figure 42. The white dots represent the piezoelectric sensors.

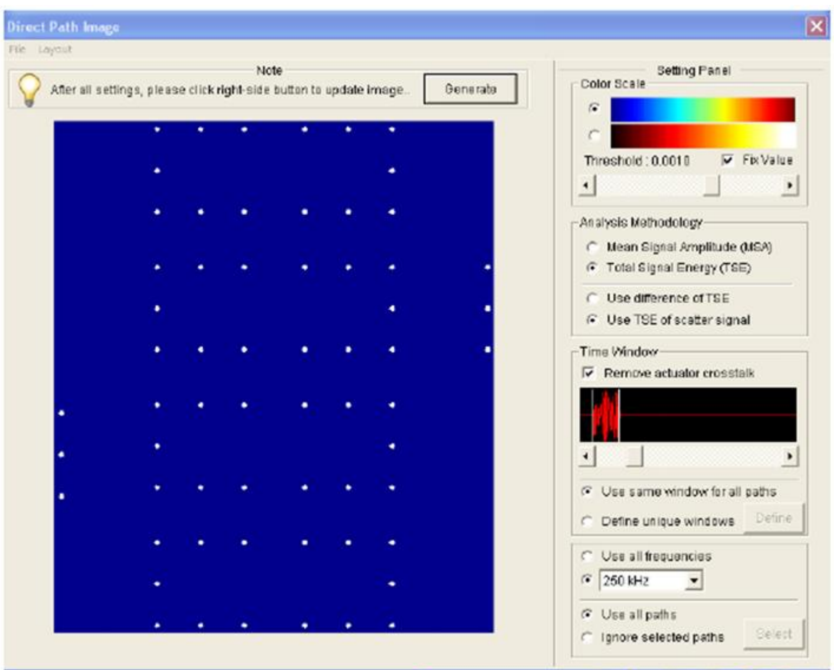


Figure 42. Diagnostic Image with no Damage (all blue)

The first damage that was introduced was a type T5 located on the left hand side of the bottom bay. Using the quick Direct Path Image method, the damage was clearly detected, but the exact location was a few inches off as shown in Figure 43 (Note: The actual locations of the damages are indicated by a yellow vertical line for circumferential cracks and a yellow horizontal line for axial cracks). Using the RBA method, the damage location is much more accurate (Figure 44).

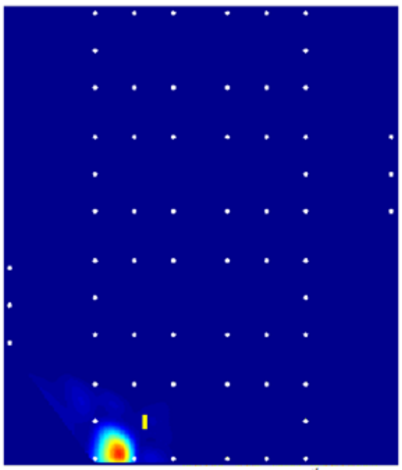


Figure 43. Direct Path Image of T5 Type Damage

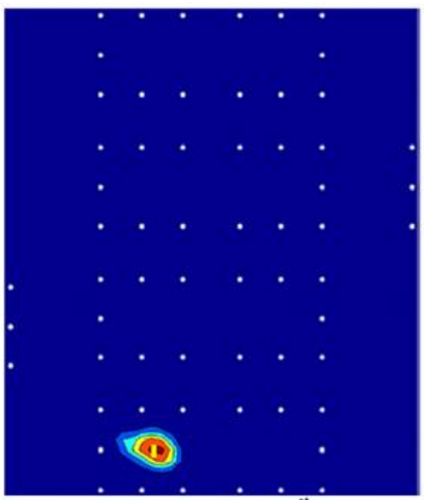
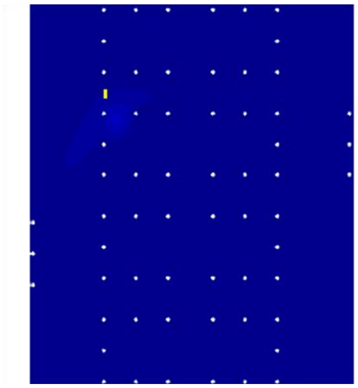
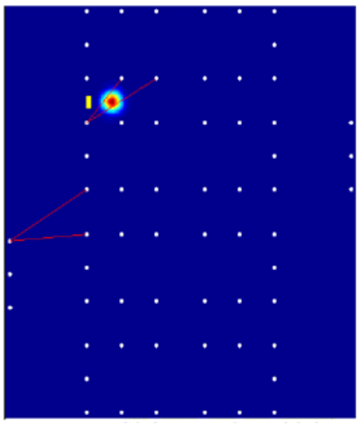
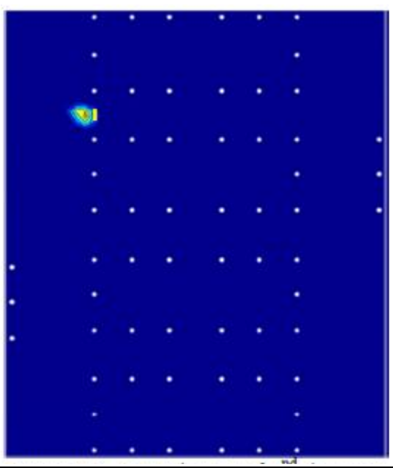


Figure 44. RBA Image of T5 Type Damage

The second damage introduced was a type T3 located on the edge of the sensor network directly in line on the path between two sensors. Since the crack is co-linear with the actuator-sensor path going directly through it, the sensor signal for that path is relatively unaffected, and the resulting Direct Path Image does not show the damage (Figure 45). If the sensitivity is increased, then the Direct Path Image can detect the damage (Figure 46), but the increased sensitivity can lead to false positives. In Figure 46, the red lines show the actuator-sensor paths that have indications of damage. Note that at this high sensitivity, the two paths going across the frame on the left hand side show indications of damage where there are none. However, using the RBA method (Figure 47), the damage can be detected and located without having to increase the sensitivity.

	**Figure 45. Direct Path Image of T3 Type Damage**
	**Figure 46. - Direct Path Image of T3 Type Damage With Increased Sensitivity**
	**Figure 47. RBA Image of T3 Type Damage**

Using the Direct Path Image, multiple damages can be detected and imaged as shown in Figure 48 with the 2nd and 3rd damage, which are both type T3, and Figure 49 with the 2nd through 6th damages shown.

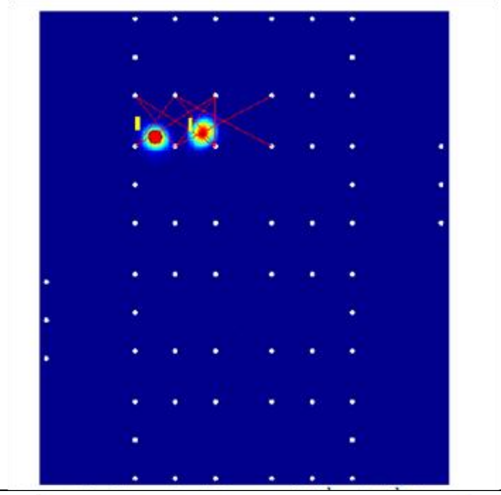


Figure 48. Direct Path Image of Two (2) Type 3 Damage Locations

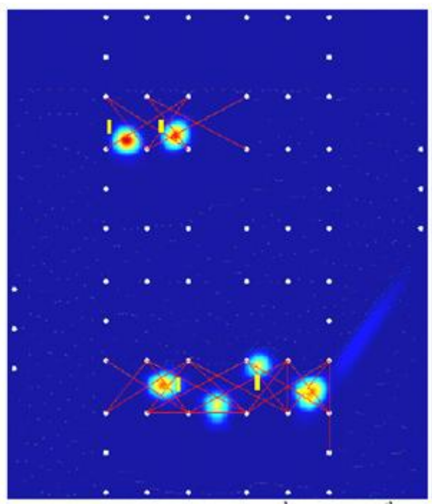


Figure 49. Direct Path Image of Multiple Damage Locations

The RBA method can be used to detect and distinguish multiple damages if the distance between the damages is greater than the length of the actuator-sensor paths, and can provide accurate detection and location of isolated damages as shown in Figure 50 of the fourth damage, which is type T4.

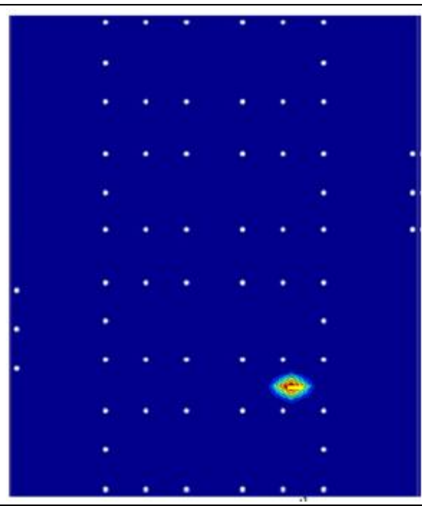


Figure 50. RBA Iimage of Type 4 Damage

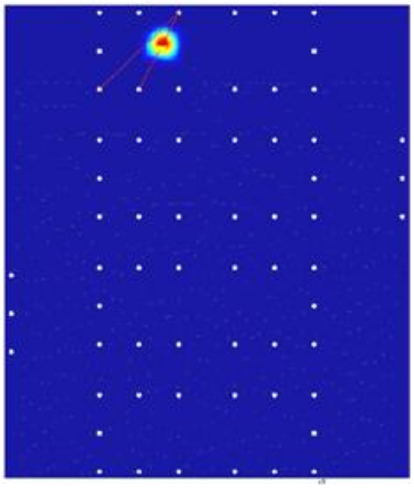


Figure 51. Direct Path Image of Type 6 Damage

The seventh damage, which is a type T6, was detected and accurately located by both the Direct Path Image and RBA methods (Figures 51 and 52). The Direct Path Image of the eighth and ninth damages (type T1 and T2) are shown in Figures 53 and 54. The Direct Path Image method is not able to detect the ninth damage (Figure 54) because the sensor signals passing through the damage must go through the gap between the two adjacent skin panels, making the signals relatively weak. But since the RBA method relies on reflections from the damage and not the through-path signals, the RBA method can detect and locate the crack (Figure 55).

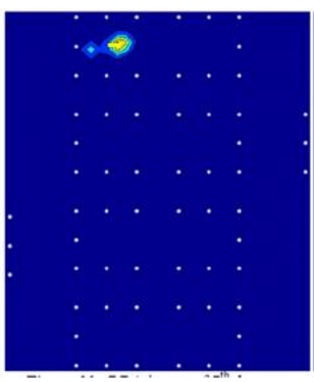


Figure 52. RBA Image of Type 6 Damage

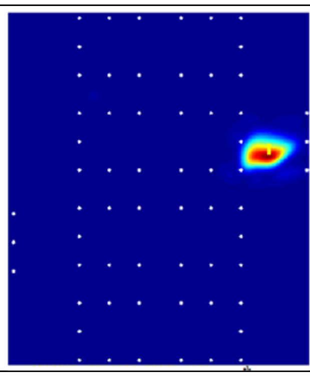
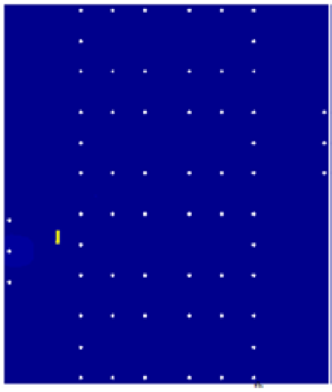
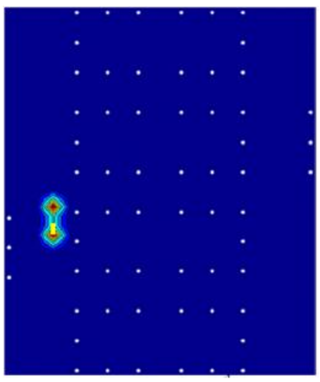
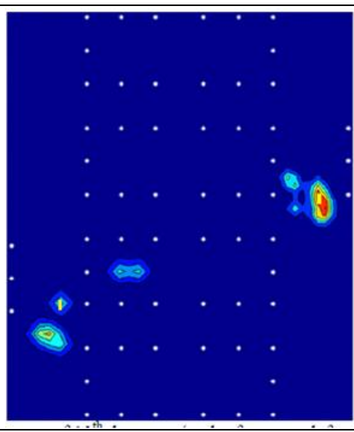
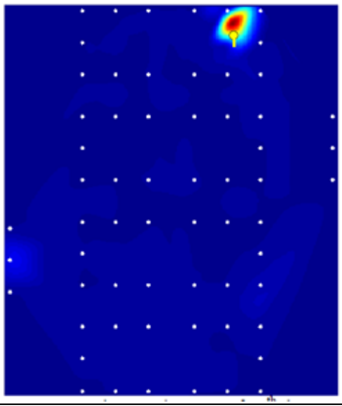
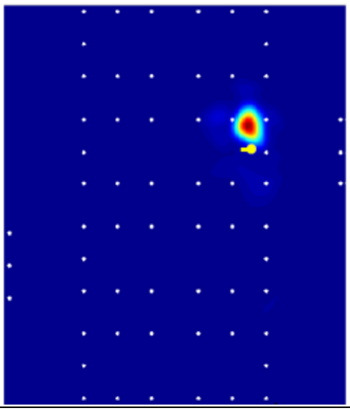
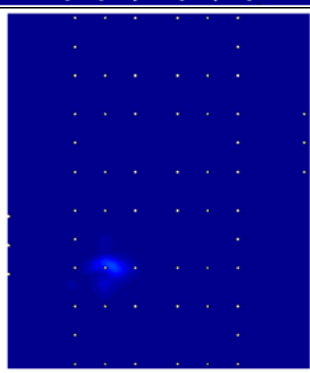


Figure 53. Direct Path Image of Type 1 Damage

	**Figure 54. Direct Path Image of Type 2 Damage**
	**Figure 55. RBA Image of Type 2 Damage**

Both of the tenth damages were induced at the same time and were also beneath the frames on the left and right hand side (similar to the eighth and ninth damages). The RBA method was used to detect both damages as shown in Figure 56.

	**Figure 56. RBA Image of Type 1 and Type 2 Damage (Under Frames at Left and Right Hand Side)**

	**Figure 57. Direct Path Image of Diamond Wire Cut**
	**Figure 58. Direct Path Image of Diamond Wire Cut**
	**Figure 59. Generated Diagnostic Image with Wet Batting Touching Surface**

The eleventh and twelfth damages were created using a small drill to first put a hole, and then the crack was grown using a diamond wire. The direct path images are shown in Figures 57 and 58.

Tests were also conducted to evaluate the effects of wet batting pressed against the surface of the undamaged structure. We were concerned that wet batting in contact with the skin would attenuate the signals and affect the performance of the Acellent System resulting in an indication of damage when no damage was present. It was discovered that the effects on the sensor signals were much smaller than those caused by damage. This reduced that concern that wet batting would introduce a false alarm or ambiguous detection results. A direct path image of the structure with the wet bating pressed against the surface is shown in Figure 59.

3.7.2.6 Summary of Acellent Testing

The purpose of the testing was to demonstrate the crack detection capability of Acellent's SMART Layer SHM system. For the demonstration, a Dremel tool was used to introduce artificial cracks in the structure. Two data analysis techniques were used to analyze the data and generate diagnostic images. The first method (Direct Path Image) determines the total signal energy in each actuator-sensor path and generates a 2-D diagnostic image. The second method (Reflection-Based Analysis) utilizes the wave velocities of the S0 and A0 Lamb wave modes in each actuator-sensor path to extract the reflections in each signal. The Direct Path Image method had difficulties detecting damage on the edges of the sensor layout, but could detect all damages within the sensor layout. The Reflection-Based Analysis could reliably detect and locate the cracks everywhere, including the edges of the sensor layout, but has difficulties imaging multiple cracks that are located near each other. For a production system, it is recommended to combine the two techniques, and develop a reasoning algorithm to give more weight to the Reflection-Based Analysis unless there are multiple cracks detected next to each other.

3.7.3 JENTEK MWM Array Evaluation

3.7.3.1 Procedure and Sensor Installation Instructions

For this C-5A aft crown evaluation, two existing MWM sensor configurations were used: the single sense-element FS36 MWM (Figure 60) and the wide-area, 36-sense-element, FA120 MWM-Array (Figure 61).

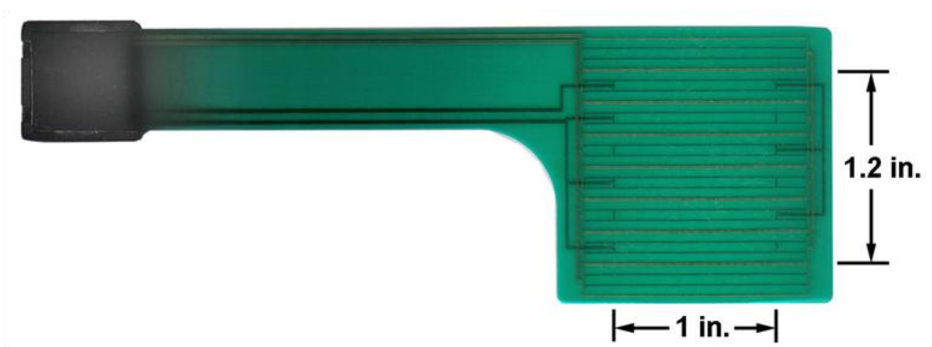


Figure 60. Single-Sensing Element FS36 MWM

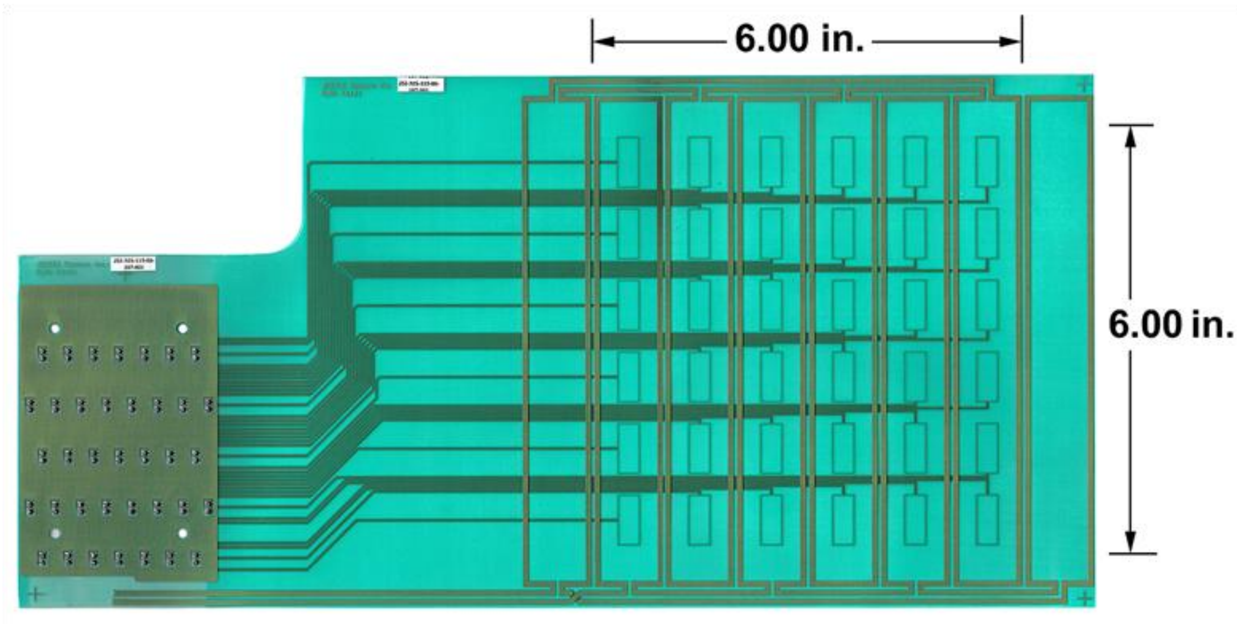


Figure 61. Wide-Area 36-Sensing Element FA120 MWM-Array

The damage was introduced by grinding an approximately 1-in. long notch through the skin of the aft crown with a Dremel tool. Also, only a pair of FS36 MWM sensors were mounted for the environmental cycling portion of the demonstration. All other data was acquired before and after notching by holding the sensors, by hand, against the inside surface of the aft crown skin. Figure 62 shows a photograph and schematic of the section of aft crown instrumented by JENTEK and the location of the notches. Photographs of notches T5, T6, T3, T31, T11, T12, and T61 can be found in Figure 63.

The GridStation System employed for the data acquisition consisted of a modified 7-channel impedance instrument configured to operate 2 single-channel probes with FS36 MWM sensors. The GridStation System was connected, in turn, to a host laptop computer operating the GridStation software environment for instrument ranging and system calibration. Calibration was performed in air without any reference standards. After calibration was complete, the ranging and calibration parameters for a system were stored on the laptop for later recall during data acquisition.

To simulate a periodic inspection of the aircraft, data acquisition for the demonstration consisted of three sets of measurements acquired before and after introduction of the notches at the location to be notched and at a neighboring location (see Figure 63). The data acquired over the neighboring locations represents, a neighboring sensing element of the planned on-aircraft sensing network that does not experience any cracking. These neighboring locations also provide a measure of the noise introduced by these "pick-and-place" measurements.

For the environmental cycling portion of the test, two FS36 sensors were mounted to the inside surface of the skin of the aft crown section (see the lower left photograph of Figure

63). Preparation of the skin surface consisted of cleaning the locations for mounting, as well as the surface of the sensors, with isopropyl alcohol. 3M VHB adhesive transfer tape (F9460PC – 0.002-in. thick) was then cut to size and applied to the sensing side of the FS36s which were then adhered to the inside of the skin.

VHB was used for the laboratory demonstration because it forms a "semi-permanent" bond that allows the sensors to be removed, without damaging them, at the conclusion of the test. For on-aircraft installation, a sealant would be used to adhere the network of sensors to the skin. For example. in related programs, Pro-Seal 870 was successfully tested with the Kapton version of an MWM-Array in a laboratory fatigue test conducted over 17 days and M-Coat J was successfully used to protect a Kapton version of an MWM-Array in a salt fog environment at an elevated temperature for 39 days.

3.7.3.2 DEMONSTRATION RESULTS

For the FS36, of the frequencies collected, 10 kHz provided the best signal-to-noise ratio (SNR). The SNR of interest compares the conductivity drop resulting from the introduction of an approximately 1-in. long, through-notch in the skin of the aft crown section to the range of conductivities produced from pick and-place measurements acquired over several locations on the inside surface of the skin. The pick-and place range is approximately ±0.4% while the conductivity drop from the introduction of the T5 and T6 notches is 6% and 5%, respectively, for a SNR of 12.5 or better.

As expected, the circumferential notch under a stringer, T3, was not detected. However, notch T31 simulated a circumferential crack extending from underneath a stringer and extended about 3/8 in. from the edge of the stringer. Although the conductivity change is much less than the 1-in. long notches of T5 and T6, the SNR is still greater than three.

The remaining FS36 data presented in Figure 64 was acquired over notches T11 and T12 which were placed in the skin beneath titanium straps (see Figure 4, top photograph) that span the center of the bays. It should also be noted that due to the presence of fasteners passing through the skin and the titanium straps, the FS36 had to be rotated 45° with respect to the crack from the preferred orientation. This did not affect the results as the measured conductivity drop over the notches was still 5% or greater. The SNR suffers over the straps because the range of pick-and-place data almost doubles to ±0.7%.

For the environmental cycling portion of the demonstration, one last notch, designated T61, was introduced into bay 2 of the aft crown section. T61 has the same orientation as T6 but was located lower in the bay to assist in the mounting of the two FS36s required for the environmental cycling. Again, data was acquired in a pick-and-place manner prior to the introduction of the notch. After notching, data was acquired periodically, once per minute, prior to heating, during heating, during cooling caused by the application of the water-soaked cotton batting, reheating, and second application of the cotton batting.

3.7.3.2.1 JENTEK PERFORMANCE ASSESSMENT

The demonstration for this program was designed to assess the capability of the MWM sensor technology to detect cracks (notches) with the sensor mounted on the inside skin surface of the C-5 aft crown. The two specific capabilities addressed and successfully demonstrated during the program were: (i) detection of through cracks in the aluminum skin with the sensor mounted on the internal surface of the aluminum skin, and (ii) detection of through cracks in the aluminum skin with the sensor mounted on the titanium alloy straps, where the cracks were buried below the straps and the sensor was mounted on the inside/exposed surface of the straps.

The SNR in each of these cases, for the crack sizes examined, was at least 5 to 1. It has been JENTEK's experience that with such a high SNR, the Probability of Detection (POD) performance would exceed 90% POD with over 95% confidence; and that filtering techniques can be used to improve the SNR. However, there is no accepted practice for POD determination using such permanently mounted sensors.

Note that JENTEK has a funded Air Force Phase II SBIR that is directly addressing this need for a method to generate such POD curves for permanently mounted sensors.

In addition to detection testing, we evaluated temperature and other environmental effects on sensor performance. We were again able to maintain a signal to noise of greater than 5 to 1 for all tested conditions. Thus, assuming we can normalize the data by a sensor in a similar location that is not seeing a crack – a likely and practical possibility, then we believe that a high POD is likely under typical operating conditions. Note that further post filtering to remove non-crack-like trends in the data is expected to provide substantial improvements in SNR performance.

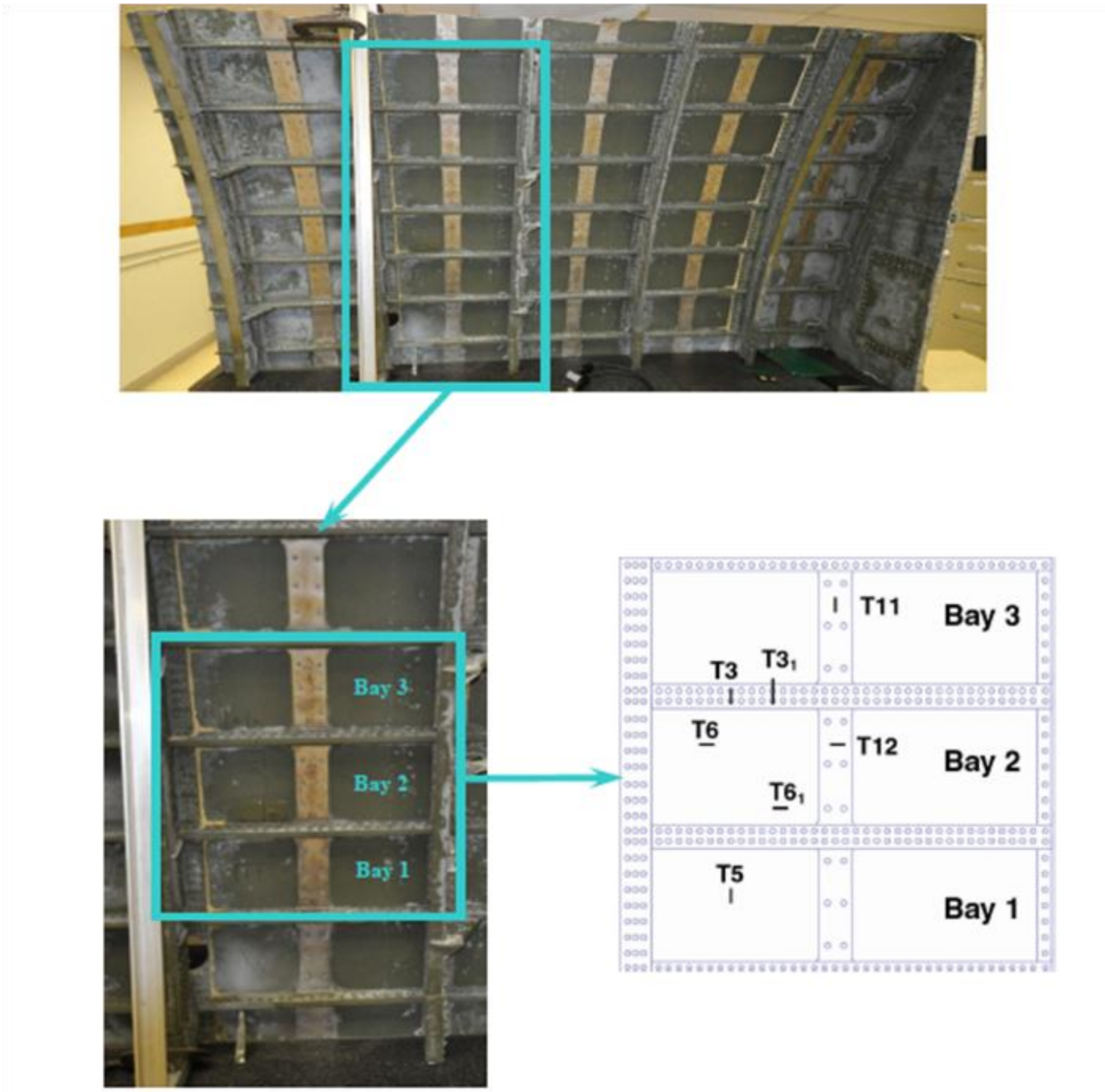


Figure 62. C-5A Aft Crown Instrumented by JENTEK

Figure 63. Notch Locations

Figure 63. Notch Locations:

- Bay 3 (top) contains notches T3, T31, and T11;
- Bay 2 (bottom of the top photograph and lower left) contains notches T6, T61, and T12; bay 1 (lower right) contains notch T5.

The mounted FS36 MWM pair for the environmental cycling portion of the test are shown in the lower left photograph in Figure 63.

Light from the heat lamps can be seen shining through the T61 notch and the channel 1 FS36. The black marker lines around the notches show where the FS36 was placed during data acquisition. For example, in the top photograph, before T3 was notched, data was acquired over T3 and T31 to serve as a comparison (and, vice versa when T31 was notched)

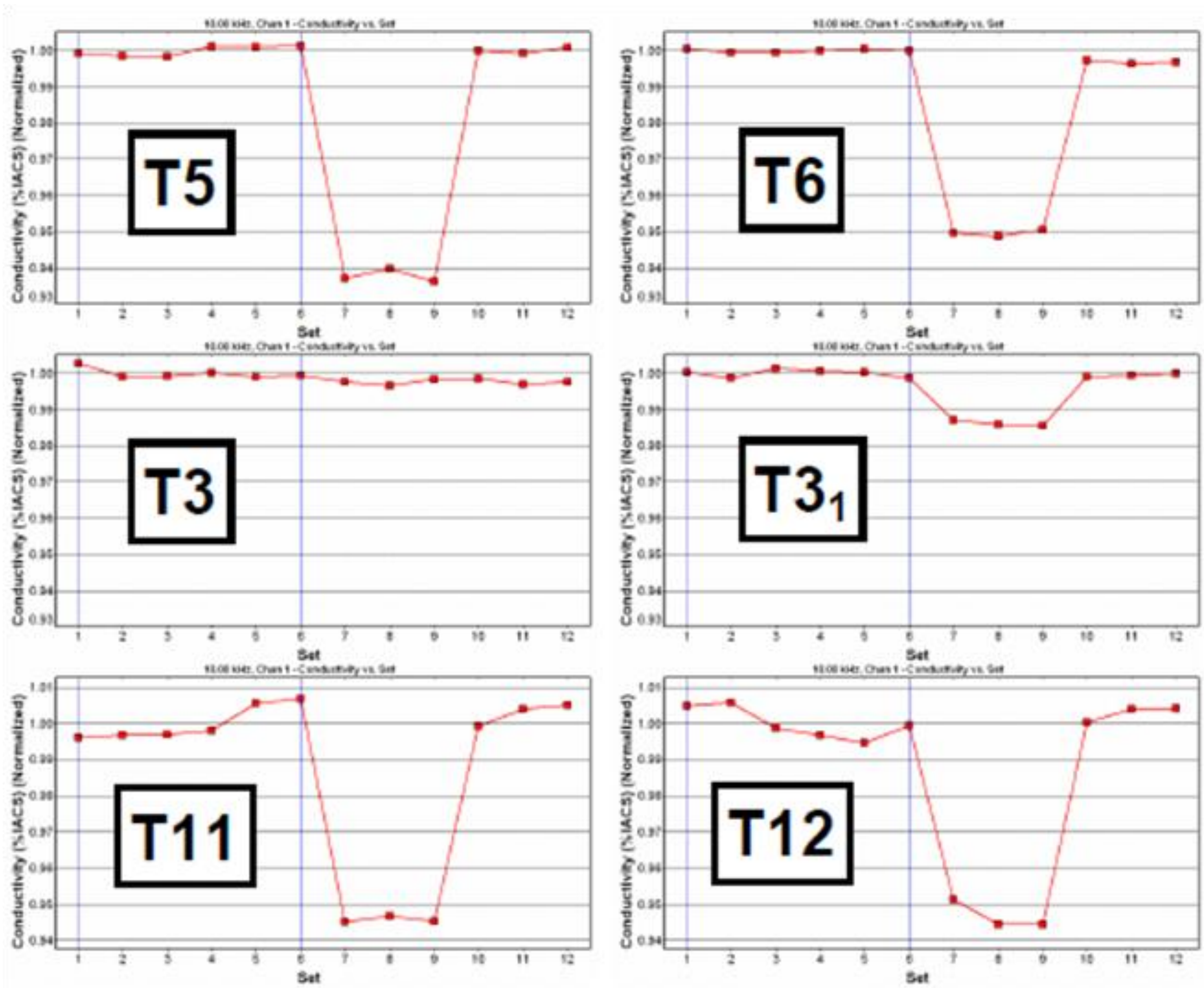


Figure 64. Normalized FS36 10 kHz Conductivity Data Simulating a Periodic Inspection for Six Notches in the Skin

Consult Figure 62 for the location and orientation of each notch. In each graph (Figure 64), 12 sets of data, depicted as the red dots on each graph, are presented that simulate a periodic inspection: prior to notching, 3 sets were acquired over the notch location and 3 sets were acquired over a neighboring location and then after notching, the sequence was repeated. In each figure, sets 1-6 and 10-12 provide a sense of the error introduced from pick-and-place measurements at different locations and the drop in conductivity of sets 7-9 provides the signal due to the notch. Effectively points 1 through 6 represents the state of the structure prior to insertion of a defect (notch) and points 7 through 12 represent the state of the structure after insertion of a defect (notch). The data clearly illustrates that JENTEK was able to detect type 5, 6, 11 and 12 defects with little difficulty. Type 3 was not detected and Type 3_1 was marginally detected.

3.7.4 Impact Technologies CBM System Mock-Up

Impact Technologies developed a simulation to illustrate maintenance ConOps using a portable maintenance device (laptop). The simulation supports various implementation approaches and fault scenarios and provides information visually. For manual scenarios the simulation provides information supporting manual (Visual and MOI) inspection. The display is modeled from the Aft Crown graphics current in AFIRM. A production implementation would allow the maintenance lap top to interface with AFIRM and allow presentation of data in AFIRM and updates of AFIRM based on manual inspection and SHM system data.

The simulation represents a possible user interface implementation that is driven by the sensor data provided by a sensor implementation that monitors 400 bays of the aft crown. A bay is the skin area bounded by stringers and frames. Each rectangle or cell represents a bay. The horizontal lines represent stringers and the vertical lines represent fuselage stations. The blue region in the screen shot on the left represents a critical area of the Aft Crown and the green a less critical area. The screen shot on the right identifies critical crack sizes with red representing areas where relatively small cracks a considered critical and the blue and green areas representing areas where relatively large cracks are not considered critical.

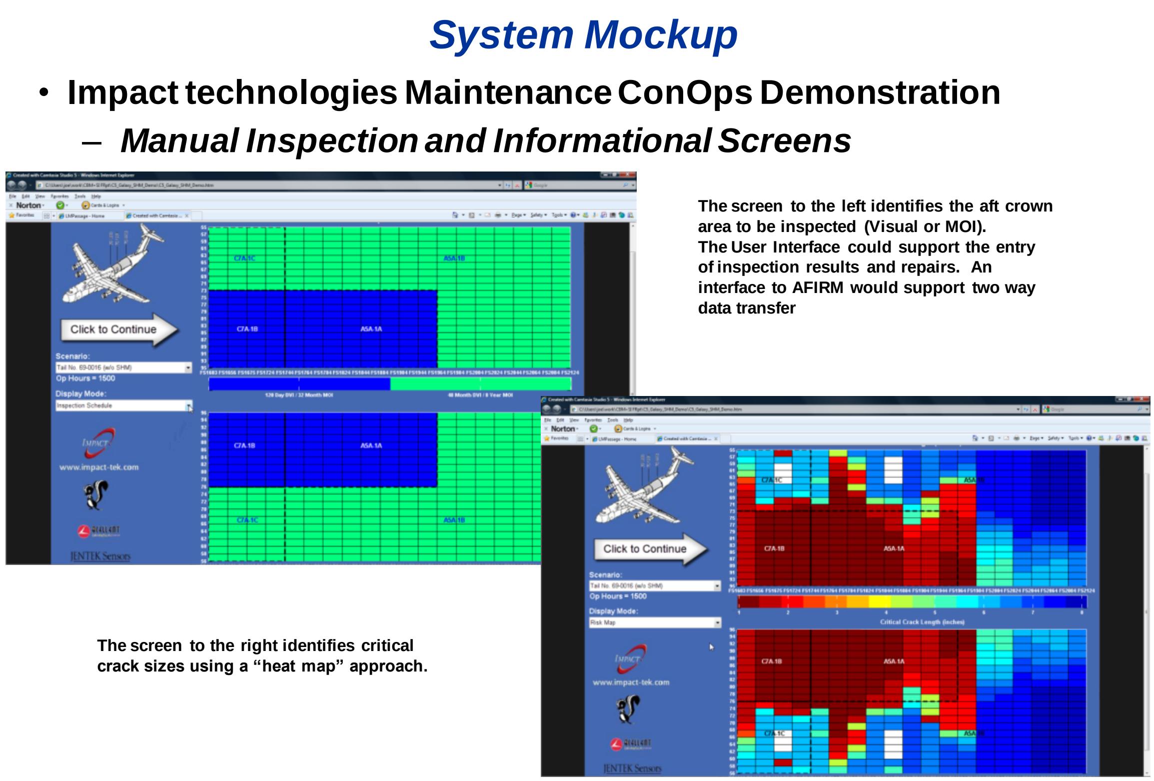


Figure 65. CBM System Mock Up – Inspection and Informational Screens

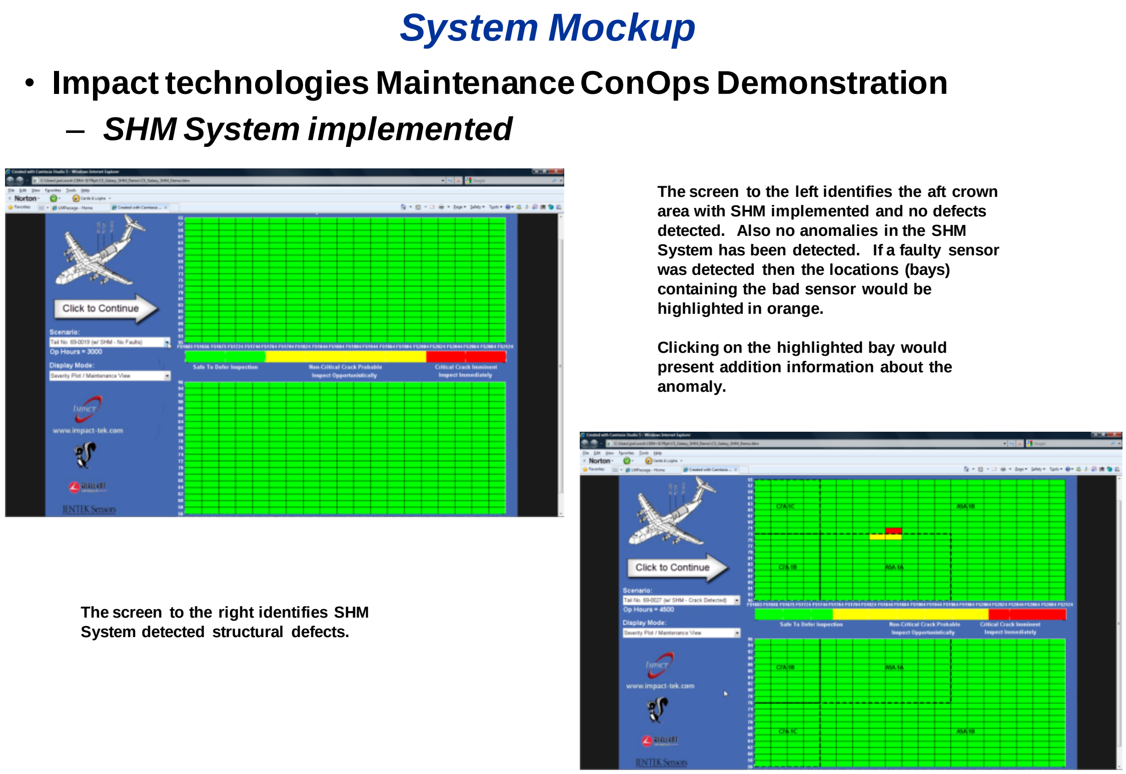


Figure 66. CBM System Mock Up – SHM System Implemented

Figure 66 illustrates the SHM System Health Report User Interface. The screen on the left illustrates the results with no anomalies in the SHM System or Structural Defects detected. The screen on the right illustrates a case where the SHM System detected defects. The locations of the defects are indicated in Red and Yellow.

3.7.5 Risk Analysis

In the final stage of the prototype development, a series of CBM+ strategies were evaluated based on structural risk (probability of fracture) calculations using a candidate SHM system (based roughly on the Acellent technology). Eight inspection protocols were defined with various combinations of visual and magneto-optic imaging (MOI) inspections and with a range of SHM system operation intervals. See Table 1. In order to support development and comparison of the various strategies, risk analyses were performed using the methods and data developed for the 2006 risk analysis of C-5A aft crown skins [3.5]. The calculations themselves were carried out using the Lockheed Martin developed computer program, SOPROF [3.6], which uses the same technical approach as, and has been validated against the USAF computer program, PROF V2 [3.7]. SOPROF is capable of analyzing inspection programs with mixed inspection types.

All of the scenarios considered were based on the 25 year period from 2015 to 2040. (The assumption was made that an SHM system could not be fielded before 2015.) The time variations of SFPoF over the 25 year period for the critical inspection zone (FS-1603 to FS-1964) were estimated using the same data that were used in the current (2006) risk assessment. At the time that this program was concluding, the risk analysis for the aft crown was being updated [3.8] based on a number of new and revised data sets, the most notable of which was the newly obtained residual strength data from National Institute for Aviation Research (NIAR) at Wichita State University [3.9].

Table 1. Inspection & SHM Strategies Considered for C-5A Aft Crown Skin Risk Analyses

no.	description
IP-1	120d GVI + 960d MOI
IP-2	120d GVI + 1460d MOI (baseline)
IP-3	960d MOI only
IP-4	1460d MOI only
IP-5	120d SHM + 1460d MOI
IP-6	365d SHM + 1460d MOI
IP-7	120d SHM only
IP-8	365d SHM only

The dependence of single flight probability of failure (SFPOF) on accumulated flight hours was calculated using the following input parameters. The fracture analysis was for a crack growing along a circumferential splice joint in two stages. In stage one, the crack grows from one fastener hole to the adjacent hole. In stage two, the crack grows between two fastener lines so that no benefit is taken for time to re-initiate after each ligament failure (this is a conservative assumption). The stress intensity factor (SIF) solution for this scenario is shown in Fig. 68. The statistical nature of the fracture toughness of the aft crown skin material (7079-T6 COS) was represented using a normal distribution with a mean of 62 ksi√in and a standard deviation of 6.2 ksi√in. See Fig. 69.

As is the standard practice for PROF, the stress exceedance curve was represented in the form of a Gumbel distribution with parameters A = 0.608 and B = 15.729. The stress exceedance diagram is shown in Fig. 70. In the 2006 risk analysis, a simplified, exponential representation of the crack growth behavior was used. The exponential curve shown in Fig. 71 was fit to a calculated fatigue crack growth curve which included both fatigue (cycle dependent) and stress corrosion (time dependent) components.

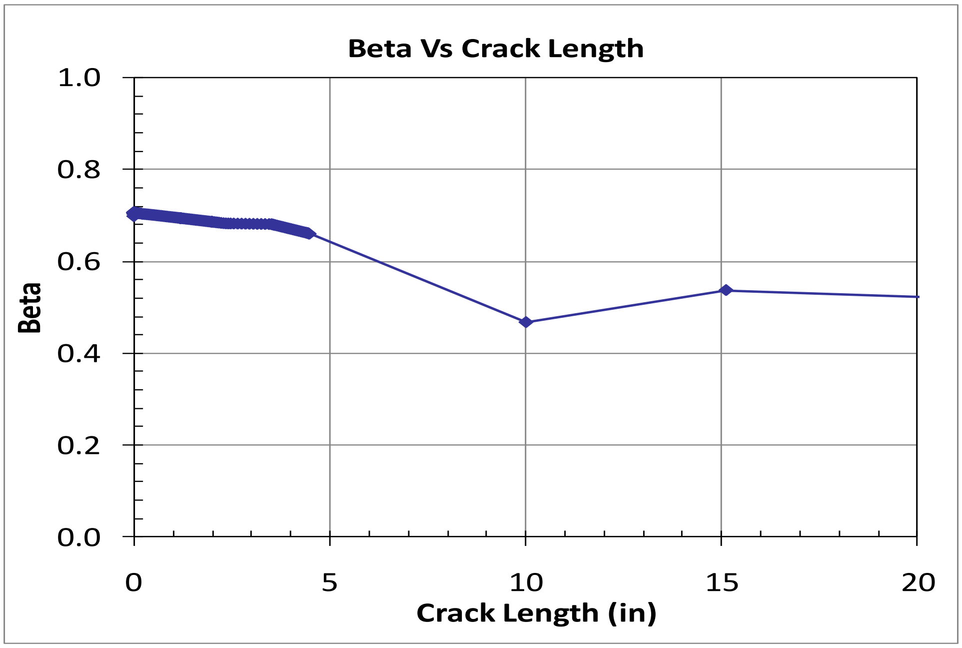


Figure 67. Stress Intensity Factor Solution Used for Aft Crown Risk Analyses

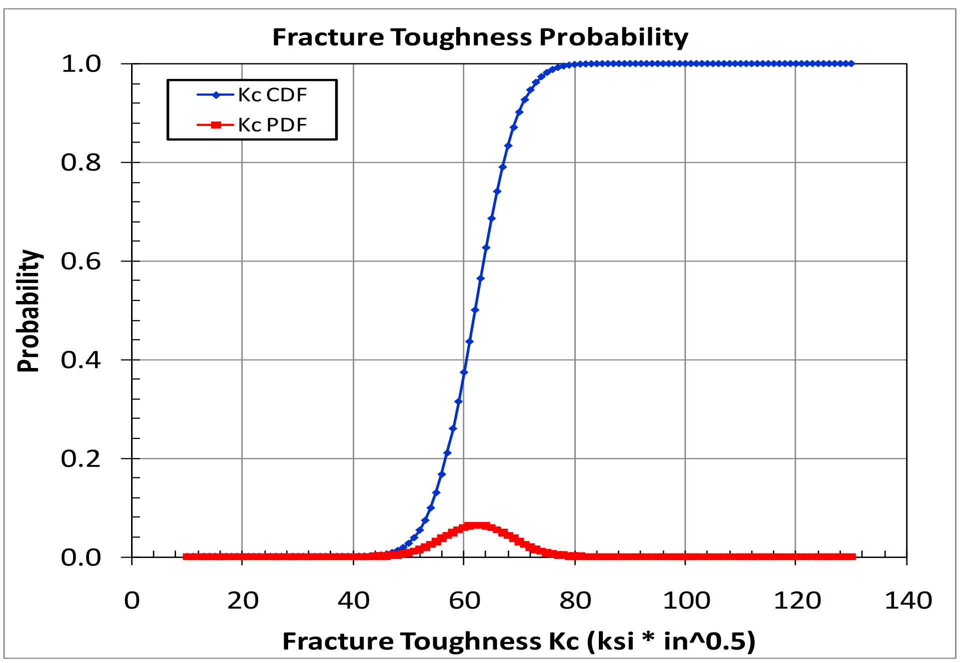


Figure 68. 7079-T6 COS Fracture Toughness

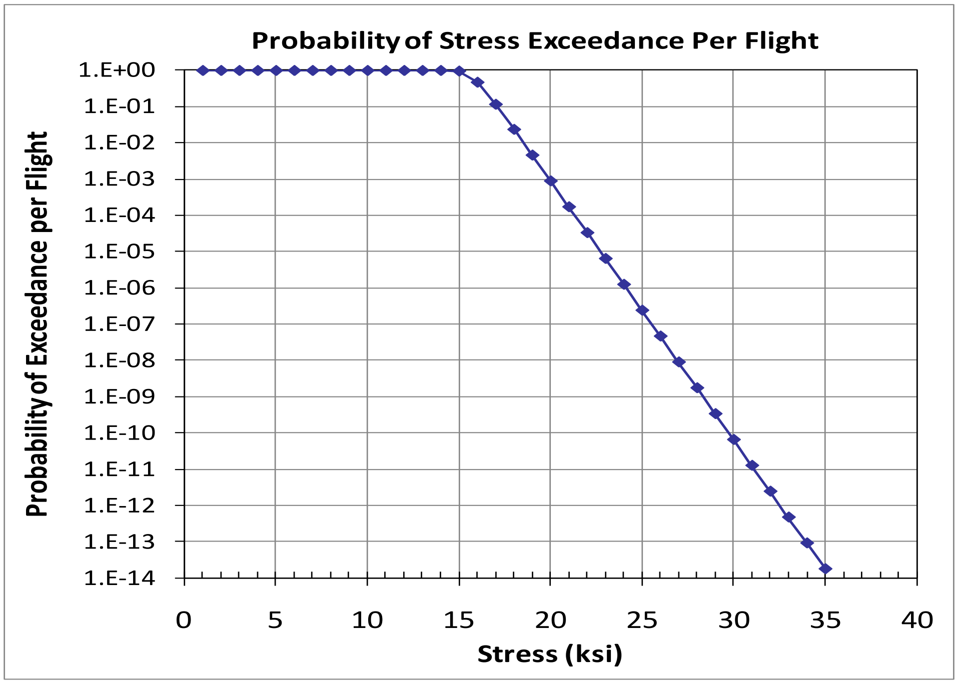


Figure 69 Probability of Stress Exceedance Per Flight

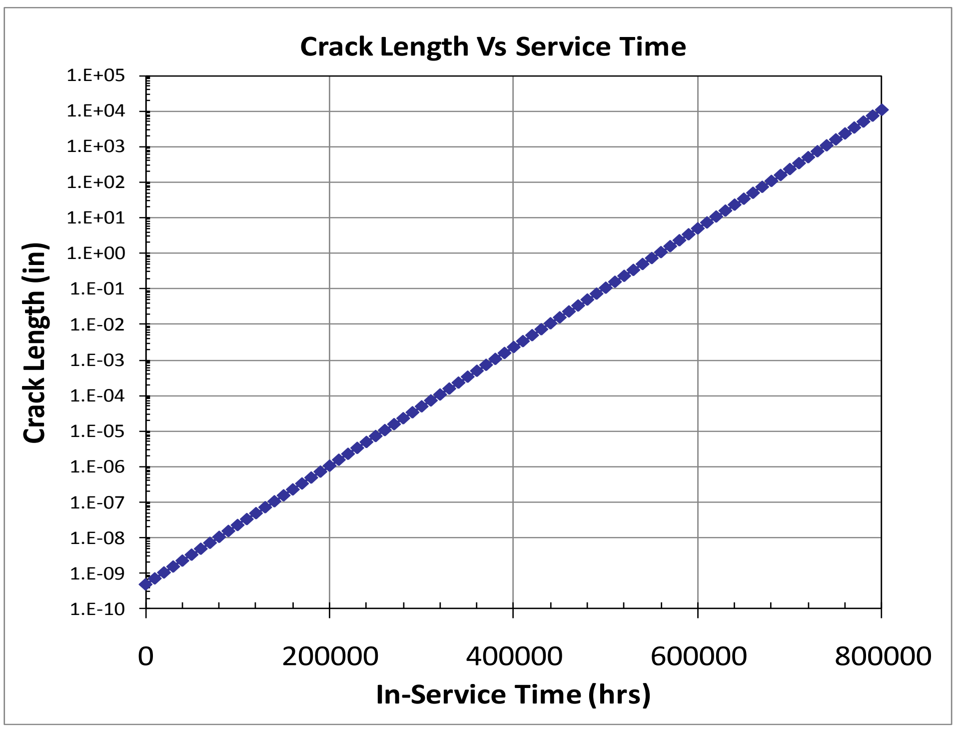


Figure 70. Crack Length vs Service Time

The assumed initial flaw size distribution (at zero hours) was developed with, and is consistent with the exponential crack growth curve. It is modeled using a Weibull distribution with shape and scale parameters of 0.8768 and 2.0987E-5 respectively. In order to be consistent with the 2006 risk assessment, the repair flaw size distribution was assumed to be uniform with a_{max} = 0.05 in.

The repairs are often done in the field in non-optimum working conditions. This is the rationale behind the more conservative choice of flaw size distributions for repairs as opposed to 'initial'.

Section 3.1.3.3 (WL-TR-91-3066,) provides guidance on Repair Flaw Size Distribution (RFSD). While we did not have access to the draft risk handbook that is currently being reviewed by the USAF, we understand that this uniform flaw distribution as well as an exponential distribution (Weibull with Sh=1) is still endorsed. As stated in WL-TR-91-3066, these RFSD are arbitrary. In our case, we did some sensitivity studies to show little effect of RFSD on SFPOF.

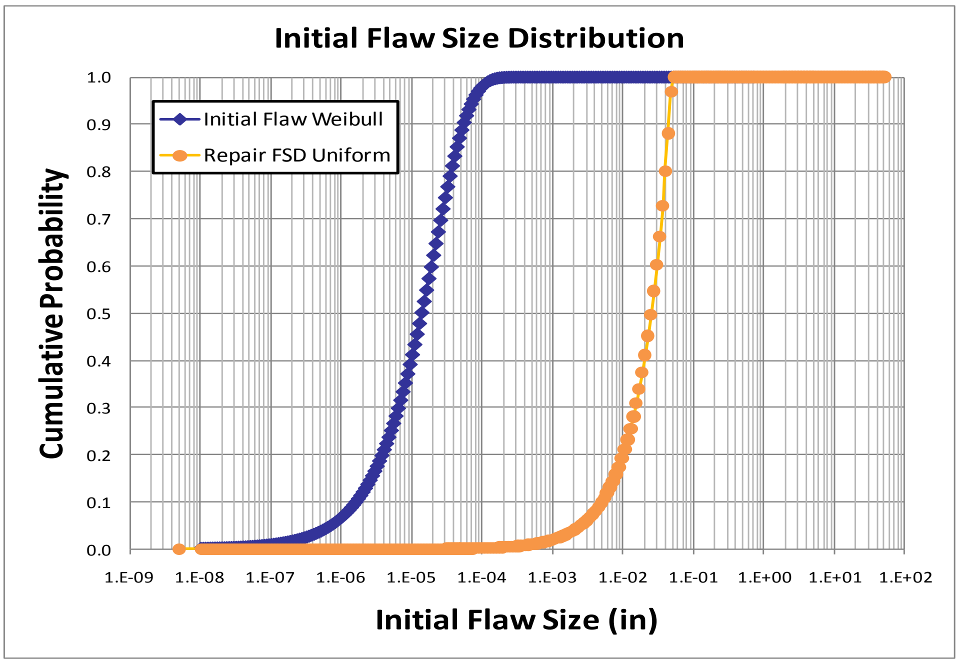


Figure 71. Initial and Repair Flaw Size Distributions

The probability of detection (POD) curves for general visual inspection (GVI) and magneto-optic imaging (MOI) were developed by the AFRL and provided to Lockheed Martin for the 2006 assessment. The POD for GVI was represented using a log odds distribution with a 50% crack size of $a_{50\%}$ = 1.5 in., std.dev. = 1.15 in., and a threshold of a_{th} = 0.25 in. The POD for MOI was also represented using a log odds distribution, in this case with a 50% flaw size of $a_{50\%}$ = 0.05876 in., std.dev. = 1.0 in., and a_{th} = 0.01 in. The cumulative probability of detection curves for these two inspection techniques are shown in Figure 72. In order to be consistent with current practice, the effects of probability of inspection (POI) being less than one were included. While there is no consensus on appropriate values for POI, the values assumed for the present analyses were 0.5 for GVI and 1.0 for MOI.

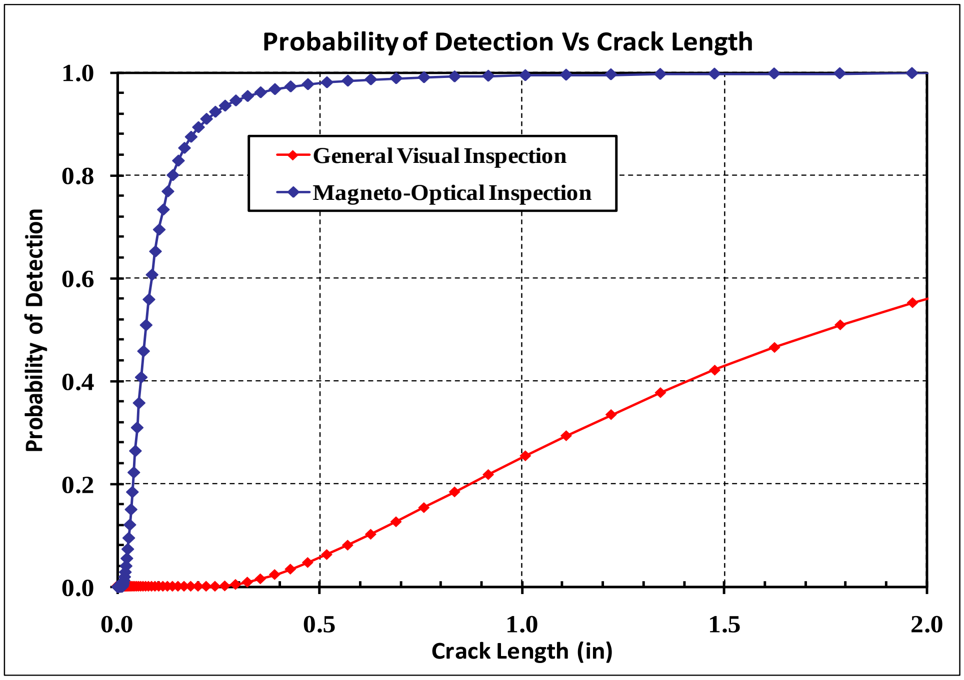


Figure 72. Probability of Detection vs Crack Length

All of the inspection scenarios considered in this study started with the same flaw growth and inspection program, which was based on the following assumptions:

- The aircraft entered service Jan. 1970
- 450d GVI w POI=0.5 started in 1995
- 105d GVI w POI=0.5 + 840d MOI w POI=1 started in 2003
- 120d GVI w POI=0.5 + 960d MOI w POI=1 started in 2010
- The inspection program (IP) option being analyzed starts in 2015

The inspection history up to 19570 flight hours is shown in Table 2. From that point forward, the inspection programs (IP) varied according to the descriptions given in Table 1. The first inspection program is the one currently in place for the management of the aft crown skins, namely GVI every 120 days with an MOI at 32 month intervals. The second program is considered the baseline since it represents what typically happens in practice (which is to defer the MOI to 48 months (major) and accept an 80% flight restriction during the period between 32 and 48 months). The third and fourth options consider the usage of MOI alone at 32 and 48 month intervals. Options 5 through 8 consider either SHM alone, or SHM in combination with MOI.

Table 2. Assumed Inspection History for C-5A Aft Crown Skin (Critical Zone) for First 19570 Flight Hours

Date	Interval (days)	Days	Months	Years	Flight Hours	Analysis Point	Calendar Hours	Inspect. Type	POI
1/1/1970		0	0	0	0				
7/3/1995	9314	9314.0	306.0	25.50	11475.2	1	223536	GVI	0.5
8/26/1996	420	9734.0	319.8	26.65	11992.6	2	233616	GVI	0.5
10/20/1997	420	10154.0	333.6	27.80	12510.1	3	243696	GVI	0.5
12/14/1998	420	10574.0	347.4	28.95	13027.5	4	253776	GVI	0.5
2/7/2000	420	10994.0	361.2	30.10	13545.0	5	263856	GVI	0.5
4/2/2001	420	11414.0	375.0	31.25	14062.4	6	273936	GVI	0.5
5/27/2002	420	11834.0	388.8	32.40	14579.9	7	284016	GVI	0.5
7/21/2003	420	12254.0	402.6	33.55	15097.3	8	294096	GVI	0.5
11/3/2003	105	12359.0	406.0	33.84	15226.7	9	296616	MOI	1
2/16/2004	105	12464.0	409.5	34.12	15356.1	10	299136	GVI	0.5
5/31/2004	105	12569.0	412.9	34.41	15485.4	11	301656	GVI	0.5
9/13/2004	105	12674.0	416.4	34.70	15614.8	12	304176	GVI	0.5
12/27/2004	105	12779.0	419.8	34.99	15744.1	13	306696	GVI	0.5
4/11/2005	105	12884.0	423.3	35.27	15873.5	14	309216	GVI	0.5
7/25/2005	105	12989.0	426.7	35.56	16002.9	15	311736	GVI	0.5
11/7/2005	105	13094.0	430.2	35.85	16132.2	16	314256	GVI	0.5
2/20/2006	105	13199.0	433.6	36.14	16261.6	17	316776	MOI	1
6/5/2006	105	13304.0	437.1	36.42	16391.0	18	319296	GVI	0.5
9/18/2006	105	13409.0	440.5	36.71	16520.3	19	321816	GVI	0.5
1/1/2007	105	13514.0	444.0	37.00	16649.7	20	324336	GVI	0.5
4/16/2007	105	13619.0	447.4	37.29	16779.1	21	326856	GVI	0.5
7/30/2007	105	13724.0	450.9	37.57	16908.4	22	329376	GVI	0.5
11/12/2007	105	13829.0	454.3	37.86	17037.8	23	331896	GVI	0.5
2/25/2008	105	13934.0	457.8	38.15	17167.1	24	334416	GVI	0.5
6/9/2008	105	14039.0	461.2	38.44	17296.5	25	336936	MOI	1
9/22/2008	105	14144.0	464.7	38.72	17425.9	26	339456	GVI	0.5
1/5/2009	105	14249.0	468.1	39.01	17555.2	27	341976	GVI	0.5
4/20/2009	105	14354.0	471.6	39.30	17684.6	28	344496	GVI	0.5
8/3/2009	105	14459.0	475.0	39.59	17814.0	29	347016	GVI	0.5
11/16/2009	105	14564.0	478.5	39.87	17943.3	30	349536	GVI	0.5
3/16/2010	120	14684.0	482.4	40.20	18091.2	31	352416	GVI	0.5
7/14/2010	120	14804.0	486.4	40.53	18239.0	32	355296	GVI	0.5
11/11/2010	120	14924.0	490.3	40.86	18386.9	33	358176	MOI	1
3/11/2011	120	15044.0	494.3	41.19	18534.7	34	361056	GVI	0.5
7/9/2011	120	15164.0	498.2	41.52	18682.5	35	363936	GVI	0.5
11/6/2011	120	15284.0	502.1	41.85	18830.4	36	366816	GVI	0.5
3/5/2012	120	15404.0	506.1	42.17	18978.2	37	369696	GVI	0.5
7/3/2012	120	15524.0	510.0	42.50	19126.1	38	372576	GVI	0.5
10/31/2012	120	15644.0	514.0	42.83	19273.9	39	375456	GVI	0.5
2/28/2013	120	15764.0	517.9	43.16	19421.8	40	378336	GVI	0.5
6/28/2013	120	15884.0	521.9	43.49	19569.6	41	381216	MOI	1

In all of the risk analyses performed for this study, the results were normalized with respect to the maximum SFPOF value for the current (baseline) inspection program (i.e. the max SFPOF value from option 2). Thus, if the calculated normalized risk for any given proposed inspection / SHM program exceeded a value of 1.0, then the risk exceeded that of the current policy and would not be considered viable and/or acceptable. This threshold is shown as a red in the figures that follow.

The normalized risk vs. time for options 1 and 2 are shown in Figure 73. These results indicate that MOI, even at 48 month intervals, is very effective at managing risk. (This obviously is a direct result of the very high POD for MOI used in the analysis.) Comparisons of the normalized risk vs. time for options 3 and 4 against the baseline are shown in Figures. 74 and 75, respectively. In Figure 74 we see that MOI alone at 32 month intervals is nearly as effective at mitigating risk as the current baseline inspection program. (This is a significant finding because it has an impact on the business case.) However, as shown in Figure 75, MOI alone at 48 month intervals results in risk values that exceed those of the current baseline. Based on this analysis, IP-4 is not considered a viable option.

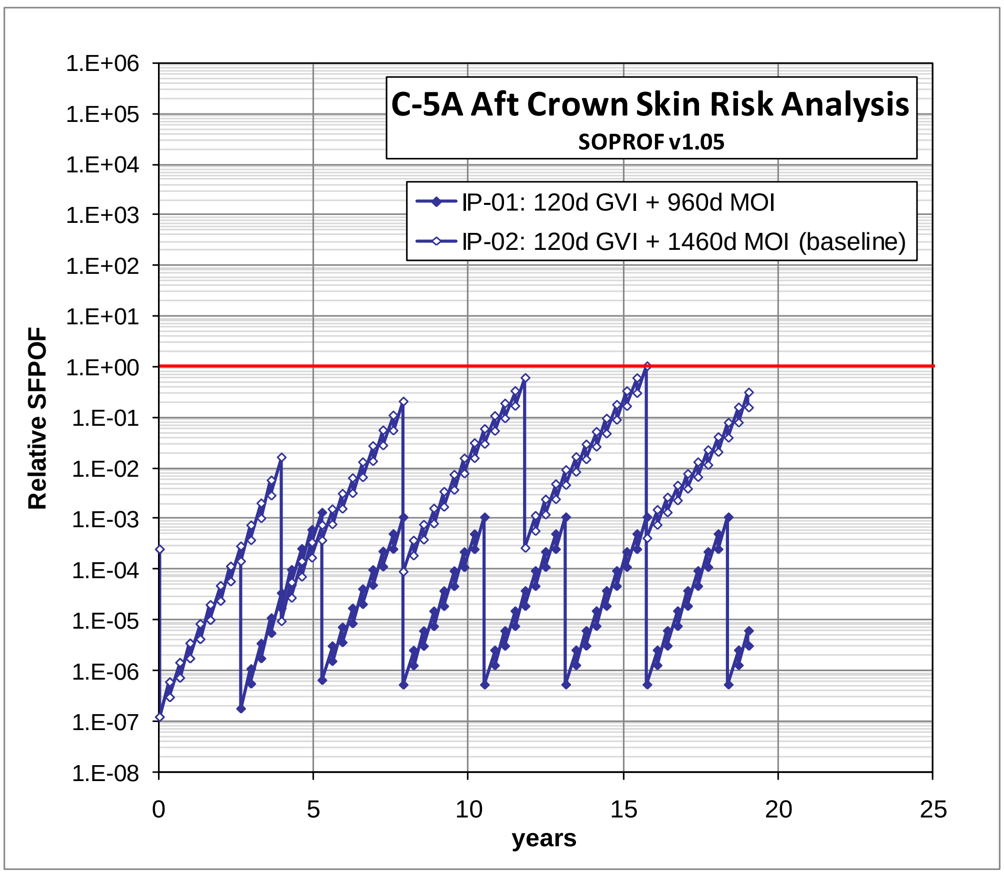


Figure 73. - Normalized SFPOF for Option 1 and Option 2 (baseline) Inspection Programs

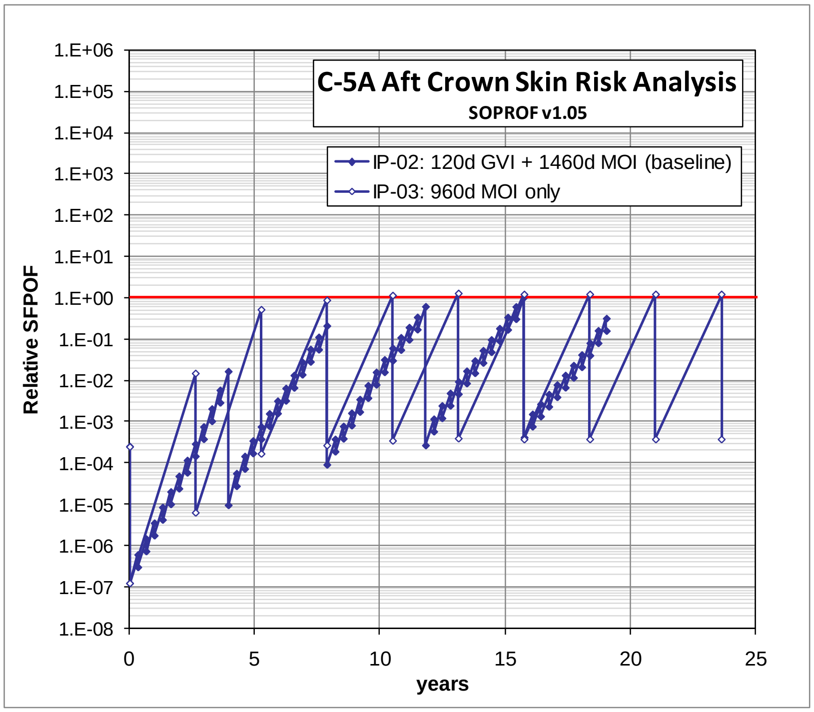


Figure 74. Normalized SFPOF for Option 2 (baseline) and Option 3 Inspection Programs

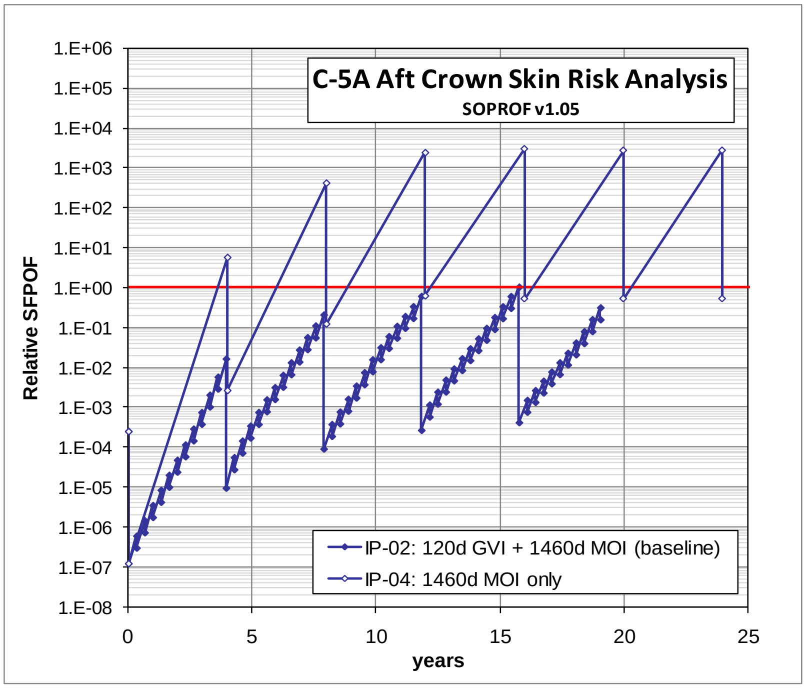


Figure 75. Normalized SFPOF for Option 2 (baseline) and Option 4 Inspection Programs

In order to perform the risk analyses for the SHM options, it was necessary to estimate the detection reliability of the candidate SHM system. Since a full POD study for a system of sensors installed on the aft crown was well beyond the scope of this program, a probability of detection profile for the Acellent Smart Layer system in the "A" configuration (as described in the previous section) was developed as follows:

- Assumed log-odds distribution [3.10]
- A review of the Acellent test report showed 7 out of 9 'hits' using the DPI method for 1 inch notches in various orientations and positions in the aft crown skin. Based on this, a 50% flaw size of 1 inch was assumed. This is a conservative assumption. However, the fact that the hit/miss data is based on notches, and not natural fatigue cracks, is unconservative.
- Assumed threshold flaw size is 0.2 inch.
- Assumed standard deviation is 1.0 inch.

The resulting POD curve, assuming a lognormal distribution, is shown in Figure 76. The cumulative probability of detection curves for general visual and magneto-optical inspection are also shown for comparison. The assumed POI for SHM was 1.0.

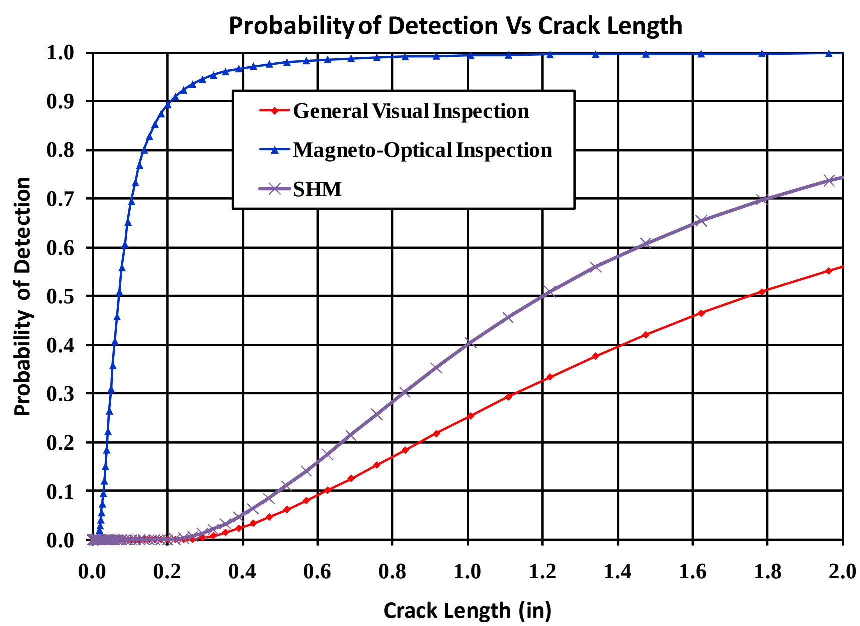


Figure 76. Estimated Crack Detection Reliability for SHM Sensor System on C-5A Aft Crown Skins

SOPROF calculations were made using the SHM POD curve given in Figure 76 with all other input parameters the same as described above. The calculated normalized risk vs. time for options 5 through 8 are shown in Figures 77 through 80.

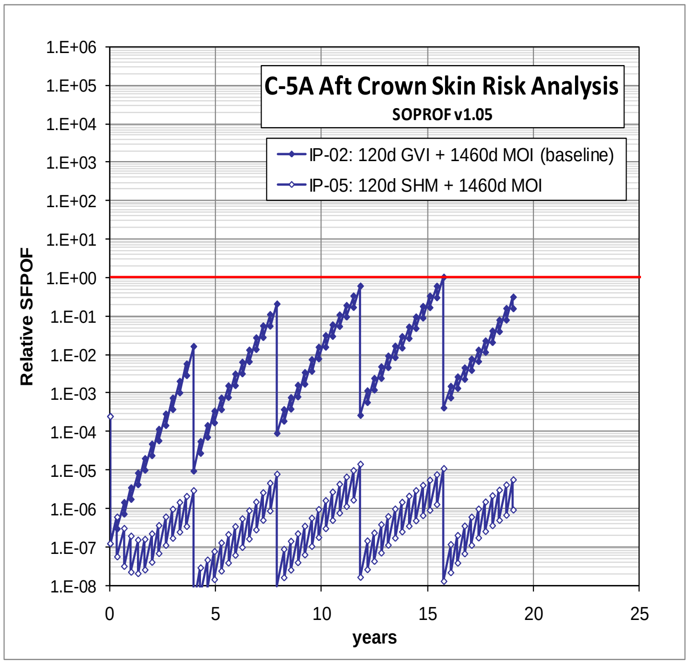


Figure 77. Normalized SFPOF for Baseline and Option 5 SHM / MOI Inspection Programs

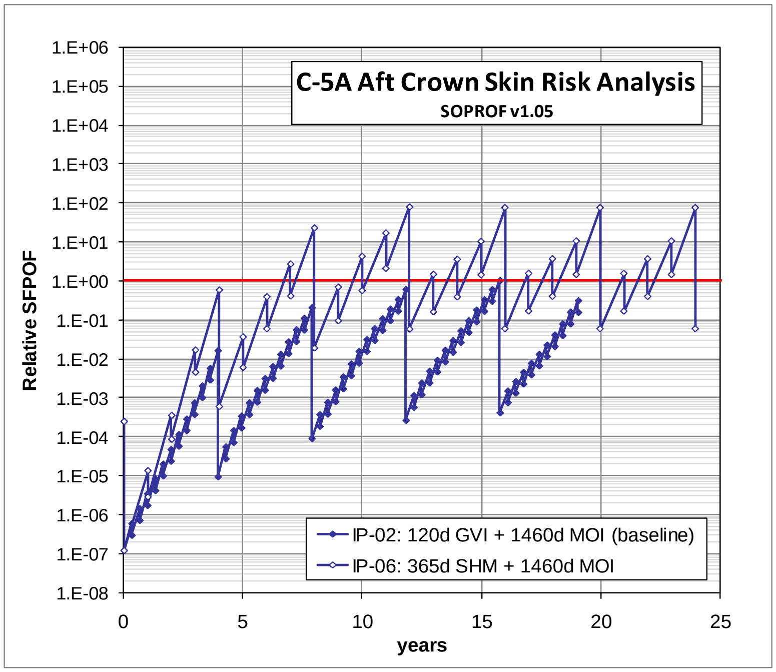


Figure 78. Normalized SFPOF for Baseline and Option 6 SHM / MOI Inspection Programs

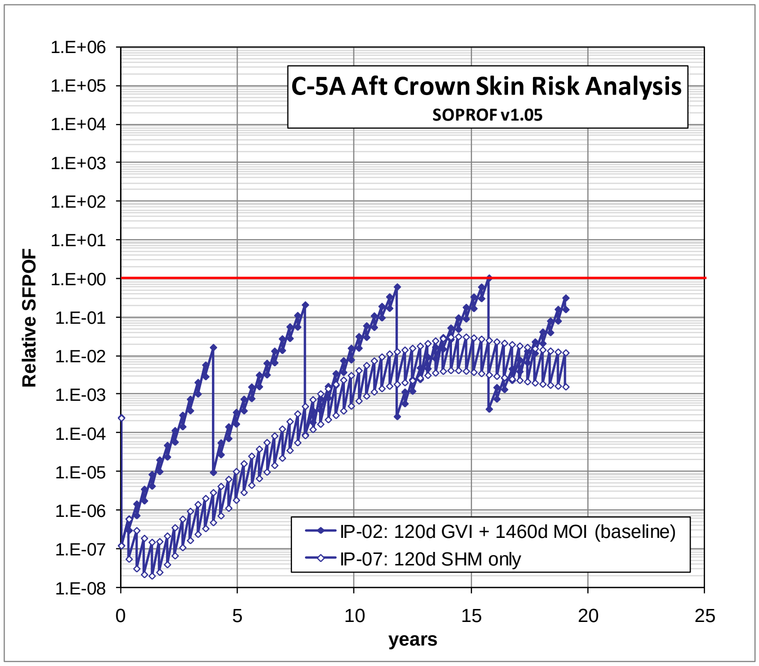


Figure 79. Normalized SFPOF for Baseline and Option 7 SHM / MOI Inspection Programs

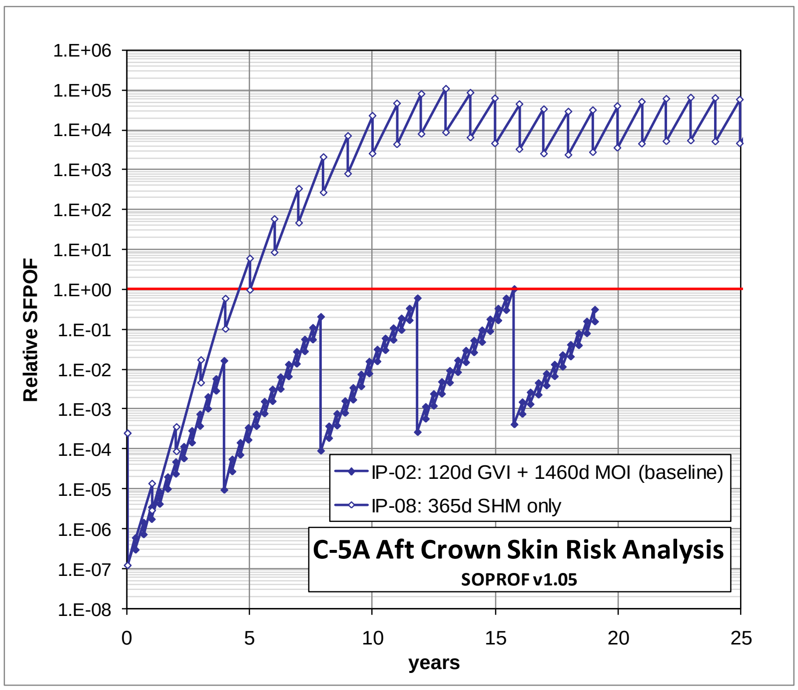


Figure 80. Normalized SFPOF for Baseline and Option 8 SHM / MOI Inspection Programs

As discussed above, for an SHM enabled CBM program to be considered a viable option, it had to satisfy the criterion that it would limit structural risk (as quantified by SOPROF calculated SFPOF) to as good or better values than are currently being achieved through combined visual plus magneto-optical inspections. Inspection of Figures 77 through 80 shows that by this criterion, only two of the four SHM options considered are viable. See Table 3. The included cost metric will be discussed later.

Table 3. Viability of Inspection / SHM Strategies for C-5A Aft Crown Skin Based on Relative Structural Risk

no.	description	cost metric ($M)	max relative SPOF
IP-1	120d GVI + 960d MOI	25.3	1.3E-03
IP-2	120d GVI + 1460d MOI (baseline)	18.3	1.0E+00
IP-3	960d MOI only	20.5	1.3E+00
IP-4	1460d MOI only	13.5	3.0E+03
IP-5	120d SHM + 1460d MOI	22.9	1.4E-05
IP-6	365d SHM + 1460d MOI	21.5	7.8E+01
IP-7	120d SHM only	22.2	3.1E-02
IP-8	365d SHM only	20.9	1.1E+05

Notes:
1) All options start with 450d GVI w POI=0.5 starting in 1995, followed by 105d GVI w POI=0.5 + 840d MOI w POI=1 starting in 2003, followed by 120d GVI w POI=0.5 + 960d MOI w POI=1 starting in 2003
2) 120d GVI is at HSC, 960d MOI is at minor, 1460d MOI is at major
3) Cost metric is estimated cumulative cost over 2015 to 2040 (25 yr) time period
4) Options 5 thru 8 include the costs of developing, installing, operating and maintaining the SHM system according to the parameters of each row
5) Costs are escalated for inflation in each year
6) All SPOF values based on SOPROF v1.05 risk analyses using "Babish" exponential crack growth curve and IFSD and 2006 data
7) GVI POD ref AFRL, POI=0.5
8) MOI POD ref AFRL, POI=1.0
9) SHM POD estimated based on Acellent test results, POI=1

3.8 Task 8 - Determine Benefits/Risks and Establish Business Case

3.8.1 Introduction

If ISHM/CBM sensors and data collection can be implemented in a technically suitable manner for the C-5A Aft Crown Skin, is an ISHM/CBM system profitable to implement? To answer this, factors which should be considered include:

- Areas of Cost for ISHM/CBM System:
- Non-recurring development and testing
- Recurring fabrication and implementation
- System operational cost
- System maintenance cost
- Areas of Savings from ISHM/CBM System:
- Elimination or reduction of Detailed Visual Inspections (DVI)
- Elimination or reduction of Magnetic Optical Imaging (MOI) and Eddy Current Surface Scan (ECSS)
- Reduction of ASIP reporting man-hours

Areas less easy to quantify include reduction in repairs and increase of availability due to more flexible scheduling of Aft Crown Skin repairs.

SHM Objectives:

- SFPoF – As good as or better than Visual/MOI Inspections
    - This was demonstrated to be feasible
- Improve Availability
    - Due to other activities performed during HSC a limited reduction in overall span time is possible, but is small
- Eliminate O Level MOI and Visual Inspection (HSC)
    - Demonstrated to be feasible
- Reduce number of Partial Mission Capable (PMC) to Fully Mission Capable (FMC) aircraft
    - This is feasible, but current operations are not affected by aircraft being partially mission capable due to deferring the current 120 visual inspection in the HSC
- Aircraft RE/Mod Cost $100K per Aircraft
    - Possible if investment in technology and manufacturing are made

3.8.2 Approach

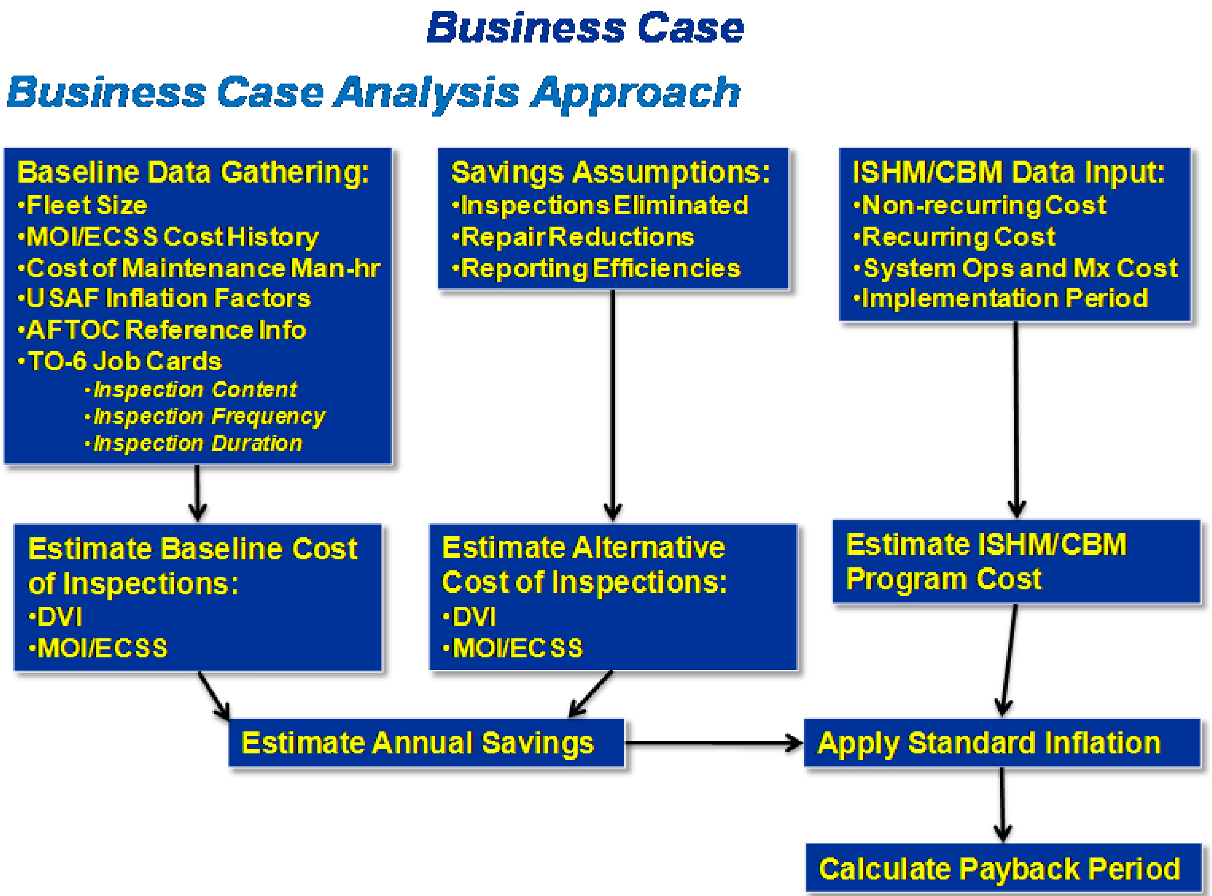


Figure 81. Business Case Analysis Approach

Figure 81 above describes our approach to developing a business case. The first step is to gather baseline data and estimate current costs of doing business. Next after establishing assumptions and estimating cost of proposed alternatives the Annual Savings are estimated. The cost of developing and installing the new system along with estimated O&S costs are established and then finally the payback period is calculated and presented as depicted below in Figure 82.

Business Case

Payback Results (IP-7 versus IP-2)

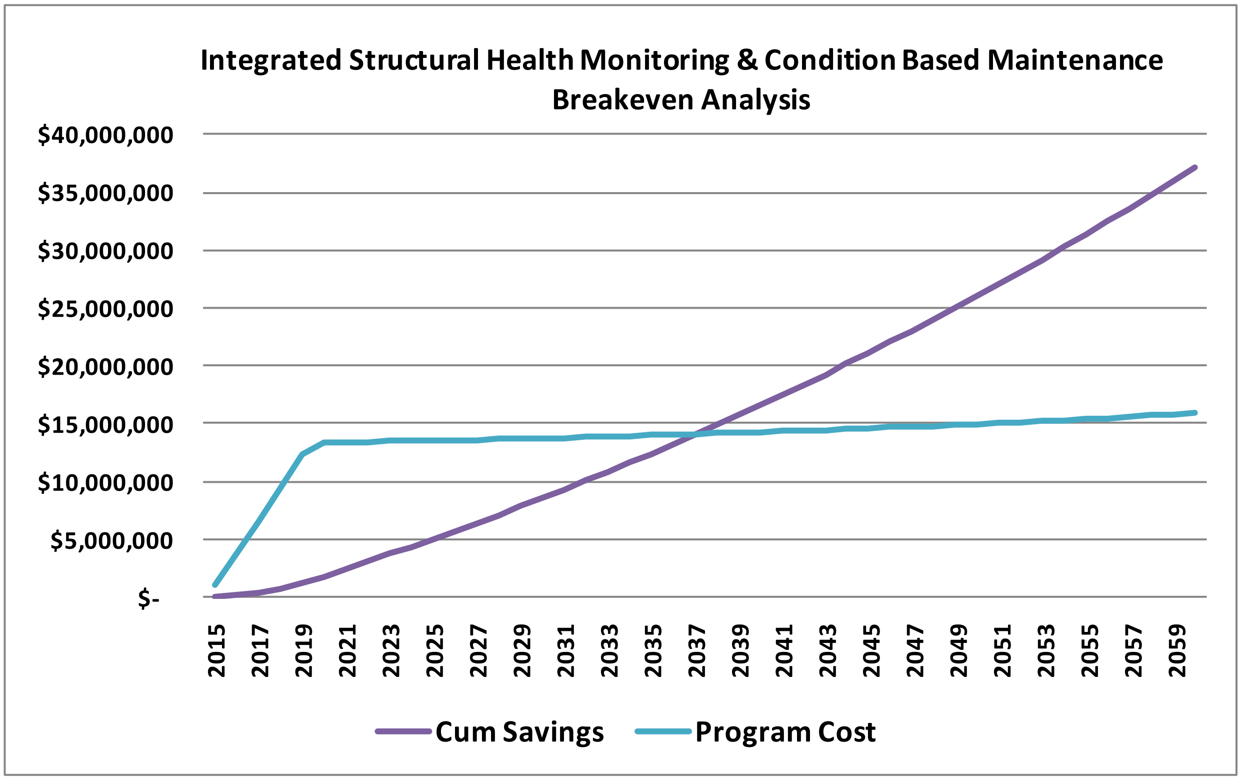


Figure 82. Payback Analysis

For the Aft Crown we identified 8 scenarios or alternate Inspection Programs (IP). These are identified below.

Table 4. Alternate Inspection Programs

no.	description
IP-1	120d GVI + 960d MOI
IP-2	120d GVI + 1460d MOI (baseline)
IP-3	960d MOI only
IP-4	1460d MOI only
IP-5	120d SHM + 1460d MOI
IP-6	365d SHM + 1460d MOI
IP-7	120d SHM only
IP-8	365d SHM only

IP-2 is the current baseline or how business is currently being done. With a 120 day visual inspection as part of the Home Station Check (HSC) and a 1460 Day (48 Months) MOI performed in the field. The MOI is a NDI performed in the field and must be done in a hangar.

IP-1, IP3 and IP-4 are alternative inspection intervals. IP-5 through IP-8 all involve SHM with SHM only options and combinations of SHM and traditional inspections.

For each option the costs and the risks were evaluated.

Part of the initial data needed was the fleet size (Figure 83), Inspection intervals and descriptions and man-hour data.

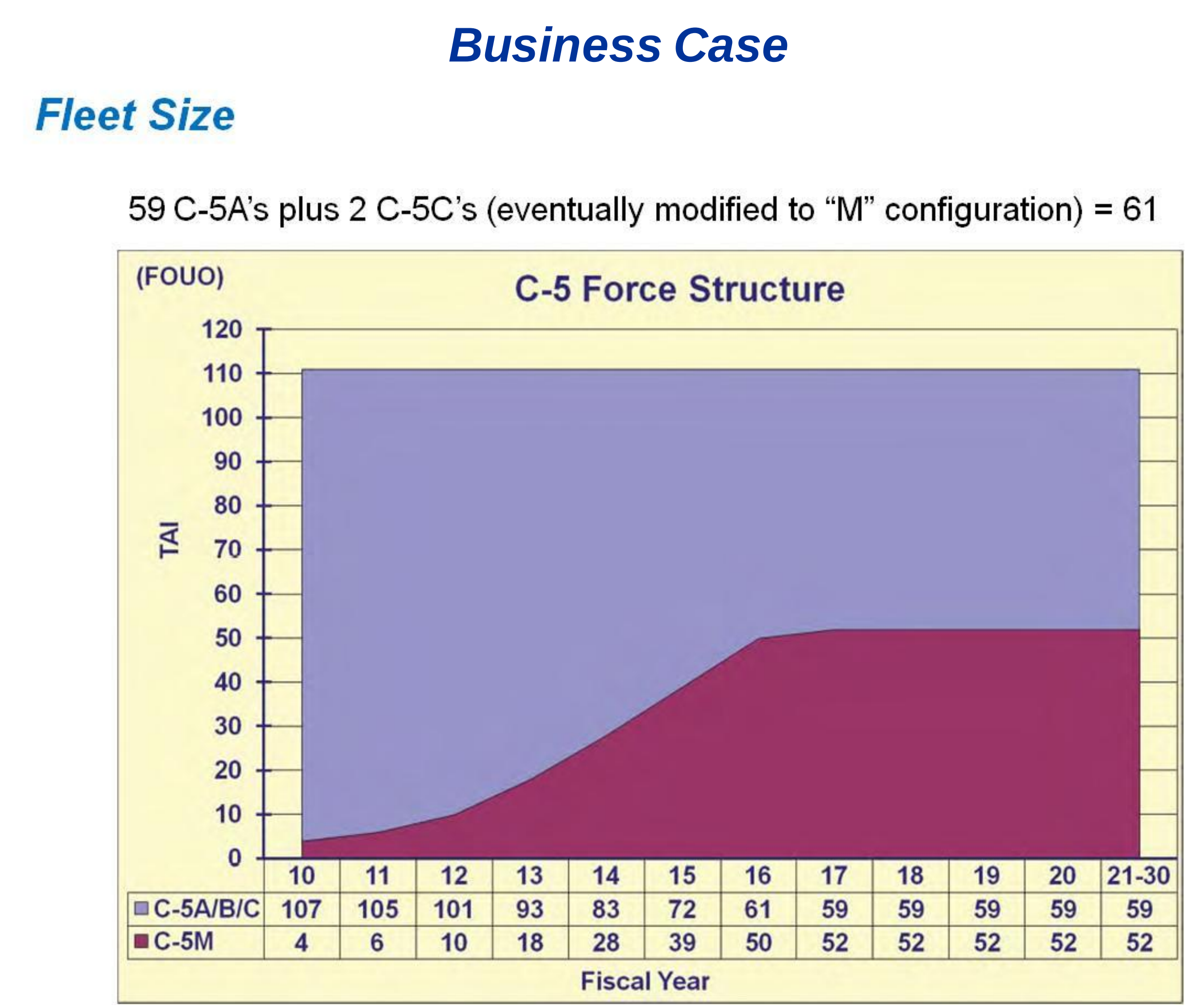


Figure 83. Fleet Size

This data along with other data is used to calculate Annual Fleet Savings.

Business Case

Savings Calculation (IP-7 versus IP-2)

Baseline: Current C-5A/C Configuration and Inspection Requirements, October 2010

	Frequency			Tolerance		Duration		Mx Crew Size		Per Inspection Calc Base MH Cost	Per Year Calc Base MH Cost
Detailed Visual Inspections (DVI)											$ 138,003.45 (DVI)
Home Station Checks (HSC):	120	days,	+ or -	10	days	480	min	2	man	$ 743.79	
Minor Isochronal (Incl HSC):	487	days,	+ or -	21	days	480	min	2	man	$ 743.79	
Major Isochronal (Incl Minor & HSC):	1460	days,	+ or -	21	days	480	min	2	man	$ 743.79	
Magnetic Optical Imaging (MOI) & Eddy Current Surface Scan	1460	days,	+ or -	21	days	-	min	2	man	$ 25,246.25	$ 385,005.31 (MOI + ECSS)
Programmed Depot Maintenance (PDM):	2920	days,	+ or -	-	days	-	min	2	man	$ 25,246.25	-
											$ 523,008.77 (Aft Crown Inspect)

Alternative: Integrated Structural Health Monitoring and Condition Based Maintenance

	Frequency			Tolerance		Duration		Mx Crew Size		Per Inspection Calc Base MH Cost	Per Year Calc Base MH Cost
Detailed Visual Inspections (DVI)											$ 1.66 (DVI)
Home Station Checks (HSC):	9999999	days,	+ or -	10	days	480	min	2	man	$ 743.79	
Minor Isochronal (Incl HSC):	9999999	days,	+ or -	21	days	480	min	2	man	$ 743.79	
Major Isochronal (Incl Minor & HSC):	9999999	days,	+ or -	21	days	480	min	2	man	$ 743.79	
Magnetic Optical Imaging (MOI) & Eddy Current Surface Scan	9999999	days,	+ or -	21	days	-	min	2	man	$ 25,246.25	$ 56.21 (MOI + ECSS)
Programmed Depot Maintenance (PDM):	2920	days,	+ or -	-	days	-	min	2	man	$ 25,246.25	-
											$ 57.87 (Aft Crown Inspect)

Annual Whole Fleet Savings: $ 522,951

Figure 84 – Savings Calculations

Then the new system costs are applied and the payback period is calculated.

Business Case

SHM Program Cost Inputs and Calculations (IP-7 versus IP-2)

Non-Recurring Cost of Alternative:	$1,000,000			
Alternative Recurring Cost per Aircraft:	$200,000			
Alternative Implementation Start Date:	1/15/2015			
Alternative Implementation Finish Date:	1/15/2020			
	1826	days		
ISHM/CBM Operation: Interval in days:	120		Manhours:	1
ISHM/CBM Inherent Maint: Interval in days:	360		Manhours:	3
ISHM/CBM Induced Maint: Interval in days:	360		Manhours:	3
Alternative Annual Ops & Mx Cost:	$ 25,876			
Mod Program Cost:	$13,200,000			
Annual Whole Fleet Savings:	$ 522,951			
Year of Payback:	2037			
Years to Payback:	17	years		

Figure 85 – Payback Calculations

The results for each alternative, along with the risk assessment, are shown below.

Business Case

COST ANALYSIS (2015 – 2040)

- **Of acceptable options considered, current inspection program (120d GVI + 48 mo MOI) is least expensive**
- **SHM options IP-05 and IP-07 meet hazard rate requirement, but are not viable based on cost**

no.	description	max relative SPOF	cost metric ($M)
IP-01	120d GVI + 960d MOI	1.3E-03	25.3
IP-02	120d GVI + 1460d MOI (baseline)	1.0E+00	18.3
IP-03	960d MOI only	1.3E+00	20.5
IP-04	1460d MOI only	3.0E+03	13.5
IP-05	120d SHM + 1460d MOI	1.4E-05	19.8
IP-06	365d SHM + 1460d MOI	7.8E+01	18.5
IP-07	120d SHM only	3.1E-02	19.2
IP-08	365d SHM only	1.1E+05	17.8

Figure 86 – Results

Referring to Figure 86 it can be seen that IP-1 and IP-2 are acceptable in terms of risk with IP-2 the preferred in terms of cost. For the SHM options IP-5 and IP-7 are acceptable in terms of risk, but neither are cheaper than IP-2.

COST ANALYSIS:

Estimate total cost for each inspection program

Select inspection program with lowest cost that meets USAF structural integrity requirements (max SFPOF=1e-7)

Cost metric is estimated cumulative cost over 2015 to 2040 (25 yr) time period

Options 5 thru 8 include the costs of developing, installing, operating and maintaining the SHM system according to the parameters of each row

Costs are escalated for inflation in each year

Factors which impacted analysis:

Areas of Cost for ISHM/CBM System:

Non-recurring development and testing

Recurring fabrication and implementation
System operational cost
System maintenance cost
Areas of Savings from ISHM/CBM System:
Elimination or reduction of Detailed Visual Inspections (DVI)
Elimination or reduction of Magnetic Optical Imaging (MOI) and Eddy Current Surface Scan (ECSS)
Reduction of ASIP reporting man-hours

Note that development costs include costs associated with development for this specific application on the C-5. Validation and Qualification costs of the Acellent System, including formal establishment of POD were excluded.

The payback period for the primary analysis performed was 15 years after mod completion. Incorporation of additional saving areas could reduce this period. Multiple Case runs show a variety a cost results

Areas less easy to quantify include reduction in repairs and increase of availability due to more flexible scheduling of Aft Crown Skin repairs.

3.8.3 Acellent Business Case Inputs

To support development of a business case Acellent provided inputs that are identified and described in the following sections.

3.8.3.1 Determination of POD

Probability of detection (POD) is being introduced as a standard measurement for quantifying the reliability and robustness of built-in structural health monitoring systems. However, traditional NDE POD curves are generated through extensive testing which is not practical for every structure and sensor configuration. To overcome this difficulty, model-based methods can be used to help compute the POD. The concept of computing POD for SHM systems is still new, and the computational methods greatly depend on the underlying damage detection algorithms. Current damage detection algorithms for detecting area-type damages (such as delamination in composites or corrosion in metals) can be used to compute model-based POD curves using the methodology shown in Figure 87. The computation is based on the geometry of the sensor configuration and the actuator-sensor paths, along with the logic in the damage detection-reasoning algorithm. While this can be readily computed for algorithms to detect area-type damage, it is a bit more complicated for linear-type damage, such as cracks. This is because the cracks to detect are typically much smaller than the sensor spacing, and the crack orientation plays a critical role in detectability. Since traditional NDE techniques are based on single point measurements for detecting damage, their associated POD can generally be measured through experimental testing alone. This traditional testing approach to determine the POD is difficult to adopt for random crack detection using an SHM sensor network because of the variety of:

- Sensor positions and network arrangements

- Structural geometries and boundary conditions
- “Randomness” of crack location, orientation, and severity

Therefore, a hybrid approach utilizing numerical simulation coupled with experiments can be used to determine the POD of cracks for an SHM system. The result will be POD curves for the entire structure as a function of sensor arrangements (Figure 88).

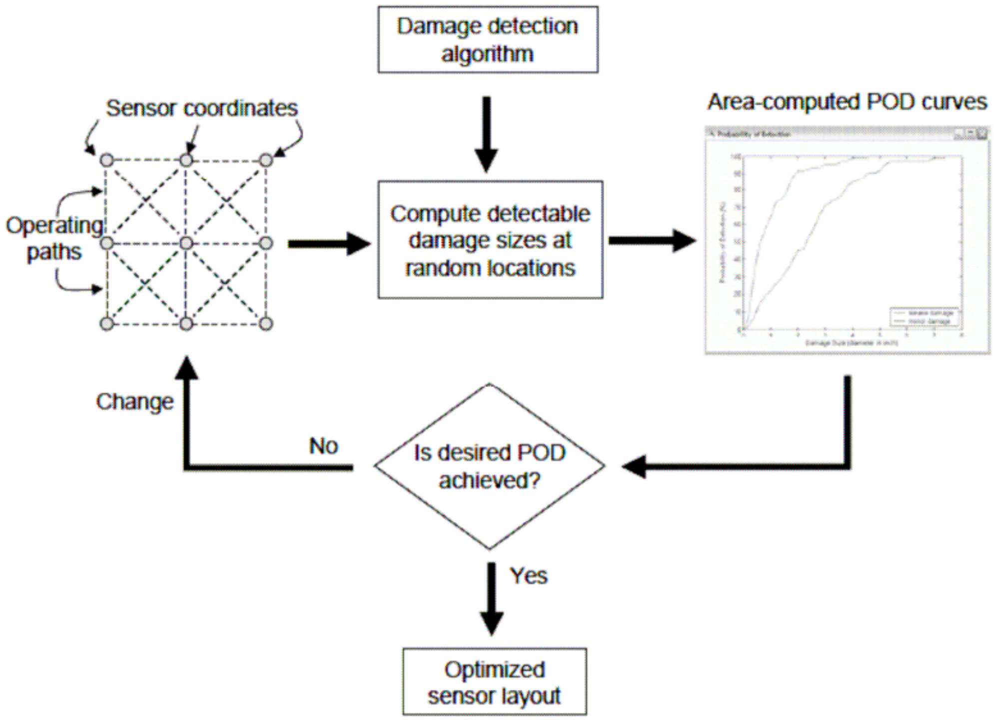


Figure 87. Methodology to optimize sensor layout for detecting area-type damages

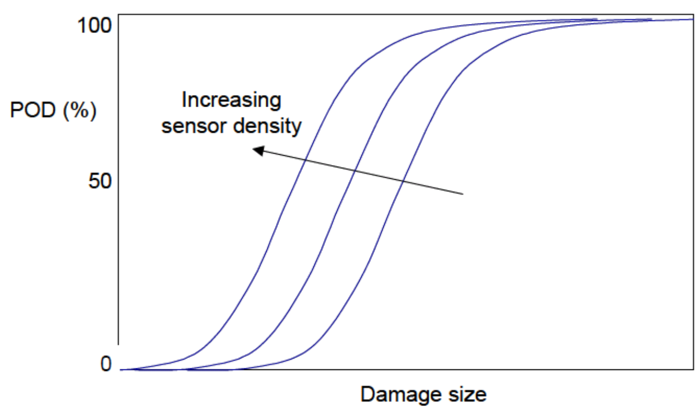


Figure 88. POD Curves are a Function of Sensor Density

In an SHM sensor network, the minimum measurement unit or building block is an individual actuator-sensor path. A network can always be decomposed into individual paths, regardless of the geometry. Therefore, if we can determine the POD of a single path, then it's possible to synthesize the POD of the entire network (Figure 89). More specifically, the approach is to first create the POD for a single path and then compute the POD of the network through probability computation.

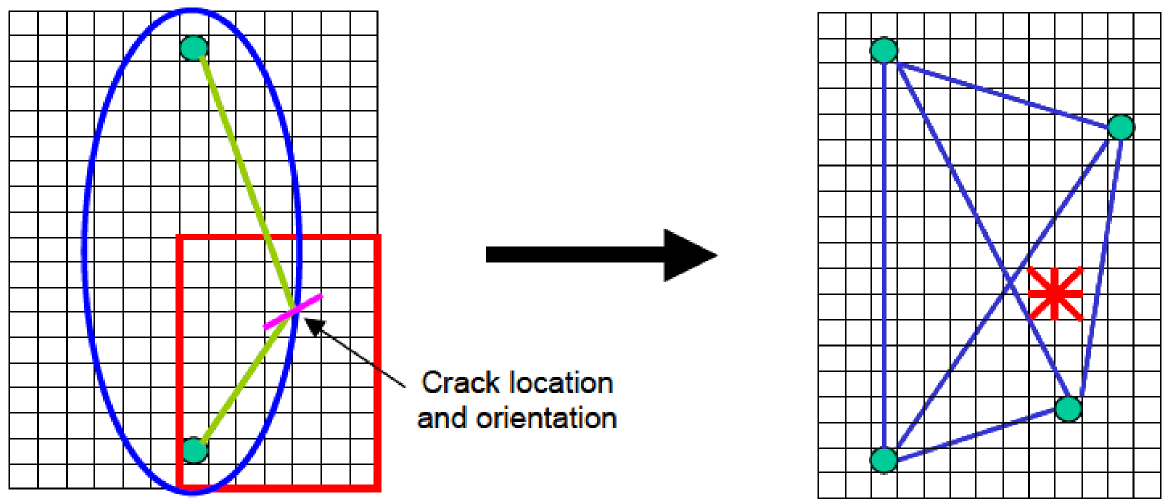


Figure 89. POD of Single Path can be used as Building Block to Compute POD of Entire Network

The damage sensitivity of an actuator-sensor path is dependent on three parameters including: 1) the distance from a crack to the nearest sensor, 2) the propagation length from the actuator to the damage and from the damage to the sensor, and 3) the angle between the crack orientation and the actuator-sensor path direction. Tests were conducted on specimens with an overpopulation of sensors, such that a single experiment with one crack can produce a multitude of data. This is because each actuator-sensor path will have a different length and orientation to the crack. For a network with ***n*** sensors, there will be ***n!/(2(n-2)!)*** actuator-sensor paths with unique lengths and orientation to the crack. Using this approach, the POD for any sensor network arrangement can be constructed with a minimal number of experiments.

3.8.3.2 Implementation for C-5 Aft Crown

3.8.3.2.1 Monitoring Areas

From the sensor data collected during the laboratory tests, it was found that a SMART Layer ring design with 8 sensors in each bay could reliably detect a 0.5" crack occurring anywhere on the skin (within the bay, under a frame, and under a stringer). Because the skin is relatively thin, and the sensor signals are strong and can propagate across multiple bays, it is estimated that using a single strip with 4 sensors in each bay can reliably detect a 1.0" crack occurring anywhere. Cost estimates for these two approaches (using an 8-sensor ring or using a 4-sensor strip in each bay) are given in following sections. The total number of sensors, and the number of bays that SMART Layers are applied to can be reduced depending on the crack sizes that needs to be detected. Ultimately, the best approach may have higher sensor density in critical regions and a sparser network in less critical areas.

3.8.3.3 Estimated Hardware Costs for 8-Sensor SMART Layer Ring Architecture

3.8.3.3.1 Sensor Layout

To detect a 0.5" crack anywhere in the aft crown area, an 8-sensor SMART Layer Ring can be mounted in each bay. The rings can be connected through cabling or a flex circuit to a multiplexer switch amplifier (SA) box, which in turn is connected to a switch hub (SH). A schematic of the aapproach is shown in Figure 90.

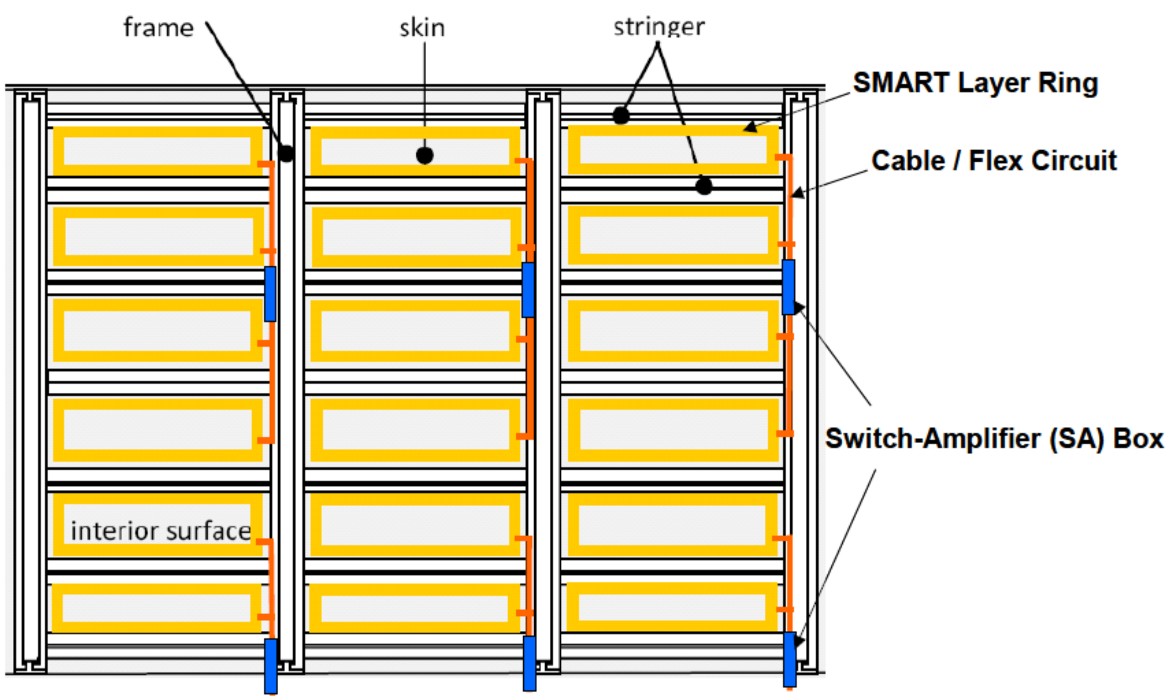


Figure 90. Sensor Layout and Architecture for using 8-Sensor SMART Layer Ring in Each Bay

3.8.3.3.2 System Cost

Assuming 400 bays in the aft crown, the sensor network will include 400 SMART Layer rings with 8 PZT sensors each. The cost estimation of the sensor network for each C-5 is listed below.

Cost Estimation of SMART layer	Unit Price ($)	Qty	Total Price ($)
SMART Layer Ring	180	400	72,000

Each SMART Layer will be connected to a switch amplifier (SA) box. The average cost for a connection cable is about $35. The total cost for all 400 cables is about $14,000.

100 SA boxes, each connecting to 32 sensors and costing $1,200, will be needed. The total cost of SA boxes will be $120,000. If a switch hub (SH) can handle 10 SA boxes, 10 SH are needed. Each SH costs $6,000. The total cost of SH will be $60,000.

The estimation of total cost for the SHM system is given below.

Cost Estimation of SHM System (hardware only)

Components	**Cost**
SMART Layers	$72,000
Cables	$14,000
SA Boxes	$120,000
SH	$60,000
Total	$266,000

3.8.3.3.3 Estimated Hardware Costs for 4-Sensor SMART Layer Strip Architecture

3.8.3.3.4 Sensor Layout

To detect a 1.0" crack anywhere in the aft crown area, a 4-sensor SMART Layer Strip can be mounted in each bay. The strips can be connected through cabling or a flex circuit to a multiplexer switch amplifier (SA) box, which in turn is connected to a switch hub (SH). A schematic of the architecture is shown in Figure 91.

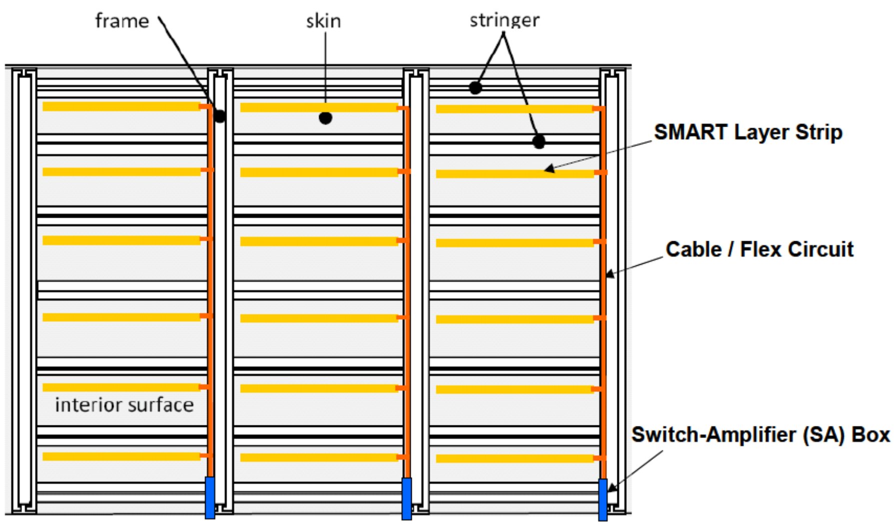


Figure 91. Sensor Layout and Architecture for using 4-Sensor SMART Layer Strip in Each Bay

3.8.3.3.5 System Cost

Assuming 400 bays in the aft crown, the sensor network will include 400 SMART Layer strips with 4 sensors each. The cost estimation of the sensor network for each C-5 is listed below

Cost Estimation of SMART layer	Unit Price ($)	Qty	Total Price ($)
SMART Layer Strip	100	400	40,000

Each SMART Layer will be connected to a switch amplifier (SA) box. The average cost for a connection cable is about $25. The total cost for all 400 cables is about $10,000. 50 SA boxes, each connecting to 32 sensors and costing $1,400, will be needed. The total cost of SA boxes will be $70,000. If a switch hub (SH) can handle 10 SA boxes, 5 SH are needed. Each SH costs $7,000. The total cost of SH will be $35,000.

he estimation of total cost for the SHM system is given below.

Cost Estimation of SHM System (hardware only)

Components	**Cost**
SMART Layers	$40,000
Cables	$10,000
SA Boxes	$70,000
SH	$35,000
Total	$150,000

3.8.3.3.6 System Installation and Connections

In order to monitor large-scale structures, such as the aft crown of the C-5A, there needs to be practical methods to install and connect large sensor networks. The architecture options shown above are modular in design, i.e. they are applicable to localized areas requiring large number of sensors and scalable to large areas. Also, the architecture options consist of different components that are considered to be on-board (permanently integrated with the vehicle structure) or off-board (detachable from the vehicle structure).

The major advantages of off-board architecture include:

1) Minimal weight is added to the aircraft
2) Lower system cost because one diagnostic hardware can be shared with multiple aircraft

The major advantage of on-board architecture include:

1) Active scan for damage at any time
2) Easier to collecting baseline
3) Easier integration with CBM system

3.8.3.4 Targeted Upgrades of SHM Components

ASIC-based hardware system will be developed in the future for low-cost and lightweight application. The development roadmap for Acellent active SHM hardware is shown in the following picture.

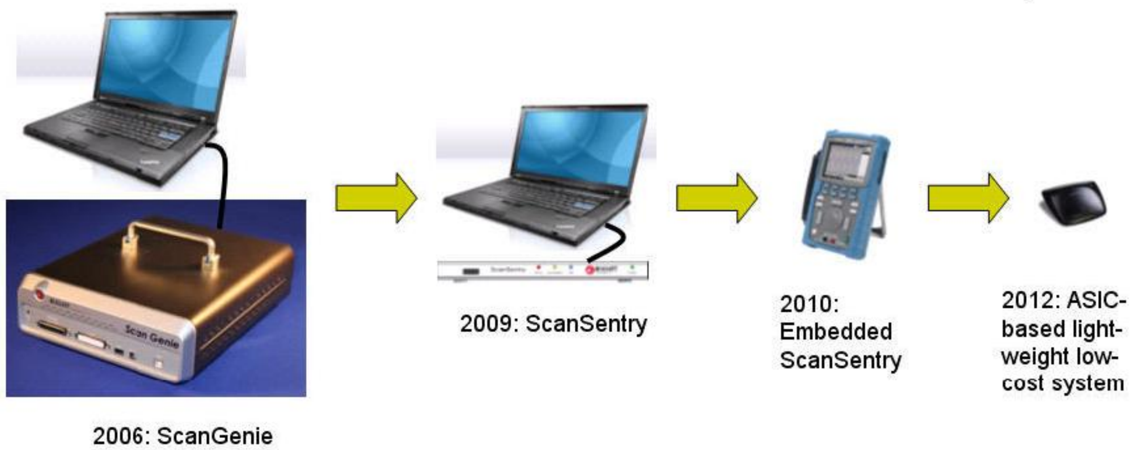


Figure 92. Acellent Upgrade Path

Ultimately, if the ASIC-based hardware system is developed, then the weight of the control hardware system can be reduced to less than 1 lb. It is expected the ASIC-based data acquisition system will be less than $1,000 per unit.

3.8.4 JENTEK Inputs to Business Case

JENTEK Sensors, Inc. is developing a miniaturized impedance instrument for which a single channel with self-contained cabling could be located in each bay with multiplexing to support sensors and serial communications (Ethernet, USB, etc.) cabling between sensors.

JENTEK believes that this could be miniaturized sufficiently to be practical. Power could be provided by Power Over Ethernet (POE) or a two-wire cable. Also, off loading of data could be supported by an optical bus or wireless transmission. Note that one alternative is to have a self-contained impedance instrument and multiplexing unit in each bay. Another alternative is to share some of the duplicate functions of the impedance instrument and multiplexing control circuitry between several bays to reduce costs and weight.

This approach would satisfy the needs for accessible aluminum skin and monitoring through titanium straps. Note that for cracks that grow under the aluminum frames and stringers, we did not attempt detection in this program. To address these cracks would require a substantial development of a low frequency sensor construct to see through the aluminum frame and stringer materials. This is possible and could be supported by similar electronics and system designs.

The estimated costs of the JENTEK system were considerably greater that the Acellent system, which is inherently better suited for wide area coverage. Due to this we did not pursue a business case for use of the JENTEK technology.

3.8.5 Summary

An update of aft crown skin risk analysis, performed during the preparation of this report, will have an impact on the business case. Updated analysis is based on:

- Improved crack growth curves (significant additional fleet data since 2006)
- Improved estimate of residual strength (based on NIAR test results)

It is likely that new, less frequent inspection inspections will be adopted, thus establishing new a baseline:

32 mo. DVI
8 year MOI

Comparative risk analyses for SHM options will have to be re-run – however, given that new baseline will be less expensive than current baseline, it will be even more difficult to demonstrate a cost benefit.

3.9 Task 9 - Transition Roadmap and Strategy

The initial business case, including inputs from Acellent and JENTEK describing development under way internally as well as their estimates of current costs, suggest that implementation costs can be significantly reduced over the next few years

The performance demonstrated and the improvements that would occur during a focused development effort suggest that a system would work.

The high level Transition Roadmap and Strategy is to:

- Engage with the C-5 Program and the C-5 ALC to present findings from the CBM+SI CRAD to gain support for further development
- Engage with AFRL and pursue a follow-on Contract Research and Development (CRAD) or Cooperative Research and Development Agreement (CRADA)
- Pursue use of Advanced Composite Cargo Aircraft (ACCA) as a vehicle for further maturation and development
- Engage with LM Aero Improvements and Derivatives to identify needs within programs
- Engage with other LM Business Units to identify more potential customers
- Develop a comprehensive Technology Roadmap

4. CONCLUSIONS AND RECOMMENDATIONS

4.1 Conclusions

LM Aero developed an approach and framework fore integration of CBM+ with SI that included investigation and development of:

Modified ASIP standard
Maintenance Concept of Operation
Availability, Operation and Support Cost Models
Approaches for Information Architectures
CBM+SI Candidate Selection Approaches
Business Case for a CBM+SI Application
Initial Transition Plan for a CBM+SI Application

Developed a structural application prototype:

- Conducted laboratory demonstrations for two SHM sensor types
- Developed mock-up of SHM system and supporting software
- Conducted structural risk analyses to estimate SFPoF for each of the CBM+ ConOps scenarios prescribed in the requirements phase – this allowed quantitative comparison of risk for inspection programs, with and without SHM, against current baseline inspection program

Implementation of CBM+ requires a systems engineering approach, however, each potential application will be unique and each will require individual assessment – for example applications on F-22 will not have same requirements or results as those on C-5.

Implementation of CBM+ requires reliable cost modeling:

- At the outset of this program, it seemed likely that a compelling business case could be made for the utilization of SHM on the aft crown. For the current state of the art, it now appears that that may not be true
- Establishing a detailed credible cost baseline was more difficult than anticipated.

The CBM+ business case can be strongly affected by:

- Accurate baseline
- Reliability and durability of SHM system
- CBM+ ConOps

4.2 Recommendations

Technical barriers to widespread application of SHM exist, but they can be overcome, technologies needed are:

Health Management Sensor development

- Damage and defect detection (location and severity)
- Global vs. local (films, layers, networks, etc.)

- Sensor size (smaller is better)
- Sensor reliability (eliminate false positives)
- Sensor durability (some critical locations are in-accessible, sensors must perform reliably for long period of time)
- Power requirements (low or self-powered is better, energy harvesting)

Health Management System development

- Signal transmission (wireless)
- AV data processing / storage (on-board resources are extremely hard to acquire due to competition with other systems)

4.2.1 Specific Recommendations

Develop sensor technologies - Shift investment toward reducing hardware cost and qualification

Explore applications where physical access is restricted - Extensive teardown and reassembly as well as LO Restoration impacts

Application of CBM+ for a single item may not make a compelling business case. Consider multiple applications that have cumulative benefits. Explore potential for amortizing some implementation costs across multiple applications.

- Develop SHM systems
- Quantify reliability, test, MAPOD
- Quantify durability
- Continue SHM integration into ASIP

Apply CBM+ Implementation Process for other candidate structural applications on different platforms such as F-16, F-22, C-130, other C-5 applications.

5. REFERENCES

[3.1] Condition Based Maintenance Plus DoD Guidebook, May 2008.

[3.2] DoDI 4151.22, "Condition Based Maintenance Plus (CBM+) for Materiel Maintenance," 2 Dec. 2007.

[3.3] MIL-STD-1530C (USAF), "Department of Defense Standard Practice – Aircraft Structural Integrity Program (ASIP), 1 Nov. 2005.

[3.4] Haugse, E. and Derriso, M., "Health Monitoring for Structural Hot Spots, A Systems Approach," USAF ASIP Conference, Dec. 2007, San Antonio TX.

[3.5] Ingram, J.E., Babish, C.A., et.al., Risk Assessment for C-5A Aft Crown Skins. 2006.

[3.7] Hovey, P.W., Berens, A.P. and Loomis, S.J., "Update of the Probability of Fracture (PROF) Computer Program for Aging Aircraft Risk Analysis," AFRL-VA-WP-TR-1999-3030, Nov. 1998.

[3.8] Ingram, J.E., Johnson, J. and Brussat, T.R., "Updated Risk Assessment for C-5A Aft Crown Skins," 2010.

[3.9] NIAR final report

[3.10] MIL-HDBK-1823A "Department of Defense Handbook – Nondestructive Evaluation System Reliability Assessment," 7 Apr. 2009.

[3.11] MIL-HDBK-516B "Department of Defense Handbook – Airworthiness Certification Criteria," 26 Sept. 2005.

ATTACHMENT A

PROPOSED
AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP)
WITH CBM+

The following section contains a proposed modification of the Aircraft Structural Integrity Program standard (Mil-Std-1530C) to include CBM+. The proposed changes to the text of the standard are in blue.

CONTENTS

TABLES

FIGURES

1. SCOPE

1.1 Scope

This standard describes the USAF Aircraft Structural Integrity Program (ASIP) which defines the requirements necessary to achieve structural integrity in USAF aircraft while managing cost and schedule risks through a series of disciplined, time-phased tasks. It provides direction to government personnel and contractors engaged in the development, production, modification, acquisition, and/or sustainment of USAF aircraft.

1.1.1 Application

This standard applies to the entire structure of an aircraft, as defined in section 3.1, regardless of aircraft type or procurement strategy, for the entire service life of the aircraft.

1.1.2 Tailoring

Every aircraft program must address all sections of this standard (including all tasks and elements within each task) and document this in its ASIP Master Plan. An ASIP Master Plan is required for all programs. Tailoring is only permitted when all of the following conditions exist:

a. The overall aircraft reliability (probability of failure) is established and approved by the appropriate Risk Approval Authority as defined in MIL-STD-882, "Standard Practice for System Safety."

b. The aircraft structure reliability is defined and supports the overall aircraft reliability requirement.

c. The effect of each tailored ASIP task and/or element and its associated impact on aircraft structure is determined.

d. The combined impact of all tailored ASIP tasks and/or elements on aircraft structural reliability is determined and achieves the allocated overall aircraft reliability requirement.

e. The tailored ASIP tasks and/or elements and the impact of this tailoring on aircraft structural reliability is documented in the ASIP Master Plan and approved in accordance with AFPD 63-1 and AFI 63-1001.

2. APPLICABLE DOCUMENTS

2.1 General

The documents listed in this section are specified in sections 3, 4, or 5 of this standard. This section does not include documents cited in other sections of this standard or recommended for additional information or as examples. While every effort has been made to ensure the completeness of this list, document users are cautioned that they must meet all specified requirements of documents cited in sections 3, 4, or 5 of this standard, whether or not they are listed.

2.2 Government documents.

2.2.1 Specifications, standards, and handbooks.

The following specifications, standards, and handbooks form a part of this document to the extent specified herein. Unless otherwise specified, the issues of these documents are those cited in the solicitation or contract.

DEPARTMENT OF DEFENSE SPECIFICATIONS

JSSG-2006	Aircraft Structures
MIL-DTL-87929	Technical Manuals, Operation and Maintenance Instructions in Work Package Format (for USAF Equipment)

STANDARDS

MIL-STD-882	Standard Practice for System Safety

HANDBOOKS

MIL-HDBK-516	Airworthiness Certification Criteria
MIL-HDBK-1568	Materials and Processes for Corrosion Prevention and Control in Aerospace Weapon Systems
MIL-HDBK-1823	Nondestructive Evaluation System, Reliability Assessment
MIL-HDBK-6870	Inspection Program Requirements, Nondestructive, for Aircraft and Missile Materials and Parts

(Copies of these documents are available online at http://assist.daps.dla.mil/quicksearch/ or from the Standardization Document Order Desk, 700 Robbins Avenue, Bldg 4D, Philadelphia PA 19111-5098; [215] 697-2179.)

2.2.2 Other Government documents, drawings, and publications.

The following other Government documents, drawings, and publications form a part of this document to the extent specified herein. Unless otherwise specified, the issues of these documents are those cited in the solicitation or contract.

DEPARTMENT OF DEFENSE POLICY DIRECTIVES AND INSTRUCTIONS

DFARS 207.105(b)(13)(ii) Oct 04

Defense Federal Acquisition Regulation, Part 207-

Acquisition Planning, Subpart 207.1 – Acquisition Plans
DoD Corrosion Prevention and Control Planning Guidebook
(Spiral II)

(Copies of DFARS Part 207 are available online at: http://farsite.hill.af.mil/reghtml/regs/far2afmcfars/fardfars/dfars/dfars207.htm; copies of the DoD Corrosion Prevention and Control Planning Guidebook are available at: http://www.dodcorrosionexchange.org/.)

U.S. AIR FORCE POLICY DIRECTIVES AND INSTRUCTIONS

AFPD 21-1	Air and Space Maintenance
AFMCI 21-102	Analytical Condition Inspection (ACI) Programs
AFI 21-105	Air and Space Equipment Structural Maintenance
AFPD 63-14	Aircraft Information Programs
AFI 63-1001/ AFMC SUPPLEMENT 1	Aircraft Structural Integrity Program
AFI 63-1401	Aircraft Information Programs

(Copies of Directives and Instructions are available from the U.S. Air Force Publications Distribution Center, 2800 Eastern Blvd, Baltimore MD 21220-2898; [410] 687-3330; http://afpubs.hq.af.mil.)

U.S. AIR FORCE TECHNICAL ORDERS

T.O. 1-1B-50	Basic Technical Order for USAF Aircraft Weight and Balance
T.O. X-YY-38*	Aircraft Structural Integrity Program Technical Order

(*The T.O. number will correlate to the weapon system. Copies of T.O.s are available from Oklahoma City Air Logistics Center (OC-ALC/LGLDT); 3001 Staff Drive STE 1AB1 100; Tinker AFB OK 73145-3042; [405] 736-3779; https://wwwmil.tinker.af.mil/til/tild/tildt-home.html#techorder.)

FEDERAL AVIATION ADMINISTRATION

MMPDS-Handbook	Metallic Material Properties Development and Standardization

(Copies are available from Battelle Memorial Institute, 505 King Avenue, Columbus OH 43201-2681; [614] 424-5000.)

U.S. AIR FORCE TECHNICAL REPORTS

WL-TR-94-4052/3/4/5/6 (Accession Number ADA311686/87/88/89/90)	Damage Tolerant Design Handbook (5 Volumes)

http://www.dtdesh.wpafb.af.mil/Handbook.asp

Damage Tolerant Design Handbook: Guidelines for the Analysis and Design of Damage Tolerant Aircraft Structures

(Copies of the five-volume DT Design Data Handbook are available from the Defense Technical Information Center [DTIC], 8725 John J. Kingman Road, Suite 0944, Fort Belvoir VA 22060-6218, 1-800-CAL-DTIC, http://stinet.dtic.mil; and from the Center for Information and Numerical Data Analysis and Synthesis (CINDAS) at Purdue University, 500 Center Drive, West Lafayette IN 47907-2022; 1-765-494-7039; https://engineering.purdue.edu/MSE/Research/CINDAS/Pubs_html).

2.3 Non-Government publications.
The following documents form a part of this document to the extent specified herein. Unless otherwise specified, the issues of these documents are those cited in the solicitation or contract.

Center for Information and Numerical Data Analysis and Synthesis

CINDAS Aerospace Structural Metals Handbook (6 Volumes)
CINDAS Structural Alloys Handbook (3 Volumes)

(Copies are available from Center for Information and Numerical Data Analysis and Synthesis (CINDAS) at Purdue University, 500 Center Drive, West Lafayette IN 47907-2022; 1-765-494-7039;
https://engineering.purdue.edu/MSE/Research/CINDAS/Pubs_html).
International Society of Allied Weight Engineers, Inc.

SAWE RP No. 7 Mass Properties Management and Control for Military Aircraft

(Copies are available from Society of Allied Weight Engineers, P.O. Box 60024, Terminal Annex, Los Angeles CA 90060-0024; http://www.sawe.org.)

2.4 Order of precedence.
In the event of a conflict between the text of this document and the references cited herein, the text of this document takes precedence. Nothing in this document, however, supersedes applicable laws and regulations unless a specific exemption has been obtained.

3. DEFINITIONS

3.1 Aircraft structure.

The structure of an aircraft includes the fuselage, wing, empennage, landing gear, rotorcraft rotor and drive systems, propellers, control systems and surfaces, airframe-engine interface components (including engine mounts), nacelles, air induction components, weapon mounts, structural operating mechanisms, components that perform a structural function, and other components as described in the contract specification.

3.2 Baseline operational loads/environment spectrum (baseline spectrum).

The baseline operational loads/environment spectrum is an update of the design spectrum based on measured data from operational aircraft (e.g., data obtained from the loads/environment spectra survey).

3.3 Baseline service life.

The baseline service life is the period of time (e.g., years, flight cycles, hours, landings, etc.) established subsequent to design, during which the structure is expected to maintain its structural integrity when flown to the baseline loads/environment spectrum.

3.4 Certification.

Certification is a repeatable process implemented to verify an aircraft can be safely maintained and operated within its described operational envelope.

3.5 Condition-based maintenance.

Condition-Based Maintenance (CBM) is the set of maintenance processes and capabilities derived from assessment of system or component condition obtained from embedded sensors and/or external tests and measurements using portable equipment. The embedded sensors and/or external test equipment comprise a structural health monitoring (SHM) system; CBM is informed by SHM. The goal of CBM is to perform maintenance only upon *evidence of need*. CBM+ refers explicitly to the combination of CBM with prognostics; it extends the utility of CBM from asset health *assessment* to asset health *management*.

3.6 Corrosion.

Corrosion is the deterioration of a material or its properties due to the reaction of that material with its chemical environment.

3.7 Critical location.

A critical location in an aircraft structure is one that has been identified through analysis, test, or service history as a being especially sensitive to the presence of damage.

3.8 Damage.

Damage to aircraft structure is any crack, flaw, corrosion, disbond, delamination, and/or other feature that degrades, or has the potential to degrade, the performance of the affected component.

3.9 Damage tolerance.
Damage tolerance is the attribute of a structure that permits it to retain its required residual strength for a period of unrepaired usage after the structure has sustained specific levels of fatigue, corrosion, accidental, and/or discrete source damage.

3.10 Design loads/environment spectrum.
The design loads/environment spectrum is the spectrum of external loads and environments (chemical, thermal, etc.) used in the design of the aircraft and is representative of the spectrum that the typical force aircraft is expected to encounter within the design service life.

3.11 Design service life.
The design service life is the period of time (e.g., years, flight cycles, hours, landings, etc.) established at design, during which the structure is expected to maintain its structural integrity when flown to the design loads/environment spectrum.

3.12 Durability.
Durability is the ability of the aircraft structure to resist cracking, corrosion, thermal degradation, delamination, wear, and the effects of foreign object damage for a prescribed period of time.

3.13 Economic life.
The economic life is the period during which it is more cost-effective to maintain and repair an aircraft than to replace it. Economic life can be applied on a component, aircraft, or force basis.

3.14 Equivalent flight hours.
Equivalent flight hours are the actual flight hours accumulated by an aircraft adjusted for the actual usage severity compared to the design spectrum or to the baseline spectrum.

3.15 Equivalent initial flaw size (EIFS) distribution.
The equivalent initial flaw size distribution is a characterization of the initial quality of the aircraft structure. The EIFS distribution is derived by analytically determining the initial flaw size distribution that must be used to obtain the measured flaw size distribution discovered following exposure to the test or actual usage stress spectra.

3.16 Fail-safe structure.
A fail-safe structure is a structure that retains its required residual strength for a period of unrepaired usage after the failure or partial failure of safety-of-flight structure.

3.17 Force Structural Maintenance Plan.

The Force Structural Maintenance Plan (FSMP) defines when, where, how, and the estimated costs of all inspections, modifications, and CBM required to preserve structural integrity. It identifies any critical areas missed during design that require additional analysis, in-service inspections and perhaps production and/or in-service modifications. It identifies all areas that will be subject to in-service inspections and/or Structural Health Monitoring. It also describes the structural maintenance program which entails all condition-based maintenance as well as recurring maintenance (for items not addressed with condition-based maintenance) (i.e., periodic, minor and major inspections, program depot maintenance (PDM), the CPCP, etc.).

3.18 Fracture-critical part.

As shown in figure 1, a fracture-critical part is a safety-of-flight structural component that is not single load path nor sized by durability or damage tolerance requirements but requires special emphasis due to the criticality of the component.

3.19 Fracture-critical traceable part.

As shown in figure 1, a fracture-critical traceable part is a safety-of-flight structural component that is either single load path or sized by durability or damage tolerance requirements.

3.20 Initial quality.

Initial quality is a measure of the condition of the aircraft structure relative to flaws, defects, or other discrepancies in the basic materials or introduced during manufacture of the aircraft structure.

3.21 Inspectability.

Inspectability means that materials and manufactured components (relative to geometry and access) which result from the application of processes and joining methods can be reliably inspected for applicable sources and types of structural flaws using available inspection procedures that meet the minimum probability of detection requirements.

3.22 Maintenance-critical part.

As shown in figure 1, a maintenance-critical part is a structural component whose failure will not cause a safety-of-flight condition but is sized by durability requirements and would not be economical to repair or replace.

3.23 Mission-critical part.

As shown in figure 1, a mission-critical part is a structural component in which damage or failure could result in the inability to meet critical mission requirements or could result in a significant increase in vulnerability.

3.24 Multiple load path.

Multiple load path is structural redundancy in which the applied loads are distributed to other load carrying members in the event of failure of individual elements.

3.25 Non-Destructive inspection (NDI).
Non-Destructive Inspection describes an inspection process or technique designed to reveal the condition at or beneath the external surface of a part or material without adversely affecting the material or part being inspected. NDI generally refers to inspections that are conducted using equipment that is not part of or permanently affixed to the article being inspected. NDI techniques such as ultrasonic or eddy current inspection utilize specialty ground support equipment operated by trained personnel to reveal material distress that may be missed by visual inspection alone.

3.26 Onset of widespread fatigue damage (WFD).
Onset of widespread fatigue damage is the point at which there are cracks at multiple structural details, and these are of sufficient size and density, such that the structure will no longer meet its damage tolerance requirement (e.g., maintaining required residual strength after partial structural failure).

3.27 Probability of detection (POD).
A POD is a statistical measurement of the likelihood, with a specified confidence level, of finding a flaw of a defined size using a specific inspection technique.

3.28 Producibility.
Producibility means that materials, processes, and/or joining methods are able to support current and future production rates without adversely affecting costs and/or quality.

3.29 Risk analysis.
Risk analysis is an evaluation of a potential hazard severity and probability of occurrence. For aircraft structural applications, the potential hazards include structural failures that can cause injury or death to personnel, damage to or loss of the aircraft, or reduction of mission readiness/availability.

3.30 Rotorcraft dynamic component.
A rotorcraft dynamic component is a structural part of the rotorcraft's drive train or lift system that experiences dynamic loading.

3.31 Safe-life.
Safe-life of a structure is that number of events such as flights, landings, or flight hours, during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue cracking.

3.32 Safety-of-flight structure.
Safety-of-flight structure is that structure whose failure could cause loss of the aircraft or aircrew, or cause inadvertent store release. The loss could occur either immediately upon failure or subsequently if the failure remained undetected.

3.33 Single load path.
Single load path is the distribution of applied loads through a single member, the failure of which would result in the loss of the structural capability to carry the applied loads.

3.34 Slow damage growth structure.
Slow damage growth structure is structure in which damage is not allowed to attain the critical size required for unstable rapid damage propagation. Safety is assured through slow damage growth for specified periods of usage depending upon the degree of inspectability. The strength of slow damage growth structure with damage present is not degraded below a specified limit for the period of unrepaired service usage.

3.35 Stability.
Stability means that materials, processes, and joining methods have matured to where consistent and repeatable quality, and predictable costs have been achieved to meet system production requirements. Also, process parameters and methods are understood, and robust and documented approaches for control of these factors (i.e., specifications) exist.

3.36 Structural health monitoring (SHM).
The process of monitoring, detecting and isolating the onset and/or growth of structural damage (i.e. cracks, flaws, corrosion, disbonds, delaminations, and/or other features that degrade, or have the potential to degrade, the performance of the affected structure) during operational usage. Direct SHM involves the use of in-situ sensing devices that may or may not complement NDI and are capable of detecting flaws/faults in current material condition, i.e. the formation and growth of defects. Indirect SHM involves the monitoring of loads / mechanical strains / environments and the conversion of these quantities to stress by parametric analysis.

3.37 Structural integrity.
Structural integrity is the condition which exists when a structure is sound and unimpaired in providing the desired level of structural safety, performance, durability, and supportability.

3.38 Structural operating mechanisms.
Structural operating mechanisms are those operating, articulating, and control mechanisms which transmit structural forces during actuation and movement of structural surfaces and elements.

3.39 Supportability.
Supportability means that thermal, environmental, and mechanical deterioration of materials and structures fabricated using the selected manufacturing processes and joining methods have been identified and that acceptable quality and cost-effective preventive methods and/or in-service repair methods are either available or can be developed in a timely manner.

4. GENERAL REQUIREMENTS

4.1 ASIP goal and objectives.

The effectiveness of any military force depends, in part, on the safety and operational readiness of its weapon systems. One major item of an aircraft system that affects its operational readiness is the condition of the aircraft structure. Its capabilities, condition, and operational limitations must be established to maintain operational readiness. Potential structural or material problems must be identified early in the life-cycle to minimize their impact on the operational force. In addition, a preventive maintenance program must be developed and implemented to provide for the orderly scheduling of inspections and replacement or repair of life-limited elements of the aircraft structure. The overall program to provide USAF aircraft with the required aircraft structural characteristics is referred to as the Aircraft Structural Integrity Program, or “ASIP.”

The goal of the ASIP is to ensure the desired level of structural safety, performance, durability, and supportability with the least possible economic burden throughout the aircraft’s design service life.

The objectives of the ASIP are to:

a. define the structural integrity requirements associated with meeting Operational Safety, Suitability and Effectiveness requirements;
b. establish, evaluate, substantiate, and certify the structural integrity of aircraft structures;
c. acquire, evaluate, and apply usage and maintenance data to ensure the continued structural integrity of operational aircraft;
d. provide quantitative information for decisions on force structure planning, inspection, modification priorities, risk management, expected life cycle costs and related operational and support issues; and
e. provide a basis to improve structural criteria and methods of design, evaluation, and substantiation for future aircraft systems and modifications.

4.2 Primary tasks.

The ASIP consists of the following five, interrelated functional tasks as delineated in table I and table II, and on figure 2 and figure 3:

a. Task I (Design Information). Task I is development of those criteria which must be applied during design to ensure the overall program goals will be met.
b. Task II (Design Analysis and Development Testing). Task II includes the characterization of the environment in which the aircraft must operate, the initial testing of materials, components, and assemblies, and the analysis of the aircraft design.
c. Task III (Full-Scale Testing). Task III consists of flight and laboratory tests of the aircraft structure to assist in determining the structural adequacy of the analysis and design.
d. Task IV (Certification & Force Management Development). Task IV consists of the analyses that lead to certification of the aircraft structure as well as the

development of the processes and procedures that will be used to manage force operations (inspections, maintenance, modifications, damage assessments, risk analysis, etc.) when the aircraft enters the inventory.

e. Task V (Force Management Execution). Task V executes the processes and procedures developed under Task IV to ensure structural integrity throughout the life of each individual aircraft. This task may involve revisiting elements of earlier tasks, particularly if the service life requirement is extended or if the aircraft is modified.

TABLE I. USAF Aircraft Structural Integrity Program Tasks.

Note: Beige boxes identify modified sections. Yellow boxes are entirely new sections.

TASK I DESIGN INFORMATION	TASK II DESIGN ANALYSES & DEVELOPMENT TESTING	TASK III FULL-SCALE TESTING	TASK IV CERTIFICATION & FORCE MANAGEMENT DEVELOPMENT	TASK V FORCE MANAGEMENT EXECUTION
5.1.1 ASIP Master Plan	5.2.1 Material and Joint Allowables Testing	5.3.1 Static Tests	5.4.1 Certification Analyses	5.5.1 Individual Aircraft Tracking (IAT) Program
5.1.2 Design Service Life & Design Usage	5.2.2 Loads Analysis	5.3.2 First Flight Verification Ground Tests	5.4.2 Strength Summary & Operating Restrictions (SSOR)	5.5.2 Rotorcraft Dynamic Component Tracking (RDCT) Program
5.1.3 Structural Design Criteria	5.2.3 Design Service Loads Spectra	5.3.3 Flight Tests	5.4.3 Force Structural Maintenance Plan (FSMP)	5.5.3 Loads/Environment Spectra Survey (L/ESS)
5.1.4 Durability and Damage Tolerance Control Program	5.2.4 Design Chemical/Thermal Environment Spectra	5.3.4 Durability Tests	5.4.4 Loads/Environment Spectra Survey (L/ESS) Development	5.5.4 ASIP Manual
5.1.5 Corrosion Prevention & Control Program (CPCP)	5.2.5 Stress Analysis	5.3.5 Damage Tolerance Tests	5.4.5 Individual Aircraft Tracking (IAT) Program Development	5.5.5 Aircraft Structural Records
5.1.6 Nondestructive Inspection Program	5.2.6 Damage Tolerance Analysis	5.3.6 Climatic Tests	5.4.6 Rotorcraft Dynamic Component Tracking (RDCT) Program Development	5.5.6 Force Management Updates
5.1.7 Structural Health Monitoring Program	5.2.7 Durability Analysis	5.3.7 Interpretation and Evaluation of Test Results		5.5.7 Recertification

5.1.8 Selection of Materials, Processes, Joining Methods, & Structural Concepts	*5.2.8* Corrosion Assessment
	5.2.9 Sonic Fatigue Analysis *5.2.10* Vibration Analysis *5.2.11* Aeroelastic and Aeroservoelastic Analysis *5.2.12* Mass Properties Analysis *5.2.13* Survivability Analysis *5.2.14* Design Development Tests *5.2.15* Production NDI Capability Assessment *5.2.16* Production SHM System Capability Assessment *5.2.17* Initial Risk Analysis

TABLE II. USAF ASIP Tasks and elements aligned with U.S. Department of Defense acquisition events.

	MILESTONES >>>	A	B						C		
	ACQUISITION PHASES >>>	ech Developmer	System Development & Demonstration (SDD)						luction & Deployrerations & Support		
	REVIEWS & AUDITS >>> (per MIL-STD-1521: Cancelled; cited for reference)	SRR System Requirements Review	SDR [SFR] System Design Review {a.k.a. System Functional Review]	PDR Preliminary Design Review	CDR Critical Design Review	TRR Test Readiness Review	FCA [SVR] Functional Configuration Audit [a.k.a. System Verification Review]	PRR Production Readiness Review	PCA Physical Configuration Audit		
I -Design Information	5.1.1 ASIP Master Plan	INITIAL	UPDATE	UPDATE	UPDATE	UPDATE	UPDATE	UPDATE	UPDATE	UPDATE	
	5.1.2 Design Service Life and Design Usage	FINAL									
	5.1.3 Structural Design Criteria	INITIAL	UPDATE								
	5.1.4 Damage Tolerance & Durability Control Program	INITIAL	UPDATE	UPDATE	UPDATE						<<< REVIS
	5.1.5 Corrosion Prevention & Control Program	INITIAL	UPDATE	UPDATE	UPDATE						<<< REVIS
	5.1.6 Nondestructive Inspection Program	INITIAL	UPDATE	UPDATE	UPDATE						<<< REVIS
	5.1.7 Structural Health Monitoring Program	INITIAL	UPDATE	UPDATE	UPDATE						<<< NEW
	5.1.8 Selection of Materials, Processes, Joining Methods, & Structural Concepts	INITIAL	UPDATE								<<< REVIS
II - Design Analyses & Development Testing	5.2.1 Material & Joint Allowables		INITIAL	UPDATE	UPDATE						
	5.2.2 Loads Analysis		INITIAL	UPDATE	UPDATE						
	5.2.3 Design Service Loads Spectra		INITIAL	UPDATE	UPDATE						
	5.2.4 Design Chemical/Thermal Environment Spectra		INITIAL	UPDATE	UPDATE						
	5.2.5 Stress Analysis		INITIAL	UPDATE	UPDATE						
	5.2.6 Damage Tolerance Analysis			INITIAL	UPDATE						
	5.2.7 Durability Analysis			INITIAL	UPDATE						
	5.2.8 Corrosion Assessment				INITIAL						
	5.2.9 Sonic Fatigue Analysis				INITIAL						
	5.2.10 Vibration Analysis				INITIAL						
	5.2.11 Aeroelastic & Aeroservolelastic Analysis			INITIAL	UPDATE						
	5.2.12 Mass Properties Analysis		INITIAL	UPDATE	UPDATE						
	5.2.13 Survivability Analysis				INITIAL						
	5.2.14 Design Development Tests		INITIAL	UPDATE	UPDATE						
	5.2.15 Production NDI Capability Assessment				FINAL						
	5.2.16 Production SHM System Capability Assessment				FINAL						<<< NEW
	5.2.17 Initial Risk Analysis				INITIAL						<<< REVIS

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TABLE II. USAF ASIP Tasks and elements aligned with U.S. Department of Defense acquisition events – Continued.

III - Full Scale Test	5.3.1 Static Tests					INITIAL	CONTINUE				
	5.3.2. First Flight Verification Ground Tests					FINAL					
	5.3.3 Flight Tests						FINAL				<<< REVIS
	5.3.4 Durability Tests						FINAL				<<< REVIS
	5.3.5 Damage Tolerance Tests						FINAL				
	5.3.6 Climatic Tests						FINAL				<<< REVIS
	5.3.7 Interpretation & Evaluation of Test Results						FINAL				<<< REVIS
IV - Certification & Force Management Development	5.4.1 Certification Analyses						FINAL				
	5.4.2 Strength Summary & Operating Restrictions (SSOR)							FINAL			
	5.4.3 Force Structural Maintenance Plan (FSMP)							FINAL			
	5.4.4 Loads/ Environment Spectra Survey (L/ESS) Development							FINAL			<<< REVIS
	5.4.5 Individual Aircraft Tracking (IAT) Program Development							FINAL			<<< REVIS
	5.4.6 Rotorcraft Dynamic Component Tracking (RDCT) Program Development							FINAL			<<< REVIS
V - Force Management Execution	5.5.1 Individual Aircraft Tracking (IAT) Program									EXECUTE	<<< REVIS
	5.5.2 Rotorcraft Dynamic Component Tracking (RDCT) Program									EXECUTE	
	5.5.3 Loads/Environment Spectra Survey (L/ESS)									EXECUTE	<<< REVIS
	5.5.4 ASIP Manual									CREATE/MAINTAIN	<<< REVIS
	5.5.5 Aircraft Structural Records									CREATE/MAINTAIN	<<< REVIS
	5.5.6 Force Management Updates									UPDATE	<<< REVIS
	5.5.7 Recertification									CREATE/UPDATE	
	REVIEWS & AUDITS >>> (per MIL-STD-1521)	SRR	SDR [SFR]	PDR	CDR	TRR	FCA [SVR]	PRR	PCA		
	ACQUISITION PHASES >>>	ech Developmer	System Development & Demonstration (SDD)						luction & Deployerations & Support		
	MILESTONES >>>	A	B						C		

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5. DETAILED REQUIREMENTS

Detailed guidance for the establishment and verification of aircraft structural requirements and for the planning and execution of ASIP tasks is documented in JSSG 2006, Aircraft Structures.

5.1 Design information (Task I).

The design information task encompasses those efforts required to apply the existing theoretical, experimental, applied research, and operational experience to specific criteria for materials selection and structural design for an aircraft. The objective is to ensure appropriate criteria and planned usage characteristics are applied to an aircraft's design to meet specific operational requirements. This task begins as early as possible in the Technology Development phase and is finalized in subsequent phases of the aircraft's life cycle.

5.1.1 ASIP Master Plan.

The ASIP Manager shall translate the requirements defined by this standard and AFI 63-1001 into a program for each aircraft and document these in the ASIP Master Plan. Each aircraft program must have an ASIP Master Plan. This plan shall be integrated into the Integrated Master Plan (IMP) and Integrated Master Schedule (IMS). The purpose of the ASIP Master Plan is to define and document the specific approach to accomplish the various ASIP tasks throughout the life-cycle of each individual aircraft. The plan shall depict the time-phased scheduling and integration of all required ASIP tasks for design, development, certification, Structural Health Monitoring, damage surveillance, and tracking of the aircraft structure. The plan shall also include discussion of unique features, exceptions to this standard and the associated rationale including risk assessments, and any problems anticipated in the execution of the plan. The development of the schedule shall consider all interfaces, the impact of schedule delays (e.g., delays due to test failure), mechanisms for recovery programming, and other problem areas.

5.1.1.1 Tailoring.

The ASIP Master Plan shall specify how an ASIP is tailored for a specific aircraft program.

5.1.1.2 Approval.

For all acquisition programs, the initial ASIP Master Plan shall be written and approved prior to the System Requirements Review (SRR).

5.1.1.3 Updates.

The ASIP Master Plan shall be updated annually throughout the service life of the aircraft.

5.1.1.4 Responsibility.

Air Force Instruction 63-1001 describes the organizations responsible for the creation, review, and approval of the ASIP Master Plan.

5.1.2 Design service life and design usage.
The USAF shall provide the design service life and design usage/environments as part of the contract. These data shall be used in the initial design and analysis for strength, rigidity, durability, corrosion prevention and control, damage tolerance, etc. The design service life and design usage/environment shall be established through close coordination between the acquisition and operational organizations. Design mission profiles, mission mixes, and environmental exposure mixes which are realistic estimates of expected service usage shall be established based on aircraft requirements.

5.1.3 Structural design criteria.
Detailed structural design criteria for the specific aircraft shall be established in accordance with the requirements of the applicable contracts. These shall include design criteria for loads, dynamics, strength, durability, damage tolerance, and mass properties.

5.1.3.1 Loads criteria.
Criteria shall be established such that all critical limit load conditions are developed. These limit loads are those which can result from authorized ground and flight usage of the aircraft including maintenance activity, system failures from which recovery is expected, and those that occur within the design service life. Ultimate loads for the aircraft shall be obtained by multiplying the limit loads by the appropriate factor of safety.

5.1.3.2 Dynamics criteria.
Criteria shall be established to ensure the aircraft in all configurations including store carriage is free from flutter, whirl flutter, divergence, and other related aeroelastic or aeroservoelastic instabilities for all combinations of altitude and speed within the approved flight envelope by the required airspeed margin of safety. Criteria shall be established such that the aircraft structure can withstand the aeroacoustic loads and vibrations due to aerodynamic and mechanical excitations throughout the design service life.

5.1.3.3 Strength criteria.
Criteria shall be established to ensure the aircraft structure has adequate strength capability. This capability requires that, for the design environments, no detrimental deformation or damage occurs at 115-percent design limit loads and no structural failure occurs at design ultimate loads.

5.1.3.4 Durability criteria.
Criteria shall be established to ensure the aircraft structure can achieve the design service life and that inservice maintenance is economically viable. In addition, durability criteria shall be established to ensure the aircraft structure can achieve the damage tolerance criteria described in 5.1.3.5. Durability criteria apply to all airframe structural components and shall include criteria that pertain to the onset of WFD as described in 5.1.3.4.1 and economic life as described in 5.1.3.4.2.

5.1.3.4.1 Onset of WFD.
Criteria shall be established to ensure the onset of WFD does not occur within the design service life of the aircraft structure. The onset of WFD is the end of the service life for the affected component. Fullscale durability testing described in 5.3.4 shall demonstrate that the onset of WFD occurs at a time equal to or greater than the design service life by the specified margin.

5.1.3.4.2 Economic life.
Additional criteria shall be established to ensure the aircraft structure's economic life as defined in 3.12 is greater than the design service life by the specified margin. This shall be demonstrated by the full-scale durability test described in 5.3.4.

5.1.3.5 Damage tolerance criteria.
Criteria shall be established to ensure the aircraft structure can safely withstand undetected flaws, corrosion, impact damage, and other types of damage throughout its design service life. The damage tolerance criteria shall be applied to all safety-of-flight structure and other selected structure. Criteria shall consider establishment of a minimum critical flaw size for those locations which are difficult to inspect or detect with SHM. The damage tolerance evaluation criteria for rotary-wing aircraft dynamic components are addressed in 5.1.3.5.2.

5.1.3.5.1 Damage tolerance design concepts.
The aircraft structural damage tolerance design shall be categorized into either of the general design concepts which follow:

a. fail-safe concepts where the required residual strength of the remaining intact structure shall be maintained for a period of unrepaired usage through the use of multiple load paths or damage arrest features after a failure or partial failure. The period of unrepaired usage necessary to achieve fail-safety must be long enough to ensure the failure or partial failure will be detected either visually, or by an SHM system that has been fully demonstrated and qualified for the structure in question, and repaired prior to the failure of the remaining intact structure. b. slow damage growth concepts where flaws, defects, or other damage are not allowed to attain the size required for unstable, rapid propagation failure. This concept must be used in single-load-path and non-fail-safe multiple load path structures. No significant growth which results from manufacturing defects or from damage due to high-energy impact shall be allowed for composite structures.

5.1.3.5.2 Special applications.
The safe-life design methodology may be used on a limited basis. It is expected that it will be used to establish replacement times for some specifically-approved structural components (e.g., landing gear components and rotorcraft dynamic components). Damage tolerance evaluations are required for all safelife designed components and other selected structure. These evaluations shall define critical areas, fracture characteristics, stress spectra, maximum probable initial material and/or manufacturing defect sizes, and options for either eliminating defective components or otherwise mitigating threats to

structural safety. Such options may include design features, manufacturing processes, inspections or structural health monitoring.
Additionally, the damage tolerance evaluation shall establish individual aircraft tracking requirements so that the safe-life component replacement times and any scheduled safety inspections can be adjusted based on actual usage. Use of a safe-life approach for a structural component must be identified in the ASIP Master Plan.

5.1.3.6 Mass properties criteria.
Criteria shall be established to ensure the aircraft can accommodate aerodynamic, center of gravity, and inertia changes which result from fuel usage, store expenditure, asymmetric fuel and store loading, fuel migration at high angles of attack and roll rates, and aerial refueling.

5.1.4 Durability and Damage Tolerance Control Program.
A Durability and Damage Tolerance Control Program shall be established for the aircraft structure. This program shall identify and define all the tasks necessary to ensure compliance with the durability requirements as described in 5.1.3.4 and the damage tolerance requirements as described in 5.1.3.5. The disciplines of fracture mechanics, fatigue, materials and processes selection, environmental protection, corrosion prevention and control, design, manufacturing, quality control, nondestructive inspection, structural health monitoring and probabilistic methods shall be considered when the durability and damage tolerance control processes are developed. This program shall include the requirement to perform durability and damage tolerance design concept, material, weight, performance, and cost trade studies early during the aircraft's design so as to obtain structurally-efficient and cost-effective designs.

5.1.4.1 Durability and Damage Tolerance Control Plan.
A Durability and Damage Tolerance Control Plan that is consistent with the design service life shall be prepared and executed throughout the System Development & Demonstration and the Production & Deployment phases. The plan shall establish a Durability and Damage Tolerance Control Board (DDTCB) responsible for establishment and oversight of the administration of the specific controls that will be applied in accordance with the plan. The board shall be comprised of representatives from engineering, manufacturing, quality assurance, and others involved in the design, engineering development, and production of the aircraft structure. The board's decisions are subject to USAF approval.

5.1.4.2 Critical part/process selection and controls.
Criteria shall be established to select aircraft structural critical parts/processes and the controls for these critical parts/processes. The DDTCB described in 5.1.4.1 shall oversee this selection and control process. The impact on safety-of-flight, mission completion, and production and maintenance costs shall be considered in the selection of critical parts/processes. Figure 1, along with the analyses described in 5.2, shall be part of the selection process and establishment of controls. The DDTCB shall ensure the critical part/process list is updated as the design matures.

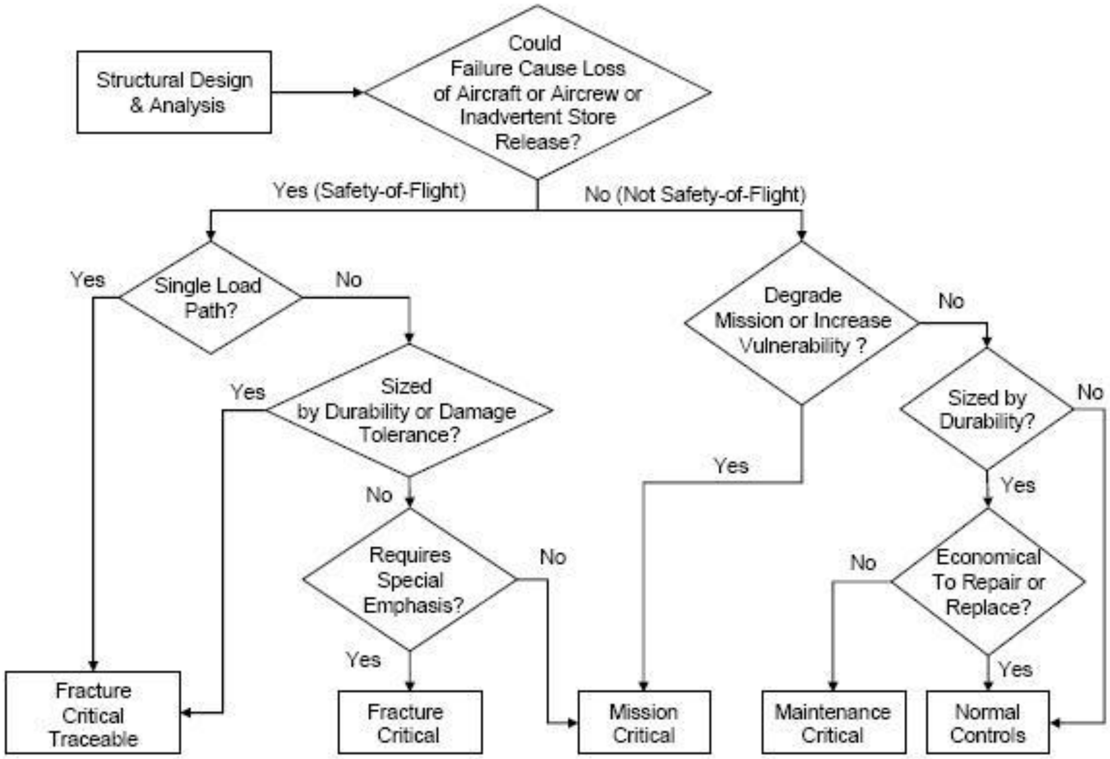

FIGURE 1. Critical part selection flow chart.

5.1.5 Corrosion Prevention and Control Program (CPCP).

A Corrosion Prevention and Control Program shall be established for the aircraft structure. The program shall establish a Corrosion Prevention Advisory Board (CPAB) responsible for establishment and oversight of the execution of the program. The board shall be comprised of representatives from engineering, manufacturing, quality assurance, and others involved in the design, engineering development, and production of the aircraft structure. The board's decisions are subject to USAF approval. Corrosion prevention shall be a primary consideration in the development and implementation of the durability and damage tolerance control process and the force management process. Materials and processes, finishes, coatings, and films which have been proven in service or by comparative testing in the laboratory shall be selected to prevent corrosion as described by the Corrosion Prevention and Control Plan described in 5.1.5.1. Results of the susceptibility to corrosion evaluation described in 5.1.5.2 shall be used to control the impact of corrosion damage. Corrosion prevention and control guidelines are provided in JSSG-2006, the DoD Corrosion Prevention and Control Planning Guidebook (Spiral II), MIL-HDBK-1568, DFARS 207.105(b)(13)(ii) Oct 04, AFPD 21-1, and AFI 21-105.

5.1.5.1 Corrosion Prevention and Control Plan.

A Corrosion Prevention and Control Plan shall be prepared and corrosion prevention and control processes shall be used in accordance with this standard, MIL-HDBK-1568, and JSSG-2006. The plan shall be consistent with the design service life and shall define corrosion prevention and control requirements and considerations for the System

Development & Demonstration and Production & Deployment phases. The plan shall specify actions to delay the onset of corrosion and minimize corrosion maintenance costs through the selection of materials, fabrication techniques, sealants, protective coatings, design features, SHM technologies and other measures that minimize the potential for corrosion throughout the structure.

5.1.5.2 Evaluation of corrosion susceptibility.

An evaluation of the susceptibility of the aircraft structure to corrosion shall be conducted as part of the CPCP. The evaluation shall identify locations where the structure might be susceptible to corrosion and the expected type(s) of corrosion (e.g., exfoliation, uniform, crevice, intergranular, and stress-corrosion cracking, etc.) that could occur at these locations. To identify potential corrosion damage locations, the evaluation shall account for the materials, manufacturing processes, corrosion prevention systems (e.g., coatings, sealants, etc.), preventative maintenance approaches (e.g., hangaring, wash cycles, wash fluids, etc.), the inspectability of the location, application of SHM and structural fabrication techniques as well as the expected operational environments to which the aircraft are subjected.

5.1.6 Nondestructive Inspection (NDI) Program.

An NDI Program shall be established in accordance with MIL-HDBK-6870 and consistent with direction in AFI 21-105. The NDI Program will consider and implement appropriate nondestructive inspection processes into all phases of the program (design, engineering development, production, and in-service operation). The program shall establish a Nondestructive Inspection Requirements Review Board (NDIRRB) responsible for oversight and execution the program. The NDIRRB shall be formed early in the design phase to review and assess product form concepts for inspectability in terms of production process control and quality monitoring. The NDIRRB shall also be responsible for review and approval of inspection methods and detectability assumptions implemented in the Force Structural Maintenance Plan. The board’s decisions are subject to USAF approval.

5.1.7 Structural Health Monitoring (SHM) Program.

If the maintenance planning for all or any part of the aircraft structure will be informed by structural health monitoring, then a structural health monitoring (SHM) Program shall be established. This program shall identify and define all of the tasks necessary for the development, test, qualification, and deployment of a SHM system. System qualification shall include demonstration of capability (in terms of POD and false call rates for specified damage types) as well as demonstration of durability (in terms of sensor life and maintenance requirements). The purpose of the SHM system will be to ensure compliance with the durability requirements as described in 5.1.3.4 and the damage tolerance requirements as described in 5.1.3.5 and to enable both condition-based and inspection-based force structural maintenance planning. The SHM Program will address all phases of the aircraft program (design, engineering development, production, and in-service operation). The program shall establish a Structural Health Monitoring Requirements Review Board (SHMRRB) responsible for oversight and execution the program. The SHMRRB shall be formed early in the design phase to review and assess material selections, structural design concepts for compatibility with SHM sensor capability and requirements and SHM system architecture and requirements. The SHMRRB shall also be responsible for review and approval of SHM sensor selection and capability assumptions implemented in the Force Structural Maintenance Plan. The board's decisions are subject to USAF approval.

5.1.8 Selection of materials, processes, joining methods, and structural concepts.
Materials, processes, joining methods, and structural concepts shall be selected to result in a structurally efficient, cost-effective aircraft structure that meets the strength, rigidity, observability, durability, and damage tolerance requirements of the applicable specifications. Prior to a commitment to new materials, processes, joining methods, and/or structural concepts (i.e., those not previously used in the military and/or commercial aviation industry), an evaluation based on their stability, producibility, inspectability, supportability, and mechanical and physical properties shall be performed. The risk associated with the selection of the new materials, processes, joining methods and/or structural concepts shall be estimated and risk mitigation actions defined. The trade studies performed as part of the durability and damage tolerance control described in 5.1.4 shall be a major driver in the final selection of materials, processes, joining methods, and structural concepts. The detailed rationale for the individual selections and any proposed risk mitigation actions shall be submitted in the proposals of prospective contractors. Each rationale and all supporting data shall become part of the design database after contract award and during the design of the aircraft.

5.1.8.1 Stability.
Maturity of material, process, and joining method choices shall be assessed by determining if these choices result in consistent and repeatable quality and if predicable costs are likely to be achieved to meet production requirements. This can be demonstrated by robust and documented approaches for controlling process parameters/methods, i.e., specifications exist.

5.1.8.2 Producibility.
Quality control shall be ensured through the use of appropriate process control measures employed during the manufacture of the aircraft structure.

5.1.8.3 Mechanical and physical properties.
Mechanical and physical properties include all the key mechanical and physical properties that have been determined in the appropriate environments in the as-fabricated condition using the manufacturing processes and joining methods that will be utilized. Key mechanical properties include but are not limited to: strength, elongation, fracture toughness, damage growth rates, stress corrosion and fatigue crack growth thresholds. Key physical properties include but are not limited to: density, corrosion resistance, defect population, reflectivity, and surface roughness.

5.1.8.4 Supportability.
Legacy experience and health/environmental regulations must be considered when the capability of proposed approaches is evaluated. The selection of preventive and repair

methods shall consider the potential for repeated use on individual aircraft. These preventive and repair methods include corrosion preventive coatings, hot bonding of composites, mechanical fastened repairs, field welding and stress relief, grinding, shot peening, etc.

5.1.8.5 Risk mitigation actions.
Risk mitigation actions shall be defined and implemented in the program based on an estimate of the level of risk associated with the selection of the new materials, processes, joining methods, and/or structural concepts. The specific actions required will depend on the classification of the structural component (e.g., safety-of-flight structure), the design concept (i.e., safe-life, fail-safe, or slow damage growth), the estimated risk level and the effectiveness of the means by which to mitigate failure risk with SHM. Possible risk mitigation actions include: using higher factors of safety, fabrication and durability testing of one or more large structural components in the appropriate thermal/chemical environments, special in-service inspections of test and/or operational aircraft, SHM technology implementation and special in-process testing (such as periodic strength proof testing of bonded joints) conducted throughout the production of the aircraft. In addition, the use of fail-safe design concepts is preferred when flight safety depends on the integrity of the bonded joint.

5.2 Design analyses & development testing (Task II).
The objectives of the design analyses and development tests task are to: 1) determine the environments in which the aircraft structure must operate (load, temperature, chemical, abrasive, and vibratory and aeroacoustic environment), 2) perform preliminary and final analyses and tests based on these environments, and 3) size the aircraft structure to meet the strength, rigidity, damage tolerance, and durability requirements. Test plans, procedures, and schedules shall be approved by the USAF.

5.2.1 Material and joint allowables.
Material and joint allowables data identified in FAA MMPDS-Handbook, Damage Tolerant Design Handbook, CINDAS Aerospace Structural Metals Handbook, and CINDAS Structural Alloys Handbook may be used to support the use of existing materials in design analyses. Other data sources may also be used, but shall first be reviewed by the USAF and the contractor. Experimental programs to obtain the data and generate analysis test data shall be formulated and performed for new materials and those existing materials for which there are insufficient data available. The variability in material properties shall be considered when material and joint allowables are established.

5.2.2 Loads analysis.
Loads analysis shall determine the magnitude and distribution of significant static and dynamic loads which the aircraft structure may encounter when operated within the envelope established by the structural design criteria. This analysis consists of a determination of the flight loads, ground loads, powerplant loads, control system loads, and weapon effects. When applicable, this analysis shall include the effects of temperature, aeroelasticity, and dynamic response of the aircraft structure.

5.2.3 Design service loads spectra.
Design service loads spectra shall be developed to establish the distribution, frequency, and sequencing of loadings that the aircraft structure will experience based on the design service life and usage. The design service loads spectra and chemical/thermal environment spectra as defined in 5.2.4 shall be used to develop flight-by-flight stress/environment spectra, as appropriate, to support the analyses and tests described herein.

5.2.4 Design chemical/thermal environment spectra.
Design chemical/thermal environmental spectra shall be developed to establish the intensity, duration, frequency of occurrence, etc., of the environment which the aircraft structure will experience based on the design service life and usage.

5.2.5 Stress analysis.
A stress analysis shall include the analytical determination of the internal loads, stresses, strains, deformations, and margins-of-safety which result from the external loads and environments imposed on the aircraft structure. In addition to verification of strength, the stress analysis shall be used as a basis for durability and damage-tolerance analyses, selection of critical structural components for design development tests, material review actions, and selection of loading conditions to be used in the structural strength tests. The stress analysis shall be used as the basis to determine the adequacy of structural changes throughout the life of the aircraft and to determine the adequacy of the structure for new loading conditions which result from increased performance or new mission requirements. The stress analysis shall be revised to reflect any major changes to the aircraft structure or to the loading conditions applied to the aircraft structure.

5.2.6 Damage tolerance analysis.
Damage tolerance analysis shall be conducted to substantiate the ability of the structural components to comply with the detail requirements for damage tolerance. The design flight-by-flight stress/environment spectra based on the requirements of 5.2.3 and 5.2.4 shall be used in the damage growth analysis and verification tests. The calculations of critical flaw sizes, residual strengths, safe damage growth periods and inspection intervals shall be based on demonstrated NDI/SHM capabilities and existing fracture test data and basic fracture allowables data generated as a part of the design development test program.

5.2.7 Durability analysis.
Durability analysis shall be conducted to substantiate the ability of the structure to comply with the detail requirements for durability. The design flight-by-flight stress/environment spectra based on the requirements of 5.2.3 and 5.2.4 shall be used in the durability analysis and verification tests. Durability analysis shall be performed for all airframe structural components and shall include analysis that pertains to the onset of WFD as described in 5.2.7.1 and economic life as described in 5.2.7.2.

5.2.7.1 Onset of Widespread Fatigue Damage (WFD).

The analysis shall account for those factors which affect the time for typical-quality structure to experience the onset of WFD. These factors shall include initial quality and initial quality variations, chemical/thermal environment, load sequence and environment interaction effects, material property variations, and analytical uncertainties.

5.2.7.2 Economic life.

The analysis shall account for those factors that affect the time for cracks or equivalent damage to reach sizes large enough to necessitate maintenance actions.

5.2.8 Corrosion assessment.

An assessment shall be conducted to identify the failure modes associated with the type(s) of corrosion identified by the CPCP described in 5.1.5 and the structural integrity consequences associated with the failure modes. Special attention should be given to those safety-of-flight and mission-critical aircraft structural locations where corrosion damage could affect the onset of fatigue cracking or lead to stress corrosion cracking, and especially to those locations where corrosion could accelerate the onset of WFD. The assessment shall be utilized to evaluate accessibility for inspection, to determine applications for SHM technology insertion, establish rework limits, and ensure component replaceability (if necessary) in the design of the aircraft structure.

5.2.9 Sonic fatigue analysis.

Sonic fatigue analysis shall be conducted to ensure the aircraft structure is resistant to sonic fatigue cracking throughout the design service life. The analysis shall define the intensity of the aeroacoustic environment from potentially critical sources and shall determine the dynamic response, including significant thermal effects. Potentially critical sources include but are not limited to powerplant noise, aerodynamic noise in regions of turbulent and separated flow, exposed cavity resonance, and localized vibratory forces.

5.2.10 Vibration analysis.

Vibration analysis shall be conducted to predict the resultant environment in terms of vibration levels in various areas of the aircraft such as the crew compartment, cargo areas, equipment bays, etc. The vibration analyses, in conjunction with the durability analyses of 5.2.7, shall show that the structure in each of these areas is resistant to cracking due to vibratory loads throughout the design service life. In addition, the analyses shall show that the vibration levels are acceptable for the reliable performance of personnel and equipment throughout the design service life of the aircraft.

5.2.11 Aeroelastic and aeroservoelastic analysis.

Analysis shall be conducted to determine the characteristics of the aircraft for flutter, divergence, and other related aeroelastic or aeroservoelastic instabilities. The primary objective of the analysis is to evaluate potential aeroelastic and aeroservoelastic instabilities and substantiate the ability of the aircraft structure to meet the specified aeroelastic airspeed margins, damping requirements, and aeroservoelastic stability margins for all design conditions. Analysis for design failure conditions shall also be conducted.

5.2.12 Mass properties analysis.
A mass properties analysis shall be conducted to determine the aircraft weight and balance. This analysis shall be based on estimates of the aircraft's design, construction, and usage at the time of Initial Operational Capability (IOC). In addition, a Mass Properties Control and Management Plan (MPCMP) shall be established and implemented throughout the life of the aircraft. Detailed guidance may be found in the Society of Allied Weight Engineers Recommended Practice Number 7 (SAWE RP No. 7).

5.2.13 Survivability analysis.
Survivability analysis shall be conducted to ensure the aircraft structure can perform effectively in a combat environment.

5.2.13.1 Vulnerability analysis.
Vulnerability analysis shall be conducted to verify that the aircraft structure can withstand the operational loads after being damaged by specific threats.

5.2.13.2 Weapons effects analysis.
Weapons effects analysis shall be conducted to ensure the aircraft structure can withstand the loads due to thermal transients, overpressure, and gust associated with weapon detonation. Nuclear weapons effects analysis shall be conducted to determine the capability envelope for the aircraft structure and crew radiation protection for the specified range of variations of weapon delivery trajectories, weapon size, aircraft escape maneuvers, and the resulting damage limits.

5.2.14 Design development tests.
Design development tests shall be conducted to establish material, process, and joint allowables; to verify analysis methods and procedures; to obtain early evaluation of allowable stress levels, material selection, fastener systems, and the effect of the design chemical/thermal environment spectra; to establish aeroelastic and loads characteristics through wind tunnel tests; and to obtain early evaluation of the strength, durability, fatigue (sonic and vibratory), and damage tolerance capabilities of critical structural components and assemblies. Examples of design development tests are tests of coupons; small elements; splices and joints; panels; fittings; control system components and structural operating mechanisms; and major components such as wing carry through, horizontal tail spindles, wing pivots, and assemblies thereof. The plans shall consist of information such as rationale for selection of scope of tests; description of test articles, procedures, test loads and test duration; and analysis directed at the establishment of cost and schedule trade-offs used to develop the program.

5.2.14.1 Duration of durability tests.
Durability testing shall determine initial estimates of the onset of WFD and of the equivalent initial flaw size (EIFS) distribution.

5.2.14.2 Corrosion tests.
Corrosion testing shall be conducted to evaluate the effectiveness of the corrosion protection system to meet design service life requirements for the defined service environments. For corrosion protection system effectiveness, comparative tests on representative structure (including fasteners and dissimilar material contacts) shall be used to evaluate corrosion protection system choices. The comparative tests shall include corrosion protection systems used on legacy aircraft to ensure the new corrosion protection system provides adequate corrosion protection and to provide insight into the degree the new systems will protect aircraft.

5.2.15 Production NDI capability assessment.
The capability of nondestructive inspection processes used for production process monitoring and quality control of structural component shall be established to mitigate risk of missing defects. Special emphasis shall be given to fracture- and mission-critical parts. Capability demonstration of production NDI processes shall be addressed within the NDI Program.

5.2.16 Production SHM system capability assessment.
If the maintenance planning for all or any part of the aircraft structure will be informed by structural health monitoring, then the capability of the SHM system shall be established to mitigate risk of missing defects. Special emphasis shall be given to systems used to monitor fracture- and mission-critical parts. Capability / reliability demonstration of production SHM processes shall be addressed within the SHM Program. Both shall be demonstrated as a function of time / usage. Capability / reliability shall be quantified in terms of metrics (POD, false call rates, sensor failure rates, etc.) suitable for formal risk analyses. As determined practical and feasible by the SHMRRB, component level SHM system capability tests shall be conducted to mitigate the risk that full-scale testing does not yield measurable fatigue, impact, corrosion or other damage in monitored regions. In addition, component level durability tests shall be conducted in order to establish sensor / system durability.

5.2.17 Initial risk analysis.
An initial risk analysis shall be performed using the EIFS distribution developed under 5.2.14.1 and 5.3.4 and combined, when appropriate, with data from similar aircraft. A primary objective of this analysis is to demonstrate a low risk of both WFD and loss of fail-safety during the design service life when the aircraft is flown to the design loads/environment spectrum. Also, the analysis should estimate the time beyond the design service life when the risk of loss of fail-safety will become unacceptable. For non-failsafe structures, the analysis should estimate the time beyond the design service life when required safety inspections and/or modifications would result in an unacceptably high risk of aircraft unavailability and/or adverse economic consequences. All significant variables impacting risk shall be included in the risk analysis. Examples of such variables include: EIFS distribution, load spectra, chemical and thermal environment, material properties, the NDI probability of detection (POD) and the SHM sensor / system probability of detection (POD).

5.3 Full-scale testing (Task III).

The objective of this task is to assist in the determination of the structural adequacy of the design through a series of ground and flight tests. Test plans, procedures, and schedules shall be approved by the USAF. Test results shall be used to validate analytical design data and to verify requirements are achieved.

5.3.1 Static tests.

A static test program shall be conducted on an instrumented aircraft using simulated loads derived from critical flight and ground handling conditions. Thermal environment effects shall be simulated in addition to the load application on aircraft structures where operational environments impose significant thermal effects. The primary purpose of the static test program is to verify the static strength analyses and the design ultimate strength capabilities of the aircraft structure. Deletion of the full-scale ultimate load static tests is generally unacceptable. However, a separate full-scale static test is not required if any of the following conditions are met and specifically approved by the acquisition authority:

a. where it is shown that the aircraft structure and its loading are essentially the same as that of a previous aircraft structure which was verified by full-scale tests; or

b. where it is shown that the strength margins (particularly for stability-critical structures) have been demonstrated by major assembly (i.e., entire wing, fuselage, and/or empennage component) tests; or

c. strength demonstration proof tests are performed to 115 percent of design limit load on every flight aircraft to be operated. These proof tests shall demonstrate that deformation requirements have been met and shall validate the accuracy of the strength predictive methods.

Major repairs, extensive reworks and refurbishments, and component modifications which alter the structural load paths, or which represent significant changes in structural concept, shall require a static ultimate load test of the affected component.

5.3.1.1 Selection of test article.

The test article shall be an early System Development & Demonstration phase test aircraft structure and shall be representative of the operational configuration (including all significant structural details) and manufacturing processes. If there are significant design, material, or manufacturing changes between the test article and production aircraft, static tests of an additional article or selected components and assemblies thereof shall be required.

5.3.1.2 Schedule requirements.

Full-scale static tests and/or strength demonstration proof tests shall be scheduled such that the tests are completed in sufficient time to support removal of flight restrictions on flight test and operational aircraft in support of program requirements.

5.3.2 First flight verification ground tests.

The following verification tests shall be conducted prior to first flight.

5.3.2.1 Mass properties tests.
Mass properties tests shall be conducted to verify the aircraft weight and balance are as predicted and within limits for all design conditions.

5.3.2.2 Functional proof tests.
Functional proof tests shall be conducted to design limit load to demonstrate the functionality of flight critical structural systems, mechanisms, and components whose correct operation is necessary for safe flight. These tests shall demonstrate the deformation requirements have been met.

5.3.2.3 Pressure proof tests.
Each pressurized compartment of each pressurized flight aircraft shall be pressure proof tested to the maximum pressure limit loads. These proof tests shall demonstrate that deformation requirements have been met and shall validate the accuracy of the strength predictive methods.

5.3.2.4 Strength proof tests.
Strength proof tests of selected aircraft structural components and systems (e.g., flight control surfaces, hydraulic systems, etc.) shall be conducted when the full-scale static test schedule does not allow for adequate testing prior to first flight, when the full-scale test will not adequately demonstrate strength capability, or when flight restrictions to limit these component loads may be difficult to achieve without unreasonably restricting the aircraft.

5.3.2.5 Control surface rigidity and free-play tests.
Control surface rigidity and free-play tests shall be conducted to verify the flutter analysis as well as to ensure safe free-play limits. These tests should be conducted prior to ground vibration tests and should be conducted for both design failure and normal conditions. If mass balancing of controls surfaces is used to prevent any aeroelastic instability, stiffness tests of the mass balance attachments shall be conducted. In addition, the mass and inertia of the control surfaces shall be measured in support of the flutter analysis and to verify the mass property analysis.

5.3.2.6 Ground vibration tests.
Ground vibration tests shall be conducted to verify the natural frequencies, mode shapes, and structural damping of the aircraft. Test results are to be correlated against the structural model used in all aeroelastic analyses. Consideration of the aircraft supporting system is required to ensure rigid body modes of the aircraft do not interfere with the capture of aircraft elastic modes. To allow for changes in the structural models, component ground vibrations tests shall be conducted prior to aircraft assembly and well in advance of full-scale aircraft tests.

5.3.2.7 Aeroservoelastic tests.

Aeroservoelastic ground tests to include open-loop transfer (frequency response) tests and closed-loop coupling (structural resonance) tests shall be conducted to correlate and verify the aeroservoelastic analysis.

5.3.3 Flight tests.
Flight tests shall be conducted on a fully-instrumented aircraft. An additional aircraft, sufficiently late in the production program to ensure obtainment of the final configuration, shall be the backup aircraft for these flight tests and shall be instrumented similarly to the primary test aircraft. These tests shall include dynamic response, flutter, and aeroacoustic and vibration tests, as well as a flight and ground loads survey. These tests will serve to flight qualify any SHM equipment which is a part of the production configuration of the aircraft.

5.3.3.1 Flight and ground loads survey.
The flight and ground loads survey program shall consist of an instrumented and calibrated aircraft operated within and to the extremes of its limit structural design envelope to measure the resulting loads and, if appropriate, to also measure pertinent temperature profiles on the aircraft structure. Load measurements shall be made in a build-up fashion by the strain gage or pressure survey methods commensurate with the state-of-the-art, usually installed during production buildup. The objectives of the loads survey are to:

a. verify the structural loads and thermal analyses used in the design of the aircraft structure;

b. evaluate loading conditions which produce the critical structural load and temperature distribution; and

c. determine and define suspected new critical loading conditions which may be indicated by the investigations of structural flight conditions within the design-limit envelope.

5.3.3.2 Dynamic response tests.
The dynamic response tests shall consist of an instrumented and calibrated aircraft operated to measure the structural loads and inputs while flown through atmospheric turbulence; and during taxi, takeoff, towing, landing, refueling, store ejection, etc. The objectives shall be to obtain flight verification and evaluation of the elastic response characteristics of the structure to these dynamic load inputs.

5.3.3.3 Flutter tests.
Flight flutter tests shall be conducted to verify the aircraft structure is free from aeroelastic instabilities and has satisfactory damping throughout the operational flight envelope. Test aircraft should have sufficient instrumentation installed and acceptable methods of in-flight excitation shall be used to determine the frequency and amount of damping of the primary modes of interest at each flight test condition. The tests shall be performed with test data taken at predetermined test points, defined by Mach number and altitude, in a prescribed order of ascending criticality. For aircraft with a flight control augmentation system, flight aeroservoelastic stability tests shall be conducted in conjunction with flight flutter testing.

5.3.3.4 Aeroacoustic tests.
The aeroacoustic environments shall be measured on a full-scale aircraft to verify the acoustic loads/environment used in the sonic fatigue analysis. Measurements of sound pressure levels shall be made of those areas determined to be sonic-fatigue critical. Sufficient instrumentation shall be in place for both flight and ground operations which produce the significant aeroacoustic loads.

5.3.3.5 Vibration tests.
Flight vibration tests shall be conducted to verify and correct analysis of the vibration environment. Measurements shall be made at a sufficient number of locations to define the vibration characteristics of the aircraft structure with the test results being the basis for equipment environmental requirements. In addition, the test results shall be used to demonstrate that vibration control measures are adequate to prevent cracking and to provide reliable performance of personnel and equipment throughout the design service life.

5.3.4 Durability tests.
A durability test program shall be conducted on an instrumented aircraft using the repeated application of the flight-by-flight design service loads/environment spectrum. Thermal environment effects shall be simulated, along with the load application on aircraft structures where operational environments impose significant thermal effects. The objectives of the full-scale durability tests are to:

a. demonstrate that the onset of WFD does not occur within the design service life by the specified margin;
b. demonstrate that the economic life of the test article is equal to or greater than the design service life by the specified margin;
c. identify critical areas of the aircraft structure not previously identified by analysis or component testing;
d. provide a basis for special inspection and modification requirements for force aircraft;
e. demonstrate the capability of the SHM system to operate successfully under design loading/environmental conditions for the design service life; and to
f. obtain crack growth data to validate analysis methods and EIFS distribution data to support risk analyses. If no cracks are detected, or an insufficient number of cracks occur during the full-scale test, the data obtained from the design development testing described in 5.2.14 shall be used for verification.

Major component modifications which alter the structural load paths or which represent significant changes in structural concept shall require a durability test of a full-scale component.

5.3.4.1 Selection of test article.
The test article shall be an early System Development & Demonstration phase test aircraft structure and shall be representative of the operational configuration (including

all significant details) and manufacturing processes. It is not required that the test article include systems, but the article must include system attach structures and associated details representative of the operational configuration and manufacturing process. If there are significant design, material, or manufacturing changes between the test article and production aircraft, durability tests of an additional article or selected components and assemblies thereof shall be required.

5.3.4.2 Test scheduling and duration.
One lifetime of durability testing plus an inspection of critical structural areas shall be completed prior to a full production go-ahead decision. Two lifetimes of durability testing plus an inspection of critical structural areas shall be scheduled to be completed prior to delivery of the first production aircraft. If the economic life of the test article is reached prior to two lifetimes of durability testing, sufficient inspection in accordance with the inspection program described in 5.3.4.3 and data evaluation shall be completed prior to delivery of the first production aircraft to estimate the extent of required production changes and retrofit. It may be advantageous to continue testing beyond the minimum requirement to:

1) determine life-extension capabilities,
2) validate design-life capability for usage that is more severe than design usage,
3) validate repairs, modifications, inspection methods, and changes,
4) support damage-tolerance requirements, and
5) determine the onset of WFD.

In the event the original schedule for the production decision and production delivery milestones becomes incompatible with the above schedule requirements, a study shall be conducted to assess the technical risk and cost impacts of changing these milestones. An important consideration in the durability test program is that it be completed at the earliest practical time, but after Critical Design Review (CDR).

5.3.4.3 Inspection program.
An inspection program shall be conducted as an integral part of the full-scale aircraft structure durability test. The inspection program shall be approved by the USAF. The objectives of the inspection program shall be to detect damage as early as possible, to provide crack growth data, and to minimize the risk of a catastrophic failure during testing.

5.3.4.4 SHM system demonstration.
If the maintenance planning for all or any part of the aircraft structure will be informed by structural health monitoring, then a SHM system demonstration / validation shall be conducted as an integral part of the full-scale aircraft structure durability test. The objectives of the program shall be to: 1) demonstrate that the SHM system can detect damage at stages that are early enough to prevent catastrophic failure and permit remedial maintenance actions, 2) demonstrate that false positive rates remain below acceptable levels, 3) demonstrate that all non-field servicable SHM equipment remain functional for the design service life of the aircraft and that serviceable equipment (connectors, energy

sources, etc.) remain functional for a period deemed acceptable by the SHMRRB. In the event that the full scale durability test does not yield measurable damage or critical flaws, the SHM capability demonstration may rely on component level durability tests results as defined in Section 5.2.14.

5.3.4.5 Teardown inspection and evaluation.
At the end of the full-scale durability test, including any scheduled damage tolerance tests, a destructive teardown inspection program shall be conducted. This inspection shall include disassembly and laboratory-type inspection of those critical areas identified in design as well as additional critical structure identified during testing and during close visual examination while disassembly is performed. Fractographic examinations shall be conducted to obtain crack growth data and to assist in the assessment of the initial quality of the aircraft structure. The EIFS distribution shall be derived from the damage discovered during testing and the teardown inspection. The methods, procedures, and data used to determine the EIFS shall be documented and delivered to the USAF as part of the acquisition contract to serve as a basis to validate any future changes in analytical methods. Prior to teardown, consideration should be given to evaluation of the effectiveness of the anticipated NDI methods that may be applied to fielded aircraft.

5.3.5 Damage tolerance tests.
A damage tolerance test program shall be conducted using the repeated application of the flight-by-flight design service loads/environment spectrum. Thermal environment effects shall be simulated, along with the load application on aircraft structures where operational environments impose significant thermal effects. The intent shall be to conduct damage tolerance tests on existing test hardware. This may include use of components and assemblies of the design development tests as well as the full-scale static and durability test articles. When necessary, additional structural components and assemblies shall be selected, fabricated, and tested.

5.3.6 Climatic tests.
Full-scale system-level climatic testing shall be conducted to identify potential corrosion problems in the field. Identification of fluid sources, trapped fluid locations, and improper drain paths shall be performed to the maximum extent possible. The results of this testing shall provide initial input for corrosion-related tasks in the Force Structural Maintenance Plan described in 5.4.3.

5.3.7 Interpretation and evaluation of test results.
Each structural problem that occurs during the tests described by this standard shall be analyzed to determine the root cause, corrective actions, force implications, and estimated costs. Examples of structural problems include but are not limited to: analytical shortfalls (measured loads, stresses, vibrations, etc., that differ from predictions), failures, cracking, yielding, corrosion, etc. The scope of and interrelation between the various ASIP tasks within the interpretation and evaluation effort are illustrated in figure 2 and figure 3. The results of these evaluations shall define corrective actions required to demonstrate that the strength, rigidity, damage tolerance, and durability design requirements are met and the associated risk reduction is achieved. The cost, schedule, and other impacts which result

from correction of structural problems shall be used to make major program decisions such as major redesign, program cancellation, awards or penalties, and production aircraft buys. Structural modifications or changes derived from the results of the full-scale tests to meet the specified strength, rigidity, damage tolerance, and durability design requirements shall be substantiated by subsequent tests of components, assemblies, or full-scale article, as appropriate (see figure 3).

5.4 Certification & force management development (Task IV).
Aircraft structural certification is based on the results of Tasks I through III. Certification analyses described in 5.4.1 culminate in structural certification of an aircraft. An ASIP must develop an appropriate force management strategy in preparation for force management that occurs during sustainment under Task V. This strategy depends upon formal documentation of structural capability, creation of maintenance plans, and the development of data acquisition/storage/evaluation systems.

5.4.1 Certification analyses.
The design analyses described in 5.2 shall be revised to account for differences revealed between analysis and test. Selected design development tests described in 5.2, the full-scale tests described in 5.3, and the interpretation and evaluation of test results described in 5.3.7 shall be used in the certification effort. The design analyses correlated to ground and flight testing establish structural certification and are herein referred to as "certification analyses." The certification analyses provide the engineering source data for the Technical Orders that document the operational limitations/restrictions, procedures, and maintenance requirements to ensure safe operation. Approval of the certification analyses shall constitute aircraft structural certification, a critical step in achievement of airworthiness certification for the aircraft in accordance with procedures outlined in MIL-HDBK-516.

5.4.1.1 Risk analysis.
When tailoring, as described in 1.1.2, has been accomplished, a risk analysis shall be performed and utilized in the initial airworthiness certification. The objective of this analysis is to determine the combined impact of all tailored ASIP tasks and/or elements on aircraft structure reliability and to verify that the allocated aircraft structure reliability requirement has been achieved.

5.4.1.2 Quantifying the accuracy of analyses.
The accuracy of the analyses described in 5.2 shall be probabilistically quantified by direct comparison to the test results described in 5.2.14 and 5.3 and documented to support aircraft structural certification.

5.4.2 Strength Summary & Operating Restrictions (SSOR).
A Strength Summary & Operating Restrictions (SSOR) document shall summarize the final analyses and other pertinent structural data into a format which shall provide rapid visibility of the important structural characteristics, limitations, and capabilities in terms of operational parameters. The SSOR shall be primarily in a diagrammatic form that shows the aircraft structural limitations and capabilities as a function of the important

operational parameters such as speed, acceleration, center-of-gravity location, and gross weight. The summary shall include brief descriptions of each major structural assembly, in diagrammatic form, which indicate structural arrangements, materials, critical design conditions, damage tolerance and durability critical areas, and minimum margins of safety. Appropriate references to design drawings, detail analyses, test reports, and other back-up documentation shall be provided.

5.4.3 Force Structural Maintenance Plan (FSMP).
The intent during the design of the aircraft is to achieve robust structures that will require little, if any, maintenance for corrosion, fatigue cracking, stress corrosion cracking, and/or delaminations within the design service life assuming that the aircraft is flown to the design loads/environment spectrum. However, full-scale testing described in Task III and the certification analyses performed as part of Task IV may identify critical areas missed during design that would require additional analysis and in-service inspections and perhaps production and/or in-service modifications. The FSMP shall define when, where, how, and the estimated costs of these inspections and modifications. It shall also describe the inspection-based (recurring) structural maintenance program (i.e., periodic, minor and major inspections, program depot maintenance (PDM), the CPCP, etc.). If the maintenance planning for all or any part of the aircraft structure will be condition-based (i.e. CBM+), then the operation/scheduling of the structural health monitoring system shall be described. Furthermore, when both inspection-based and and condition-based maintenance are used to manage the structure, the interaction between the two and the responsibilities of each shall be described. It is intended that the FSMP will be used to establish budgetary planning, force structure planning, and maintenance planning. The initial FSMP will generally be based on the design loads/environment spectrum and shall be updated when the data from the Loads/Environment Spectra Survey (L/ESS) (described in 5.4.4) becomes available and a new baseline operational spectrum is developed. Additional updates, as described in 5.5.6, will be required when any of the following occur: 1) there are significant changes in operational usage, 2) unanticipated damage is discovered by the structural health monitoring system during operational usage, 3) unanticipated damage is discovered during scheduled inspections, 4) unanticipated damage is discovered during surveillance sampling inspections conducted using the Analytical Condition Inspection (ACI) Program, 5) unanticipated damage is discovered during structural teardown inspection programs, and/or 6) unanticipated damage is discovered during normal operational maintenance of the aircraft. The structural maintenance database required to support these updates is described in 5.4.3.1.

5.4.3.1 Structural maintenance database development.
The structural maintenance database shall be developed to capture adequate, detailed information on the aging processes (fatigue, corrosion, delaminations, etc.) which occur in the aircraft and thus support the ongoing evaluation of structural integrity during sustainment. The database shall be developed to record all significant damage findings such as cracks, corrosion, and/or delaminations discovered during program depot maintenance, analytical condition inspections, time compliance technical order (TCTO) structural inspections, teardown inspections, and normal operational maintenance. The database shall also be able to record a description of the damage types, damage sizes,

damage locations, inspection techniques (including POD information), aircraft configuration, pertinent aircraft usage history including basing information, and corrosion preventive methods (e.g., wash cycles, coatings, etc.). The database shall also be able to record all significant repairs and/or modifications so as to maintain configuration control. These records shall include a description of the repair/modification and when it was incorporated. Additional considerations for data to be recorded shall facilitate the analysis update described in 5.5.6.

5.4.3.2 Inspections.

Implicit in damage-tolerant structural designs are inspection requirements intended to ensure damage never reaches the sizes that can cause catastrophic failures. Inspections are required initially and at the repeat intervals described in 5.4.3.2.1. Such inspections shall continue to the estimated time, with the appropriate scatter factor, of the onset of WFD. At the onset of WFD, inspections are no longer sufficient to protect safety. The risk analysis of 5.2.16 shall be used to initially establish the time to onset of WFD. Upon their availability, the risk analysis updates of 5.5.6.3 shall be used to update the estimated time to onset of WFD.

5.4.3.2.1 Inspection intervals.

The criteria for the initial and repeat inspection intervals shall be as follows:

a. The initial inspections for fail-safe design concepts shall be established based on either: 1) fatigue analyses and tests with an appropriate scatter factor, or 2) slow damage growth analyses and tests assuming an appropriate initial flaw size.

b. The initial inspection for slow damage growth design concepts shall occur at or before one-half the life from the assumed maximum probable initial flaw size to the critical flaw size.

c. The repeat inspection intervals for both design concepts shall occur at or before one-half the life from the minimum detectable flaw size (based on the probability of detection) to the critical flaw size.

d. The risk analysis of 5.2.16 and 5.5.6.3 should be used to determine if a reduction in the inspection intervals are required to control the safety risk to an acceptable level or to reduce economic or availability consequences associated with damage repair.

5.4.3.2.2 Inspection methods.

Results from structural analysis shall be used to identify the inspection methods required to detect anticipated damage. Selection of the inspection methods shall consider material, geometry, accessibility, human factors, and the resulting assumed detectable flaw size. Alternate inspection capability estimates may be used if demonstrated using the guidance of MIL-HDBK-1823 and as approved by the Nondestructive Inspection Requirements Review Board.

5.4.3.3 Structural Health Monitoring.

The aircraft structural health monitoring system is intended to ensure that damage in monitored areas is discovered early enough to allow scheduling of maintenance actions and that it never reaches the sizes that can cause catastrophic failures. Development of the SHM system shall consider material, geometry, accessibility, sensor POD and the

resulting system level POD. The initial development of, as well as any subsequent modification to the SHM system will require approval by the SHMSRRB.
There are two modes for SHM system operation: 1) continuous, in-flight monitoring, 2) on-ground operation at regular intervals or after specified flight events. It is anticipated that the vast majority of SHM systems will be operated on-ground, which means that the highest possible frequency of operation would be once per flight. Less frequent intervals of operation shall be determined as follows:

5.4.3.3.1 SHM System Operation Intervals.
The criteria for establishing the frequency of SHM system operation shall be as follows:
a. The SHM system operation interval for fail-safe design concepts shall be less than or equal to one-half the life as determined by either: 1) fatigue analyses and tests with an appropriate scatter factor, or 2) slow damage growth analyses and tests assuming an appropriate initial flaw size. The required initial flaw size shall be based on the demonstrated flaw size detection capability of the SHM system. (The data obtained from the design development testing described in 5.2.16 shall be used for verification of the flaw size detection capability.)
b. The SHM system operation interval for slow damage growth design concepts shall be less than or equal to one-half the life from the assumed maximum probable initial flaw size to the critical flaw size. The required initial flaw size shall be based on the demonstrated flaw size detection capability of the SHM system. (The data obtained from the design development testing described in 5.2.16 shall be used for verification of the flaw size detection capability.)
c. The risk analysis of 5.2.16 and 5.5.6.3 should be used to determine if a reduction in the operation intervals are required to control the safety risk to an acceptable level or to reduce economic or availability consequences associated with damage repair.

5.4.3.3.2 SHM system operation for anomalous events.
If the frequency of SHM system operation is less than once per flight, then a procedure shall be established whereby an anomalous usage event will trigger an SHM interrogation after the current flight. A list of anomalous usage events (overload, hard landing, ballistic damage, etc.) shall be developed and continuously updated.

5.4.3.4 Surveillance.
A surveillance program shall be developed to improve estimates of the in-service times when damage requires maintenance actions (inspections, repairs, modifications or retirements, etc.). Two essential components of a surveillance program are an ACI Program and a Structural Teardown Program.

5.4.3.4.1 Analytical Condition Inspection (ACI) Program.
An ACI Program shall be conducted throughout the life of the aircraft per Air Force Materiel Command Instruction (AFMCI) 21-102. Corrosion, fatigue, and other damage scenarios shall be considered in the selection of inspection locations and schedules and/or SHM sensor locations and system operation schedules for the aircraft structure. Aircraft scheduled for the ACI Program based on Individual Aircraft Tracking (IAT) estimates of structural damage shall be referred to as Lead the Fleet (LTF) aircraft. The ACI Program

shall be conducted with special emphasis on determination of when and where corrosion occurs and on prototypes of NDI and repair actions.

5.4.3.4.2 Structural Teardown Program.
A Structural Teardown Program may be required if an aircraft is expected to operate beyond its design service life or if there is evidence of extensive damage that may jeopardize the aircraft's structural integrity. The need for and timing of a Structural Teardown Program shall be based on force management updates described in 5.5.6.

5.4.3.5 Repair criteria.
Allowable damage limits and damage growth rates shall be established to develop repair concepts for structural components and assemblies. Structural analyses shall be used to establish repair designs and to identify post-repair inspection requirements.

5.4.4 Loads/Environment Spectra Survey (L/ESS) development.
A system to perform a loads/environment spectra survey (L/ESS) shall be developed to obtain actual usage data that can be used to update or confirm the original design spectrum. A sufficient number of aircraft shall be instrumented to achieve a 20-percent valid data capture rate of the fleet usage data. L/ESS systems shall record time-history data such as vertical and lateral load factors; roll, pitch and yaw rates; roll, pitch, and yaw accelerations; altitude; Mach number; control surface positions; selected strain measurements; ground loads; aerodynamic excitations; etc. Data shall also be collected to characterize the thermal and chemical environments within the aircraft and associated with aircraft basing. If the IAT Program as described in 5.4.5 obtains sufficient data to develop the baseline operational loads/environment spectrum and to detect significant changes in usage and/or environment, a separate L/ESS system as described herein is not required. If instrumentation and/or sensors are part of the L/ESS Program, the instrumentation shall be incorporated into the full-scale static test described in 5.3.1, into the full-scale durability test described in 5.3.4, and into the flight and ground loads survey aircraft described in 5.3.3.1. Data systems should comply with the requirements of AFPD 63-14 and AFI 63-1401.

5.4.5 Individual Aircraft Tracking (IAT) Program development.
A program to perform individual aircraft tracking shall be developed to obtain actual usage data that can be used to adjust maintenance intervals on an individual aircraft ("by tail number") basis. All force aircraft shall have systems that record sufficient usage parameters that can be used to determine the damage growth rates throughout the aircraft structure. The total IAT control point set will include both the inspection-based maintenance (IBM) control point sets and the condition-based maintenance (CBM). The systems shall have sufficient capacity and reliability to achieve a 90-percent minimum valid data capture rate of all flight data throughout the service life of the aircraft. The systems shall include serialization of interchangeable/replaceable aircraft structural components, as required. The IAT Program shall be ready to acquire data at the beginning of initial flight operations. If instrumentation and/or sensors are part of the IAT Program, the instrumentation shall be incorporated into the full-scale static test described in 5.3.1, into the full-scale durability test described in 5.3.4, and into the flight and

ground loads survey aircraft described in 5.3.3.1. Data systems should comply with the requirements of AFPD 63-14 and AFI 63-1401.

5.4.5.1 Tracking analysis methods.

Analysis methods shall be developed which adjust the inspection and modification times based on the measured structural condition (damage state) for CBM control points, and on the actual measured usage of the individual aircraft for all (CBM and IBM) control points. These methods shall have the ability to predict damage growth in all critical locations and in the appropriate environment as a function of the total measured usage, and to recognize changes in operational mission usage. The methods shall also provide the ability to determine the equivalent flight hours. The analysis methods and accompanying computer programs shall be provided to the USAF.

5.4.6 Rotorcraft Dynamic Component Tracking (RDCT) Program development.

A program to perform rotorcraft dynamic component tracking (RDCT) shall be developed to provide data to support condition-based maintenance. One hundred percent of the rotorcraft shall be instrumented with systems that measure component responses to operations and that anticipate impending failures. The systems shall provide sufficient warning of impending failure to allow safe flight and landing. The RDCT Program shall be ready to acquire data at the beginning of initial flight operations.

5.5 Force management execution (Task V).

Task V describes the execution of the force management strategy described in Task IV. Task V will be primarily the responsibility of the USAF. Force management shall be conducted by executing the FSMP. The maintenance schedule directed by the FSMP shall be adjusted for each aircraft by data received from the IAT Program described in 5.5.1 or by the RDCT system described in 5.5.2. The FSMP shall be updated periodically to ensure it accurately and efficiently protects against structural failures. Updates to the FSMP shall be based on evaluations of changes in operational usage described in 5.5.3 as well as assessments of new damage findings documented within the structural maintenance database described in 5.5.5.1. Periodic action shall be taken to ensure the reliability of the on-board usage data-gathering equipment is sufficient to achieve the required data capture rates. Any changes to the force management strategy shall be documented in the ASIP Master Plan described in 5.1.1.

5.5.1 Individual Aircraft Tracking (IAT) Program.

The IAT Program shall be used to adjust the inspection, modification, overhaul, and replacement times based on the measured condition (damage state) for CBM control points and on the actual, measured usage of the individual aircraft for all (CBM and IBM) control points. The IAT Program shall be used to determine damage growth in the appropriate environment as a function of the total measured usage and to quantify changes in operational mission usage. The IAT Program shall also determine the equivalent flight hours (or other appropriate measures of damage such as landings, pressure cycles, etc.) and adjust the required maintenance schedule for all critical locations on each individual aircraft. The IAT Program shall forecast when aircraft structural component life limits will be reached. Data systems should comply with the requirements of AFPD 63-14 and AFI 63-1401.

5.5.2 Rotorcraft Dynamic Component Tracking (RDCT) Program.
The RDCT Program shall measure rotorcraft dynamic component responses to operations and provide data to support condition-based maintenance and to anticipate impending failures. The systems shall provide sufficient warning of impending failure to allow safe flight and landing.

5.5.3 Loads/Environment Spectra Survey (L/ESS).
The loads/environment spectra survey shall be conducted to obtain actual usage data that can be used to update the original design spectrum. A new baseline operational loads spectrum shall be developed from the in-flight measurements and the predicted operational environment updated as necessary. Significant changes to the baseline operational loads spectrum shall be used to update the analyses described in 5.5.5. Data systems should comply with the requirements of AFPD 63-14 and AFI 63-1401.

5.5.3.1 Initial Loads/Environment Spectra Survey.
The initial survey period shall last for at least 3 years after Initial Operating Capability (IOC). The length of the initial survey period shall be based on evaluations of the mission types, mission mix, and quantity of aircraft in service.

5.5.3.2 Loads/Environment Spectra Survey updates.
The stability of mission types, mixes, and severity shall be evaluated to determine the need for periodic survey updates. The ASIP Manager shall review the need for L/ESS updates annually.

5.5.4 ASIP Manual.
An Aircraft Structural Integrity Program Technical Order (T.O. X-YY-38) is required for each aircraft per MIL-DTL-87929; Appendix G of the specification provides format and content for an Aircraft Structural Integrity Program Technical Manual. The intent of a T.O. X-YY-38 is to provide the flight-line maintainer basic information on the ASIP and to define actions (if any) that maintainers must perform to support the ASIP. Periodic updates of this technical order are required as the ASIP changes.

5.5.5 Aircraft structural records.
Records which pertain to aircraft structures shall be retained to provide a historical basis for evaluation of continued airworthiness.

5.5.5.1 Structural maintenance records.
The structural maintenance database shall be used to capture detailed information on the aging processes occurring in the aircraft and support the ongoing evaluation of structural integrity during sustainment. The database shall record all significant damage findings such as cracks, corrosion, and/or delaminations discovered during PDM, analytical condition inspections, TCTO structural inspections, teardown inspections, normal operational maintenance, and by structural health monitoring. The database shall also

record a description of the damage types, damage sizes, damage locations, inspection techniques (including POD information), aircraft configuration, pertinent aircraft usage history including basing information, and corrosion preventative methods (e.g., wash cycles, coatings, etc.). The database shall also record all significant repairs and/or modifications so as to maintain configuration control. These records shall include a description of the repair/modification and when it was incorporated.

5.5.5.2 Weight and balance records.
Weight and balance records shall be maintained to ensure the aircraft remains within its approved limitations. Guidance may be found in SAWE RP No. 7 and T.O. 1-1B-50.

5.5.6 Force management updates.
Mission and usage changes, major modifications, as well as aircraft inspection and/or structural health monitoring findings shall be evaluated by analysis and/or testing (to include a possible additional full-scale static and/or durability test) to determine the need for and timing of periodic updates to the force management strategy. It is envisioned that updates will be required every 5 years or as dictated by the requirements defined in the subparagraphs below. Information which results from the updates described below shall be documented in the FSMP.

5.5.6.1 Durability and Damage Tolerance Analysis (DADTA) and IAT Program updates.
The IAT data described in 5.5.1, L/ESS data described in 5.5.3, and the aircraft structural records described in 5.5.5 shall be used to determine when Durability and Damage Tolerance Analysis (DADTA) and IAT Program updates should be conducted. Variations in the average usage from the analysis baseline and usage variation extremes from the analysis baseline shall be considered when the need for an update is determined. In addition, an update to the DADTA and IAT Program shall be conducted when aircraft damage findings (both from inspections and from structural health monitoring) indicate the accuracy of the analyses are less than expected per the results of 5.4.1.2.

5.5.6.2 Corrosion assessment updates.
Information obtained from the ACI Program and new corrosion findings documented in the structural maintenance database shall be reviewed annually. The occurrences of corrosion shall be evaluated with regard to the effectiveness of the preventive procedures (e.g., frequency of wash cycles, coatings, corrosion prevention compounds, etc.) used and, if possible, corrosion findings shall be correlated to the aircraft basing and the results of Task II and Task III environmental testing. The results of these evaluations and any observed trends will be used to develop improved maintenance procedures and adjust the corrosion inspection requirements in the FSMP. The corrosion assessment updates may also be used to define candidate structural applications for the introduction or expansion of a structural health monitoring system. In the event that a structural health monitoring system is introduced, it shall be qualified in accordance with 5.1.7, 5.2.16 and 5.3.4.

5.5.6.3 Structural Health Monitoring (SHM) updates.
Information obtained from the ACI Program and new inspection and/or structural health monitoring findings documented in the structural maintenance database shall be reviewed annually. The occurrences of damage and the effectiveness of structural health monitoring to manage them shall be evaluated. The results of these evaluations and any observed trends will be used to define candidate structural applications for the introduction or expansion of a structural health monitoring system (for both hot spot and wide area applications), and to develop improved maintenance procedures and adjust the inspection requirements in the FSMP. In the event that a structural health monitoring system is introduced, it shall be qualified in accordance with 5.1.7, 5.2.16 and 5.3.4.

5.5.6.4 Risk analysis updates.
The risk analyses described in 5.2.17 and 5.4.1.1 shall be updated and the results shall be reported for formal acceptance using MIL-STD-882 direction. The EIFS distribution developed under 5.3.4 shall be updated to include aircraft inspection results (e.g., sizes of cracks found and number of locations inspected) which account for the IAT data described in 5.5.1 to determine the probability of failure of the aircraft structure. Validation of the EIFS distribution by teardown inspection of aircraft and/or components with high levels of predicted damage shall be considered. The primary reasons to update the risk analyses are to:
a. evaluate detected and anticipated aircraft structural damage. The results shall be used in conjunction with IAT data described in 5.5.1 to establish the individual aircraft maintenance times.
b. Evaluate economic and/or availability impacts associated with maintenance options such as inspection and repair/replacement as needed versus modification.
c. Determine the structural integrity risk associated with operating the aircraft beyond the design service life.
These updates shall be used to compare the predicted probability of catastrophic failure of the aircraft structure to the following limits. A probability of catastrophic failure at or below 10-7 per flight for the aircraft structure is considered adequate to ensure safety for long-term military operations. Probabilities of catastrophic failure exceeding 10-5 per flight for the aircraft structure should be considered unacceptable. When the probability of failure is between these two limits, consideration should be given to mitigation of risk through inspection, repair, operational restrictions, modification, or replacement.

5.5.7 Recertification.
Recertification of the aircraft structure shall be performed if significant deviations from the certification baseline occur. Such deviations may include changes to usage, damage, configuration, and/or service life expectancy. The recertification analyses shall provide the engineering source data for revision of Technical Orders which document the operational limitations/restrictions, procedures, and maintenance requirements to ensure continuing safe operation. Recertification efforts should consider all ASIP tasks and elements and may require an additional full-scale static and/or durability testing, flight testing, etc.

6. NOTES

(This section contains information of a general or explanatory nature that may be helpful, but is not mandatory.)

6.1 Intended use.

This standard is intended as a foundation to establish and conduct an ASIP for all USAF aircraft developed to perform combat and support missions in environments unique to military weapons systems, and may be used by other agencies at their discretion. Contractual documents may contain tailored requirements for each program, based on the content herein.

6.2 Acquisition requirements.

Acquisition documents should specify the following: a. Title, number, and date of the standard.

6.3 Data requirements.

The long-term operation and maintenance of USAF aircraft and equipment is directly dependent on the availability of certain structural data developed during an ASIP. These data are used to establish, assess, and support inspections; maintenance activities; repairs; modification tasks; and replacement actions for the life of the aircraft structure. Contractual provisions must ensure these data are available to the USAF and to relevant contractors and subcontractors throughout the operational life of the system. The following list is provided as a general guide to the necessary data. This list may be tailored based on system operational requirements, the support concept/strategy, AFPD 63-14, AFI 63-1401, the requirements contained in this standard, and guidance in JSSG-2006.

a. ASIP Master Plan and integration with the IMP and IMS (See 5.1.1.)
b. Design service life and design usage (See 5.1.2.)
c. Structural design criteria (See 5.1.3.)
d. Durability and Damage Tolerance Control Program (See 5.1.4.)
e. Corrosion Prevention and Control Program (See 5.1.5.)
f. Nondestructive Inspection Program (See 5.1.6.)
g. Structural Health Monitoring Program (See 5.1.7.)
h. Selection of materials, processes, joining methods, and structural concepts (See 5.1.8.)
i. Material and joint allowables (See 5.2.1.)
j. Loads analysis (See 5.2.2.)
k. Design service loads spectra (See 5.2.3.)
l. Design chemical/thermal environment spectra (See 5.2.4.)
m. Stress analysis (See 5.2.5.)
n. Damage tolerance analysis (See 5.2.6.)
o. Durability analysis (See 5.2.7.)
p. Corrosion assessment (See 5.2.8.)
q. Sonic fatigue analysis (See 5.2.9.)
r. Vibration analysis (See 5.2.10.)
s. Aeroelastic and aeroservoelastic analysis (See 5.2.11.)
t. Mass properties analysis (See 5.2.12.)

u. Survivability analysis (See 5.2.13.)
v. Design development tests (See 5.2.14.)
w. Production NDI capability assessment (See 5.2.15.)
x. Production SHM capability assessment (See 5.2.16.)
y. Initial risk analysis (See 5.2.17.)
z. Static tests (See 5.3.1.)
aa. First flight verification ground tests (See 5.3.2.)
bb. Flight tests (See 5.3.3.)
cc. Durability tests (See 5.3.4.)
dd. Damage tolerance tests (See 5.3.5.)
ee. Climatic tests (See 5.3.6.)
ff. Interpretation and evaluation of test results (See 5.3.7.)
gg. Certification analyses (See 5.4.1.)
hh. Strength Summary & Operating Restrictions (SSOR) (See 5.4.2.)
ii. Force Structural Maintenance Plan (FSMP) (See 5.4.3 and 5.5.6.)
jj. Loads/Environment Spectra Survey (L/ESS) (See 5.4.4 and 5.5.3.)
kk. Individual Aircraft Tracking (IAT) Program (See 5.4.5 and 5.5.1.)
ll. Rotorcraft Dynamic Component Tracking (RDCT) Program (See 5.4.6 and 5.5.2.)
mm. Aircraft structural records (See 5.5.5.)
nn. ASIP Manual (See 5.5.4.)
oo. Probability of failure limits (See 5.5.6.3.)
pp. Recertification (See 5.5.7.)

6.4 Subject term (key word) listing.

aeroacoustics
aircraft structure
condition-based maintenance (CBM)
corrosion
cracking
damage tolerance
durability
economic life
fatigue
flight test
flutter
force management
ground test
loads
mass
nondestructive inspection (NDI)
probability of failure
proof test
risk
service life
static test
strength

structural health monitoring (SHM)
sustainment
vibration
weight and balance
widespread fatigue damage

6.5 Changes from previous issue.
Marginal notations are not used in this revision to identify changes with respect to the previous issue due to the extent of the changes.

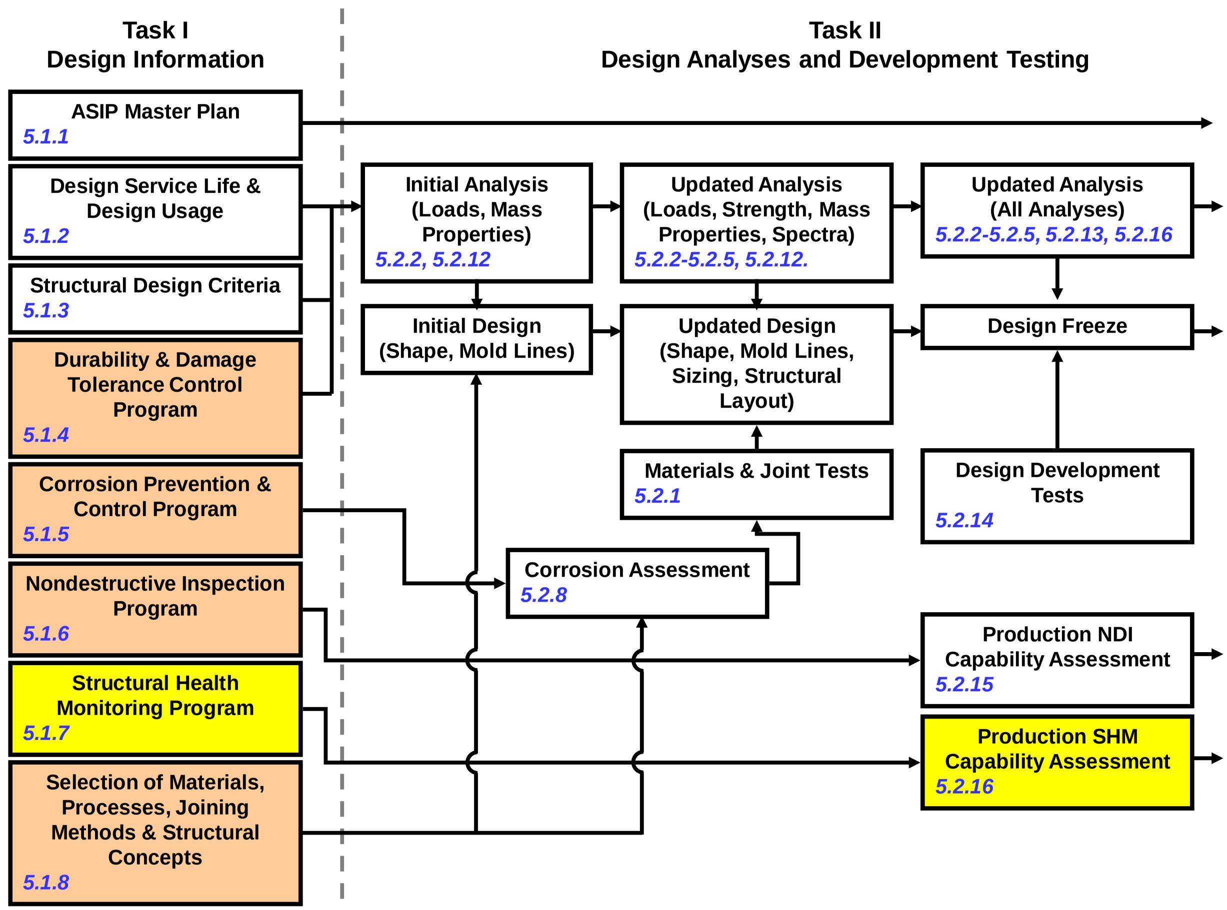

FIGURE 2. Aircraft Structural Integrity Program – Tasks I and II.
Beige boxes identify tasks that are modified; yellow boxes are new tasks.

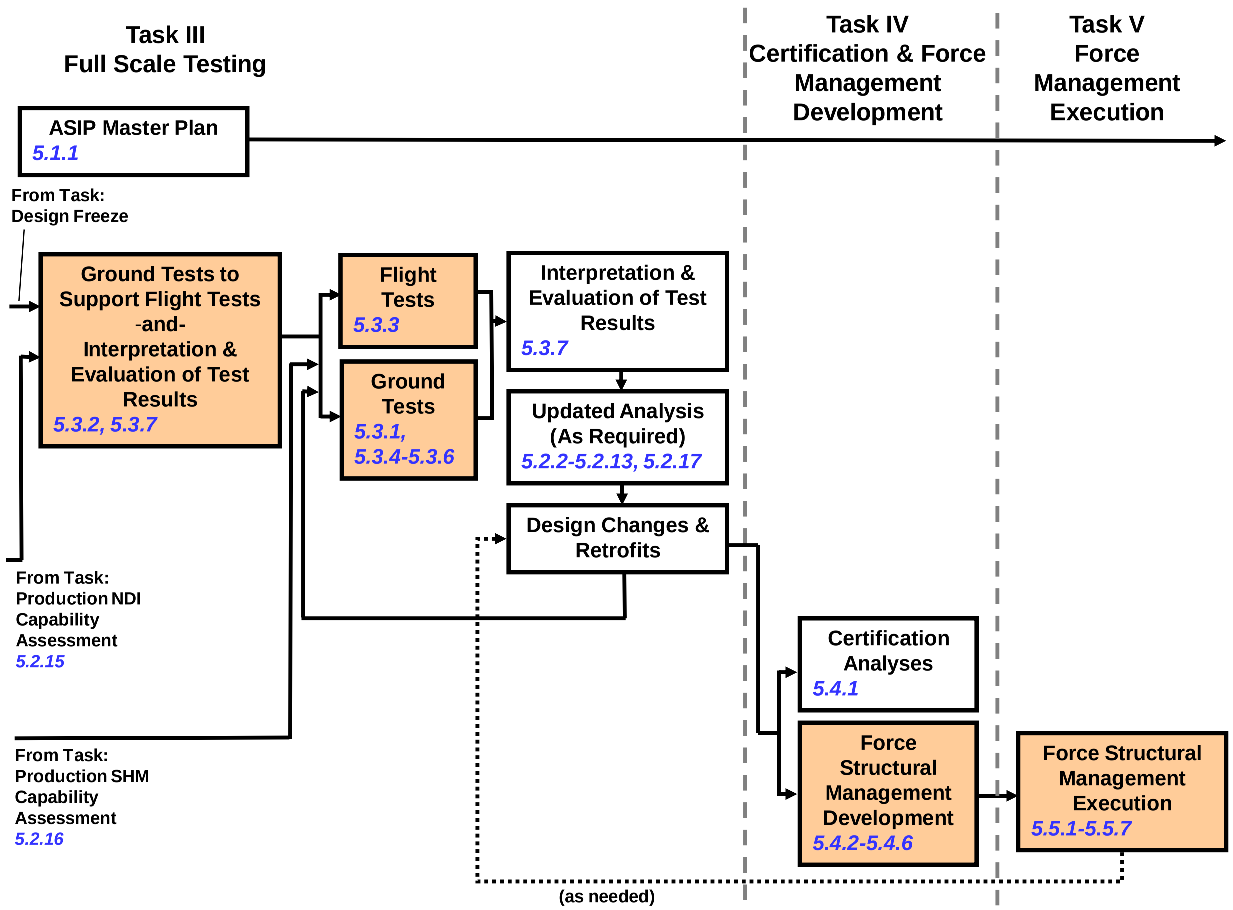

FIGURE 3. Aircraft Structural Integrity Program – Tasks III through V.
Beige boxes identify tasks that are modified; yellow boxes are new tasks.

Custodian:
Air Force – 11

Preparing activity:
Air Force – 11
(Project 15GP-2005-009)

NOTE: The activities listed above were interested in this document as of the date of this document. Since organizations and responsibilities can change, you should verify the currency of the information above using the ASSIST Online database at http://assist.daps.dla.mil

ACRONYMS

ASIP	Aircraft Structural Integrity Program
CBM	Condition Based Maintenance
DADT	Durability and Damage Tolerance
FSMP	Force Structural Maintenance Plan
IFSD	Initial Flaw Size Distribution
IVHM	Integrated Vehicle Health Monitoring
MAPOD	Model Assisted Probability of Detection
MOI	Magneto-Optical Imaging
POI	Probability of Inspection
POD	Probability of Detection
SCC	Stress Corrosion Cracking
SFPOF	Single Flight Probability of Failure
SHM	Structural Health Monitoring
SIF	Stress Intensity Factor
WFD	Widespread Fatigue Damage