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Application and Certification of Comparative Vacuum Monitoring Sensors for Structural Health Monitoring of 737 Wing Box Fittings

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ABSTRACT

Multi-site fatigue damage, hidden cracks in hard-to-reach locations, disbonded joints, erosion, impact, and corrosion are among the major flaws encountered in today's extensive fleet of aging aircraft and space vehicles. The use of in-situ sensors for real-time health monitoring of aircraft structures are a viable option to overcome inspection impediments stemming from accessibility limitations, complex geometries, and the location and depth of hidden damage. Reliable, structural health monitoring systems can automatically process data, assess structural condition, and signal the need for human intervention. Prevention of unexpected flaw growth and structural failure can be improved if on-board health monitoring systems are used to continuously assess structural integrity. Such systems are able to detect incipient damage before catastrophic failures occurs. Condition-based maintenance practices could be substituted for the current time-based maintenance approach. Other advantages of on-board distributed sensor systems are that they can eliminate costly, and potentially damaging, disassembly, improve sensitivity by producing optimum placement of sensors and decrease maintenance costs by eliminating more time-consuming manual inspections.

This report presents a Sandia Labs-aviation industry effort to move SHM into routine use for aircraft maintenance. This program addressed formal SHM technology validation and certification issues so that the full spectrum of concerns, including design, deployment, performance and certification were appropriately considered. The Airworthiness Assurance NDI Validation Center (AANC) at Sandia Labs, in conjunction with Boeing, Delta Air Lines, Structural Monitoring Systems Ltd., Anodyne Electronics Manufacturing Corp. and the Federal Aviation Administration (FAA) carried out a certification program to formally introduce Comparative Vacuum Monitoring (CVM) as a structural health monitoring solution to a specific aircraft wing box application. Validation tasks were designed to address the SHM equipment, the health monitoring task, the resolution required, the sensor interrogation procedures, the conditions under which the monitoring will occur, the potential inspector population, adoption of

CVM into an airline maintenance program and the document revisions necessary to allow for routine use of CVM as an alternate means of performing periodic structural inspects.

To carry out the validation process, knowledge of aircraft maintenance practices was coupled with an unbiased, independent evaluation. Sandia Labs designed, implemented, and analyzed the results from a focused and statistically-relevant experimental effort to quantify the reliability of the CVM system applied to the Boeing 737 Wing Box fitting application. All factors that affect SHM sensitivity were included in this program: flaw size, shape, orientation and location relative to the sensors, as well as operational and environmental variables. Statistical methods were applied to performance data to derive Probability of Detection (POD) values for CVM sensors in a manner that agrees with current nondestructive inspection (NDI) validation requirements and also is acceptable to both the aviation industry and regulatory bodies. This report presents the use of several different statistical methods, some of them adapted from NDI performance assessments and some proposed to address the unique nature of damage detection via SHM systems, and discusses how they can converge to produce a confident quantification of SHM performance.

An important element in developing SHM validation processes is a clear understanding of the regulatory measures needed to adopt SHM solutions along with the knowledge of the structural and maintenance characteristics that may impact the operational performance of an SHM system. This report describes the major elements of an SHM validation approach and differentiates the SHM elements from those found in NDI validation. The activities conducted in this program demonstrated the feasibility of routine SHM usage in general and CVM in particular for the application selected. They also helped establish an optimum OEM-airline-regulator process and determined how to safely adopt SHM solutions. This formal SHM validation will allow aircraft manufacturers and airlines to confidently make informed decisions about the proper utilization of CVM technology. It will also streamline the regulatory actions and formal certification measures needed to assure the safe application of SHM solutions.

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Acknowledgments

The operation and maintenance of commercial aircraft requires the frequent interaction of multiple agencies. The unique nature of this program accentuated the need for these ties and the program's success hinged on close cooperation among all participants. This Structural Health Monitoring initiative was sponsored by the FAA William J. Hughes Technical Center under the direction of the technical manager, Paul Swindell. This program was formulated by the Sandia National Laboratories Airworthiness Assurance Center (AANC) – operated by Sandia Labs for the FAA - in concert with the industry team including the FAA, Boeing, Delta Air Lines, Structural Monitoring Systems Ltd. (SMS), and Anodyne Electronics Manufacturing Corporation (AEM).

This report makes it clear to the reader that this program was a team effort that included multiple people from this team to address the full range of engineering issues including: validation planning, procedures, Job Card definition, guidance documents, sensor design and layout, application-specific installation drawings, inspection, Damage Tolerance Analysis, data analysis, data package preparation, formal document revisions, and final approval via Airplane Representatives (AR) at Boeing. The contributions of this team are gratefully acknowledged with key engineering work produced by the following core team members: David Piotrowski, John Bohler and Alex Melton of Delta Air Lines, John Linn of Boeing, Trevor Lynch-Staunton of AEM, Toby Chandler and Mike Reveley of SMS and Paul Swindell of the FAA.

The authors would like to also thank the Delta Air Lines mechanics and inspectors who conducted the CVM installations and flight test monitoring: Chris Coleman, John Hayes, and Joe Reeves. General SHM support within Delta's maintenance program was provided by the following Delta Air Lines management and engineering personnel: Jeff Peiter, David Fauerstenau, Paul Vaughan, Tristan Perry, David Fosse, John Brutag, and Chris Iaconis.

Special recognition goes to the FAA participants who provided guidance and direction on all aspects of this program. Program sponsors and contributors to program planning included Mark Freisthler, Ian Won, Greg Schneider, Walt Sippel and Michael Gorelik of the FAA Transport Aircraft Directorate. Additional guidance and oversight from other FAA offices came from: Patrick Safarian (FAA Air Transport Offices), Tammy Anderson (FAA Seattle Aircraft Certification Office), Nancy Marsh (FAA Seattle Aircraft Certification Office), Roy Ventura (FAA Seattle Aircraft Certification Office), Bill Heliker (FAA Seattle Aircraft Evaluation Group; currently Aircraft Maintenance Div), Linda Haynes (FAA Atlanta Aircraft Certification Office) and Keith Frable (FAA Principal Maintenance Inspector for Delta).

We would like to thank additional Boeing personnel who provided extensive guidance on this effort. Boeing ARs, covering expertise in structures, systems and interiors, provided critical information and served as reviewers and advisors for the activities described in this report.

These include: Matt Miller (Chief Structures Engineer), Mike Wineland, Dale Lemke, Jerry Kassebaum, and Mike Tovey.

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Finally, special thanks goes to Stephen Neidigk (Sandia Labs) for his assistance in program formulation and his support of the flight test activities.

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CHAPTER 1

1.0 Background on Deploying Structural Health Monitoring Solutions

The aerospace industry is striving to reduce the unit acquisition and operating costs to their customers while maintaining required safety levels. To obtain this goal, manufacturers are introducing new material and production technologies. The manufacturers are also promoting new technologies such as Structural Health Monitoring (SHM) to reduce long-term maintenance costs and increase aircraft availability [1.1 – 1.5]. Though well-established design and maintenance procedures exist to detect the effect of structural fatigue, new and unexpected phenomena must be addressed by the application of advanced flaw detection methods. Similarly, innovative deployment methods must be developed to overcome a myriad of inspection impediments stemming from accessibility limitations, complex geometries, and the location and depth of hidden flaws.

Health monitoring of structures is a major concern of the engineering community. This need is even more intense in the case of aging aerospace and civil structures many of which are operating well beyond their initial design lives. The current damage tolerance design philosophy requires that a structure be capable of sustaining small damage without failure, and that an inspection program be instituted to detect such flaws before they grow to a critical size. This damage tolerance approach recognizes the impossibility of establishing complete structural redundancy – the fail-safe design premise – and places greater emphasis on inspection to ensure safety and reliability.

Multi-site fatigue damage and hidden cracks in hard-to-reach locations are among the major flaws encountered in today's extensive fleet of aging aircraft, bridges, buildings, and civil and space transport vehicles. The costs associated with the increasing maintenance and surveillance needs of aging structures are rising at an unexpected rate. Aircraft maintenance and repairs represent about a quarter of a commercial fleet's operating costs. The application of Structural Health Monitoring (SHM) systems using distributed sensor networks can reduce these costs by facilitating rapid and global assessments of structural integrity. These systems also allow for condition-based maintenance practices to be substituted for the current time-based or cycle-based maintenance approach thus optimizing maintenance labor. Other advantages of on-board distributed sensor systems are that they can eliminate costly, and potentially damaging, disassembly, improve sensitivity by producing optimum placement of sensors with minimized human factors concerns in deployment and decrease maintenance costs by eliminating more time-consuming manual inspections. Through the use of in-situ sensors, it is possible to quickly, routinely, and remotely monitor the integrity of a structure in service. This requires the use of reliable structural health monitoring systems that can automatically process data, assess structural condition, and signal the need for specific maintenance actions.

Current aircraft maintenance operations require personnel entry into normally-inaccessible or hazardous areas to perform mandated, nondestructive inspections. To gain access for these inspections, structure must be removed, sealant must be removed and restored, fuel cells must be vented to a safe condition, or other disassembly processes must be completed. These processes are not only time consuming but they provide the opportunity to induce damage to the structure. The use of in-situ sensors, coupled with remote interrogation, can be employed to overcome a myriad of inspection impediments stemming from accessibility limitations, complex geometries, and the location and depth of hidden damage. Furthermore, prevention of unexpected flaw growth and structural failure can be improved if on-board health monitoring systems exist that could regularly assess structural integrity. Such systems would be able to detect incipient damage before catastrophic failures occur. The ease of monitoring an entire on-board network of distributed sensors means that structural health assessments can occur more often, allowing operators to be even more vigilant with respect to flaw onset. When accessibility issues are considered, distributed sensor systems may also represent significant time savings by eliminating the need for component tear-down.

While ad-hoc efforts to introduce SHM into routine aircraft maintenance practices are valuable in leading the way for more widespread SHM use, there is a significant need for an overarching plan that will guide near-term and long-term activities and will uniformly and comprehensively support the evolution and adoption of SHM practices. The Federal Aviation Administration is addressing these issues through a series of SHM validation programs. Overall, an SHM evaluation and deployment plan must contain input from aircraft manufacturers, regulators, operators, and research organizations so that the full spectrum of issues, ranging from design to deployment, performance and certification is appropriately considered. The SHM validation and utilization program described in this data package has produced guidelines for SHM system designers or procedures for assessing the performance of SHM systems. This program, involving an OEM, airline, national lab, SHM provider and the FAA is providing information and guidance that will support the adoption of SHM practices and allow the aviation industry to make informed decisions about the proper utilization of SHM. It will also be used to assess what regulatory guidance is needed to assure the safe incorporation of SHM through formal certification programs.

1.1 SHM Definition and Benefits Derived from its Use

SHM, which is often closely associated with nondestructive inspection (NDI) but which extends beyond normal NDI activities, has been defined in a wide variety of ways. Several definitions of SHM are provided below along with a definition of NDI to provide a basis of comparison and contrast.

<u>Nondestructive Inspection</u> (NDI) – examination of a material to determine geometry, damage, or composition by using technology that does not affect the future usefulness of the structure. Normal attributes of NDI deployment are:

- High degree of human interaction
- Local, focused inspections
- Requires access to area of interest (applied at select intervals)

Structural Health Monitoring (SHM) is the use of in-situ, mounted or embedded sensors and associated data analysis to aid in the assessment of structural or mechanical condition or system operation including the direct detection of structural flaws. The parameters to be monitored could indicate flaws directly or they could be physical properties such as load, strain, pressure, vibration, or temperature from which damage, malfunction, mechanical problems, or the need for additional investigation can be inferred.

The replacement of our present-day manual inspections with automatic health monitoring could substantially reduce the associated life-cycle costs. Motivated by these pressing needs, considerable research efforts are currently being directed towards development of health monitoring sensors and systems. Whether the sensor network is hardwired to an accessible location within the aircraft or monitored in a remote, wireless fashion, the sensors can be interrogated in a real-time mode. However, it is anticipated that the sensors will most likely be examined at discrete intervals; probably at normal maintenance checks. Figure 1-1 depicts a notional view of a sensor network deployed on an aircraft to monitor critical sites over the entire structure. Examples of some common flaws found in aircraft structure that could be monitored using SHM systems are shown in Figure 1-2.

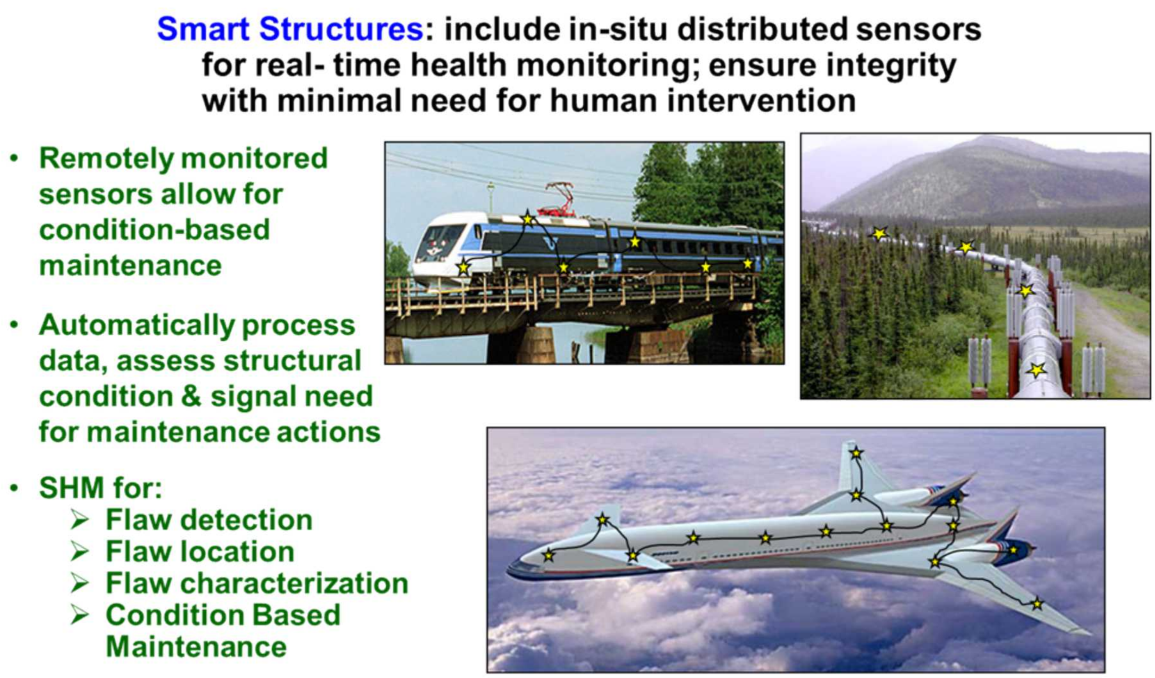

Figure 1-1: Depiction of Distributed Network of Sensors to Monitor Structural Health

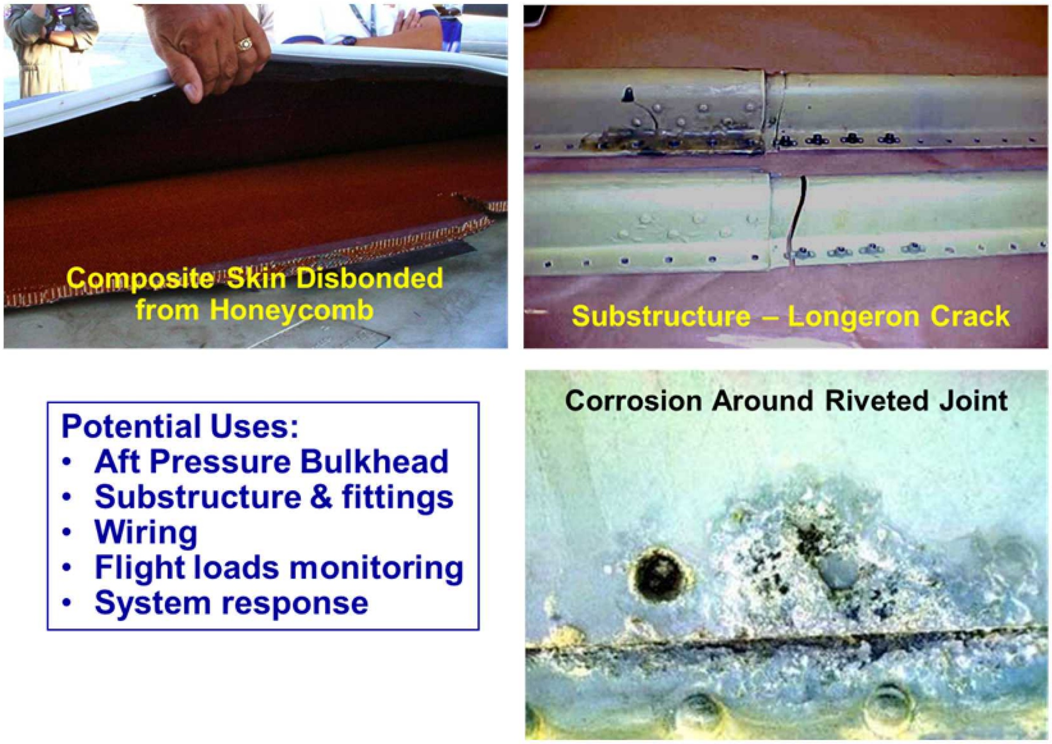

Figure 1-2: Sample Disbond, Crack and Corrosion Damage in Aircraft Structure that Could Be Monitored Using SHM Systems

A more detailed description of SHM includes:

Structural Health Monitoring (SHM) – sometimes referred to as "Smart Structures" or "Smart Systems;" involves the use of nondestructive inspection principles coupled with in-situ sensing to allow for rapid, remote, and real-time condition assessments. The sensors may record certain signatures wherein deviations from such signatures may indicate a mechanical issue which needs to be addressed. Alternately, the sensors may deterministically detect a flaw thus indicating the type of damage and location for further assessment. Such a system may be used to conduct health assessments for areas of the aircraft that have traditionally been difficult to access. SHM systems my either be used to supplement normally scheduled inspections or provide continued monitoring of a given structure.

A more succinct definition of SHM produced by the SAE Aerospace Industry Steering Committee on SHM (AISC-SHM) is:

Structural Health Monitoring (SHM) – The process of acquiring and analyzing data from on-board sensors to determine the health of a structure

There are a number of potential benefits that SHM offers regarding airplane maintenance and operation [1.1-1.11]:

Near-Term SHM Benefits

- Increased vigilance with respect to flaw onset
- Elimination of costly & potentially damaging structural disassembly
- Reduced operation and maintenance costs
- Increased availability of the aircraft fleet, by reduction of down-time after unforeseen events
- Ensure safety by identifying problems (aircraft operations, diminished structural integrity) that could threaten airworthiness
- Overcome accessibility limitations, complex geometries, depth of hidden damage
- Early flaw detection to enhance safety and allow for less drastic and less costly repairs
- Eliminate normal human factors concerns through the use of automated, uniform deployment of SHM sensors and automated data analysis (improved sensitivity)
- Detection of blunt impact events occurring during operation
- Reduction of inspection time
- Allow for maintenance-on-demand (Condition Based Maintenance) in lieu of current time- or cycle-based maintenance practices
- Accommodate performance trend analyses and timely, possibly even pre-emptive, corrective actions.

Long Term SHM Benefits

- Optimized structural efficiency (weight savings)
- New design philosophies (SHM designed into the structure)
- In-depth assessments of operational environments to produce knowledge-based maintenance processes
- Provide information to aid in-flight decisions
- Accumulate information to study performance history, automatically identify trends, and suggest corrective maintenance if necessary
- Allow for maintenance credits based on usage history and oversight provided by SHM.

In recent years, turn-key self-sufficient SHM systems have been evolved using networks of integrated sensors for the continuous monitoring, inspection and damage detection of structures to reduce labor cost and human error. Figure 1-3 summarizes some of the technology advancements that have occurred to make SHM solutions a viable alternative to traditional NDI practices. In principle, SHM in commercial airplane applications have the potential to detect structural discrepancies, determine the extent of damage, determine effects of structural usage, and eventually determine the impact on structural integrity and continued airworthiness. SHM systems can also be used to monitor loads and strain fields, or other critical environments, in order to better evaluate the state of the structure or mechanism.

Figure 1-4 through Figure 1-6 show the general architecture for an SHM system and how it might operate within an aircraft maintenance program. Note the use of multiple inputs to the aircraft health assessment via: 1) sensors that directly measure damage or provide pre-cursors to damage, 2) structural analyses, and 3) loads and environmental monitoring that can help guide and focus maintenance activities.

- Evolution of miniaturized sensors & supporting technology
- Design of turnkey systems with reasonable costs
- Ability to monitor new & unexpected phenomena (new inspection needs; DTA and rapid flaw growth)
- Promise for technical & economic gains more clearly defined
- OEM willingness to explore SHM merits
- Long-term prognosis -
  - Complete health assessment with network of SHM "nerves"
  - Automated data transmission (real-time monitoring; alarms)
  - Embedded sensors (MEMS)
  - Improved diagnostics using neural networks (historical data)
  - Direct ties to maintenance planning and actions
  - Reduction in life-cycle costs

Figure 1-3: Technology Advancements to Make SHM a Viable Alternative to Alternate Health Monitoring Methods

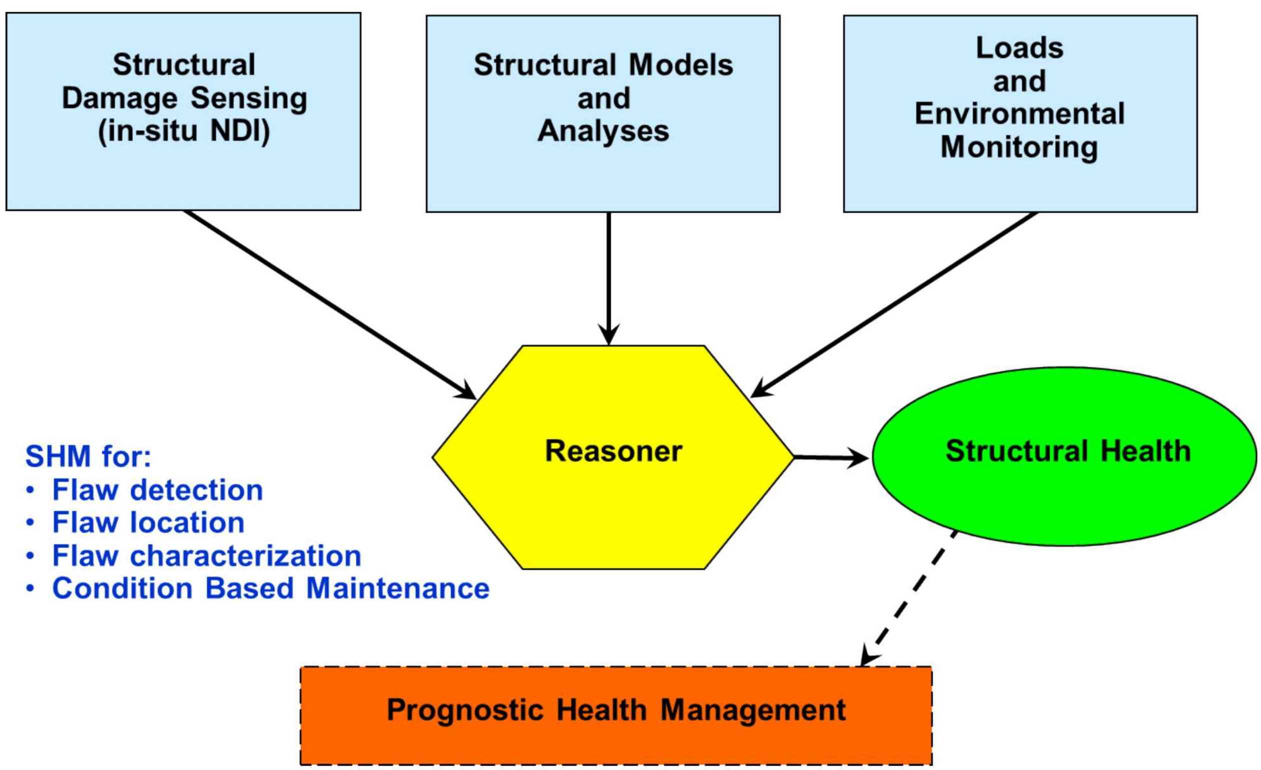

Figure 1-4: Premise of Structural Health Monitoring - Basic Operation of an SHM System within an Aircraft Maintenance Program

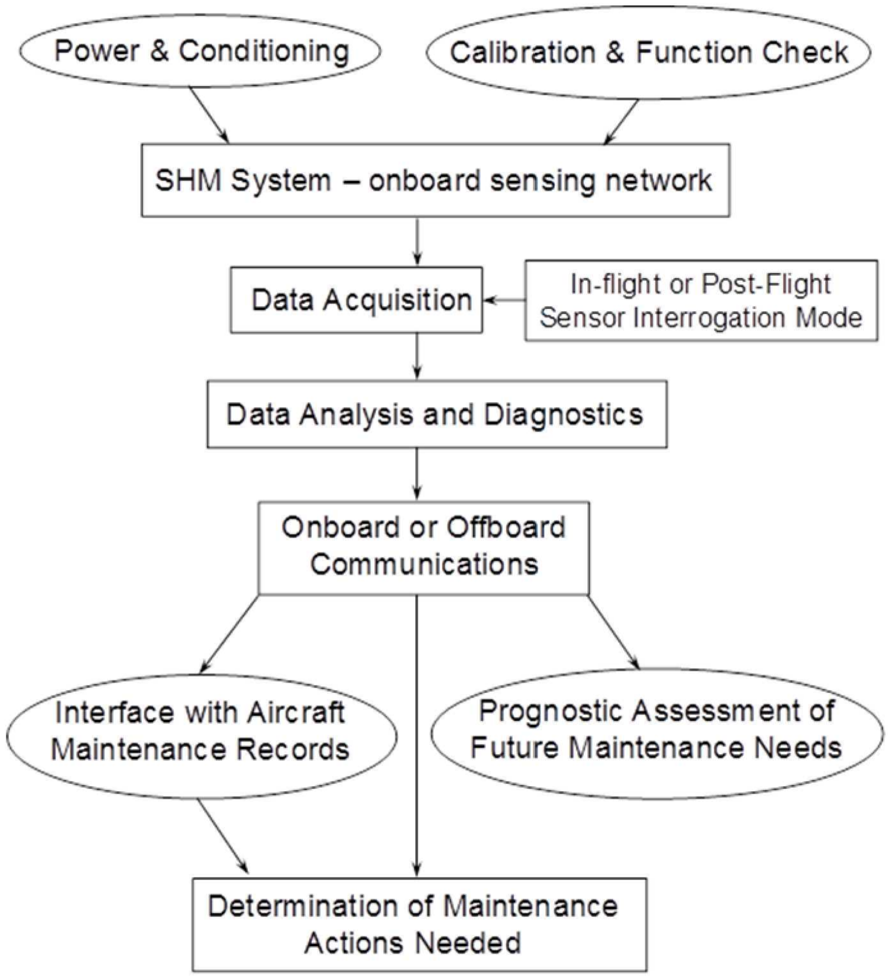

Figure 1-5: Operation of an SHM System within an Air Carrier's Maintenance Program

Figure 1-7 and Figure 1-8 show a wide range of structures from multiple industries such as oil and gas, transportation, mining, and renewable energy, where SHM solutions can address structural monitoring needs. In general, SHM sensors should be low profile, lightweight, easily mountable, durable, and reliable. To reduce human factors concerns with respect to flaw identification, the sensors should be easy to monitor with minimal need for users to conduct extensive data analysis. Figure 1-9 compares two styles of SHM sensors. The deterministic sensor is able to produce a signal (or change in signal) that directly indicates the presence of damage. Oftentimes, the parameter used to describe the sensor output is generally referred to as the Damage Index (DI). When the DI level exceeds a certain, predetermined threshold, the sensor is detecting damage in the structure. Other sensors may fall into the category of derivative. These type of sensors can use some well-defined structural response, such as strain, displacement or temperature to infer the presence of damage. These sensors can work equally as well as deterministic sensors for SHM applications, however, additional testing and calibration is required to properly relate their output to structural damage. Figure 1-10 provides several examples of mountable, in-situ SHM sensors.

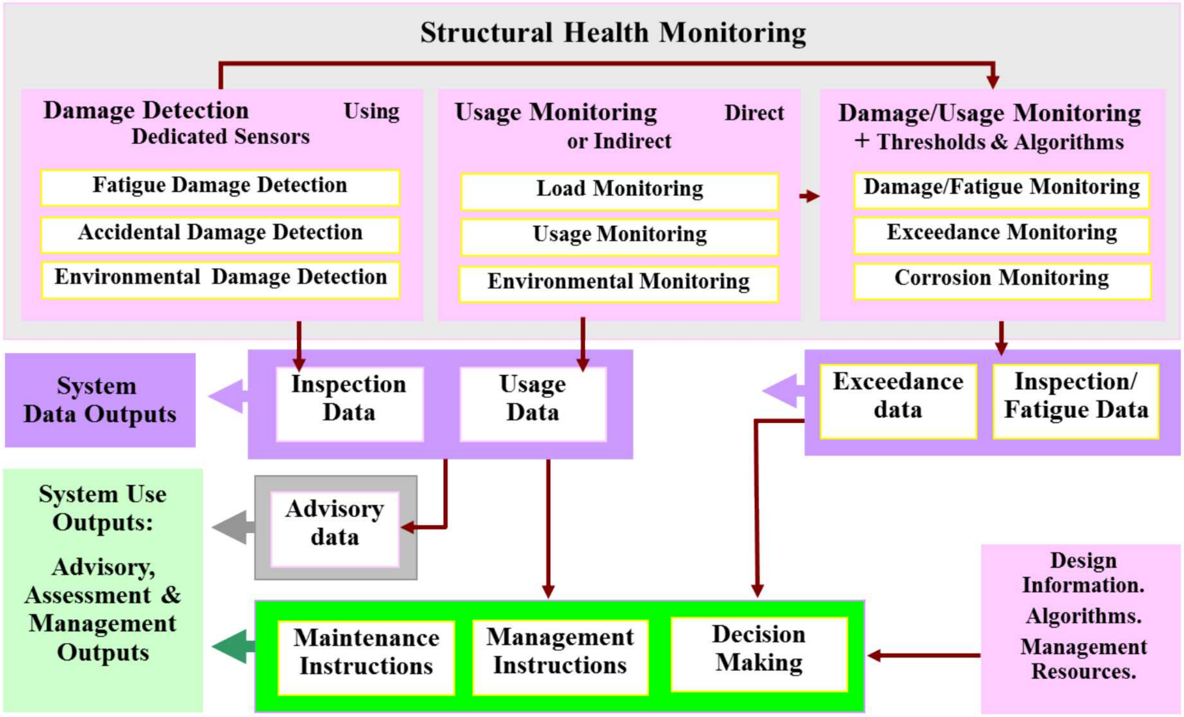

Figure 1-6: Potential Functions of SHM Systems

Figure 1-7: Sample Structures Showing a Wide Range of Uses for SHM Systems (Part A)

Figure 1-8: Sample Structures Showing a Wide Range of Uses for SHM Systems (Part B)

For optimum performance of the in-situ sensor-based approaches, the signal processing and damage interpretation algorithms must be tuned to the specific structural interrogation method. Initial research has highlighted the ability of various sensors to detect common flaws found in composite and metal structures with sensitivities that could exceed current flaw detection requirements, if needed. Use of SHM solutions in routine maintenance activities can only be achieved by overcoming the basic obstacles listed in Figure 1-11. Programs such as the one described here and many other evolution and validation efforts underway within the SHM community have addressed these potential roadblocks and have created an environment where the application of SHM systems is possible. Completed validation programs at the Sandia Labs AANC – conducted jointly with aircraft manufacturers and airlines – worked to integrate SHM sensors into aircraft maintenance programs. These evaluations incorporated both cost-benefit analyses, as well as statistically-derived performance reliability numbers.

Whether the sensor network is hardwired to an accessible location within the aircraft or monitored in a remote, wireless fashion, the sensors can be interrogated in a real-time mode. However, it is anticipated that in the initial application of SHM technology, the sensors will most likely be examined at discrete intervals; probably at normal maintenance checks. The important item to note is that the ease of monitoring an entire network of distributed sensors means that structural health assessments can occur quickly and in an automated fashion [1.12 – 1.16].

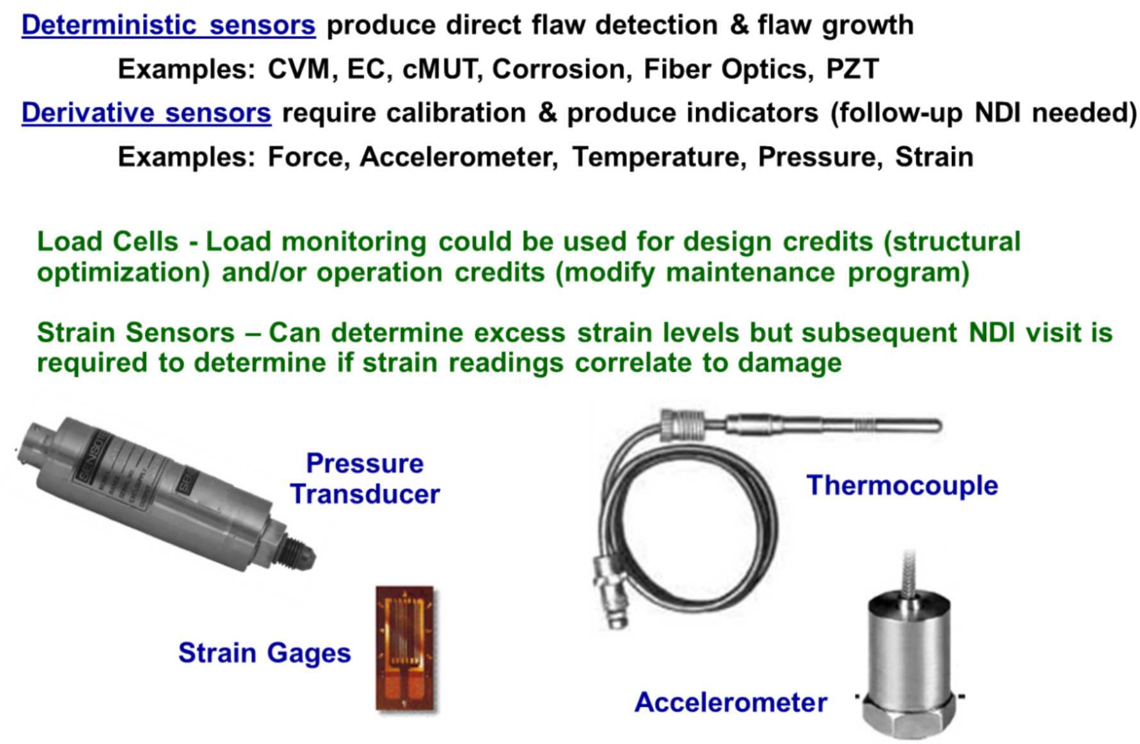

Figure 1-9: Deterministic vs. Derivative Sensors for Health Monitoring Applications

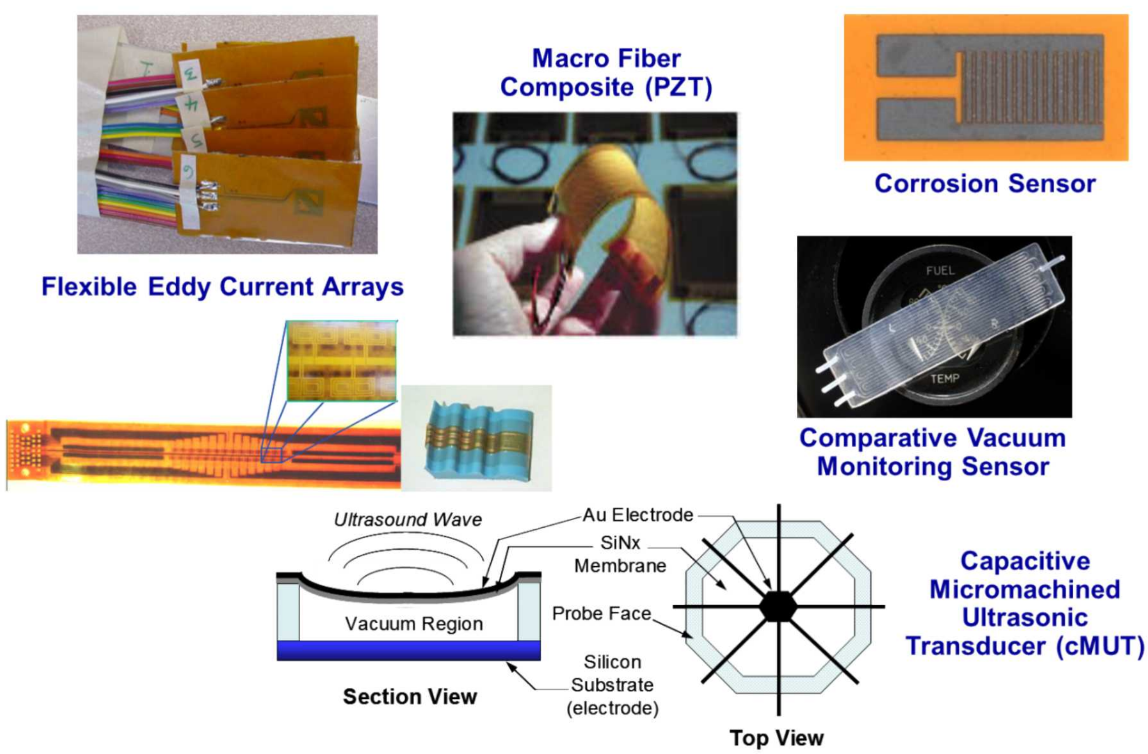

Figure 1-10: Examples of SHM Sensors

- Cost of sensors and sensor systems
- Ease of use and coverage area
- Need for rapid customization of sensors
- Need for substantial business case (cost-benefit analysis) – operators must realize benefits of multi-use
- OEMs may need to own technology
- Small-scale damage must be detected in large-scale structures
- Validation activities – general performance assessments needed; reliability of SHM systems must be demonstrated
- Validation activities – field trials on operating aircraft is necessary but time consuming
- Certification – need to streamline specific applications; technical, educational and procedural initiative (OEMs, operators, regulators)
- Standardization needed for validation and certification activities
- Technology transfer and implementation requires changes in maintenance programs

Figure 1-11: Impediments and Challenges to SHM Deployment

Several SHM sensors have been demonstrated to reliably detect damage both in the laboratory environment and in commercial applications. One example of a more mature sensor that can detect cracks and structural defects is the Comparative Vacuum Monitoring (CVM) sensor. A number of organizations have been investigating and demonstrating the use of CVM as a means for inspecting certain commercial airplane applications [1.8, 1.17 - 1.22]. In the CVM applications studied to date, the CVM technology is a permanently mounted nondestructive damage detection sensor that can be queried at the same inspection intervals as the currently accepted NDT methods. The advantage of the CVM in this case is that the inspected structure only needs to be accessed once for CVM installation. Afterward, the area is inspected by remotely connecting to the CVM without need for structural teardown. This program involved a detailed investigation into CVM technology with an emphasis on a specific aircraft application and a desire to produce approved, routine use of this SHM solution.

The interest in SHM has risen dramatically in recent years. Driven by the potential for both technical and economic solutions, OEMs and airlines currently have groups of engineers engaged in developing and applying SHM solutions to aircraft monitoring needs. Figure 1-12 shows a summary of just some of the agencies that are studying the integration of SHM into routine aircraft maintenance. Figure 1-13 shows several, traditional hand-deployed NDI equipment along with the signals generated during the inspections. It highlights some of the challenges associated with signal interpretation that can be simplified through the use of SHM systems.

Figure 1-12: Sample Organizations within the Aviation Community that are Studying the Integration of SHM into Routine Aircraft Maintenance

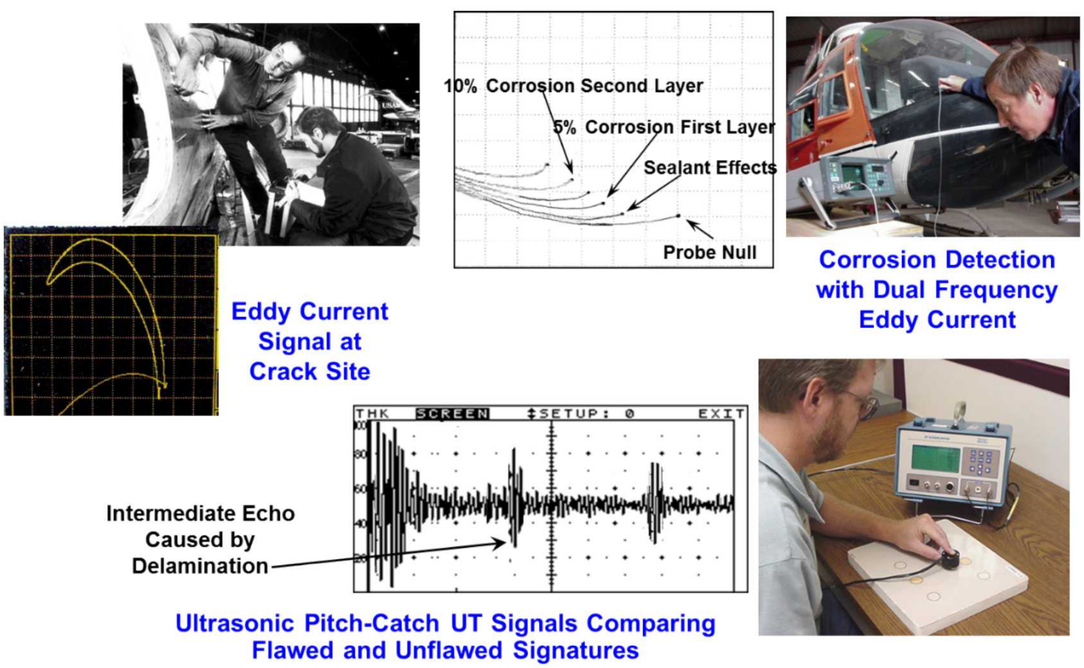

Figure 1-13: Typical A-Scan Signals Used for Flaw Detection with Hand-Held NDI Devices Highlighting Signal and Human Factors

1.2 Industry Survey and Insight on Potential SHM Usage

An important element in developing and applying SHM solutions for the aviation industry is a clear understanding of the current status of SHM technology and the pending regulatory issues facing the aviation industry to safely adopt SHM practices. To acquire such information, the AANC used a survey to collect information on industry interest in deploying SHM solutions. The comprehensive survey, implemented with the aviation industry, determined the technology maturation level of SHM, identify integration issues, and prioritize research and development needs associated with implementing SHM on aircraft. The survey was implemented via a customized, on-line web site and was sent to persons involved in the operation, maintenance, inspection, design, construction, life extension, and regulation of aircraft as summarized in Figure 1-14. Specific emphasis was placed on structural and maintenance characteristics that may impact the operational performance of an inspection process or health monitoring system. Over 450 people responded to the survey to provide industry information on SHM deployment and utilization, validation and certification, SHM standardization, sensor evolution and operation, cost-benefit analysis, and SHM system description. The survey results were initially used by the FAA to identify and prioritize research and development needs associated with implementing Structural Health Monitoring (SHM) on aircraft.

Below are just a few results excerpted from the in-depth presentation of the overall results obtained from the SHM Industry Survey [1.23]. Overall, it was determined that there is a strong interest in SHM. Industry's main concerns with implementing SHM on aircraft are achieving a positive cost-benefit and the time to obtain approval for SHM usage. OEMs and airlines felt that research and development efforts should be focused on: global systems, sensor technology, system validation and integration, and regulatory guidance. In addition, they felt that standardization and guidelines are needed in validation, certification, and sensor design with aviation in mind.

Over 200 applications, covering all aircraft structural, engine, and systems areas, were identified. The 80 applications provided as the respondent's first selection are listed below. The main trends of potential SHM applications include: general damage detection and crack detection in structural members (bulkheads), corrosion detection and coating monitoring, hard landing, load monitoring, impact detection and indication, hot spot monitoring, bolt tightness monitoring, strain levels, heat damage, monitoring of fuselage door and window areas, bond monitoring, delamination in composite structures, monitoring of existing cracks, monitoring fuselage skin repairs and flaw detection in difficult-to-inspect/access areas.

Figure 1-15 shows that the majority of respondents think SHM is a viable alternative to nondestructive testing. More than half of respondents think 5 years is a reasonable timeframe to recoup the costs of an SHM system while almost 1/3 of the respondents felt that 2 years was reasonable. Payback period is one of industry's biggest concerns in considering the use of an SHM system. Figure 1-16 shows that over 50 percent of respondents think that all of the primary structural areas are candidates for SHM applications: fuselage pressure bulkhead, frames, stringers, wing ribs and spars, landing gear, main attachments and skin areas. In fact, there were no aircraft regions that received insignificant responses. Areas where respondents are less interested in implementing SHM were: power train and nonstructural systems.

Owners/Operators	OEMs	Regulators	Maintainers
All Nippon Airways American Airlines Austrian Air Force China Airlines Continental Airlines Delta Air Lines Federal Express Finnair Hawaiian Airlines Japan Airlines Jazz Airlines Jet Blue Airways Kalitta Air LLC NASA Qantas Airways Singapore Airlines Swiss Air United Airlines US Airways USAF US Army USCG US Navy	Airbus Astronics-Adv. Electronic Systems Avensys Inc. BAE systems Bell Helicopter Textron Boeing Bombardier Aerospace Cessna Aircraft Company Dassault Aviation EADS Military Air Systems Embraer Goodrich Honeywell Lockheed Martin Aeronautics Messier-Dowty Mistras Group, Inc Polskie Zaklady Lotnicze Sp. PZL Swidnik Rolls-Royce Corp Systems & Electronics, Inc. TecScan	Air Transport Association CAA - NL CAA - Bra EASA FAA NAVAIR NAWCAD Transport Canada (TCCA) USAF US Army USCG US Navy	Aerotechnics Inc Air New Zealand China Airlines Christchurch Engine Centre Fokker Aircraft Services B.V. Fuji Heavy Industries, Ltd. Jazz Air LTD Lufthansa Technik AG NASA Olympic Airways Services S.A. SAA Technologies SR Technics Switzerland LTD Texas Aero Engine Services Timco / GSO United Airlines USAF US Army USCG US Navy

Over 450 responses from OEMs, regulators, operators, and research organizations.

Figure 1-14: SHM Survey of Aviation Industry to Gage Interest and Range of Applications for SHM

First SHM Application Listed by Survey Respondents:

1. Overload monitoring and detection
2. Overload monitoring
3. Bolt torque monitoring
4. Debonding detection and assessment in specific areas
5. Hard /heavy landing
6. Impact damage detection
7. Airframe monitoring
8. Door hinge area
9. Corrosion detection
10. Anything that reduces operating costs
11. Key hot spots (locations that are known to develop damage and require additional inspections
12. Moisture detection in wet areas (galley, lavatory etc.)
13. Landing gear overload detection
14. Corrosion detection
15. Composite structures (delamination and other damage)
16. Areas that require disassembly for routine inspection
17. Composites

18. Monitor moisture in corrosion prone areas
19. Crack detection in structural critical areas, with the evaluation of crack length or other parameter to assess criticality
20. Leading edge composite disbond
21. Inaccessible areas that require major tear down
22. Fiber breakage and delamination in composite structures
23. Monitoring known crack locations
24. Crack growth monitoring in difficult to access regions of airframe as an AMOC to manual inspection.
25. SHM of UAV composite structures
26. Crack detection in high load areas such as door cutout
27. Wing lug attach fittings
28. Landing gear attach points
29. Impact detection
30. Propagation rates of disbond/delamination of composites
31. Around fuselage doors cutout
32. Fuselage door
33. B747
34. Repair and bonded patches
35. Corrosion prevention, detection and sizing
36. Delaminations on hidden areas of honeycomb flight control structure
37. Primary structures
38. Structural damage
39. Cracks in lap-splice joints
40. Composites damage
41. Monitoring for impact damage during aircraft operation due to bird, tree, hail strike
42. Hot spots
43. Conventional NDI replacement
44. CRJ - 559 area
45. Aging aircraft with known structural health issues
46. No access (costly access) structure
47. Corrosion detection
48. Stabilizer shim migration
49. Cracks in the airframe
50. Composite structures that may get heat damaged, inner fixed structures of thrust reverser
51. Fuselage skin
52. Aft pressure bulkhead
53. Commercial aircraft
54. Compressor and turbine blades (tip timing method)
55. Frames
56. Landing gear fittings
57. Tension bolts
58. Tail-strike indicator (already in use on A340+A380)
59. Structural cracking
60. Bonded structures monitoring
61. Rotor vibration monitoring

62. Lightning hit damage detection at the location where access is difficult (e.g. top of fuselage, vertical and horizontal stabilizers)
63. Any difficult-to-access location
64. Hard landing detection
65. Corrosion in hard to access areas in bilge
66. Structure integrity in load carrying composite structures
67. Fuselage skin
68. Heavy landing event monitoring
69. Structural fatigue
70. Closed areas with no access to either side
71. Cracks in pressure bulkheads
72. Fuselage skin crack detection
73. Frame shear angles.
74. Corrosion detection
75. Monitoring structural repairs
76. Critical bolts (hot spots) - small cracks
77. Crack detection in metallic components
78. Corrosion assessment in bays
79. Multilayer crack detection at fastener holes
80. Flight control abnormal loading

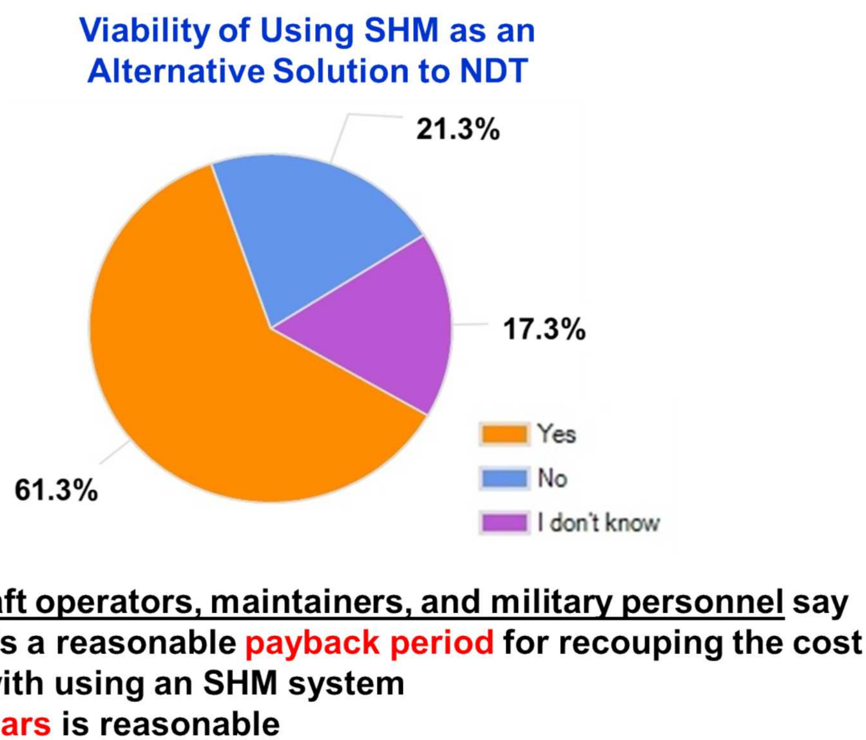

Figure 1-15: Survey Results Indicating that a Majority of Airlines are Interested in Using SHM

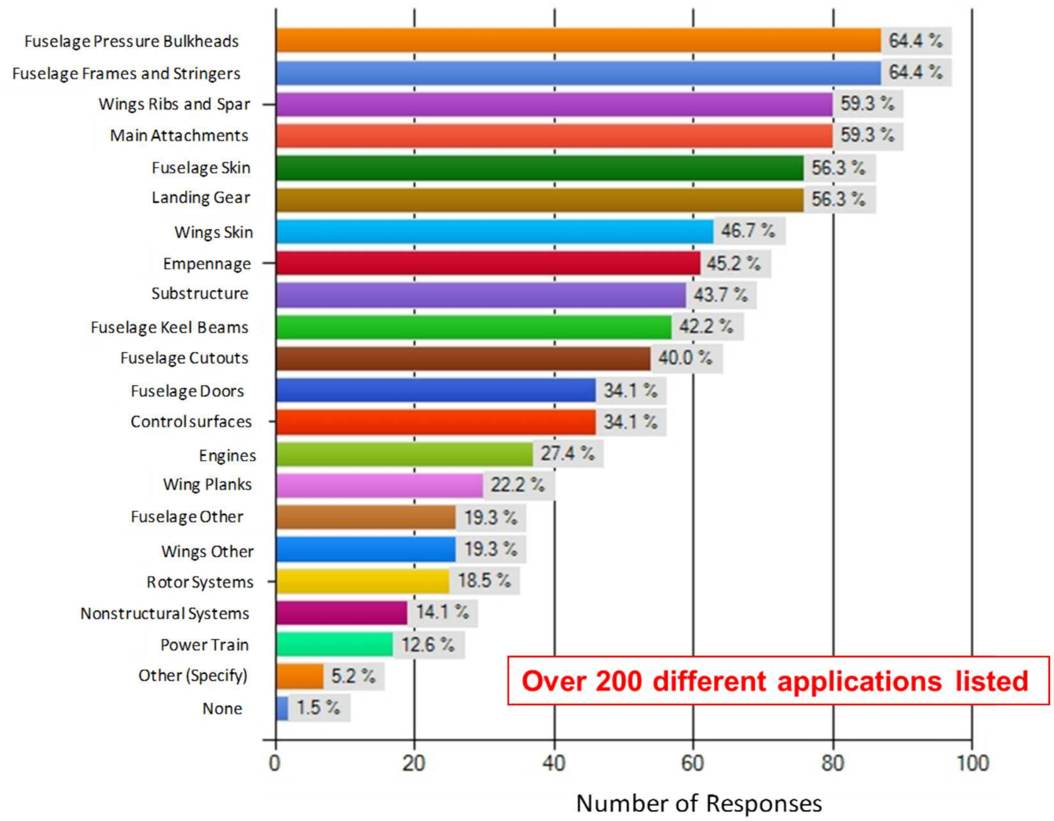

Figure 1-16: SHM Survey Result - Areas Respondents Feel SHM Solutions are Viable

Figure 1-17 summarizes the types of damage/flaws the industry is interested in detecting. It's not surprising that a large majority of the persons surveyed were interested in detecting the major damage types found on aircraft: cracks, corrosion, delaminations and disbonds. Related damage from stress risers, impact, fluid ingress, and other environments are also cited often. Damage associated with composites, exposure, mechanical malfunction and off-design conditions (e.g. ground support activities) are were also listed. Overall, the potential damage and malfunctions where respondents would like to utilize SHM covered a very broad spectrum of applications with the majority of the damage types being listed by over 1/3 of the survey participants.

In the next five years, a majority of the systems being planned for application are local or hot spot monitoring systems. Figure 1-18 shows that 85% of those surveyed anticipate applying local systems and only 15% believe that global SHM systems will be applied within the five year time frame. In the survey, local implies focused evaluation of specific areas that currently require local inspections; often associated with a Detailed Visual Inspection or a Special Detailed Inspection. Global implies evaluation of large areas such as control surfaces or fuselage panels; often associated with a General Visual Inspection or some wide-area NDI task

Figure 1-19 shows that the main reasons respondents are interested in SHM are associated with cost considerations (e.g. avoiding disassembly, reduction in labor hours) and safety/reliability considerations (e.g. early flaw detection, improved sensitivity). Another item of note is that almost all of the possible reasons for using SHM were listed in over 1/3 of the survey responses. Reasons that were deemed as less important pertained to obtaining maintenance credits, design credits or weight savings, and monitoring electrical and aircraft systems. These are mostly long-term prospects for SHM so it is not surprising that these are currently of less interest to end-users.

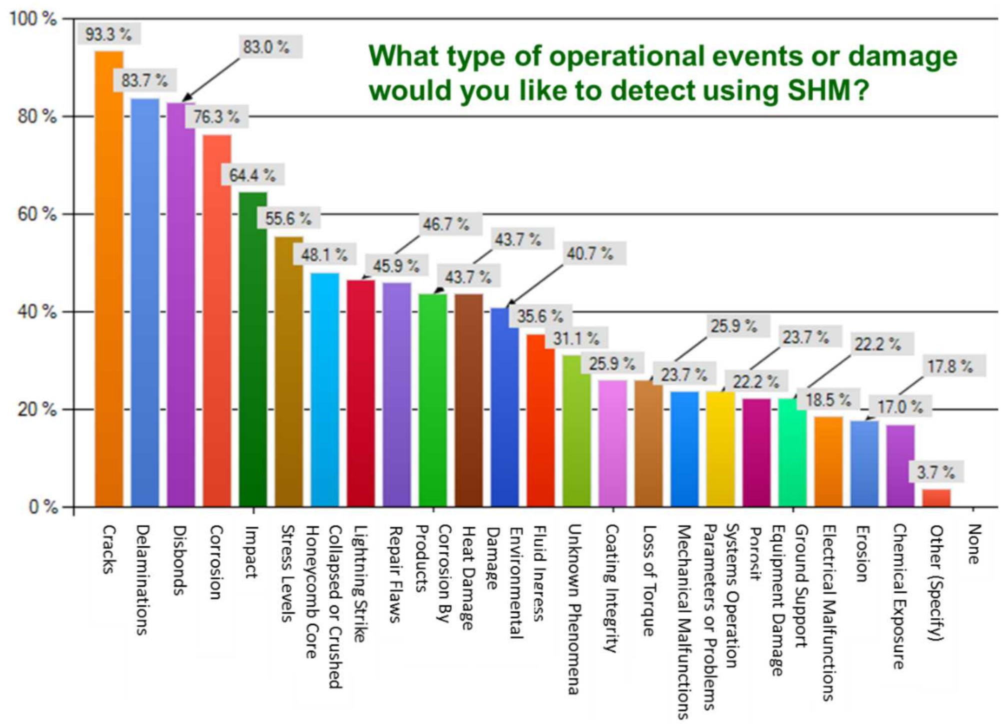

Figure 1-17: SHM Survey Results Listing the Damage that Users Would Like to Detect

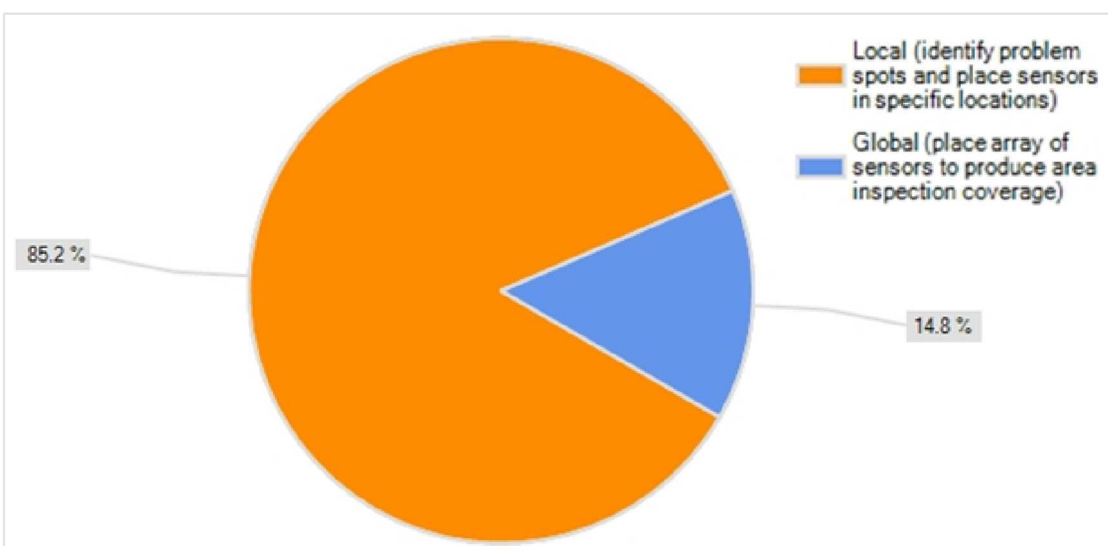

Figure 1-18: Type of SHM Expected to be Deployed in the Near-Term

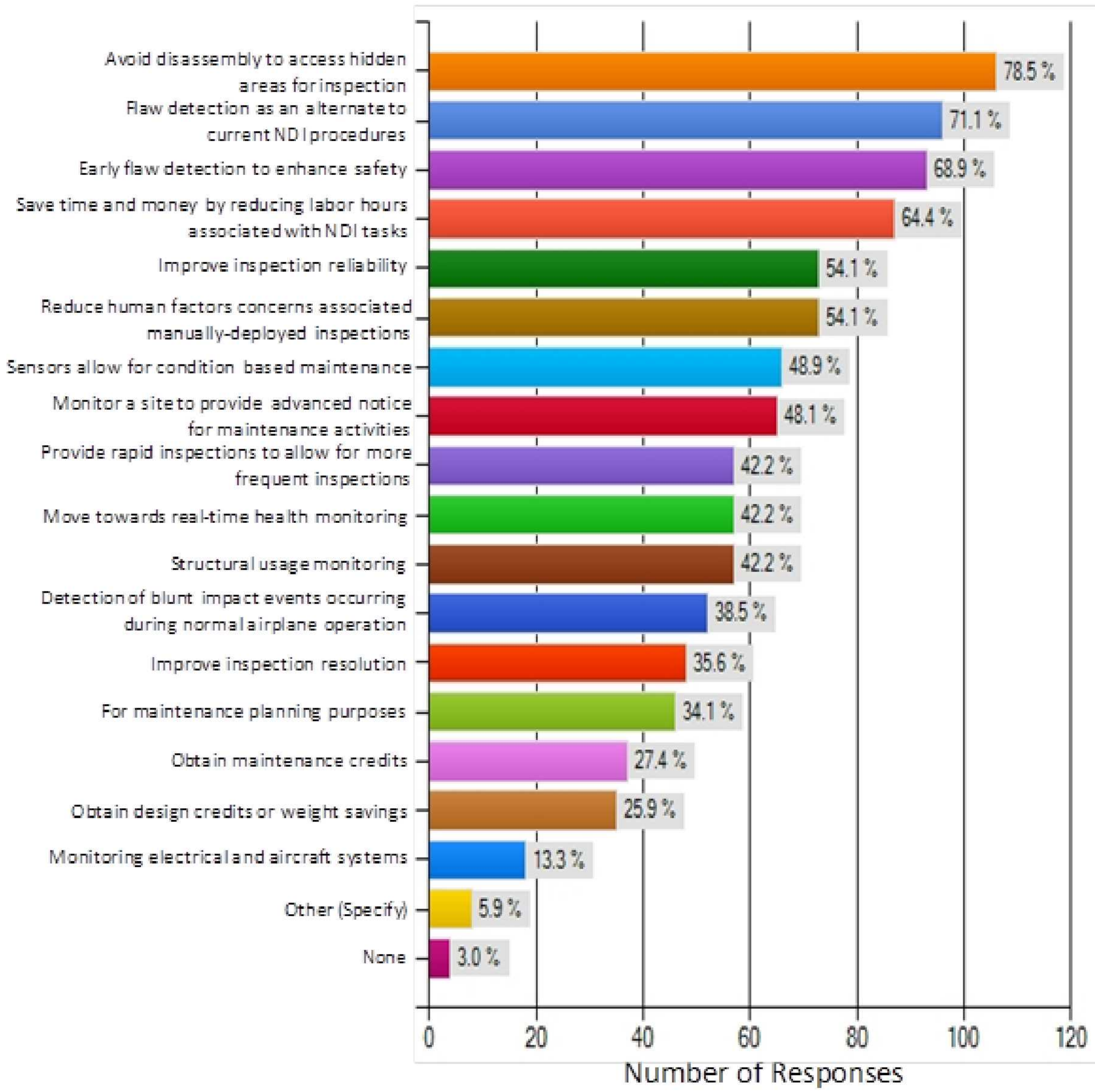

Figure 1-19: Respondents Reasons for Interest in SHM

Table 1-1 contains the prioritized list of the most important items in determining the cost-benefit of using SHM systems on an aircraft. The most important factor (52% had it as a response priority of 5) is the elimination of structural tear down to access areas to be monitored. Other items receiving at least 30% response level and a priority of 4 or 5 include: initial cost of SHM equipment, recurring cost of SHM sensors, time required for validation/qualification, time required to obtain permission for use from regulators, compliance requirements, and the frequency that the SHM system will be used.

What are the most important items when considering the cost-benefit of implementing an SHM solution?(prioritize with 1 being lowest and 5 being highest priority)					
Answer Options	**1**	**2**	**3**	**4**	**5**
Initial cost of SHM equipment	4%	7%	24%	**35%**	30%
Recurring cost of SHM sensors	0%	10%	28%	**40%**	22%
Replacing existing inspections with more rapid monitoring	1%	13%	29%	**31%**	26%
Elimination of structural teardown to access a region to be monitored	0%	1%	21%	25%	**52%**
Frequency of potential SHM utilization (inspection intervals)	1%	4%	**43%**	31%	19%
Cost of validation	4%	6%	**37%**	27%	27%
Cost of qualification	3%	8%	**33%**	29%	26%
Time required for validation/qualification	3%	9%	32%	**35%**	21%
Compliance requirements – existing or future needs	0%	13%	28%	**34%**	24%
Time required to obtain permission for use from OEM	1%	9%	**46%**	28%	16%
Time required to obtain permission for use from regulators	3%	6%	32%	**33%**	26%
Training required for maintenance personnel	7%	27%	37%	19%	9%
Need to adjust maintenance program to accommodate SHM	7%	20%	29%	29%	16%

Table 1-1: Most Important Items for Determining the Cost-Benefit of Implementing an SHM Solution

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CHAPTER 2

2.0 SHM Program Overview – CVM Validation

2.1 SHM Validation Process

The AANC's validation approach is designed to address the equipment, the inspection task, the resolution required, the inspection procedures, the conditions under which the inspection will occur, and the potential inspector population. To carry out the validation process, knowledge of aircraft maintenance practices must be coupled with an unbiased, independent evaluation. The AANC has designed, implemented, and analyzed the results from a wide range of statistically-relevant experimental programs to quantify the reliability of inspection methods as deployed at commercial aircraft maintenance facilities. Much of this methodology to quantify NDI performance can be adapted to the validation of SHM systems. However, it is important to recognize the unique validation and verification tasks that arise from distinct differences between SHM and NDI deployment and flaw detection. An important element in developing SHM validation processes is a clear understanding of the regulatory measures needed to adopt SHM solutions along with the knowledge of the structural and maintenance characteristics that may impact the operational performance of an SHM system.

The Airworthiness Assurance NDI Validation Center (AANC) at Sandia Labs, in conjunction with Boeing, Structural Monitoring Systems, and multiple, interested airlines has conducted a long-term research program to develop and validate Comparative Vacuum Monitoring (CVM) Sensors for crack detection. CVM sensors are permanently installed to monitor critical regions of a structure. The CVM sensor is based on the principle that a steady state vacuum, maintained within a small volume, is sensitive to any leakage. Vacuum monitoring is applied to small galleries that are placed adjacent to a second set of galleries maintained at atmospheric pressure. If a flaw is not present, the low vacuum remains stable at the base value. If a flaw develops, air will flow from the atmospheric galleries through the flaw to the vacuum galleries. A crack in the material beneath the sensor will allow leakage resulting in detection via a rise in the monitored pressure.

The initial goal of this project was to provide Boeing Commercial Aircraft with sufficient data to place CVM sensor technology into the Nondestructive Testing Standard Practices Manual. The test specimens included those designed to simulate the Boeing aircraft lap joint and others with single crack origination sites. The test matrix studied the effects of surface coating, skin thickness, and material type on the performance of the CVM sensors. Statistical methods using one-sided tolerance intervals were employed to derive Probability of Detection (POD) levels for each of the test scenarios. The result is a series of flaw detection curves that can be used to propose CVM sensors for aircraft crack detection. Complimentary, multi-year field tests were also conducted to study the deployment and long-term operation of CVM sensors on aircraft.

The follow-on effort looked at the application of SHM solutions to a particular aircraft application. In this case, the validation effort focused on the use of SVM sensors to detect cracks in the wing box fitting of a Boeing 737 aircraft. This data package presents the quantitative crack detection capabilities of the CVM sensor, its performance in actual flight environments, and the prospects for structural health monitoring applications on commercial aircraft.

Validation of Structural Health Monitoring Systems

The validation and certification process begins with the declared application intent, and a determination of the resultant criticality. The declared intent should specify whether this application is for credit (replaces required task or leads to changes in the requirements for a task) and if it adds to, replaces, or intervenes in maintenance practices or flight operations. When the declared intent is for credit, the end-to-end criticality for such an application should be determined and used as an input to establish the validation criteria. If the declared intent is for noncredit (provides additional data above and beyond required tasks), it may be certified as long as it can be shown that the installation of the equipment will not result in a hazard to the aircraft. Therefore, criticality describes the severity of the result of an SHM application failure or malfunction.

The program to implement SHM, and thus the validation plan, requires a clear definition of the application. There are several considerations that must be addressed when formulating this definition. These considerations include, but are not limited to, structural configuration, structural variation, usage environment, system durability requirements, configuration management, and system maintenance [2.1 – 2.7]. The SHM Validation Plan should address the following items:

1. Part Geometry –Engineering drawings that define specific dimensional information regarding the part or assembly, including the local structural interfaces, geometric interference, manufacturing variability and access. This drawings should define the geometry and composition of mating components and how these mating components are joined to the component under interrogation. The assembly defines the boundary conditions under which the SHM system must reliably function. The assembly configuration can affect the sensor design and placement
2. Material – The material description must include, in the case of metallic structure, the alloy type and heat treatment or temper condition, and may require a description of any surface treatments including coatings or plating and thicknesses. In addition, material details may be required for other structure located in the region of interest including fastener type and material composition.
3. Flaw Location and Orientation – A clear definition of both the expected flaw location and orientation is required. This information may be available in the form of damage tolerance analysis, and fatigue test results (subcomponent, component, or full scale).
4. Effectivity/Configuration Changes – A list of affected aircraft or systems by tail or serial number. This information should include a description of any deviations or configuration changes in component design, including variances in any of the items described above. Potential structural variability that could affect the reliability or repeatability of an SHM system should be defined. Sources of variability include but are not limited to variations in structural faying surface interfaces, coating systems, or part configuration often due in large part to production changes, repairs or deterioration of materials over time. Such an accounting provides a level of assurance that all affected systems are inspected and that SHM processes are appropriately adjusted to compensate for known variances.

5. Access for Installation/Stay-Out Zones – Points of access for installation and repair of the SDHM system must be identified. This description should include panels or doors that can be removed to facilitate system installation, description of local structure or subsystems that may hinder access, areas that can be used for cable routing or other system subcomponents, aircraft systems that may be affected by SHM hardware, and regions that cannot be used to mount SHM system subcomponents (stay-out zones).
6. SHM Performance/Capability – Provide both a goal and threshold $\boldsymbol{a}_{NDI}$ (or $\boldsymbol{L}_{NDI}$) value. The value of $\boldsymbol{a}_{NDI}$ has been established as the $\boldsymbol{a}_{90/95}$ Probability of Detection value determined statistically using appropriate methods. The $\boldsymbol{a}_{90/95}$ is an estimate of the crack size that will be detected 90% of the time with a statistical confidence of 95%. The goal value is the detection capability that may be very challenging to meet but would result in inspection intervals that provide an economic or maintenance benefit to the program. The goal and threshold values should be used to develop the SDHM demonstration experiment. In addition, these values should be used to develop SHM interrogation intervals.
7. False Positive Rates – False positives (also known as false alarms) can present a significant economic and availability burden if not appropriately controlled as they can drive costly and intrusive structural disassembly. The maximum rate of false positives can best be defined by the OEM or by operators.
8. Durability – System durability requirements, in terms of ability to operate in expected environments for specific periods without failure, should be defined. Failure rates must be sufficiently low to support the maintenance concept and provide long term monitoring without the need for invasive maintenance or repair of the monitoring system.
9. Usage Environment – The usage environment includes but is not limited to temperature profiles, humidity, fuel, hydraulic fluid or chemical exposure, strain and vibration. A definition of this environment will drive the design of environmental and durability testing and the qualification/airworthiness requirements.
10. Other Requirements – The SHM Validation Plan should clearly define other, pertinent aircraft specific requirements. These may include maximum system weight and size, power requirements, etc. Development of the MRD should be closely coordinated with the appropriate aircraft system and safety engineering authority within the operator's maintenance program.

The SHM Validation Program should use a multi-phased approach that includes controlled, representative laboratory testing that will eventually lead to on-aircraft flight tests. Each phase must address various aspects of the four critical factors (detection capability, durability, installation/supportability, safety) with a successful outcome supporting a decision milestone to move to the next phase. Validation testing can consist of mounting SHM sensors to representative specimens and cyclically loading the specimens to generate and grow fatigue damage. Preliminary testing may involve the use of simulated defects (e.g. electro-discharge machined (EDM) notches, simulated disbonds/delaminations) to represent damage but should progress to use of cyclically loaded fatigue damaged specimens.

The loading spectrum used for fatigue propagation should be based on the anticipated on-aircraft load environment; however, higher load rates may be required for economy. Test specimens must be manufactured from the same material, alloy, heat treat and possess a similar microstructure as the intended application. The sample design sufficiently complex (contain stiffeners, fastener holes, tapers, curves, etc. as appropriate) to represent the intended application but may not require the detailed replication of aircraft structure geometry or assembly. The goal for this phase is to demonstrate the system detection capability to sense and reliably identify relevant damage on structures in a relevant environment. A relevant environment is defined as test conditions that closely simulate the load spectrum when the test coupons are exposed to an environment similar to the intended application. Conditions that may have to be simulated include vibration, temperature, pressure, and exposure to moisture or aircraft fluids (hydraulic fluids, fuel, greases). The test samples should represent the intended application in terms of geometry, material, and assembly, including boundary conditions.

<u>SHM Validation Process Tasks</u>

The objective of any SHM technology validation exercise is to provide quantifiable evidence that a particular inspection or maintenance methodology (equipment plus its operation) is capable of achieving a satisfactory result. The validation process must consider the numerous factors that affect the reliability of an inspection methodology including the individual inspector/operator, his equipment, his procedures and the environment in which he is working. It also accounts for the viability of the SHM approach within the aircraft's maintenance program. The approach is based on the use of real-life Validation Assemblies which are full-scale structural assemblies containing known, realistic defects or other operational malfunctions which the SHM system is intended to monitor.

The validation process should: 1) provide a vehicle in which skills, automation of instrumentation and human error can be evaluated in an objective and quantitative manner, 2) produce a comprehensive, quantitative performance assessment of the SHM system and utilization procedure in a systematic manner, 3) provide an independent comparison between SHM solutions and alternate maintenance and monitoring methodologies, 4) optimize SHM utilization methodologies through a systematic evaluation of results obtained in laboratory and field test beds, 5) produce the necessary teaming between the airlines, aircraft manufacturers, regulators, and related SHM development and research agencies to ensure that all airworthiness concerns have been properly addressed.

The process of validating SHM techniques involves the specification of a structure with defects or containing the appropriate boundary conditions and features to allow for the assessment of whatever physical parameter the SHM system is monitoring. The validation process may involve the production of full size sections of airframes or appropriate laboratory test samples which contain natural, fully characterized defects or realistic, engineered defects. Inspection or monitoring of these Validation Assemblies must occur under conditions identical to those of the day-to-day inspection environment. The validation process is a full-scale, realistic mockup of the daily activities of the maintenance personnel involved in the proposed SHM application. The tests performed are then independently assessed against industry standards in terms of personnel and instrument performance. In this regard, independence and objectivity are essential. Some validation efforts may include the use of airline maintenance personnel who will perform the monitoring tasks using normal working practices and under normal working conditions (lighting, heating, noise, work shifts, etc.).

- Declared Intent - application is for credit (replaces task or leads to changes in the requirements for a task); criticality describes the severity of the result of an SHM application failure or malfunction

- Usage Mode for SHM System
  - "Hot spot" or local monitoring (S-SHM)
  - Prognostic and condition-based health monitoring (P-SHM and C-SHM) - shift to predictive and continuous monitoring will require extensive validation and successful in-service experience so that regulatory agencies and operators can acquire confidence in these SHM approaches

- Aircraft Maintenance Practices – change in programs; how to adopt

- Deployment – operational performance & repeatability

- Regulatory Actions and Industry Acceptance – depends on certification process (AMOC, NDT SPM, SB/AD, STC)

- Key element in an SHM system is a calibration of sensor responses so that damage signatures can be clearly delineated from sensor data produced by undamaged structures

- Commercial implementation of SHM needs to be proven through statistically-viable lab performance data and successful field operation data

- Data requirements need to be established for determining the applicability of SHM (boundaries) and to address certification requirements

- Educational initiatives with key players – understanding of SHM, its usage and its limitations

Figure 2-1: Considerations for Producing an SHM Validation Plan

- **Validation Process should:**
  1) **provide a vehicle in which skills, instrument deployment & human error can be evaluated in an objective and quantitative manner**
  2) **provide an independent comparison between SHM solutions and alternate maintenance and monitoring methodologies**
  3) **optimize SHM utilization methodologies through a systematic evaluation of results obtained in laboratory and field test beds**
  4) **produce the necessary teaming between the airlines, aircraft manufacturers, regulators, and related SHM development and research agencies to ensure that all airworthiness concerns have been properly addressed**
- **Validation Assemblies – Assess technology and process; deployed under conditions identical to those of the day-to-day maintenance environment; use airline maintenance personnel who will perform the monitoring tasks using normal working practices and under normal working conditions**
- **Comprehensive Evaluation - Assess performance, training and integration into maintenance program (technical and admin)**

Figure 2-2: Considerations for SHM Validation Process Tasks

- **SHM Method - SHM solution, device, sensor spacing, data acquisition process, data analysis method, data interpretation (thresholds, S/N), use of baselines**
- **Structural Configuration – geometry, material type, number of layers, fastener types and spacing, hole geometry, assembly specifics (fit/gaps), surface condition, coating changes**
- **Flaw/Damage Condition – type, X-Y location, depth, orientation, dimensions, morphology, presence of by-products**
- **Environmental Conditions – load scenario to generate damage, impact, environment to generate damage & establish durability**

Complex Structure Requires Detailed SHM Validation

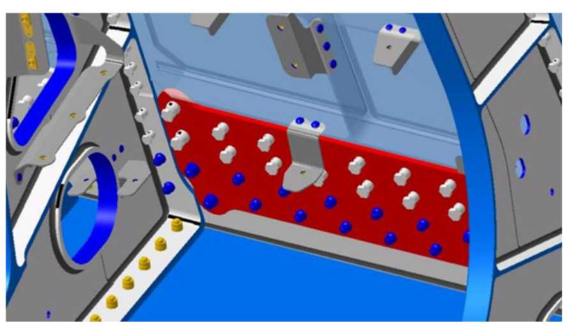

Figure 2-3: SHM Validation Process Must Account for All Factors That Can Affect Performance

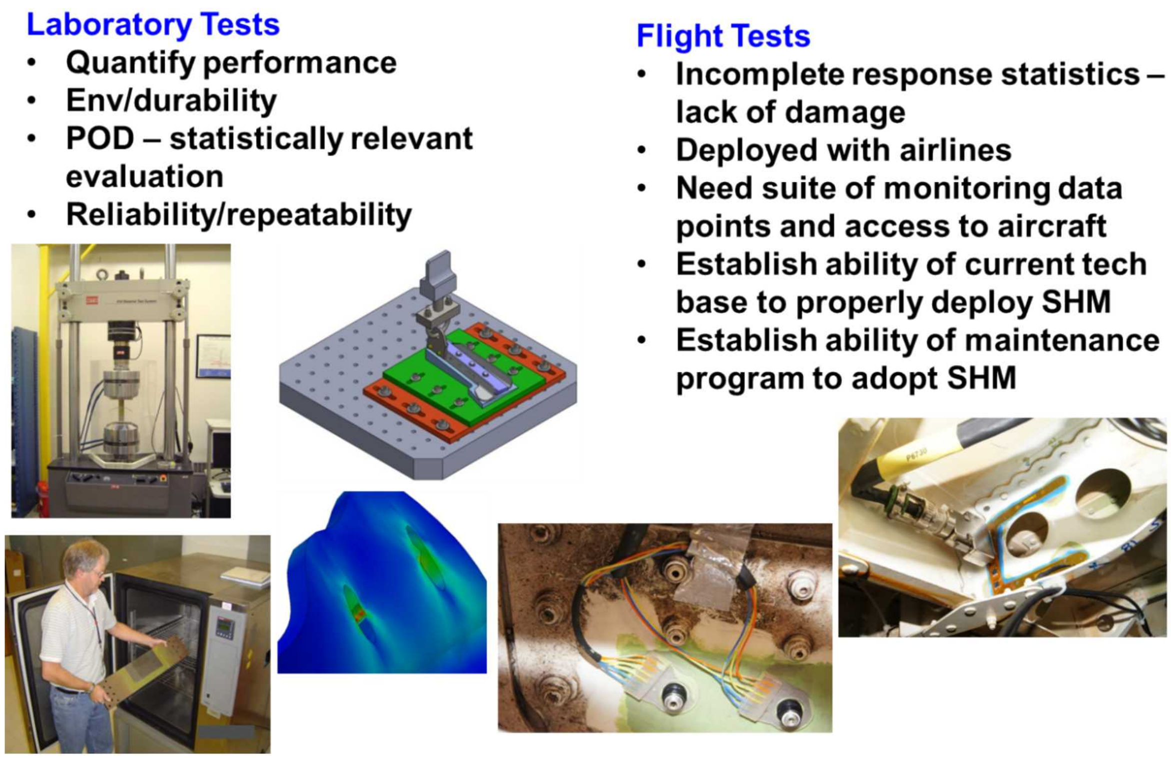

Figure 2-4: Two Major Components for Validation of SHM Capability

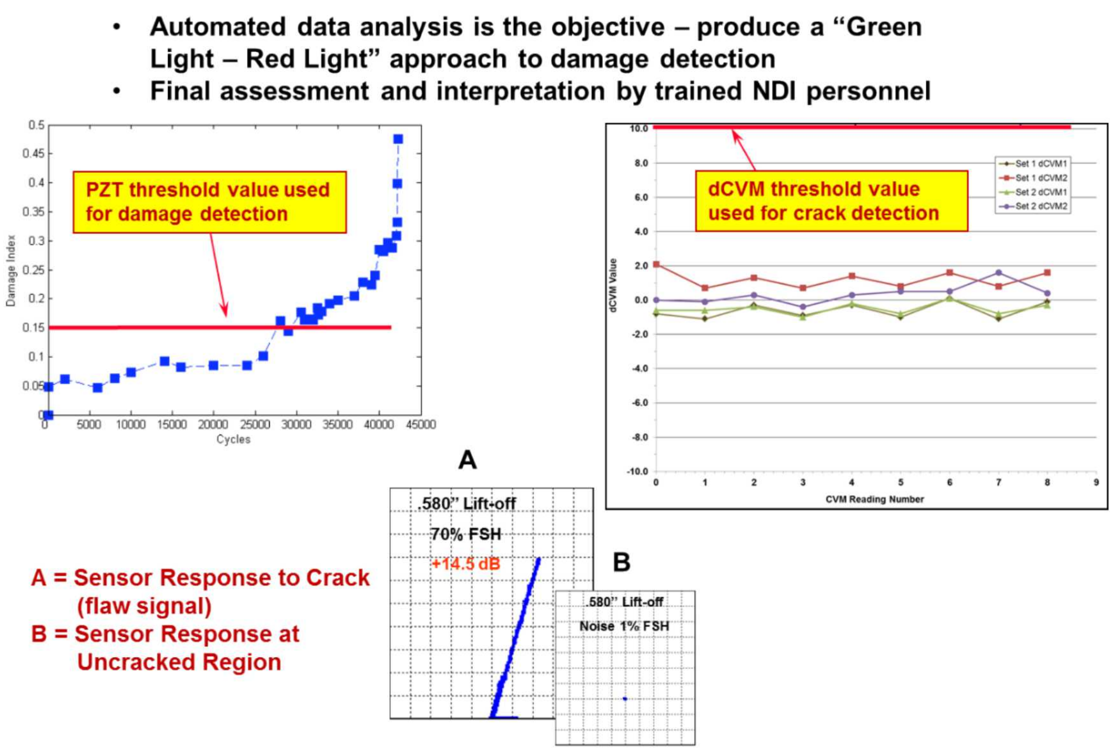

Figure 2-5: SHM Information – Importance of Establishing Damage Detection Thresholds and Minimizing Data Interpretation or Data Analysis

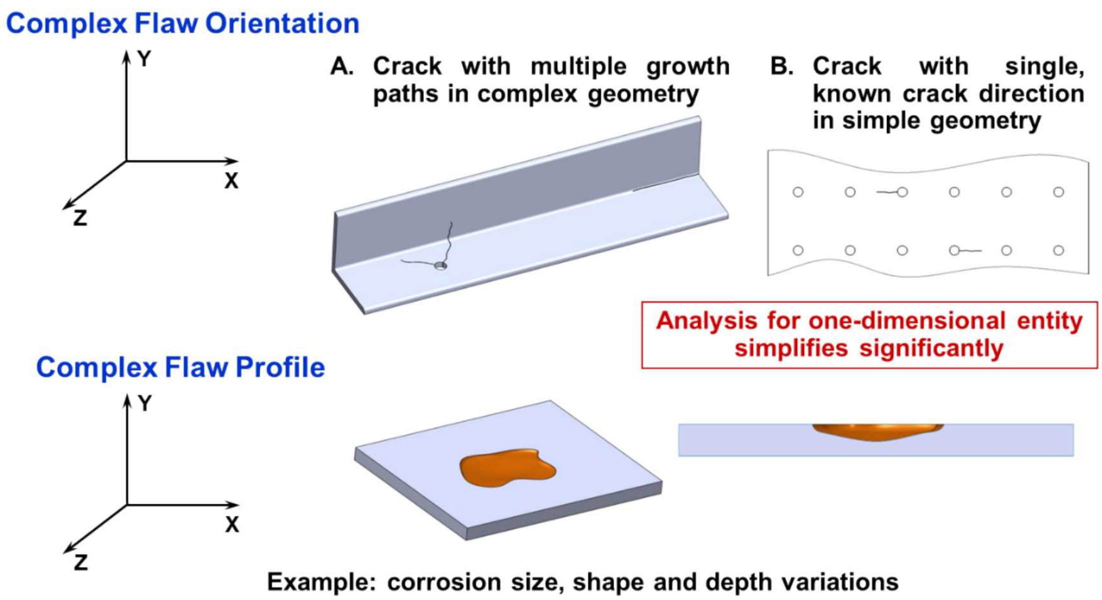

Figure 2-6: Reliability Assessment for Simple and Complex SHM Solutions

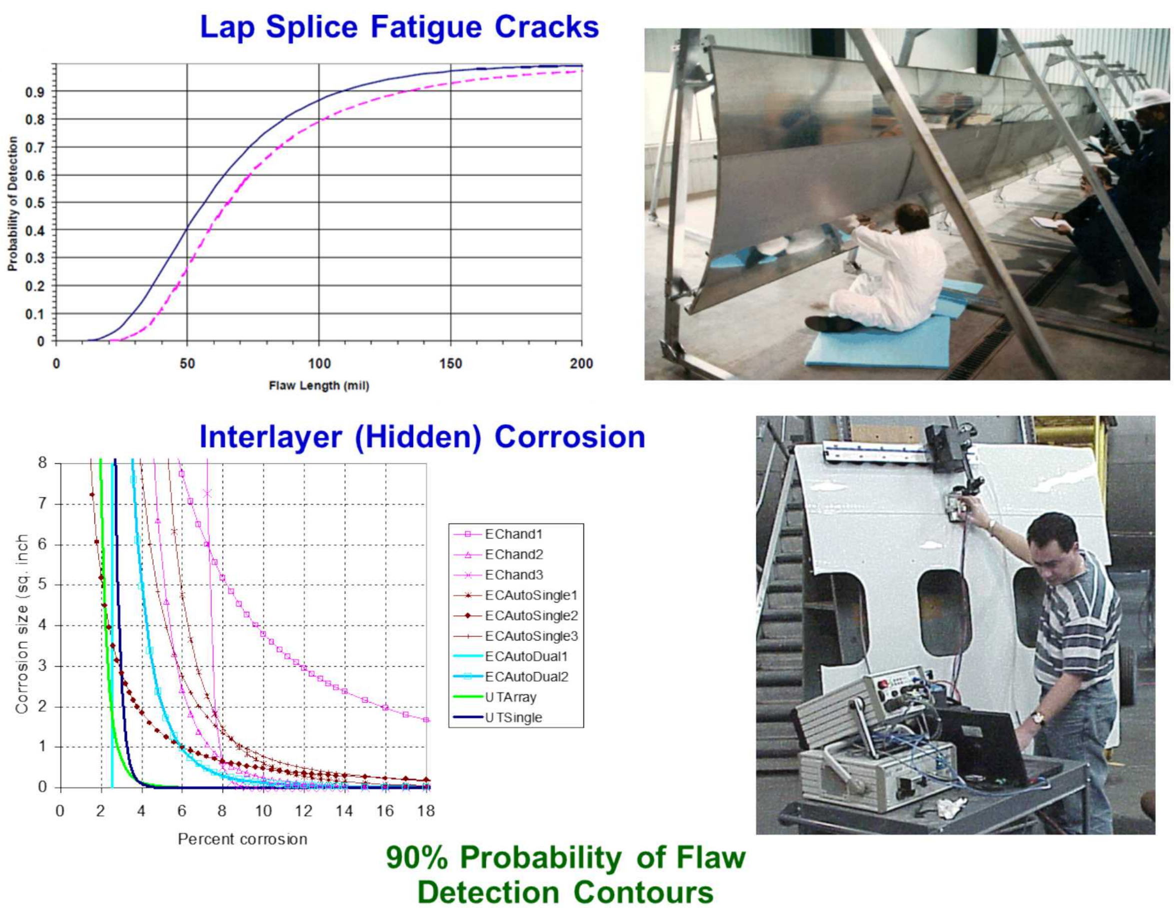

Figure 2-7: Approaches to Present NDI POD Values for Different Flaw Geometries

2.2 CVM Technology Description

Comparative Vacuum Monitoring (CVM) is a simple pneumatic sensor technology developed to detect the onset of cracks. CVM sensors are permanently installed to monitor critical regions of a structure. The CVM sensor is based on the principle that a steady state vacuum, maintained within a small volume, is sensitive to any leakage. A crack in the material beneath the sensor will allow leakage resulting in detection via a rise in the monitored pressure. The following graphics show top-view and side-view schematics of the self-adhesive, elastomeric sensors with fine channels etched on the adhesive face along with a sensor being tested in a lap joint panel. When the sensors are adhered to the structure under test, the fine channels and the structure itself form a manifold of galleries alternately at low vacuum and atmospheric pressure. Vacuum monitoring is applied to small galleries that are placed adjacent to the set of galleries maintained at atmospheric pressure. If a flaw is not present, the low vacuum remains stable at the base value. If a flaw develops, air will flow from the atmospheric galleries through the flaw to the vacuum galleries. Figure 2-8 and Figure 2-9 show top-view and side-view schematics of the self-adhesive, elastomeric sensors with fine channels on the adhesive face along with a CVM sensor being tested in on an aircraft panel. The graphics show results from this crack detection monitoring and the pressure response used to indicate the presence of a crack. It is important to note that the sensor detects surface breaking cracks once they interact with the vacuum galleries. When a crack develops, it forms a leakage path between the atmospheric and vacuum galleries, producing a measurable change in the vacuum level. This change is detected by the CVM monitoring system (PM-200 device) shown in Figure 2-9. Figure 2-8 also shows a photo of a fatigue crack as it engages the first vacuum gallery of a CVM sensor. A pressure rise, corresponding to a rupture in the gallery and a leakage path to atmospheric pressure, occurs at this same time. The large increase in the pressure corresponds to crack detection as shown in the Figure 2-8 plot. One signal (blue curve) corresponds to vacuum levels produced when there is no crack indication and the other signal (red curve) occurs when a vacuum is not achievable. This latter signal is produced when the CVM detects a crack.

These sensors can be attached to aircraft structure in areas where crack growth is known to occur. On an OEM-established engineering interval, a reading will be taken from an easily accessible point on the aircraft. Each time a reading is taken, the system performs a self-test. This inherent fail-safe property ensures that the sensor is attached to the structure and working properly. Since the sensor physics is based on pressure measurements, there is no electrical excitation involved. This can be important in areas where electrical signals can create interference (near avionics) or where electrical connections may pose a hazard (fuel tanks). Each time a reading is taken, the system performs a self-test to ensure: 1) there is no blockage in the galleries which would affect and subsequent vacuum measurements and 2) proper adherence of the sensor to the surface it is monitoring. This initial check provides and inherent fail-safe property that ensures the sensor is attached to the structure and working properly prior to any data acquisition.

Drivers for Application of CVM Technology

- Overcome accessibility problems; sensors ducted to convenient access point
- Improve crack detection (easier & more often)

- Real-time information or more frequent, remote interrogation
- Initial focus – monitor known fatigue prone areas
- Long term possibilities – distributed systems; remotely monitored sensors allow for condition-based maintenance

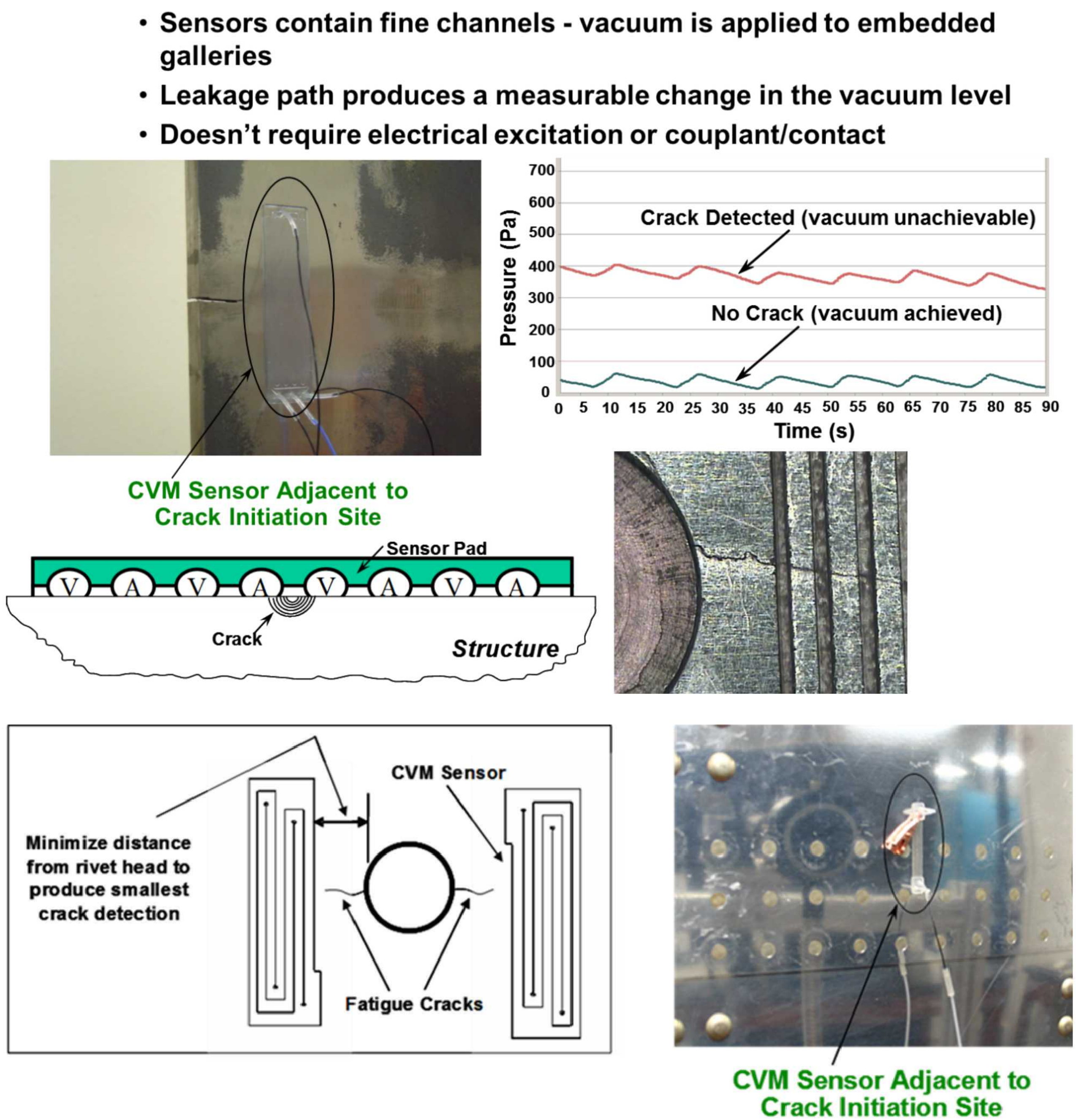

Figure 2-8: Schematic Depicting Operation of CVM Sensor and Polymer Sensor Mounted on Surface of an Aircraft Panel

Figure 2-9: Comparative Vacuum Monitoring System

Historical Perspective - The Federal Aviation Administration's Airworthiness Assurance Center at Sandia Labs, in conjunction with industry and airline partners, completed the first series of validation tests on the CVM system in the 2000 to 2004 timeframe in an effort to adopt Comparative Vacuum Monitoring as a standard NDI practice [2.8 - 2.10]. Subsequent Testing was conducted at Sandia Labs, in concert with Embraer and the Agencia Nacional de Aviação Civil (ANAC) regulatory agency in Brazil, to complete validation testing of CVM sensors for a variety of potential applications on Embraer aircraft [2.11-2.12]. In all programs, fatigue tests were completed on simulated aircraft panels to grow cracks in riveted specimens (see Figure 2-10) while the vacuum pressures within the various sensor galleries were simultaneously recorded. A fatigue crack was propagated until it engaged one of the vacuum galleries such that crack detection was achieved and the sensor indicated the presence of a crack by its inability to maintain a vacuum. In order to properly consider the effects of crack closure in an unloaded condition (i.e. during sensor monitoring), a crack was deemed to be detected when a permanent alarm was produced and the CVM sensor did not maintain a vacuum even if the fatigue stress was reduced to zero. This prior test program produced a statistically-relevant set of crack detection levels for 0.040" to 0.100" (1.02 mm to 2.54 mm) thick panels in both the bare and primed configurations. The results from these validations tests are described in Section 4.3.

Figure 2-11 and Figure 2-12 summarize another proof-of-concept program which was driven by an actual inspection need on a commuter aircraft which involved a structure that was difficult to access. The program involved Bombardier and Transport Canada and proved the viability of the CMV system for monitoring the main engine beam in the empennage region of the CRJ aircraft platform. The initial goal of this project was to provide Bombardier and regulatory agencies with sufficient data to certify CVM sensor technology for specific aircraft applications. Probability of flaw detection assessments were coupled with on-aircraft flight tests to study the performance, deployment, and long-term operation of CVM sensors on aircraft. From a maintenance planning perspective, the objective was to eliminate access difficulties associated with this inspection and to provide an early indication of a flaw onset to properly schedule maintenance tasks. The derived benefit was a reduction in the rate of aircraft grounded after an inspection by allowing repairs to be scheduled in advance. By using CVM measurements as an alternate method of inspection (meet the inspection requirements of a Principal Structural Element), the goal was to: 1) reduce maintenance costs associated with the inspection tasks, and 2) increase threshold and repeat intervals for Fatigue Driven PSEs. Figure 2-12 shows the CVM sensor design and placement and also highlights the crack detected on an operating aircraft. The lower left image is a photo of a dye penetrant inspection showing the crack engaging the CVM galleries

Additional programs conducted to produce approval for CVM sensor use on rotorcraft and fixed wing aircraft are depicted in Figure 2-13and Figure 2-14. Additional information on these and other CVM applications will be discussed in this report.

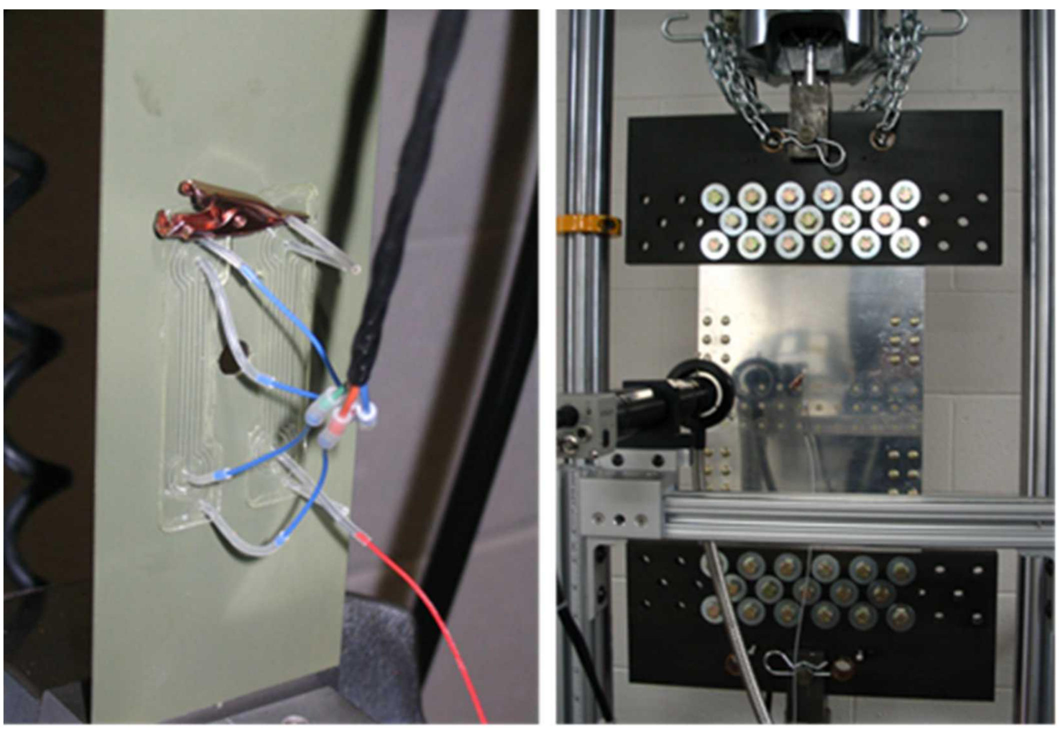

Figure 2-10: CVM Sensors Monitoring Crack Growth on Aluminum Test Specimens

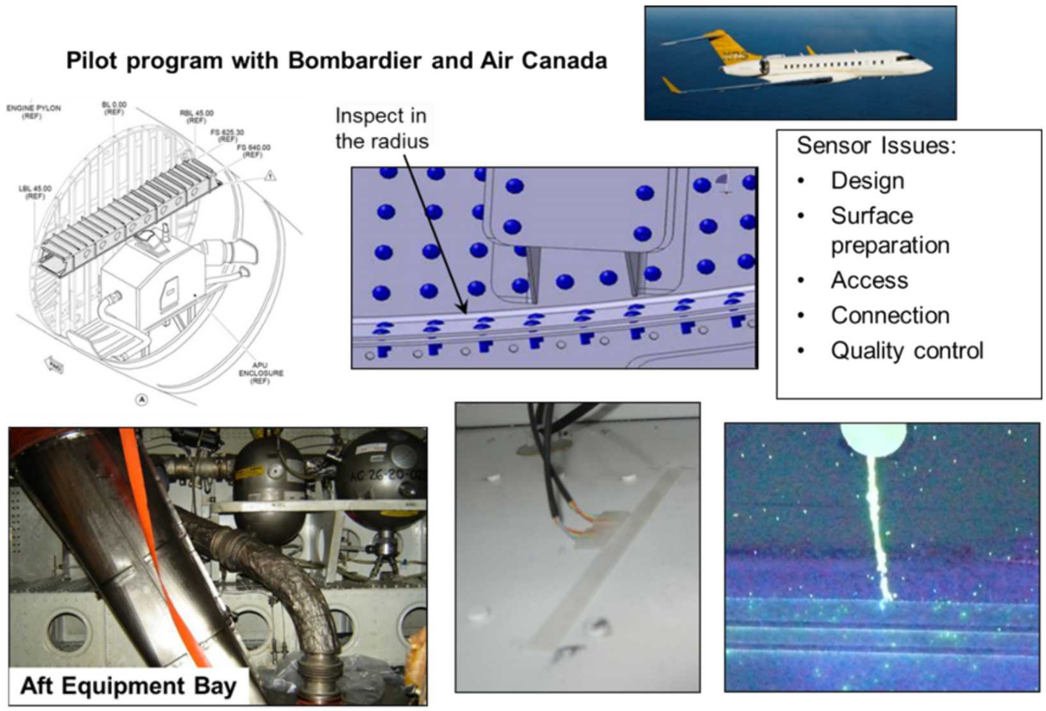

Figure 2-11: Sample Program that Produced a Successful Crack Detection by CVM Sensor on an Operating CRJ Aircraft

LHS SLS ID#1	Conductivity Control Gallery	Conductivity Gallery 1	Conductivity Gallery 2	CVM Gallery 1	CVM Gallery 2	Gallery 1 Status	Gallery 2 Status	Conventional NDT Result
Date	CI	CI	CI	CI	CI			
25-02-09	9300	10686	9930	2419	0.7	IF	NC	NC
26-06-09	6947	7339	7270	4276	1.1	NA	NC	NC

RHS SLS ID#2	Conductivity Control Gallery	Conductivity Gallery 1	Conductivity Gallery 2	CVM Gallery 1	CVM Gallery 2	Gallery 1 Status	Gallery 2 Status	Conventional NDT Result
Date	CI	CI	CI	CI	CI			
25-02-09	10192	10827	10692	0.9	1.9	NC	NC	NC
26-06-09	6702	7807	6930	2947*	12.3	C	C	C

Crack Indication**	2
No Crack Indication	4
True Positive Or False Positive	2/2 (100%)
True Negative	2/2 (100%)
False Negative	0/2 (0%)

** All galleries are considered independent

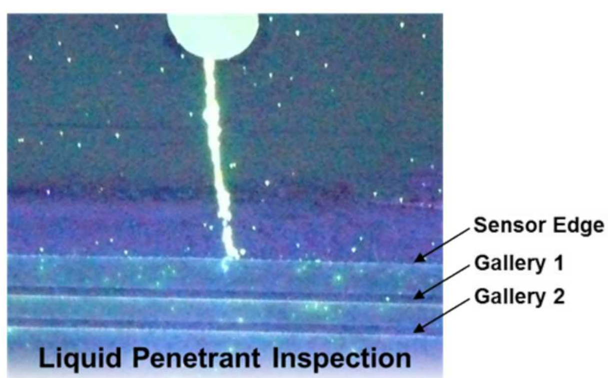

Figure 2-12: Flight Test Results from CRJ Aircraft Showing Crack Detection by CVM Sensor

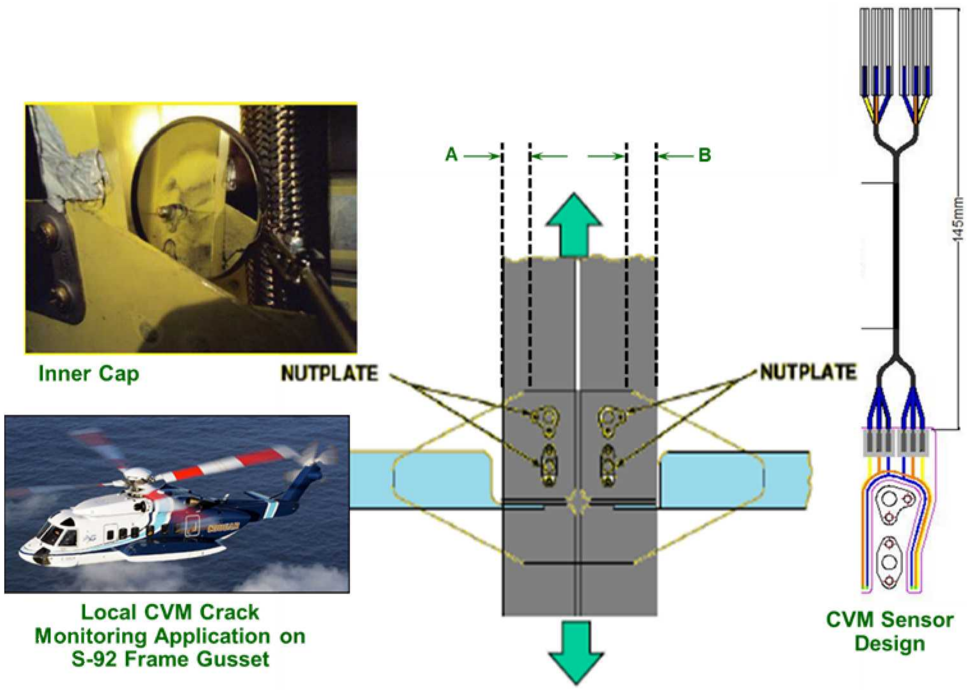

Figure 2-13: Sample Rotorcraft Application Deploying CVM System to Monitor Cracks on Rotorcraft Component

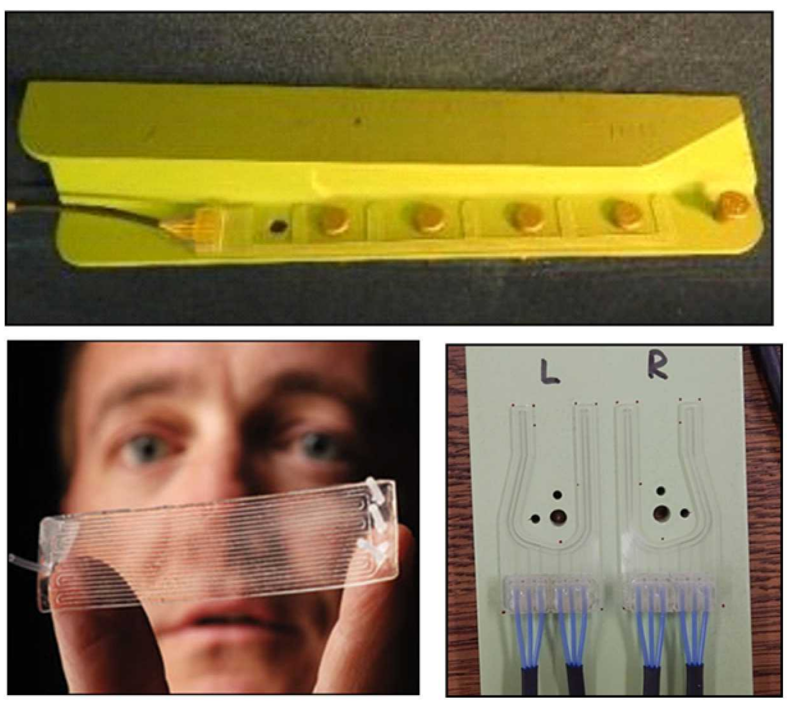

Figure 2-14: Sample Custom CVM Sensor Designs and Installations on Aircraft (Wing Box Fitting) and Rotorcraft (Frame Gusset) Applications

Multi-CVM Switch-Based System for Remote Bridge Monitoring – As mentioned above, SHM systems can be used to monitor a wide variety of structures that may benefit from periodic, remote inspections. A real-time monitoring system was developed for remotely interrogating a distributed array of CVM sensors. It used a series of pressure switches that can continuously monitor structures remotely via a wireless transmitting device. Sensors were placed in known fatigue critical locations on the bridge structure shown in Figure 2-15. If a crack breaches a CVM sensor, the pressure switch would be opened and, in turn, a message would automatically be sent to a maintenance center and any cell phone that was programmed into the firmware.

Up to 50 switches can be powered by one vacuum pump. The CVM monitoring system, shown in Figure 2-16, was mounted at a central point on the bridge structure. Multiple sensors were arranged to monitor the growth of any crack. In this design, a known crack can be monitored for a particular length when a sensor placed ahead of the crack is triggered when the crack grows. In this bridge application, known, critical locations at welded joints required periodic monitoring and their location over 100 feet from the road surface made manual on-site inspections impractical. The installed CVM monitoring system could continuously update web sites or send automated text messages or e-mails so that operators can quickly and remotely ascertain the condition of the bridge structure and determine if maintenance action is required.

Figure 2-15: Placement of CVM Sensor Network for Monitoring Critical Bridge Welds

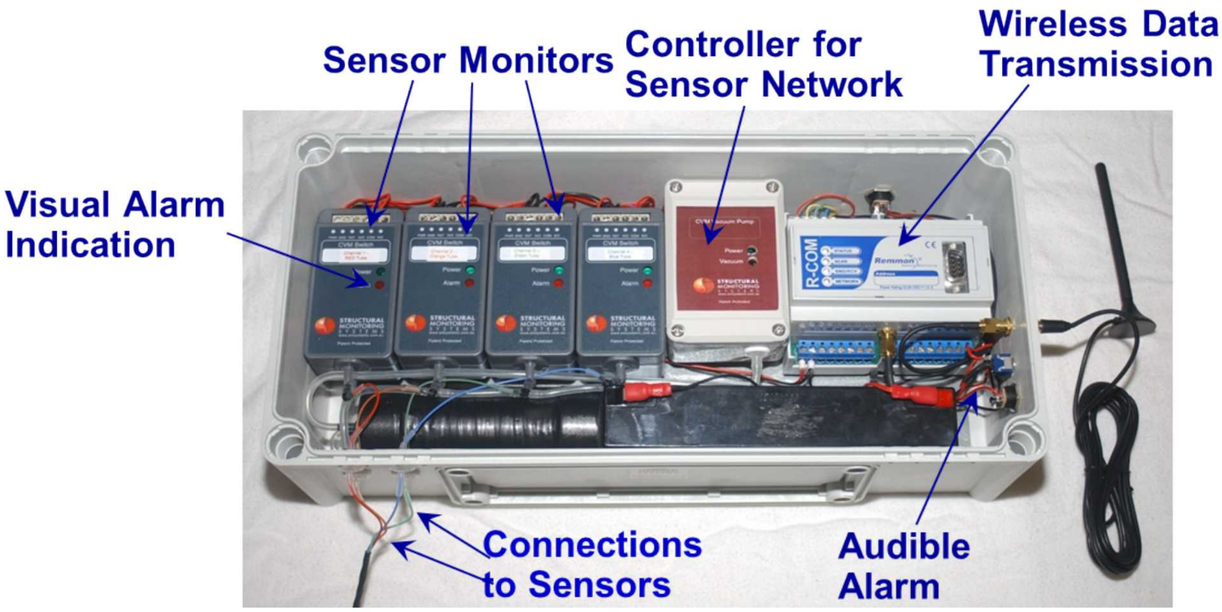

Figure 2-16: Real-Time, Remote Monitoring System for a Network of CVM Sensors

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CHAPTER 3

3.0 B737 Wing Box Fitting Application

This SHM certification and integration activity was a joint effort that leveraged existing airline maintenance programs and involved Delta Air Lines, Boeing, the FAA and the Sandia Labs AANC. It included the following activities:

- Certification/usage effort intended to investigate, exercise and evolve the SHM certification path – address all “cradle-to-grave” issues for airlines, OEMs, and regulators
- Identification of SHM applications – assess positive cost-benefit analysis
- Customize SHM system to the selected application(s)
- Develop validation/certification plan – utilize precedents from existing sensors
- Complete SHM indoctrination and training for Delta personnel (engineering, maintenance, NDI) and FAA as needed
- Hardware specifications, installation procedures, operation processes, continued airworthiness instructions
- Complete modifications to Delta maintenance program as a result of SHM use
- Assessment of aircraft maintenance depots’ ability to adopt SHM and the FAA support needed to ensure airworthiness.
- Formal paperwork and sign-offs associated with OEM approval for routine use of CVM technology for a specific application.

The selection of the application was done carefully and purposefully. Since the regulatory guidance was to-be-determined, the application could not be a safety critical area. The safety-critical or ‘hot-spot’ areas, often mandated by Airworthiness Directives, have become the current focus of SHM deployment. The bulk of the ~20 proposed applications fell into this category. So, the B737 Wing Shear Fittings were selected since it was determined by Boeing to be an ‘economic’ application and not a safety-critical area. Since the damage detection process required a time-consuming inspection and logistically complex repair, Boeing suggested inspecting or replacing these fittings at Heavy Maintenance. This program served as an approval process to provide an alternative to the existing eddy current and visual inspections without requiring burdensome and time-consuming access.

Figure 3-1 shows sample aircraft locations and loading/damage drivers that drove the selection of the SHM application. The selection of the application required a unique, custom CVM sensor design. The shear fitting was known to crack and propagate between fasteners; therefore, the sensor was designed with ‘fingers’ to fit in between each of the fasteners. Additionally, the area is a high vibration area, however, confidence was high due to the previous, successful flight tests at Delta and Northwest that occurred between 2004 and 2011 (see Chapter 6) [3.1, 3.2]. Despite the complex geometry of the CVM sensors needed to monitor all required regions on the wing box fitting and the extreme operating environment, the installations were generally without any issues.

In summary, the background on the selection of an SHM application and solution:

- Boeing issued an inspection Service Bulletin for the 737 Wing Box fitting as a result of cracking experienced in the field after 21K cycles. Figure 3-2 through Figure 3-5 provide a thorough description of the Wing Box fittings, potential crack onset areas and the inspection requirements to be addressed by the CVM sensors.
- Team selected the Comparative Vacuum Monitoring (CVM) system to find cracks in known hot spots as it was able to address the crack detection needs.
- Through previous research in FAA programs, this sensor was successfully flown for over 6 years on several Delta and Northwest Airlines' aircraft (See Section 6.1).
- Demonstrated performance of CVM:
    - Completed testing provided environmental reliability data.
    - Completed testing established general Probability of Detection performance.
    - CVM had already been adopted into Boeing's NDT Standard Practices Manual as an approved "generic" method (tool) for crack detection (See Section 4.3, Figure 3-6 and Ref. [3.3, 3.4]).
- FAA Transport Aircraft Directorate was a participant and helped coordinate meetings to include both Seattle and Atlanta Aircraft Certification Offices (ACO).

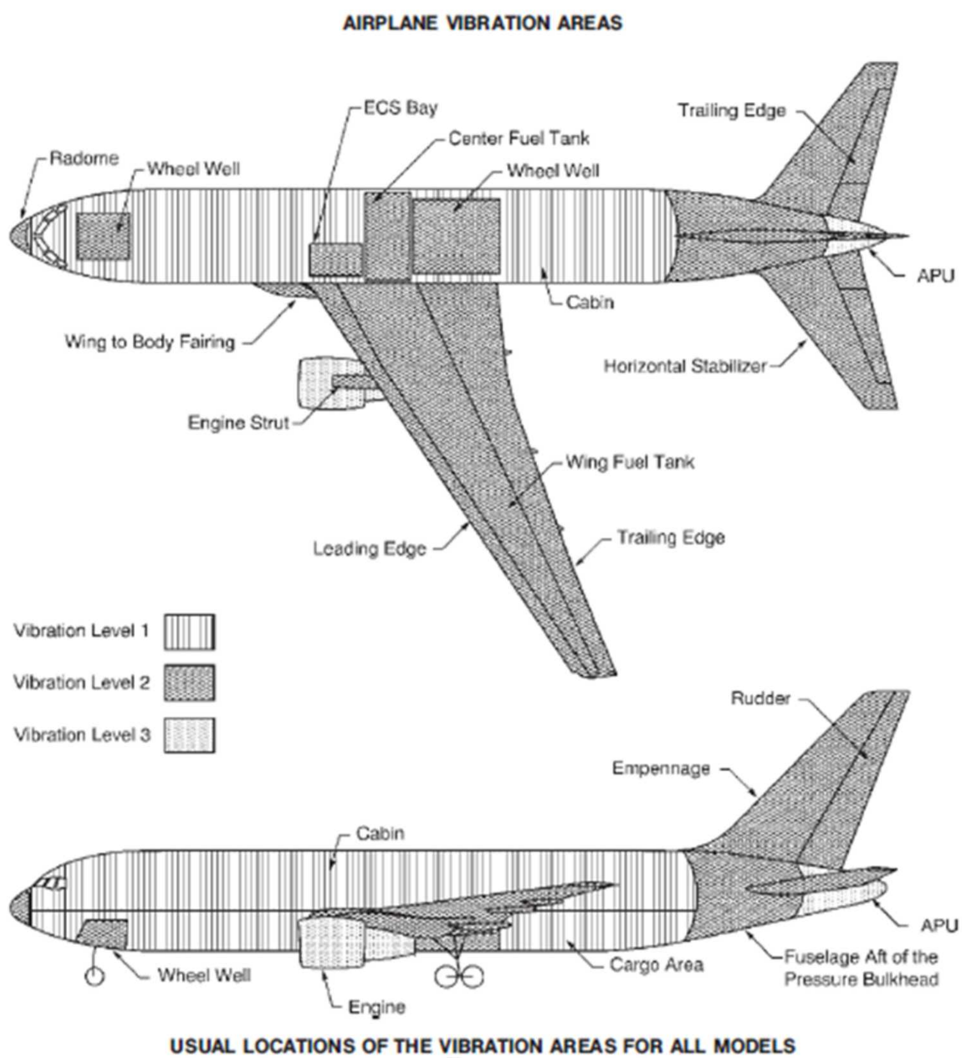


Figure 3-1: Sample Considerations in Delta Selection of SHM Application

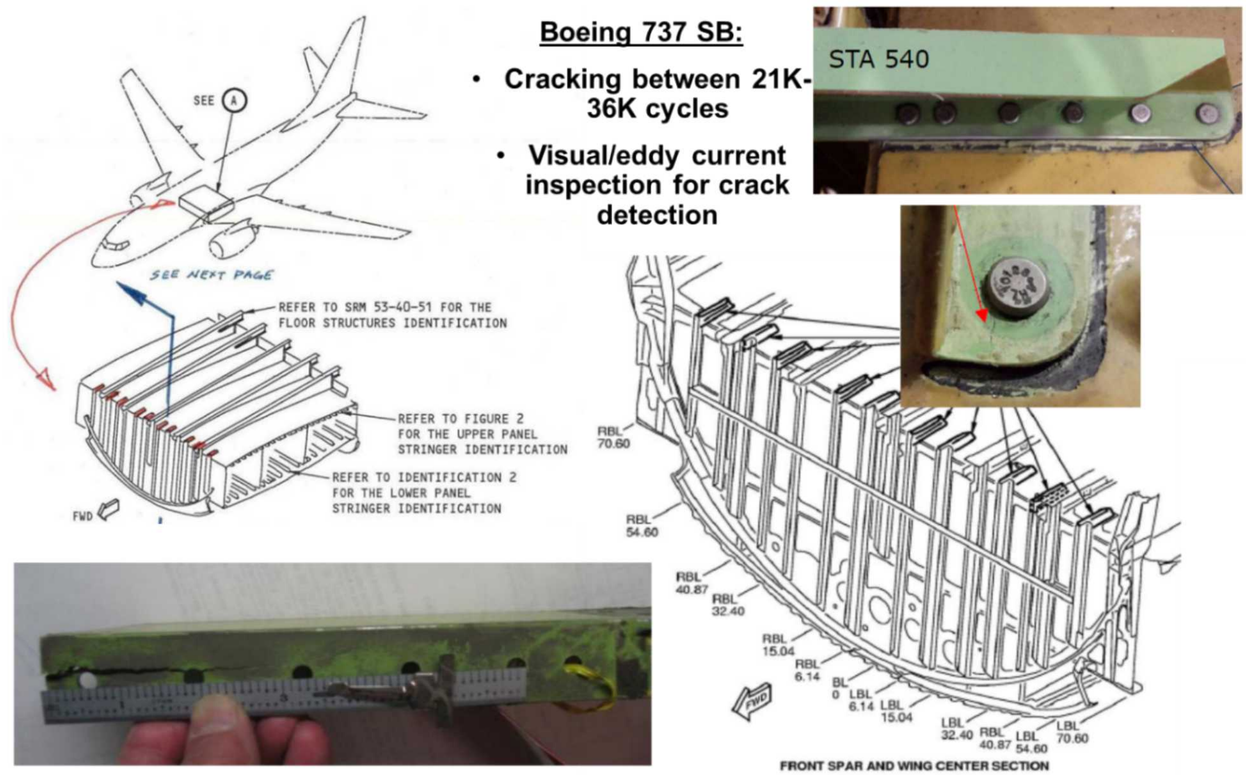


Figure 3-2: 737NG Center Wing Box, Front Spar Shear Fitting and Sample Cracks Experienced on this Structural Component

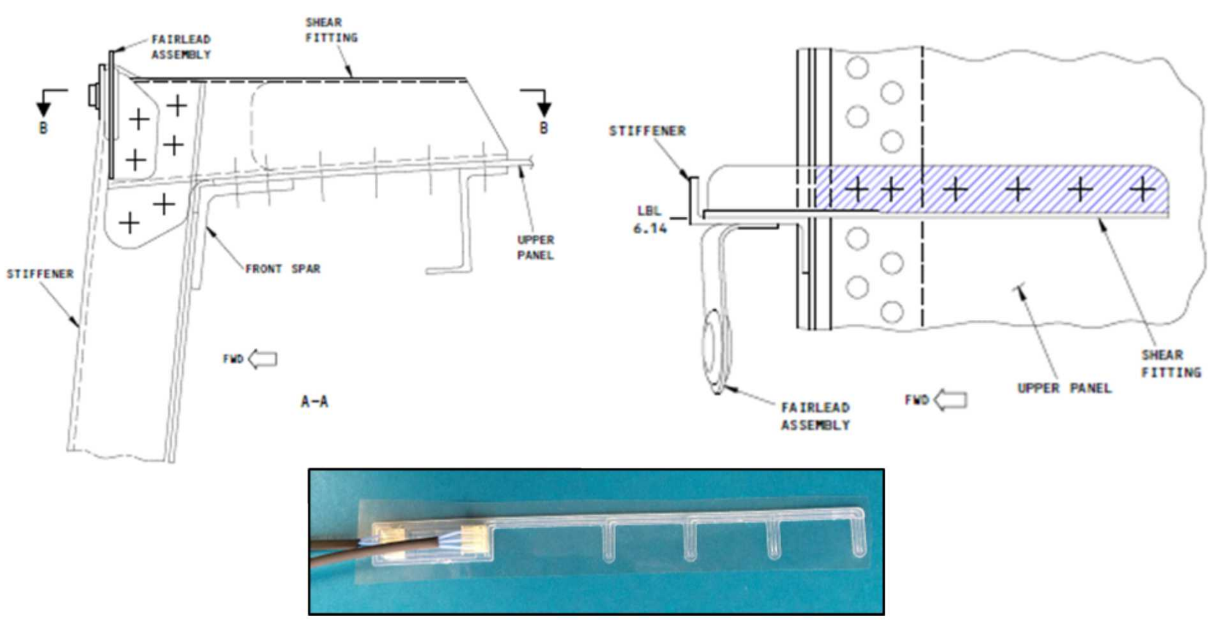


Figure 3-3: Side View of the 737 Center Wing Box Fitting with CVM Sensor Design Used to Monitor the Crack Regions

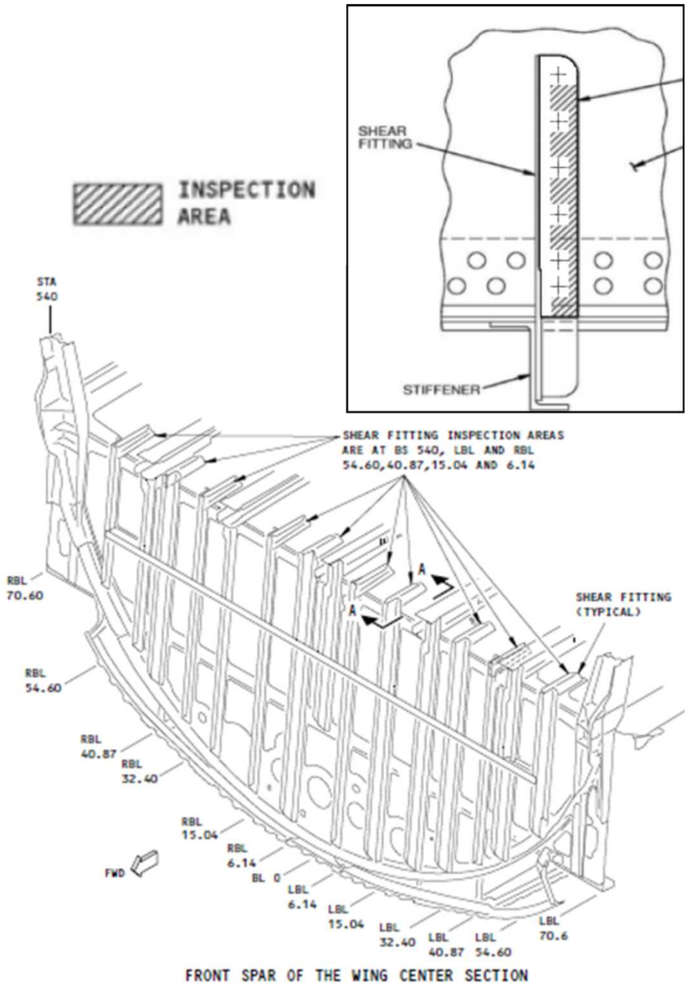


Figure 3-4: Set of Ten Fittings and Specific Inspection Regions

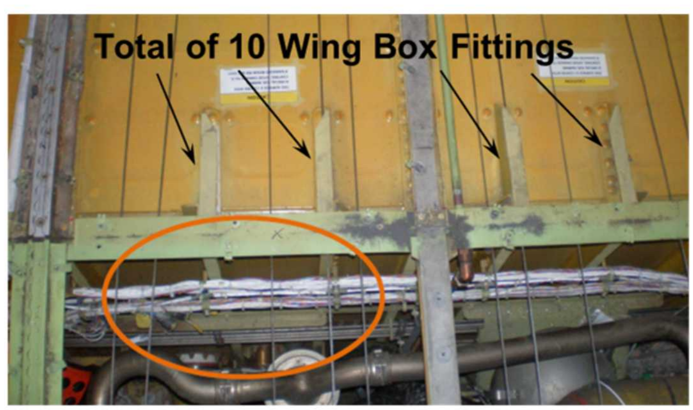


Figure 3-5: CVM Sensor on 737 Wing Box Fitting, View of Fittings Along the Forward Section of Wing Box and Top View of Location on Forward Spar to Mount the SLS Connectors for Data Acquisition

BOEING

NONDESTRUCTIVE TEST

PART 1 - GENERAL
COMPARATIVE VACUUM MONITORING (CVM) SYSTEM
ALUMINUM PART SURFACE INSPECTION

1. Purpose

   A. This procedure uses the Comparative Vacuum Monitoring (CVM) System to examine aluminum parts for surface cracks. The CVM system uses adhesive sensors that stay attached to the part to be examined.

   B. This procedure uses linear and curved-linear type sensors. It will be necessary to use a different procedure for other sensor types. For each use of CVM, a sensor or sensors must be specially made.

   C. This procedure can be used to examine internal aircraft structure that is free from the effects of external environmental conditions. The inspection surface can only have primer on it to use this procedure; this procedure cannot be used if the inspection surface has an enamel top coat.

   D. Use this procedure to find surface cracks in aluminum parts that are 0.09 inch (2.29 mm) thick or less.

2. Equipment

   A. General

      (1) It is necessary to use a periodic monitoring inspection instrument that can be calibrated to measure vacuum differential.

   B. Instrument

      (1) Use a CVM instrument that:

         (a) Has applicable firmware version installed

         (b) Operates with an impedance of 2 T PASM (Terra Pa.second/meter^2)

         (c) Has the recommended equipment for the inspection

         (d) Has the CVM instrument Operations Manual

         (e) Has the correct Test Point Plug (TPP)

      (2) The instrument that follows was used to help prepare this procedure.

         (a) PM200; Structural Monitoring Systems (see Fig. 1)

CMN NDT
Nov 15/08

Part 1
51-11-01
Page 1

Figure 3-6: General CVM Procedure in NDT Manual

The final CVM sensor designed to meet the crack detection needs at each of the fasteners in the Wing Box fittings is shown in Figure 3-7. The Boeing Service Bulletin number 737-57-1309 requiring periodic inspections of the Wing Box fittings is introduced in Figure 3-8.

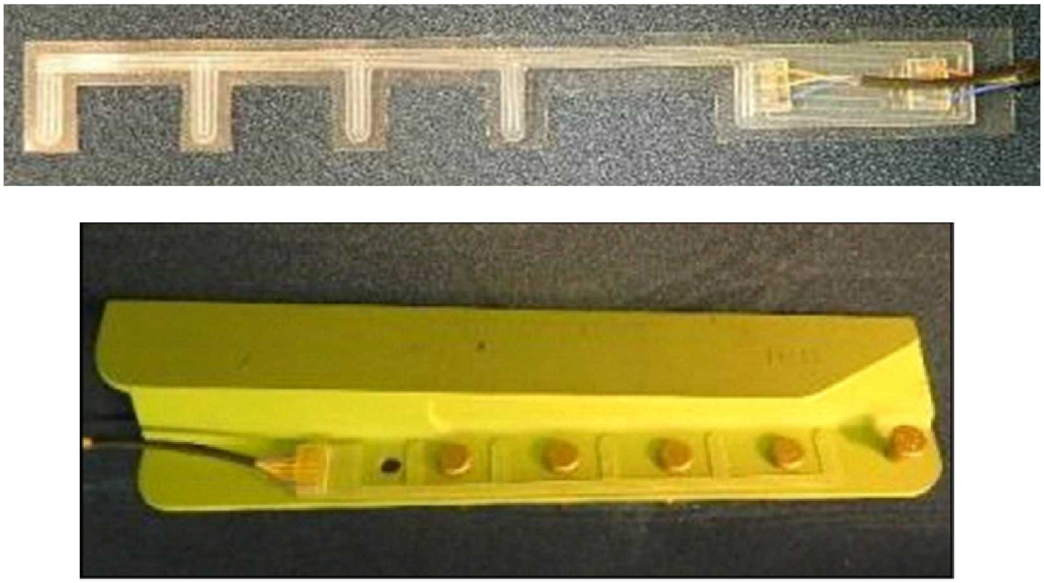
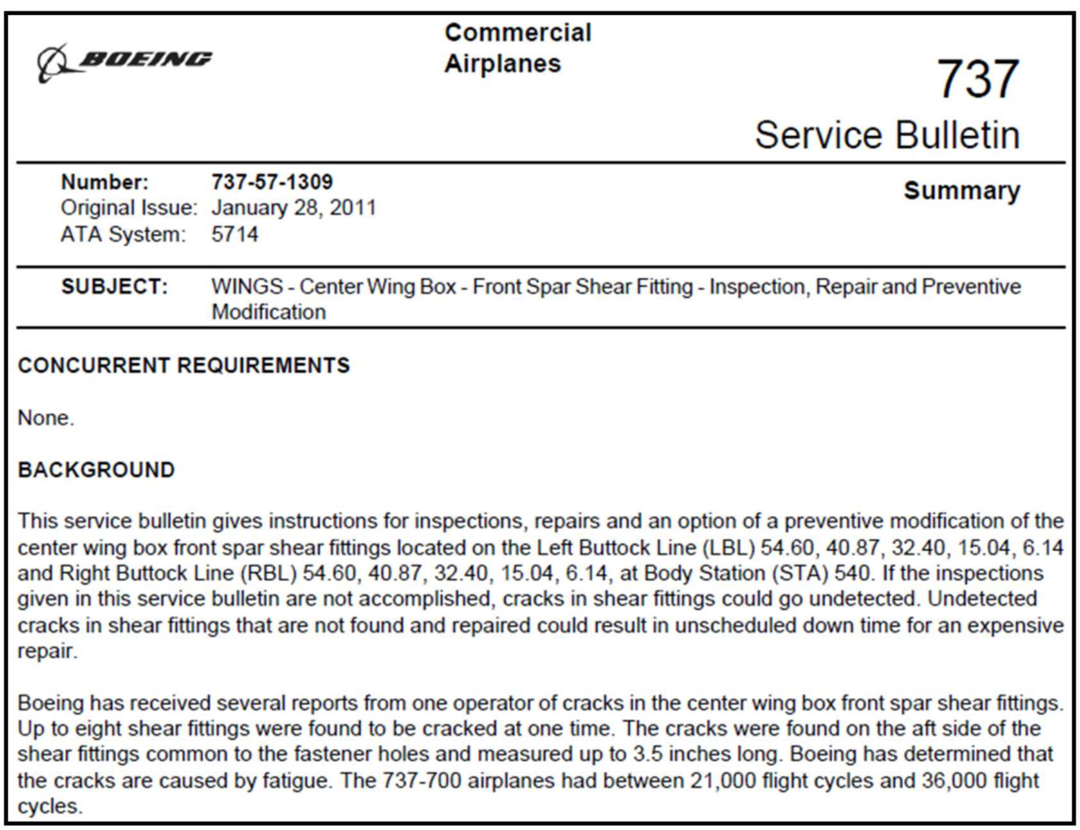


Figure 3-7: CVM Sensor Mounted on Wing Box Fitting to Monitor All Inspection Regions



BOEING

Commercial
Airplanes

737

Service Bulletin

Number: **737-57-1309**
Original Issue: January 28, 2011
ATA System: 5714

Summary

SUBJECT: WINGS - Center Wing Box - Front Spar Shear Fitting - Inspection, Repair and Preventive Modification

CONCURRENT REQUIREMENTS

None.

BACKGROUND

This service bulletin gives instructions for inspections, repairs and an option of a preventive modification of the center wing box front spar shear fittings located on the Left Buttock Line (LBL) 54.60, 40.87, 32.40, 15.04, 6.14 and Right Buttock Line (RBL) 54.60, 40.87, 32.40, 15.04, 6.14, at Body Station (STA) 540. If the inspections given in this service bulletin are not accomplished, cracks in shear fittings could go undetected. Undetected cracks in shear fittings that are not found and repaired could result in unscheduled down time for an expensive repair.

Boeing has received several reports from one operator of cracks in the center wing box front spar shear fittings. Up to eight shear fittings were found to be cracked at one time. The cracks were found on the aft side of the shear fittings common to the fastener holes and measured up to 3.5 inches long. Boeing has determined that the cracks are caused by fatigue. The 737-700 airplanes had between 21,000 flight cycles and 36,000 flight cycles.

Figure 3-8: Boeing Service Bulletin Addressing Center Wing Box Fittings

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CHAPTER 4

4.0 CVM Crack Detection Performance Assessment

4.1 Performance Assessment Methodology (POD)

Considerations for an SHM POD Study - Some portions of the normal POD methodology needed to quantify NDI performance can be adapted to the validation of SHM systems. However, it is important to recognize the unique validation and verification tasks that arise from distinct differences between SHM and NDI deployment and damage detection. SHM reliability calculations will depend greatly on the complexity of the structure and geometry of the damage profile. For example, corrosion damage has a widely-varying damage shape, both in the surface dimensions and in the changing depth. Contrast this with a fatigue crack that grows in a known propagation path such that the damage scenario can be described in a single parameter: crack length. In this latter case, the simplicity of such a one-dimensional entity allows for a more direct calculation of the reliability of the SHM system detecting such damage. Statistical performance assessments of damage detection sensors that are permanently mounted in a fixed position must be handled differently than similar studies using hand-held or other deployed NDI transducers that are moved along the structure being inspected. In the case of in-situ SHM sensors, the damage of interest originates, and may even propagate, into the region being monitored by the SHM sensor. Performance analyses then considers the response of the sensor or damage detection and correlates this response with the size of the damage when detected. For example, a crack in the material beneath or in the vicinity of an SHM sensor will allow for detection. The POD data could then consist of fatigue cracks that were propagated in various metal specimens with the direction of growth aligned with the mounted sensors.

Because of physical, time or cost constraints, it is often impractical to inspect an entire population. Instead, a small sample of the total population is tested and the data is used to gauge how well the entire population conforms to specifications. In traditional statistical process control, a significant number of data points are required in order to get a reasonably accurate estimate of process capability. This is because capability is usually calculated to cover a fixed multiple standard deviations. But this percentage only holds true for larger sample sizes; that is, greater than 50. As the sample size decreases, there is greater uncertainty in knowing the true location of the mean and the true magnitude of the population variance. Therefore, the estimate of the range of values encompassing a given percentage of the population must necessarily increase to compensate. In order to maintain a reasonably accurate estimate of the capability of a process for smaller sample sizes, it is necessary to adjust the number of multiple sample standard deviations used to define the region covering the desired proportion of the population distribution with a given confidence.

An SHM POD experiment for aerospace applications will generally consist of fixed sensors being placed on specimens (e.g., flat plates) with starter-cracks (e.g., EDM notches). Then the cracks could be grown in fatigue over time by applying cyclic mechanical loads to the plate. SHM signal data would be taken periodically over time and related to the length of the crack at that time. Similarly, for pipeline applications, fixed sensors could be used to monitor a corrosion process where SHM signals would be related to the amount of metal loss.

The SHM-POD experiment should accurately simulate the actual SHM process. Again, it is important that the experiment capture relevant sources of variability. For example, the variability in cracks grown in the experiment should accurately represent the variability seen in actual cracks. The observational units in SHM POD studies will be crack/sensor combinations (where there may be an array of sensors in some applications).

In any SHM application, there will be an important consideration of how to map the SHM signal(s) into a detect/no-detect decision at each inspection opportunity (however inspection opportunity is defined), typically referred to as a damage index. The POD will then depend on the (joint) probability distribution of the inputs to that decision-making mechanism. Data from one or more SHM sensors may be mapped into one or more scalar damage indices that can be used for decision making, however, each damage index would produce separate individualized POD results. In this chapter, we will assume that the decision-making response is a scalar and that the crack length, which is known, adequately describes crack properties (i.e., truth data).

Factors Affecting Detection Sensitivity and Sources of Variability in SHM - To properly quantify POD from a POD study, it is essential that all important sources of variability that could affect detection are explicitly captured. Omitting influential sources of variability in a POD study could result in overestimating the probability of detecting smaller cracks or underestimating the potential for false alarm indications.

Factors relating to damage and system properties that could affect SHM signals include:

1. Damage size, shape, and orientation (including changes in these characteristics over time). We note that this is typically the dominant source of variability in traditional NDE and it is expected that this will be true also for SHM applications.
2. Damage location relative to sensor location (including the distance between the sensor and the damage).
3. Environmental variables such as temperature and humidity.
4. Mechanical variables such as strain conditions (due to variable fuel loading, etc.).
5. Variability in sensor signal responses due to sensor-to-sensor manufacturing variability.
6. Change in the structural configuration where the sensors are located as a function of time and that could have an effect on SHM signal.
7. Changes in sensor performance over time due to maintenance repair, re-painting, etc.
8. Sensor aging and degradation.
9. Sensor, adhesive and other characteristics relating to installation-to-installation variability.

Then, for those factors that are not assumed to be held constant across inspections or compensated for by a calibration operation, it is essential that there be an accurate characterization of the joint probability distribution. For example, in traditional NDE, to crack-to-crack variability arising from differences in crack morphology (different cracks with nominally the same size can have signal responses that vary enormously) tends to be the dominant source of variability and this is also expected to also be the case in SHM applications.

Many factors are involved in obtaining viable SHM data for POD calculations. Some of these factors depend on the SHM system itself and some depend on the type of testing, the complexity of the test article used in the assessment and the type, location, and orientation of damage being detected. Factors to be considered include, but are not limited to, determining the boundaries for the SHM system applications, producing validation tests that are representative of the actual structure, establishing proper damage detection thresholds, utilizing data with appropriate signal content compared to system noise, and data analyses methods.

Parameters to be Considered for Effect on Crack Detection - Statistical performance assessments of flaw detection sensors that are permanently mounted in a fixed position must be handled differently than similar studies using hand-held or other deployed NDI transducers that are moved along the structure being inspected. In the case of in-situ SHM sensors, the flaw of interest originates, and may even propagate, into the region being monitored by the SHM sensor. Performance analyses then considers the response of the sensor or flaw detection and correlates this response with the size of the flaw when detected.

It is important to recognize the unique validation and verification tasks that arise from distinct differences between SHM and NDI deployment and flaw detection. SHM reliability calculations will depend greatly on the complexity of the structure and geometry of the flaw profile. For example, corrosion damage has a widely-varying flaw shape, both in the surface dimensions and in the changing depth. Contrast this with a fatigue crack that grows in a known propagation path such that the damage scenario can be described in a single parameter: crack length. In this latter case, the simplicity of such a one-dimensional entity allows for a more direct calculation of the reliability of the SHM system detecting such damage.

Note: crack damage is usually repeatable, and many variables will not play a role in detection, depending on the sensor system. New variables come into play on a case-by-case basis. They must be properly controlled and uncoupled for proper performance assessments.

- SHM system side – 1) design and position of sensors relative to damage, 2) density/layout of network is applicable, 3) data analysis methods, 4) repeatability of sensor fabrication & associated response, 5) repeatability of sensor placement (assume conservative variations & assess), 6) repeatability of the sensor readout device (DAQ), 7) effects of environment (temperature, vibration, stress, chemicals) on sensor/hardware response, 8) selection of DI threshold for assigning detection (permanent, unloaded condition), 9) spatial resolution to properly capture changes associated with damage onset/growth, 10) statistics needed for sufficient data.
- Structural response side – 1) complexity of structure (layers, gaps, bushings, adjacent fasteners, hole size, nearby repairs), 2) damage onset mode & loads that generate the damage, 3) residual stress levels (crack closure), 4) stress reapportion with changing flaw profile, 5) repeatability of crack response/morphology (variations in the defect), 6) damage orientation, 7) presence of chemical by-products (e.g. aluminum oxide from corrosion), 8) presence of coatings, 8) simultaneous/multiple damage sites which could make it difficult or impossible to uncouple the SHM response for each individual damage occurrence (main affect is on testing which should include singular damage sites), 9) geometry of the monitored region (could produce signal reflections).

One-Sided Tolerance Interval (OSTI): Use of Confidence Bounds to Calculate Specific POD Values – The Length at Detection (LaD) method for repeated inspections of cracks growing under or near fixed sensors provides a simple, statistically-valid method to compute POD for SHM applications [4-1]. This method, summarized in Figure 4-1, was originally suggested in Reference [4-2] and first applied to POD assessments of SHM systems in References [4-3 – 4.4]. This method uses only the crack-length values when cracks are first detected. Similar to other POD applications, the underlying statistical model is that there is a population of crack/sensor combinations and that the POD study is based on a sample of these crack/sensor combinations. Each crack has a length, random from crack to crack, at which the crack will be detected. Because only one observation is taken from each crack/sensor combination, the issue of dealing with the dependency of repeated measures data does not arise.

Because of the close relationship between confidence intervals for probability distribution quantiles and tail probabilities, the computation of the lower confidence bounds for POD in the LaD method, or upper bounds on the crack length associated with that POD, can also be done by using statistical methods for computing a one-sided tolerance bound, as described in Reference [4.4]. The computation of a one-sided 100(1-α)% tolerance bound to exceed at least 100p% of a normal population corresponds to the computation of a one-sided confidence bound for the 100pth percentile of the normal distribution. The one-sided tolerance bound is equal to the LaD value associated with the lower confidence limit of the POD curve at the 100pth percentile of interest. With these assumptions, there exists a distribution on the flaw lengths at which detection is first made. In this context, the probability of detection for a given flaw length is just the proportion of the flaws that have a detectable length less than that given length. That is, the reliability analysis becomes one of characterizing the distribution of damage lengths and the cumulative distribution function is analogous to a Probability of Detection (POD) curve.

In previous applications of this tolerance bound calculation, it has been termed a "One-Sided Tolerance Interval" (OSTI) because it estimates the upper bound, from the LaD distribution, which should contain 100p% of all the measurements in that LaD distribution with 100(1-α)% confidence. It should be noted that this approach evaluates the lower confidence limit of the POD curve at the single percentile value of interest. Since it is based on a sample of the entire population (*n* data points), the confidence is less than 100%. The tolerance bound calculation from a OSTI estimates the upper detection bound which should contain a certain percentage of all measurements in the population with a specified confidence, as described in Reference [4.5]. More specifically, the $a_{90/95}$ point (95% upper confidence bound on the crack size that will be detected with probability 0.90) can be obtained as an upper confidence bound on the 0.90 quantile of the LaD distribution and this is equivalent to a one-sided upper tolerance bound on the same distribution. Methods for computing this confidence (or tolerance) bound are given in Section 4.4 of Meeker, Hahn, and Escobar [4.6].

The Probability of Detection for a fixed sensor detecting a crack which is propagating in a known direction in the vicinity of the sensor can be determined using the One-Sided Tolerance Interval (OSTI) approach. The OSTI estimates the upper bound which should contain a certain percentage of all measurements in the population with a specified confidence. Since it is based on a sample of the entire population (n data points), the confidence is less than 100%. Thus, the OSTI is greatly affected by two proportions: 1) the percent coverage which is the percent of the population that falls within the specified range (normally chosen as 90%), and 2) the degree of confidence desired (normally chosen as 95%). A demonstration of this OSTI calculation specifically for SHM system response is provided in References [4.7, 4.8].

- **Interval to cover a specified proportion of a population distributed with a given confidence – related to measures of process capability**
- **One-sided Tolerance Interval – estimates the upper bound which should contain a certain percentage of all measurements in the population with a specified confidence**
- **Since it is based on a sample of the entire population (n data points), confidence is less than 100%. Thus, it includes two proportions:**
  - **Percent coverage (90%)**
  - **Degree of confidence (95%)**
- **The reliability analysis becomes one of characterizing the distribution of flaw lengths and the cumulative distribution function is analogous to a Probability of Detection (POD) curve:**

  $$TI = X \pm (K_{n,\gamma,\alpha})(S) \qquad \text{[log scale calculation]}$$
- **Interested in a 1-tailed interval (utilize "+" in equation); upper limit of TI. Uncertainty in knowing the true mean and population variance requires that the estimate of the range of values encompassing a given percentage of the population must increase to compensate.**

Figure 4-1: Description of Confidence Bounds and Use of One-Sided Tolerance Interval to Determine POD for Sensor Systems in Fixed Locations

Assuming that the distribution of damage is such that the logarithm of the lengths has a Gaussian distribution, it is possible to calculate a one-sided tolerance bound for various percentile flaw sizes. To do this, it is necessary to find factors $K_{n,\gamma,\alpha}$ to determine the probability γ such that at least a proportion (1-α) of the distribution will be less than $X - K_{n,\gamma,\alpha}$ where X and S are estimators of the mean and the standard deviation computed from a random sample of size n. There may also be situations where the process capability is measured relative to a single-sided limit. These situations arise when a product characteristic need only meet a minimum specification limit or remain below a maximum specification limit. In this case, the desired POD value is the maximum crack length associated with the 90% POD level so an upper bound tolerance interval can be used. From the reliability analysis a cumulative distribution function is produced to provide the maximum likelihood estimation (POD). So, the tolerance interval, which represents the actual POD value for the damage of interest, can be derived from Equation 4.1:

$$T_{POD(90,\,95)} = X + (K_{n,\gamma,\alpha})(S) \tag{4.1}$$

T = Upper tolerance bound for crack length corresponding to 90% POD with a 95% confidence
X = Mean of detection lengths
K = Tolerance factor (~ function of sample size, detection level desired and confidence level desired)
S = Standard deviation of detection lengths
n = Sample size
1- α = Detection level
ɣ = Confidence level

Using Equation 4.1, it is possible to quantify the 90% POD level (e.g. crack length) for a sensor with a desired confidence level. The value for *T* is related to the number of samples tested and the range in detection levels observed. Thus, the performance is penalized – and the resulting POD increases - if the results are obtained with only a few samples and/or if there is a high degree of variability in the results. As the number of data points increases, the K value will decrease and the POD numbers could also decrease if the mean and standard deviation remain consistent. K can be calculated as follows:

$$K = t_{n-1,\gamma}\left(\sqrt{n}\,\Phi_{norm}^{-1}(\alpha)\right)/\sqrt{n} \tag{4.2}$$

Where,
t = non-central t-distribution with degrees of freedom n-1 and γ
Φ^{-1} = inverse CDF of a standard normal (gaussian) distribution
α = percent coverage or detection level

The data captured is that of the flaw length at the time for which the SHM sensor provided sustainable detection. R function normQuantileCI in R package StatInt is available to do the needed computation. Tables of the probability factor, K, needed to compute such tolerance bounds are also available in References [4.6, 4.9 – 4.10], and some engineering statistics textbooks. Corresponding estimation and confidence bound and confidence interval methods for the other location-scale and log-location-scale distributions are described and illustrated in Chapter 14 of Meeker, Hahn, and Escobar [4.6].

Data conditions necessary for a POD assessment using this approach are that the distribution of flaws is such that the logarithm of the lengths (strictly positive sizes) has a Gaussian distribution (log-normal distribution). The data should plot linearly on a semi-log scale (or the log values plot in a linear fashion on a linear scale) and the data should be clustered near the 50th percentile. Data conditions necessary for a POD assessment using this approach are that the distribution of flaws is such that the logarithm of the lengths (strictly positive sizes) has a Gaussian distribution (log-normal distribution). In order to ensure the validity of a log-normal, or Gaussian, distribution on the damage lengths, the data should plot linearly on a semi-log scale (or the log values plot in a linear fashion on a linear scale) and the data should be clustered near the 50th percentile. The assumption of normality can also be tested by applying the Anderson-Darling test. The Anderson-Darling test yields a P-value that can be compared to the chosen significance level to determine whether or not the assumption of normality should be rejected. The significance level, ψ, is chosen to be 0.05. Any value of P less than $\psi = 0.05$ indicates that there is sufficient evidence to reject the assumption of normality. An A-D calculation that determines a P value that is greater than 0.05 supports the assumption of a Gaussian data distribution. A normal probability plot can be created using statistical software such as Minitab®. Figure 4-2 shows two plots of sample SHM sensor crack detection data which indicates that a log-normal distribution is a correct assumption. In addition, the Anderson-Darling test returns the required value of $P > 0.05$. It shuld be noted that Kolmogorov-Smirnov or Cramer-Von Mises tests can also be used to check the normality assumption.

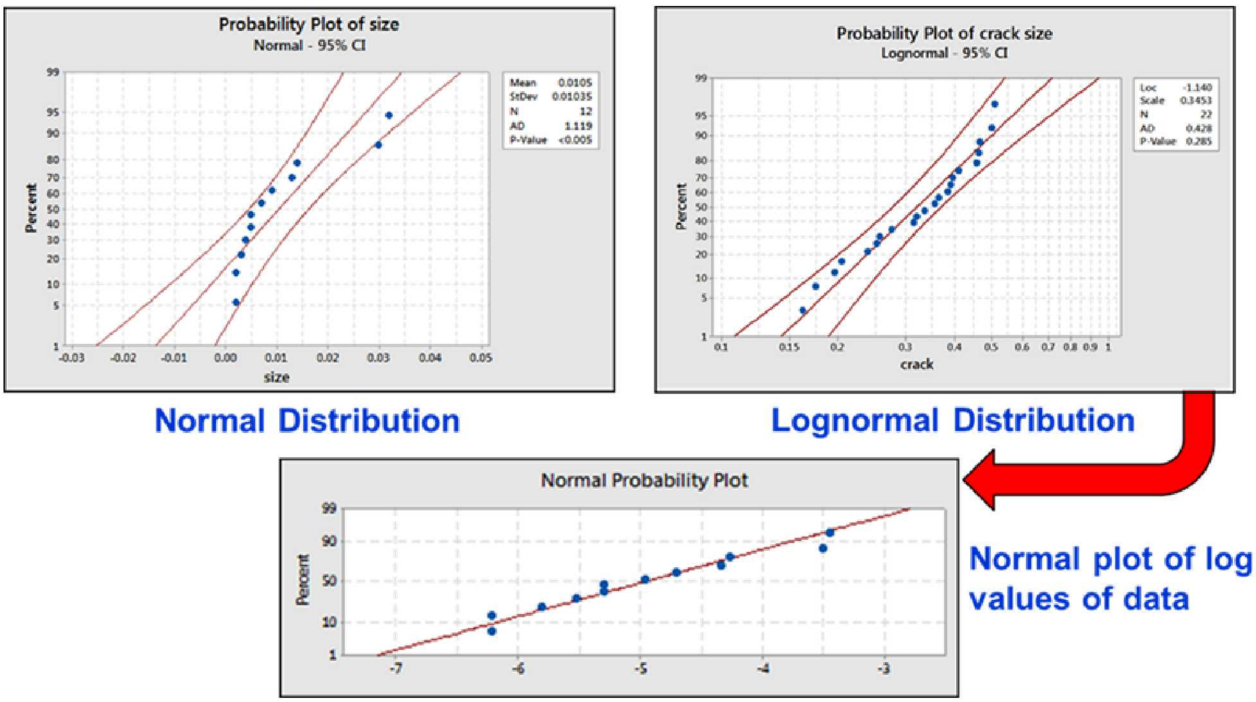


Figure 4-2: Sample Plot of SHM Data Indicating a Gaussian Distribution of Data

The discussion above shows how it is possible to calculate a one sided tolerance bound for various percentile flaw sizes - find factors $K_{n,\gamma,\alpha}$ to determine the confidence ɣ such that at least a proportion (α) of the distribution will be less than $X + (K_{n,\gamma,\alpha})S$ where X and S are estimators of the mean and the standard deviation computed from a random sample of size n. The reliability analysis becomes one of characterizing the distribution of flaw lengths and the cumulative distribution function is analogous to a Probability of Detection (POD) curve. A two-sided tolerance interval, used to indicate values at which certain compliance is met, is shown in Figure 4-3. In this case, the POD corresponds to a 1-tailed interval (utilize "+" equation 4.1) or the upper limit of tolerance interval. The uncertainty in knowing the true mean and population variance requires that the estimate of the range of values encompassing a given percentage of the population must increase to compensate. The capability of the process is determined not only by the location of the sample mean but also by the tail areas of the distribution. Recommended sampling includes the use of at least 8 data points to calculate $\boldsymbol{T}_{\mathbf{POD(90,\ 95)}}$ to gage an entire population from a small sampling. In the case of the subject CVM testing, convergence of the

POD values and the needed performance levels were used to determine the number of data points to include in the POD calculations.

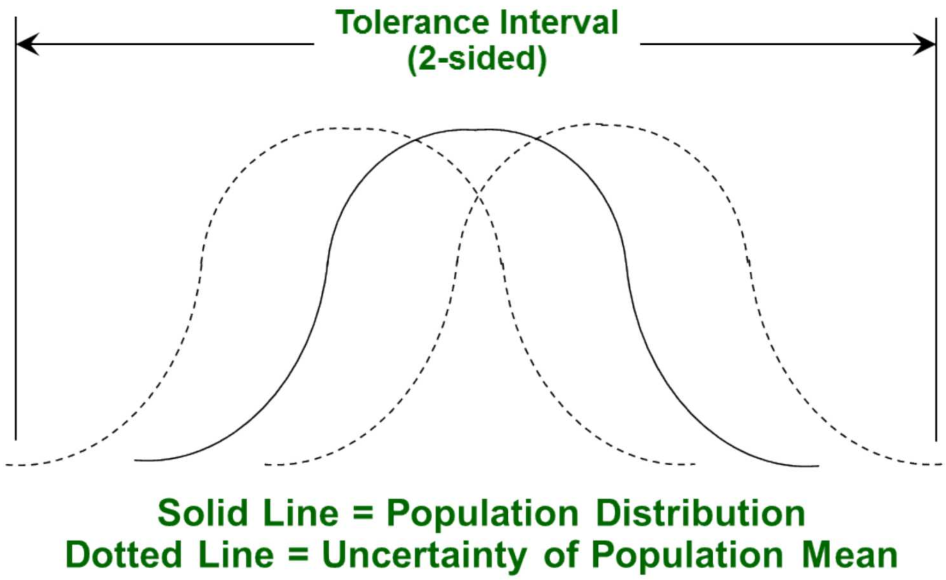


Figure 4-3: Two-Sided Tolerance Interval where the Upper Confidence Bound is Used to Describe the POD Level

Setting Appropriate Thresholds for Crack Detection – For the CVM technology, the key parameter, or Damage Index, for determining crack detection is the dCVM level measured by the PM200 device (see also Section 8.2). Preliminary testing is conducted to acquire dCVM values at different measured crack lengths for validation trending. The crack length "a" is the independent variable. So, system response tests are conducted initially to determine the all-important threshold for assigning "official crack detection." Normally the threshold level is set to provide Signal-to-Noise levels of 3 or greater (as per normal NDI rules-of-thumb) without sacrificing the sensitivity of the system or, conversely, inducing any false calls. Towards that end, the preliminary testing is used to identify any possible signal deviations during crack growth that might induce false calls and then set a threshold to stay above those levels. This has never been an issue with CVM sensors as the sensor dCVM/da plots have always been quadratically increasing plots (no up-down deviations).

In some tests on other SHM sensors, some signal reversals in DI values have been observed during early portions of the crack growth. So, in such cases, the crack detection levels (DI threshold) are set at higher DI values to avoid this gray area. As the crack continues to grow, SHM responses (DI values) tend to rapidly and continuously increase so the safe level to set DI thresholds is normally quite evident.

During the initial tests, dCVM values are acquired as the crack increases in length so that we can assess where to set the threshold. The plots in Figure 4-4 show some sample data where one can place a horizontal line to determine viability as a crack detection threshold. For this data, a dCVM value = 4 was conservatively chosen as the threshold. This produced S/N ratios of 10 to over 100.

The chosen DI threshold will change for different applications depending on the sensor design (number of galleries and associated volume), the length of the small tubes (associated volume) and the structural response (material, crack opening). So, the initial response tests are essential to properly setting these thresholds.

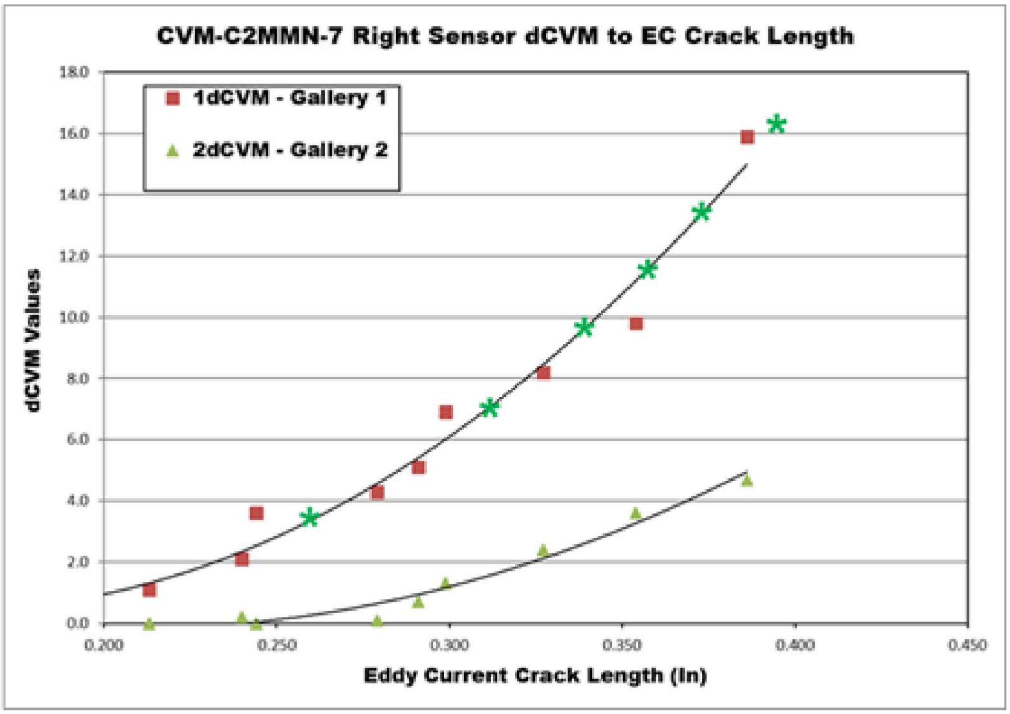


Figure 4-4: Response Relating dCVM Values to Fatigue Crack Length – Used to Establish Proper Threshold to Use for Crack Detection

Sample Application of OSTI to POD Study Based on Comparative Vacuum Monitoring (CVM) Sensors - Comparative Vacuum Monitoring (CVM) sensors provide another method to detect cracks in structures. CVM is a pneumatic, elastomeric sensor with fine channels etched on its adhesive face. Figure 2-9 and Figure 2-9 show top-view and side-view schematics of the self-adhesive, elastomeric sensors with fine channels on the adhesive face along with a CVM sensor being tested in a lap joint panel. When the sensors are adhered to the structure under test, the fine channels and the structure itself form a manifold of galleries alternately at low vacuum and atmospheric pressure. When a crack develops, it forms a leakage path between the atmospheric and vacuum galleries, producing a measurable change in the vacuum level. This change is detected by the CVM monitoring system.

In the sample performance tests discussed here, a CVM sensor was mounted adjacent to a 5mm edge notch on a series of 600 x 40 X 2mm Al-Li coupons. The CVM sensor used a 20mm L crack intercept region with two 0.32mm W sensing galleries to produce the crack detection response. Each test specimen was subjected to tension-tension cyclic loading to initiate and grow natural fatigue cracks. Vacuum levels (Damage Index = dCVM level) were measured every 1,000 cycles and a calibration exercise was used to determine the dCVM value corresponding to sensor crack detection. Figure 4-5 show plots of the CVM data from the subject test series that reveals that the damage index increases exponentially in crack length.

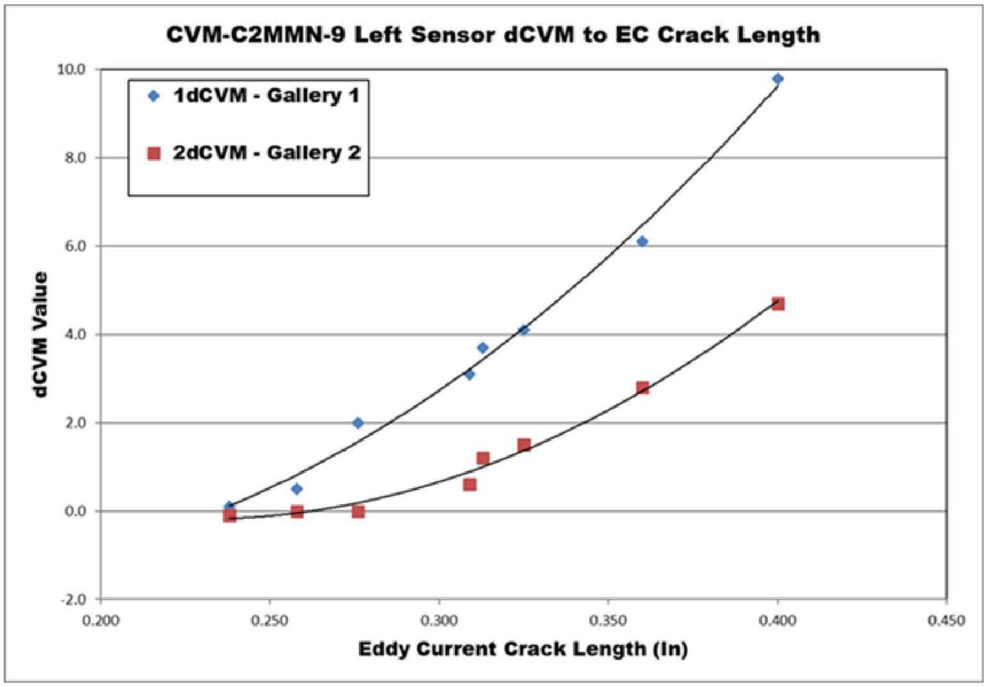


Figure 4-5: Response from CVM Sensors During Performance Testing that Relate dCVM Values to Fatigue Crack Length

Sample response data from the CVM sensor tests described above are shown in Figure 4-6. The crack detection threshold of 1.5 is superimposed on the response data to demonstrate how detection and corresponding crack length are achieved. Table 4-1 summarizes the results from the CVM performance testing described above for one particular specimen. It shows the changing damage detection parameter (dCVM) as the crack grows along with a highlighted level when the dCVM value exceeds the established threshold of dCVM = 1.5 for crack detection. For the example shown of Specimen 6, the crack length at CVM sensor detection is 0.08".

Table 4-2 shows the set of data from all test specimens where only the crack lengths at CVM detection are listed. That is, the crack length listed corresponding to the dCVM value that exceeded the threshold (i.e. crack detection). Since the OSTI calculation is performed in the log domain, the log values are also listed in this table. These crack lengths, which are the actual values measured during testing and not an extrapolated value down to the exact threshold level, were input in the OSTI calculation equation (4.1). The value for the tolerance factor, K, is a function of sample size, detection level desired and confidence level desired. For the data shown, the number of data points is 11, the desired POD level is 90% and the desired confidence level is 95%. The methods described above can be used to determine this K value. The resulting $POD_{(90/95)}$ = 3.35 mm (0.132"). The tabulated detection values listed in Table 4-2 indicate an average crack detection value of 1.85 mm (0.073”) with a standard deviation of 0.47 mm (0.018”), however, as a rough comparison, the statistical POD calculation produces a higher value due to the limited number of data points and the standard deviation in those data points [4.11 – 4.12].

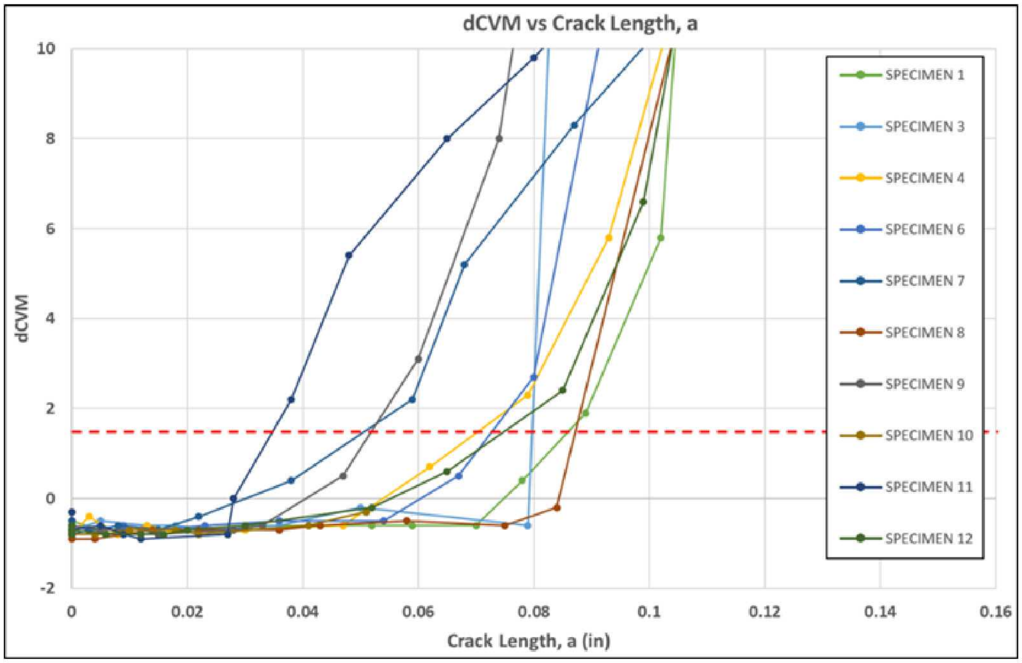


Figure 4-6: Responses from Series of CVM Sensors (dCVM) Monitoring Specimen Crack Growth During Fatigue Tests

SPECIMEN 6		
Cycles	Crack Length	Gallery 1 dCVM
0	0	-0.7
1000	0	-0.7
2000	0	-0.7
3000	0	-0.7
4000	0	-0.6
5000	0.003	-0.7
6000	0.009	-0.6
7000	0.015	-0.7
8000	0.023	-0.6
9500	0.036	-0.5
11000	0.054	-0.5
12500	0.067	0.5
14000	0.08	2.7
15500	0.093	11.2
17000	0.102	25.9
18500	0.12	72
19500	0.148	169.6

Table 4-1: CVM Crack Detection Using Established Damage Threshold

Specimen	Damage Index (dCVM) Level	Crack Length at CVM Detection a (in)	Log of Crack Length at CVM Detection a (in)
1	1.9	0.089	-1.051
2	1.7	0.061	-1.215
3	25.0	0.090	-1.046
4	2.3	0.079	-1.102
5	1.6	0.059	-1.229
6	2.7	0.080	-1.097
7	2.2	0.059	-1.229
8	6.5	0.100	-1.000
9	3.1	0.060	-1.222
11	2.2	0.038	-1.420
12	2.4	0.085	-1.071

Table 4-2: Summary of CVM Crack Detection Levels for Each Test Specimen

4.2 CVM Performance Testing for Specific 737 Wing Box Fitting Application

The goal of this project was to produce sufficient data and to conduct the proper interface with regulatory agencies to certify CVM sensor technology for specific aircraft applications. Towards that end, probability of flaw detection assessments were coupled with on-aircraft flight tests to study the performance, deployment, and long-term operation of CVM sensors on aircraft. Statistical methods using one-sided tolerance intervals were employed to derive Probability of Detection (POD) levels for SHM sensors. The result is a series of flaw detection curves that can be used to propose CVM sensors for aircraft crack detection [4.13]. The test specimens were wing box fittings from the Boeing 737 which was the chosen CVM application from Delta's fleet. The figures below show the details of the wing box fitting application and installation of CVM sensors for the flight test program. Fatigue tests were completed on the wing box fittings using flight load spectrums while the vacuum pressures within the various sensor galleries were simultaneously recorded. A fatigue crack was propagated until it engaged one of the vacuum galleries such that crack detection was achieved and the sensor indicated the presence of a crack by its inability to maintain a vacuum.

Figure 4-7and Figure 4-8 show various views of the wing box fitting, the CVM sensor used to monitor for cracks stemming from any of the attachment holes, and the specialized test fixtures used to apply the proper fatigue stress field. Note the use of an angled mounting fixture on the hydraulic machine's platen to produce the representative ratio of tension and bending stresses. The bolts used to attach the load plate to the machine provided the suitable surface impingement as it was determined that fatigue cracks originated from both the bolt hole and at the point where the induced bending caused the rivet upset region to press on the surface of the wing box fitting flange.

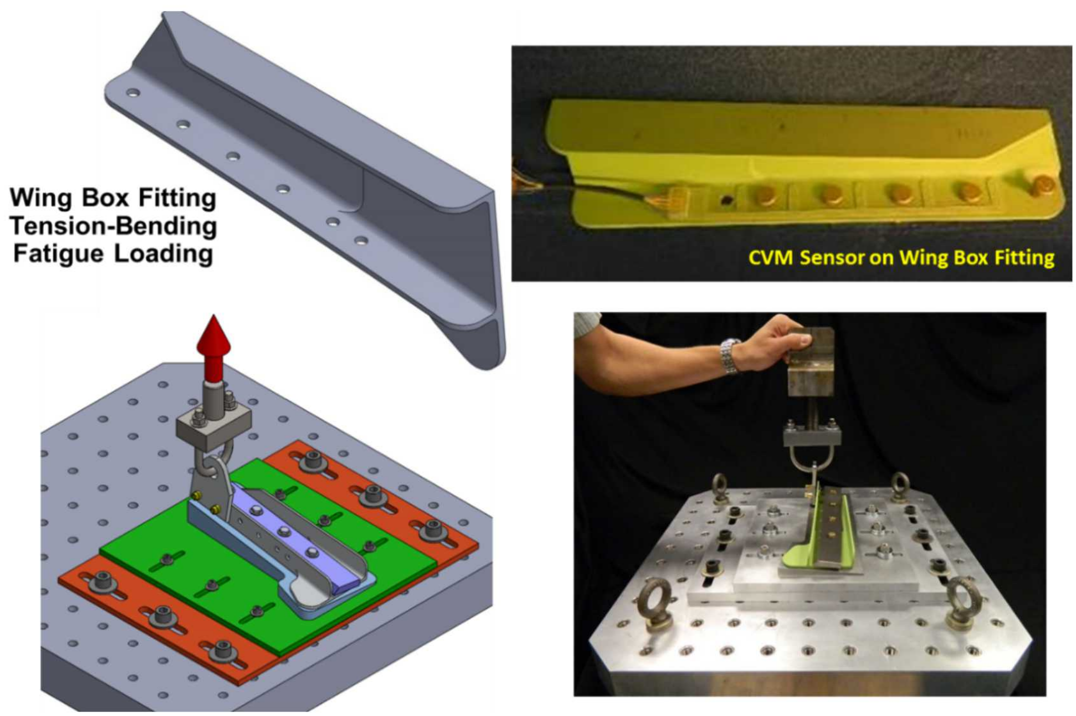


Figure 4-7: CVM Performance Tests 737NG Center Wing Box Fittings – Note use of Angled Mounting Fixture to Produce Proper Ratio of Tension and Bending Stress

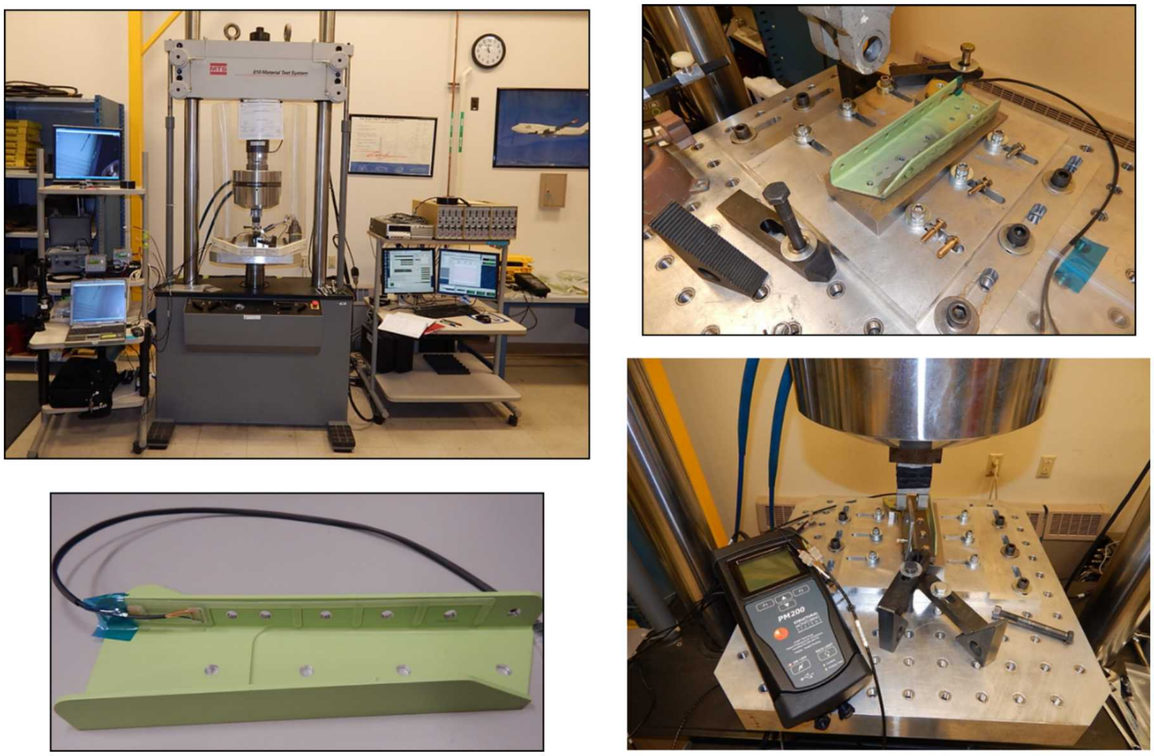


Figure 4-8: Fatigue Test Set Up and CVM Monitoring for Performance Tests on 737NG Center Wing Box Fittings

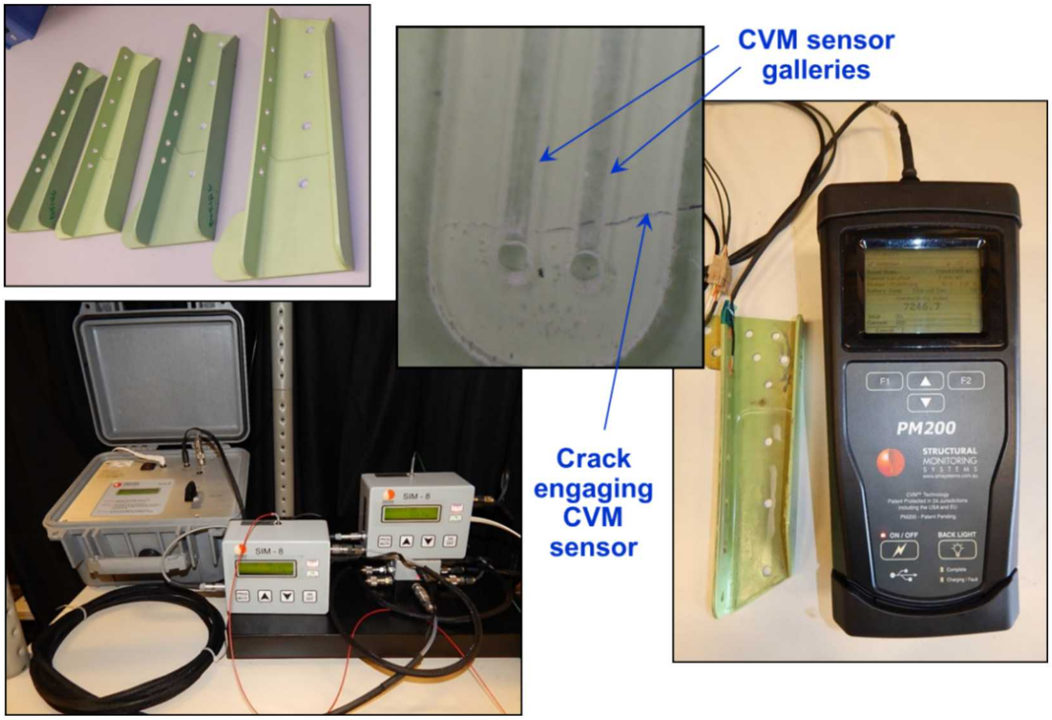


Figure 4-9: CVM Monitoring with Sim-8 (real-time feedback) and PM-200 (field equipment and basis for final crack detection)

Figure 4-9 shows various close-up views of the CVM sensor and the galleries along each "finger" that monitors the region between each bolt hole. Also shown are the monitoring equipment used during testing. The Sim-8 devices monitor the vacuum level in each gallery in real time so that any deviations can be used to stop the fatigue tests as needed for final monitoring. The PM200 device is connected to the CVM sensor at various intervals much as it would be during monitoring (inspections) on an aircraft. Final determination of a crack detection was associated with direct readings and associated failure messages from the PM200 device.

Figure 4-10 and Figure 4-11 show different photos of fatigue cracks engaging the CVM sensor along with close-up views of the test set-up. Crack length measurements, used to relate CVM response levels (dCVM) to actual crack lengths in the structure, were determined using: 1) eddy current inspections to identify the crack tip, and 2) calibrated, high fidelity micro-scales as shown in Figure 4-12 and Figure 4-13.

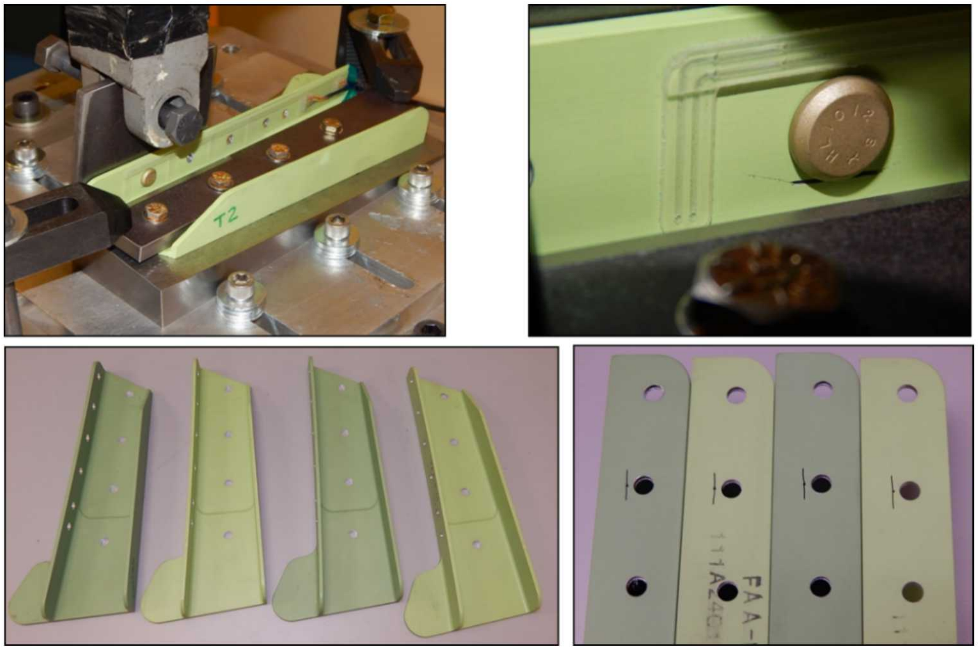


Figure 4-10: CVM Wing Box Fitting Test Specimens with Sensors Installed and Mounted in Fatigue Test Fixtures

Figure 4-11: Fatigue Cracks Fatigue Intercepting CVM Dual Gallery Arrangement

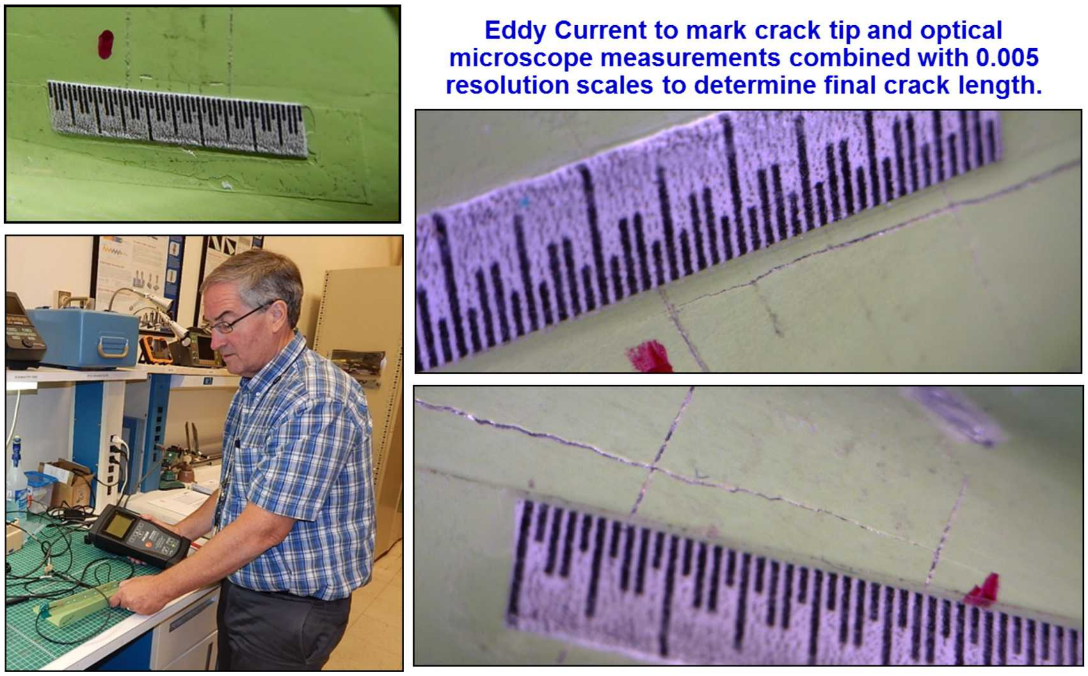


Figure 4-12: Eddy Current Inspections to Measure Crack Length

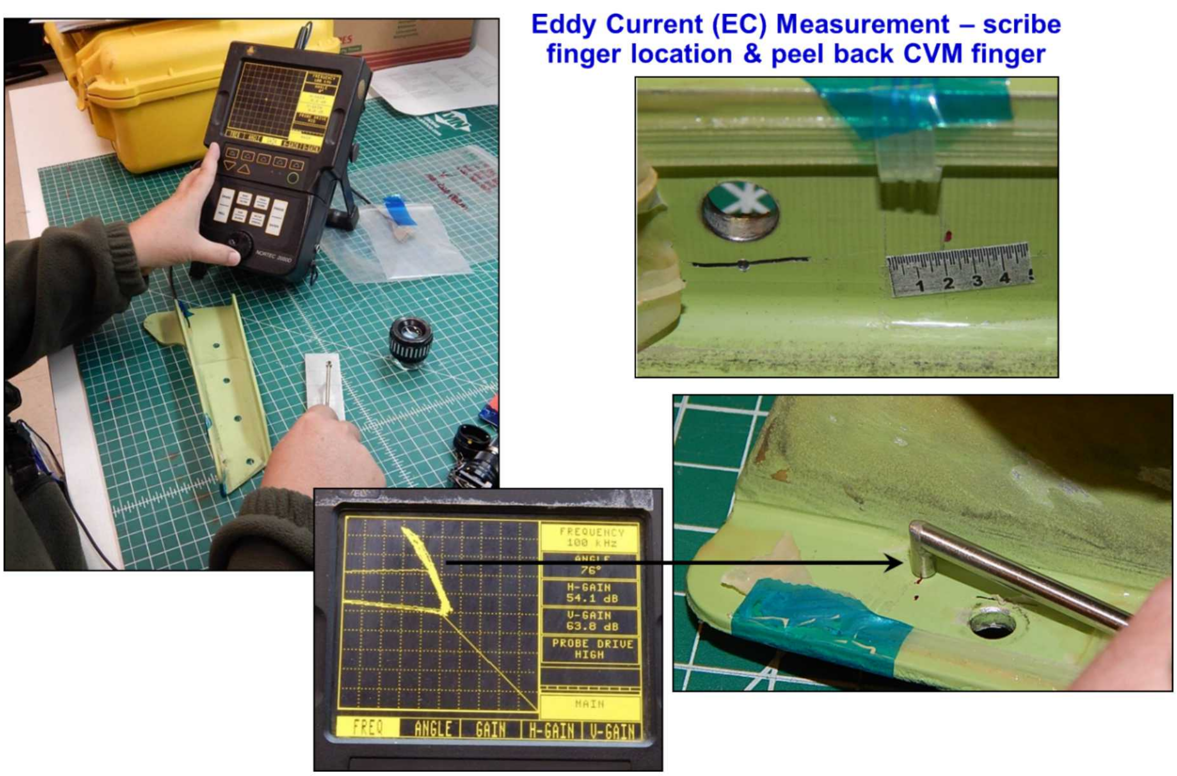
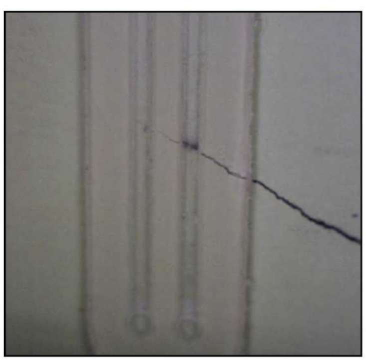


Figure 4-13: Crack Length Measured with Eddy Current Inspections to Plot Data with Associated dCVM Levels Measured by PM-200

A detailed explanation of the components in the total crack length and the method used to determine the total crack length at CVM detection is presented in Figure 4-14 and Figure 4-15. The critical measurement is the excursion of the crack into the CVM sensor (crack length under the sensor). The distance from the crack origin to the edge of the sensor can then be added to determine the total crack length at detection. This approach allows for the distance from the crack origin to the edge of the sensor to be a variable that can be adjusted to accommodate the expected placement variations in the CVM sensor. Worst case conditions can be used when calculating the final POD level such that the final performance assessment is arrived at in a conservative manner.

Fatigue tests were halted at regular intervals and also when any indications from the real-time Sim-8 monitoring equipment indicated that the CVM sensors were changed response due to cracks engaging the galleries.

- Monitoring for permanent crack detection – unloaded, unfastened and multiple day lag in readings
- Sealant (FVB) applied to determine crack detection when entire surface is sealed. Figure 4-16 shows the application of normal Fuel Vapor Barrier (FVB) that is applied to all surfaces in the wing box region. Since the CVM sensors rely on a crack connection to atmospheric pressure – either via a crack connecting two adjacent galleries or via a crack path to a region outside the sensor – the use of the sealant layer was critical to producing accurate crack detection values.
- Crack detection thresholds were chosen such that the resultant POD $_{[90/95]}$ for 1st & 2nd gallery produced an S/N > 10.



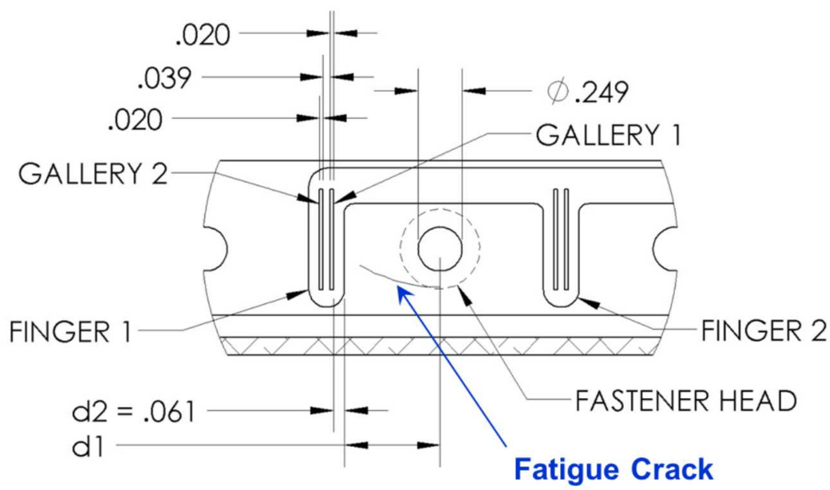


Figure 4-14: Explanation of Crack Engagement with CVM Sensor and Method Used to Determine Total Crack Length at CVM Detection

In order to properly consider the effects of crack closure in an unloaded condition (i.e. during sensor monitoring), a crack was deemed to be detected when a permanent alarm was produced and the CVM sensor did not maintain a vacuum even if the fatigue stress was reduced to zero. Crack detection lengths ranged from 0.145” to 0.245” in length for the wing box fitting application. Data acquired from CVM fatigue tests were used to calculate the 90% POD level for CVM crack detection on the wing box fittings subjected to tension-bending fatigue loading described above.

- All crack detections are for the most conservative unloaded state, fasteners loose and entire part sealed with FVB (crack, sensor & fitting). When all fasteners and fixtures were loosened, the SIM-8 and PM-200 values occasional was reduced (crack closure occurred) so "permanent detection" required acceptable PM-200 readings with all fasteners loose. Even in an unloaded aircraft, fittings are still fastened in place. Thus, the crack detection lengths recorded are the most conservative values.
- Over the course of the fatigue tests, crack detection was achieved from multiple CVM fingers in the sensor (crack originated at different holes in the fitting).
- For the official POD testing of a wing box fitting coated with FVB, matching its condition on an operating aircraft, eight of the specimens were pristine wing box fittings where starter notches were used to initiate the fatigue crack. The other twelve were removed from aircraft and 5 of those contained natural fatigue cracks which were propagated into the CVM sensor. One specimen did not use a starter notch to determine any differences in crack growth and sensor response - there was none.

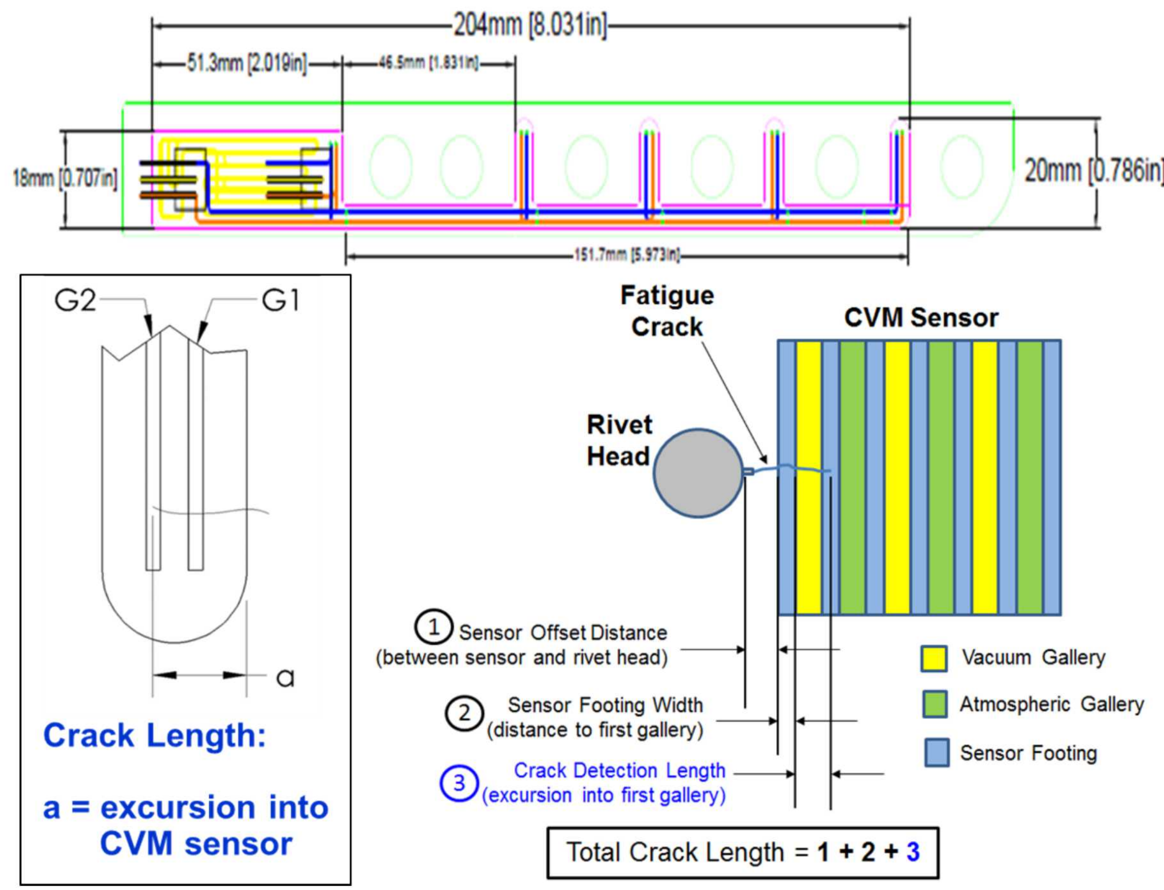


Figure 4-15: Crack Length Measured (Excursion into Sensor) and Additional Components forming Total Crack Length

- All SIM-8 and PM-200 readings listed correspond to the final, permanent crack detection. For an idea of the signal-to-noise ratios, the baseline (pristine) SIM-8 and PM200 readings without any cracks present were approximately: (SIM1 = 52 Pa, SIM2 = 44 Pa) and (1dCVM = 0.4, 2dCVM = 0.1) for SIM-8 and PM200, respectively.
- Crack detection requirement for this inspection corresponds to a fastener-to-fastener crack and visual inspections. In the wing box fitting fastener holes, the distance from center hole to center hole is 1.329 inches. Thus, any POD level less than 1.3" was deemed as good as or better than the current inspection requirement.

Figure 4-16: Application of Fuel Vapor Barrier on CVM Sensor and Wing Box Fitting

CVM Sensor Installation and Fatigue Test Specimen Preparation - CVM sensor installations were completed as per SMS documents that are described in the quality assurance portion of Section 8.0. The basic steps for the surface preparation and sensor installation are as follows:

1) Apply primer to test specimen.
2) Remove grease, dirt or any contaminants using a clean, lint-free cloth and Acetone or Rhodiasolve.
3) Use 600 grit sandpapers to sand the CVM installation area.
4) Clean the sanded surface again using a clean, lint-free cloth and Acetone.
5) Conduct final cleaning with deionized water.

6) Apply self-adhering sensor to the surface. Use and guides (e.g. templates, surface markings, hole dowels, specialized tools) that are helpful in ensuring the accurate and repeatable placement of the sensor.
7) Use a flat-ended spatula to press down on the sensor and produce an air-tight seal with the structure surface.
8) Allow sensor to sit for at least 15 minutes. Connect CVM sensor to PM200 unit and measure the baseline (no crack) readings for proper dCVM and continuity levels.

CVM Sensor Fatigue Test Procedure - The testing and data acquisition steps were as follows:

1. Load each test specimens into the 110,000 pound MTS machine in accordance with the test configuration shown in Figure 4-7.
2. Set the load to produce the desired stress level that is representative of the stress levels experienced in the structure during operation. Determine and apply a suitable R ratio for the fatigue loading to establish the lower load in the fatigue tests. Apply a fatigue cycle frequency while maintaining the ability to apply the proper maximum load levels and R value.
3. Take a digital USB microscope picture of each test location (the intended crack site) prior to fatigue testing.
4. Measure the distance between the rivet hole and the edge of the sensor. This distance is the Sensor Offset Distance.
5. Verify initial installation and sensor function prior to data acquisition (see Figure 4-17). Take PM200 measurement on un-cycled test coupon to ensure proper seal between the sensor and coupon. Record dCVM values and continuity vales as determined by the PM200. Connect sensors to Kvac-4 and Sim-8 fatigue test set-up and record the baseline Sim-8 readings for the real time sensor monitoring.
6. Connect the Sim-8 units to the MTS load machine such that the machine will automatically stop if the Sim-8 detects the initial presence of a crack.
7. Fatigue cycle the specimen while taking measurements with Sim-8 devices to determine the point at which the SHM sensors detect the presence of a crack (Figure 4-9). Continue this process until a sensor initial crack detection has occurred as indicated by Sim-8 real-time, dynamic reading of 12,000 to 15,000 Pa.
8. Bring the specimen to an unloaded state and use Sim-8 indications to determine if there is still an initial crack detection. Continue the fatigue cycling at very low intervals until a crack is detected by the Sim-8 device when the specimen is in an unloaded state.
9. Connect the CVM to the PM200 CVM monitoring device and determine if the PM200 is able to detect the crack. Use a dCVM reading of +/- 10.0 as the threshold for PM-200 crack detection. This value is determined in calibration testing as described in Section 4.1. If the PM-200 does not detect a crack, continue to fatigue cycle the sample in small increments until the PM200 CVM system properly detects the presence of a crack when the specimen is in an unloaded state (dCVM > 10.0).
10. Confirm the location and presence of damage (fatigue crack), along with the crack length using conventional eddy current NDI methods and an optical microscope (see Figure 4-12 and Figure 4-13).
11. Record the crack lengths at CVM detection. Log any false calls where the CVM system indicates a crack detection when a crack is not actually present.

12. Use One-Sided Tolerance Interval method to calculate CVM POD level when all crack detections are included.

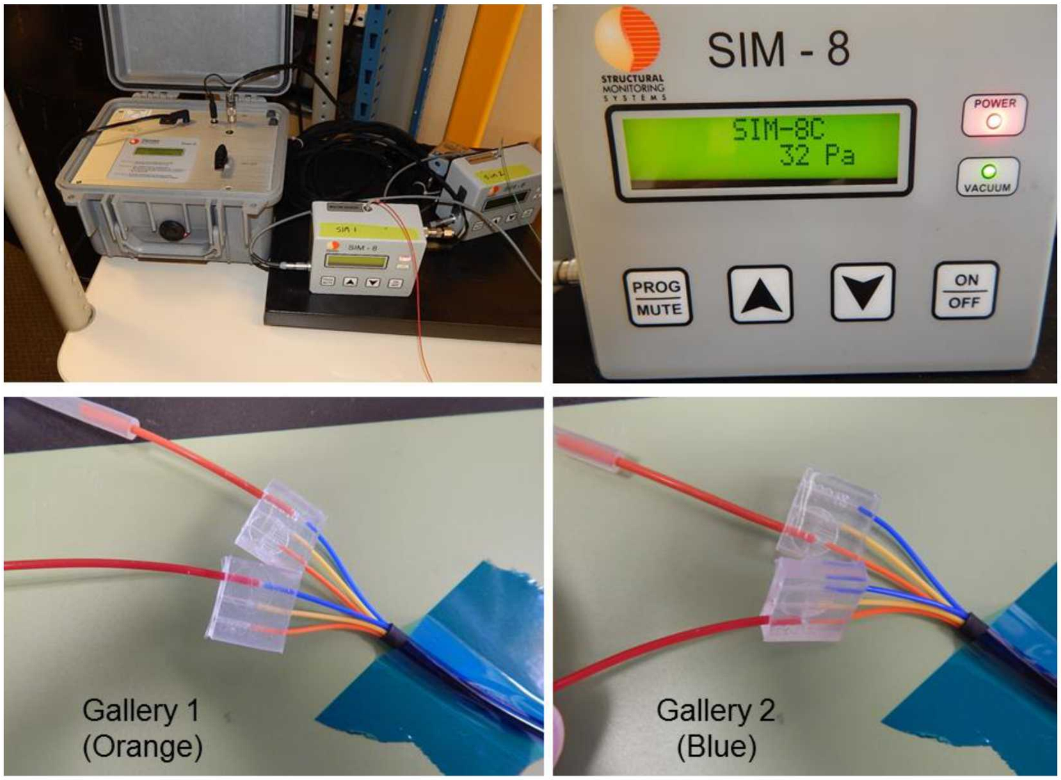


Figure 4-17: Connect CVM Sensors to Sim-8 and Kvac-4 Units for Real Time Monitoring and Measure the Baseline Value for All Galleries

Table 4-3 summarizes some preliminary results from fatigue testing that was conducted without the FVB sealant covering the CVM sensors. In this case, crack detection was achieved when the crack engaged the first gallery to provide the path to atmosphere. Note average total crack length at detection of 0.095”. Figure 4-16 show the process used to seal the sensors as they would be when operating in the wing box area (i.e. fuel vapor barrier installed). In this case, crack detection was achieved when the crack engaged and connected Galleries 1 and 2 to provide the path to atmosphere.

Twenty data points were acquired from fatigue tests on actual wing box fitting test specimens to produce the POD calculations. Table 4-4 summarizes the test data including the crack lengths at detection and the corresponding PM-200 readings. Note that the average crack length at CVM detection was 0.209”.

With the use of 20 data points, the reliability calculations include a corresponding magnitude of the K (probability) factor. As a result, while most of the crack detection levels were less than

0.2”, the overall POD value (95% confidence level) for CVM crack detection was calculated from equation (1) as 0.258”. The K values correspond to the desired ɣ (confidence level) of 95%. As the number of data points increases, the K value will decrease and the POD numbers could also decrease.

Results for Gallery 1 Permanent Detection (viable without complete FVB seal)

CVM Sensor Wing Box Fitting Performance Tests					
Test No.	CVM Finger Location	Sensor Distance from Fastener d_1 (In)	Crack Length at CVM Detection a (In)	SIM-8 Reading (Pa)	PM200 Reading (dCVM)
T1	2	0.488	0.084	282	7.4
T2	1	0.524	0.109	496	35.5
T3	1	0.550	0.089	2017	157.5
T4	1	0.570	0.094	330	14.4
T5	1	0.574	0.084	285	8.9
T6	1	0.580	0.079	2901	264.8
T7	2	0.546	0.124	318	22.5
* Final values being confirmed					
** Detection for unloaded state with sealed crack and sensor					

Avg CVM Gal 1 Detect Length = 0.095”
Distance from CVM Edge through Gal 1 = 0.081”

Table 4-3: Sample Crack Detection on Wing Box Fitting Without Fuel Vapor Barrier in Place (Gallery 1 Detection)

In order to obtain the total crack length at CVM detection, the distance from crack origin (rivet head or rivet hole center) to the sensor edge must be added to the POD level described above. This is because the POD value calculated from the data presented in Table 4-4 corresponds to the crack excursion (length) into the sensor itself (see also Figure 4-14 and Figure 4-15). Table 4-4 lists the average distance from the hole center to the edge of the CVM sensor finger as 0.552”. Thus, the final $POD_{(90/95)}$ value for the wing box application is determined to be 0.81”. In this particular instance, it was desired to achieve crack detection before the crack reached 1.3” in length so this goal was achieved. There were no False Calls associated with these tests where the CVM sensor indicated the presence of a crack when actually none was present. In over 200 fatigue tests conducted using CVM sensors there have been no false calls produced by the sensors in any of the tests.

Oversight and review of all performance assessment testing was conducted by Boeing in conjunction with Sandia Labs. Boeing review of the CVM validation and performance testing included:

- Review of all CVM Validation Test Plans

- Review of fatigue test set-up, loading, equipment and calibration
- Review of data acquisition for CVM crack detection via SIM-8 and PM-200 devices
- Review of crack measurement for POD assessment
- Participation in fatigue testing on one specimen until permanent crack detection by CVM is achieved – crack growth through Gallery 1 and Gallery 2
- Review of environmental testing procedures, equipment and calibration
- Trail run of environmental chamber to demonstrate feedback and control of temperature and humidity
- Review and observation of application of fuel vapor barrier to wing box fitting such that CVM and backside of fitting are coated with FVB

CVM Sensor Wing Box Fitting Performance Tests								
Test No.	**Fatique Cycles at permanent CVM Crack Detection**	**Sensor Distance from Fastener d_1 (In)**	**Eddy Current Crack Length at CVM Detection a (In)**	**Total Crack Length (In)**	**SIM-8 Reading Gallery 1 (Pa)**	**SIM-8 Reading Gallery 2 (Pa)**	**PM200 Reading (1dCVM)**	**PM200 Reading (2dCVM)**
T1	87,098	0.488	0.215	0.703	338	720	18.3	44.5
T3	58,528	0.550	0.193	0.743	1468	1456	130.7	129.3
T4	53,726	0.570	0.193	0.763	318	330	14.1	14.4
T5	91,273	0.574	0.205	0.779	232	228	12.9	13.6
T6	84,277	0.580	0.200	0.780	2602	2692	257	264.8
T7	69,459	0.546	0.243	0.789	271	310	12.4	15.7
T9	105,239	0.605	0.180	0.785	560	390	37.6	25.1
T10	59,392	0.570	0.205	0.775	271	277	17.4	18.3
T11	20,225	0.621	0.238	0.859	261	274	15.5	16.2
T12	21,229	0.569	0.240	0.809	2451	2491	253.6	258
T13	39,553	0.528	0.258	0.786	304	227	19	19.7
T14	79,508	0.588	0.218	0.806	N/A	N/A	10.4	10.3
T15	148,139	0.566	0.178	0.744	200	205	9.9	11.1
T16	131,596	0.481	0.175	0.656	332	309	20.2	19.4
T17	26,367	0.566	0.220	0.786	243	258	14.2	14.7
T19	300,292	0.584	0.198	0.782	1328	511	97	29.3
T20	79.413	0.572	0.208	0.780	278	270	15.7	16.6
T21	191,030	0.526	0.193	0.719	244	255	13.9	14.9
T22	192,987	0.432	0.235	0.667	252	234	13	10.8
T23	213,030	0.529	0.183	0.712	205	214	13.3	13.8

Avg Distance From Finger to Center Hole	**0.552**

Avg CVM Detection	**0.209**
CVM Detect Std Dev	**0.024**

Table 4-4: CVM Crack Detection Performance with FVB Installed (Crack Detection at Gal1 to Gal2 Connection)

CVM Crack Detection Data

Eddy Current Crack Length at CVM (In)	Log of Crack Length at CVM Detection a (In)
0.215	-0.66756154
0.193	-0.714442691
0.193	-0.714442691
0.205	-0.688246139
0.200	-0.698970004
0.243	-0.614393726
0.180	-0.744727495
0.205	-0.688246139
0.238	-0.623423043
0.240	-0.619788758
0.258	-0.588380294
0.218	-0.661543506
0.178	-0.749579998
0.175	-0.756961951
0.220	-0.657577319
0.198	-0.70333481
0.208	-0.681936665
0.193	-0.714442691
0.235	-0.628932138
0.183	-0.73754891

Statistic Estimates on Log Scale

Statistic	Value in Log Scale	Value in Linear Scale
Mean (X)	-0.682724025	0.209
Stnd Deviation (S)	0.049124663	0.023962471

POD Detection Levels
(γ = 95%, n = 20)

Flaw Size: POD = X + K(S)	0.258160667

Overall POD (with sensor offset from crack origin):
Final POD $_{(90/95)}$ = 0.258" + 0.552" = 0.81"

It is possible to calculate a one sided tolerance bound for various percentile flaw sizes - find factors $K_{n,\gamma,\alpha}$ to determine the confidence γ such that at least a proportion (α) of the distribution will be less than $X + (K_{n,\gamma,\alpha})S$ where *X* and *S* are estimators of the mean and the standard deviation computed from a random sample of size *n*

Table 4-5: POD Determined Using the One-Sided Tolerance Interval Method Applied to the CVM Response Data on Wing Box Fitting

POD Analysis Using Standard Hit-Miss Methodology (Mil-HDBK-1823) – Traditional methods for calculating POD values from NDI tests are described in Reference [4.14]. One of these methods is called the Hit-Miss or Log-Regression analysis. In this model, the *POD(a)* function is defined as the proportion of all cracks of size *a* that will be detected in a particular application of an SHM system. Analysis of data from reliability testing indicates that the *POD(a)* function can be reasonably modeled using the log normal distribution function or a Log Regression analysis. Thus, if the SHM system can produce output (detection) that can be reduced to a binary response, such as the CVM data, a Log-Regression (***hit/miss)*** analysis can be used [4.14]. The conditional probability of a randomly selected crack population having detection probability of p and being detected at the inspection is given by $p\, f_a(p)$. The unconditional probability of a randomly selected crack from the population being detected is the sum of the conditional probabilities over the range of p, that is:

$$\mathrm{POD}(a) = \int_0^1 p\, fa(p)\,dp \qquad (4.3)$$

The Log Regression *Hit/Miss* POD model is used to analyze binary (detect/no detect) data using the following underlying mathematical relationship between POD and crack size:

$$\text{POD}(a) = \frac{\exp[\alpha + \beta[ln(a)]}{1 + \exp[\alpha + \beta[ln(a)]} \quad (4.4)$$

A brief overview of the Hit-Miss method follows:

- Early attempts to quantify probability of detection, POD, considered the number, n, of cracks detected, divided by the total number, N, of cracks inspected, to be a reasonable assessment of system inspection capability, POD = n/N. This resulted in a single number for the entire range of crack sizes. Grouping specimens this way improved the resolution in crack size, but the resolution in POD suffers because there were fewer specimens in each range & many factors influence the probability of detecting any one given flaw
- If the SHM system can produce output (detection) that can be reduced to a binary response, a *hit/miss* analysis can be used (*Hit/Miss* POD model)
- A perfect inspection produces is a step function, as shown in Figure 4-18, with POD = 1 for $a > a_{crit}$ and POD = 0 when $a < a_{crit}$. It is *not* a POD(a) = constant = 1 because an inspection that finds everything is useless since it cannot discriminate between an actual crack and a benign microstructural artifact, an edge, or a surface blemish.
- An efficient use of the binary (***hit/miss***) data is to produce an underlying mathematical relationship between POD and size.
- Logistic Regression ***Hit/Miss*** **POD model** is used to analyze binary (detect/no detect) data

$$ln[POD(a)/(1\text{-}POD9a))] = \alpha + \beta[ln(a)] \quad (4.5)$$

Where,

a = crack length

α and β are estimated by maximum likelihood estimates.

- Assumption is for no variation in equipment or procedures
- Assumption is all critical factors are controlled in the testing so there is no need for additional φ function to describe other factors on the RHS of log regression formula
- Each flaw is either detected or not detected so the best estimate for POD(a) is either 0 or 1. A range of flaw sizes are used to determine the α and β that maximize the likelihood of the particular sequence of 0's (misses) and 1's (detects) that were observed. Figure 4-19 shows a typical POD curve determined by the Hit-Miss analysis.

The Hit-Miss POD analysis method requires the use of approximately 50 independent data points from 50 different crack sites. In order to create a comparison that relates the POD calculated from the OSTI method to traditional POD assessments, the data from the POD testing described above was applied in a Hit-Miss POD analysis. Some extrapolation of the CVM crack detection data was necessary to produce sufficient data using only 20 independent crack detection tests. Thus, it must be stressed that the exercise of conducting the Hit-Miss POD calculations is used here for simple comparisons to the methodology used in Mil-Hnbk-1823. Following are some considerations when using the Hit-Miss methodology with the 20 data points acquired in the SHM testing.

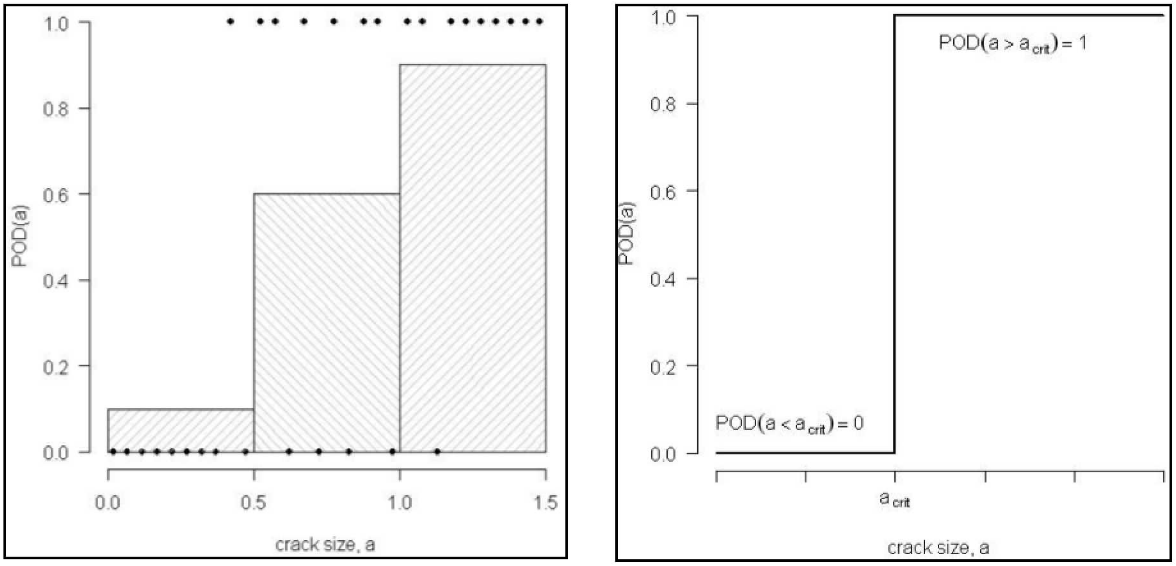


Figure 4-18: Comparison of Initial Simple POD (n/N) with a Step Function from a Perfect Inspection

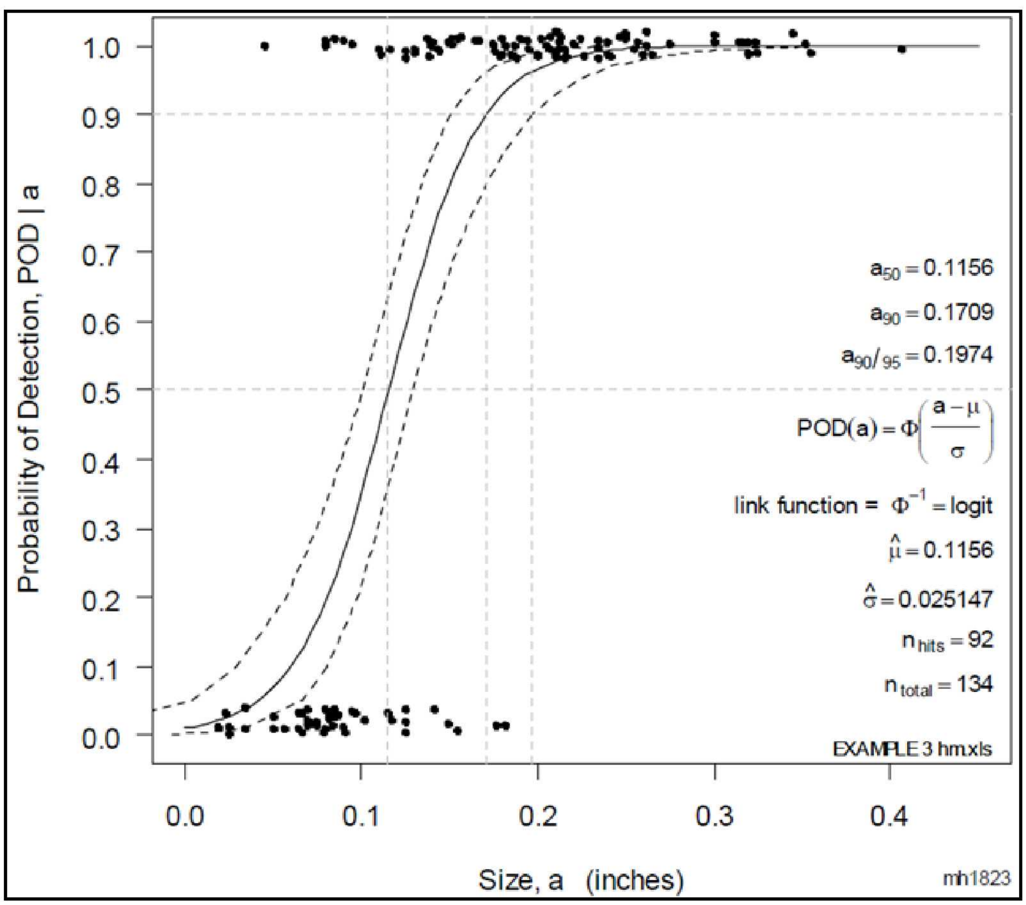


Figure 4-19: Construction of POD Curve from Hit (1) and-Miss (0) Inspection Data

A Gaussian distribution of hit-miss data was compiled using crack CVM detection length from each test augmented by assumed, missed crack detections below the actual CVM detection level & assumed, hit crack detections at lengths above the actual CVM detection level. Following is a description on the use of this resulting data set in the Hit-Miss POD assessments:

- Normal NDI POD values are calculated using only independent data points. This includes and independent distribution of seeded cracks, unique signals at detection are logged (one reading on each target), test series accounts for operator-to-operator (sensor-to-sensor) variability, array of specimens is sufficiently large to account for crack-to-crack variability.
- Log-Regression (hit-miss) Model – For CVM data, there were 20 independent tests (cracks) as presented in Table 4-4. The corresponding 65 hit-miss data points were acquired from these 20 tests and thus, not all independent. Additional extrapolated data at extremes (small & large cracks) were used to populate a complete POD curve
- In the Hit-Miss assessment conducted with the limited CVM response data, the calculations are carried out with the assumption that each data point is independent and is produced by a separate crack (separate specimen). This is not the case because the Hit-Miss analysis took credit for the additional, extrapolated data as independent data points (Mil-HDBK-1823 calculation).
    - ➢ If the sensors, their location, the cracks, and the sensor response is consistent enough that the assumed data is representative (additional tests produced independent data that is equivalent to the repeated measures assumed data), then the resulting “hit-miss” calculations are close to the truth.
    - ➢ If the actual responses – should many additional tests be conducted – exhibit lack consistency (deviate significantly from the assumed response), then the Hit-Miss calculations will have a much larger deviation from the truth.
    - ➢ Results obtained in the significant test set from multiple years of CVM performance testing [4-4 – 4-8, 4-12 – 4-15] gives us confidence in the extrapolations listed here and used in the “hit-miss” calculations. The assumption of consistent, additional data, based on the existing set of 20 data points, is a justified assumption but only for comparison purposes.
- Repeated measures data (multiple data points from a single crack profile and CVM response) are used in these calculations; this is an assumption that is not statistically valid. It does not account for possible crack-to-crack variations from different specimens.
- However, these results are for illustrative purposes only and not for any certification of performance. The results calculated from this hit-miss analysis are for general comparisons only.
- *Certification results are to be taken only from the OSTI method already presented above.*

Table 4-6 and Figure 4-20 summarize the results for each individual test specimen where the hit-miss data surrounding the CVM crack detection has been extrapolated from the raw test data. Equation (4.3) was used to calculate the individual and compiled POD values. The spray (variation) in $POD_{(90/95)}$ values show the consistency of the results while the overall $POD_{(90/95)}$ value of 0.247” can be used for comparison to the OSTI $POD_{(90/95)}$ value of 0.258”. These results from the Hit-miss POD method, which represents traditional POD analyses, compare well with the OSTI method (within 4.3%).

CVM Sensor Wing Box Fitting Performance Tests Hit/Miss POD Values		
Test No.	Eddy Current Crack Length at CVM (In)	$POD_{90/95}$
T1	0.215	0.252
T3	0.193	0.232
T4	0.193	0.232
T5	0.205	0.243
T6	0.200	0.224
T7	0.243	0.279
T9	0.180	0.220
T10	0.205	0.243
T11	0.238	0.274
T12	0.240	0.276
T13	0.258	0.293
T14	0.218	0.255
T15	0.178	0.228
T16	0.175	0.225
T17	0.220	0.257
T19	0.198	0.236
T20	0.208	0.246
T21	0.193	0.232
T22	0.235	0.271
T23	0.183	0.223
Cumulative		0.247

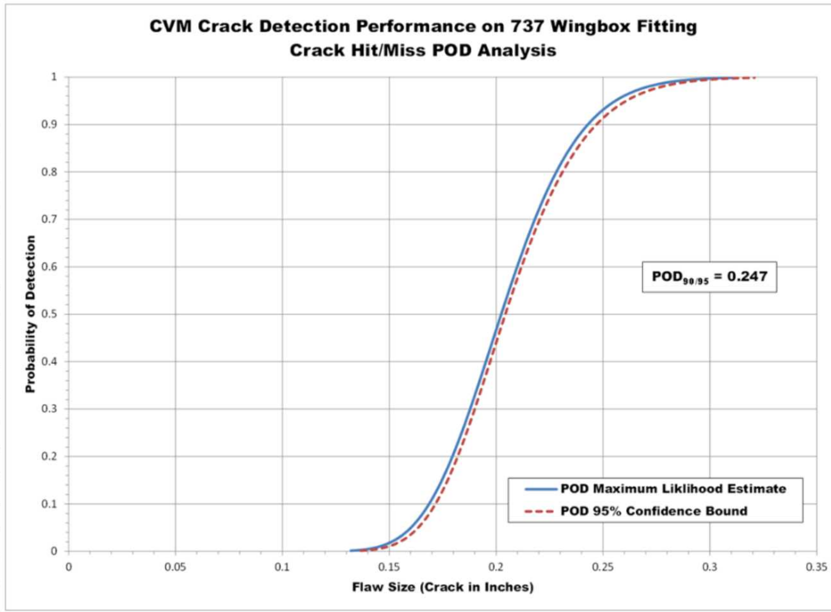


Table 4-6: CVM $POD_{(90/95)}$ Values Determined from Hit-Miss Analysis Method Performed on Wing Box Fitting Crack Detection Data and Extrapolated Results

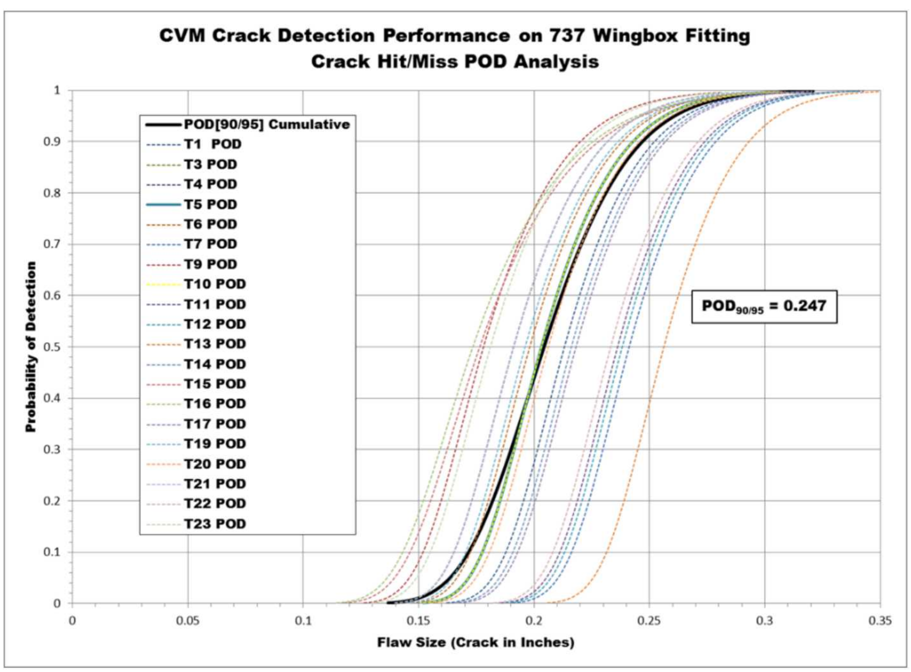


Figure 4-20: Hit-Miss POD Approximation - Spray of Results from Individual Inspectors and Cumulative POD from All Performance Data Combined

Damage Detection with SHM Signals – The important aspect of SHM damage detection relates to determining how the SHM signal gets translated into flaw detection. From a simple sensor standpoint, SHM is very analogous to NDI where the set of signals represent first a baseline, corresponding to a pristine structure, and later a deviation from the baseline, possibly corresponding to a damaged structure. This deviation is used to infer the presence of a flaw. Depending on the equipment and the type of inspection being conducted, the guidance on how to delineate a flaw may differ but it is normally rooted in some desired signal-to-noise ratio which has been determined to produce the best POD while minimizing false calls. Some transducer/sensor signals may provide a more direct measure of damage (e.g. abnormal reflection peak that is absent in a pristine part) and some may be secondary and require extensive calibration (e.g. change in strain level created by nearby damage). These are more sensitivity issues which affect POD assessments but still follow the process of using deviations in signal signatures to identify flaws. Similarly, Damage Indices or other parameters based on the sum total of signals received may aid sensitivity but should not change the process for quantifying performance. Quantifying SHM performance using the Log Regression Method only requires that the signal deviation can be reduced to produce a simple detection (hit) or no-detection (miss). Thus, the mapping of SHM signals to flaw detection is key. It is possible to lower damage detection threshold in both NDI and SHM in order to improve POD, possibly at the expense of increasing false calls. However, the normal rule of thumb is that it is best to maintain a signal-to-noise ratio of at least 3 to avoid misinterpretation of data that may stem from normal deviations in the SHM system.

With the above hit-miss data description in mind, it is important to highlight a few differences in how this final, binary data is produced. For example, when deploying scanning NDI methods (e.g. pulse-echo ultrasonics on an X-Y motion gantry), it is possible to set gates during data acquisition based on the amplitude and/or time-of-flight information where the presence or absence or change in signals creates a variation in the resulting color-coded image. Such changes are used to identify a flaw. This is similar to an SHM threshold. However, in the case of NDI, it is possible for the inspector to revisit the potential damage location (or the data corresponding to the potential damage location) and conduct additional evaluations to further convince himself that damage is actually present. Even hand-held NDI deployment allows for multiple passes of the NDI transducer around the same area in question and each time may involve transducer motion from a slightly different direction. This type of human feedback loop is missing in SHM as SHM methods utilize automated data acquisition and analysis to arrive at a final hit-miss assessment. This highlights the fixed nature of SHM deployment where a well-designed SHM sensor network aims to properly model these different paths to adequately capture the necessary signal variations for analysis. So, the success in applying the Log Regression Method lies in the ability of the SHM system to produce acceptable binary data. Thus, the POD testing must accommodate all of the key variations within the set of POD specimens using statistical distribution. Such variations include flaw type, size, orientation, depth and location within the sensor coverage area.

4.3 CVM Performance Testing for General Fuselage Skin Crack Detection

Prior to moving into the wing box fitting application, a laboratory and field evaluation program was conducted for the purposes of including usage of CVM as an option in the Boeing NDT Standard Practices Manual. This activity involved a set of general skin crack detection tests and the results are presented here for completeness. This was the first CVM validation program. It occurred in the 2001 to 2003 time frame and was intended to establish the overall capability of CVM sensors such that CVM technology could be included in Boeing's NDT "tool box" (NDT Standard Practices Manual). The testing was designed to establish the ability of CVM sensors to detect cracks in fuselage skin structure and to determine the limits on skin thickness applications such that a crack of 0.10" length could be reliably detected. The end result of the laboratory and flight testing was that Boeing's NDT Standard Practices Manual was revised to include CVM sensors as a possible structural monitoring option.

The AANC at Sandia Labs, in conjunction with Boeing, Northwest Airlines, Delta Airlines, Bombardier, Structural Monitoring Systems, the University of Arizona, and the FAA, completed validation testing on the Comparative Vacuum Monitoring (CVM) system in an effort to adopt CVM as a standard SHM practice. Fatigue tests were conducted on simulated aircraft panels to grow cracks in riveted specimens while the vacuum pressure within the various sensor galleries was simultaneously recorded. The fatigue crack was propagated until it engaged one of the vacuum galleries such that crack detection was achieved (sensor indicates the presence of a crack by its inability to maintain a vacuum). In order to properly consider the effects of crack closure in an unloaded condition (i.e. during sensor monitoring), a crack was deemed to be detected when a permanent CVM alarm was produced even if the fatigue stress was reduced to zero.

Table 4-7 summarizes the two specimen designs and the matrix of tests that were conducted to assess CVM crack detection for different materials, skin thicknesses and surface coatings. This test program produced a statistically-relevant set of crack detection levels for 0.040", 0.070" and 0.100" thick panels in both the bare and primed configurations. It was determined that CVM crack detection performance was better on primed surfaces than on bare metal surfaces. This is attributed to the brittle nature of primer which will readily rupture to match the crack beneath the primer. Since a primed surface is more desirable than a bare surface, due to the corrosion protection capability of primer coatings, the primed surface results are emphasized here.

The 90% POD levels for crack detection on aluminum structures of various thicknesses and surface conditions were calculated using the OSTI described above. As a preliminary example, data acquired from CVM fatigue tests were used to calculate the 90% POD level for CVM crack detection on 0.1" thick 2024-T3 bare aluminum structure subjected to tension-tension fatigue loading. Table 4-8 summarize the crack detection data and shows the calculated quantities for equation (4.1) in the log transform. Twelve data points (bare surface) were used for the OSTI POD calculations. Due to the reduced number of data points compared to conventional Log

Regression POD calculations described above (~ 50 data points), the reliability calculations induce a penalty by increasing the magnitude of the K (probability) factor. As a result, while most of the crack detection levels listed in Table 4-8 are less than 0.015”, the overall POD value (95% confidence level) for CVM crack detection was calculated from equation (4.1) as 0.023”. The K values correspond to the desired ɣ (confidence level) of 95%.

Test Scenarios:

Material	Thickness	Coating
2024-T3	0.040”	bare
2024-T3	0.040”	primer
2024-T3	0.071”	primer
2024-T3	0.100”	bare
2024-T3	0.100”	primer
7075-T6	0.040”	primer
7075-T6	0.071”	primer
7075-T6	0.100”	primer

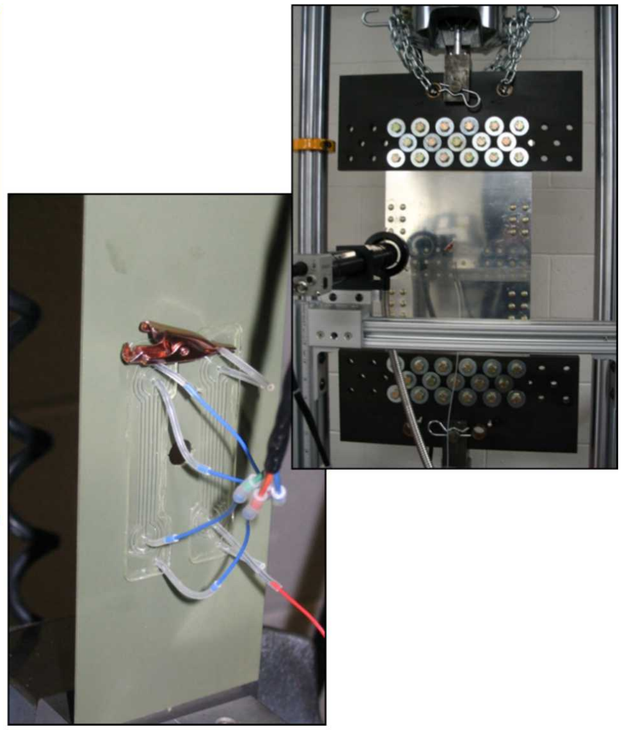


Table 4-7: Test Matrix to Quantify CVM Probability of Crack Detection for Different Structural Configuration

CVM Crack Detection Data (0.040” th)

Bare Metal		Over Primer	
Flaw size (inch)	Log (flaw size)	Flaw size (inch)	Log (flaw size)
0.003	-2.52	0.002	-2.70
0.007	-2.15	0.007	-2.15
0.002	-2.70	0.010	-2.00
0.030	-1.52	0.009	-2.05
0.009	-2.05	0.004	-2.40
0.005	-2.30	0.006	-2.22
0.004	-2.40	0.010	-2.00
0.002	-2.70	0.009	-2.05
0.014	-1.85	0.011	-1.96
0.005	-2.30	0.007	-2.15
0.013	-1.89		
0.032	-1.49		

Statistic Estimates on Log Scale

Statistic	Over Bare metal	Over Primer
Mean	-2.1566	-2.1679
Stnd deviation	0.40889	0.22809

POD Detection Levels
(ɣ = 95%, n = 12 for bare, n=10 for primer)

Detection level $(1-\alpha)$	$K_{n,0.95,\alpha}$		$\bar{X}+K_{n,0.95,\alpha}\cdot S$ (log scale)		Flaw size in inches	
	bare	primer	bare	primer	bare	primer
0.75	1.366	1.465	-1.598	-1.834	0.025	0.015
0.90	2.210	2.355	-1.253	-1.631	0.056	0.023
0.95	2.736	2.911	-1.038	-1.504	0.092	0.031
0.99	3.747	3.981	-0.624	-1.260	0.237	0.055
0.999	4.900	5.203	-0.153	-0.981	0.703	0.104

Table 4-8: Sample POD Calculations Using the One-Sided Tolerance Interval Method and Data Acquired from CVM Sensors Monitoring Cracks in Bare 2024-T3 Aluminum

With the same parameters described above, the maximum likelihood estimate describing the upper bound or optimal performance on the Probability of Detection for the OSTI approach can be calculated as:

$$\text{POD}_{\text{(Max Likelihood Est)}} = \frac{1}{xS\sqrt{2\pi}}\, EXP\left(\frac{-(ln(x)-\bar{X})^2}{2S^2}\right) \tag{4.4}$$

The maximum likelihood estimated POD function, representing the optimum performance for CVM crack detection, was calculated from equation (4.4) and is plotted alongside the 95% confidence bound (cumulative distribution function) in Figure 4-21. As the number of data points increases, the K value will decrease and the POD numbers could also decrease. In this particular instance, it was desired to achieve crack detection before the crack reached 0.1” in length so this goal was achieved.

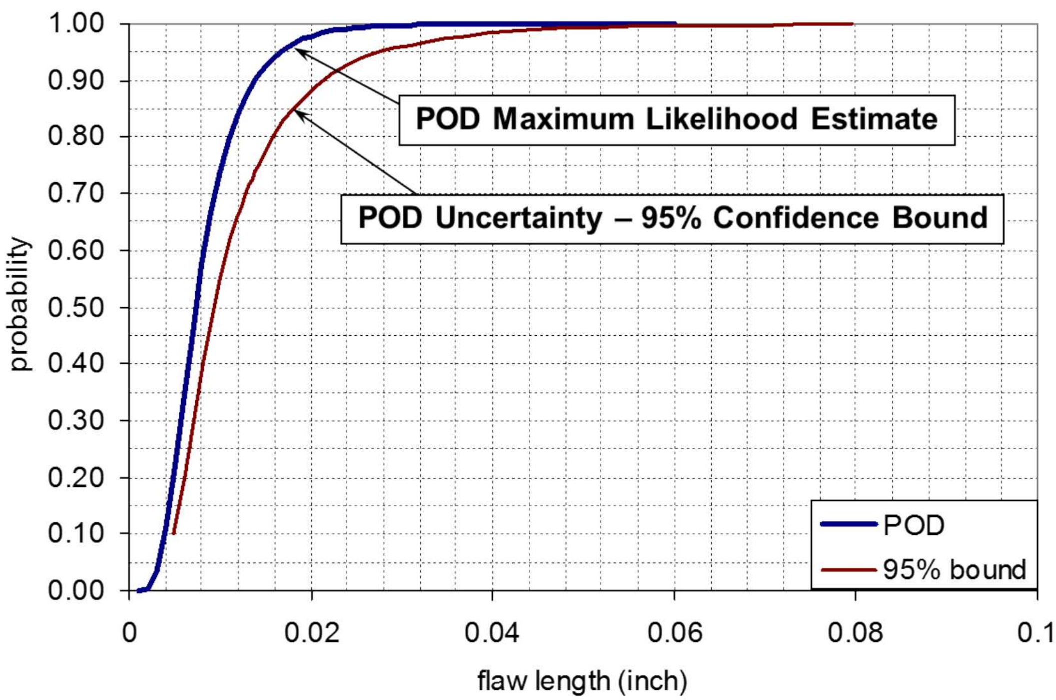


Figure 4-21: Sample Probability of Detection Curves for CVM Comparing the POD Maximum Likelihood Estimate with the POD Estimated with the OSTI Cumulative Distribution Function

Table 4-9, Table 4-10 and Table 4-11 provide the crack detection results for 2024-T3 aluminum skin at thicknesses of 0.040”, 0.71” and 0.1”, respectively. The crack lengths listed in these tables correspond to the crack length into the CVM gallery. In order to produce the total crack length at CVM detection, the distance from the crack origin to the near-side sensor gallery (Gallery 1) must be added to this value as shown in Figure 4-22. The final set of $POD_{(90/95)}$

values for 2024-T3 material at different thicknesses is shown in Figure 4-23. The construction lines indicate the required $POD_{(90/95)}$ will be equal to or better (less) than the required 0.1" crack length for all skins up to a thickness of 0.090". This encompasses most of the fuselage skin.

Description: 0.040 inch thick panel (primer surface)

PHASE 2 TESTS						
Panel	Fastener Crack Site	Distance from Fastener (inches)	Crack Length at CVM Detection (growth after install in inches)	SIM-8 Reading ΔPa (Pasm)	PM-4 Read-out	PM-4 Indicate Crack (Y or N)
4018	5R	0.040	0.002	400-500	1607	Y
4018	6R	0.014	0.007	1700-1800	2847	Y
4018	7R	0.040	0.010	400-500	1704	Y
4018	5R(2)	0.050	0.009	1700-1800	2768	Y
4018	6L	0.052	0.004	1000-1100	2161	Y
407	7L	0.118	0.006	3758-3786	4790	Y
407	5L	0.125	0.010	654-695	1769	Y
407	7R	0.147	0.009	345-375	1426	Y
407	5R	0.139	0.011	374-409	1391	Y
4018	6L	0.194	0.007	530-560	1628	Y
4018	5L	0.253	0.006	380-430	1553	Y
4018	8R	0.262	0.011	320-360	1452	Y
407	6R	0.189	0.012	450-510	1661	Y

90% POD Level	False Calls
0.021"	0

Table 4-9: Summary of CVM Crack Detection and Overall $POD_{(90/95)}$ for a 0.040" Thick, 2024-T3 Skin

Description: 0.071 inch thick panel (primer surface)

PHASE 3 TESTS				
Panel	Sensor	Crack Length at CVM Detection (growth after install in inches)	PM-4 Read-out	PM-4 Indicate Crack (Y or N)
1	1-R	0.043	1507	Y
1	1-L	0.019	1535	Y
1	2-L	0.020	1639	Y
1	2-R	0.021	1673	Y
1	3-L	0.019	2332	Y
1	3-R	0.007	1469	Y
2	1-R	0.015	1335	Y
2	1-L	0.007	1441	Y
2	2-L	0.009	1526	Y
2	2-R	0.012	1424	Y
2	3-L	0.009	1390	Y
2	3-R	0.012	1311	Y
3	1-L	0.035	1339	Y
3	1-R	0.015	1376	Y
3	1-L	0.012	1388	Y
3	1-R	0.008	3405	Y

90% POD Level	False Calls
0.0423"	0

Table 4-10: Summary of CVM Crack Detection and Overall POD$_{(90/95)}$ for a 0.071" Thick, 2024-T3 Skin

Description: 0.100 inch thick panel (primer surface)

PHASE 2 TESTS						
Panel	Fastener Crack Site	Distance from Fastener (inches)	Crack Length at CVM Detection (growth after install in inches)	SIM-8 Reading ΔPa (Pasm)	PM-4 Read-out	PM-4 Indicate Crack (Y or N)
1001	5L	0.350	0.065	773-825	1713	Y
1001	7R	0.206	0.054	697-722	1768	Y
1001	8R	0.115	0.060	560-600	1609	Y
1003	8L	0.044	0.068	297-320	1410	Y
1003	7L	0.086	0.058	342-386	1411	Y
1003	8L	0.187	0.069	~1800	3391	Y
1003	6L	0.061	0.065	476-500	1846	Y
1003	6L	0.131	0.076	800-946	2117	Y
1003	8R	0.160	0.045	380-420	1508	Y

90% POD Level	False Calls
0.090"	0

Table 4-11: Summary of CVM Crack Detection and Overall POD$_{(90/95)}$ for a 0.100" Thick, 2024-T3 Skin

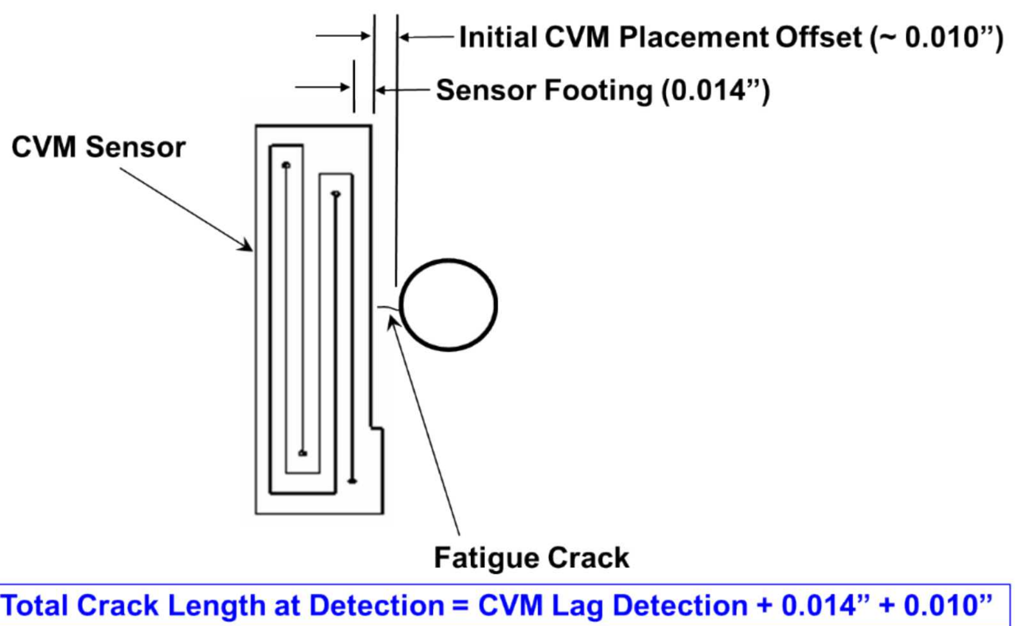


Figure 4-22: Determining Final CVM Crack Detection Level from Crack "Lag" Values (Crack Excursion into Gallery 1)

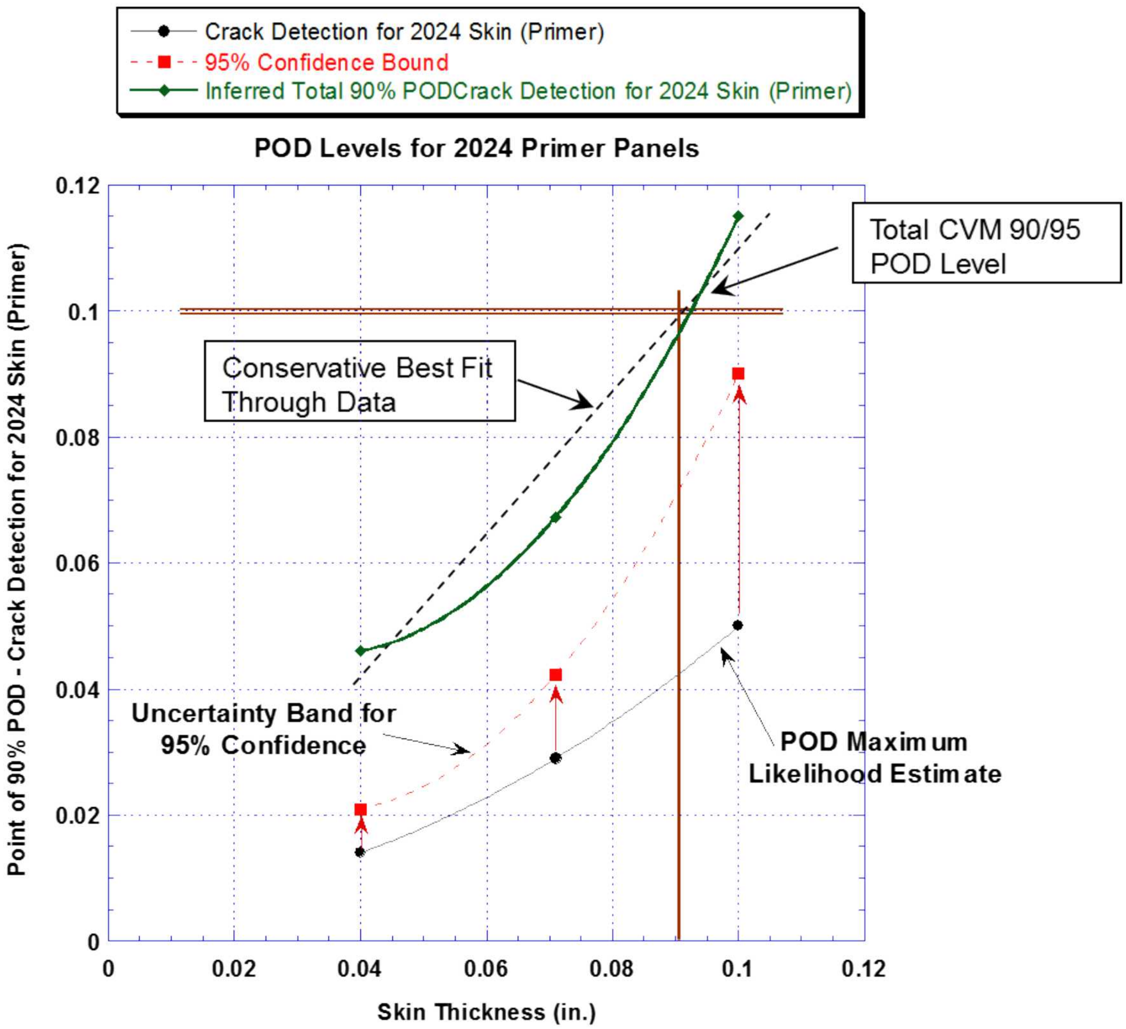


Figure 4-23: Overall CVM $POD_{(90/95)}$ Values as a Function of 2024-T3 Material Thickness

Table 4-12, Table 4-13, and Table 4-14, provide the crack detection results for 7075-T6 primed aluminum skin at thicknesses of 0.040", 0.71" and 0.1", respectively. The crack lengths listed in these tables correspond to the crack length into the CVM gallery. In order to produce the total crack length at CVM detection, the distance from the crack origin to the near-side sensor gallery (Gallery 1) must be added to this value as shown in Figure 4-22. The final set of $POD_{(90/95)}$ values for 7075-T6 material at different thicknesses is shown in Figure 4-24. The construction lines indicate the required $POD_{(90/95)}$ will be equal to or better (less) than the required 0.1" crack length for skin thicknesses in excess of 0.100".

Table 4-15 lists the overall crack detection/sensitivity results for CVM sensors on 2024-T3 and 7075-T6 aluminum skins of different thicknesses and possessing two different surface coatings (none/bare and primer). With respect to reliability and repeatability of the sensor operation, it was noted that, in over 150 fatigue tests conducted using CVM sensors on these various skin structures, there were no false calls produced by the sensors.

Description: 0.040 inch thick panel (primer surface)

PHASE 3 TESTS					
Panel	Fastener Crack Site	Number of Fatigue Cycles	Crack Length at CVM Detection (growth after install in inches)	PM-4 Read-out (Pasm)	PM-4 Indicate Crack (Y or N)
1	1-L	3400	0.009	1738	Y
1	1-R	2400	0.011	1706	Y
1	2-L	6200	0.013	2109	Y
1	2-R	6000	0.014	2415	Y
1	3-L	6702	0.015	2346	Y
1	3-R	6702	0.004	1680	Y
2	1-R	3200	0.010	1611	Y
2	2-R	4850	0.006	1658	Y
2	3-L	5450	0.014	2506	Y
2	3-R	5450	0.018	4058	Y
3	1-L	3725	0.012	1731	Y
3	1-R	2925	0.006	1679	Y
3	2-L	4800	0.004	1833	Y
3	2-R	4600	0.008	1750	Y
3	3-L	5325	0.016	2946	Y
3	3-R	5230	0.005	2150	Y

90% POD Level	False Calls
0.0255"	0

Table 4-12: Summary of CVM Crack Detection and Overall $POD_{(90/95)}$ for a 0.040" Thick, 7075-T6 Skin

Description: 0.071 inch thick panel (primer surface)

PHASE 3 TESTS					
Panel	Fastener Crack Site	Number of Fatigue Cycles	Crack Length at CVM Detection (growth after install in inches)	PM-4 Read-out (Pasm)	PM-4 Indicate Crack (Y or N)
1	1-L	2600	0.008	1439	Y
1	1-R	2500	0.007	1341	Y
1	2-L	4100	0.014	1411	Y
1	2-R	3900	0.011	1484	Y
2	1-L	3800	0.012	1825	Y
2	1-R	3500	0.017	2056	Y
2	2-L	4800	0.003	2618	Y
2	2-R	5000	0.005	2634	Y
2	3-L	5900	0.007	4142	Y
2	3-R	6100	0.003	6012	Y
4	1-L	3500	0.004	1589	Y
4	1-R	3400	0.013	1706	Y
4	2-L	5600	0.007	3035	Y
4	2-R	5600	0.027	2734	Y
4	3-L	6400	0.003	2778	Y
4	3-R	6400	0.020	11380	Y

90% POD Level	False Calls
0.033"	0

Table 4-13: Summary of CVM Crack Detection and Overall $POD_{(90/95)}$ for a 0.071" Thick, 7075-T6 Skin

Description: 0.100 inch thick panel (primer surface)

PHASE 3 TESTS								
Panel	Fastener Crack Site	Number of Fatigue Cycles	Crack Length at CVM Detection (growth after install in inches)	PM-4 Read-out (Pasm)	PM-4 Indicate Crack (Y or N)		90% POD Level	False Calls
1	1-L	3505	0.007	2123	Y		0.023"	0
1	1-R	3205	0.007	1938	Y			
1	2-L	5350	0.010	2251	Y			
1	2-R	5550	0.011	1954	Y			
1	3-L	6650	0.009	4526	Y			
1	3-R	7099	0.016	7099	Y			
2	1-L	3100	0.011	1786	Y			
2	1-R	3400	0.014	1707	Y			
2	2-L	5300	0.005	2383	Y			
2	2-R	5300	0.016	2204	Y			
3	1-L	4475	0.019	1790	Y			
3	1-R	4825	0.013	1904	Y			
3	2-L	7025	0.008	2100	Y			
3	2-R	7878	0.010	4302	Y			

Table 4-14: Summary of CVM Crack Detection and Overall $POD_{(90/95)}$ for a 0.100" Thick, 7075-T6 Skin

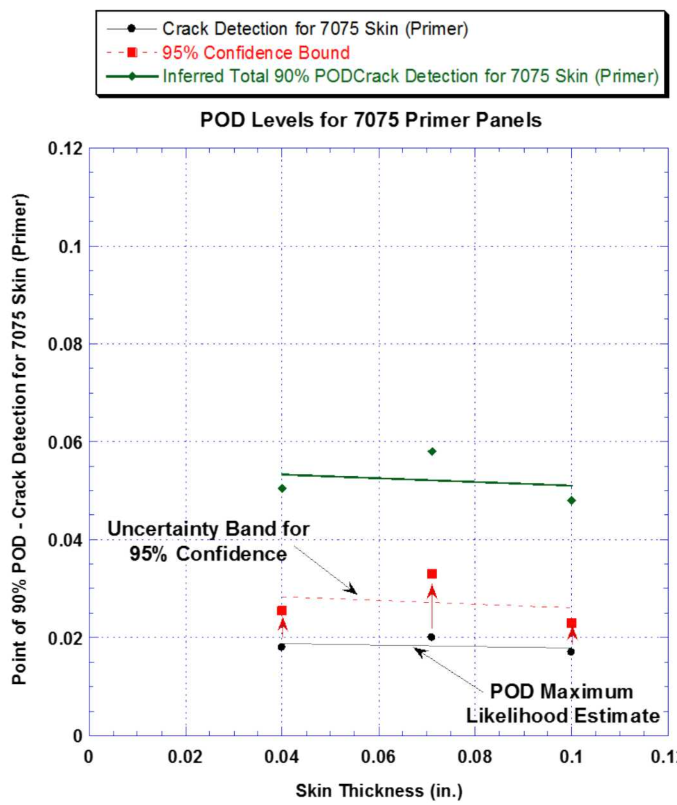


Figure 4-24: Overall CVM $POD_{(90/95)}$ Values as a Function of 7075-T6 Material Thickness

Material	Plate Thickness (mm)	Coating	90% POD for Crack Detection (mm)
2024-T3	1.02	Bare	1.24
2024-T3	1.02	Primer	0.53
2024-T3	1.80	Primer	1.07
2024-T3	2.54	Bare	6.91
2024-T3	2.54	Primer	2.29
7075-T6	1.02	Primer	0.66
7075-T6	1.8	Primer	0.84
7075-T6	2.54	Primer	0.58

Table 4-15: Summary of Crack POD Levels for CVM Deployed on Different Materials, Surface Coatings, and Plate Thicknesses

The end result of this test series is that curves shown in Figure 4-23 and Figure 4-24, along with the successful flight testing described in Section 6.1, were used to establish the overall capability of CVM sensors allowed CVM technology to be included in Boeing's NDT "tool box" as a viable crack detection methodology. The testing establish the ability of CVM sensors to detect cracks in fuselage skin structure and to determine the limits on skin thickness applications such that a crack of 0.10" length could be reliably detected. For 2024-T3 material, the allowable skin thickness for the 90/95 POD level of 0.1" was up to 0.09" thick. For 7075-T6 material, the allowable skin thickness for the 90/95 POD level of 0.1" was up to 0.10" thick. In the 2005-2006 time frame, Boeing's NDT Standard Practices Manual was revised to include CVM sensors as a possible structural monitoring option.

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CHAPTER 5

5.0 CVM System Durability Results

5.1 Extreme Environment Cycling

In addition to the crack detection performance data, this program also conducted tests to evaluate the environmental durability of the CVM system. It is an indispensable step to carry out validation tests for any SHM systems under operational environments before it becomes an application-ready product. This testing is meant to establish the durability level of the sensors so that Delta could ensure that it was proposing something that will sustain operations over a long period of time and not be a major inconvenience to its users. The environmental tests were part of the overall performance testing. The team also compiled data from the flight test program to use as proof of successful sensor function in an actual aircraft operating environment. This is important information to add to the laboratory results for any certification package being reviewed by regulatory authorities approving CVM sensor use on aircraft.

When considering overall durability assessments, it is important to make sure that all operating conditions that may affect SHM system response are properly included in the test program. This has been recommended in numerous SHM performance processes and demonstrated in studies on specific SHM sensor systems [5.1 – 5.4] Structural health monitoring systems often experience harsh environments which can, even in the absence of damage, create varying, nonlinear and nonstationary behaviors. These response changes must be understood and either mitigated or incorporated into any damage detection algorithms to avoid any reduction in the performance of the SHM system.

The overall goal is to assess the topics of durability, reliability, and longevity and to develop and apply a suitable criteria to properly assess SHM system performance in representative operating environments. Environmental tests may include, for example, temperature extremes, humidity, fluid susceptibility, altitude, mechanical connections, structural strain and component vibration. Application of these environments may be static or cyclic if fatigue response is an important consideration. A criteria to assess particular changes in sensor response, which involves pre- vs. post-test and intermittent measurements, are useful in assessing the SHM system's performance.

Durability testing of CVM has been addressed in a number of studies [5.4 - 5.5]. This program completed its own set of tests to comprehensively and independently add to this referenced database and arrive at a proper conclusion about the operation of CVM sensors over long periods of time. There are existing standards that address testing for the durability of commercial and military aircraft components and these were utilized in the CVM durability testing [5.6]. The test plan is summarized in Figure 5-1 while the test specimens are described in Figure 5-2 through Figure 5-4. The environmental conditioning tests consisted of the following elements:

1) Hot-Wet Conditioning (55°C ± 3°C and 95% ± 3% RH) - 28 days, monitor every 7 days.
2) Cold/Freeze/Icing - (8 hours @ -18°C) followed by monitoring after each freeze cycle.
3) Heat Exposure (8 hours @ 74°C) - followed by monitoring after each extreme heat exposure.

- Part of overall performance testing - meant to establish durability of sensor systems
- Sensor fail-safe feature is critical item – will be proven
- Temperature environment selected to match similar testing to certify metal primer
- Environmental elements:
  - Hot-Wet (7 days @ 60°C and 95% ±3% RH)
  - Freeze (8 hours @ -18°C)
  - Heat (8 hours @ 74°C)

    Three components equal one total ENV cycle
  - Environmental elements repeated 4 times (total of 28 day hot-wet environment and 36 day minimum total ENV TEST)
- CVM Sensor function measurements were acquired before each overall cycle, at the end of each cold exposure component and after total ENV test completion (total of 9 CVM monitoring events)
- Test specimens include all hardware that remains on the aircraft during flight operations

Figure 5-1: Test Plan for Environmental Durability Performance Assessment

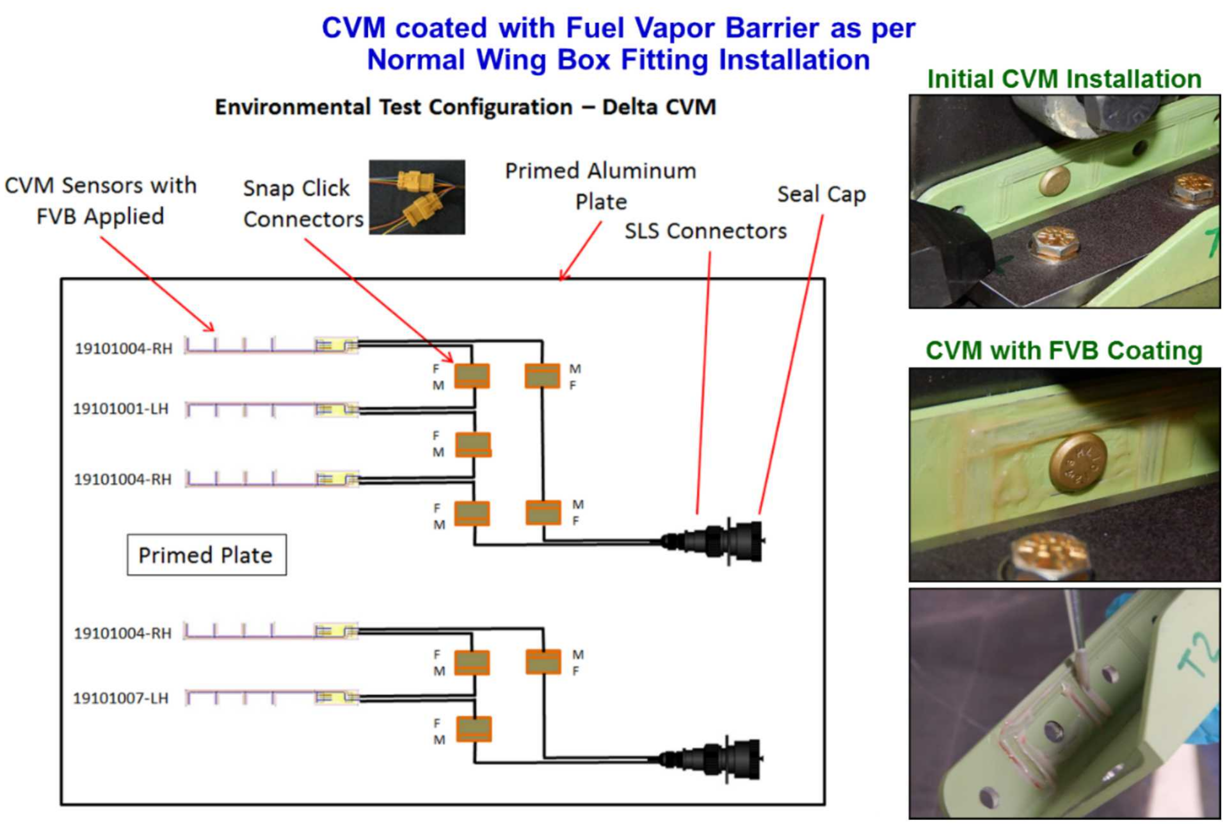


Figure 5-2: Coating CVM Sensors on Environmental Test Specimens

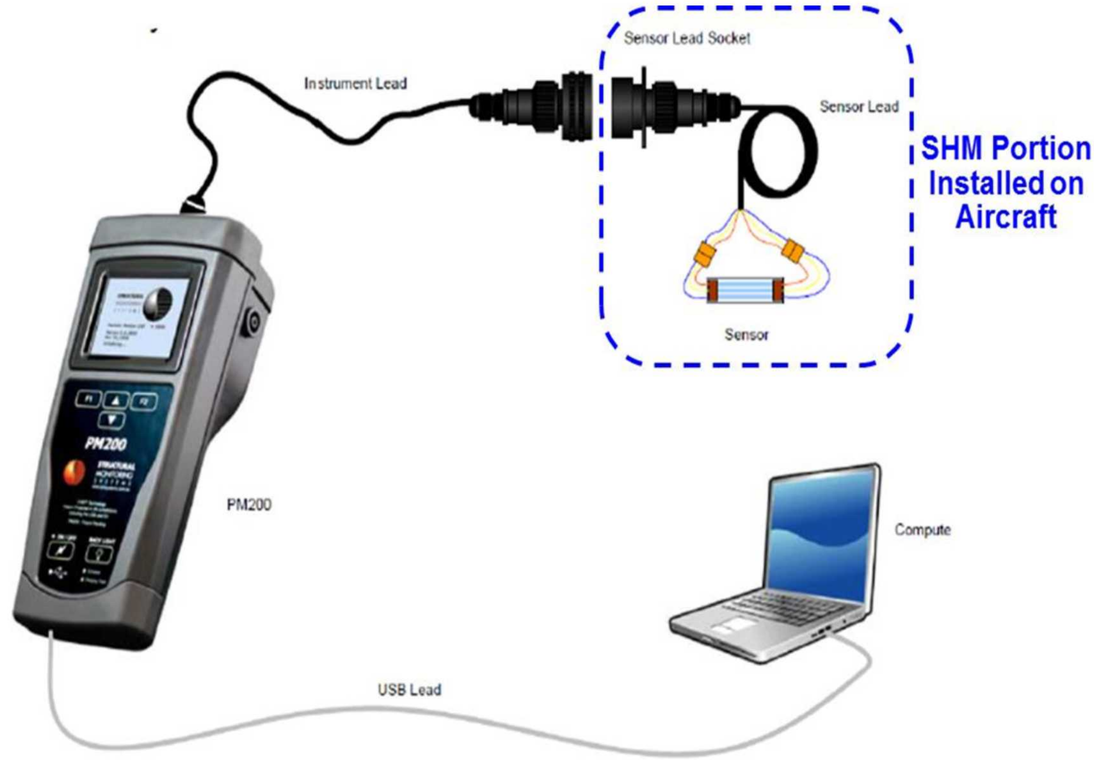


Figure 5-3: Schematic of Full CVM System Highlighting the Portion Subjected to Durability Testing and Subsequent Monitoring

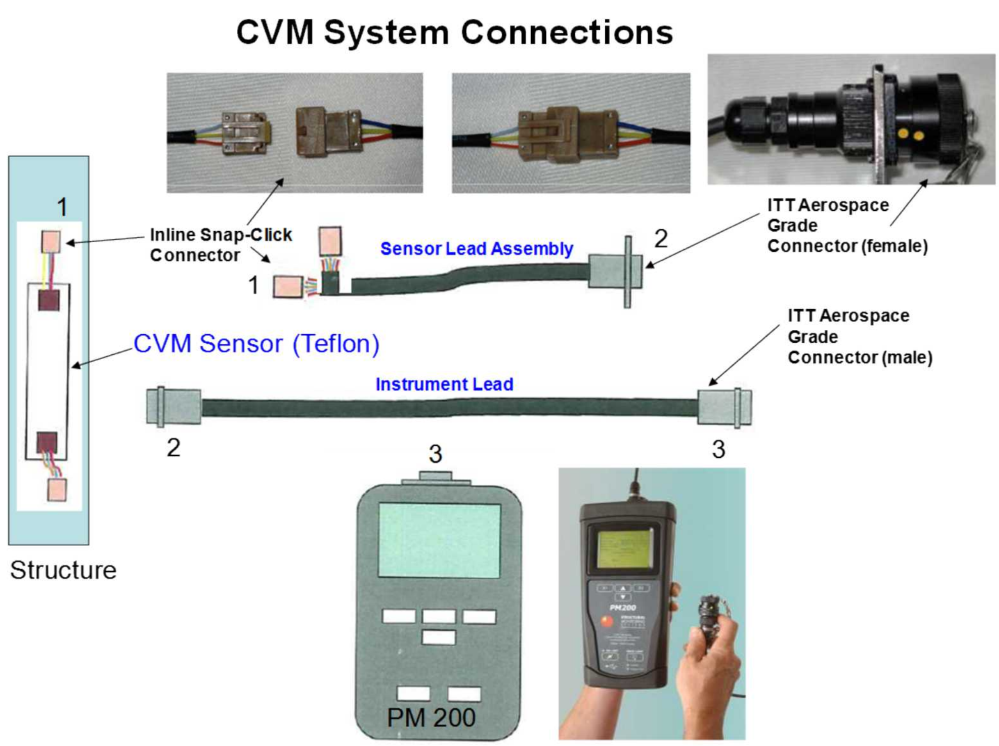


Figure 5-4: CVM System Connections

Figure 5-5 depicts all of the elements of the temperature and humidity environments along with the data acquisition points for each 9-10 day cycle. Each hot-cold-wet cycle was repeated four times to produce the full 28 days of hot-wet conditioning used in normal environmental tests. The minimum and maximum temperatures correspond to, or exceed, the DO-160 environment used to certify primer materials [5.6].

Figure 5-3 captures the components involved in the environmental testing. Complete connection routing showing CVM sensors, Sensor Lead, Instrument Lead and Snap-Click connectors to connect CVM sensors to data acquisition equipment, data analysis and logging. Figure 5-4 and Figure 5-6 focus on the CVM connection method where Snap-Click connectors are used to connect each CVM sensor to the CVM Sensor Leads (see items 1 and 2 which are mounted on-aircraft). Then an instrument lead is used to link the on-aircraft, aerospace grade connector to the PM-200 device for sensor monitoring.

Five sensors, arranged into groups as they normally are on an operating aircraft, were placed on a single panel as shown in Figure 5-7 through Figure 5-9. They were connected in a daisy-chain fashion and to the SLS on-board connector as shown in Figure 5-6.

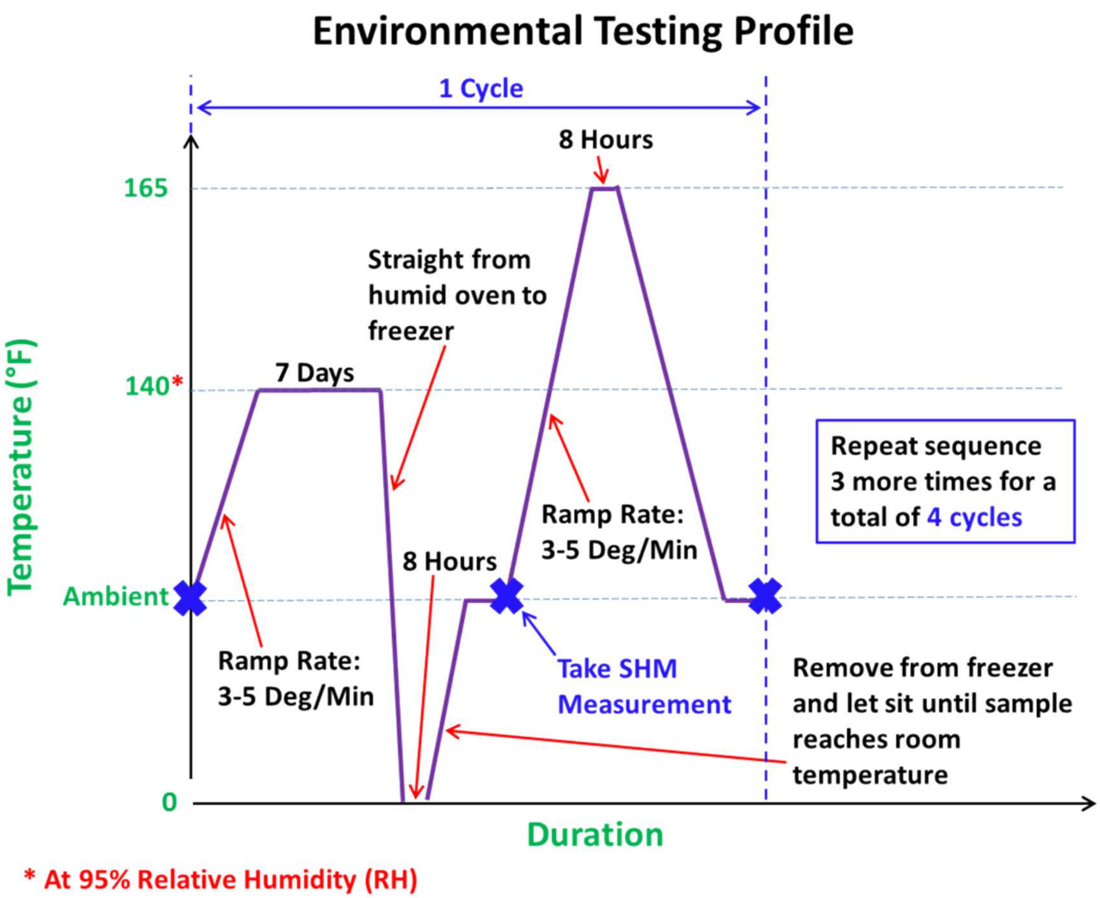


Figure 5-5: Description of Cyclic Environmental Extremes for CVM Durability Tests

The sensors were subjected to the environmental test environment shown in Figure 5-5. First, baseline continuity and dCVM values were acquired (see Table 5-1) to ensure suitable sensor installation and to establish data for future comparisons after ENV exposure. Figure 5-10 through Figure 5-12 show placement of the sensors into the environmental chambers while Figure 5-13 through Figure 5-18 show the resulting environment as measured by calibrated thermocouples and humidity transducers.

Sensor response measurements were made after each of the three environments listed above (hot-wet, cold, heat) and this process was repeated for a total of four cycles. The tests evaluated sensor ability to function after severe exposure to humidity, temperature variations, icing/freezing and heat.

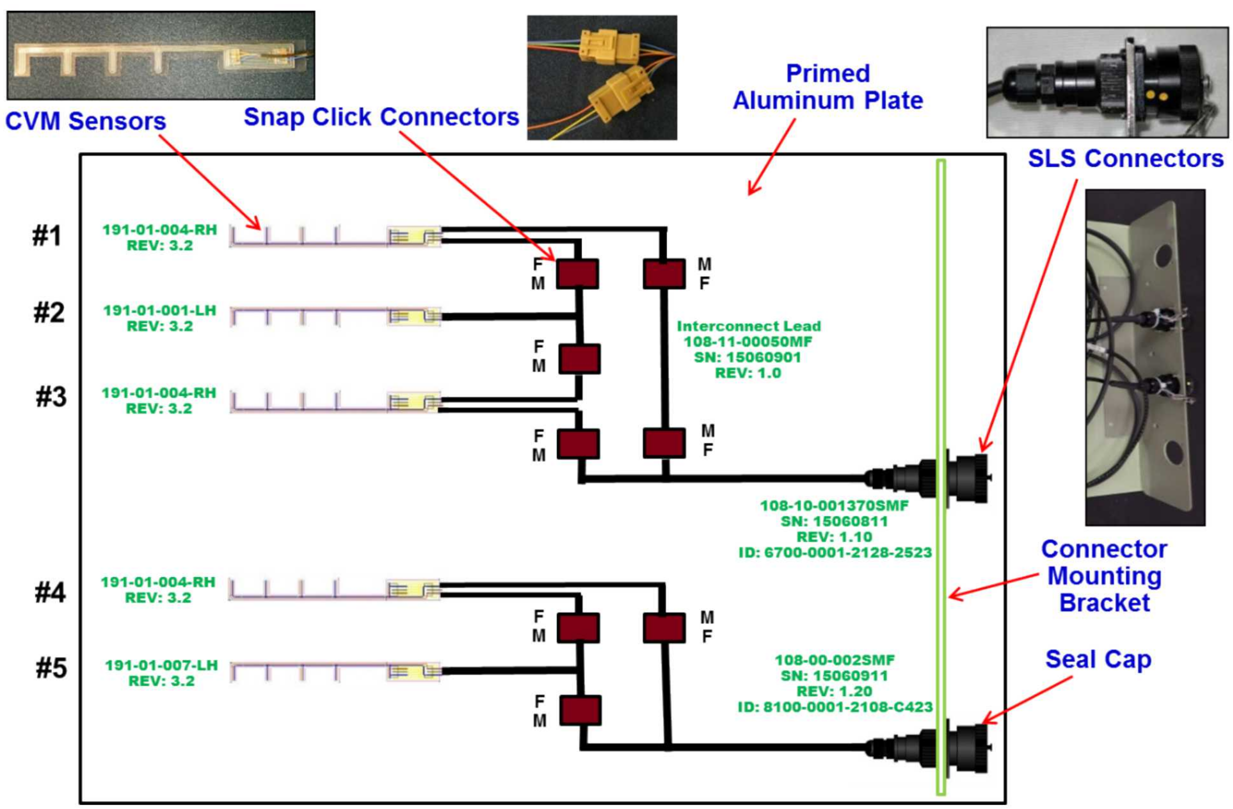


Figure 5-6: Multi- Sensor Test Specimen Configuration for Environmental Durability Performance Tests

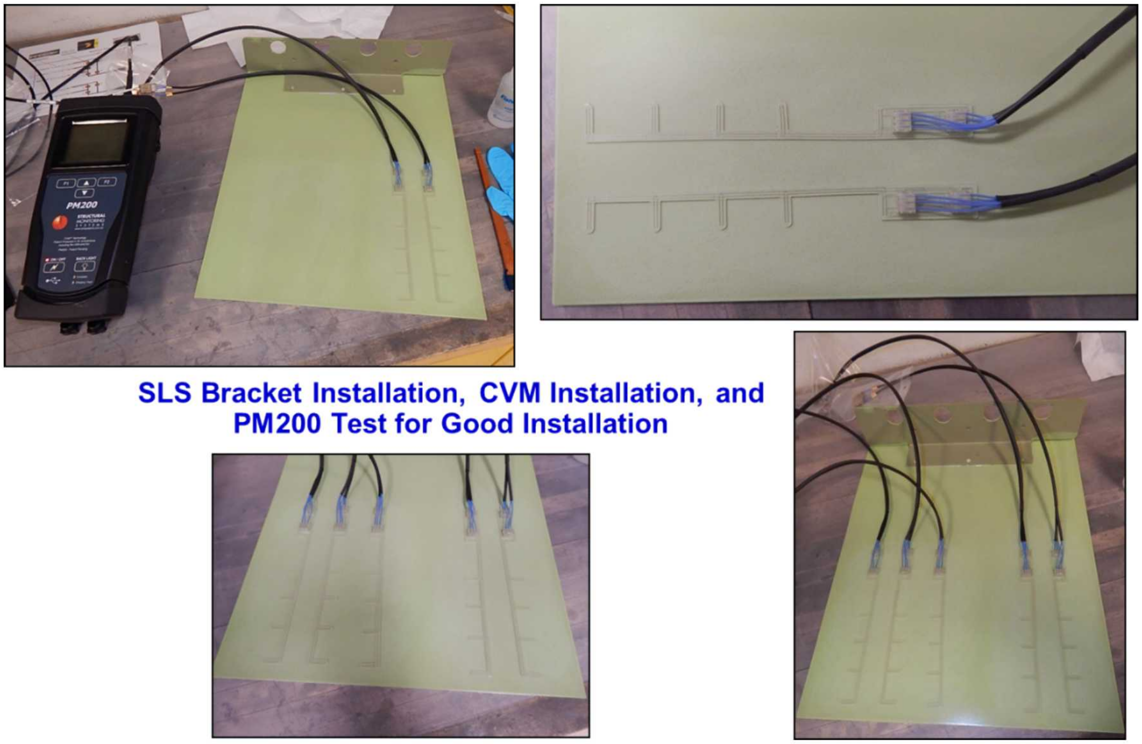


Figure 5-7: Installation of Five Wing Box CVM Sensors for Environmental Durability Test

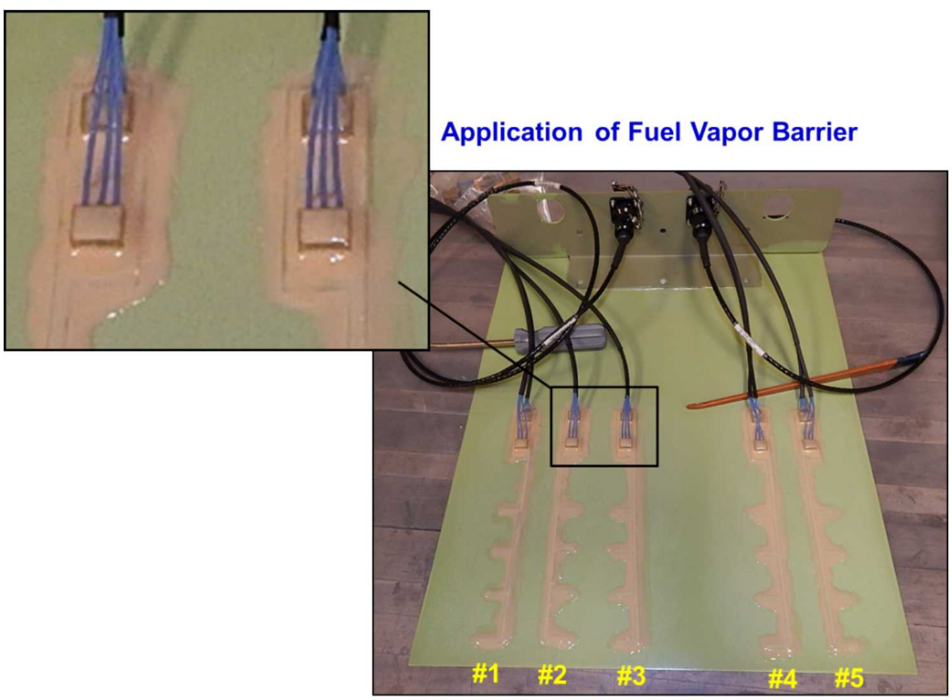


Figure 5-8: Typical Sensor Installation, Fuel Vapor Barrier Coating and Vacuum Tube Cable Connection to SLS Connectors

Figure 5-9: Final Sensor Set-Up into Groupings as those used on Operational Aircraft

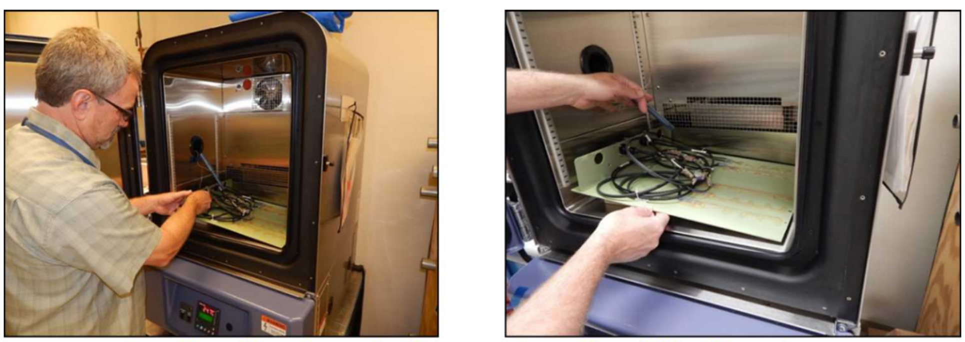
Environmental Tests - Delta Program							
Sensor #	Ccont	1Cont	2Cont	dCVM1	dCVM2	CVM Screen Reading	Notes:
Individual CVM Sensor Readings on PM200 Device After Installation							
1	Max CI	Max CI	16939.0	-0.6	0.7	Pass	Cable ID: AB00-0001-2133-D323
2	16737.0	Max CI	15966.0	-0.7	0.0	Pass	Cable ID: AB00-0001-2133-D323
3	Max CI	17087.0	15190.0	-0.6	-0.1	Pass	Cable ID: AB00-0001-2133-D323
4	Max CI	13546.0	16740.0	-0.4	0.1	Pass	Cable ID: AB00-0001-2133-D323
5	Max CI	13521.0	Max CI	-0.8	-0.1	Pass	Cable ID: AB00-0001-2133-D323
3-2 Grouping of CVM Sensors - Readings on PM200 Device After FVB Coating							
1 2 3	6589.0	6010.0	6715.0	-1.0	1.9	Pass	Cable ID: 6700-0001-2128-2523
4 5	10605.0	10927.0	10431.0	-0.8	-0.1	Pass	Cable ID: 8100-0001-2108-C423

All tests passed – high conductivity (flow rate) and low dCVM (vacuum level) on all sensor sets

Table 5-1: Baseline Data Prior to Environmental Exposure - Initial PM200 Tests on Individual Sensors and Grouping of Sensors as Per Normal 737 Wing Box Installation

- Used to subject CVM sensor system to controlled temperature and humidity environments
- Programming feature provides repeatable conditions with required temperature ramping (rate of increase) and length of soak times

Figure 5-10: Environmental Test Chamber



Loading Specimen in Temperature-Humidity Chamber

Logging Data to Ensure Proper Environment

Figure 5-11: Set-Up of Durability Specimen in Environmental Test Chamber

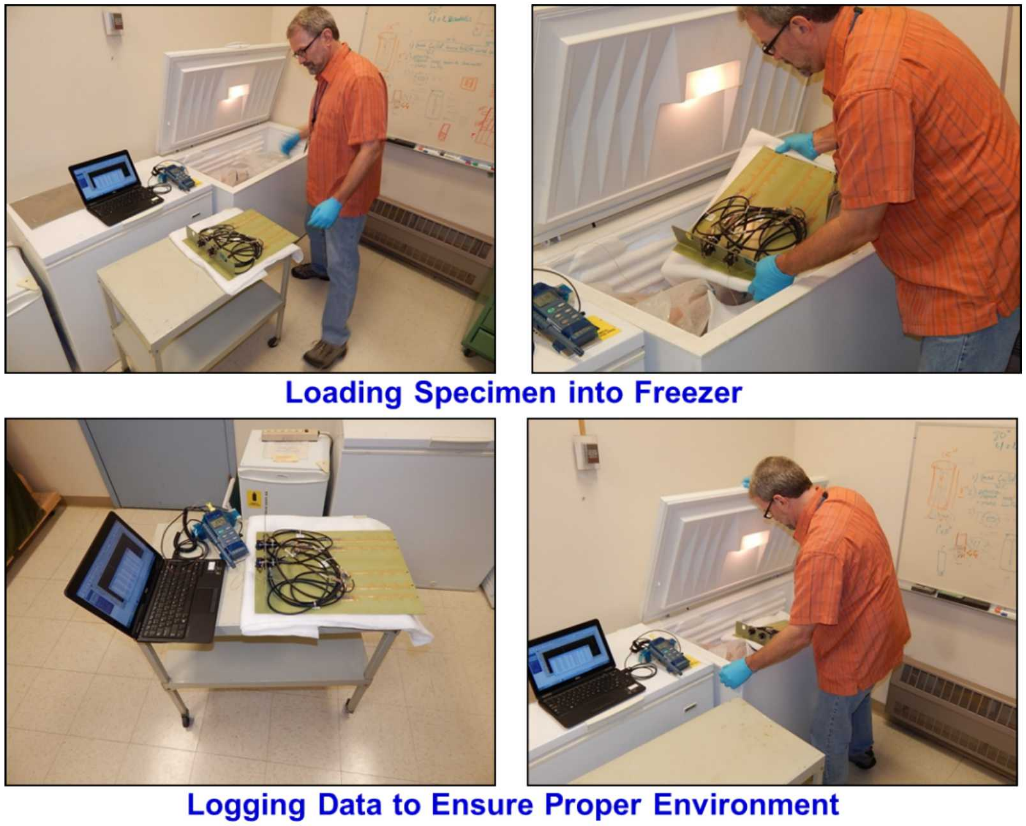


Figure 5-12: Controllable Freezer Used to Expose Sensor Specimen to Extreme Cold Environment

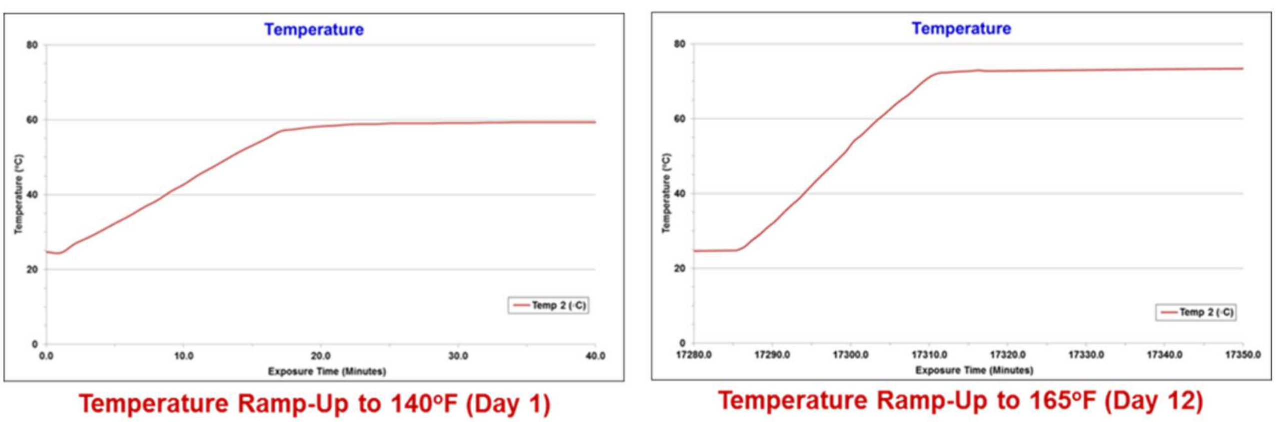


Figure 5-13: Data Plot Showing Ramp-Up of Temperature During Hot-Wet and Extreme Heat Portions of Durability Tests

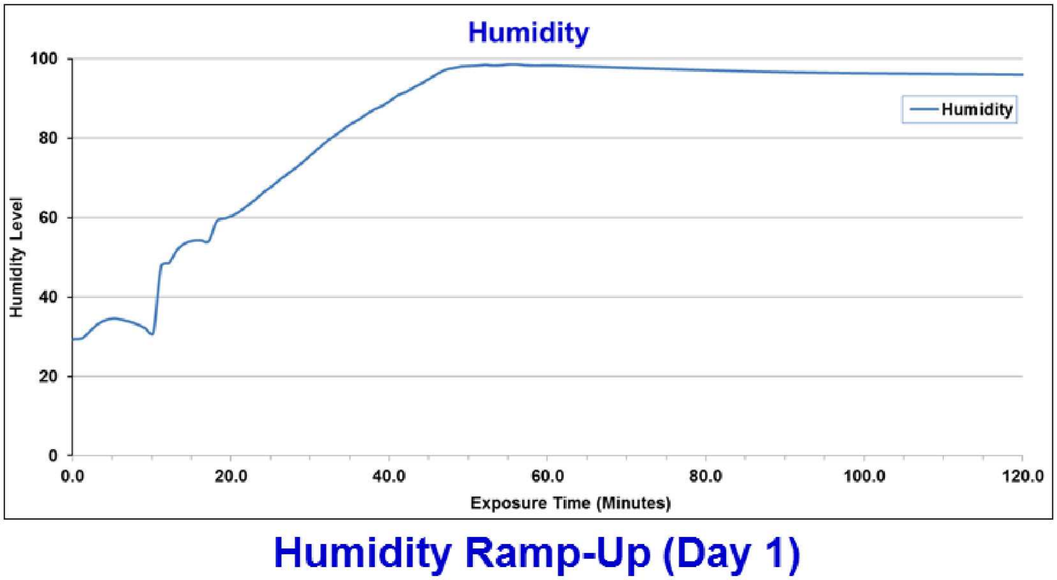


Figure 5-14: Data Plot Showing Ramp-Up of Humidity During Hot-Wet Portion of Durability Tests

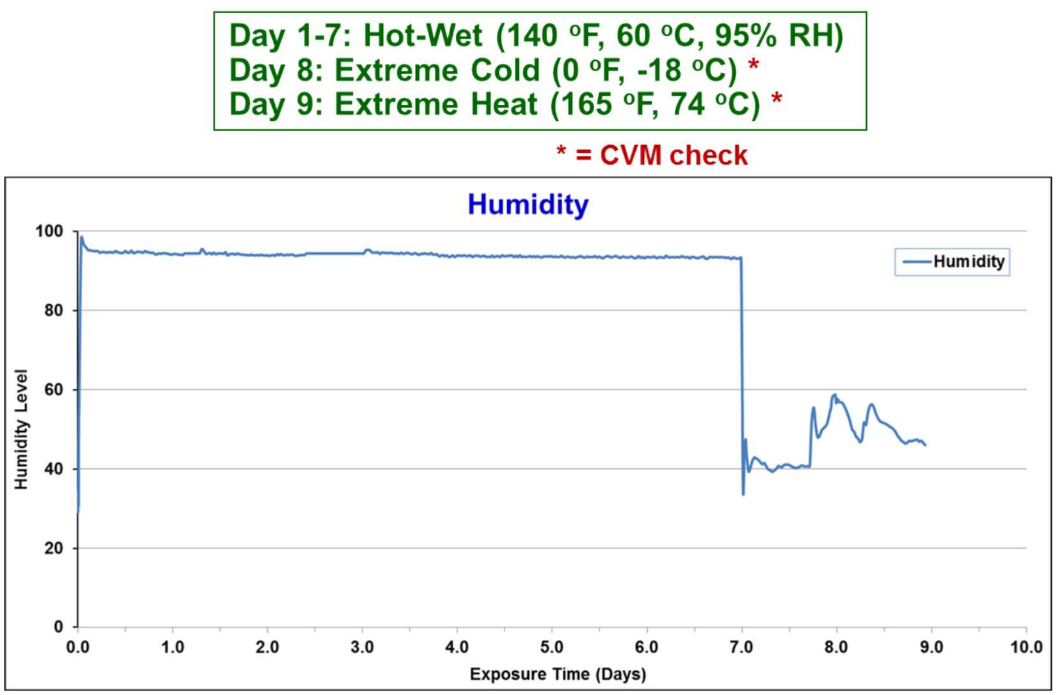


Figure 5-15: View of Consistent Humidity Level During the Hot-Wet Portion of the Durability Tests (Cycle 1)

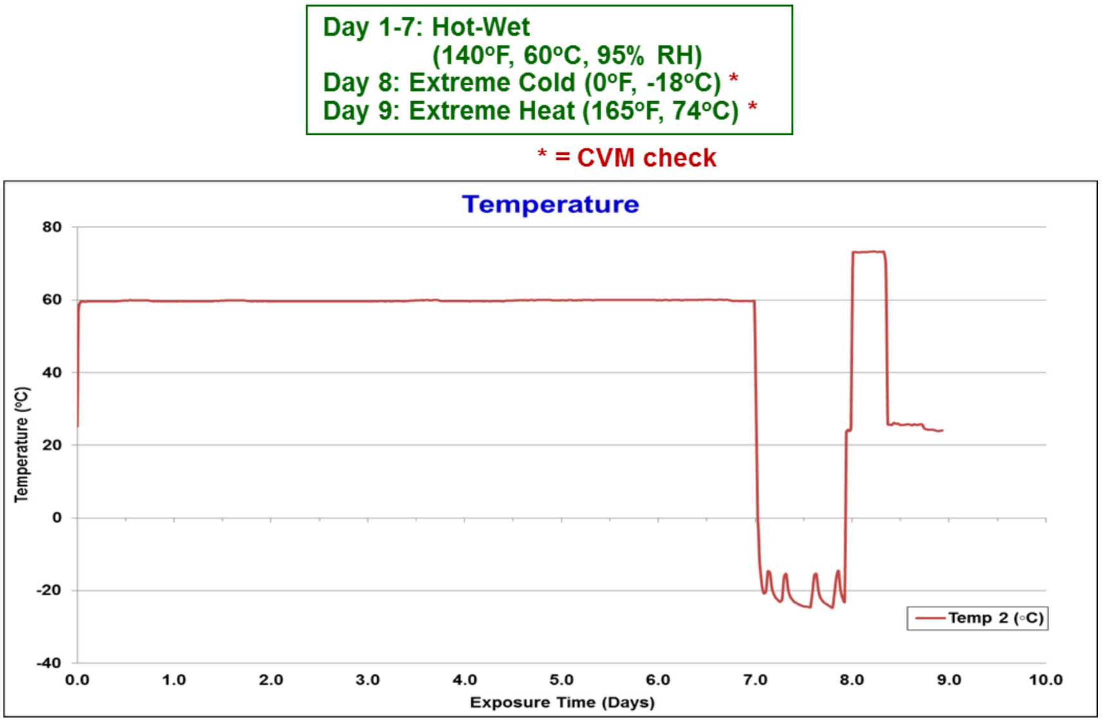


Figure 5-16: View of Consistent Temperature Levels During the Hot-Wet, Freezing and Extreme Heat Portions of Durability Tests (Cycle 1)

The CVM sensors were monitored during the time periods indicated in Figure 5-5. Recall that the sensors were installed on undamaged structure and the status of that structure did not change during the course of the 40 environmental testing. Thus, the optimal results would be for the CVM sensors to function properly and also produce consistently low dCVM values (i.e. no crack detected) over the entire time of the tests. Results from CVM readings during the Environmental Durability tests indicate that:

- Sensor readings during 40 day environmental tests remained small compared to the threshold level required for crack detection (see Figure 5-19)
- dCVM values ranged +/- 2.0 while the crack detection threshold was set for dCVM = 10.0
- Good durability of SHM system; no degradation
- Signal-to-noise (S/N) for crack detection is a minimum of 5 (most exceeded 20 in fatigue tests)
- Desired S/N for normal NDI operations is a minimum of 3.

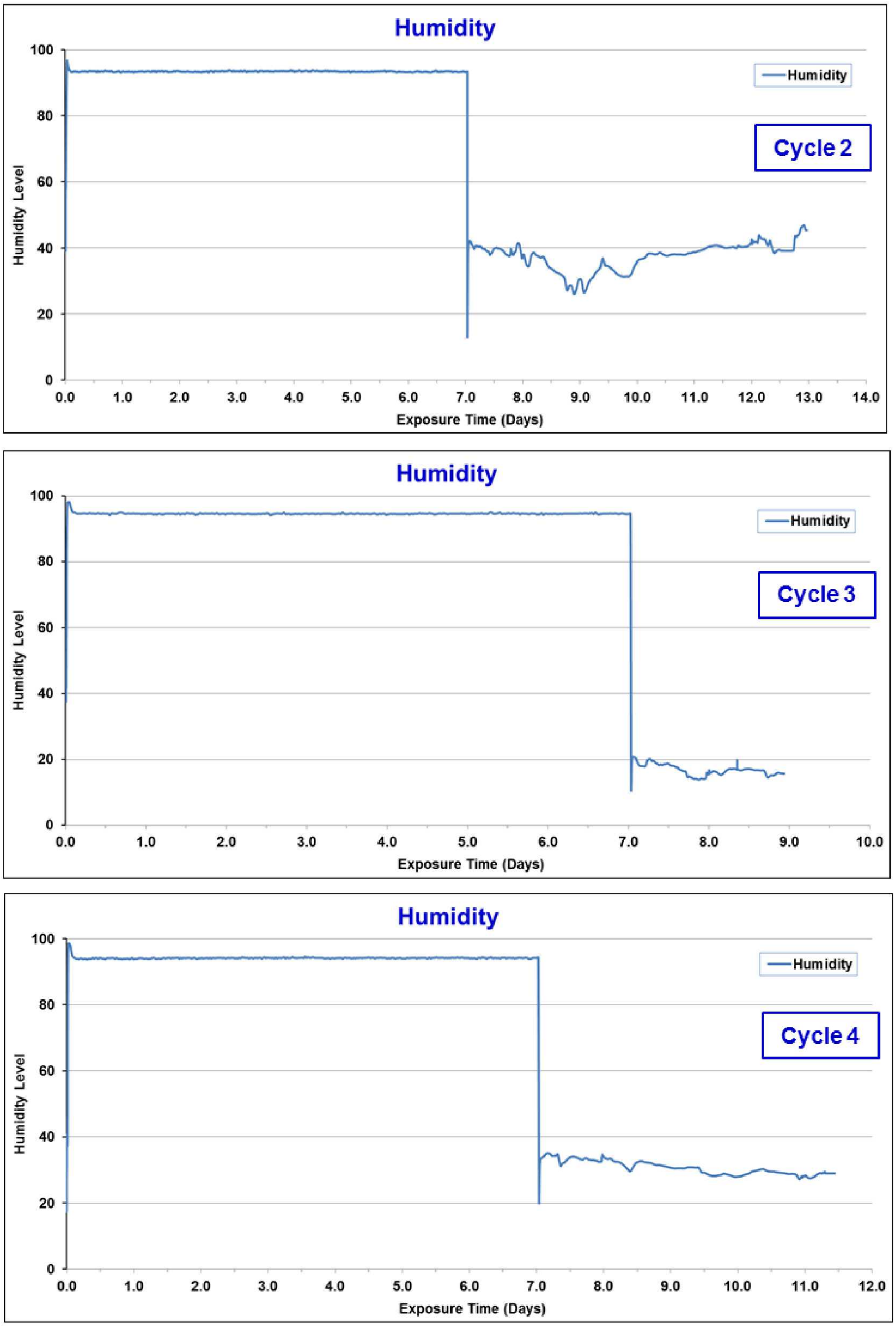


Figure 5-17: Humidity Level During the Hot-Wet Portion of the Durability Tests (Cycles 2, 3, and 4)

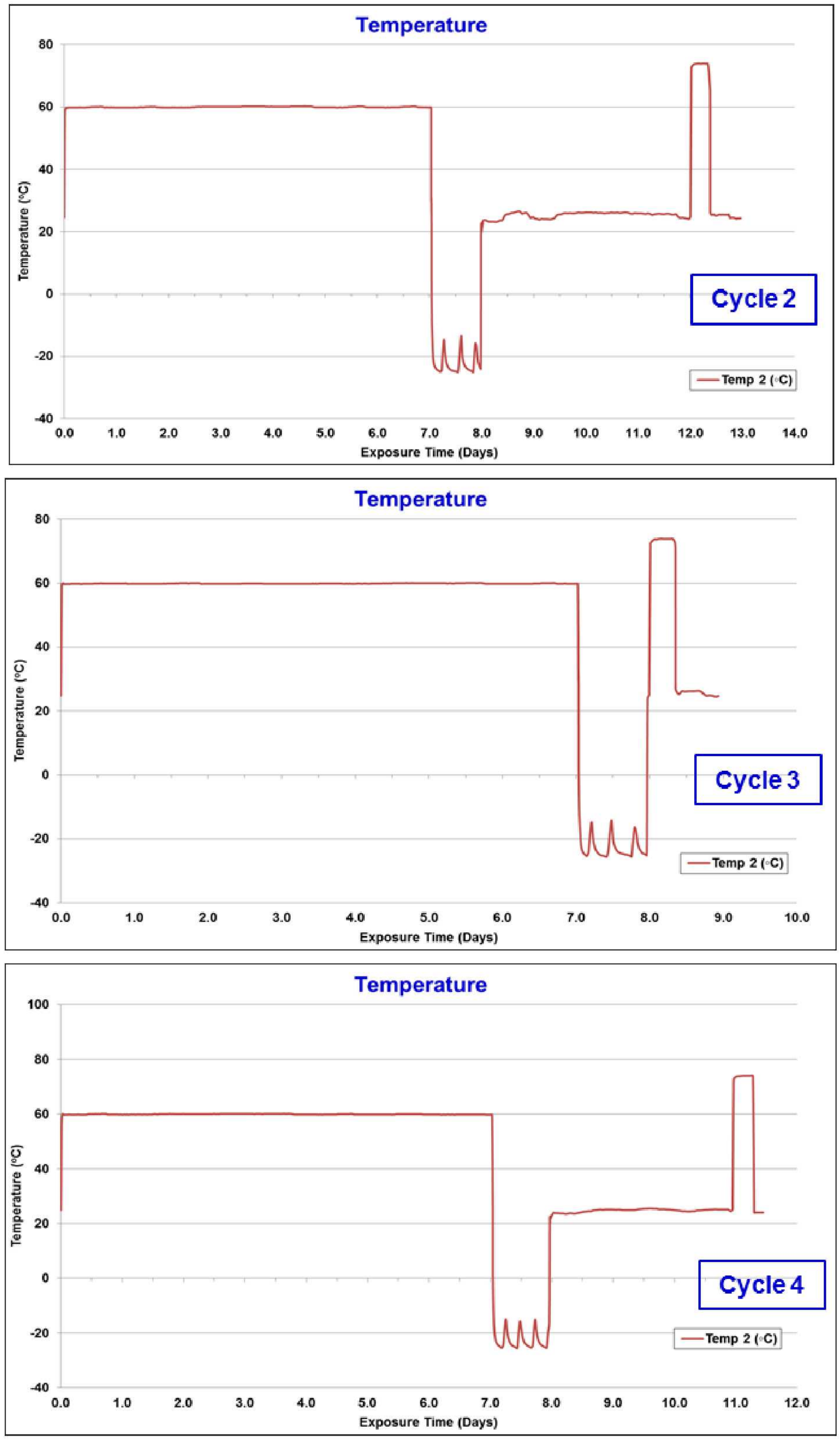


Figure 5-18: Temperature Levels During the Hot-Wet, Freezing and Extreme Heat Portions of Durability Tests (Cycles 2, 3, and 4)

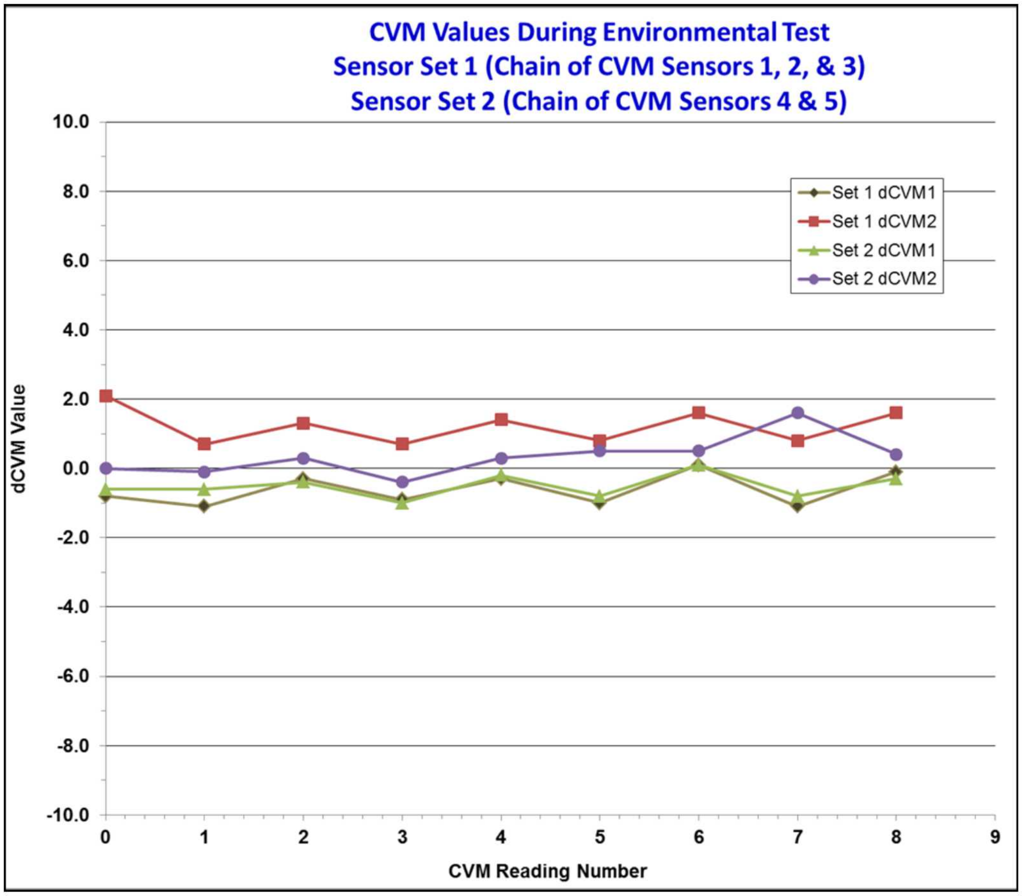


Figure 5-19: CVM Sensor Readings Remain Unchanged During Environmental Tests

Similarly, there should be no change in the status of the galleries over the course of the durability testing. Continuity checks are conducted by the PM-200 device to ensure that each gallery has proper flow and is not blocked or otherwise restricted in any way. This test must be passed before any crack detection readings are acquired. Regardless of the status of the structure (damaged or undamaged), the optimal results would be for the CVM sensors to provide consistently high continuity (flow rate) values over the entire time of the tests. Results from CVM readings during the Environmental Durability tests indicate that:

- Sensor continuity measures for possible gallery blockage. During 40 day environmental tests, continuity remained large indicating proper sensor functioning and no blockage in the galleries (see Figure 5-20).
- Continuity values ranged 6,000 to 12,000; minimum levels allowed were Cont = 2,000.
- Good durability of SHM system; no degradation.

The data above corresponds to the sensor groupings of Set 1= CVM1, CVM2, CVM3 and Set 2= CVM4, CVM5 as they are grouped on the 737 wing box fittings. Data was also acquired to show that the individual sensors maintained consistent dCVM and continuity readings before and after 40 day environmental tests. Figure 5-21 and Figure 5-22 show that there was no change in either the dCVM or continuity values and thus, no effect of 4 cycles of extreme hot-wet-cold-heat environment on CVM performance.

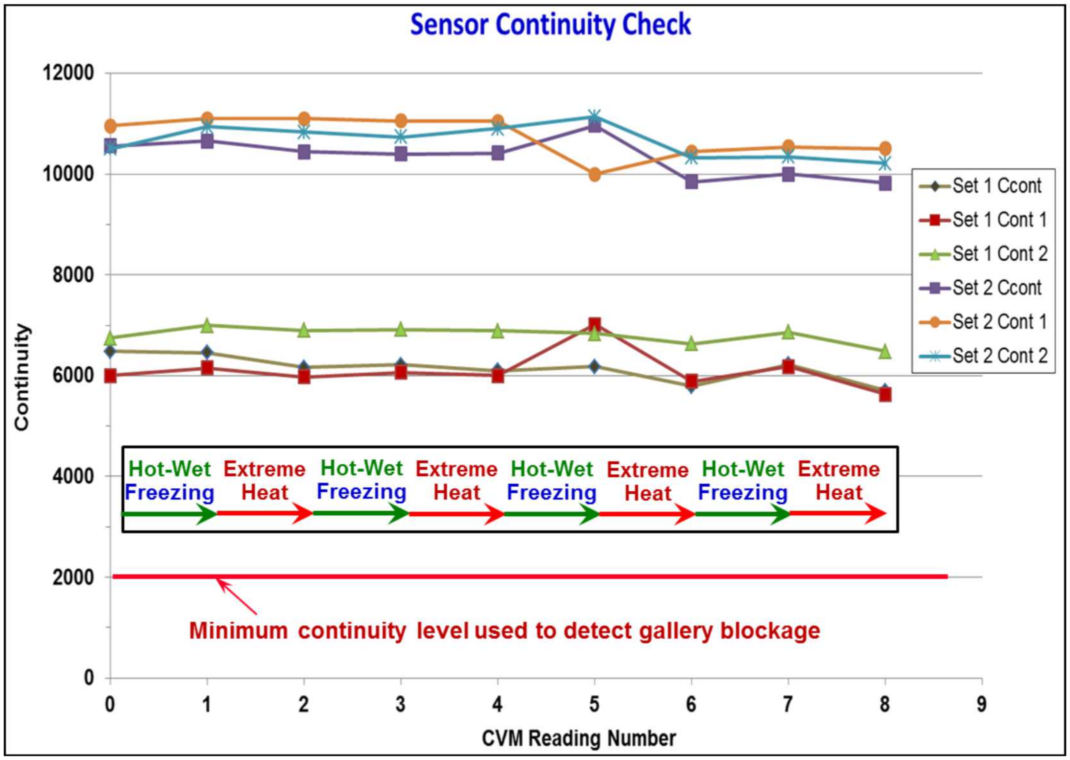


Figure 5-20: CVM Sensor Continuity Levels Remain Unchanged During Environmental Tests

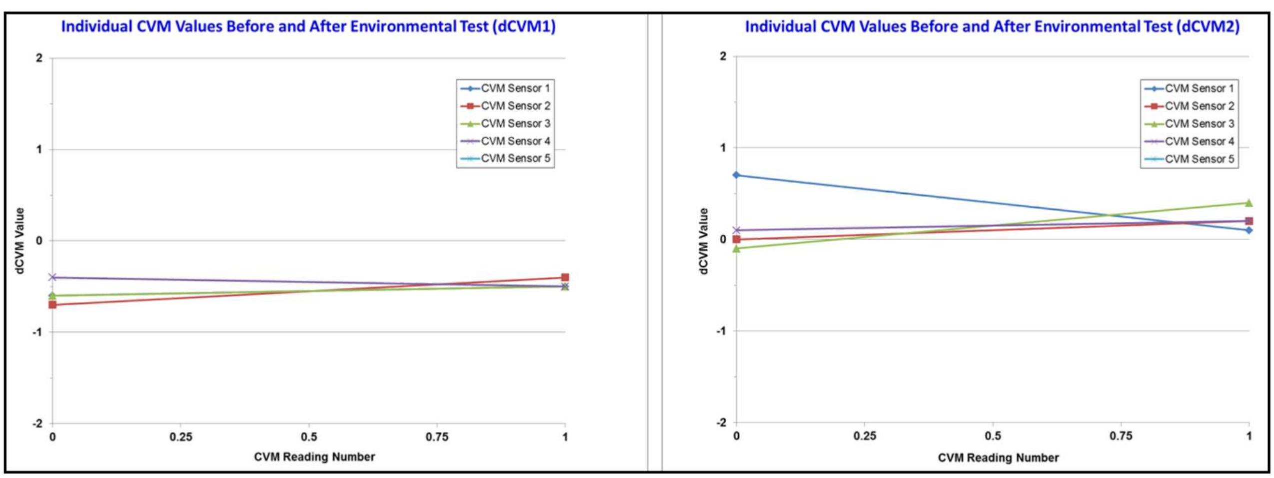


Figure 5-21: Individual CVM Sensor Readings Remain Unchanged

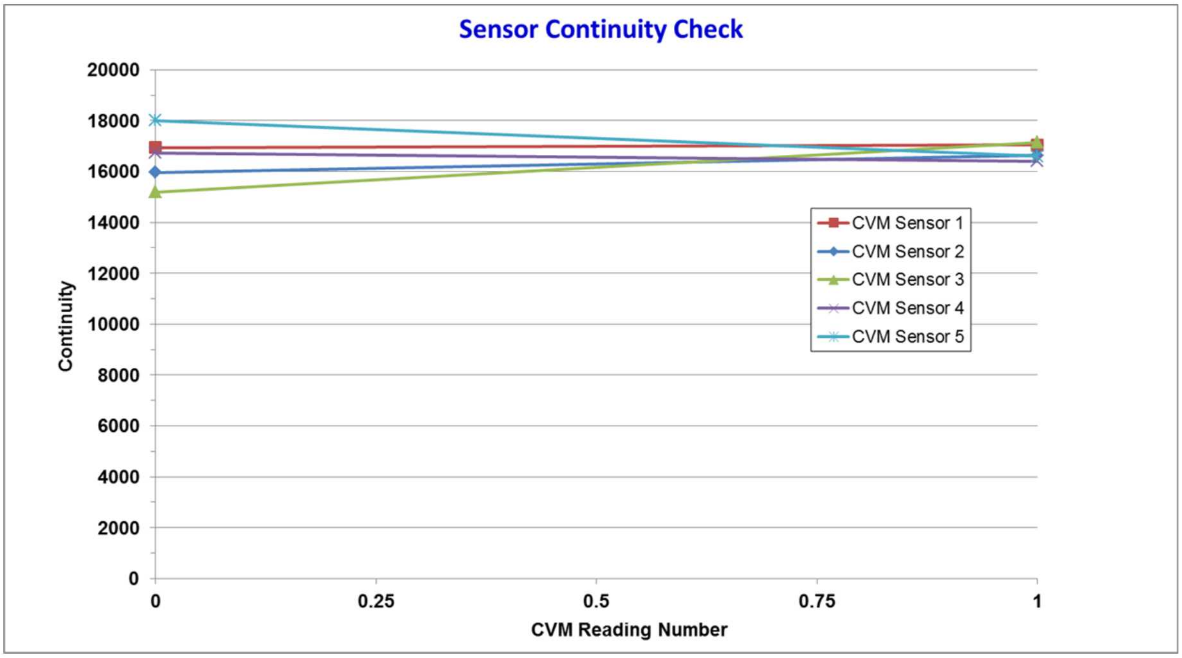


Figure 5-22: Individual CVM Continuity Readings Remain Unchanged

5.2 Exposure to Corrosion Inhibiting Compounds

Effect of Corrosion Inhibiting Compounds on CVM POD Performance - A focused study was also conducted to assess the effects of exposure to other materials they may exist in the wing box area. Specifically, CVM sensors were exposed to an array of Corrosion Inhibiting Compounds (CIC) to assess any effect on sensor performance. The objective was to provide confidence in the ability of CVM sensors to function properly and detect cracks even in the presence of CICs during crack growth. Following is a description of the CICs used in this study and the test set-ups used to produce extreme exposure levels for conservative assessments [5.7]. One of the key assumptions was that a small crack exists in the structure such that it is currently not detectable by CVM but could possibly allow for CIC ingress. Will CIC continue to wick into a growing crack and, if so, will it "fill" the crack to make it transparent to the CVM sensor? Tests were conducted to assess this.

<u>CICs Selected</u>:

- BMS 3-35 which is Ardrox AV15 or Corban-35 (Zip Chem)
- BMS 3-23 which is LPS-3 or Ardrox AV-8 or Dinitrol

<u>Assumptions on Worst Case Conditions</u>:

- CIC has access to CVM via wicking into a joint and along a rivet shank.
- Greatest opportunity for CIC wicking is in a joint where there is no sealant at all.
- Some CICs remain liquid for extended periods thus providing the opportunity to wick into cracks that were not present when it was initially applied.

Test Specimen: The test specimens were composed of a 2.5” wide plate with a doubler plate riveted to the back (material = 7075-T6). Figure 5-23 shows the test specimen design. Two rows of rivets were used to connect the two plates, however, the upper rivet row was only the single center hole to ensure controlled crack growth at this hole with the highest center stress. The single rivet also provided more space for additional CVM sensor placement as the cracks grew so that more data could be acquired from each specimen.

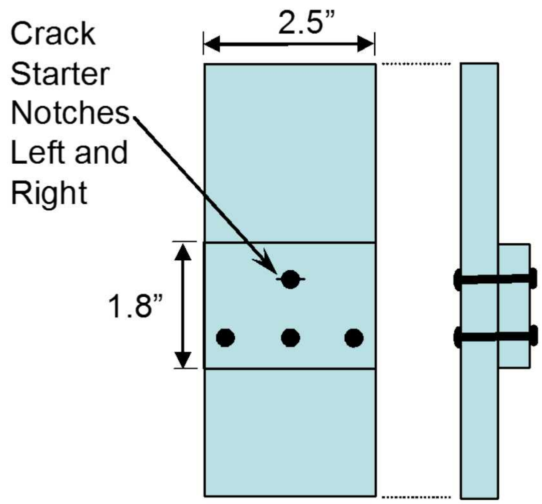


Figure 5-23: Schematic of Test Specimen used to Assess CIC Affects on CVM Operation

Test Procedure:

- Coupon plate and doubler plate were both coated with primer.
- No sealant was placed in the faying surface between the parent plate and the doubler to allow for maximum fluid ingress.
- Fatigue cracks were initiated in the specimen from the starter notches in the upper rivet hole. Cracks were propagated to a length of 0.050” or slightly longer but kept to a length that might exist prior to CVM crack detection. In other words, such a crack could exist in the field and be coated with CIC prior to CVM application.
- CIC was applied in normal application spray fashion with no intent to avoid nor excessively inject CIC between the faying surfaces. The CIC was applied to the front and back side of the test specimens. Because a normal joint would include three fasteners in the upper row of the test specimen, the upper left and right regions of doubler plate in the schematic above were clamped to eliminate any excessive gaps between the two plates (abnormal CIC ingress)
- CIC was allowed to cure as per the manufacturer’s specifications.
- The area for CVM application was prepped as per normal field installation procedures: sand surface, clean surface, apply primer. A CVM sensor was placed adjacent to each crack tip (i.e. no CVM detection or engagement at this point). The area marked with a red crosshatch in Figure 5-24 were prepped for the application of several CVM sensors.
- Fatigue loads were applied to grow the crack until permanent alarm (crack detection) was achieved by the CVM sensor.

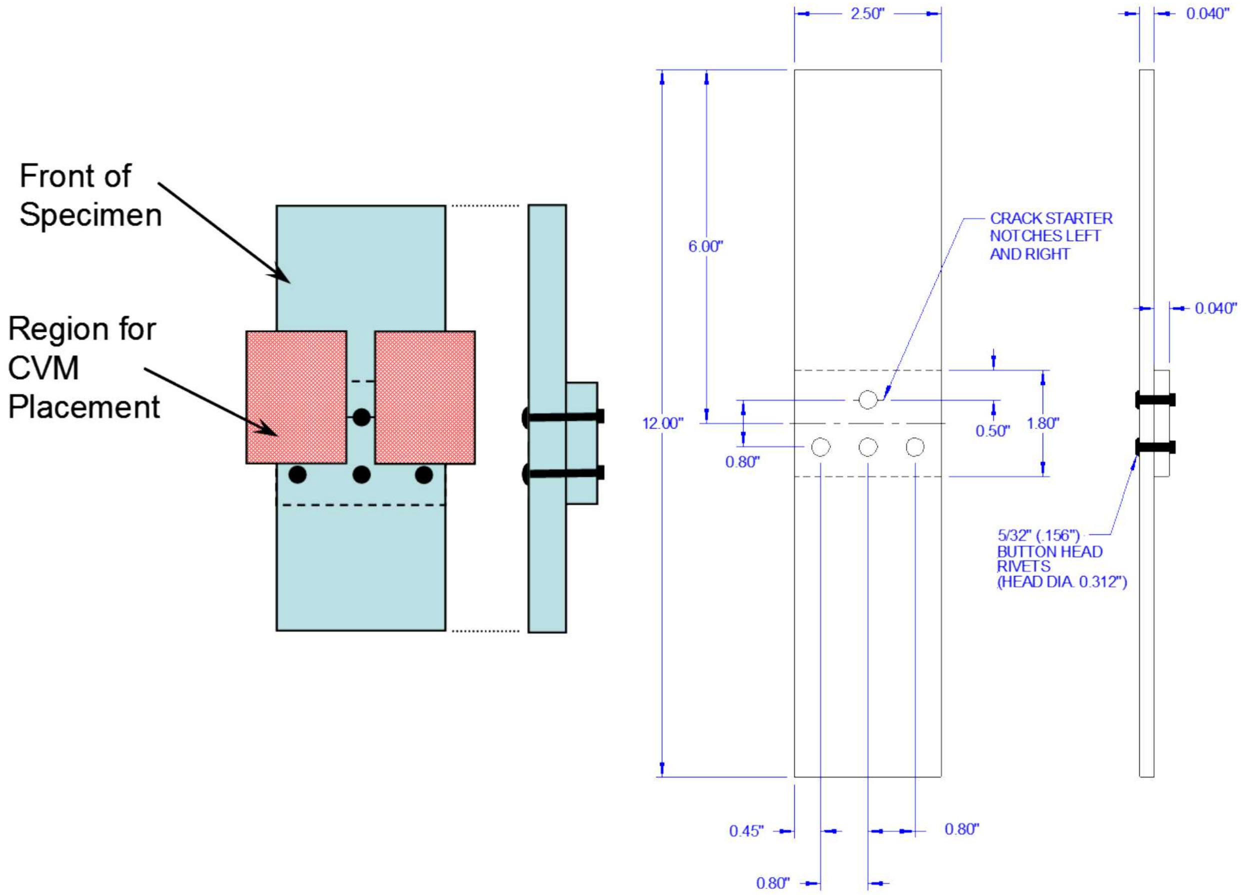


Figure 5-24: Dimensions of Test Specimen to Study Effects of CIC and CVM Placement for Crack Detection

<u>CVM Detection</u>: (same procedure used as in normal POD testing)

- Apply CVM to primer surface and measure baseline pressure levels.
- Use SIM-8 for real-time crack detection with max sensitivity.
- Apply PM-200 to determine final, permanent detection in an unloaded specimen.
- Measure crack length using eddy current inspections to establish CVM detection length.

<u>Data</u>:

- Figure 5-25 shows some of the assembled test specimens prior to CVM installations while Figure 5-26 shows the application of the CIC to produce a permanent elastomeric coating on the primer surface.
- Acquire 8-16 data points to produce a statistically-relevant data set that quantifies any affects from the presence of CIC.
- Repeat entire test series using both identified CIC compounds.

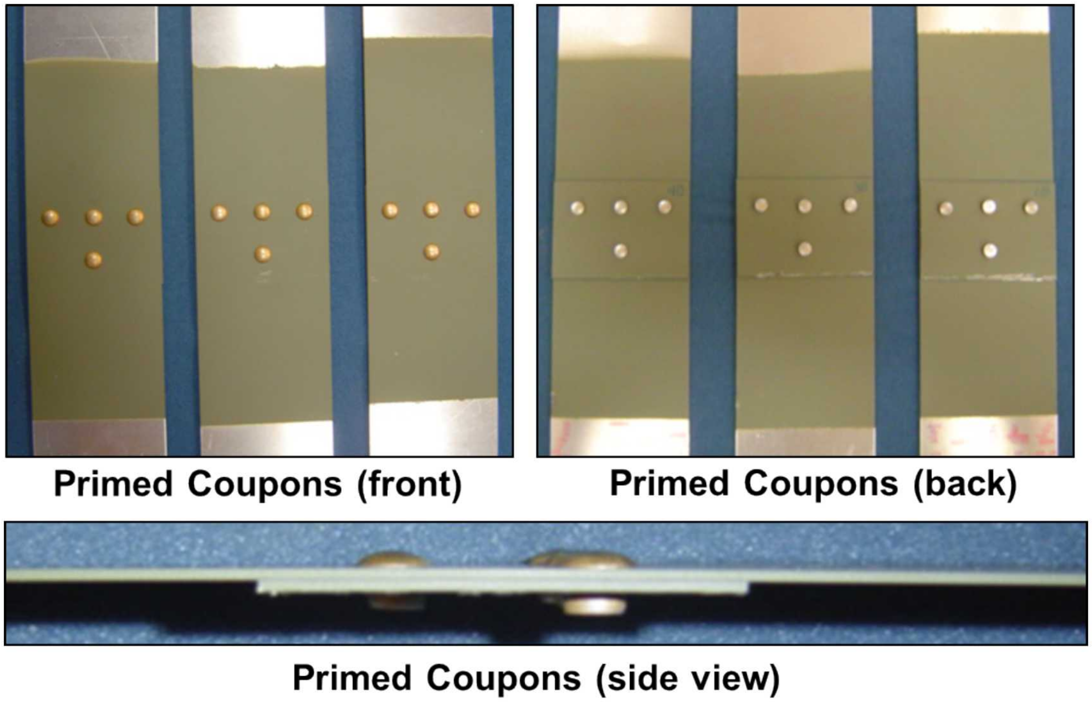


Figure 5-25: Photos of Riveted, Cracked Test Specimens to Study Effects of CIC on CVM Crack Detection

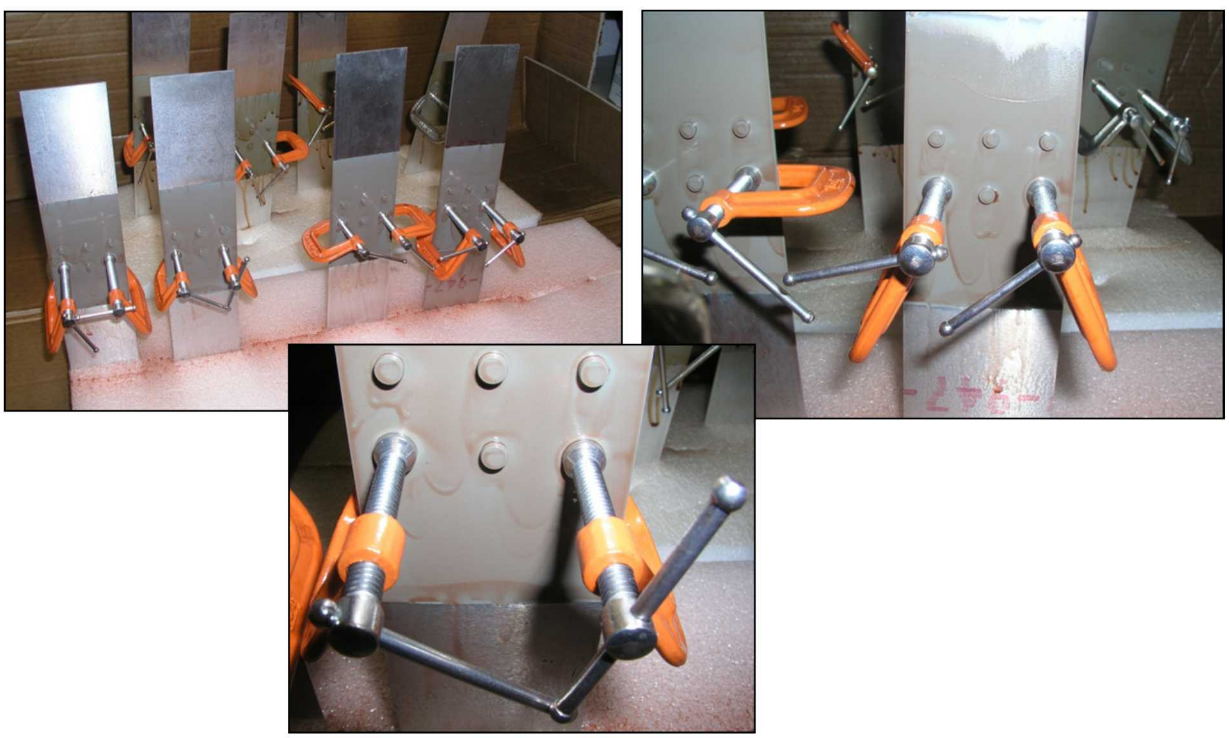


Figure 5-26: Application of CIC Compounds (Corban-35 and AV-8) to Test Specimens Prior to Fatigue Crack Growth

Crack Detection Results with and without CIC:

- After the application of the CIC to the cracked specimens, it was observed that no CIC was drawn into CVM galleries. Related to this, the galleries did not experience any blockage during the CIC testing.
- The crack detection results from all test specimens are summarized in Table 5-2 and Table 5-3 for Corban-35 and AV-8 CIC liquids, respectively. POD levels determined from testing in with and without CIC:

$POD_{(90/95)}$ = 0.011" without CIC (16 data points)
$POD_{(90/95)}$ = 0.013" with Corban-35 CIC in place (10 data points)
$POD_{(90/95)}$ = 0.018" with AV-8 CIC in place (6 data points)
$POD_{(90/95)}$ = 0.015" with any CIC in place (16 data points)

Since this POD variation is within experimental deviations, the conclusion is that there is no appreciable difference in CVM crack detection performance (POD) with or without the presence of CIC. CIC did not affect normal CVM operation.

CVM Results without CIC Present			
Panel	**Fastener Crack Site**	**Number of Fatigue Cycles**	**Crack Length at CVM Detection (growth after install in inches)**
1	1-L	3400	0.009
1	1-R	2400	0.011
1	2-L	6200	0.013
1	2-R	6000	0.014
1	3-L	6702	0.015
1	3-R	6702	0.004
2	1-R	3200	0.010
2	2-R	4850	0.006
2	3-L	5450	0.014
2	3-R	5450	0.018
3	1-L	3725	0.012
3	1-R	2925	0.006
3	2-L	4800	0.004
3	2-R	4600	0.008
3	3-L	5325	0.016
3	3-R	5230	0.005
Average Crack Length			0.011

Description: 0.040" thick panel (primer surface)

7075-T6 Alum.

CVM Results in Presence of CIC (Corban-35 CIC)		
Panel	**Sensor**	**Lag (inch)**
3C	1-R	0.012
4C	1-L	0.016
3C	2-R	0.010
4C	2-L	0.009
4C	3-L	0.019
3C	3-R	0.012
3C	4-R	0.026
4C	4-L	0.013
2C	1-L	0.010
2C	2-L	0.006
Average Crack Length		0.013

Table 5-2: CVM Performance in the Presence of CIC Compounds (Corban 35)

CVM Results without CIC Present			
Panel	Fastener Crack Site	Number of Fatigue Cycles	Crack Length at CVM Detection (growth after install in inches)
1	1-L	3400	0.009
1	1-R	2400	0.011
1	2-L	6200	0.013
1	2-R	6000	0.014
1	3-L	6702	0.015
1	3-R	6702	0.004
2	1-R	3200	0.010
2	2-R	4850	0.006
2	3-L	5450	0.014
2	3-R	5450	0.018
3	1-L	3725	0.012
3	1-R	2925	0.006
3	2-L	4800	0.004
3	2-R	4600	0.008
3	3-L	5325	0.016
3	3-R	5230	0.005
Average Crack Length			0.011

Description: 0.040" thick panel (primer surface)

7075-T6 Alum.

CVM Results in Presence of CIC (AV-8 CIC)		
Panel	Sensor	Lag (inch)
1D	1-L	0.007
1D	2-L	0.014
2D	1-R	0.030
2D	2-R	0.020
2D	3-R	0.017
2D	4-R	0.018
Average Crack Length		0.018

Table 5-3: CVM Performance in the Presence of CIC Compounds (AV-8)

Assessment Cure Time for Corrosion Inhibiting Compounds - For these tests, a series of simulated lap splice joints were used to provide faying surfaces for evaluating various cure times for normal and excessive amounts of Corrosion Inhibiting Compounds (CIC). Figure 5-27 shows the test specimen components while Figure 5-28 shows the assembled joint ready for application of the CIC. The joint was assembled with Clecos in lieu of actual rivets to allow the joint to be disassembled for cure assessments over time.

<u>Cure Trial #1 – Normal Application of CIC</u> -

- These trials involved CIC applied as per normal specifications. The CIC application involved a spray of Corban-35 using 3 to 4 passes over the assembled joint at a distance of 8 – 10 inches.
- CIC was applied to specimens Cor-35-Cure-1 thru Cor-35-Cure-4.
- Result: With normal rivet clamp-up spacing, the CIC did not penetrate far into the faying surface (wicking at edges only). In addition, all CIC cured to a hardened coating in 24 hours.

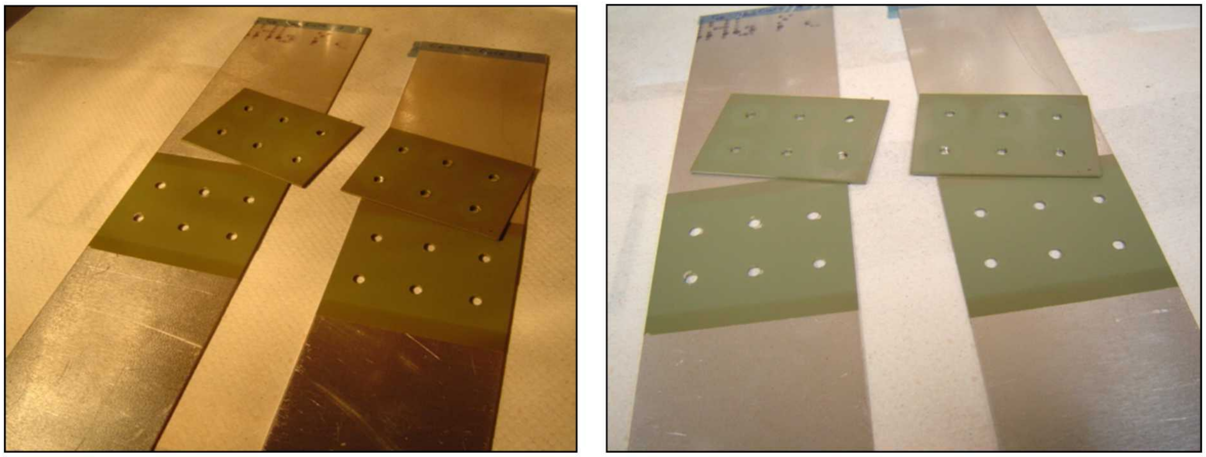


Figure 5-27: Components of CIC Cure Test Specimens Prior to Assembly

<u>CIC Cure Trail #2 - CIC Applied to Extreme Levels (Corban-35)</u> -

a) Cor-35-Cure-1 thru Cor-35-Cure-3: These trials involved a non-normal CIC application where the <u>inside</u> of the faying surfaces were sprayed directly (5 passes used to produce thick coating), as shown in Figure 5-29, and then assembled into the panel shown in Figure 5-28. This application method produced high amounts of CIC within the faying surface joint and exhibited excessive accumulation/pooling on specimens after 5 passes.

   The result for the Cor-35-Cure-1 through Cor-35-Cure-3 specimens was that after 24 hours the CIC both inside and outside was mostly cured. The outside of the specimen was dry and the inside faying surface area was hardened and dry in some spots. Some inside spots were very tacky such that it would not flow or wipe off. After 48 hours, all CIC was hardened into a coating like nail polish.

b) Cor-35-Cure-4 and Cor-35-Cure-5: These trials involved a non-normal CIC application on assembled specimens where the CIC was sprayed to excess (10 passes) until it was flowing. The application method produced high amounts of CIC liquid accumulation at the plate edge in the lap joint as shown in Figure 5-30.

The result for Cor-35-Cure-4 thru Cor-35-Cure-5 specimens revealed the following CIC cure rates. The specimens were disassembled after 24 hours and excessive pooling of CIC was found inside the faying surface region. After 24 hours, the outside of the specimen was cured and dry. The inside (faying surfaces) was very tacky such that it would not flow or wipe off with the exception of a few accumulation areas along the edge where the CIC could be wiped off with cloth and could possibly flow. After 48 hours, the CIC on the hidden, inner faying surfaces was mostly cured into a hardened coating like nail polish. The few CIC accumulation areas along the edge were tacky to the point of not flowing (would not wipe off with cloth; would not flow) as shown in Figure 5-31. After 96 hours of cure time, even the non-natural accumulation areas were cured to a hardened coating.

Figure 5-28: CIC Cure Assessment Coupons with Clecos at Rivet Points

Figure 5-29: Application of BMS 3-35 (Corban-35) Corrosion Inhibiting Compound to Evaluate Cure Time and Flow of CIC Compounds

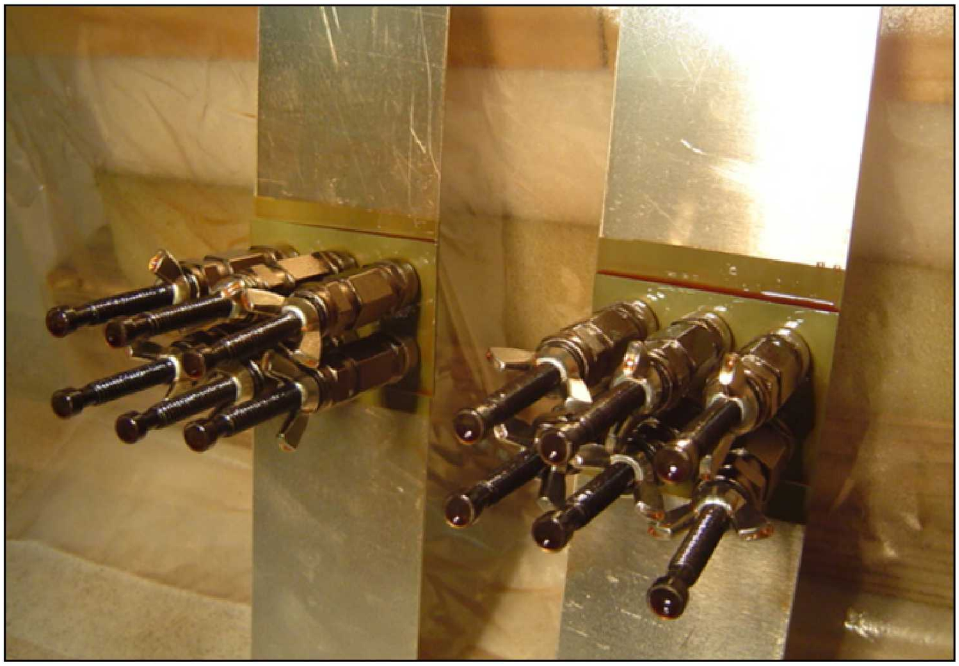


Figure 5-30: Accumulation of Excessive CIC in Assembled Joint After a Non-Normal Number of Spray Passes

<u>CIC Cure Trail #3 - CIC Applied to Extreme Levels with CVM Installed Directly on CIC</u> -

- These tests involved specimen Cor-35-Cure-5 where a non-normal CIC application was produced by spraying an excessive amount of CIC (10 passes) until it was flowing (see Figure 5-30). After waiting 48 hours, the specimen was disassembled and a CVM sensor was installed on the tacky CIC found on the inside, faying surface as shown in Figure 5-32.
- Recall that after 48 hours, the CIC on the hidden, inner surfaces was mostly cured into a hardened coating like nail polish. The few CIC accumulation areas along the edge were tacky to the point of not flowing (would not wipe off with cloth; would not flow) as shown in Figure 5-31. After the CVM was applied directly to the tacky inside surface at CIC accumulation points, a normal data acquisition process was carried out using the PM-200 device. A vacuum was drawn on both of the CVM galleries, however, no CIC was drawn into the CVM galleries after two hours at full vacuum (530 Pa). The clear galleries are shown in Figure 5-32.

<u>CIC Cure Trail #4 - CIC Applied to Extreme Levels (AV-8)</u> -

- These tests involved specimen AV-8-Cure-1 where a non-normal CIC application was produced by spraying an excessive amount of CIC (10 passes) until it was flowing. The application method produced high amounts of CIC liquid accumulation at the plate edge in the lap joint as shown in Figure 5-30. During CIC application, the overlapping plate was facing upwards to allow the CIC to pool and wick into the faying surfaces.

- Result for AV-8-Cure-1 specimen. After 24 hours, the outside of the specimen was cured and dry. The inside (faying surfaces) was very tacky such that it would not flow or wipe off with the exception of a few accumulation areas along the edge where the CIC could be wiped off with cloth and could possibly flow. After 48 hours, the CIC on the hidden, inner surfaces was mostly cured into a hardened coating like nail polish. The few CIC accumulation areas along the edge were tacky to the point of not flowing (would not wipe off with cloth; would not flow) as shown in Figure 5-33

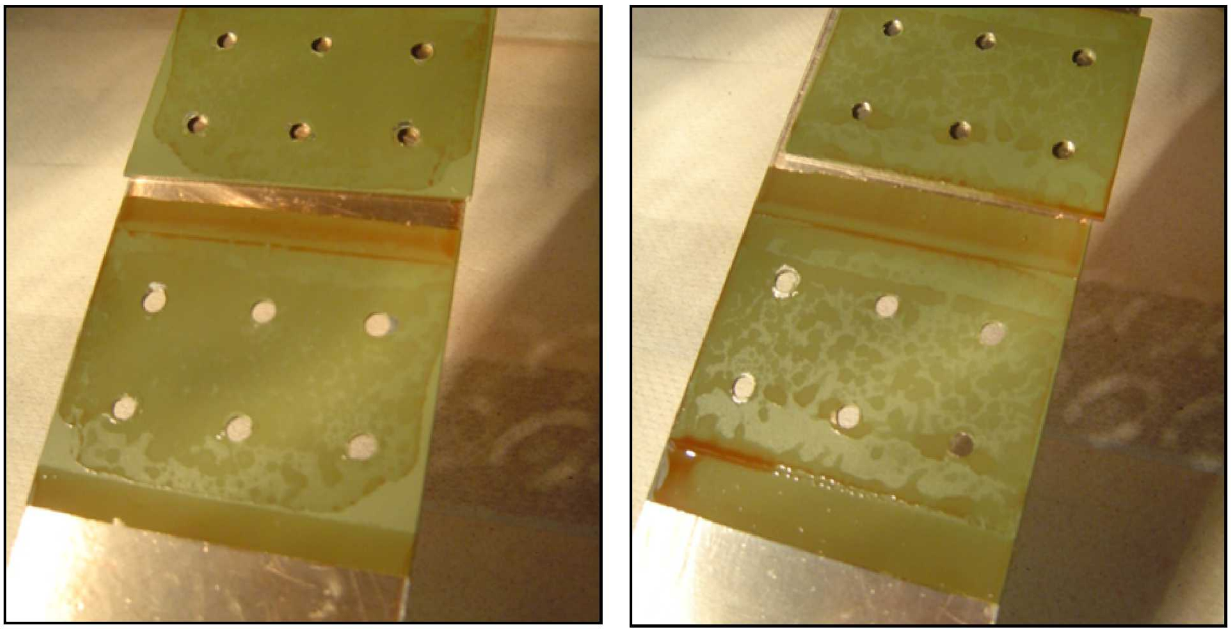


Figure 5-31: View of Cure Levels at Extreme Accumulation Points of Corban-35 CIC Inside the Faying Surfaces of the Specimen Joint

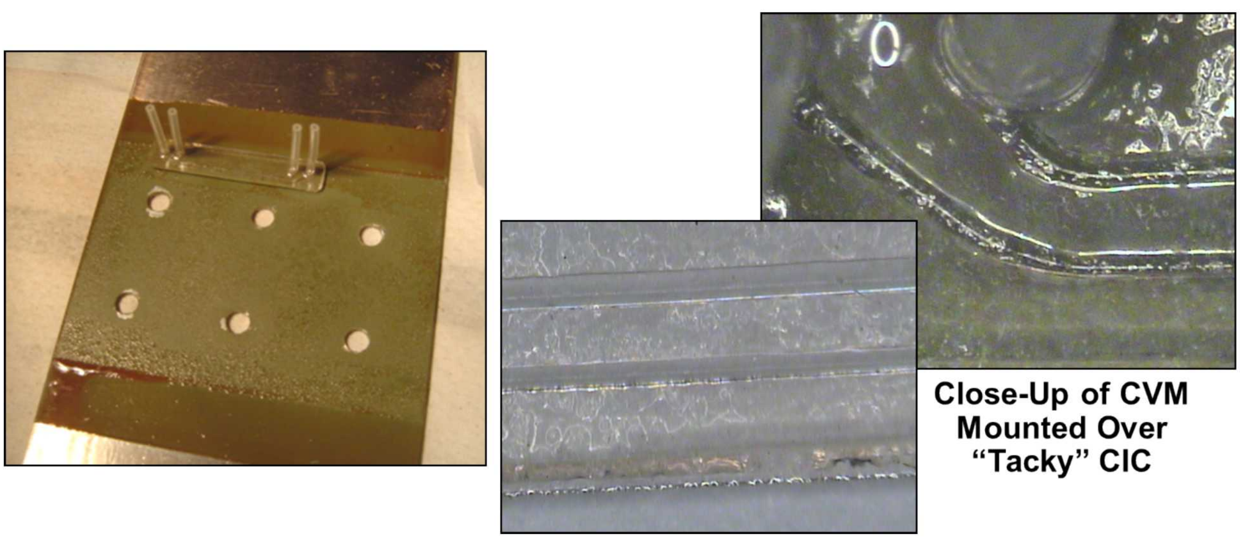


Figure 5-32: Installation of CVM Directly on a Surface Coated with CIC

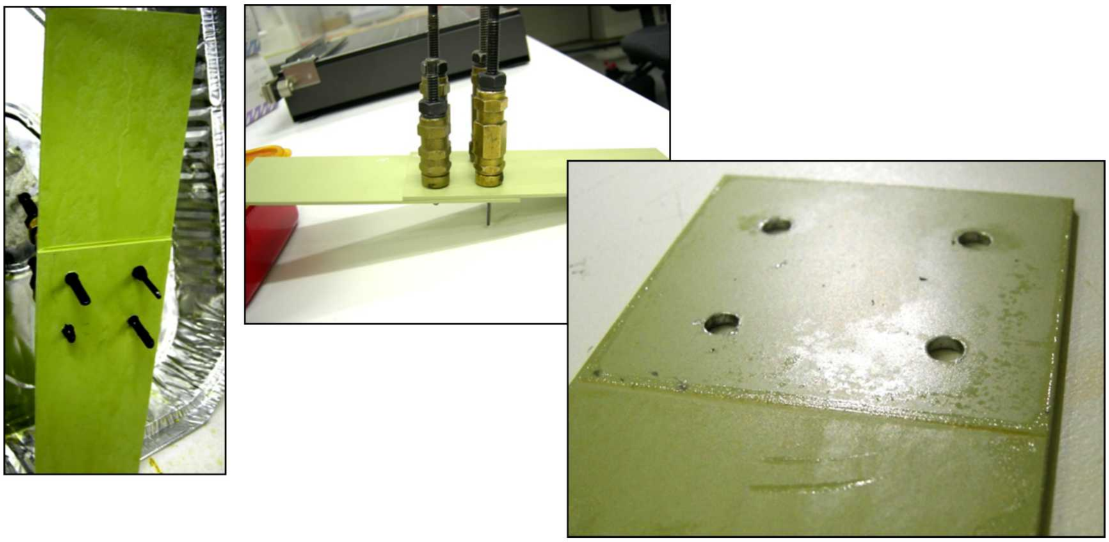


Figure 5-33: View of Cure Levels at Extreme Accumulation Points of AV-8 CIC Inside the Faying Surfaces of the Specimen Joint

Assessment Corrosion Inhibiting Compounds on Newly-Installed CVM Sensors - SMS and its subsidiary, AEM, conducted a separate study to assess any effects of CIC on the installation of a CVM sensor [5.8]. Specifically, the study was intended to determine if a CIC could possibly wick under a CVM sensor or otherwise affect the adhesive layer between the sensor and the surface to which it is applied. It is a possible occurrence for CIC to be sprayed in the area of CVM sensors after the sensors are installed in an aircraft. In this study, test specimens containing CVM sensors were sprayed with normal-to-excessive amounts of Boeing BMS 3-23 G Type II Corrosion Inhibiting Compound to ensure full coverage, and even pooling, along the FEP material edges. The CIC was applied to different specimens after a predetermined amount of time (2 hours, 6 hours and 24 hours). Pull tests were then performed on each sensor. Figure 5-34 shows some of the coupon test specimens along with a typical pull test used to quantify the adhesive strength between the sensor and the primed, metal surface.

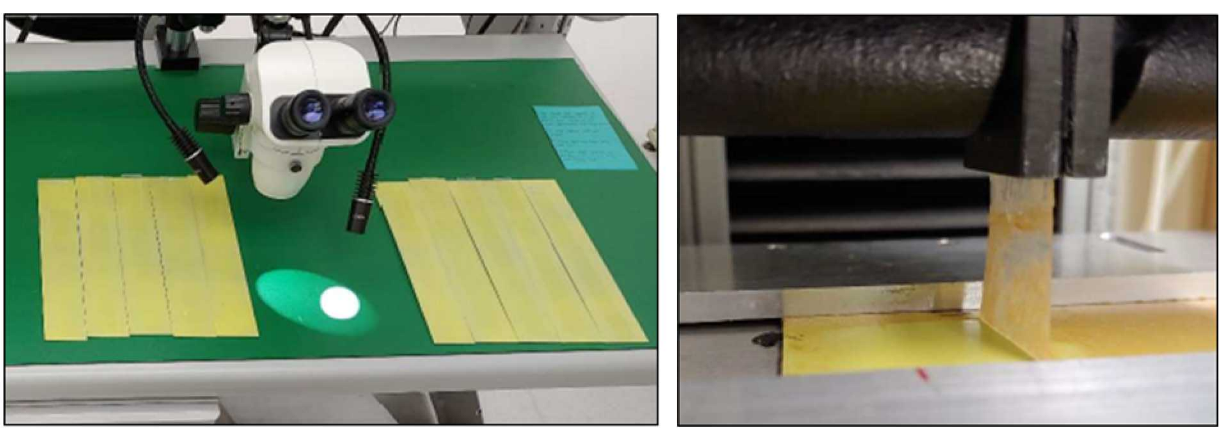


Figure 5-34: Test Specimens and Pull Tests Used to Assess Effects of CIC on the Bond Strength of CVM Sensors

Results from the specimens coated with CIC and those left uncoated (control specimens) were compared. Microscopic inspections confirmed that no wicking took place under the sensor FEP material. In addition, no discernible degradation was observed visually under the FEP or exposed during bond pull testing. All specimens had acceptable sensor bond strength indicating that the CIC did not adversely affect the sensor installation. Based on these results, the CVM installation procedure was enhanced to reduce the CVM installation cure time from the initial 36-hour cure time down to 2 hours before CIC is applied.

Overview of Effects of CIC on CVM - It should be noted that while no wicking of CIC or otherwise adverse effects of CIC on CVM performance were noted, any wicking of a liquid or other obstruction such as dust particles into a CVM sensor will result in blockage of the galleries. In such cases, the sensor will fail the initial positive flow test prior to any acquisition of data. The PM-200 device will indicate a failure in this positive flow measurement (Continuity level too low) and the sensor will need to be revisited – and possibly replaced – before and crack detection data can be acquired. Thus, blockage in the sensor galleries will produce a fail-safe action and not result in the acquisition of erroneous data.

In the case of CIC application over CVM sensors, it was determined that CIC coatings can be safely applied two hours after CVM sensor installation. However, in the event that a CIC – or any other liquid application around CVM sensors – is applied such that it affects the adhesive between the sensor and the surface it is monitoring, the vacuum readings (dCVM values) will be affected and revealed during the PM200 monitoring process. This will indicate that the sensor needs to be checked. Once again, this result will correspond to a fail-safe response and a reinstallation of the sensor before any erroneous data or false calls are recorded.

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Chapter 6

6.0 CVM Flight Tests

This chapter will discuss the first two flight test series that were conducted to assess the performance of CVM sensors on operating aircraft. The first test series placed CVM sensors in regions that were not expected to experience any cracking. For this reason, the flight tests were considered CVM installations in "decal mode" (i.e. no damage). The purpose of this initial test series was to explore general installation, operation and monitoring of CVM sensors by airline personnel while also assessing the durability of the sensors when exposed to real flight conditions. Different sensor designs were installed in various aircraft regions without any particular application in mind. The second test series was conducted in association with the 737 wing box fitting program. CVM sensors, designed to monitor the actual wing box fitting, were installed on the set of ten wing box fittings, on seven different 737 aircraft that were operating in the Delta Air Lines fleet. Overall, these flight tests have allowed for the accumulation of over 1.5 million successful flight hours of CVM operation. In general, flight tests provide critical information about the long-term performance, reliability, durability and continued airworthiness of flying components [6.1 – 6.5]. They are a key element in establishing inspections (or maintenance actions) needed to avoid catastrophic failure during the operational life of the airplane.

6.1 First Flight Test Series - Sensors in Decal Mode

These CVM functional and environmental durability tests were conducted to assess the long-term viability of CVM sensors in an actual operating environment. Twenty-two sensors were installed for functional evaluation on DC-9, 757 and 767 aircraft as summarized in Table 6-1. Figure 6-1 and Figure 6-2 show photos of some of the sensor installations. Figure 6-3 shows some additional details on the sensor data acquisition connection points and the monitoring process which was conducted from the baggage compartment, thus eliminating the need to disassemble the aircraft to conduct inspections.

Aircraft	Tail	Operator	Date	# Sensors	Status
DC-9	9961	NWA	Feb 04	6 (4 remaining)	2 sensors removed by NWA
DC-9	9968	NWA	Apr 05	6	3 sites
B757	669	Delta	Apr 05	8	4 sites in empennage on stringers, frames & near APB
B767	1811	Delta	Apr 05	6 (4 connected)	3 sites in empennage

Table 6-1: Summary of CVM Installations on Aircraft Operating in the Delta Air Lines and Northwest Airlines Fleets

Typical CVM data is shown in Figure 6-4 through Figure 6-7. The continuity data shows the positive flow through the sensor galleries which ensures that there is no blockage in the sensor prior to taking crack detection data. These values should be consistent and high. The CVM data displays the vacuum level (pressure) in the sensor. Extremely low levels indicate a successful ability to pull a vacuum on the sensor (i.e. no crack is present to produce leakage and abnormally high pressure. So, in the absence of any damage, these values should be consistent and low.

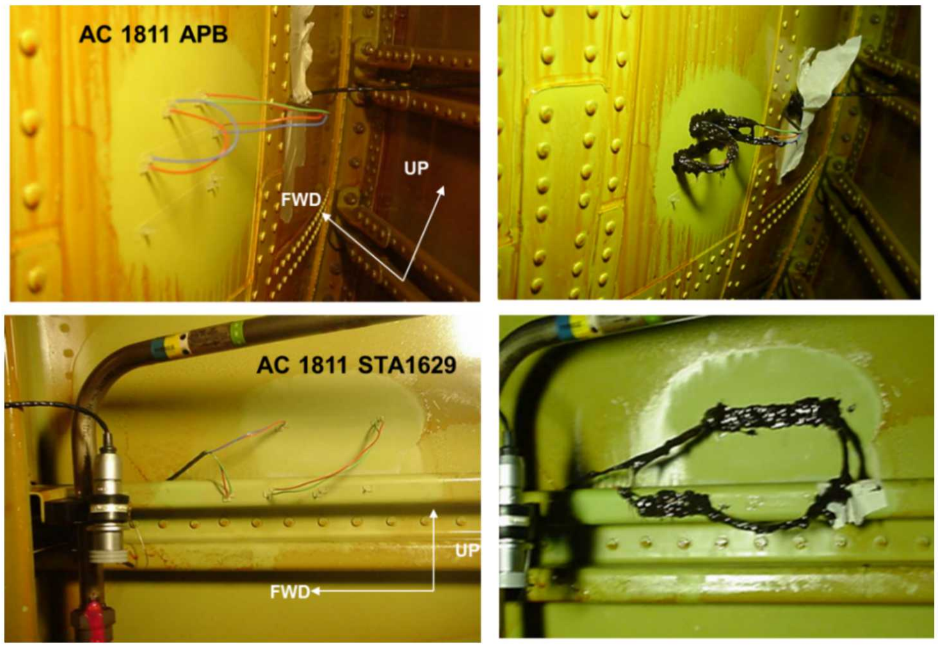


Figure 6-1: Sample CVM Installations on Aircraft in the Delta Air Lines Fleet

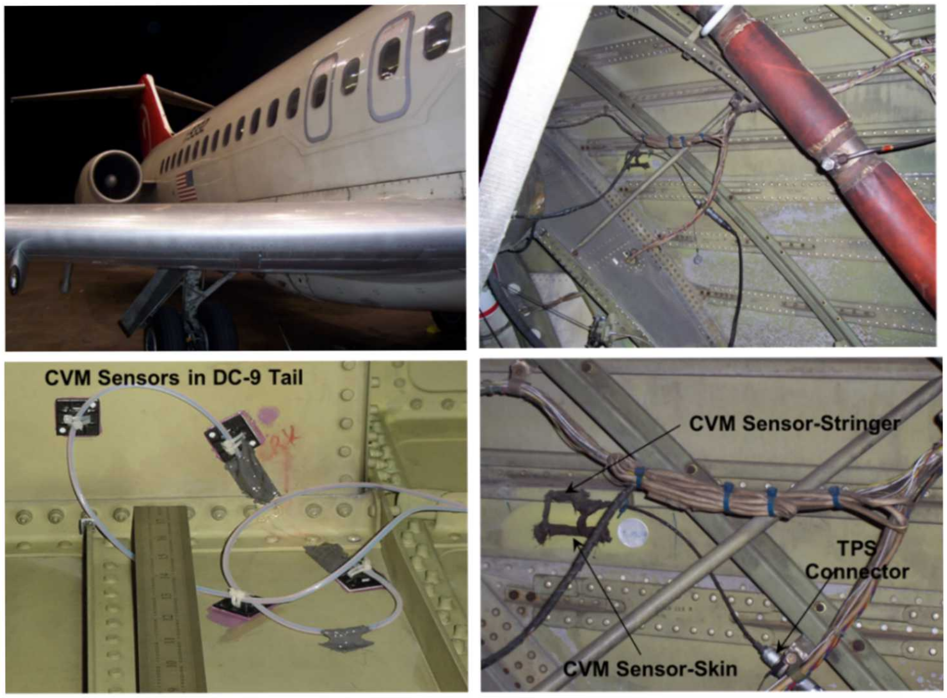


Figure 6-2: Sample CVM Installations on Aircraft in the Northwest Airlines Fleet

Figure 6-3: Field Evaluation of CVM Sensor Applications

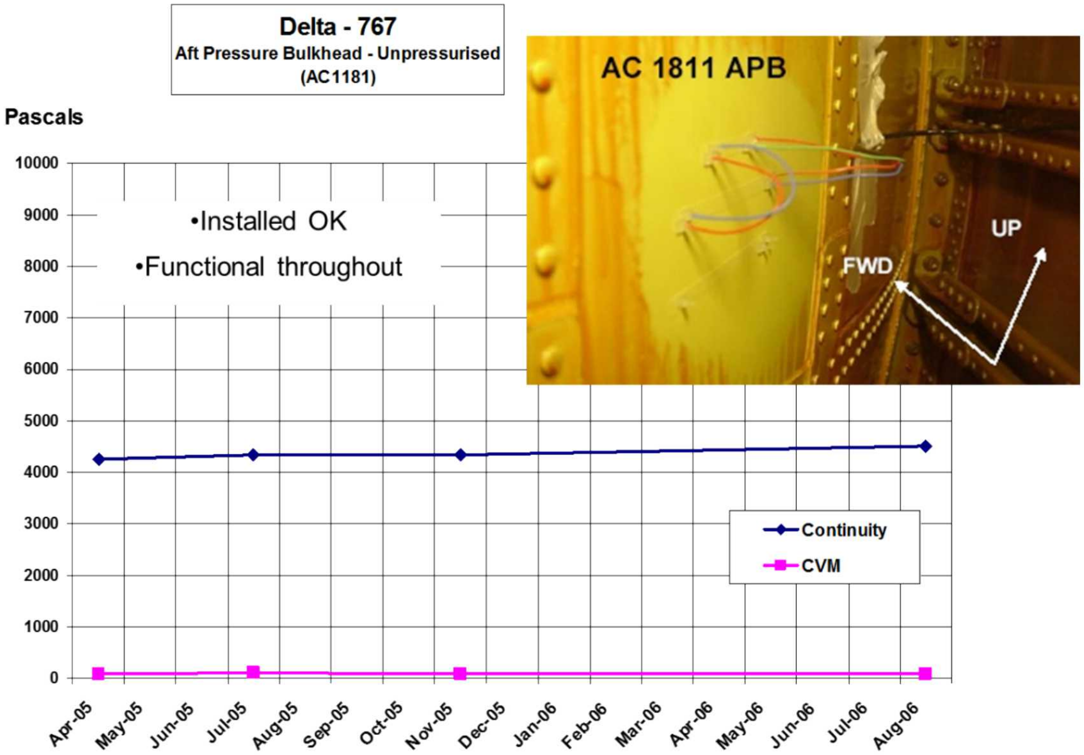


Figure 6-4: Sample CVM Data Acquired from an Aft Pressure Bulkhead Installation

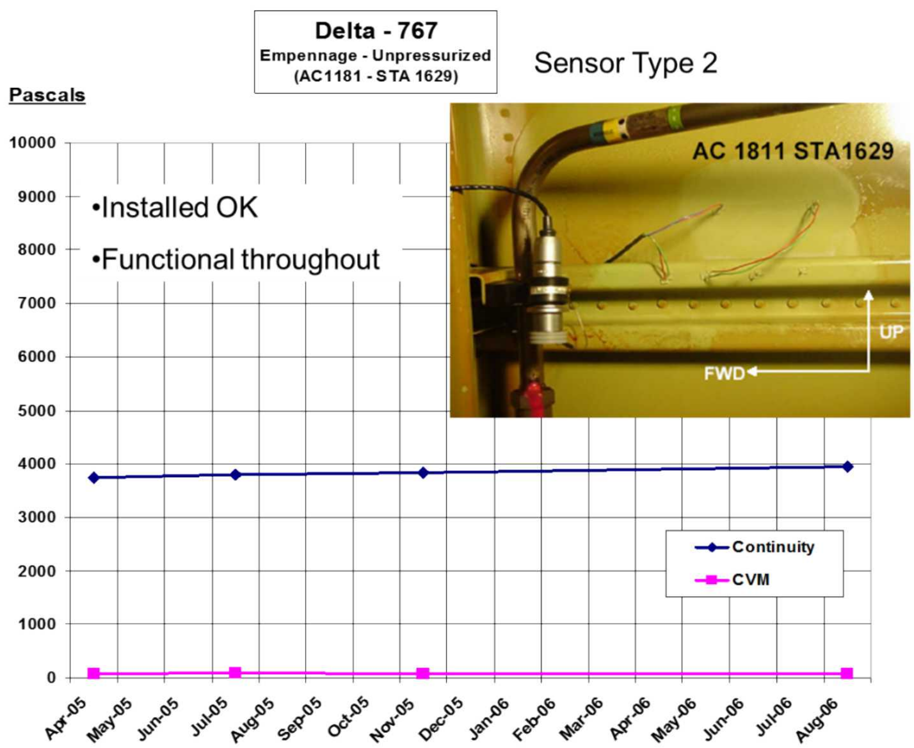


Figure 6-5: Sample CVM Data Acquired from an Empennage Installation

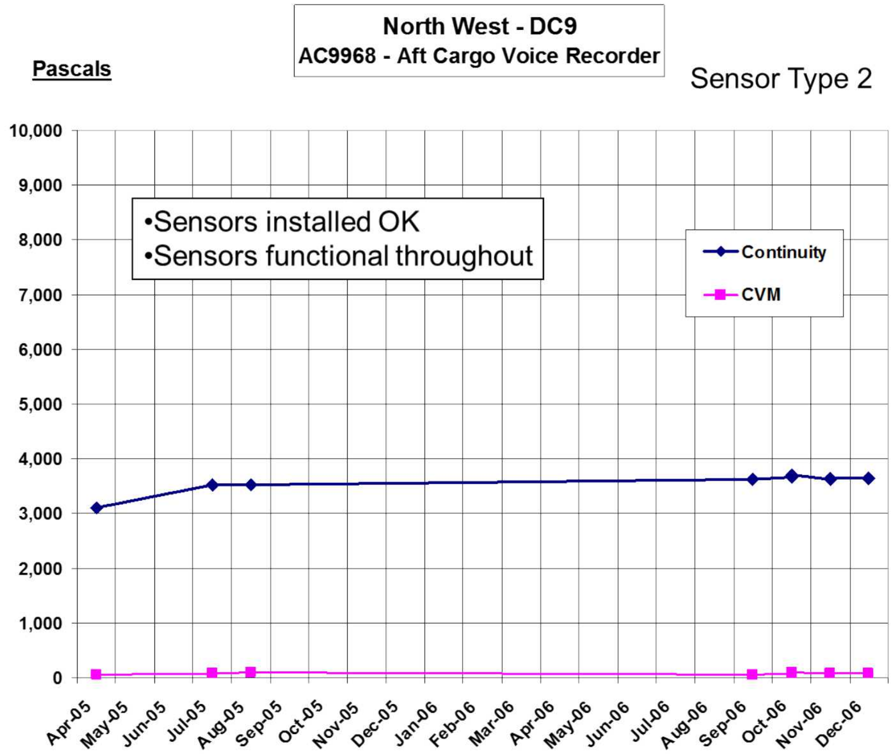


Figure 6-6: Sample CVM Data Acquired from an Aft Cargo Region Installation

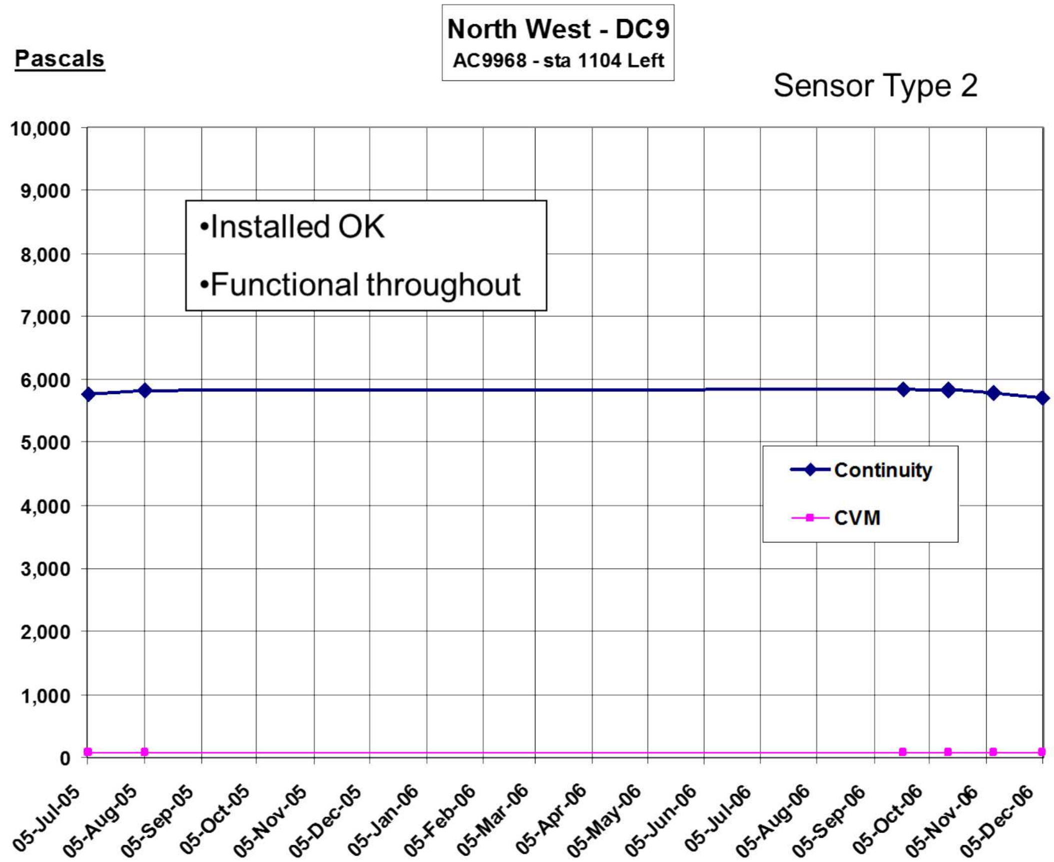


Figure 6-7: Sample CVM Data Acquired from Stringer Substructure Installation

This initial flight test series demonstrated the ability of CVM sensors to: 1) operate successfully on operating aircraft over long periods of time, 2) produce consistent data and 3) be properly installed and monitored by airline personnel.

6.2 Flight Test of CVM Sensors for Wing Box Fitting Application

The SHM certification and integration activity for the 737 wing box fitting included both controlled laboratory-based testing and field testing. In addition to the lab performance tests described above, a set of 68 sensors were mounted on wing box fittings in seven different B-737 aircraft in the Delta Air Lines fleet. The sensors have been monitored every 90 days for over four years, producing over 1,200 sensor response data points. These flight tests demonstrated the successful, long-term operation of the CVM sensors in actual operating environments. This environmental durability study compliments the laboratory flaw detection testing described above as part of the overall CVM certification effort.

The joint effort of Delta, Boeing, the FAA, SMS Ltd. and Sandia National Labs was established to leverage airline interest and associated activities aimed at the adoption of SHM solutions along with the desire of the FAA to oversee the safe usage of SHM. By including an airline, aircraft OEM, a regulator and a research agency on this team and addressing all "cradle-to-grave" issues associated with SHM use, the goal was to produce a roadmap or guideline procedures for the industry to use when certifying SHM systems. One key portion of that roadmap procedure includes the use of trial installations and flight tests at Delta Air Lines to identify the field-based portion of SHM deployment. Important topics to study during the flight tests include:

- Complete SHM indoctrination and training for Delta personnel (ranging from management to A&P mechanics) in all pertinent departments such as engineering, maintenance, NDI, supply and logistics.
- Complete indoctrination of FAA personnel who may be involved in the certification process such as Aircraft Certification Office, Transport Directorate, and Principle Maintenance Inspectors.
- Hardware specifications, installation procedures, operation processes, continued airworthiness instructions
- Complete formal modifications to integrate SHM into airline maintenance programs. This includes: 1) production of all documents needed to install and monitor CVM sensors, 2) hardware specifications to control all items being installed on the aircraft, 3) Job Cards needed to guide and ensure all tasks to be conducted by maintenance personnel for the specific application, 3) Engineering Authorization document to oversee the SHM deployment and, 4) Instructions for Continued Airworthiness which specifies all actions needed to maintain the SHM system in working order for the duration of its use on the aircraft.
- Identify and complete the approval processes within an airline maintenance depot to obtain the necessary signatures needed from all oversight management and engineers.
- Assess aircraft maintenance depots' ability to safely adopt SHM and the FAA support needed to ensure airworthiness.

A total of 10 sensors were used to monitor the 10 wing box fittings in each 737 aircraft. Several sensors on each side of the aircraft were connected in series (daisy-chained) to single SLS connectors. The ten sensors were daisy-chained into sets of 2 and 3 (left side of wing box) and sets of 2 and 3 (right side of wing box) such that they could be monitored by 4 SLS connectors. Figure 6-8 through Figure 6-10 provide an overview of the CVM installations on the Wing Box fittings including the routing of all connection lines to the SLS connectors. Figure 6-11 and Figure 6-12 show the daisy-chain arrangements used to group the ten sensors into sets of 2 and 3 sensors (4 groups = full 10 sensor set of each aircraft). Figure 6-13 and Figure 6-14 provide additional detail on the sensor groups along with schematics showing how the SLS connectors were mounted on the forward Wing Spar and the tube routing between each sensor and the set of four SLS connectors.

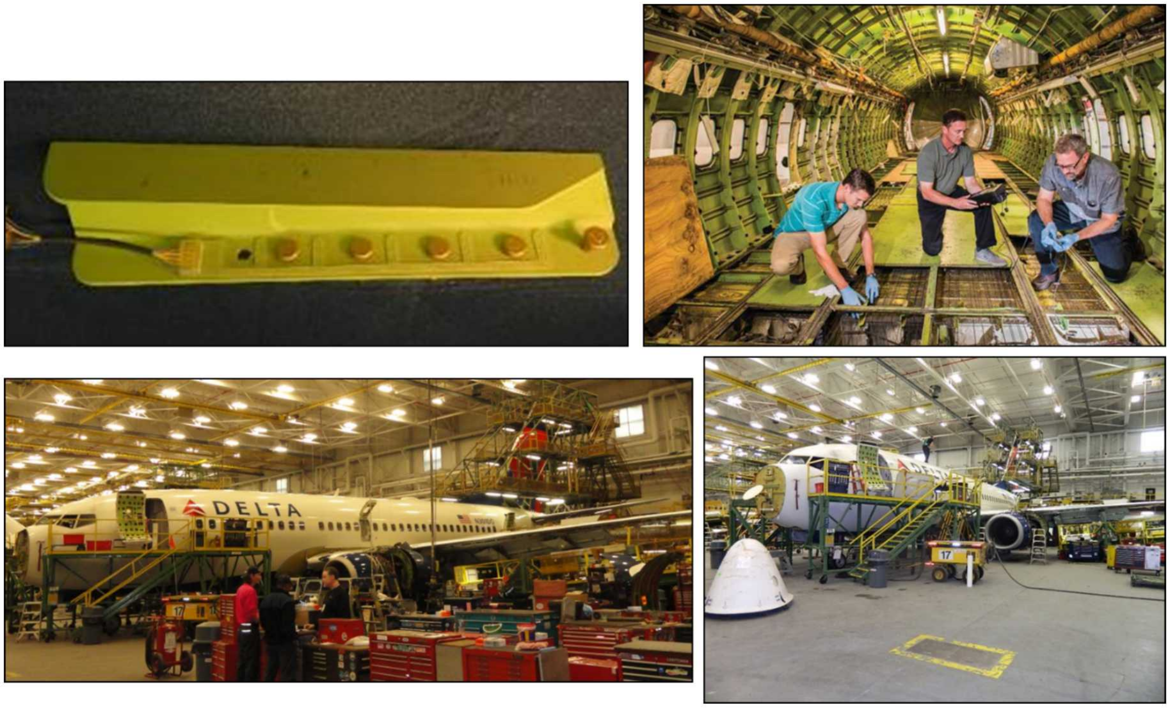


Figure 6-8: Overview of CVM Sensor Installations on Wing Box Fittings at Delta Facility

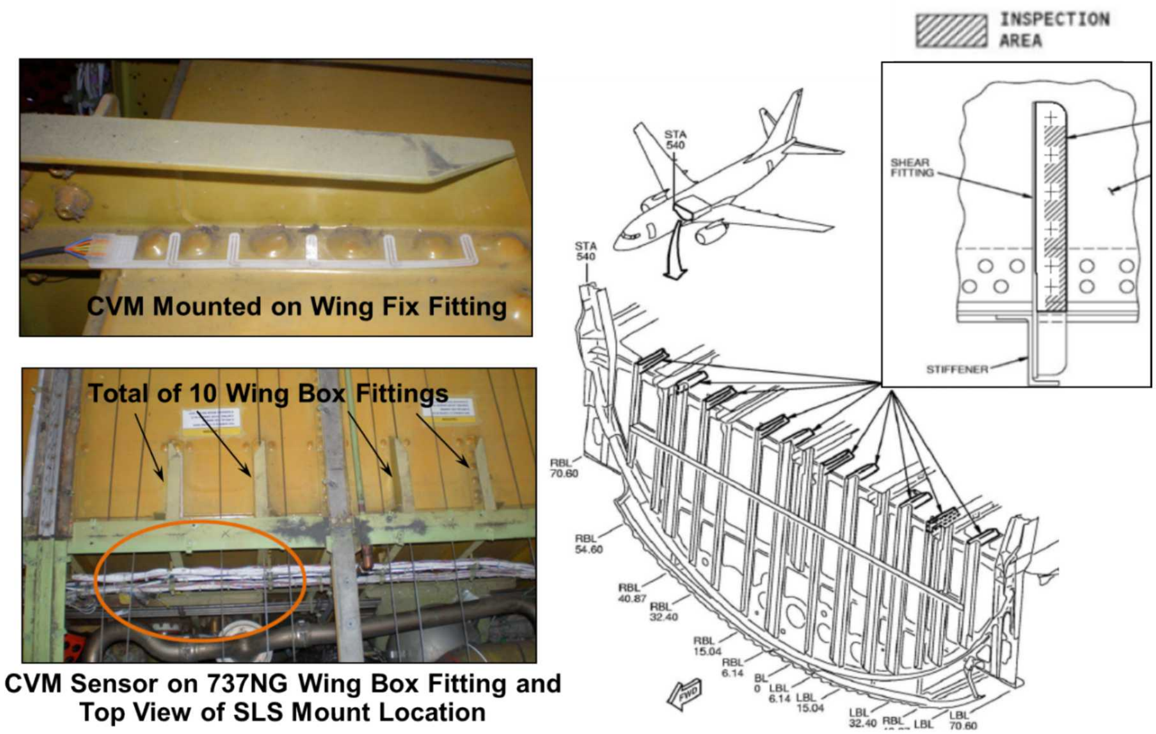


Figure 6-9: 737 Wing Box Area – Location of Ten Fittings and CVM Sensor to Monitor for Cracks in the Inspection Area Highlighted

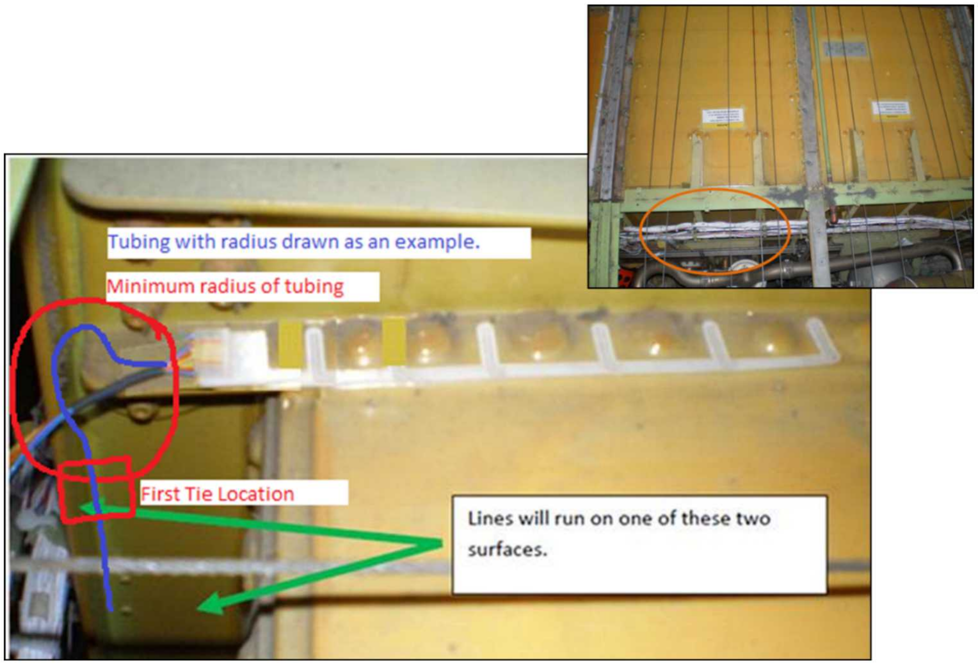


Figure 6-10: Close-Up of CVM Sensor to Show Tube Routing and Top View of SLS Mount Location Along Forward Wall of Wing Box

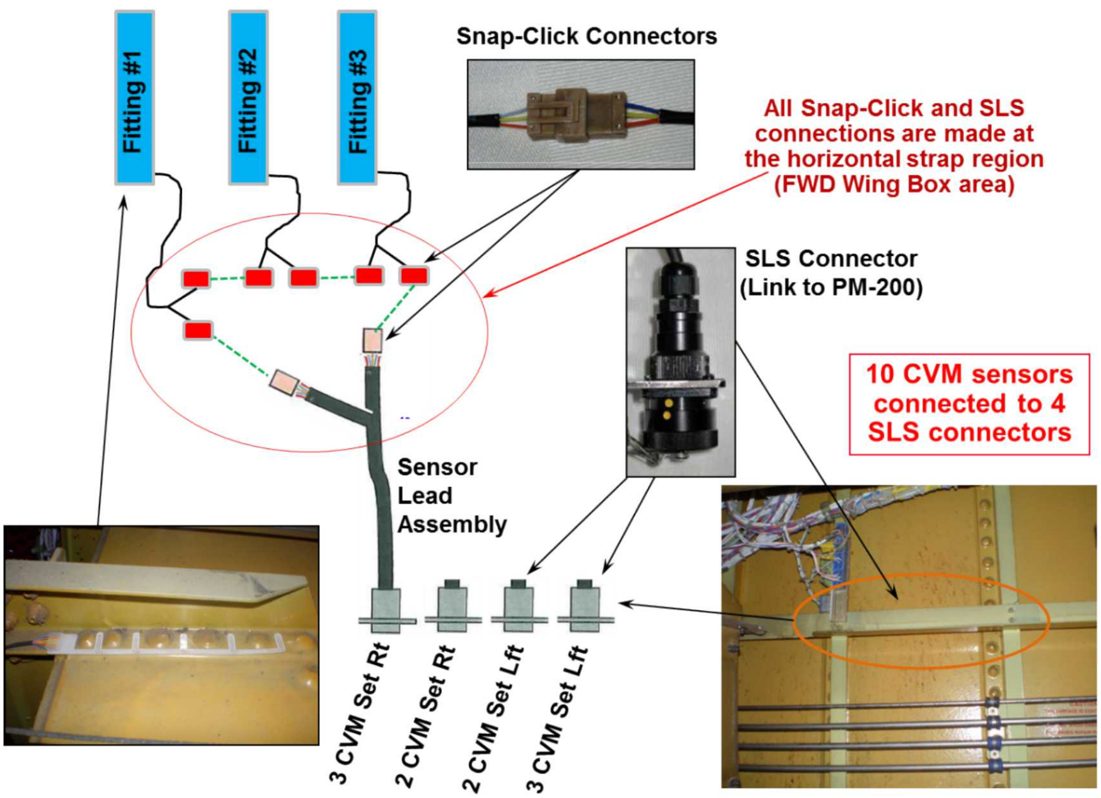


Figure 6-11: Interconnection Schematic Showing Daisychain Set-Up for Ten Wing Box Sensors and Connection to Four SLS Connectors for Data Acquisition

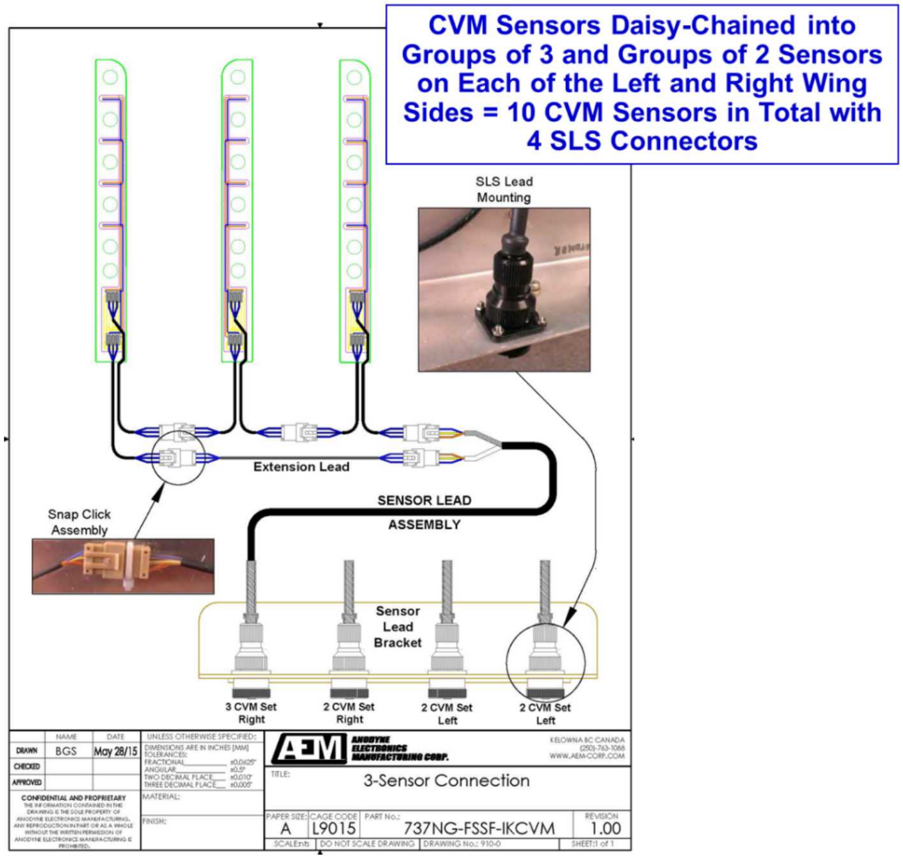


Figure 6-12: Formal Schematic of Sensor Connection into Groups

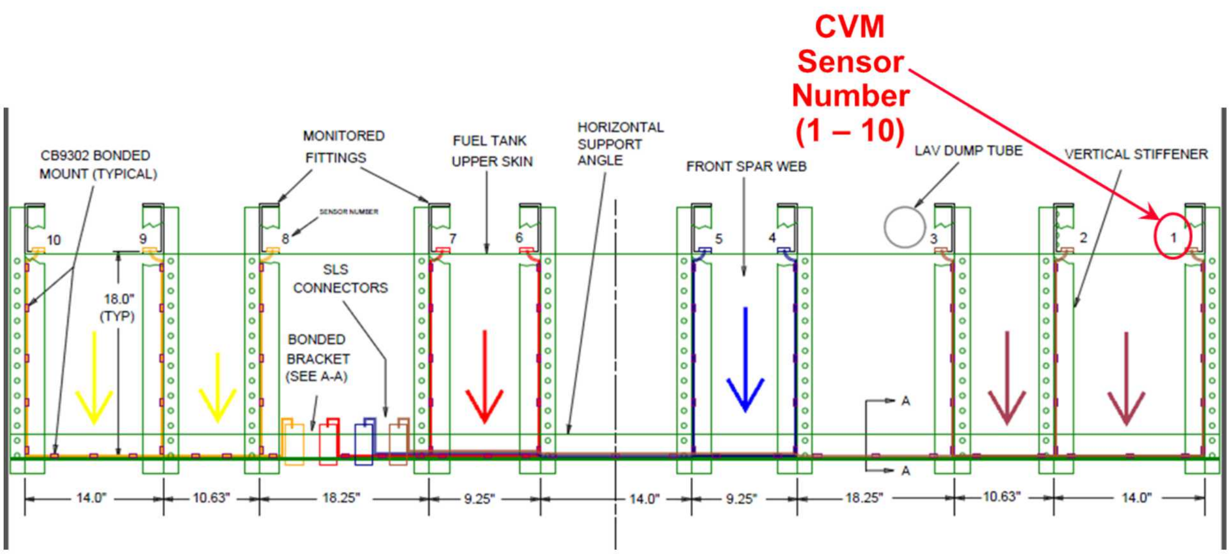


Figure 6-13: Sensor Layout on the Ten Wing Box Fittings and Routing Plan to the Four SLS Connectors

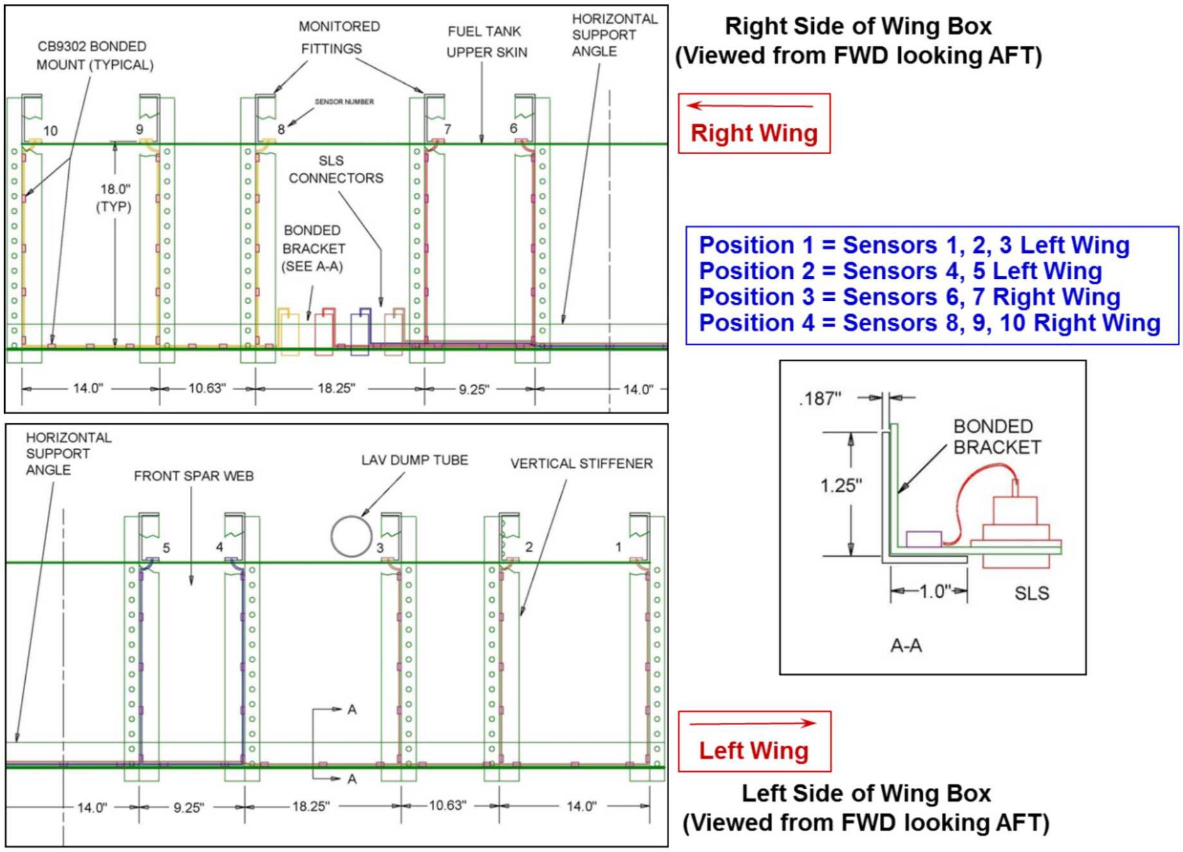


Figure 6-14: Sensor Locations and Associated Four SLS Groups

The installation of the CVM sensors is described in Figure 6-15 through Figure 6-18. The basic installation steps include:

1) Remove the rivet head sealant, fuel vapor barrier and primer in the region of the sensor installation.
2) Inspect for any existing cracks in the wing box fitting using High Frequency Eddy Current (HFEC). This will ensure that the wing box fittings have no cracks, do not need to be replaced and are good candidates for subsequent CVM monitoring.
3) Prime the surface with fresh primer to produce a good foundation for CVM installation. For applications with smooth surfaces and an intact primer in place, Steps (1) through (3) may not be necessary.
4) Prepare the surface using a light sanding of the primer with a super-fine grit sandpaper to produce a smooth surface. Clean the surface with an Acetone wipe and deionized water.
5) Apply the self-adhering CVM sensor on the wing box fitting. Apply light and localized pressure to achieve an optimal seal between the sensor and the surface being monitored.
6) Connect CVM sensor to monitoring lines using air-tight Snap-Click connectors. Small tubing and Snap-Click connectors can be customized to produce any desired daisy-chain grouping of sensors.
7) Re-apply rivet head sealant and Fuel vapor barrier as per normal specifications in the wing box region.

8) Install the set of four SLS connectors to monitor the ten wing box fitting sensors arranged in the four groupings described in Figure 6-11 through Figure 6-14. The four SLS connectors are mounted to the forward section of the wing box as shown in Figure 6-17 and Figure 6-18.
9) Connect multiple CVM sensors, as per custom manifold, to each of the four SLS connectors. The SLS connectors are placed along the forward spar of the wing box to allow for easy access from the forward baggage compartment.
10) Monitor the ten CVM sensors with the PM-200 by connecting the PM-200 device to each SLS connector.

Subsequent monitoring of the CVM sensors was conducted during an overnight stay of the aircraft at the Atlanta airport. Rapid sensor interrogation with minimum access time allowed the inspections to be completed at the airport gate during overnight parking. After the aircraft arrived at its gate from its final flight of the night, Delta personnel from the Delta-Atlanta maintenance facility (Delta Tech Ops) performed the CVM data acquisition. The basic steps in the CVM monitoring process are shown in Figure 6-19 through Figure 6-22 and include:

1) Complete routine calibration of PM-200 equipment before acquiring any data.
2) Access the SLS connectors from the forward baggage compartment.
3) Connect to each SLS connector and acquire CVM data on PM-200 device
4) Log all results. Data is stored on PM-200 for future plotting and comparisons for desired data trending. System responds with "Green Light" – "Red Light" message to indicate any cracks that are detected. Aircraft is available for its next flight.

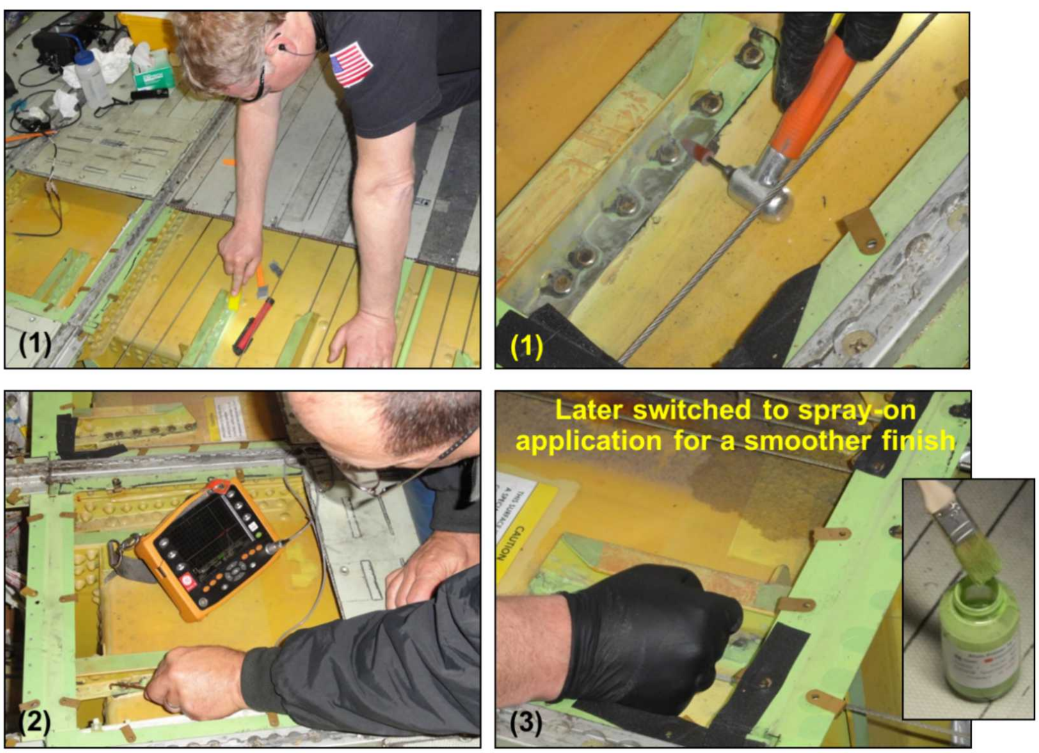


Figure 6-15: CVM Installation Steps – 1) Remove Rivet Head Sealant, Fuel Vapor Barrier and Primer, 2) Inspect for Cracks with HFEC, and 3) Re-Prime Surface

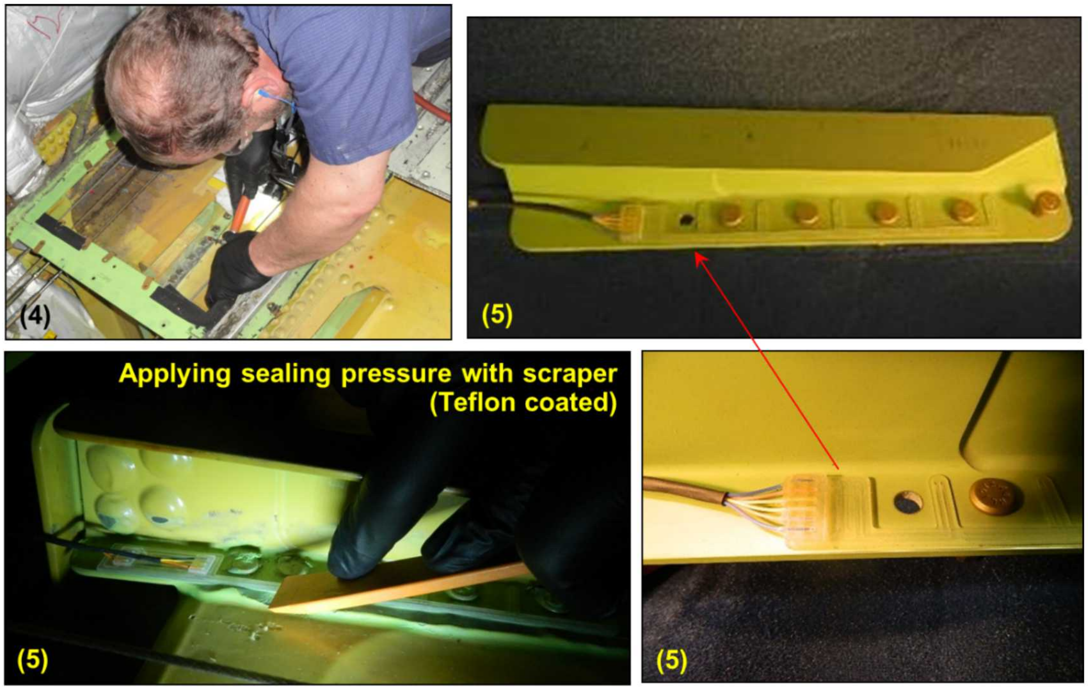
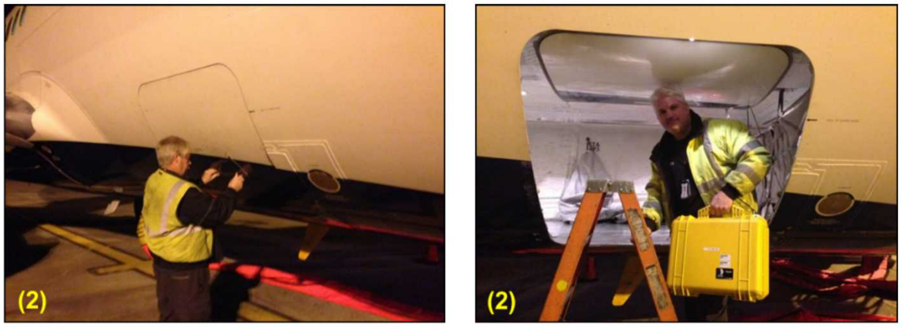


Figure 6-16: CVM Installation Steps - 4) CVM Surface Preparation (light sanding, acetone & deionized water clean), and 5) CVM Sensor Placement on Wing Box Fitting

Figure 6-17: CVM Installation Steps - 6) Connecting CVM to Monitoring Lines Using Snap-Clicks, 7) Reapplication of Rivet Head Sealant and Fuel Vapor Barrier, and 8) Installation of SLS Connector Set

Figure 6-18: CVM Installation and Monitoring - 9) Connection of Multiple CVM Sensors to Individual SLS Connectors and 10) Monitoring CVM with PM-200 Device

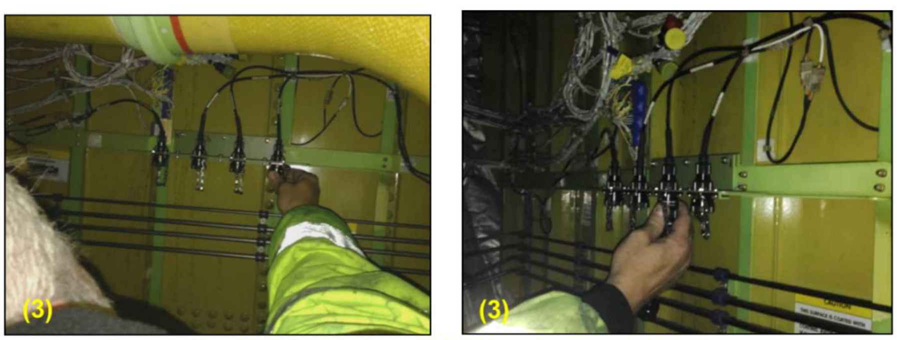
Figure 6-19: Monitoring CVM Sensors on 737NG Center Wing Box Fittings – Set-Up



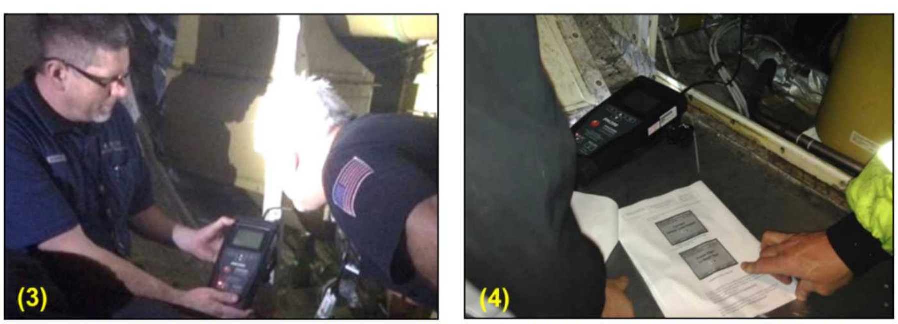
Access to SLS Connectors Through Forward Baggage Compartment

Removal of Baggage Liner to Access 4 SLS Connectors Mounted to Bulkhead

Figure 6-20: Monitoring CVM Sensors on 737NG Center Wing Box Fittings – Access to SLS Connectors through Forward Baggage Compartment



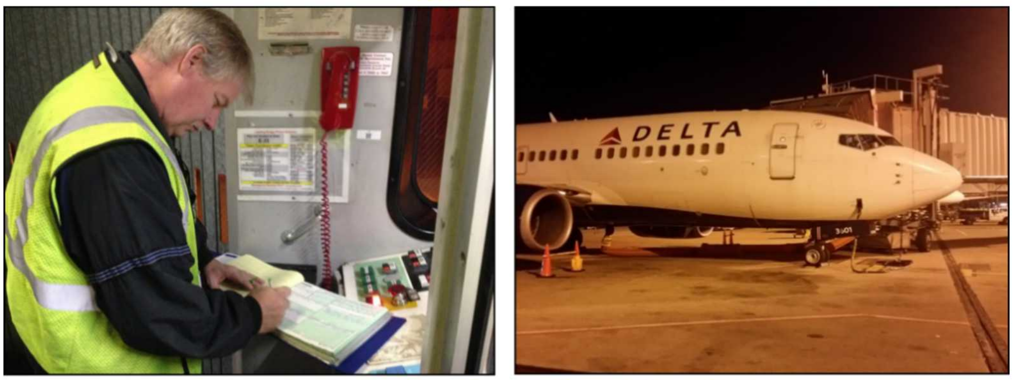
Connecting SLS Leads from PM-200 to On-Board SLS Connectors



Running PM-200 Monitoring Device to Measure dCVM Levels of Each Sensor Group

Figure 6-21: Monitoring CVM Sensors on 737NG Center Wing Box Fittings – Connecting to SLS Connectors and Acquiring Data on PM-200 Monitoring Device

Reinstallation of Forward Baggage Liner and Close-Up of Compartment



Logging Inspection Completion at Aircraft Gate

Figure 6-22: Monitoring CVM Sensors on 737NG Center Wing Box Fittings – Logging CVM Results and Closing Up Aircraft to Return to Service

The CVM system and aircraft installation process described above were completed on seven aircraft in the Delta Air Lines fleet. A summary of the flight test series follows:

- Sensors installed on 7 aircraft in Delta fleet (A/C #3601 to #3607)
- Repetitive inspections conducted every 90 days
- Goal - produce a data package with 1 to 1.5 years of monitoring (5-7 readings after installation).
  - ➢ Flight test CVM data (desired data is 5 checks for a total of 68 sensors X 5 checks = 340 data points)
  - ➢ Combine flight test data with lab performance data in the overall certification Data Package.
- Review/approval by Boeing Aviation Representatives (ARs); data presented to the FAA -
  - ➢ Current requirement is a visual inspection (assume DVI = 2" long crack)
  - ➢ Sensor designed with fingers placed between fasteners to produce 0.5" crack detection.
- Normal monitoring of CVM sensors –
  - ➢ Sensor function check: want continuity (flow) high = no gallery blockage
  - ➢ Crack detection: if dCVM (vacuum) is low = no crack
- Note that any CVM sensor failure results in a Fail-Safe Failure (false call) which will induce a site visit for eddy current inspection for confirmation. This feature prevents any erroneous data from being recorded (i.e. a missed crack).

CVM sensor installations for flight testing:

- Sensors installed on 10 wing box fittings on 7 aircraft in Delta fleet (A/C #3601 to #3607).
- Aircraft were available for 1 ½ to 2 days during a 7 day check for sensor installation.
- Two instances where faulty sensor installations occurred with no remaining time to remove and install new sensor – sensors were removed from data acquisition plan. Final total of 68 sensors installed and monitored for the flight test program.
- Post-installation failures occurred on 3 aircraft after several monitoring intervals (1 sensor each).
- Failure rate (excluding initial faulty installations) = 2.9% (66 out of 68 sensors functioning).
- Total data points acquired = 385 (out of a possible 398) = 96.7% data success rate.

While the failure rate was determined to be within "normal" failure rates requiring sensor replacement, it was noted that each of the three sensor failures produced borderline-acceptable vacuum levels during initial check of the sensor. Normally, the sensors would have been replaced, however, the aircraft was completing its short, seven-day maintenance and was returning to service. As a result, there was no time to remove and install new sensors. So, the three CVM failures were attributed to the difficult installation coupled with challenging surface prep (due to existing coatings) and access time constraints. It was noted early on that the brush application of the primer left an uneven surface with very small "grooves" in the coating. This made it more challenging to produce an optimal seal with the CVM sensor. A spray-on primer was used on the latter four aircraft. This produce a smoother surface and eliminated any future sensor failures on these aircraft.

Summary of Data Acquired During Delta Air Lines Flight Test Program							
Aircraft	CVM Readings (Number of Monitors)	Number of Sensors at Beginning	Number of Sensors at End	Number of Data Points	Date Range for Flight Operation	Duration of Monitoring	Notes
3601	7	10	9	65	2/14 to 6/15	15 months	Sensor failed at 2nd check (6 months)
3602	6	9	9	54	2/14 to 4/15	14 months	No time to replace faulty CVM installation (9 sensors)
3603	5	10	10	50	3/14 to 5/15	14 months	All sensors functioned throughout
3604	5	10	9	46	3/14 to 3/15	12 months	Sensor failed at first check - faulty CVM installation
3605	6	9	9	54	3/14 to 8/15	17 months	No time to replace faulty CVM installation (9 sensors)
3606	6	10	9	56	3/14 to 5/15	14 months	Sensor failed at 2nd check (6 months)
3607	6	10	10	60	4/14 to 10/15	18 months	All sensors functioned throughout

Table 6-2: Summary of CVM Installations and Data Acquired During Flight Testing

Results from the first two years of operation for the seven Delta aircraft are presented in Figure 6-23 through Figure 6-36. The continuity values from each sensor is shown first followed by the dCVM values. Note that in all cases the continuity should be a high number, at least above 1,000; while the dCVM levels should be low (for no crack detection; threshold for crack detection was determined to be 10). In all cases, the continuity values were at least 2,000 for all readings (i.e. no blockage) and the dCVM values were less than 2 (i.e. no crack detection). In addition to the general high continuity (no flow blockage) and low dCVM (no cracks present) readings, the data was observed to be repeatable and consistent during the monitoring period. While the initial goal was to acquire 18 months of operational data from the seven sensor networks, Delta Air Lines continued to monitor these aircraft for additional months. Several examples of extended data results are also presented here. Figure 6-37 through Figure 6-44 show additional CVM data points for aircraft 3601, 3602, 3603 and 3605, respectively. The 90 day inspection cycles for obtaining CVM data was continued so each data point represents approximately 90 days of operation. Thus, the 22 to 23 data points in these figures represent almost 6 years of proper CVM sensor operation on the 737 Wing Box fitting installations.

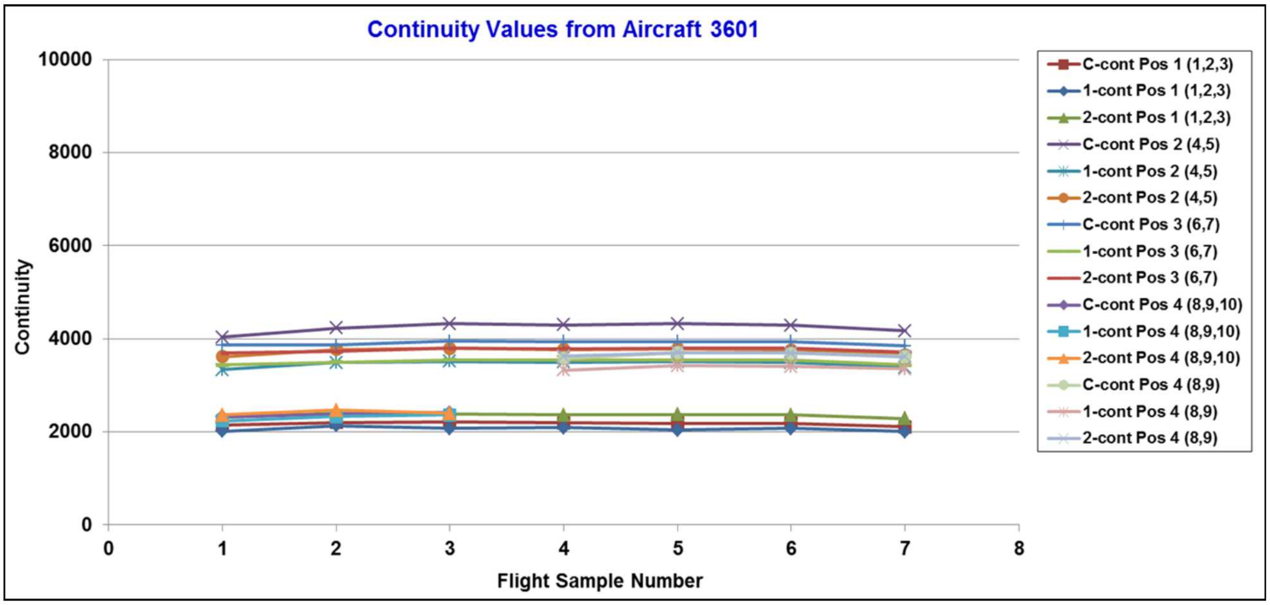


Figure 6-23: CVM Sensor Continuity Check for Delta Air Lines Aircraft 3601

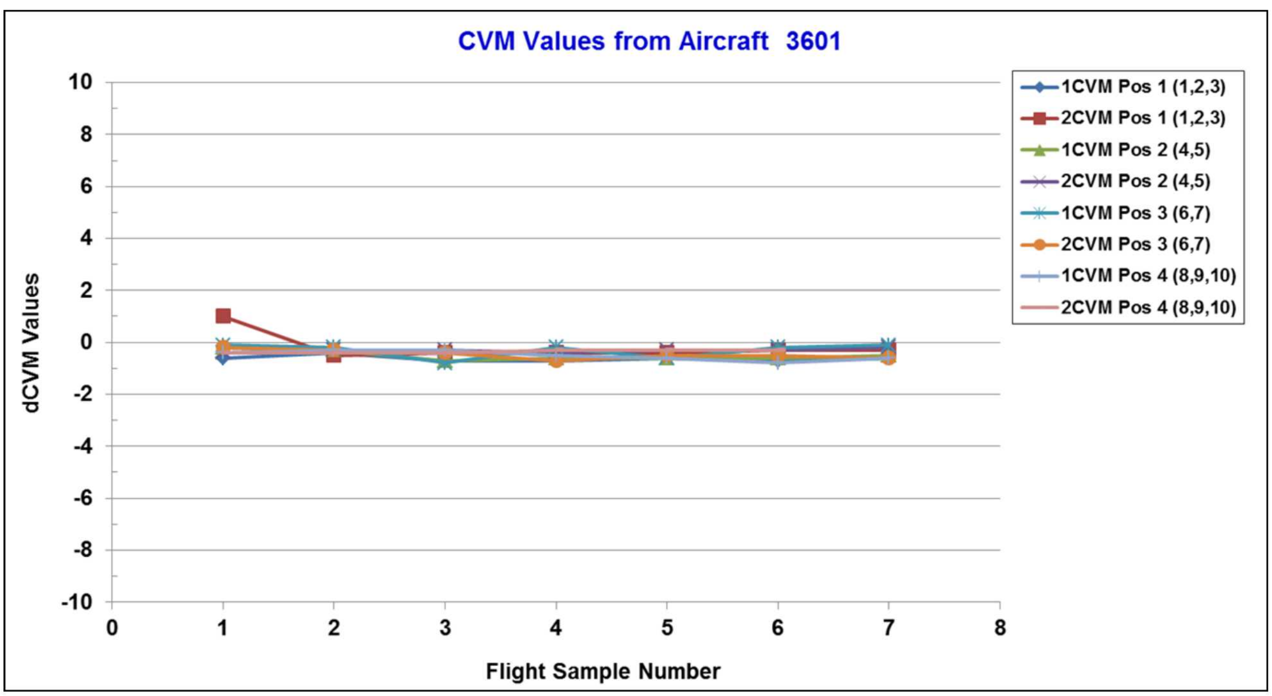


Figure 6-24: CVM Sensor dCVM Check for Delta Air Lines Aircraft 3601

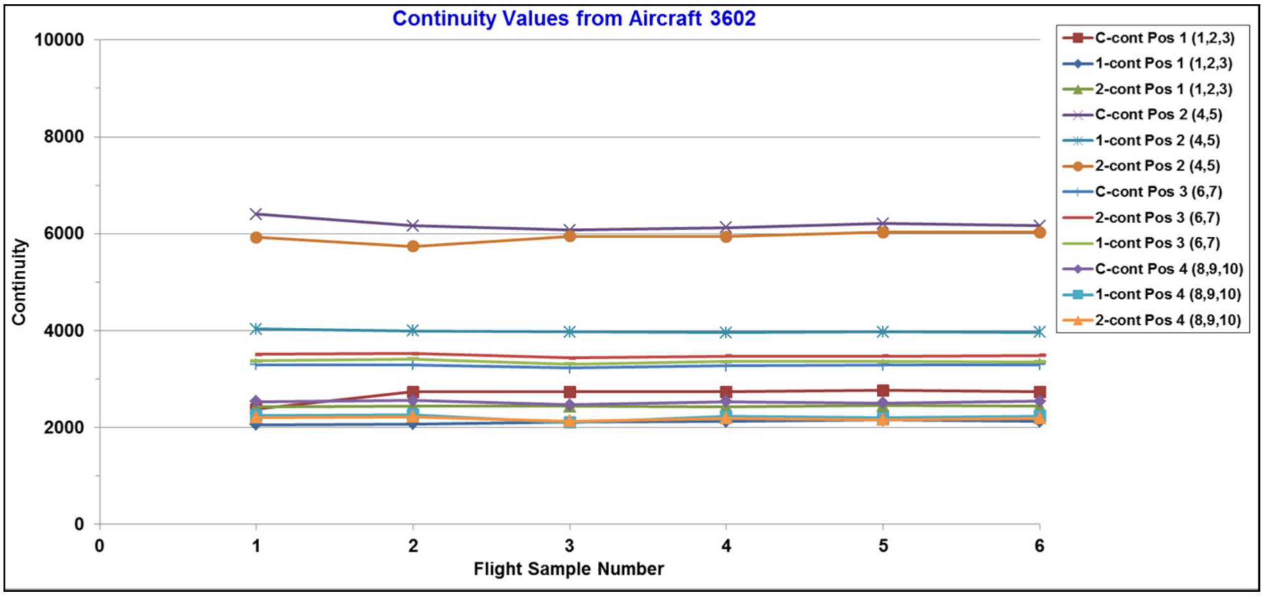


Figure 6-25: CVM Sensor Continuity Check for Delta Air Lines Aircraft 3602

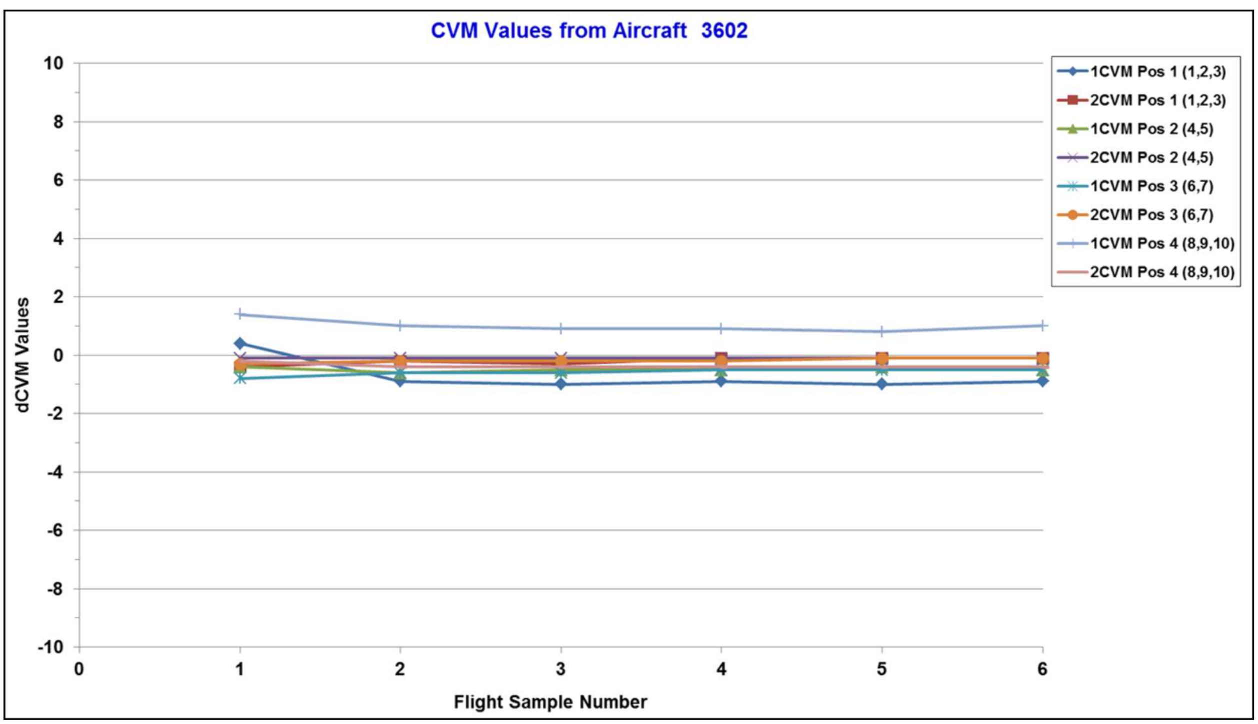


Figure 6-26: CVM Sensor dCVM Check for Delta Air Lines Aircraft 3602

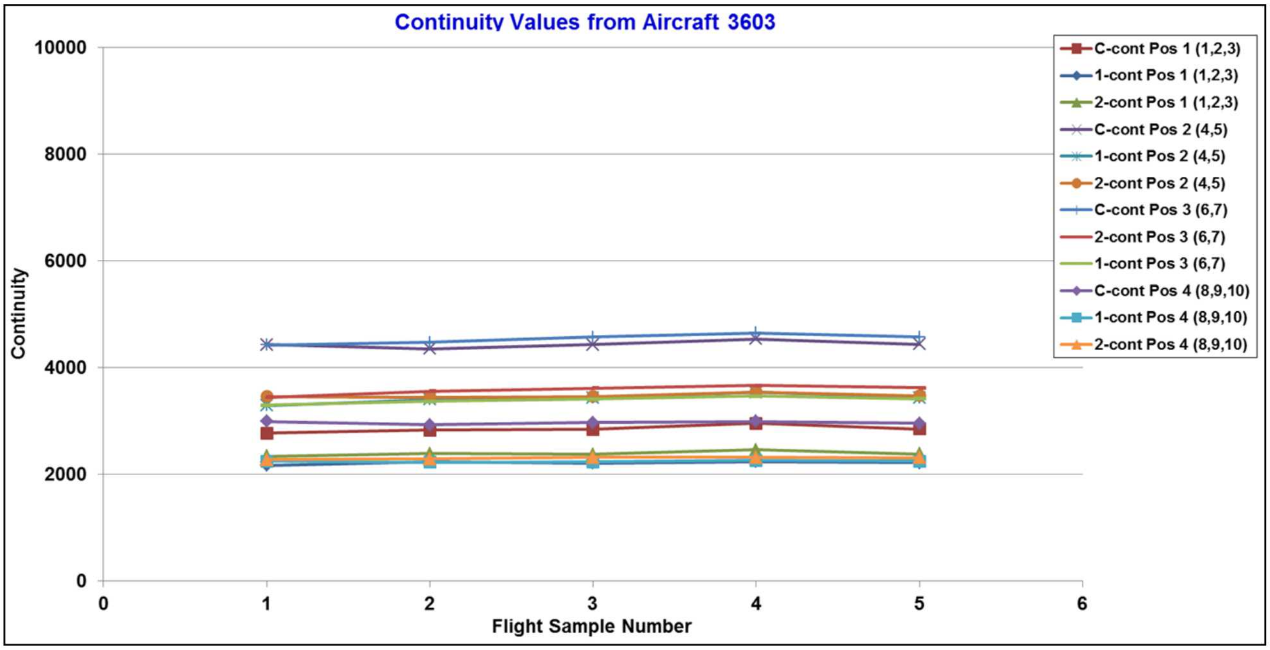


Figure 6-27: CVM Sensor Continuity Check for Delta Air Lines Aircraft 3603

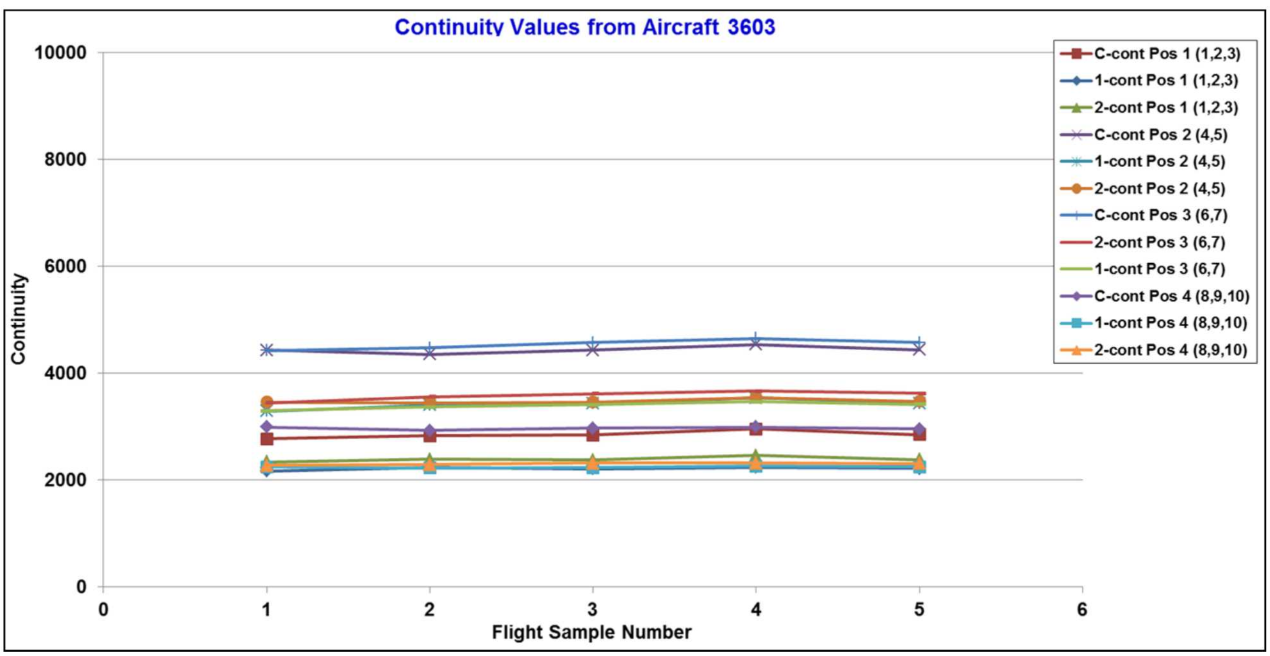


Figure 6-28: CVM Sensor dCVM Check for Delta Air Lines Aircraft 3603

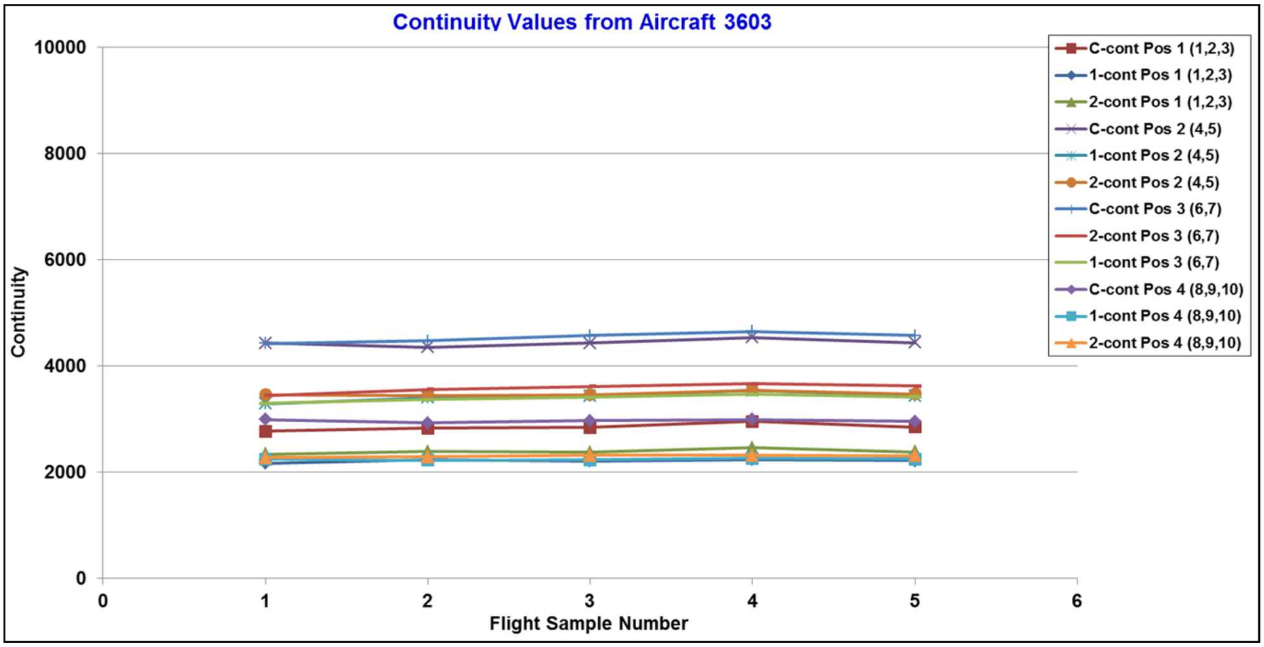


Figure 6-29: CVM Sensor Continuity Check for Delta Air Lines Aircraft 3604

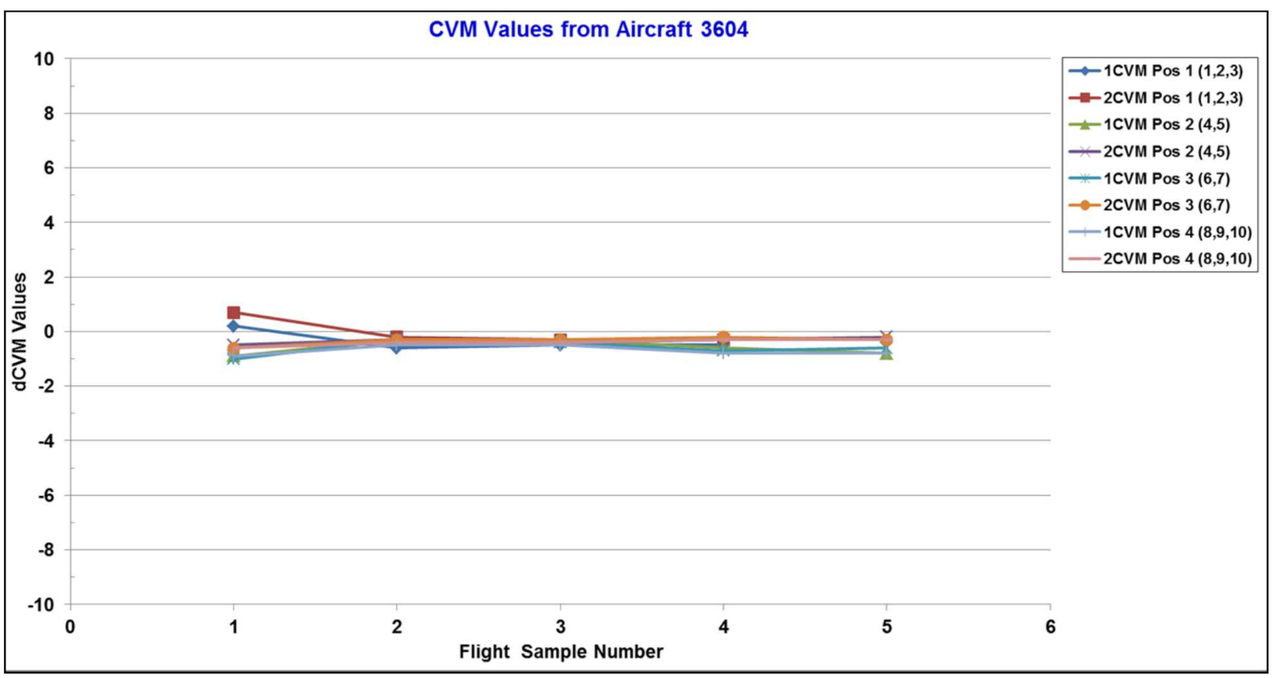


Figure 6-30: CVM Sensor dCVM Check for Delta Air Lines Aircraft 3604

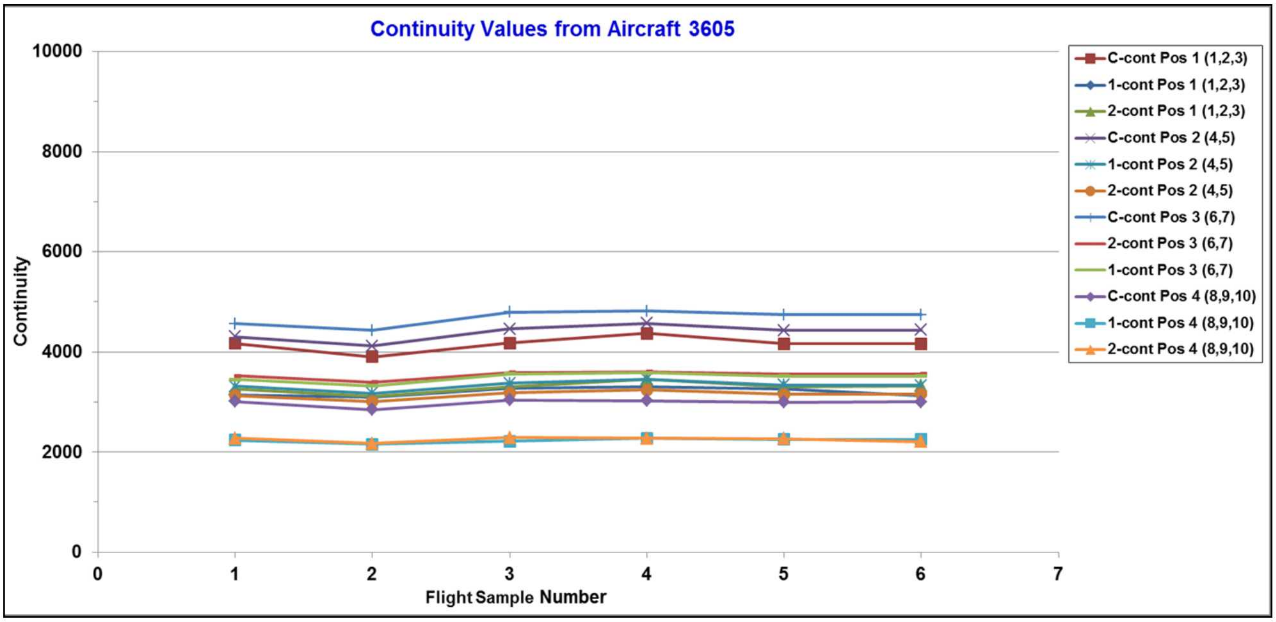


Figure 6-31: CVM Sensor Continuity Check for Delta Air Lines Aircraft 3605

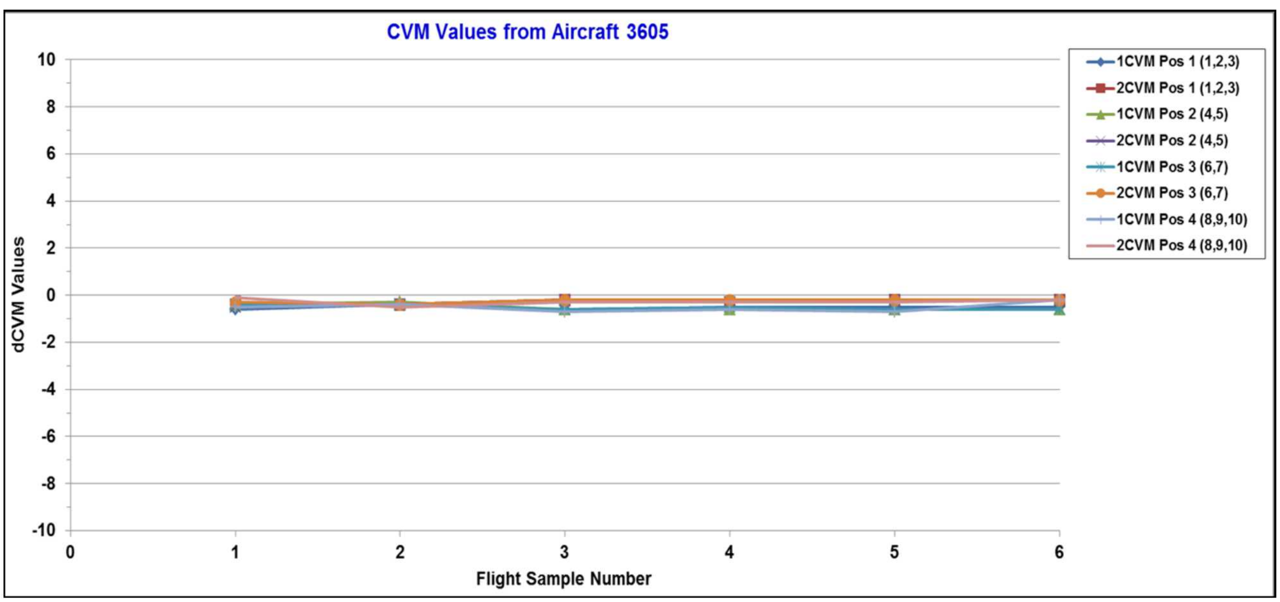


Figure 6-32: CVM Sensor dCVM Check for Delta Air Lines Aircraft 3605

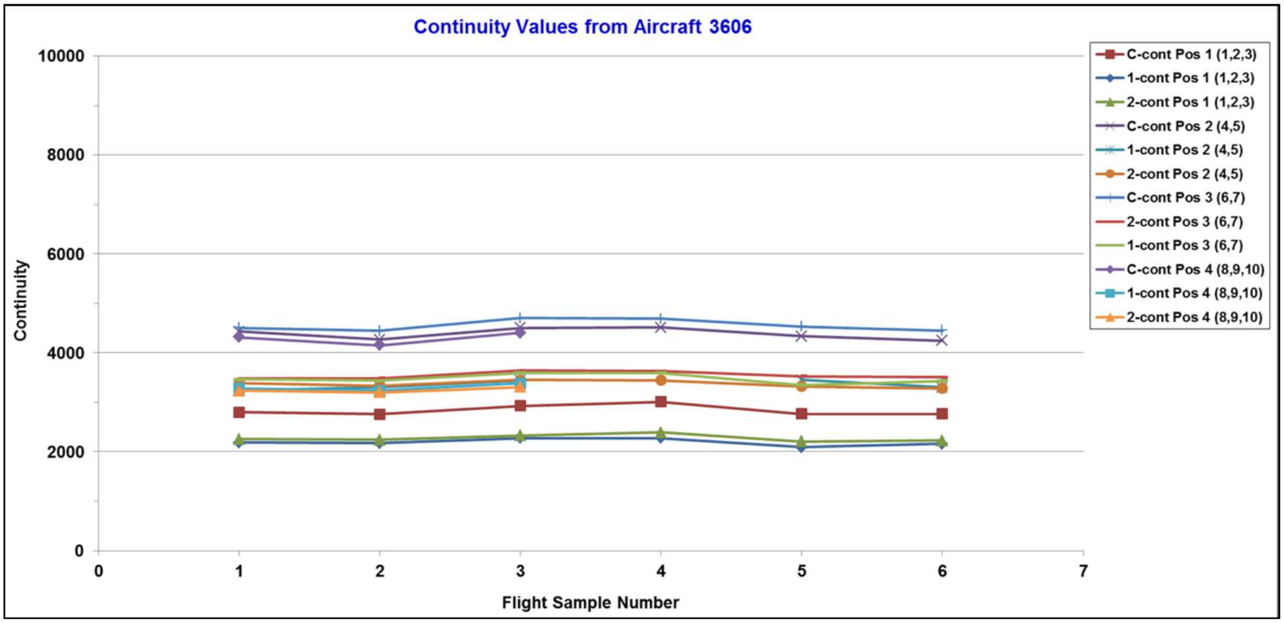


Figure 6-33: CVM Sensor Continuity Check for Delta Air Lines Aircraft 3606

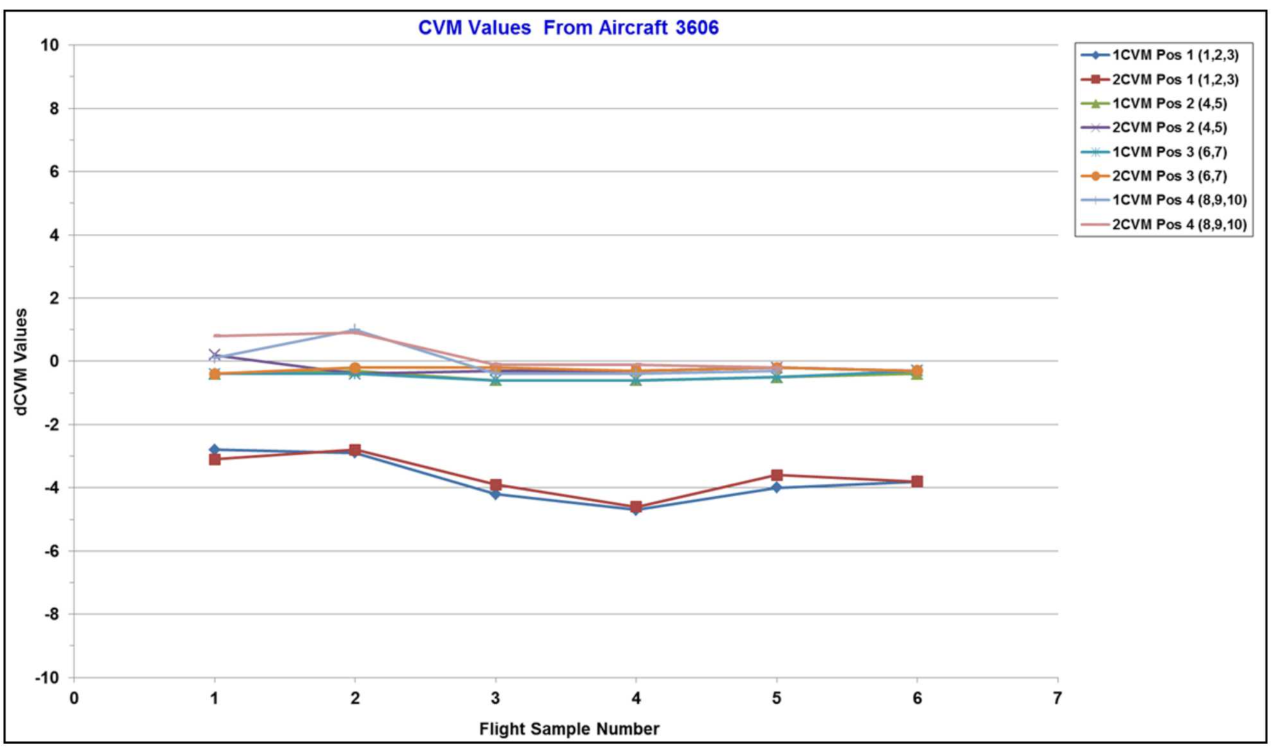


Figure 6-34: CVM Sensor dCVM Check for Delta Air Lines Aircraft 3606

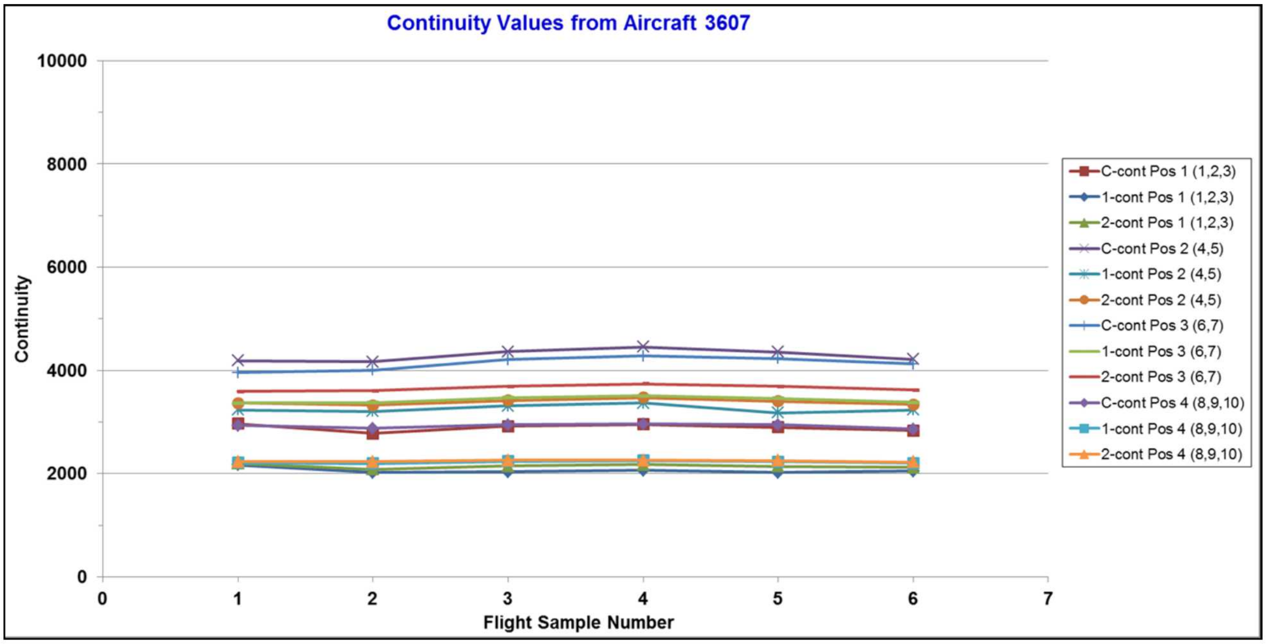


Figure 6-35: CVM Sensor Continuity Check for Delta Air Lines Aircraft 3607

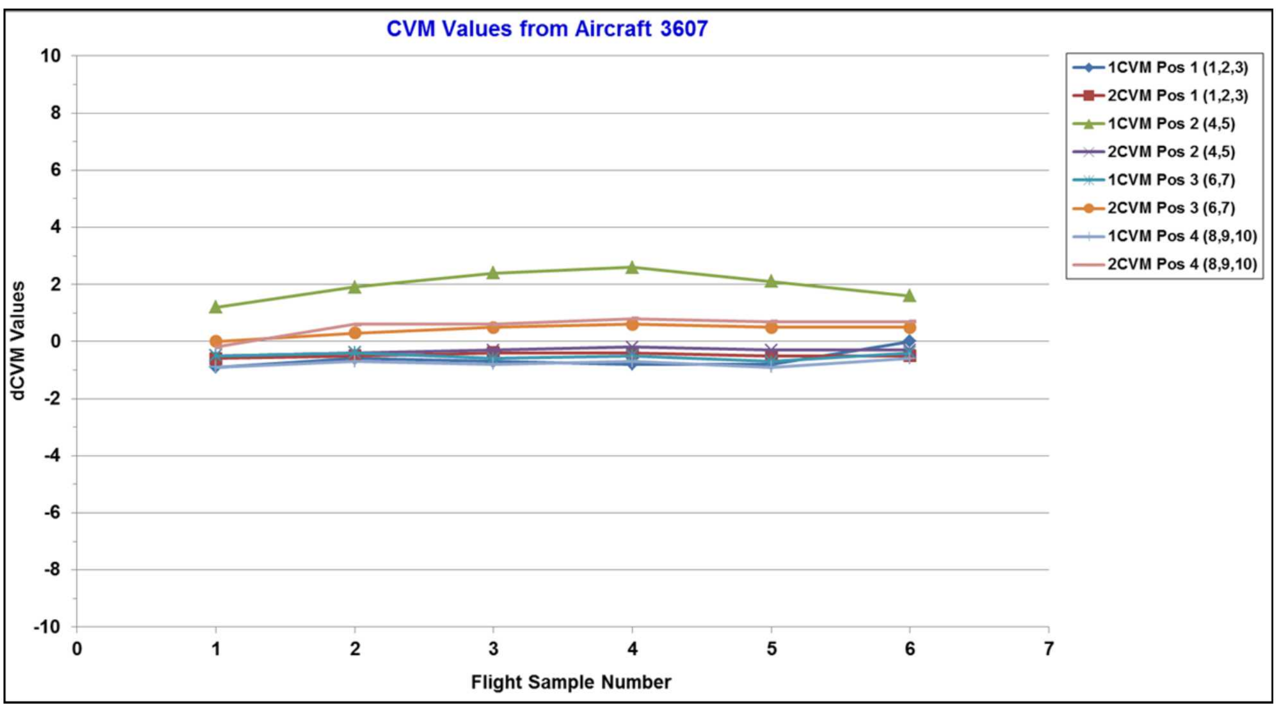


Figure 6-36: CVM Sensor dCVM Check for Delta Air Lines Aircraft 3607

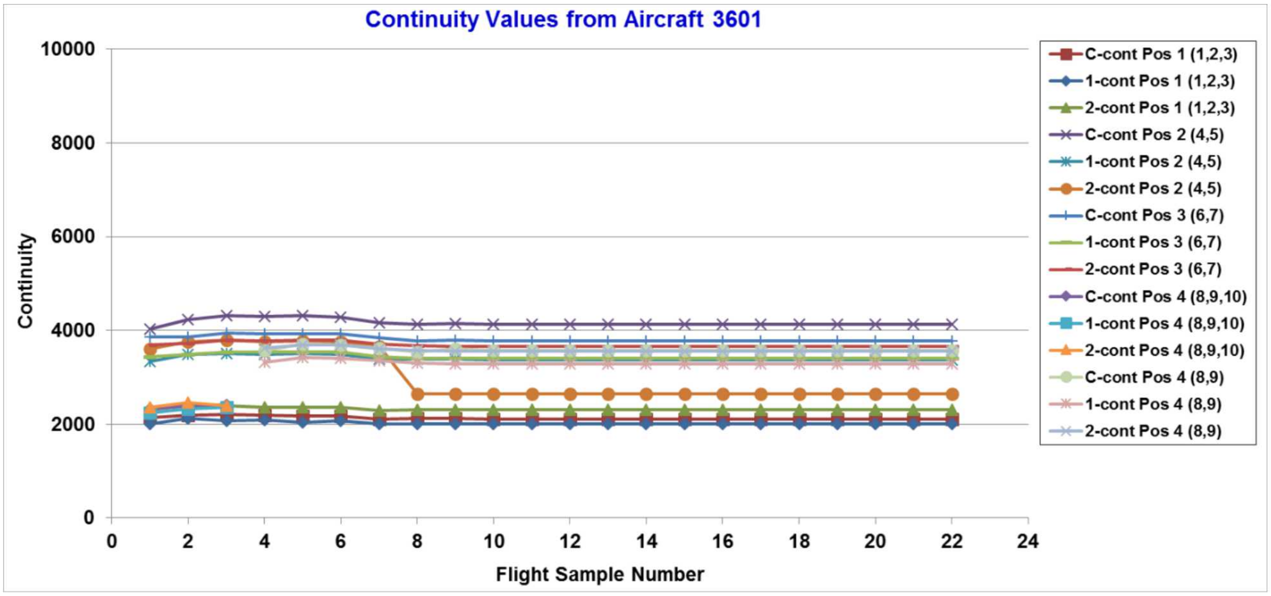


Figure 6-37: Long Term CVM Sensor Continuity Check for Delta Air Lines Aircraft 3601

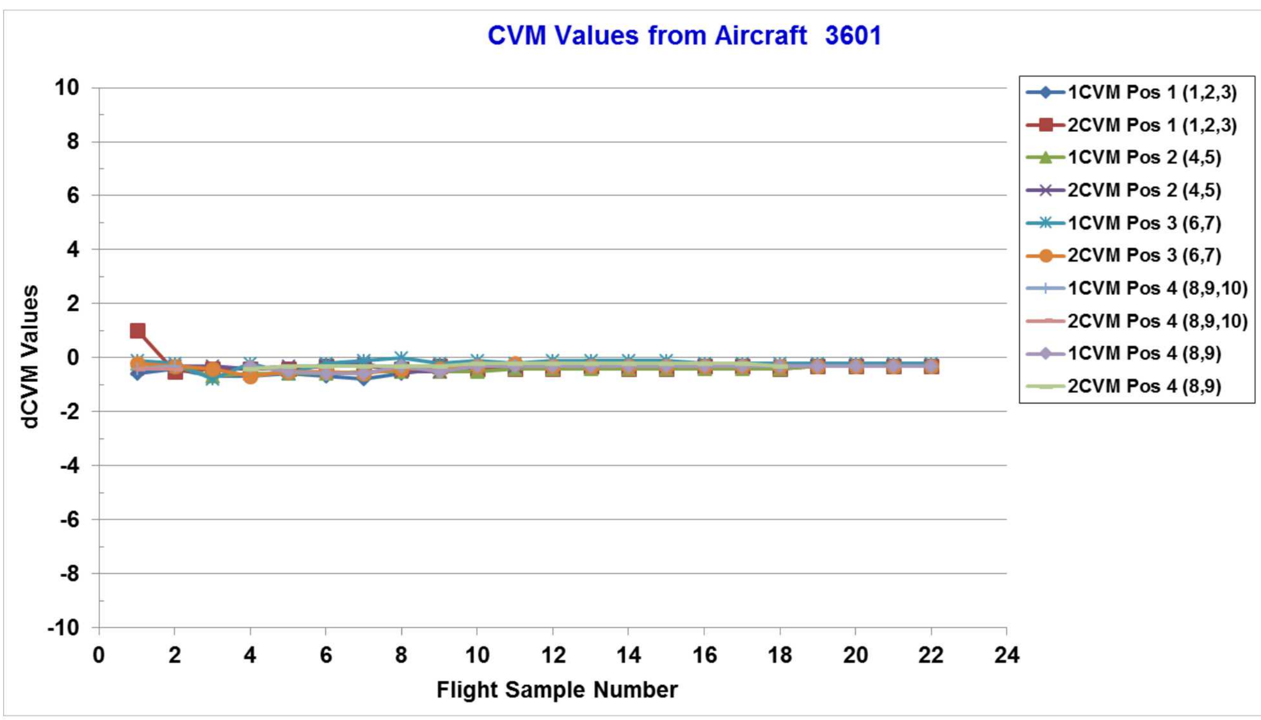


Figure 6-38: Long Term CVM Sensor dCVM Check for Delta Air Lines Aircraft 3601

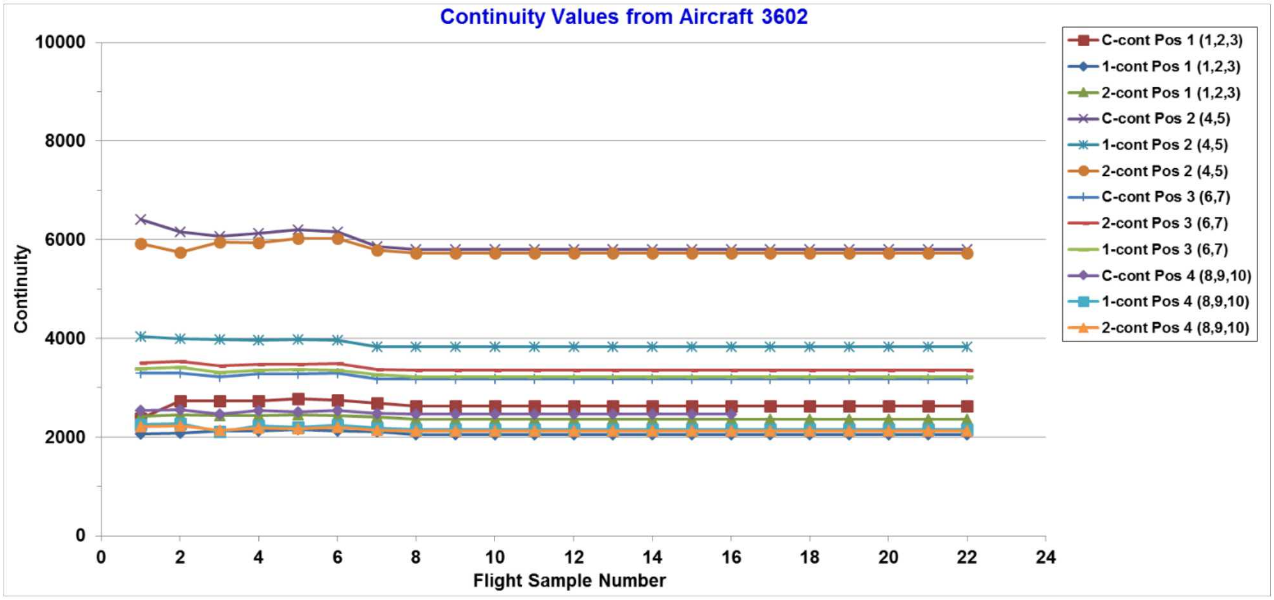


Figure 6-39: Long Term CVM Sensor Continuity Check for Delta Air Lines Aircraft 3602

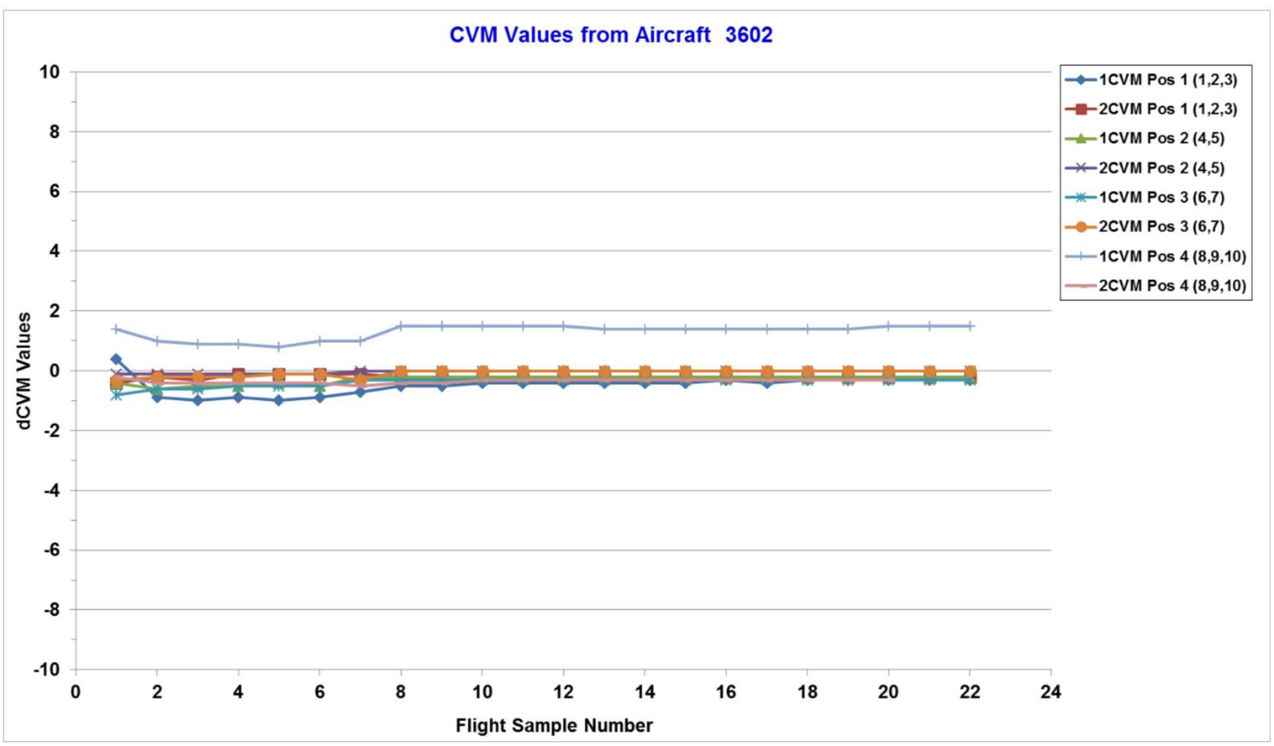


Figure 6-40: Long Term CVM Sensor dCVM Check for Delta Air Lines Aircraft 3602

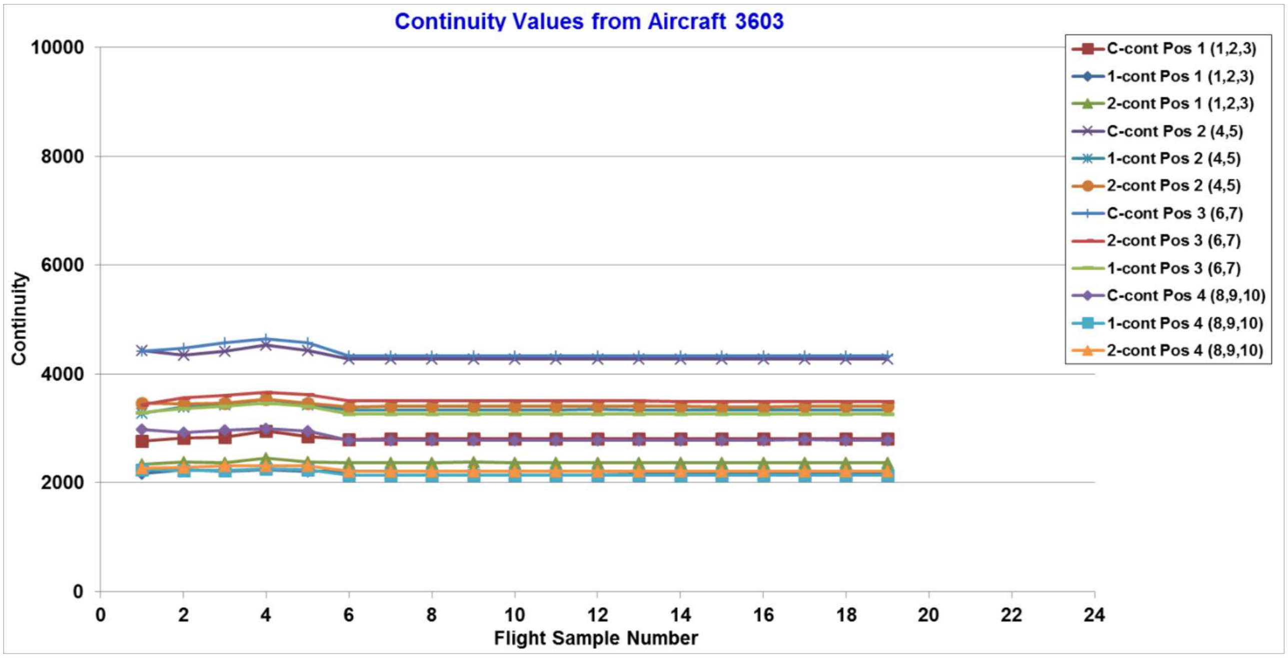


Figure 6-41: Long Term CVM Sensor Continuity Check for Delta Air Lines Aircraft 3603

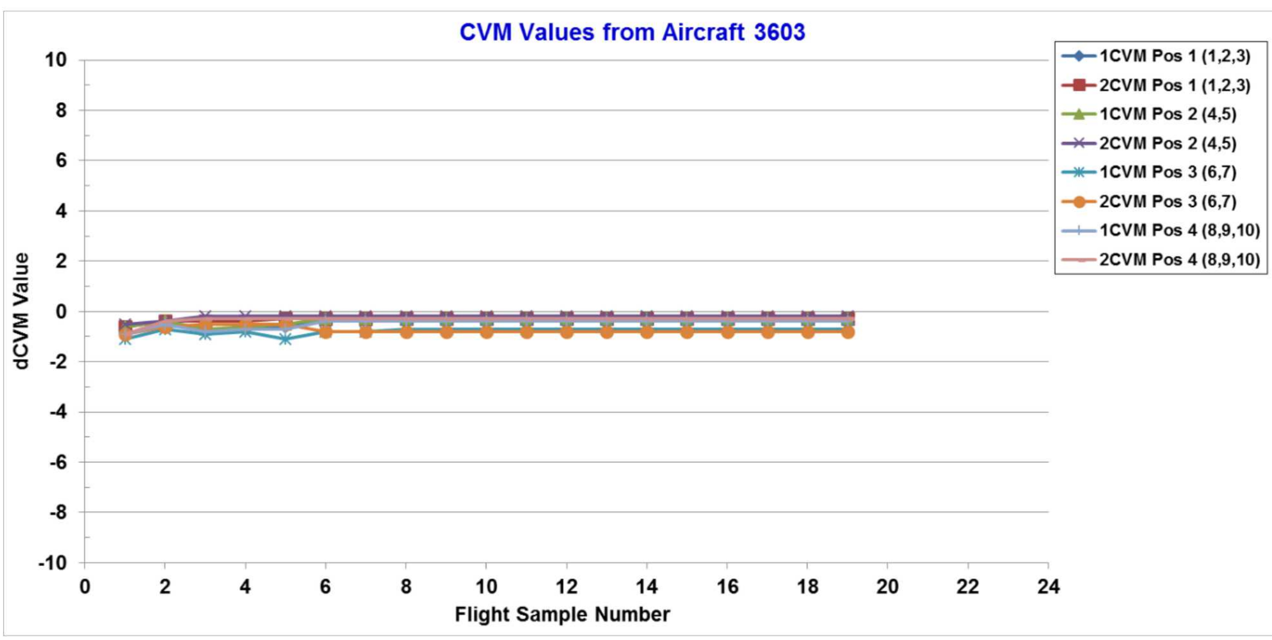


Figure 6-42: Long Term CVM Sensor dCVM Check for Delta Air Lines Aircraft 3603

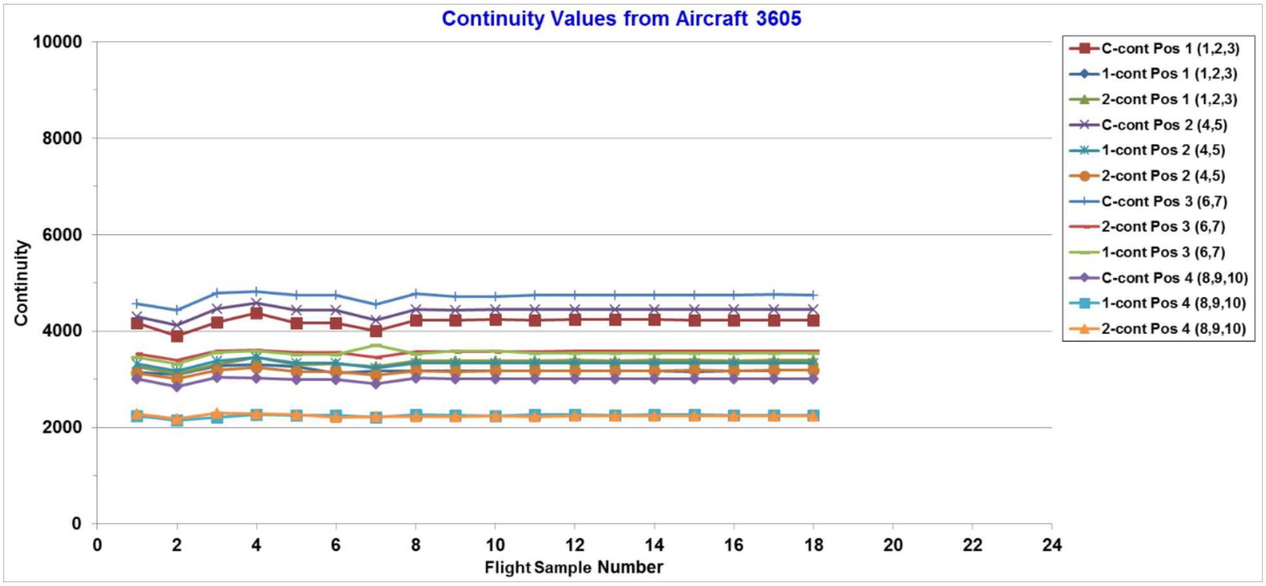


Figure 6-43: Long Term CVM Sensor Continuity Check for Delta Air Lines Aircraft 3605

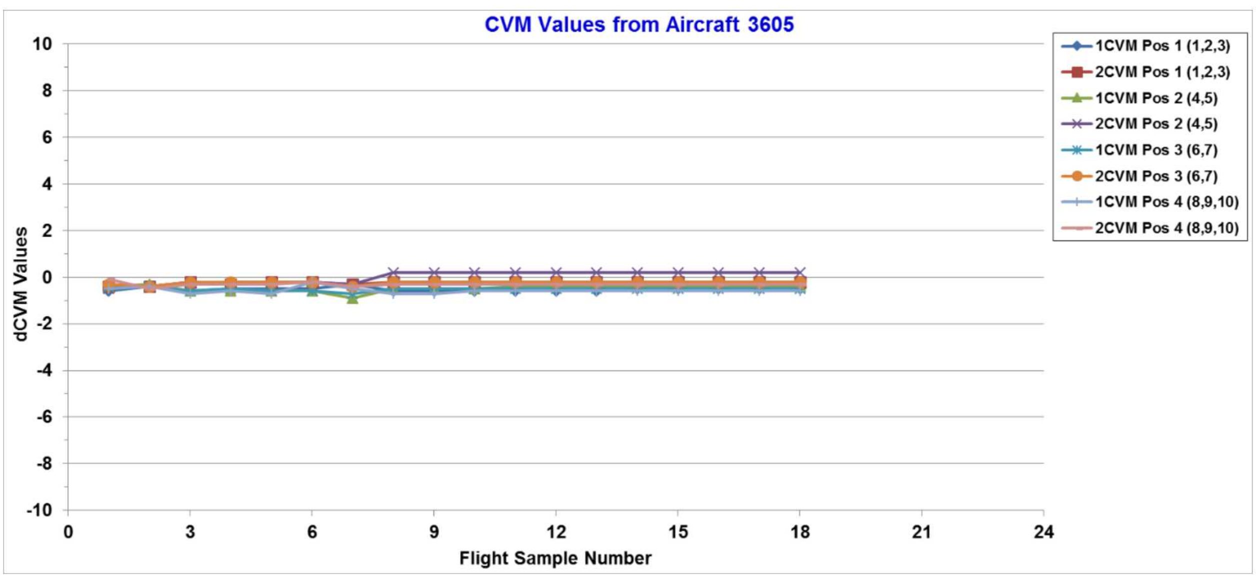


Figure 6-44: Long Term CVM Sensor dCVM Check for Delta Air Lines Aircraft 3605

Overview of CVM Operation from Flight Test Series - Flight testing of the CVM system helped prove the technology and produce the following general items in support of routine SHM use on aircraft.

- Multiple aircraft applications addressed

- Comprehensive performance assessments completed – sensitivity, reliability, durability
- Over 50 combined years of successful operation on flying aircraft
- Formal approval from aircraft manufacturers and aviation regulators
- Reached routine use on aircraft.

This chapter presents the results from 90 CVM sensor installations which were monitored for 5 years on 14 commercial aircraft. These flight test programs resulted in the accumulation of over 1.5 million hours of successful operation (representing 50 combined years of operation on flying aircraft) and the acquisition of over 3,000 sensor monitoring data points. Two different flight test series were conducted to explore general installation, operation and monitoring of CVM sensors by airline personnel while also assessing the durability of the sensors when exposed to real flight conditions. Such flight tests provide critical information about the long-term performance, reliability, repeatability and continued airworthiness of flying CVM systems. Other important issues for CVM adoption that were also studied during the flight tests include:

- Complete SHM indoctrination and training for Delta personnel.
- Complete indoctrination of FAA personnel who were involved in the certification process.
- Completion of formal hardware installation and operation procedures.
- Completion of formal modifications to integrate SHM into an airline maintenance program.
- Assessment of an aircraft maintenance depots' ability to safely adopt SHM and the FAA support needed to ensure airworthiness.

Data from the monitored sensors showed that, in all cases, the continuity numbers maintained the desired high levels while the dCVM levels remained in the low numbers associated with no crack detection. In addition to the general high continuity (no flow blockage) and low dCVM (no cracks present) readings, the data was observed to be repeatable and consistent during the monitoring period. Results from the flight test series demonstrated the ability of CVM sensors to: 1) operate successfully on operating aircraft over long periods of time, 2) produce consistent data and 3) be properly installed and monitored by airline personnel. Flight testing is a key element, in combination with controlled laboratory tests, in establishing proper inspections (or maintenance actions) needed to avoid catastrophic failure during the operational life of the airplane.

References

6.1 NATO Research & Technology Organization, "Flight Test Techniques Series," AC/323(SCI-FT3)TP/74, July 2005.
6.2 Federal Aviation Regulations (FAR) Part 25 - "Airworthiness Standards: Transport Category Airplanes", U.S. Department of Transportation, Federal Aviation Administration (FAA).
6.3 Federal Aviation Administration Advisory Circular, "Damage Tolerance and Fatigue Evaluation of Structure," AC §25.571-1, January 2011.
6.4 Pettit, D., Turnbull, A., "General Aviation Reliability Study," NASA/CR-2001-210647, February 2001.
6.5 Padfield, G. D., "Flight Testing for Performance and Flying Qualities," Advisory Group for Aircraft Research & Development LS-139, AD-A155 946, January 1

CHAPTER 7

7.0 Adoption of CVM SHM System by Delta Air Lines

Delta Air Lines has investigated Structural Health Monitoring and CVM sensors for many years [7.1 - 7.3]. Two main hurdles were identified as an impediment to incorporation: 1) establishing a proper business case, and 2) lack of regulatory guidance and a need for industry education, and 3) the time required to obtain approval for routine use of SHM on commercial aircraft. This program was established to join OEMs, regulators, airlines and research agencies and address all three of this impediments. Detailed studies on a wide array of applications have identified good business cases where inspection access difficulties, short repeat inspection intervals, heightened interest in more frequent structural monitoring, and inspection complexity have produced positive cost advantages of deploying SHM solutions. Items (2) and (3) above have been addressed via streamlining and precedent-setting brought about by programs such as this, along with a number of formal, regulatory and industry documents that have been produced to comprehensively guide the safe adoption of SHM practices [7.4 – 7.15].

CVM technology has been widely researched, analyzed, tested and even incorporated into Boeing's general practices. Mainstream adoption through formal, approved, routine use was the logical, last step and thus, the genesis of this program. Towards that end, Sandia National Laboratories, in concert with the FAA, formed a partnership with Delta Air Lines, Boeing, Anodyne Electronics Manufacturing Corp., Structural Monitoring Systems, to safely utilize CVM sensors. The goal was to move beyond the traditional prototype field testing completed in the first decade of 2000 and move into mainstream, industry-wide adoption of SHM. The information and process stemming from this program was used to help produce the industry and regulatory guidance that will enable widespread adoption of SHM across the commercial aviation industry.

As described in Chapter 6, Comparative Vacuum Monitoring (CVM) sensors were applied to ten wing center section shear fittings, a known area of cracking, on seven B737-700s on the Delta Air Lines fleet. This flight test program was designed to provide ample data which can be used to provide an approval basis for maintenance program changes. The passive CVM SHM system has been flown since February 2014 and periodic interrogation has occurred through connector access in the cargo bin. The data was downloaded during overnight checks, on a 90-day repetitive schedule. The objective was to combine the flight data from the sensors and the performance tests also described above to generate produce the data package needed for formal approval of SHM usage in lieu of traditional NDI processes.

7.1 Integration of SHM Systems into Airline Maintenance Programs

The maintenance program instituted by each air carrier is the means used by operators to ensure the proper performance and long-term reliability of their aircraft. Maintenance programs are intended to produce the maximum aircraft availability while ensuring compliance with FAA regulations. Specifically, the maintenance programs, which are based on the manufacturer's instructions for continued airworthiness, seek to guarantee the safety and reliability of all aircraft systems and structures, repair any damage or operational problems identified, and accommodate continuous improvements to enhance reliability or advance aircraft designs. The maintenance program must be modified to accommodate the unique operation, use, and maintenance associated with SHM systems. In turn, SHM systems can help carriers achieve their goal of increasing the usage of their aircraft. Currently, some aircraft experience over 5,000 flight hours in a year placing added emphasis on cost-effective and streamlined maintenance practices.

Operators organize their inspection and maintenance tasks in order to achieve compliance with regulations and OEM recommendations while maximizing aircraft availability. The various checks associated with general aircraft maintenance are as follows:

- Walk Around – visual checks conducted prior to each flight
- Service Checks – brief checks conducted every several days to service consumable items like fluids and to check for wear
- A-Checks – scheduled line maintenance check conducted every 25-40 days
- B-Checks - scheduled line maintenance check conducted every 45-75 days
- C-Checks - detailed maintenance and inspection visit conducted every 12-15 months
- D-Checks – heavy maintenance visit or complete aircraft overhaul conducted every 2-5 years.

The intervals between services are dependent upon aircraft utilization, flight cycles, and required aircraft maintenance tasks. Activities up through B-Checks can normally be accomplished during overnight stays for the aircraft. C-checks can take up to one week to complete while D-Checks require approximately one month to complete. Operators may choose to implement their maintenance activities in block, segmented, phased, or continuous maintenance visits. These options allow the various maintenance tasks to be broken into different intervals and completed in segments over the required interval.

The objectives of a maintenance program are: a) to ensure realization of the inherent safety and reliability levels of the equipment; and b) to restore safety and reliability to their initial levels when deterioration has occurred. The application of SHM methods provides the potential to reduce aircraft maintenance tasks and down time but the promise of new technology must always be reviewed in light of airworthiness compliance issues. The effects of SHM on the Instructions for Continued Airworthiness (ICA) must be addressed whenever a commercial aircraft application is pursued and associated modifications to the maintenance program are made.

The scope of the maintenance programs include three major areas: 1) scheduled maintenance tasks including inspections, function checks and other maintenance based on time or flight cycle limitations or other prescribed intervals, 2) unscheduled maintenance tasks that are based on the findings from scheduled maintenance or that arise from unforeseen events (e.g. high loads, bird strike, hard landing, over-temperature condition), and 3) maintenance requirements for major components including engine overhaul, propeller overhaul, and airframe maintenance. The maintenance manual includes instructions on what to do, when to do it, how to do it, and checks to ensure that the work was done properly.

With respect to SHM utilization, Delta Air Lines is pursuing an operator perspective that includes:

- Initially: alternate inspections of difficult to access areas.
    - Inconvenient maintenance visits such as short repeat inspection intervals.
    - Hotspot monitoring through Alternate Means of Compliance (AMOC).
- Medium term:
    - Provide early warning of issues.
- Future: Condition Based Maintenance & Crack Monitoring.
    - 'Smart Signal' for engines.
    - Allow time for OEM support.
    - Two main hurdles to implementation.
- Conduct sweep of all maintenance activities to identify possible SHM applications. Utilize widespread use approach when generating SHM business cases (payback).
- Continue regulatory interface to help induce regulatory guidance. Aid SHM education process.
    - Pursue proof-of-concept and SHM approval programs such as the wing box fitting to demonstrate viability of SHM systems and move into formal, routine use.
- Utilize experience with SHM to allow Delta to help write the guidance/blueprint for SHM certification.

Delta lists its reasons for pursuing SHM as follows:

- Airlines/MRO are under constant pressure to reduce costs.
- Many sensor technologies appear ready for implementation.
    - Not implemented in industry on wide-scale which would demonstrate an airline/MRO cost benefit analysis, use of SHM in lieu of traditional NDI, generate necessary changes in an airline's maintenance program and produce certification guidance in regulations.
- FAA sponsored this program to move SHM from 'prototype' status to 'mainstream' use.
- Partnership included agencies with expertise and experience in SHM: Boeing, FAA, Sandia Labs' AANC, Delta, Structural Monitoring Systems Ltd (AEM).
    - Delta will 'live through certification' of SHM application.
    - All vendor items provided separately: Instrument, sensors.
    - Boeing provides program oversight, review.
    - FAA-SACO review.
    - FAA-TAD is the customer. Guidance is the goal.
    - Information provided to SAE G-11 SHM Aerospace Industry Steering Committee on SHM for use in industry standards (guideline documents).

SHM programs must meet a strict cost-benefit analysis, the trickiest parameter of which is payback. Full payback for a typical 'alternate inspection' program would not be until the first repetitive inspection. In general, the cost-benefit to the airlines will be directly proportional to the industry acceptance and comfort level. For example, in order for the FAA and aircraft OEMs to approve SHM, they may initially request a period of time with 'side-by-side' inspections using both the SHM system and the current NDI before granting approval. From the airlines' perspective, this means double the cost for that period of time, which typically means the program will not go forward. It is imperative that the industry comfort level is established quickly and solidly. This will require extensive cooperation between the OEMs, the airlines, the regulators and the sensor vendors.

With respect to Condition-Based Maintenance, a proactive program designed to be an 'early warning system' would have economic value as well, but this is complicated by the pervasive philosophy of 'if you find it, you must fix it before further flight'. This philosophy must be adapted to the increased monitoring that can be accomplished with SHM systems in order for a 'monitoring' program (active or passive) and true condition-based maintenance program to be approved. For the airline, the largest cost-benefit lies in going to true condition-based-maintenance. However, to get to that point, programs such as the one described in this report, must be completed, and subsequent regulatory guidance recommendations endorsed by the FAA. Thus, Delta is making small steps but is taking a longer term view of SHM, hoping to compress the timeline needed for acceptance and widespread adoption.

The basic aspects of this Delta Air Lines SHM certification and integration activity were:

- Certification/usage effort intended to investigate, exercise and evolve the SHM certification path – address all "cradle-to-grave" issues for airlines, MRO vendors, OEMs, and regulators.
- Identify SHM applications with positive cost-benefit analysis.
- Customize SHM system to the selected applications.
- Develop validation/certification plan – utilize precedents from existing sensors.
- Complete SHM indoctrination and training for Delta personnel (engineering, maintenance, NDI), MRO vendors and FAA as needed.
- Address hardware specifications, installation procedures, operation processes, continued airworthiness instructions.
- Complete modifications to Delta maintenance program as a result of SHM use.
- Assess aircraft maintenance MROs to determine their ability to adopt SHM and the FAA support needed to ensure airworthiness.

The listed airline requirements for deployment of SHM were:

- SHM system provides "Equivalent or better level of safety." Performance matches current inspection practices.
- Sensitivity is appropriate for application and defect definition.
- Low false calls.
- System must be 'fail-safe' such that damage detection is not affected.
- Other airworthiness requirements: low-flammability, environmental and vibrational durability, and no loss of data due to electromagnetic interference.
- Flexible financial payback.
- Approval to replace existing inspections in maintenance program
    - No requirements for side-by-side (simultaneous) use of SHM vs current inspection.
    - Ability to move toward long-term goal: Change in philosophy to allow 'monitoring' or Condition-Based Maintenance

Approval and implementation of a new technology, and resulting maintenance approach, included the completion of a series of approvals and technical checks. Figure 7-1 and Figure 7-2 show the rigorous financial, technical and logistical internal signatures necessary to adopt SHM while obtaining guidance, feedback and approval from multiple, pertinent departments within maintenance operations. The Delta Lead Engineer began by generating a project overview and financial form. Coordination meetings lead to signatures and approvals from Engineering Management, Finance, Maintenance, Demand Planning, and Materials Planning. Once these steps were successfully completed, the project was presented to the Operational Reliability Team (ORT) for consideration. Finally, the project was approved by the Project Approval Board (PAB). These approvals are required prior to writing the Engineering Orders and Job Cards used to guide all maintenance activities associated with the use of the SHM system. Upon completion of the paperwork, it is submitted for checking and approval before submitting to "Process Control". This forms an additional check on the previous coordination with maintenance, materials, inspection, etc., prior to the project being allowed to 'go open'. Once the project is open, Scheduling begins assigning the work to specific aircraft, location of the work, and determines the time of accomplishment.

Figure 7-1: Internal Approvals and Signature Flow at Delta Air Lines

7.2 Engineering Order – AMDS Work Cards

After the approval process described above was completed, the Delta Lead Engineer was able to write the Engineering Documentation and associated Job Cards. Internal Engineering Documents and Job Cards were approved as a 'minor alteration' under Delta's authority under the Code of Federal Regulations, Part 14, 121.379(b). The Engineering Authorization, introduced in Figure 7-2, were generated first and provided the formal authorization for CVM use and the production of the Job Cards. After the Job Cards were produced, they were vetted at "Process Control". The Job Card deck (See Figure 7-3) describes all of the tasks associated with CVM deployment (materials, kits, aircraft preparation, sensor installation, steps for continued airworthiness and data acquisition for inspection monitoring).

DELTA

ENGINEERING REQUEST FOR FINANCIAL APPROVAL

ER #: 5711-01044 REV:

AA/EA #: 37-509365-03 REV:

TITLE: Installation - CVM Sensors (Struct. Hlth Monitor)

CLASSIFICATION: OTHER REFERENCE DATA:

If Safety, include Probability vs. Consequences Attachment

1. WHAT IS THE PROBLEM? SUMMARIZE

Structural Health Monitoring (SHM) could be used for significant cost savings/avoidance, enhanced inspection reliability, and early warning of issues. However, no FAA guidance exists on how to handle an SHM application or maintenance program. Enabling Technologies has worked with FAA, Boeing, and a vendor to design an FAA-funded SHM project which will provide this industry guidance, which will then open a path where adoption of SHM will lead to the significant cost savings/avoidance.

Lack of FAA guidance on SHM has prevented adoption of the technology, thereby preventing costs savings. This FAA-funded program will provide the guidance necessary to ensure mainstream acceptance of SHM philosophy, and programs, enabling significant cost benefits to Delta.
Dependence of external Mtc provider (AAR-IND) for implementation.
2013 implementation required to meet FAA contract.

2. WHAT IS THE RECOMMENDED CORRECTIVE ACTION? SUMMARIZE

Propose implementing an FAA-funded SHM program on the B737NG Wing Center Section Shear Fittings via EA 37-509365-03/ AA 5711-01044. Delta will receive $100K from the FAA in exchange for assisting in developing the FAA guidance of SHM programs. Delta will lead the industry innovation and get paid to do it. Installations should occur at PSV-14 (opened access for installations), which is currently performed by AAR in Indianapolis. Schedule shall be starting Q2 2013 and ending in Q1 2014 (FAA contract for funding is Oct 2012-Sept 2014), resulting in 20 aircraft being outfitted with sensors. The remainder of the fleet will not have a convenient MTC visit again until mid-2016 thru 2018, and a future decision will be made about outfitting the remaining 737NG fleet.

3. RECOMMENDED TIME OF ACCOMPLISHMENT (PLANNING/FINANCE)

4. DOES THIS PROPOSAL ADDRESS ONE OF THE FOLLOWING?

○ ODI ○ CPI ○ TOP 5 RELIABILITY INDEX
○ AD/FAR ○ UPPER MANAGEMENT REQUESTED ITEM ○ TOP 5 PASSENGER SATISFACTION INDEX

EXPLAIN

Project supports medium to long-term vision of Structural Health Monitoring (SHM).

5. FOR AA's THAT CHANGE CONFIGURATION, WHAT MEASURES WILL BE TAKEN TO PREVENT DEMODIFICATION?

Implementation of the sensors does not change configuration.

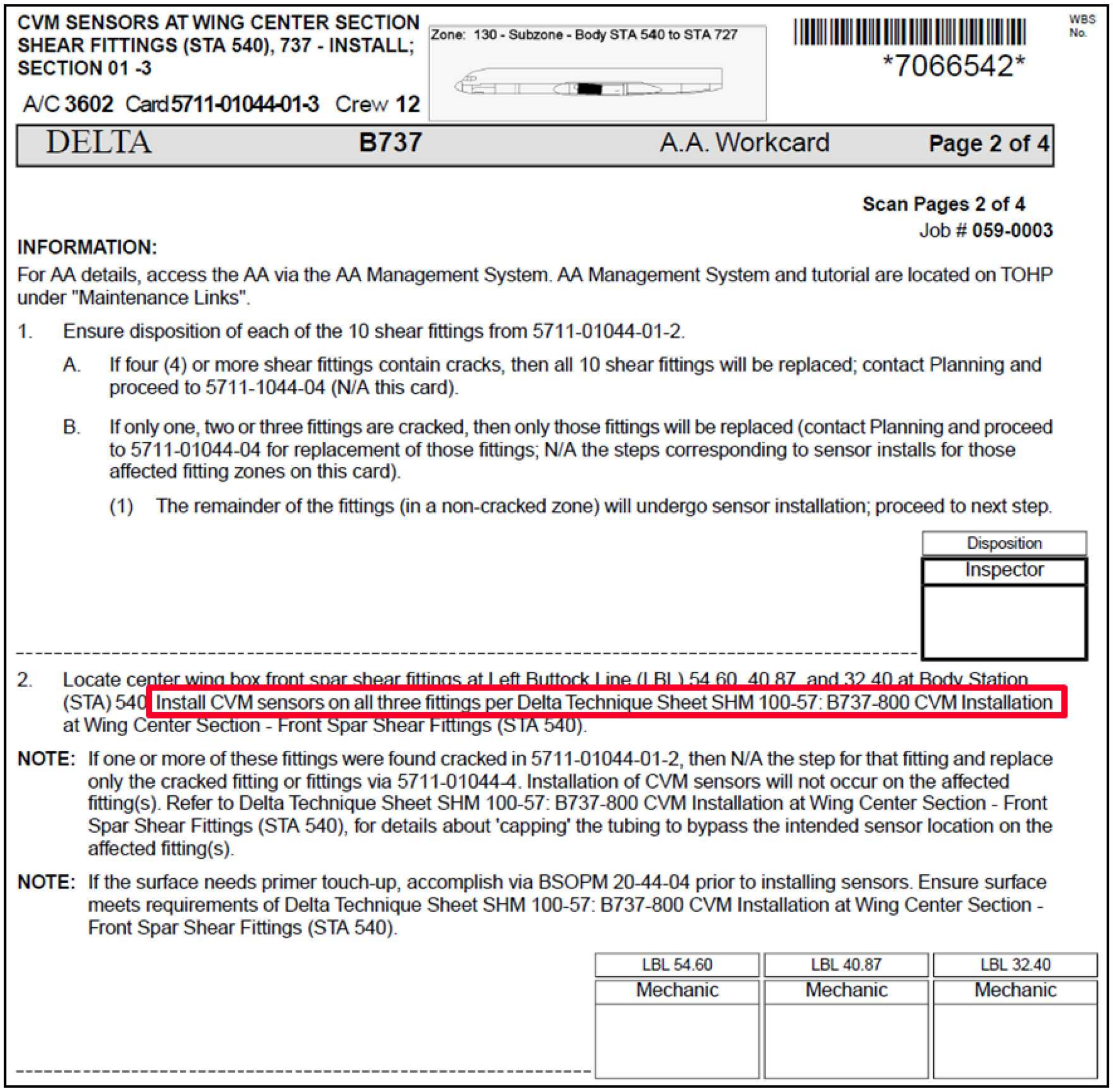
6. MATERIALS INFORMATION EXPLAIN PART/KIT LEAD TIME, AVAILABILITY, & COST

Figure 7-2: Engineering Request for Approvals to Proceed with SHM Deployment

7.3 Delta Reference Documents for Installation, Operation, & Training

Since Delta has an extensive and comprehensive process for authoring or revising Engineering Documents, job aides called "Technique Sheets" were utilized for both the installations and the monitoring. Additional detail to support the Job Card tasks is provided in referenced "Technique Sheets" (See Figure 7-4). In this program two Technique Sheets were adopted into the Delta Air Lines maintenance system: 1) CVM Installation, and 2) CVM Monitoring/Inspection. These

Technique Sheets included six different CVM documents. While these are not the actual 'sign-off' documents, they cover 'what-if' scenarios, include schematics, and other details. These are date and revision controlled, approved by the Level III inspector, and are easier to revise, if needed.



CVM SENSORS AT WING CENTER SECTION SHEAR FITTINGS (STA 540), 737 - INSTALL; SECTION 01 -3

Zone: 130 - Subzone - Body STA 540 to STA 727

7066542

WBS No.

A/C **3602** Card **5711-01044-01-3** Crew **12**

DELTA	**B737**	A.A. Workcard	**Page 2 of 4**

Scan Pages 2 of 4

Job # **059-0003**

INFORMATION:

For AA details, access the AA via the AA Management System. AA Management System and tutorial are located on TOHP under "Maintenance Links".

1. Ensure disposition of each of the 10 shear fittings from 5711-01044-01-2.

   A. If four (4) or more shear fittings contain cracks, then all 10 shear fittings will be replaced; contact Planning and proceed to 5711-1044-04 (N/A this card).

   B. If only one, two or three fittings are cracked, then only those fittings will be replaced (contact Planning and proceed to 5711-01044-04 for replacement of those fittings; N/A the steps corresponding to sensor installs for those affected fitting zones on this card).

      (1) The remainder of the fittings (in a non-cracked zone) will undergo sensor installation; proceed to next step.

Disposition
Inspector

2. Locate center wing box front spar shear fittings at Left Buttock Line (LBL) 54.60, 40.87, and 32.40 at Body Station (STA) 540. Install CVM sensors on all three fittings per Delta Technique Sheet SHM 100-57: B737-800 CVM Installation at Wing Center Section - Front Spar Shear Fittings (STA 540).

NOTE: If one or more of these fittings were found cracked in 5711-01044-01-2, then N/A the step for that fitting and replace only the cracked fitting or fittings via 5711-01044-4. Installation of CVM sensors will not occur on the affected fitting(s). Refer to Delta Technique Sheet SHM 100-57: B737-800 CVM Installation at Wing Center Section - Front Spar Shear Fittings (STA 540), for details about 'capping' the tubing to bypass the intended sensor location on the affected fitting(s).

NOTE: If the surface needs primer touch-up, accomplish via BSOPM 20-44-04 prior to installing sensors. Ensure surface meets requirements of Delta Technique Sheet SHM 100-57: B737-800 CVM Installation at Wing Center Section - Front Spar Shear Fittings (STA 540).

LBL 54.60	LBL 40.87	LBL 32.40
Mechanic	Mechanic	Mechanic

Figure 7-3: Sample Job Card Used to Guide All Aspects of CVM Deployment

Delta Technique Sheets, Job Cards, and Engineering Documents used extensive information provided by the vendor, SMS, and their production house, AEM. Included in this were Introduction to CVM Manual, PM200 Operations Manuals, CVM installation training, and CVM installation procedure. Figure 7-5 through Figure 7-9 provide an overview of the six documents that provide the comprehensive set of information needed to properly and safely utilize CVM technology. Delta and SMS/AEM jointly worked on the schematic and installation drawings specific to the B737NG Center Wing Section Shear Fitting application. When combined, these provided excellent guidance for the CVM installation and monitoring at the Delta facility. The CVM sensor layout, tube routing plan and manifolding of the 10 CVM sensors to the 4 SLS connectors, described in the Section 6.2 text and schematics were also detailed in the documents listed here.

During the flight test activity, experienced personnel from AEM and Sandia Labs participated in the initial installations, along with the Delta Air Lines A&P mechanics. Gradually, the participation from non-Delta personnel was reduced and, ultimately, eliminated such that the Delta A&P mechanics were independently installing the CVM sensors. This process aided the training process at the Delta maintenance facility while also establishing the ability of airline personnel to install CVM sensors. Similarly, Delta inspectors were solely responsible for subsequently monitoring the CVM sensors periodically. Together, the CVM installation and monitoring activities demonstrated the ability of an airline to independently and properly deploy CVM in routine maintenance operation environments.

Delta Air Lines, Inc.	NonDestructive Testing Technique Sheet	SHM 100-57 Date: 03-07-2013 J. Bohler rev. b

B737-800 CVM Installation at Wing Center Section – Front Spar Shear Fittings (STA) 540

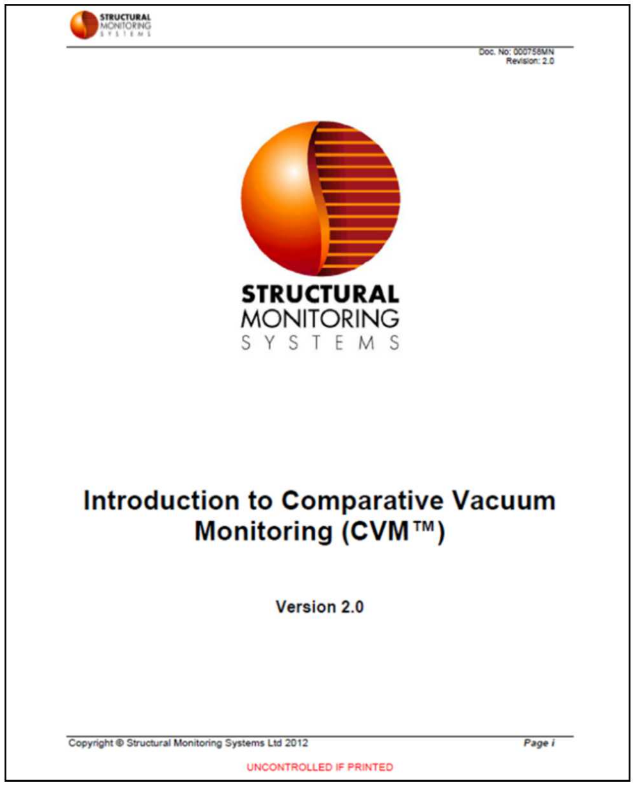
REFERENCE

EA 12-509365-03, AA 5711-01043
SB 737-57-1309
Delta PS 900-1 No. 04
PS 900-7-1-1 No. 03, Fastweld 10 option
B737-678 AMM 28-11-00-300-804
PM200 dCVM Operations Manual, available at http://dpi948/dledmprod:/Main530/equipmanuals/027

PURPOSE

This procedure provides instructions on how to prepare surfaces, install and overcoat CVM™ FEP Sensors at (LBL) 54.60, 40.87, 32.40, 15.04, 6.14 and (RBL) 54.60, 40.87, 32.40, 15.04, 6.14, at Body Station (STA) 540. The installation consists of 10 sensors. The 3 outboard sensors will be daisy chained together and the 2 inboard sensors will be daisy chained together on each side. See fig. 3. The final installation will have 4 Sensor Lead Sockets (SLS) to connect the PM200 to.

Figure 7-4: Use of Technique Sheets to Provide Additional Details and Aid the Job Card Task



STRUCTURAL MONITORING SYSTEMS

Doc. No: 000758MN
Revision: 2.0

STRUCTURAL
MONITORING
SYSTEMS

Introduction to Comparative Vacuum Monitoring (CVM™)

Version 2.0

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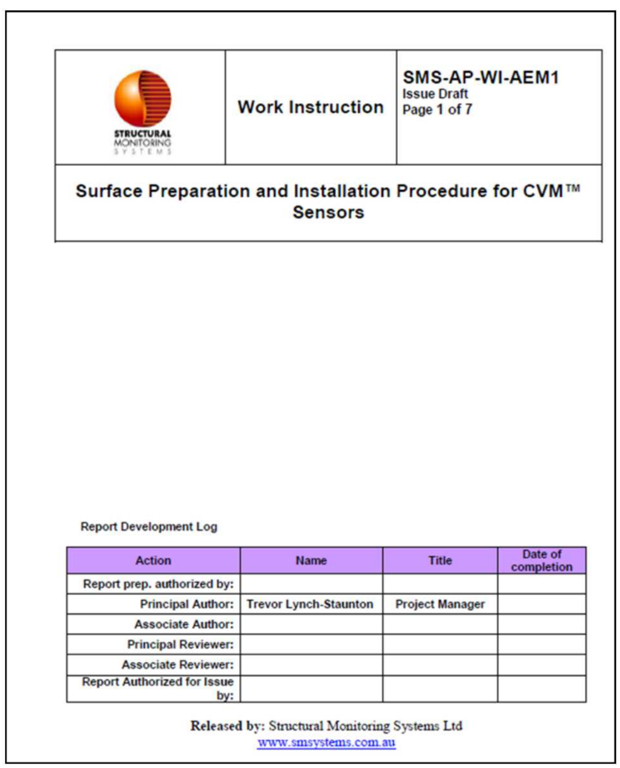
UNCONTROLLED IF PRINTED

Figure 7-5: Technique Sheets - Description of CVM Technology and Usage



STRUCTURAL MONITORING SYSTEMS	Work Instruction	SMS-AP-WI-AEM1 Issue Draft Page 1 of 7

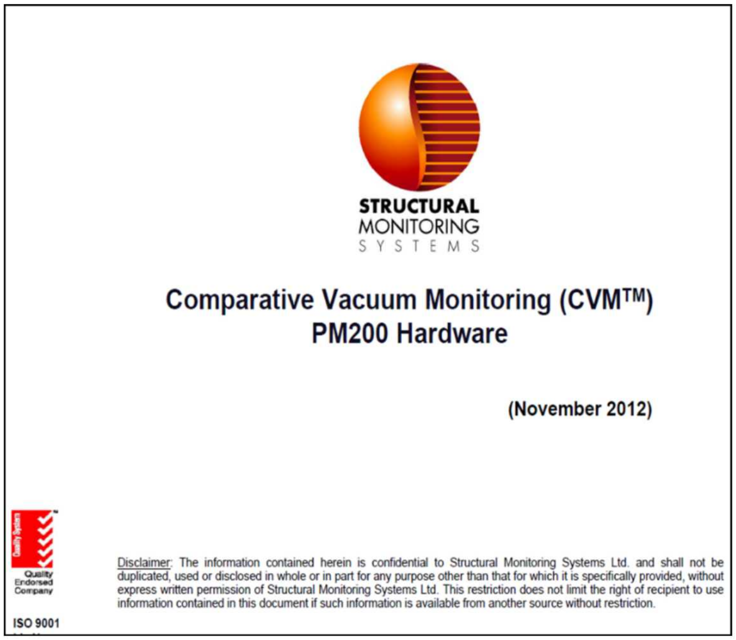
Surface Preparation and Installation Procedure for CVM™ Sensors

Report Development Log

Action	Name	Title	Date of completion
Report prep. authorized by:			
Principal Author:	Trevor Lynch-Staunton	Project Manager	
Associate Author:			
Principal Reviewer:			
Associate Reviewer:			
Report Authorized for Issue by:			

Released by: Structural Monitoring Systems Ltd
www.smsystems.com.au

Figure 7-6: Technique Sheets - CVM Installation Procedures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Comparative Vacuum Monitoring (CVM™)
PM200 Hardware

(November 2012)

Quality Endorsed Company
ISO 9001

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Figure 7-7: Description of PM200 Device


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dCVM Operations Manual

Part A – PM200 Hardware

Figure 7-8: Technique Sheets – CVM Operations Hardware

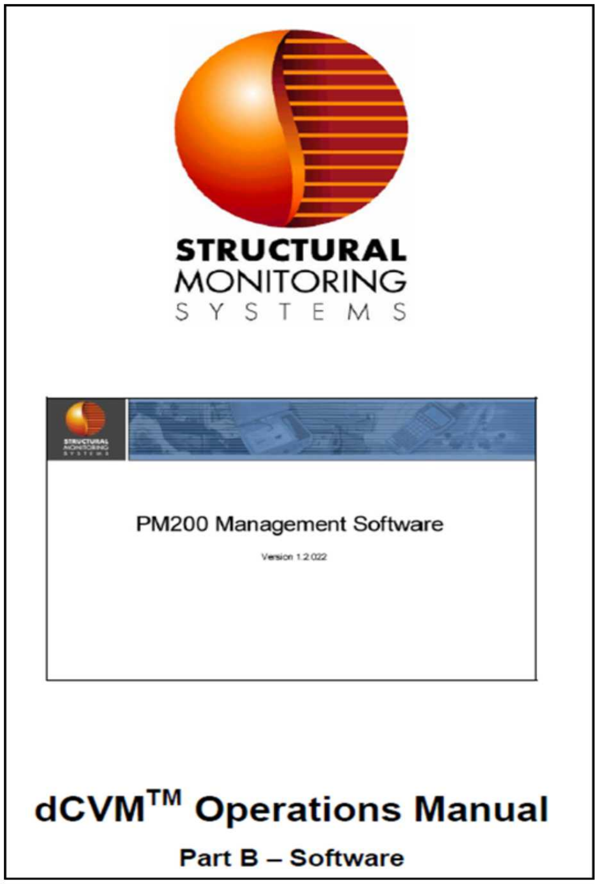


Figure 7-9: Technique Sheets - CVM Operations Software

7.4 Formal Adoption of SHM into Routine Maintenance Programs & View on Future Use

Challenges for Maintenance Program Revisions - As expected for any program that involves the introduction of new technology to maintain aircraft, there were a number of challenges experienced during this project. Internal and external challenges included:

- A cost-benefit analysis had to be tabulated and vetted by the Finance Department. A program with a payback longer than 6 to 12 months can be deemed problematic. This obstacle was overcome by obtaining Senior Management buy-in to the long-term SHM utilization vision.
- Due to the Delta merger with Northwest and resulting integration, all Job Cards and engineering documents required a 'dual process'. Beginning the project in the throes of integration was unfortunate timing that created difficulties with project flow.
- The installation approval process had to be repeated four times due to changing organizational structure and associated changing approval requirements. This involved the generation of EP-12 financial forms and dual authorship on the Job Cards. Ultimately, the

approval process was successful but required seven months; longer than the normal time frame expected even for modifications in aircraft maintenance methods.

- Once approved, the coordination of logistics continued to be a challenge. Twice, the project was 'frozen indefinitely' due to mandatory milestones associated with integration of computer systems due to the merger of Delta and Northwest.
- After final review by the Operational Reliability Team and Engineering Project Approval Board, the project was ready and CVM installations could begin.
- 737 CVM Wing Box Fitting kits were prepared in accordance, with normal Quality Assurance procedures, and assigned a formal number for ordering from the AEM supplier. The kits included the sensors along with all other on-board hardware and installation aids. Normal surface preparation tools, such as Acetone, primer and fine grit sandpaper, were checked for routine availability within Delta stores and were not included in the custom CVM kits.
- Time pressure and task descriptions provided new challenges. The aircraft installations were conducted during a 5-day visit instead of a traditional heavy maintenance visit. This only provided 2 days or less of access to the aircraft to complete all 10 sensor installations, as well as, all the routing and connection to the SLS connectors.
- The CVM sensors were installed by Maintenance personnel, but were inspected/monitored by Inspection personnel. The different job categories had to be accounted for in the Job Cards, sign-off requirements, and Engineering documentation.
- Training associated with all installation, monitoring and maintenance decisions based on CVM data was conducted with all personnel involved. This training required coordination of the entire industry team to properly hand off all duties to Delta personnel as described above.
- Finally, the target aircraft for CVM installation came into a different Delta maintenance facility than originally planned. This required some additional planning to move all activities to Atlanta.
- All aspects of the project included input from the Delta-Boeing-Sandia-SMS/AEM-FAA team in order to address the full range of engineering issues including: procedures, Job card definition, guidance documents, sensor design and layout, application-specific installation drawings, inspection, Damage Tolerance Analysis, approval via Aviation Representatives at Boeing.

Of course, it is recognized that this was the first exercise of SHM for routine maintenance and it was expected that additional attention would be required as Delta – or any airline - worked toward adoption of CVM technology. With the above process completed once, the SHM approach well-defined within Delta, Technique Sheets installed, Job Cards generated and Delta personnel trained/experienced in CVM deployment, it is expected that future CVM use will be simpler and quicker.

Lessons Learned - There were several lessons learned during the SHM adoption program. This was one of the primary goals of this activity: to identify and eliminate obstacles associated with the deployment of new aircraft monitoring methods. The specific areas ranged from optimization of the sensor installation process to streamlining the administrative, approval and oversight process.

- Sensor Installation - Surface preparation is key to a successful CVM sensor installation (obtaining a proper seal between the sensor and the surface it is monitoring). Thus, the surface preparation cannot be rushed. The time required for proper installation depends greatly on the specifics of the application. The Wing Box Fitting application was a challenging installation that involved a complex sensor design (non-symmetric and size of sensor) and a difficult-to-access area with tight geometry. In addition, the tight schedule associated with a short maintenance visit, pushed the sensors installations to less than two days. Re-installation of several sensors was needed, based on initial baseline data integrity checks, but not all of them could be accommodated due to the aircraft's return-to-service schedule. Hindsight revealed that three days would have been sufficient and probably would have eliminated the few sensor installation issues experienced (i.e. improved the 97% data success and 3% sensor failure rate).
- Primer Application - The use of a spray-on primer provided a better surface than brushed-on primer. It was noted that the brush application of the primer left an uneven surface with very small "grooves" in the coating. This made it more challenging to produce an optimal seal with the CVM sensor. A spray-on primer was used on the latter aircraft. This produce a smoother surface and eliminated any future sensor failures on these aircraft.
- Tooling Aids - A custom tool was designed as an aid to produce repeatable sensor placement with an optimal seal. However, due to the geometry, it was found that a free-hand application worked best. In future CVM applications, the use of templates and sensor installation tools may be wise.
- Proper Time Allotment - Related to the aircraft short-visit, it was noted that the 2-day time window for installation required the completion of adjacent work near the sensor installation. This created the possibility of contamination of the prepared surface. Minimizing adjacent movement and air flow in the cabin helped reduce sensor failures. Cabin interior crew was in and out of the area, requiring some coordination, but SHM installation did not impede progress on adjacent tasks.
- Post Install Treatments – After each sensor installation, the fasteners were sealed and fuel vapor barrier was reapplied to the entire area. A diluted application of the fuel vapor barrier (up to 50% dilution with acetone is allowable per Boeing specifications) and use of smaller, more delicate brushes to reapply the rivet sealant (i.e. smaller, artist paint brush) produced proper coverage without coating the entire structure. This allowed for better visual inspections immediately adjacent to the sensors in the event that visual inspections were useful or otherwise required to obtain the necessary 'comfort level' during this SHM Pilot Program.
- Post Install Connections - Once the sensors were mounted in place, tubing and SLS connectors were routed down the front spar to a bracket for convenient access through the cargo bin during future, repetitive inspections. It was determined that more detail was needed to describe the exact tubing tie-down points. This was accomplished and a sample schematic and photo description of the CVM sensor connection plan shown in Figure 7-10.
- Monitoring CVM Sensors Through SLS Connections - Each SLS instrumentation line is programmed through the PM200 device to assign proper monitoring features (e.g. crack detection threshold) and to associate each sensor to its SLS connector. This allows for automated data collection and retrieval of all data points acquired over time. It was determined that such programming is best completed prior to aircraft arrival.

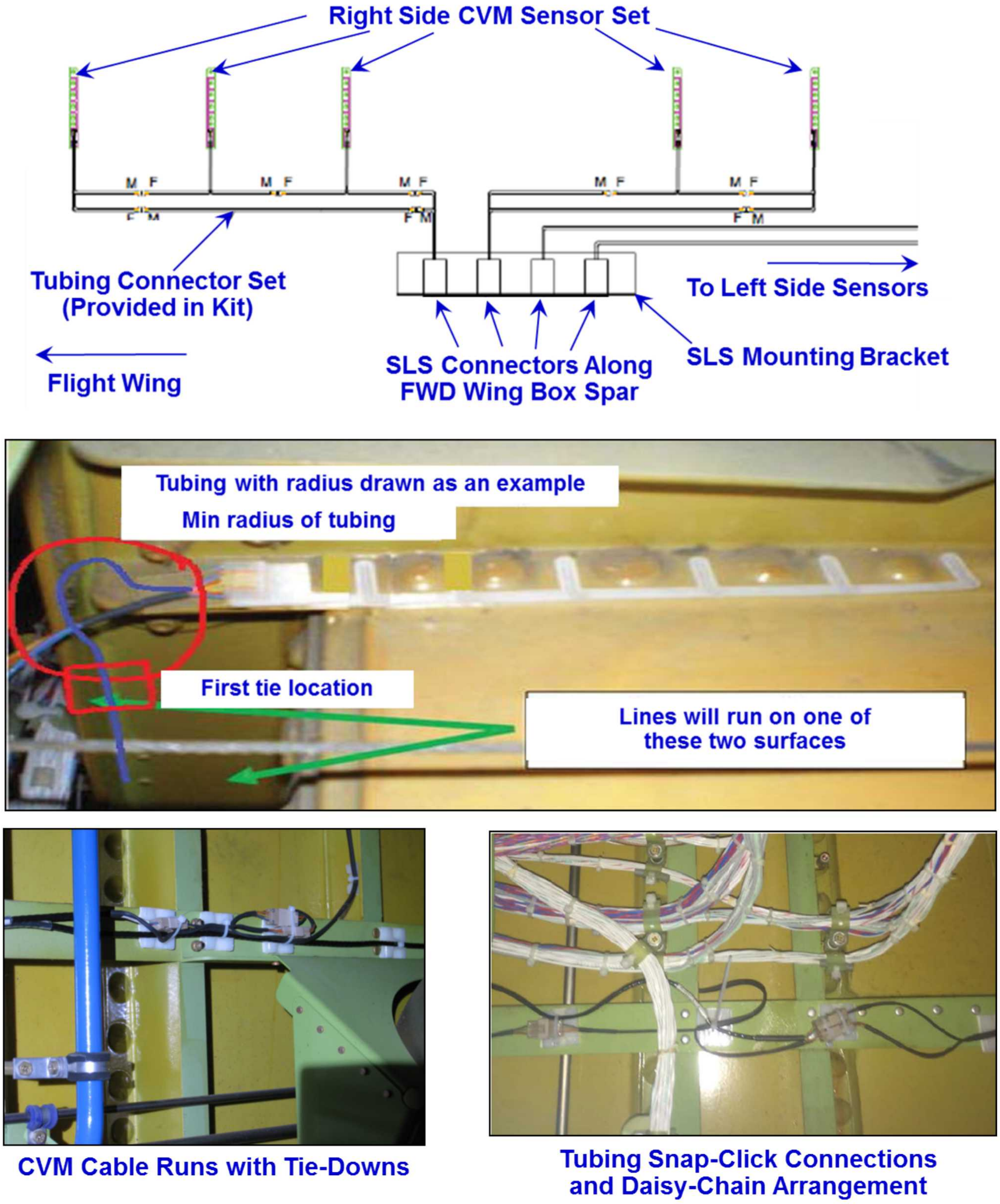


Figure 7-10: Sample of Tube Routing Diagrams and Tie-Down Points

(10 CVM sensors, 4 SLS connectors)

- Training - The technical training for the Delta personnel conducting the CVM system installation and the subsequent monitoring was critical. This is true for all new processes but especially true for the instruction of new technology that effects the core of aircraft maintenance approaches. Initially, a CVM workshop was conducted for all engineering people involved. This should have been expanded to include all A&P and inspection personnel who also had hands-on roles with the CVM deployment. Training for the latter group was conducted separately and later on when the flight test aircraft arrived at the maintenance depot. The A&P and inspection personnel would have benefitted from training during both time periods. Their questions and input early-on may have provided good guidance to streamline the program overall. In addition, it was determined that training should be implemented in both group and one-on-one sessions. Human factors concerns are always present and both classroom and field (hand-on) training were used to minimize and human factors issues.
- General SHM Education Process with Delta and FAA – Several different SHM introduction and training efforts were conducted at Delta's facilities to indoctrinate key personnel from the pertinent departments listed above. However, this process was quite ad-hoc and the timing was reactive instead of proactive. In general, the education process within Delta needed to be broader and more efficient. Many different departments were involved, creating complexities for coordination. Changing approval requirements and rotation of personnel due to merger integration made this even more important. The information and experience acquired during this program will allow future SHM integration programs to conduct more comprehensive and efficient training/education initiatives for the important airline personnel.
- CVM Wing Box Kit – As mentioned above, custom kits were generated and provided by AEM supplier using normal airline supplier QA and kitting procedures. It was noted that ten sensor kits are necessary for outfitting each aircraft but at least one additional kit should be available to accommodate any installation errors or other problems.
- Administrative Paperwork - The paperwork associated with the CVM sensor installation was well thought out. The use of a "Technique Sheet" made things more flexible. Instead of changing the Job Cards and Engineering documentation and going back through the full approval process, it was possible to quickly adjust detailed procedures by modifying the Technique Sheets described above. Some delays in the approval process were inevitable due to unique circumstances such as the Delta-Northwest merger. However, some streamlining could be done, via the comprehensive education process listed above, as well as other measures to inform management about the pending SHM deployment effort. Once again, the information and experience acquired during this program will allow future SHM approval efforts to be streamlined.

Formal OEM Paperwork for Adoption of CVM Technology - As discussed in Section 4.3 and Section 6.1, previous laboratory and flight testing on fuselage skin configurations established the overall capability of CVM sensors allowed CVM technology to be included in Boeing's NDT "tool box" as a viable crack detection methodology. As a result, Boeing's NDT Standard Practices Manual was revised to include CVM sensors as a possible structural monitoring option

[7.5]. This was the first of several formal steps that have taken place to recognize the capability of the CVM technology and allow for its use on commercial aircraft.

While the inclusion of CVM in the Boeing NDT Standard Practices Manual was a positive step, this only placed CVM sensors into the overall structural monitoring tool box for potential use. Subsequent deployment on specific applications could require some additional "gap testing" to address any unique, and currently untested, aspects of CVM performance for the materials and structural geometry of interest. Thus, the second building block to support use of CVM sensors involved inclusion of CVM technology in the Boeing Nondestructive Inspection Manual [7.6].

As a result of the CVM Wing Box Fitting program and the compiled results from completed of lab/flight testing, CVM was added to the Boeing Nondestructive Testing (NDT) Manual for the 737 aircraft platform. First, Part 5, a previously unassigned section, was added to the NDT Manual to cover "Structural Health Monitoring." Then, the first chapter within Part 5 was added to address the CVM method. This is shown in Figure 7-11 and Figure 7-12. Additional information was placed within referenceable sections inside the NDT Manual so that a comprehensive description and user-based information could be included on the CVM technology. This is shown in Figure 7-13 where the "Comparative Vacuum Monitoring" Chapter 1 within Part 5 (SHM) contains links to CVM installation and operation documents which are also contained within the NDT Manual.

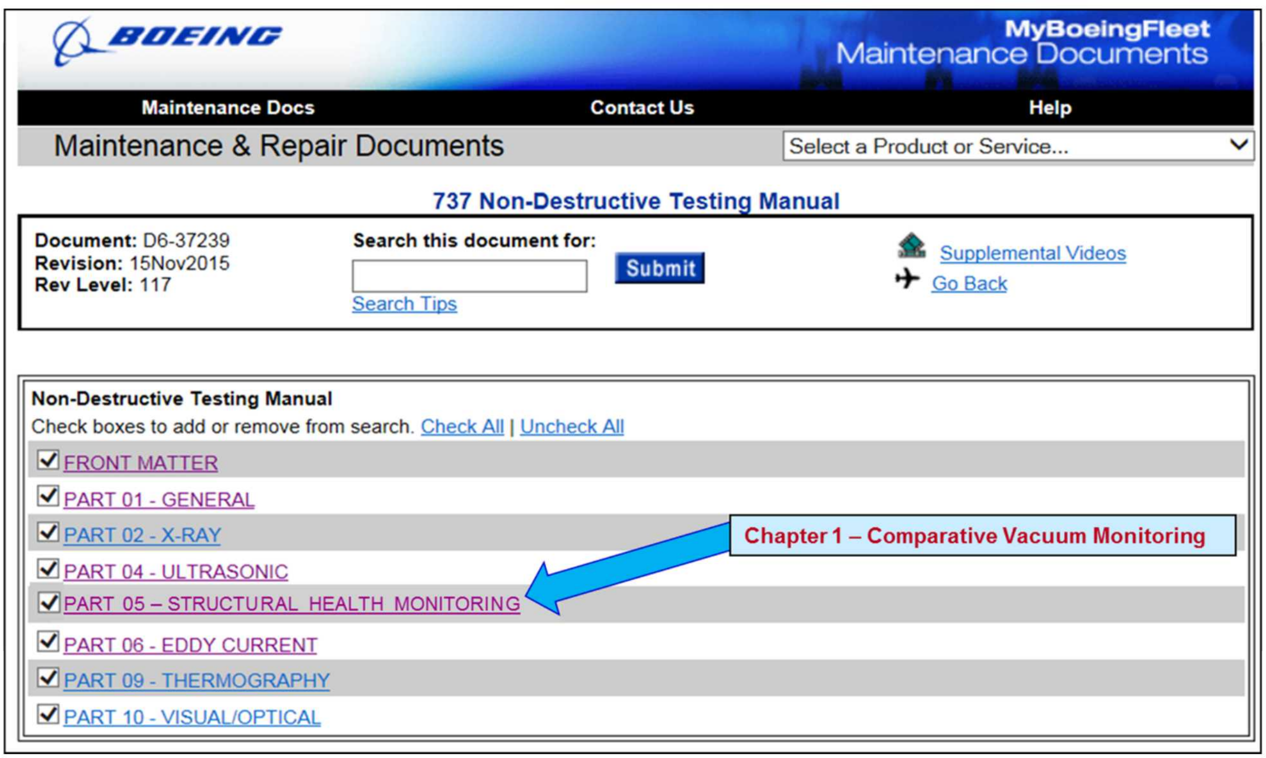


Figure 7-11: Building Block for Approval of Routine Use of SHM - New SHM Part 5 Section and CVM Chapter Published in 737 NDT Manual

BOEING

737
NON-DESTRUCTIVE TEST MANUAL

PART 5 - COMPARATIVE VACUUM MONITORING

WING CENTER SECTION - SHEAR FITTINGS AT THE FRONT SPAR

1. Purpose
   A. Use this comparative vacuum monitoring (CVM) procedure to help find cracks in the 111A2401-1 and -2 shear fittings at the front spar of the wing center section. See Figure 1 for the inspection areas.
   B. This procedure can find cracks that are 0.75 inch (19.1 mm) long or longer.
   C. The shear fittings are 7050-T7451 aluminum alloy.
   D. Service Bulletin Reference:
      (1) 737-57-1309
2. Equipment
   A. General
      (1) Comparative vacuum monitoring (CVM) is a structural health monitoring (SHM) system. The CVM system measures the different pressures between sensor galleries that have a vacuum or are at atmospheric pressure to find cracks in parts. See Figure 2 for some examples of CVM equipment.
      (2) Use the equipment specified in this inspection procedure to do this procedure.
   B. Instrument
      (1) PM200; Structural Monitoring Systems (SMS)
   C. Functional Test Socket
      (1) PM200-9 or SP1131; Structural Monitoring Systems (SMS)
   D. Comparative Vacuum Monitoring kit
      (1) 737NG-FSSF-1KCVM CVM Installation Kit; Structural Monitoring Systems (SMS)
   E. Software
      (1) PM200 Management Software version 0.0.3276 or newer
   F. Special Tools
      (1) Consumables kit. See set up file: Part 5, 57-10-01 List of Necessary Materials

Figure 7-12: Chapter 1 (CVM) of Part 5 (SHM) of 737 NDT Manual

The third building block installed to support the use of CVM technology for aircraft structural monitoring involved a revision to the Boeing Service Bulletin (SB) referenced above that addressed the wing box fitting inspection requirements. This 737 SB, number 737-57-1309, describes the inspection process for the Wing Box Fitting inspection, as well as any subsequent repair or replacement modifications [7.7]. In 2016, this SB was changed to include CVM technology as an alternate inspection method to the previously-specified visual and eddy current inspections. This is highlighted in Figure 7-14. Similarly, other SBs have been produced to acknowledge CVM usage [7.8 – 7.10]

Embedded within this SB revision, was a series of approvals from designated authorities within Boeing. The formal approval came from the appropriate set of Designated Engineering Representatives (DER) or, as they are also known, Airplane Representatives (AR). An FAA DER/AR is a person who has been given authorizations to perform certain certification functions on behalf of the FAA. They are responsible to find that a proposed aircraft design, (i.e. the engineering data, complies with the published airworthiness requirements for the aircraft type. This approved engineering data is part of the aircraft's type design. DER/AR are very specialized and are given authorizations to perform approvals of the data (instructions) used to make certain modifications or repairs to aircraft.

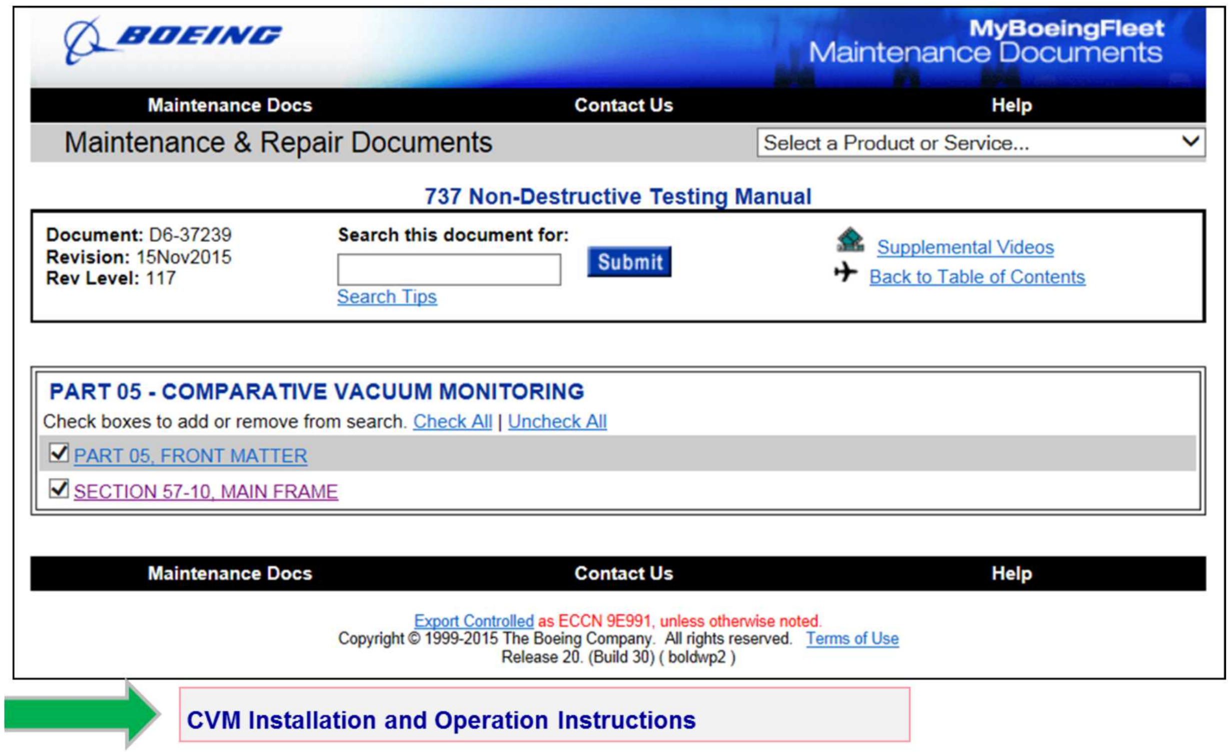


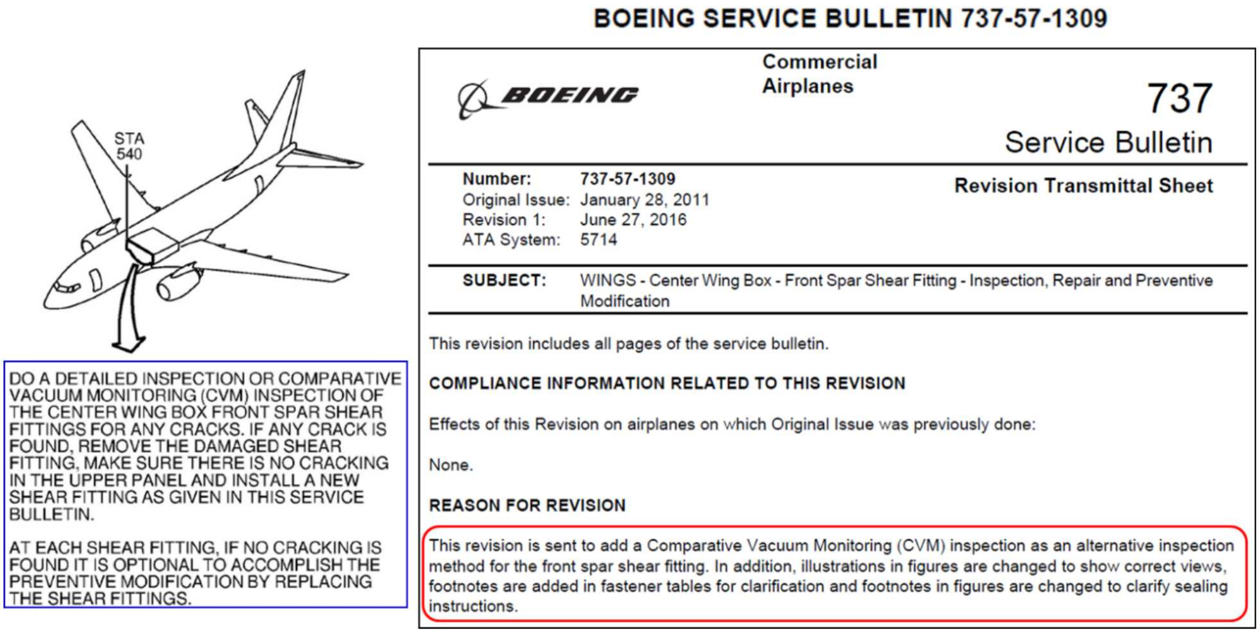
Figure 7-13: CVM Procedures Included in 737 NDT Manual

A DER may be appointed to act as a Company DER and/or Consultant DER.

- Company DERs can act as DER for their employer and may only approve, or recommend approval, of technical data to the FAA for the company.
- Consultant DERs are individuals appointed to act as an independent (self-employed) DER to approve or recommend approval of technical data to the FAA.

Some examples of the DER/AR Technical Disciplines include: Acoustical Engineering, Engine Engineering, Electrical Systems and Equipment, Flight Test Pilot, Powerplant Engineering, Propeller Engineering, Structural Engineering, System and Equipment Engineering. Depending on the equipment and application involved, approval will be required from one or more ARs. Typically the minimum is an Electrical Systems AER and Structures AR. In the case of the CVM technology, approval was obtained from a Structures AR, Interiors AR and Systems AR. Finally, the appropriate set of information along with the AR approval set was delivered to the FAA for its concurrence. No objections were received from the FAA regarding the SB revision.

The approval for CVM use could have also come via an Alternate Means of Compliance (AMOC). The AMOC would be processed and approved via the submission of an 8110-3 form [7.11]. Form 8110-3 is a "Statement of Compliance with the Federal Aviation Regulations." It is used to issue formal approval for a design of a certain piece of equipment, material etc. and indicates that the approving authority finds the subject in question to be in compliance with the FAA regulations. Similar to the SB revision process above, the AMOC would involve formal approval from appropriate ARs.



BOEING SERVICE BULLETIN 737-57-1309

BOEING Commercial Airplanes

737
Service Bulletin

Number: 737-57-1309
Original Issue: January 28, 2011
Revision 1: June 27, 2016
ATA System: 5714

Revision Transmittal Sheet

SUBJECT: WINGS - Center Wing Box - Front Spar Shear Fitting - Inspection, Repair and Preventive Modification

This revision includes all pages of the service bulletin.

COMPLIANCE INFORMATION RELATED TO THIS REVISION

Effects of this Revision on airplanes on which Original Issue was previously done:

None.

REASON FOR REVISION

This revision is sent to add a Comparative Vacuum Monitoring (CVM) inspection as an alternative inspection method for the front spar shear fitting. In addition, illustrations in figures are changed to show correct views, footnotes are added in fastener tables for clarification and footnotes in figures are changed to clarify sealing instructions.

Figure 7-14: Revision to Boeing Service Bulletin 737-57-1309 to Allow for Routine Use of CVM Solution

Formal Airline Paperwork for Adoption of CVM Technology - Within the airlines, and specifically the Delta Air Lines aircraft maintenance program, the documents described are used to institute the formal adoption of CVM technology. These documents include Engineering Authorization, Engineering Orders, Job Card deck for CVM Use on Wing Box Fitting, NDT Technique Sheets: CVM Installation, NDT Technique Sheets: CVM Operation and Monitoring, and Instructions for Continued Airworthiness. Internal Engineering Documents and Job Cards were approved as a 'minor alteration' under Delta's 14 121.379(b) authority. In order to

The ability for Delta to take credit (substitute the alternate inspection for the existing inspection) for the sensor inspections within the Maintenance Program came from the Boeing AR approval and associated SB modification described above. This allowed Delta to avoid opening up the wing box area for a HFEC or visual inspection and instead use the network of CVM sensors to appropriately monitor the region.

Formal Aviation Industry Paperwork for Adoption of CVM Technology - It is also important to note the aviation industry's response to SHM deployment. A number of different efforts are underway and agencies have put significant effort into producing guidance to safely integrate SHM methodology into aircraft maintenance activities.

In 2006, the Aerospace Industry Steering Committee on Structural Health Monitoring (AISC-SHM) was formed. The SHM-AISC is a team comprised of industry industrial, government, and academic participants with a collective vision to efficiently and effectively implement structural health monitoring for a wide variety of commercial and military applications through the development of standards, procedures, processes and guidelines for implementation and certification of SHM technologies. It currently operates under the auspices of SAE, includes over 150 members and has key representation from both government and private industry players. The Executive Management Board contains SHM experts from the FAA, EASA, airlines, aircraft manufacturers worldwide, US Army, US Navy, US Air Force, foreign military agencies, NASA, universities, research agencies, SHM developers and SHM integrators.

The mission of the SHM-AISC is to provide an approach for standardizing integration and certification requirements for SHM of aerospace structures, which will include system maturation, maintenance, supportability, upgrades and expansion. The final goals are to develop various guidebooks specifying approaches for SHM usage on Air and Space vehicles and to identify technology gaps leading to SHM utilization. Towards that end, several SHM guidebook documents are in process or have already been published. These documents are published in the form of Aerospace Recommended Practices (ARP) or Aerospace Information Reports (AIR), both of which can be referenced within OEM and regulatory documents/manuals. Two of the important documents supporting the commercial SHM deployment discussed in this report are: 1) ARP 6461 "Guidelines for Implementation of Structural Health Monitoring on Fixed Wing Aircraft" [7.12], and 2) ARP 6821"Guidance for Assessing the Damage Detection Capability of Structural Health Monitoring Systems" [7.13]. An introduction to the content of these documents is provided in Figure 7-15 through Figure 7-20. ARP 6461 has been published while ARP 6821 is currently in draft form with a goal of publication in 2021.

SAE Aerospace An SAE International Group	AEROSPACE RECOMMENDED PRACTICE	SAE ARP6461 Issued Proposed Draft v.2 2012-01-07
Guidelines for Implementation of Structural Health Monitoring on Fixed Wing Aircraft		

RATIONALE

The development of Structural Health Monitoring (SHM) technologies to achieve Vehicle Health Management objectives in aerospace applications is an activity that spans multiple engineering disciplines. It is also recognized that many stakeholders: Regulatory Agencies, Airlines, Original Equipment Manufacturers (OEM), Academia and Equipment Suppliers are crucial to the process of certifying viable SHM solutions. Thus a common language (definitions), framework of solution types, and recommended practices for reaching those solutions, are needed to promote fruitful and efficient technology development.

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Figure 7-17: Overview of ARP Guideline for SHM on Fixed Wing Aircraft – Part 3

SAE Aerospace An SAE International Group	AEROSPACE RECOMMENDED PRACTICE	SAE ARP6821 Issued Draft 4 29-February-2020

Guidance for Assessing the Damage Detection Capability of Structural Health Monitoring Systems

RATIONALE

The goal of this document is to provide guidance for establishing the damage detection capability of Structural Health Monitoring (SHM) systems to assist in the safe adoption of SHM solutions for assessing aircraft integrity. The maturity of SHM systems has evolved to the point where some SHM systems have demonstrated sensitivities that meet or exceed current damage detection requirements and are being considered for on-aircraft use. As a result, there is a growing need for well-defined methods to statistically quantify the capabilities of SHM systems. However, while there are many agreed-upon procedures for quantifying the performance of NDI techniques, there are limited guidelines for assessing SHM systems. While the intended function of the SHM and NDI systems may be very similar, there are distinct differences in the parameters that affect their capability and differences in their implementation that require special consideration. For example, some factors that affect SHM sensitivity include the flaw size, shape, orientation and location relative to the sensors, equipment hardware and software, the structural configuration, variability in the damage, residual strains, stress fields, operational and environmental variables, and issues related to the presence of multiple flaws within a sensor network. The objective is to apply statistical methods to laboratory and flight test data to derive Probability of Detection (POD) values for SHM sensors in a fashion that agrees with current nondestructive inspection (NDI) validation requirements. It is anticipated that all people in the aviation industry that are working in the SHM arena will benefit from standardized methods to establish the performance and reliability of SHM systems. They should be able to use the SHM Reliability document to clearly define the different types of SHM damage detection approaches and to assess the damage detection capability of each SHM application.

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Figure 7-19: Overview of ARP Guideline for Assessing SHM Performance – Part 2

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Figure 7-20: Overview of ARP Guideline for Assessing SHM Performance – Part 3

FAA Issue Paper Addressing SHM - Recently, the FAA has produced an Issue Paper (IP) in response to a formal request for use of CVM technology to address structural inspections associated the WiFi installations [7.14]. The IP represents the first formal set of guidelines from the FAA to produce the data necessary for certification of SHM systems in routine maintenance activities. The IP contains the general guidelines for producing SHM performance data to ensure that the proposed SHM system can adequately and reliably detect damage for compliance with §§ 25.571 and 25.1529. Specifically, the IP addresses the use of "Comparative Vacuum Monitoring (CVM) for Damage Detection in Structure of Antenna Installations."

The background on the development of the IP is as follows. Title 14, Code of Federal Regulations (14 CFR) 25.1529 requires applicants to prepare Instructions for Continued

Airworthiness (ICA) per Appendix H of part 25 that are acceptable to the Administrator. The Federal Aviation Administration (FAA) approves certain portions of the ICA, such as the Airworthiness Limitations Section (ALS). These ICA may include damage tolerance based inspections developed in accordance with the requirements of § 25.571(b). SHM are relatively new technologies for conducting inspections. Therefore, applicants need to demonstrate that SHM systems effectively and reliably detect damage. Applicants need to show the proposed SHM system to be as good as the current inspection program that the SHM system is replacing. This IP specifies key elements and criteria the applicant must address to demonstrate that their proposed SHM system adequately replaces existing ICA that are necessary for compliance with §§ 25.1529 and 25.571. The primary intent of §§ 25.1529 and 25.571 is to ensure an airplane's structural maintenance program will prevent catastrophic failure due to fatigue damage over the operational life of the airplane. The elements and criteria identified in this IP (FAA Position) will guide the applicant's comprehensive assessment of the functionality, reliability, durability, and maintainability of the proposed SHM system.

The use of SHM has also been recognized within the aviation industry's MSG-3 document [7.15]. MSG-3 (Maintenance Steering Group) 'Operator/Manufacturer Scheduled Maintenance Development' is a document developed by the Airlines for America (A4A; formerly Air Transport Association). It aims to present a methodology to be used for developing scheduled maintenance tasks and intervals, which will be acceptable to the regulatory authorities, the operators and the manufacturers. The main idea behind this concept is to recognize the inherent reliability of aircraft systems and components, avoid unnecessary maintenance tasks and achieve increased efficiency. The underlying principles are that:

- Maintenance is only effective if task applicable
- No improvement in reliability by use of excessive maintenance
- Needless tasks can also introduce human error
- Few complex items exhibit wear out
- Monitoring generally more effective than hard-time overhaul - Condition-Based Maintenance (sometimes known as CBM)
- Reliability only improved by modification
- Maintenance may not be needed if failure is acceptable and cheaper

MSG-3 is widely used to develop initial maintenance requirements for modern commercial aircraft which are published as a Maintenance Review Board Report (MRBR). It has two Volumes (1 for Fixed Wing Aircraft and 2 for Rotorcraft), and its application will proceed alongside the Type Certification process. In 2015, SHM was added as a recognized and potential alternative to conventional NDT procedures.

Finally, it is important to note that SHM systems are being adopted to conduct structural monitoring in other industries ranging from wind energy to oil and gas to civil structures such as bridges and buildings. Such work is complimentary to the aviation SHM efforts and helps lay the foundation for universal SHM usage by accumulating additional, successful field history on SHM systems.

One sample document that supports and guides the use of SHM systems on bridges is included in [7.16]. This document is the first technical SHM code by a national government that enforces sensor installation on highway bridges. Because budgets for replacing railroads, bridges, and other structures are limited, advances in sensors and monitoring technologies make a strong argument for the fact that they should be included as part of future civil engineering structures, rather than just being added to existing infrastructure. If governments could regulate and standardize SHM systems in the context of new design standards, their application: (1) could help to identify existing decaying infrastructure, (2) become an integral component of design requirements, and (3) regulate the methodology needed to quantify structural damage in future possible scenarios. The rapidly-increasing code work can serve as a model for transportation agencies in other countries in writing, approving, and implementing their own SHM regulations and guidelines to optimize SHM deployment.

Future SHM Use: Sample Applications – As described in Section 1.0, numerous airlines around the world are now considering the use of SHM solutions in their maintenance programs. Many of them have conducted an analysis of their maintenance programs and aircraft fleets to identify potential applications for SHM that provide both technical and economic value. Figure 7-21 through Figure 7-28 list several of the applications identified by airlines for possible SHM use.

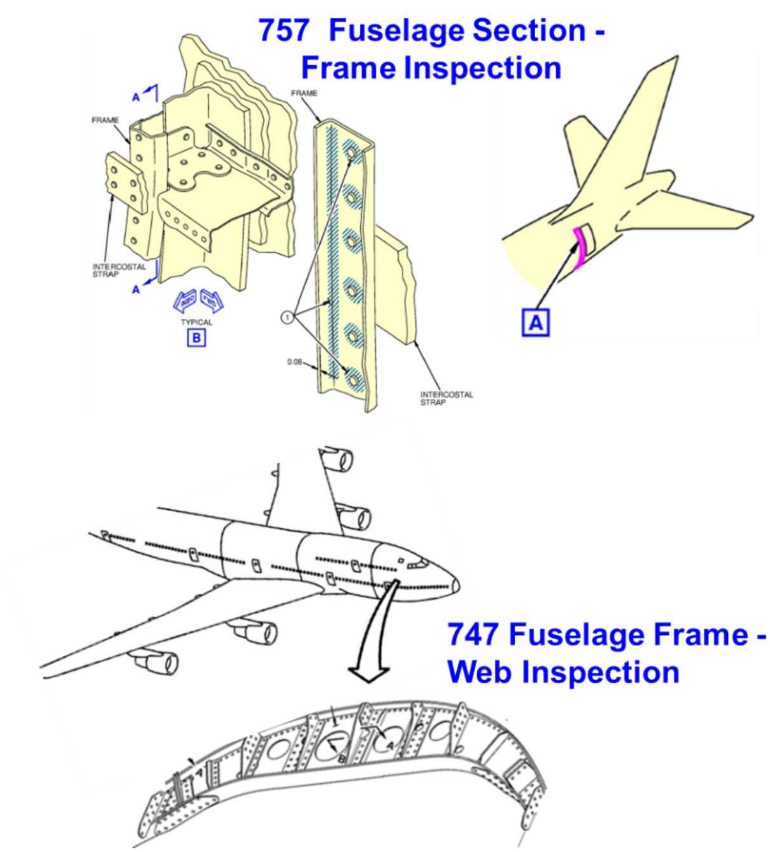


Figure 7-21: Possible SHM Applications – Frame and Substructure Inspections

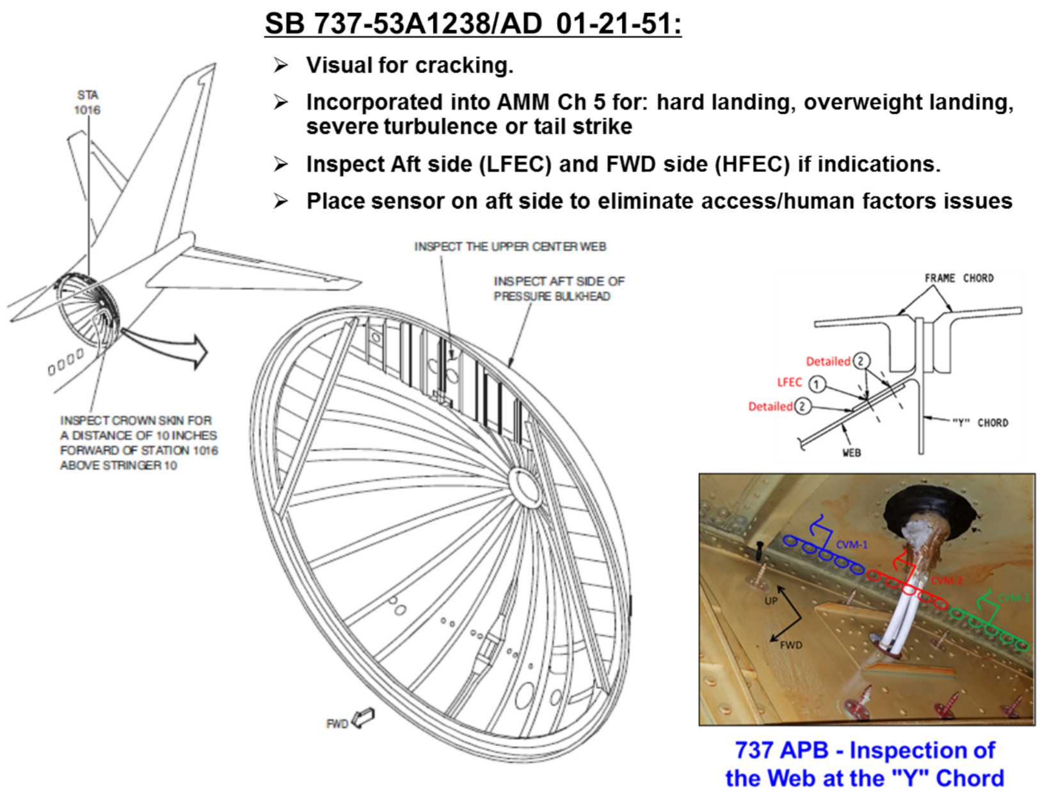


Figure 7-22: Possible SHM Application - Aft Pressure Bulkhead Inspections

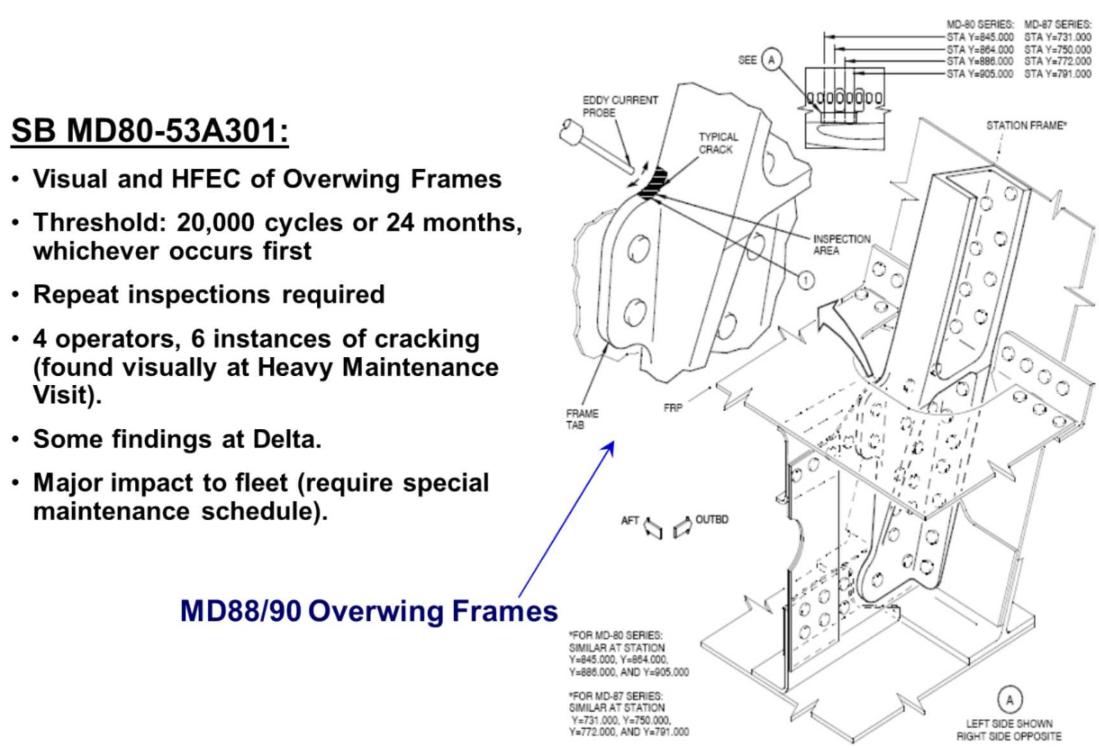


Figure 7-23: Possible SHM Application – Overwing Frame Inspection

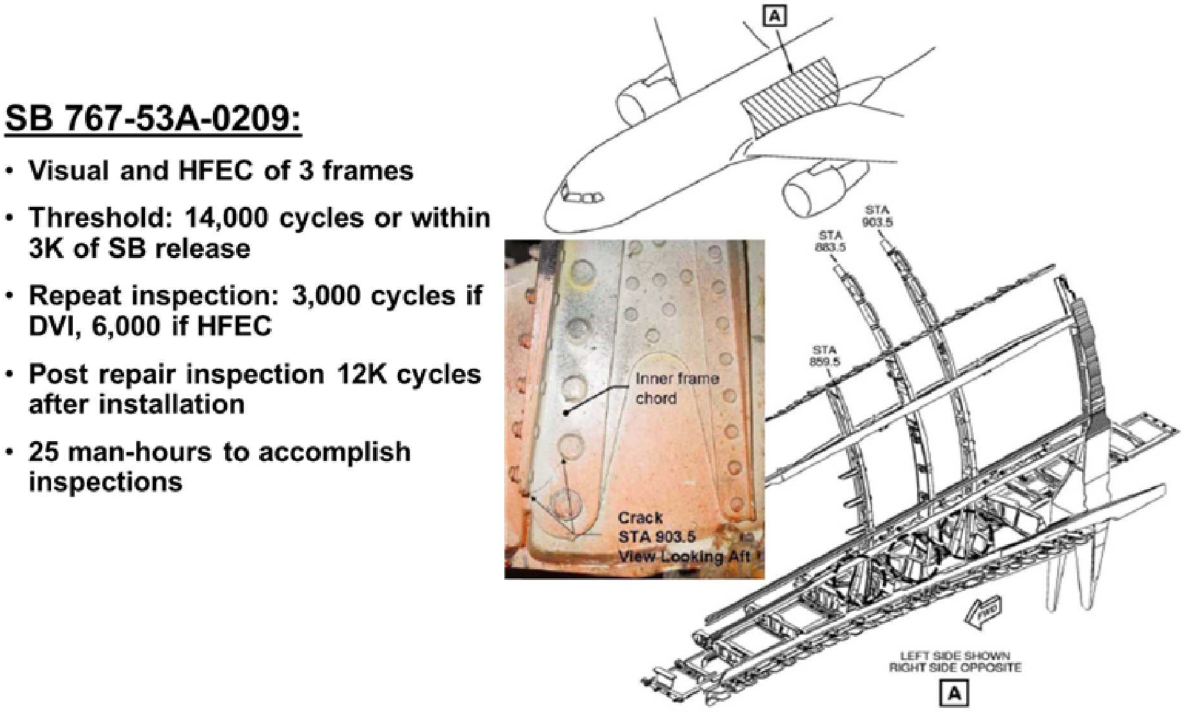


Figure 7-24: Possible SHM Application –Frame Inner Chord Inspection

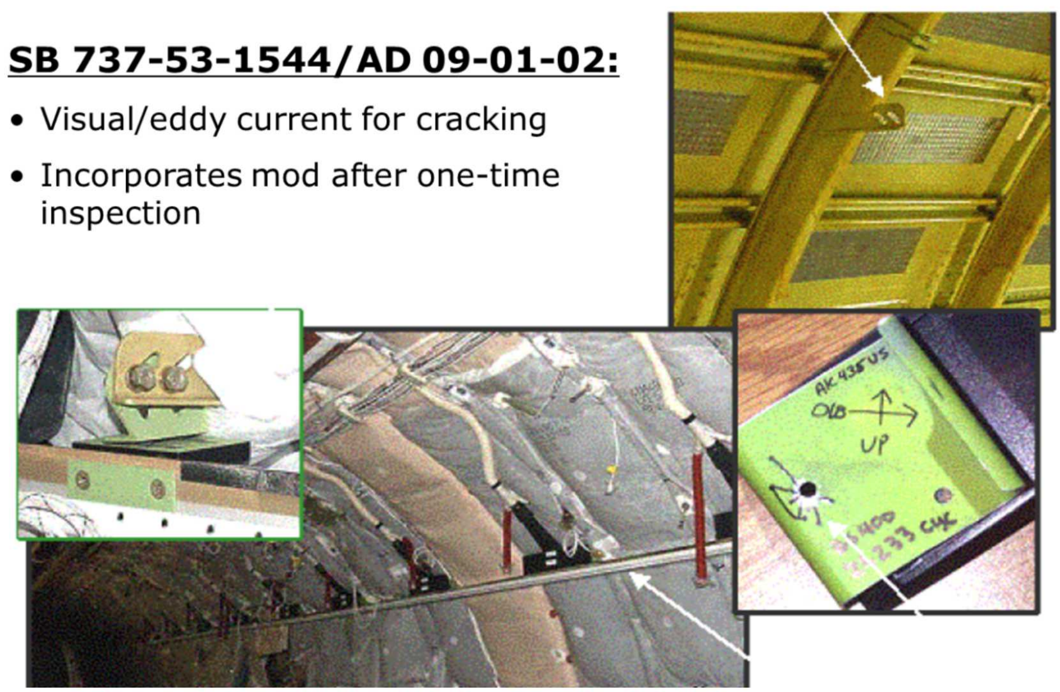


Figure 7-25: Possible SHM Application – Frame Cracks at AC Attach Brackets

SB 767-53A0078/AD 05-11-02 and
SB 767-53A0131/AD 06-24-04:

- Eddy current for cracking
- Threshold =3000 cycles of SB,
- 2100 cycle repeats

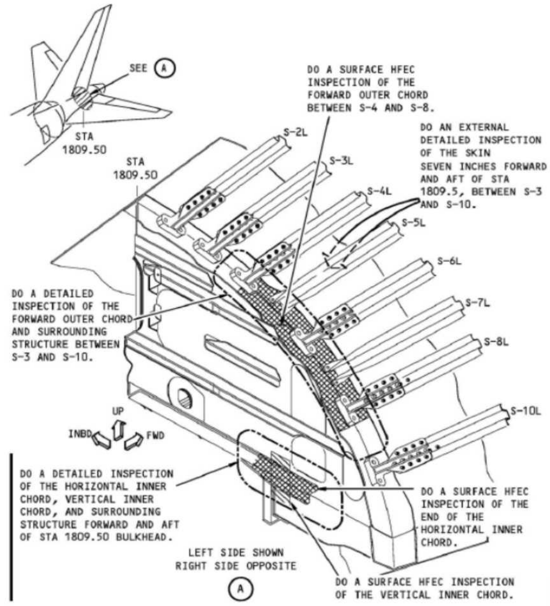
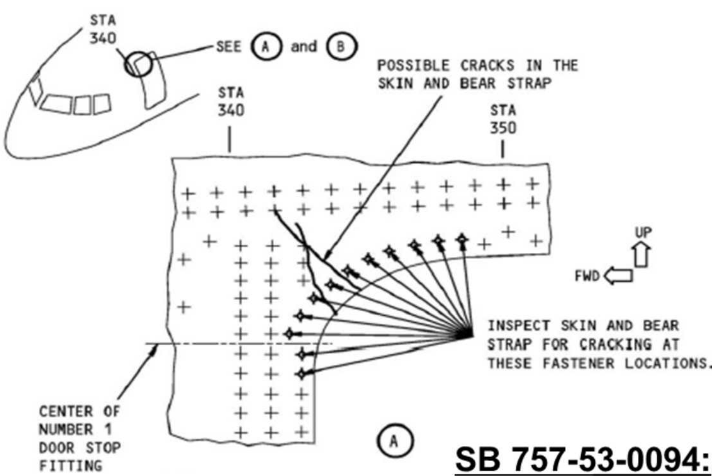
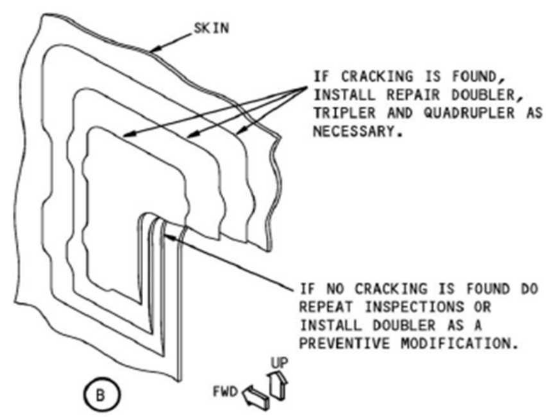


Figure 7-26: Possible SHM Application – Cracks in Bulkhead Outer Chord

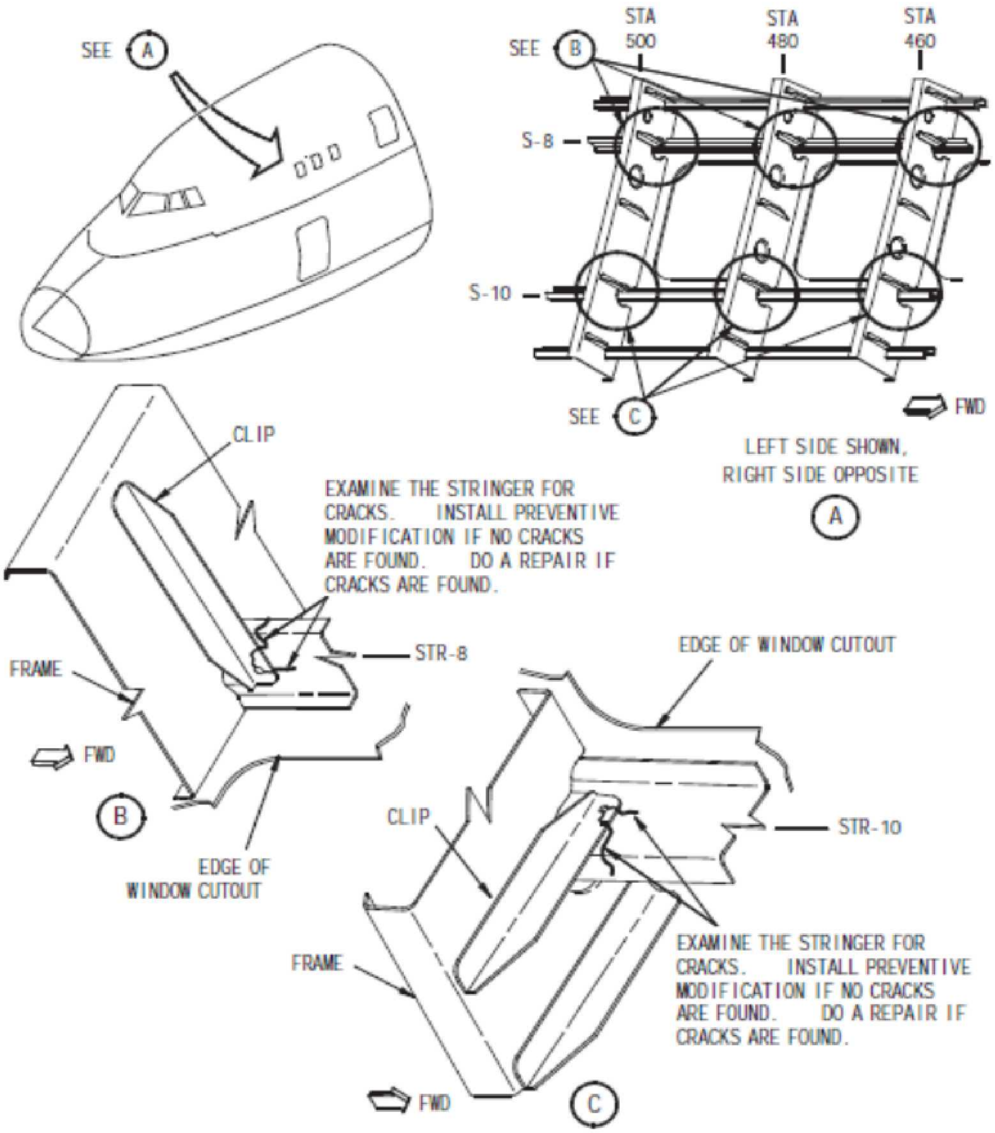




SB 757-53-0094:

- Eddy current for cracking of skin, bearstrap; LFEC of bearstrap
- 1400 cycle repeats

Figure 7-27: Possible SHM Application – Passenger Door Frame Inspection



SB 747-53A2484/AD05-15-08:

- DVI/Eddy current of stringer at several frames
- 3000 cycle repeats

Figure 7-28: Possible SHM Application – Stringer Inspections

References

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7.2 Piotrowksi, D., Roach, D., “Implementation of Structural Health Monitoring (SHM) into an Airline Maintenance Program,” Structural Health Monitoring, DEStech Publisher, Lancaster, PA, September 2015.

7.3 Piotrowski, D., Bohler, J., Melton, A., Roach, D., Rice, T., “Structural Health Monitoring of B737NG Wing Center Section Shear Fittings with Comparative Vacuum Monitoring,” Airlines for America Nondestructive Testing Forum, September 2013.

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7.5 Boeing NDT Standard Practices Manual; Section 51-11-01, General Methods for Inspection of Aircraft, Release 104.00, July 20, 2011.

7.6 Boeing 737 NDT Manual, PART 5 - STRUCTURAL HEALTH MONITORING; Chapter 1: Comparative Vacuum Monitoring, 51-11-01, Center Wing Section 57-01-01.

7.7 Boeing 737 Service Bulletin, “WINGS – Center Wing Box – Front Spar Shear Fitting – Inspection, Repair and Preventive Modification,” Number 737-57-1309, June 2016.

7.8 Embraer, “CVM Structural Health Monitoring System Installation on Wing Structure,” Embraer 190 Aircraft Service Bulletin SB190-00-0027 R00, June 2013.

7.9 Embraer, “CVM Structural Health Monitoring System Installation in the Forward Fuselage Structure,” Embraer 190 Aircraft Service Bulletin SB190-00-0028 R00, June 2013.

7.10 Embraer “CVM Structural Health Monitoring System Installation in Central Fuselage Structure,” Embraer 190 Aircraft Service Bulletin SB190-00-0029 R00, June 2013.

7.11 Form 8110-3, "Statement of Compliance with the Federal Aviation Regulations," issued to indicate appropriate DER/AR approval.

7.12 “Guidelines for Implementation of Structural Health Monitoring on Fixed Wing Aircraft,” SAE Aerospace Recommended Practice ARP 6461, September 2013

7.13 “Guidance for Assessing the Damage Detection Capability of Structural Health Monitoring Systems,” SAE Aerospace Recommended Practice ARP 6821, publish date TBD.

7.14 Federal Aviation Administration, “Comparative Vacuum Monitoring for Damage Detection in Structure of Antenna Installations,” FAA Issue Paper, A-1 under project # ODA-2499-01 for Pre-STC Compliance, Publication TBD (in process).

7.15 Airlines for America (A4A), “MSG-3: Operator/Manufacturer Scheduled Maintenance Development, Volume 1 – Fixed Wing Aircraft; Volume 2 – Rotorcraft,” Rev 2015.1, July 2015.

7.16 Moreu F, Li X, Li S and Zhang D, “Technical Specifications of Structural Health Monitoring for Highway Bridges: New Chinese Structural Health Monitoring Code.” *Front. Built Environ.* 4:10. doi: 10.3389/fbuil.2018.00010, March 2018

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CHAPTER 8

8.0 CVM Equipment Description and Quality Assurance

All hardware associated with the CVM technology is manufactured by Anodyne Electronics Manufacturing Corporation (AEM), a division of Structural Monitoring Systems Ltd. (SMS). AEM is registered as an ISO9001/AS9100C company with many years of experience in producing aviation equipment, and associated kits for airline turnkey use, using proper quality assurance measures. Documents governing general sensor production, quality assurance, installation, and use, along with specifications and hardware kit details for the Wing Box fitting application are provided in [8.1-8.8].

8.1 CVM Sensor Manufacture and Quality Assurance

The CVM sensor is manufactured from multiple layers of Teflon FEP sheets, where 2 to 8 sheets are laminated together with an acrylic pressure sensitive adhesive, the same adhesive is on the bottom layer and facilitates the adhesion to the aircraft.

Sensor production is sensors are controlled by a Manufactured Part Build Standard AEM Doc Code 191XXXXXXX-721-2, and the Assembly Procedures are contained in AEM Doc 191XXXXXXX-630-X. The basic production steps are:

1. Adhesive backed FEP sheet temporarily affixed to an aluminum plate.
2. The pattern for the lower base geometry layers are laser cut, then cleaned.
3. Another layer of FEP is laminated on top of the base geometry layers. This new layer is then laser cut to provide any required bias and outline, creating a partnered layer set.
4. The layers from step 3 are then stacked in proper sequence to create the desired sensor stack up.

Figure 8-1 shows a close-up of a CVM sensor which highlights the ability of the sensor to be customized to any shape and crack detection gallery layout. The photo also shows the “sensor header” region where the tubes, used for connection to the PM200 monitoring device, are interfaced with the thin galleries and built-in gallery routing within the sensor. The inner (Gallery 1) and outer (Gallery 2) galleries are also evident. Figure 8-2 clarifies the sensor design further by showing the custom CVM designed to monitor each Wing Box Fitting and the associated engineering drawings used to guide fabrication.

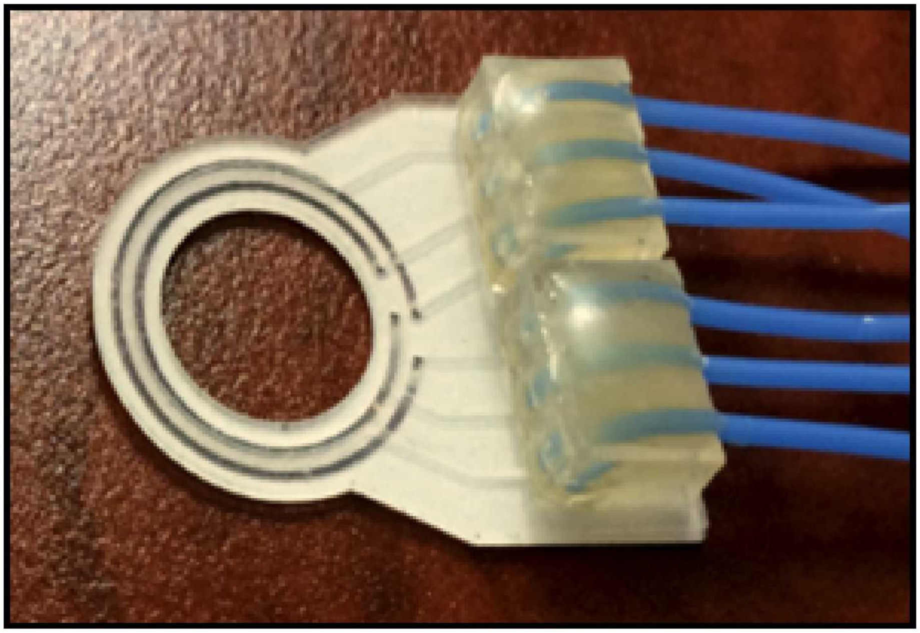


Figure 8-1: Teflon CVM Sensor and Tube Header Assembly

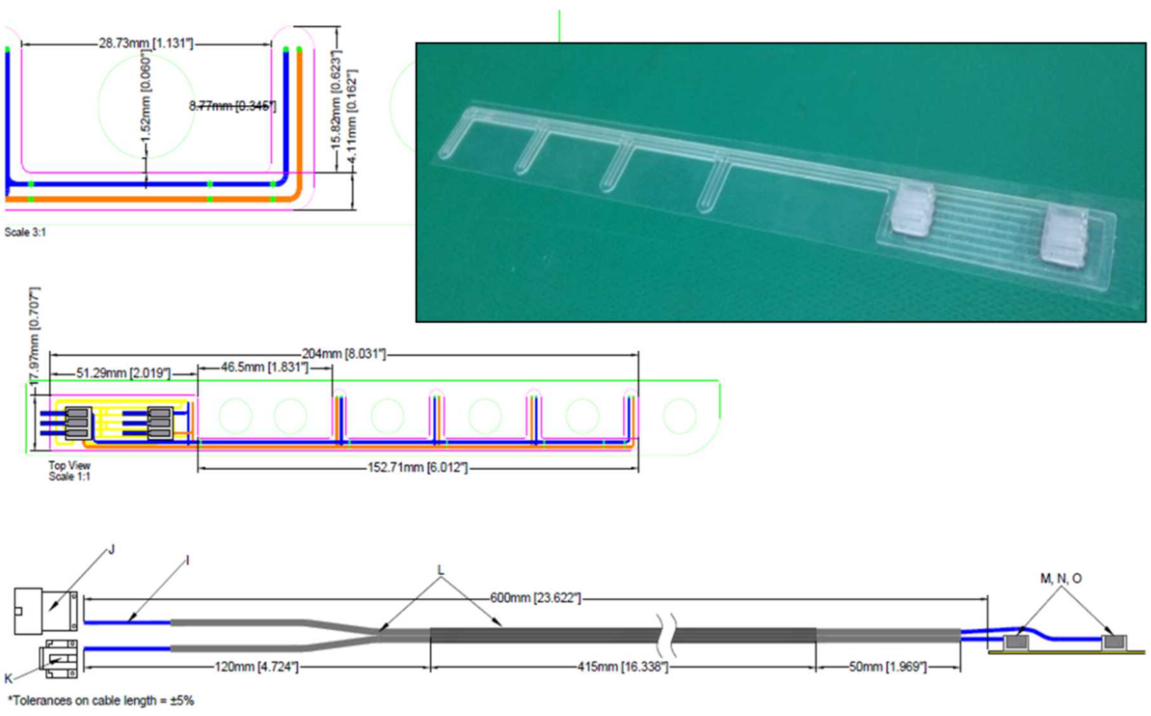


Figure 8-2: Custom CVM Sensor for Wing Box Fitting Application

The Quality Assurance (QA) measures used to ensure repeatable and properly functioning CVM sensor involves a visual inspection under a microscope with simultaneous use of sensor dimension measurement devices. These steps include:

<u>Visual Inspection</u>

1. Layer Alignment – Tolerance for layer alignment is ±0.2mm.
2. Bubbles within 0.2mm of 2 galleries or gallery/sensor edge.
3. Contamination within 0.2mm of 2 galleries or gallery/sensor edge.
4. Contamination within a gallery, blocking it by at least 50%.
5. Glue voids in the header which extend at least 50% of the glue channel.
6. Damaged F.E.P. Teflon.
7. Damaged tubing, or tubing not pressed to the end of the glue channel.

Figure 8-3 shows this process along with some non-conformance issues that can be revealed during the QA inspections and may result in the rejection of a sensor.

Once the CVM sensor is assembled into the final product shown in Figure 8-1, additional functionality checks are conducted. Each sensor is temporarily mounted to a thin plastic sheet for safe, secure storage until use. First, a sensor continuity check is conducted to ensure proper air flow through all galleries. Second, a leak test is conducted with the sensor secured to the temporary, plastic base sheet. This ensures the entire sensor mounting process and ability to seal to a surface to produce its crack detection capability. Figure 8-4 depicts these CVM sensor functionality checks.

The background on the sensor continuity check follows. One important consideration of relying on the detection of airflow (level of vacuum) to detect structural defects is that a blockage anywhere in the pneumatic circuit will mask its presence. For example, a blockage may occur in the connecting tubes, the flow restrictor, or the sensor galleries stopping (or at least artificially restricting) any airflow even if a crack exists. Using the flow meter to check the continuity of the pneumatic circuit mitigates this problem. A Continuity Test is performed by connecting one end of a sensor gallery to the flow meter inlet while leaving the other end open to atmosphere. If there are no blockage between the open end of the gallery and the vacuum reference, air will flow at the maximum rate possible through the flow restrictor. If no flow were registered by the flow meter, this would be indicative of a complete blockage while a flow rate less than the maximum would be indicative to a partial blockage. This type of test is referred to as a continuity test and a basic test schematic is shown in Figure 8-5.

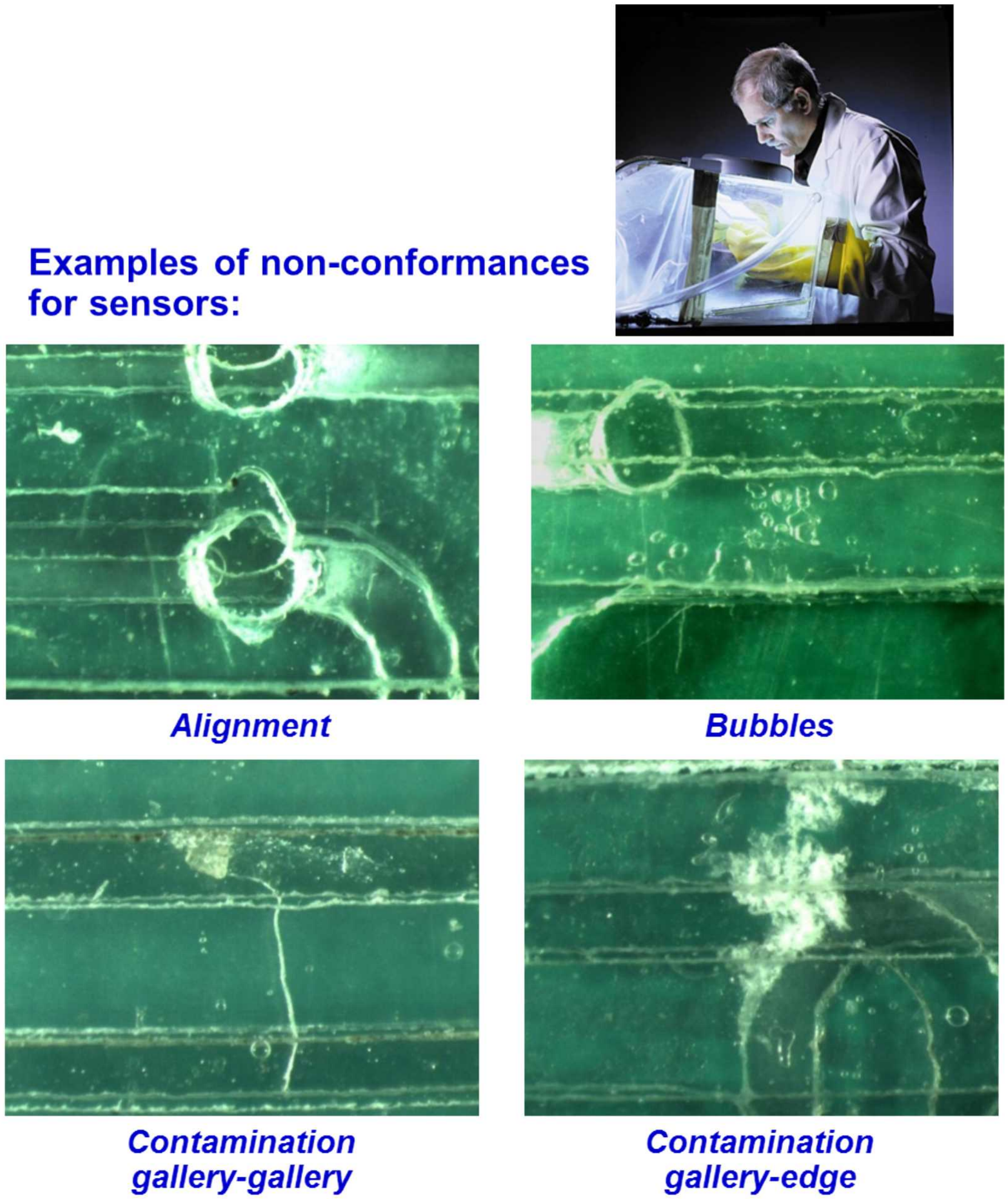


Figure 8-3: CVM Quality Assurance Inspection

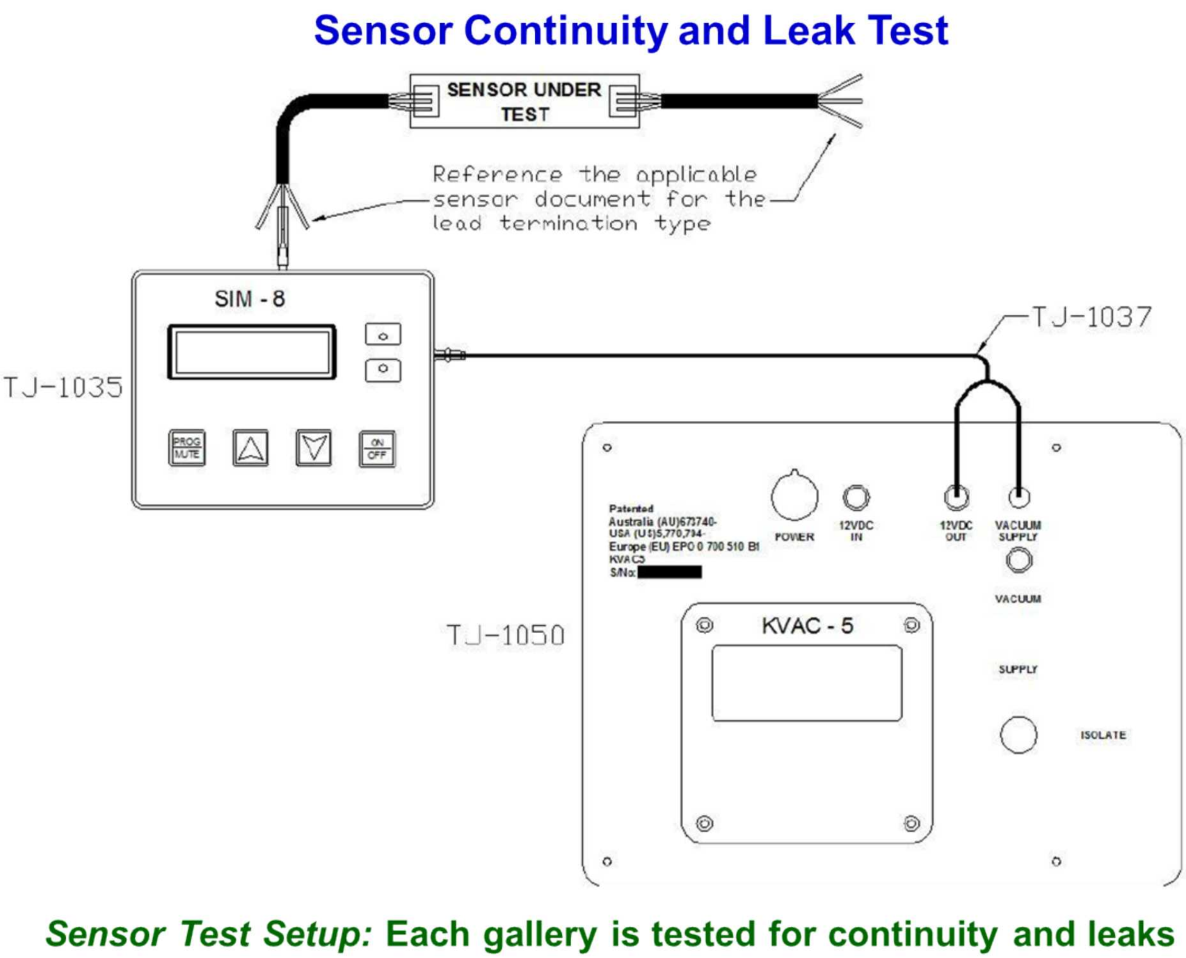


Figure 8-4: CVM Quality Assurance Functionality Checks

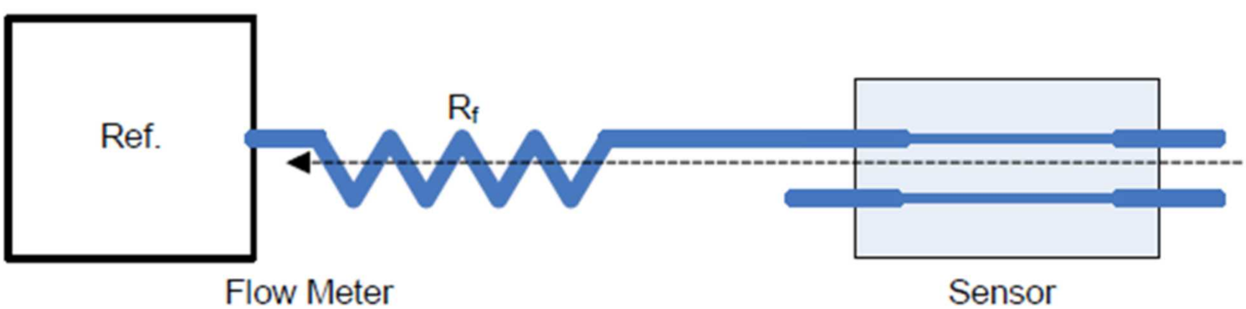


Figure 8-5: Continuity Check - CVM Quality Assurance Check Prior to Each Sensor Monitoring

8.2 PM200 Instrument – Manufacture, Usage, and Calibration

PM200 QA and Calibration - The other important piece of hardware associated with CVM use, is the monitoring device known as the PM200. The PM200, shown in Figure 8-6, is a handheld battery-operated electronic instrument which uses the principles of dCVM to detect structural defects in mechanical components. The PM200 has a built-in, sensitive air flow meter. An air tank and a vacuum pump to provide the vacuum source. The partial vacuum pressure is maintained by the vacuum pump which draws air out of the tank thus lowering the air pressure inside the tank. The PM200 belongs to the Periodic Monitor class of instruments. Periodic Monitoring involves the use of a small number of test instruments (such as the PM200) to monitor the state of many sensors. That is, the state of a particular sensor is determined (i.e. inspected) periodically, perhaps in accordance with a predefined inspection schedule. When an inspection on a sensor has been completed, the instrument can be easily disconnected, transported and reconnected to another sensor. This process is repeated until all sensors have been inspected.

Figure 8-6: PM200 Device Used to Interrogate CVM Sensors and Perform Structural Monitoring

All new PM200 builds contain circuit boards and metalwork that are assembled at AEM under strict QA control. Production Personnel are certified to J-STD-001 and Inspection Personnel are certified to IPC A-610. The AEM manufacturing facility and the PM200 instrument are certified intrinsically safe to IECEx Ex ic IIA T2 and the Certificate of Conformance number is IECEx TSA 08.0012X.

A PM200 Verification Block (PM200-9) is used to provide a set of known and controlled/repeatable reference value flow restrictors. It is used to confirm that the PM200 is operating as intended. The user verifies the operation of the PM200 using the Verification Block at the beginning of each usage (e.g. each day) and once more at the end of each usage (e.g. at the end of each day of testing). This verification provides an assurance that the measurements performed on that day are within specifications. Calibration of the Verification Block occurs annually at AEM's facility. Figure 8-7 shows the Verification Block connected to the PM200 device for calibration prior to use. If the PM200 device fails to produce the proper data (response) to tests with the Verification Block, the user would suspend use of that device until additional calibration and possible refurbishment at the AEM facility.

Figure 8-7: Verification Block for In-Situ Calibration of PM200 Instrument

PM200 Users Manuals - The PM200 has a 5 year cycle for recalibration by the manufacturer and this includes new batteries and a check of the Instrument Lead. Calibration certificates are provided to accompany each instrument. Users Manuals have been prepared for the PM200 device. Section 7.3 introduces the various manuals used to ensure proper, repeatable and safe installation and use of CVM technology. As mentioned in Section 7.3, Delta Air Lines added these CVM manuals to their maintenance documentation to guide: 1) CVM Installation, and 2) CVM Monitoring/Inspection. Included in this were Introduction to CVM Manual, PM200 Operations Manuals, CVM installation training, and CVM installation procedures. Figure 7-5 through Figure 7-9 provide an overview of the six documents that provide the comprehensive set of information needed to properly and safely utilize CVM technology. The information included in the PM200 User's Manual (hardware) is summarized in Figure 8-8. There is a mating PM200 software User's Manual that describes the software operation. On overview of the PM200 Management Software Manual is shown in Figure 8-9. A brief description of the software manual contents follows:

- Chapter 1: Getting Started - Chapter 1 introduces the software. This chapter describes how to start the program and to configure the application for use.
- Chapter 2: Application Overview - Chapter 2 describes the PM200 Management Software's main screen. Elements of the main screen such as the toolbar, menus and details section are explained to give the reader a full understanding of how to navigate through the software.
- Chapter 3: PM200 Configuration - Chapter 3 describes how to configure a PM200. This chapter describes the instruments details, inspection list, operator list, test point update list and date /time. Common operations such as adding, modifying and deleting PM200 data items are also covered.
- Chapter 4: PM200 Database - The Chapter 'PM200 Database' describes the PM200 database and how to view inspections contained within the database. The chapter on 'Data Sharing' expands on this chapter by describing how to share inspection data with others.
- Chapter 5: Data Sharing - The chapter 'Data Sharing' covers how to import and export inspection data.

Physics of PM200 Operation – The Comparative Vacuum Monitoring (CVM) sensor has been developed on the principle that a small volume maintained at a low vacuum is extremely sensitive to any ingress of air and is thus sensitive to any leakage. CVM is the measurement of tiny airflows that are used to detect defects in structures. Figure 2-9, Figure 8-10 and Figure 8-11 show top-view and side-view schematics of the flexible, self-adhering, elastomeric sensors with fine channels etched on the adhesive face. When the sensors are adhered to the structure under test, the fine channels and the structure itself form a manifold of galleries alternately at low vacuum and atmospheric pressure. Generally, a sensor has two galleries which are spaced a small distance apart depending on the application. If a flaw is not present, the low vacuum remains stable at the base value. If a flaw develops, air will flow from the atmospheric galleries through the flaw to the vacuum galleries. When a crack develops, it forms a leakage path between the atmospheric and vacuum galleries, producing a measurable change in the vacuum level. This change is detected by the CVM monitoring system (PM200 device).

PM200 Hardware Manual Contents

Figure 8-8: Contents of PM200 User's Manual Describing System Hardware

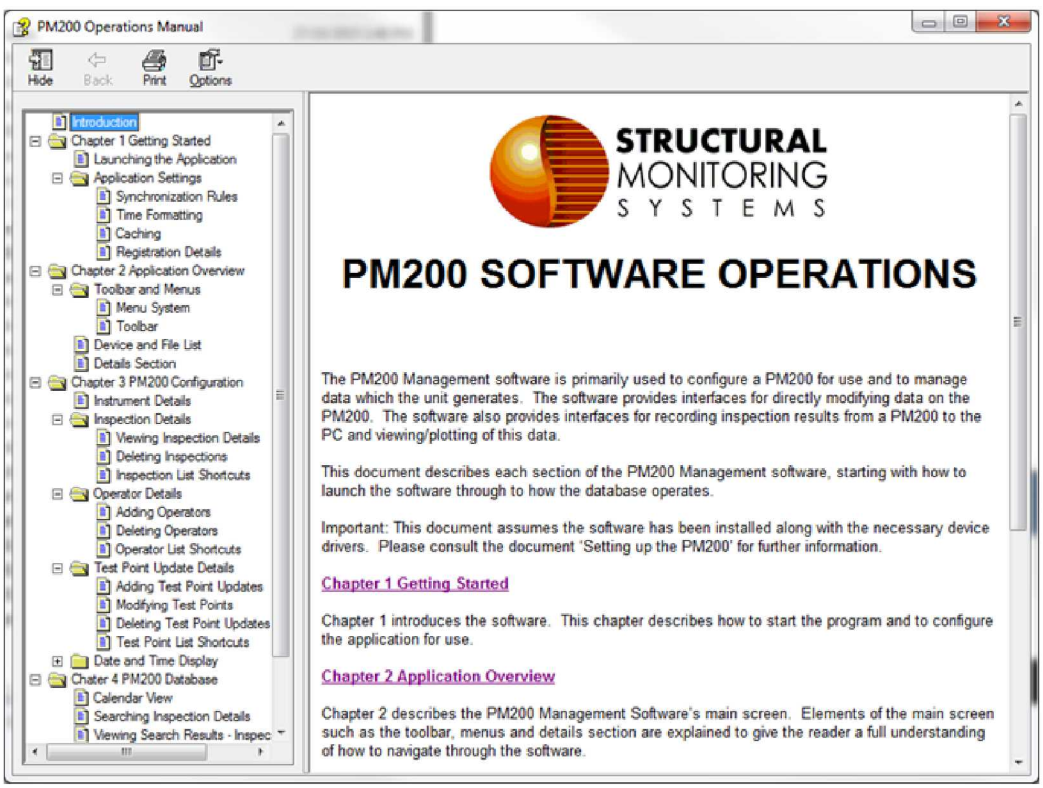
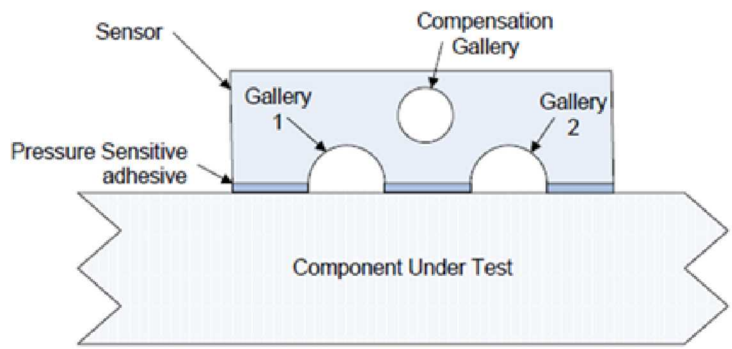


Figure 8-9: Description of Device Software in PM200 Management Software Manual

Since the sensor physics is based on pressure measurements, there is no electrical excitation involved. These sensors can be attached to a structure in areas where crack growth is known to occur. On a pre-established engineering interval, a reading will be taken from an easily accessible point on the structure. Each time a reading is taken, the system performs a self-test. This inherent fail-safe property ensures the sensor is attached to the structure and working properly prior to any data acquisition.



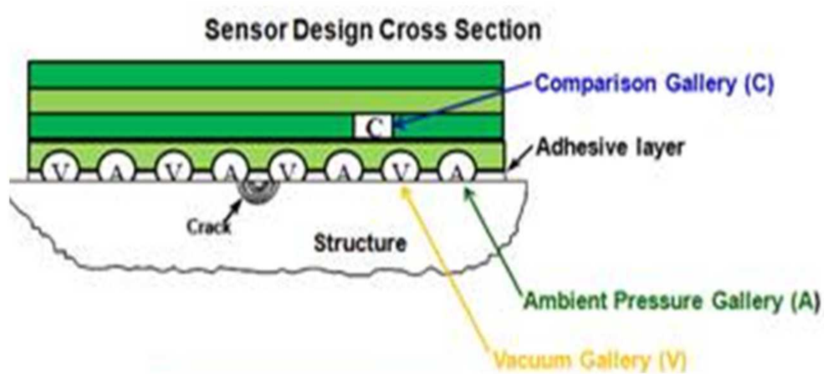


Figure 8-10: Schematic Showing Cross-Section of a CVM Sensor

Consider the case where one end of a gallery is sealed by inserting an airtight plug into the connection tube, while the other end of the same gallery is connected to a vacuum source that is constant relative (partial vacuum) to ambient (barometric) pressure. Leaving the connection tubes at each end of the second gallery open keeps the gallery at ambient pressure. If it is assumed that the sensor is properly fitted to the surface of a component, then no air will flow between the two galleries. However, consider the scenario where a structural defect in the component has propagated to the surface in the form of a crack. Depending on the surface dimensions of the crack, it is possible for the crack to breach the airtight seal between the galleries resulting in airflow from the atmosphere to the negative pressure source. Detecting such air flows in order to indicate the presence of a structural defect in mechanical components is the primary function of the PM200.

Measurement of dCVM Values in Sensor are Used to Indicate the Presence of a Crack - Differential Comparative Vacuum Monitoring (dCVM) extends this concept to include an extra pneumatic circuit called the compensation gallery (see Figure 8-10 and Figure 8-11). This gallery is the same length and size and constructed from the same materials as the measurement galleries, however, it is not in contact with the component surface or any cracks in this surface. The sensitivity of the CVM technique means that outside influences such as temperature changes are seen by the instrument as a noise signal which reduces signal to noise ratio. By calculating the difference between the measured gallery and the compensation gallery, these noise signals are reduced or eliminated, thus improving the sensitivity of the sensor. It is important that the measurement and compensation galleries match as well as possible.

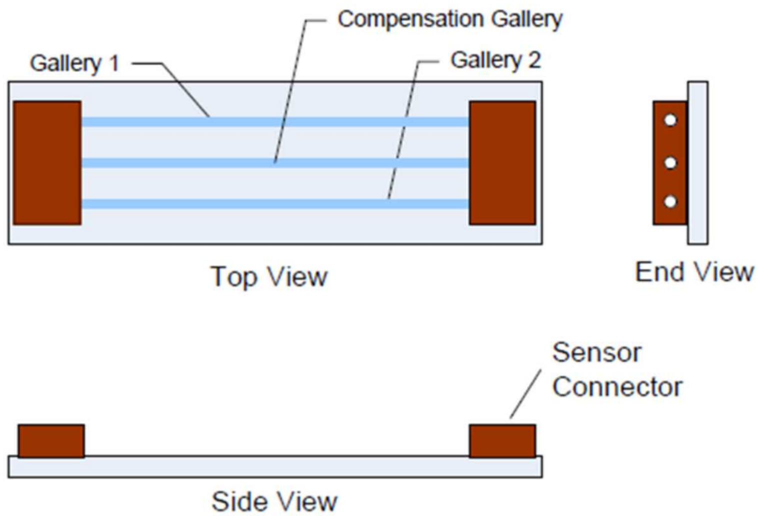


Figure 8-11: Schematic of CVM Sensor Showing Custom Positioning of Galleries to Meet Crack Detection Needs

Figure 8-11 shows a top view of a differential sensor. The sensors are made of a transparent material so it is possible to see the route taken by the galleries when looking down onto the pad. As indicated by the side and end views, the ends of each gallery are brought out to a connector that enables the external tubing to be connected to the sensor. The end goal of the PM200 is to provide crack detection using a loss of vacuum in the sensor gallery. It does this by measuring the dCVM parameter whose value is related to the ability to pull a vacuum on the gallery. The sensors include three separate pneumatic galleries. Two of these galleries are open channels that are directly exposed to the substrate the user intends to monitor. The other gallery is the compensation gallery. It is an isolated gallery of the same physical dimensions and environmental exposure as the measurement galleries.

When a measurement is made, the PM200 simultaneously pulls a vacuum on the compensation gallery as well as one of the measurement galleries. A differential measurement (P1 – P2) is calculated. This differential measurement allows cancellation of environmental effects as well as faster measurement times as the gallery degassing profiles are matched. Figure 8-12 and Figure 8-13 are schematics that clarify the measurement process.

The pneumatic values within the CVM system measured in the PM200 are as follows:

- Pressure measured in Pa (Pascals)
- Flow measured in m^3/sec
- Impedance measured in (Pa) (m^3/sec)

These pneumatic values are then used to calculate the dCVM parameter as follows. Impedance is the resistance to flow and can be used to gauge crack size. Crack size indication is expressed as CI (Conductivity Index) and is achieved by inverting impedance:

$$CI = 1/\text{Impedance} \qquad (8.1)$$

The PM200 is used to measure the difference between two separated channels, the measured channel and the compensation channel which is sealed and provides a clear indication of low vacuum level. Thus, CVM is a measurement of air flow, in units CI where 0.1CI = 135 nL/min. And, dCVM is the difference between the reference Compensation Channel value in CI and the Measurement Channel value in CI.

$$\mathbf{dCVM} = CI_{(\text{Compensation})} - CI_{(\text{Measurement})} \qquad (8.2)$$

This subtraction of the Measurement Channel from the reference Compensation Channel provides cancellation of temperature and humidity effects so it compensates for measurements at different conditions. This approach also provides extreme sensitivity to any leakage in the galleries which, in turn, provides high Signal-to-Noise ratios for crack detection. In summary, the CVM sensor design is based on the principle that a steady-state vacuum, maintained within a small volume, is extremely sensitive to any leakage.

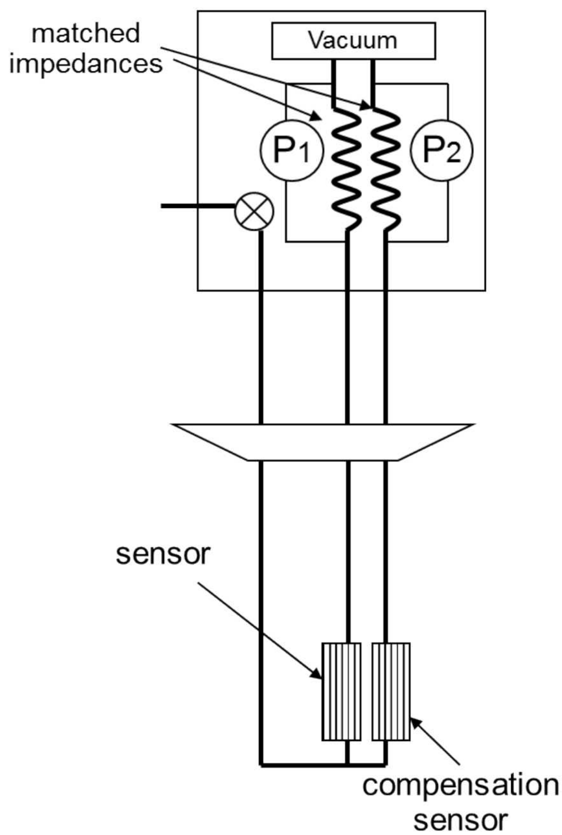


Figure 8-12: Differential Pressure Measurements Used to

Produce dCVM Parameter for Crack Detection

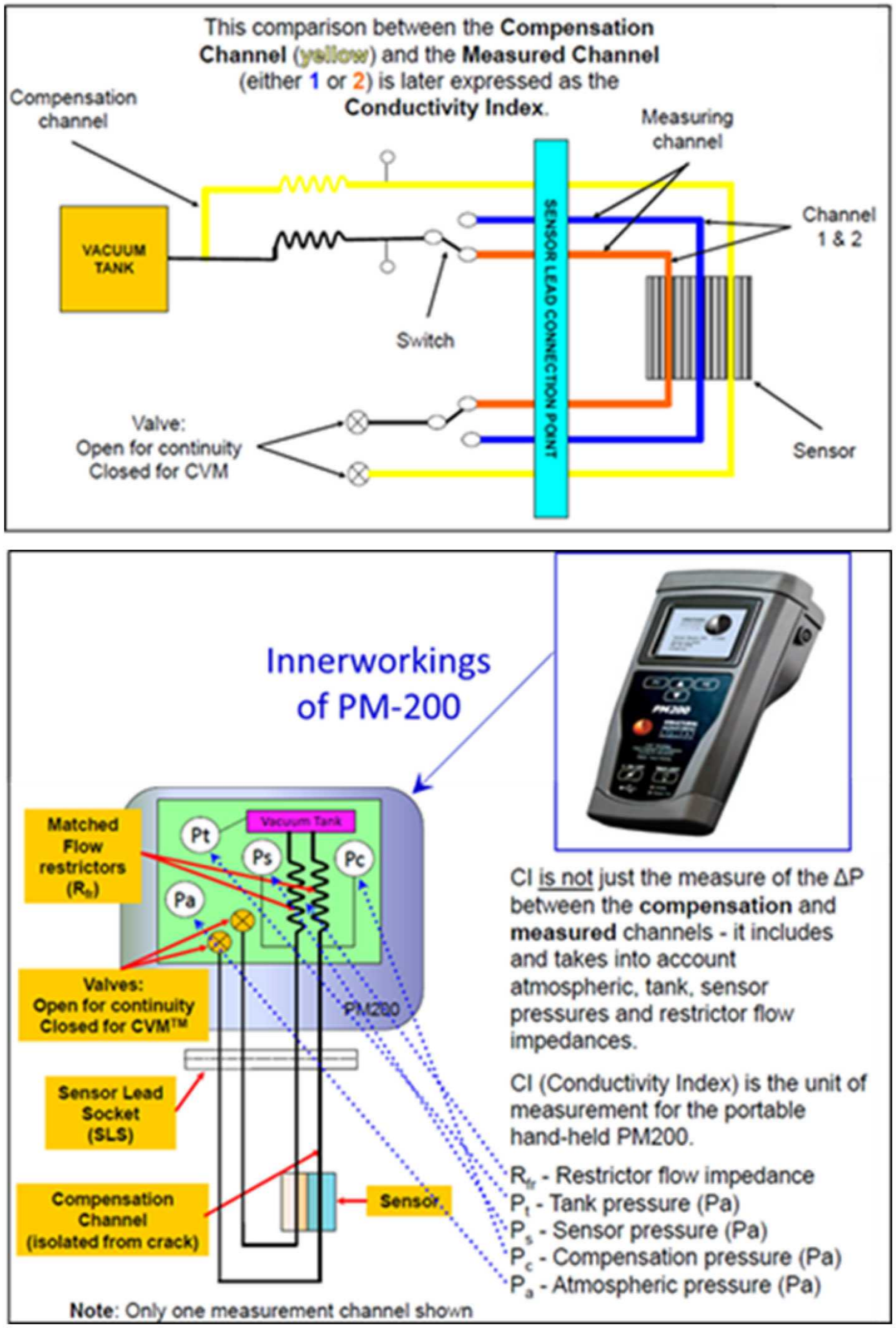


Figure 8-13: Overview of PM200 Measurements to Assess the Presence of a Crack

8.3 CVM Sensor and Connectors Durability

The Teflon CVM sensors use a bonding process to interface the PFA tubing to the sensor. The Sandia environmental tests described in Section 5.0 and the flight tests validated the proper performance of this header design shown in Figure 8-1. Header bonding tests and tubing retention tests were also performed. Figure 8-14 shows some sample results from the structural tests used to ensure sufficient strength of the CVM header interface.

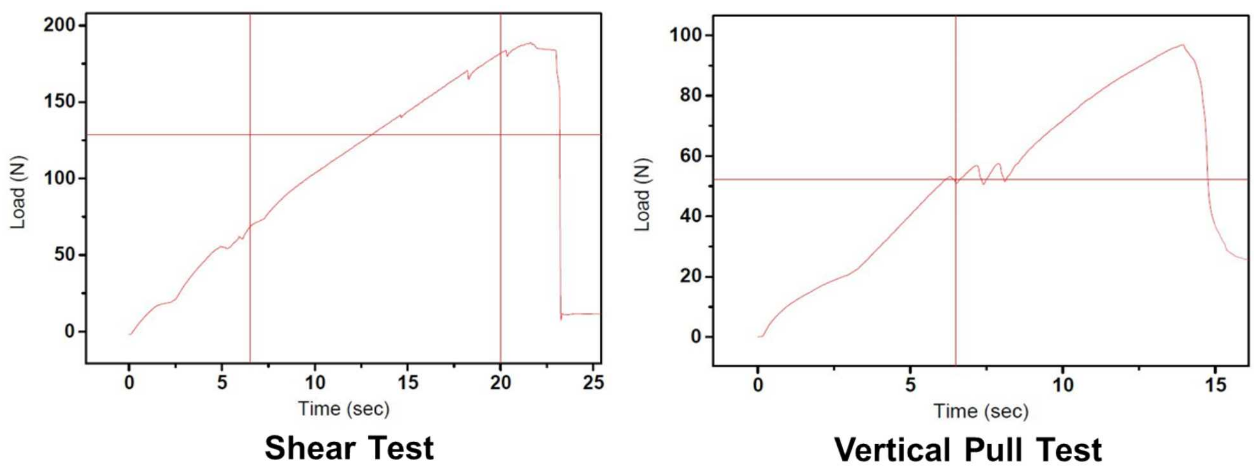


Figure 8-14: Pull and Shear Tests of CVM Header to Assess the Bonding Interface to CVM Sensor

Temperature testing for the CVM sensor (Teflon sheet with galleries and tube interface header) was conducted to conduct durability evaluations and to support engineering analyses. These tests were performed to DO-160G Cat C4 using the temperature profile shown in Figure 8-15. Functionality testing during the temperature exposure tests, revealed that the pneumatic seal in the header and all CVM sensor features remained flawless during this testing.

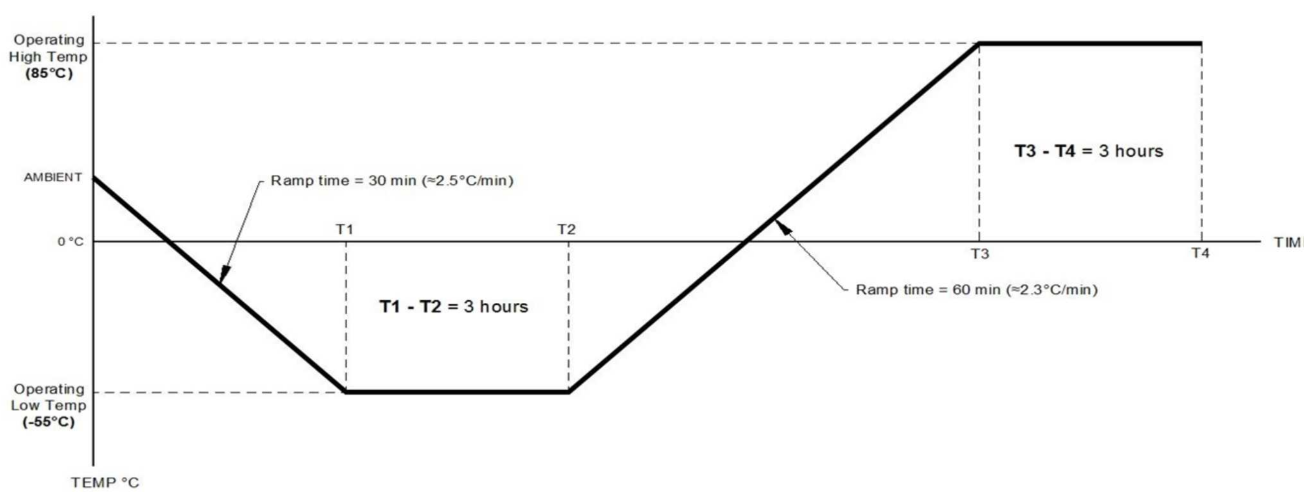


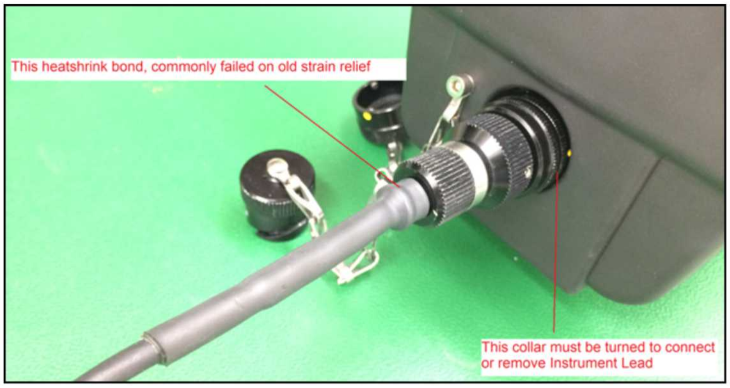
Figure 8-15: DO160 Temperature Profile Used to Test CVM Sensors and Header

The SLS connectors form the completion end of the sensor leads. These connectors, shown in Figure 5-3, Figure 5-4, Figure 5-6, Figure 6-11, Figure 6-12 and Figure 6-18, are left onboard the aircraft and are already aviation-certified connectors. They are ITT Aerospace Grade connectors that were tested in all of the environmental and durability tests – both laboratory and flight tests – described above. AEM completed additional testing on the SLS connectors with the objective of exercising the mating of these connectors at the temperature extremes, thus challenging the O-ring performance in worst case conditions.

Five SLS connectors were cycled 100 times at alternating hot and cold extremes. Each SLS connector was also fitted with an ID chip as it would be in normal operation. A normal female was mounted on an aluminum bracket. This female was manually connected to a PM200 for analysis after each cycle was complete. The test protocol was the same as it would be for normal field inspections. The brackets, together the SLS connectors, were placed in an environmental chamber overnight at -20 ^{o}C and +65 ^{o}C alternating nightly. These temperatures were selected as they are the normal operating temperatures specified for the PM200. After each 100 mating cycles, all SLS connectors on the bracket were connected to a PM200 and tested. The initial objective was to perform 1000 mating cycles on each connector. No SLS failures were found in the initial set, so testing was extended to 1400 cycles. Even after 1,400 connection mating cycles, no failures were observed in the SLS connectors. Thus, it was concluded that users should not experience any problems with SLS connection within the anticipated 1000 mating cycle lifetime of this connector.

The PM200 Instrument Lead, shown in Figure 5-3, Figure 5-4, Figure 8-6 and Figure 8-7, connects the on-board SLS connector to the PM200 unit. It does not fly with the aircraft but is brought to the airplane, along with the PM200 device, for sensor interrogation and data acquisition. The original design of the PM200 Instrument Lead used multiple layers of adhesive lined with heatshrink tubing to create a strain relief at the cable-to-PM200 connector interface. The bonding of this adhesive to the metal connector backshell was inadequate and frequently failed. Further reports from customers indicated that the available grip on the connector was too small to engage/disengage the connector from the PM200. A new connector shell was developed, along with a custom in-house, molded, strain relief. Subsequent testing of the PM200 Instrument Lead indicated that the current design eliminates any anticipated problems in connecting the PM200 to the on-board SLS connector for data acquisition. Figure 8-16 compares the two designs and shows the current configuration of the PM200 connection.

The last piece of hardware in the sensor-to-PM200 connection chain is the Snap-Click connector. The Snap-Click connector, and its use to make custom daisy-chains of CVM sensors or to mate the sensor tubes to the SLS connector, is shown in Figure 5-4, Figure 5-6, Figure 6-11, Figure 6-17, Figure 8-2 and Figure 8-17. Testing was conducted on the Snap-Click Connector to determine if it could survive 100 mating cycles. A set of connector pairs were mated 100 times each. The leakage from this connection was then measured and determined to be less than 20Pa after each connection. No degradation in performance was found during the fatigue, durability tests. Additional Snap-Click connectors were monitored while they were temperature cycled from room temperature down to -55 ^{o}C and no performance degradation was observed.



Old Strain Relief and Connector Backshell

New Strain Relief and Connector Backshell

Figure 8-16: Connection and Strain Relief of PM200 Instrument Lead

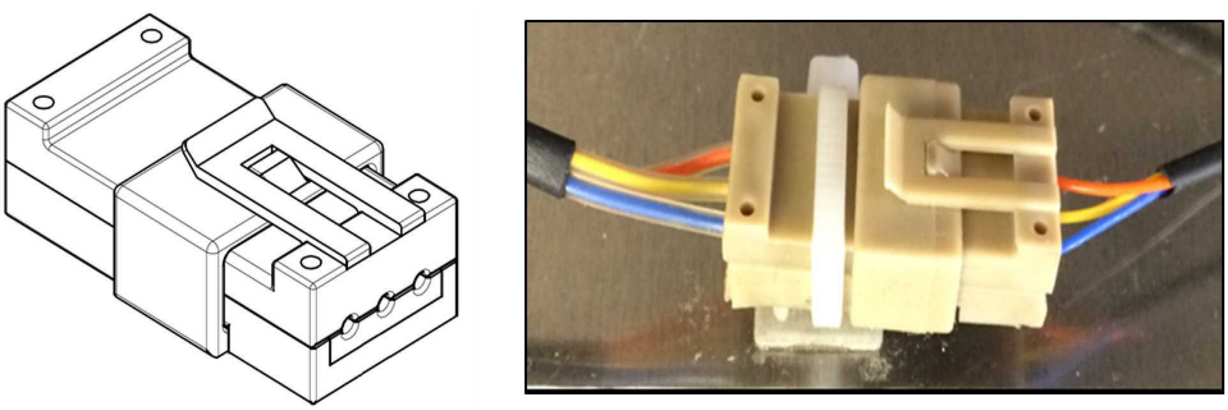


Figure 8-17: Snap-Click Pneumatic Connector

8.4 737 Wing Box Fitting Installation Kit

As introduced in Section 7.4, 737 CVM Wing Box Fitting kits were prepared in accordance, with normal Quality Assurance procedures, and assigned a formal number for ordering from the AEM supplier. The kits included the sensors along with all other on-board hardware and installation aids. Normal surface preparation tools, such as Acetone, primer and fine grit sandpaper, were checked for routine availability within Delta stores and were not included in the custom CVM kits. The 737NG-FSSF-IKCVM Installation Kit contains all hardware, sensors, and cables required to equip a Boeing 737-NG with CVM at (LBL) 54.60, 40.87, 32.40, 15.04, 6.14 and (RBL) 54.60, 40.87, 32.40, 15.04, 6.14, at Body Station (STA) 540.

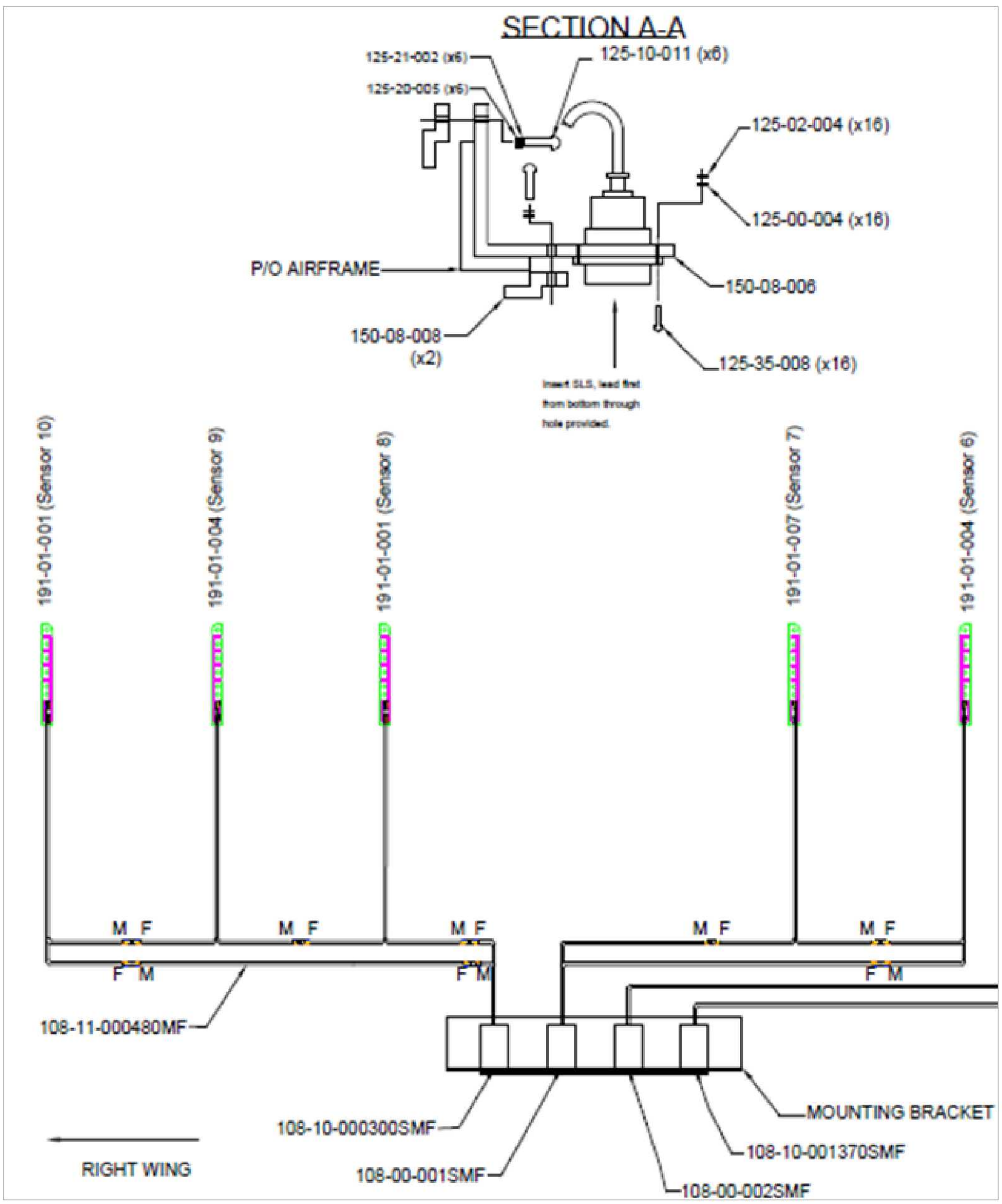


Figure 8-18: 737 Wing Box Fitting Installation Kit – Right Wing Side

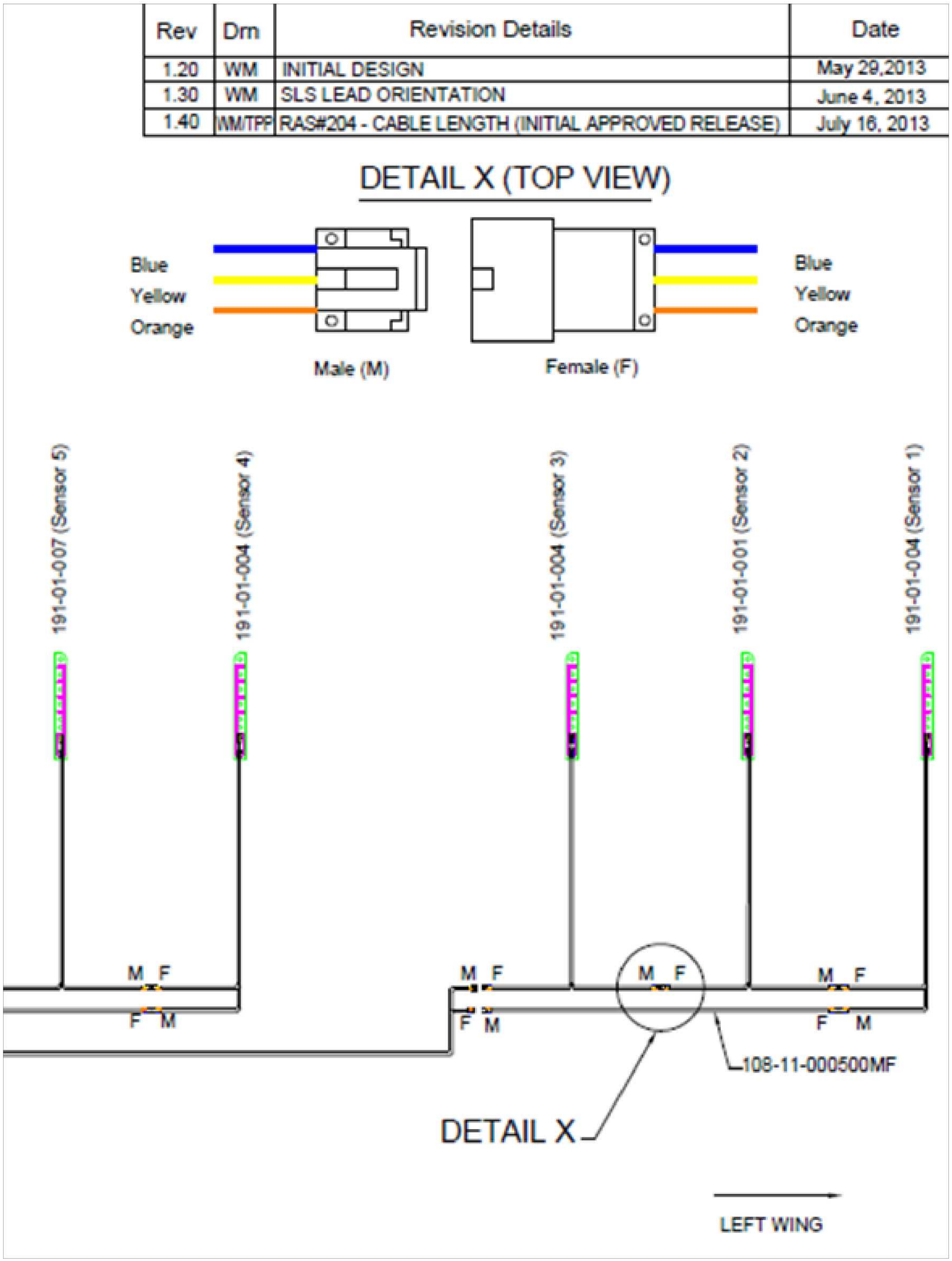
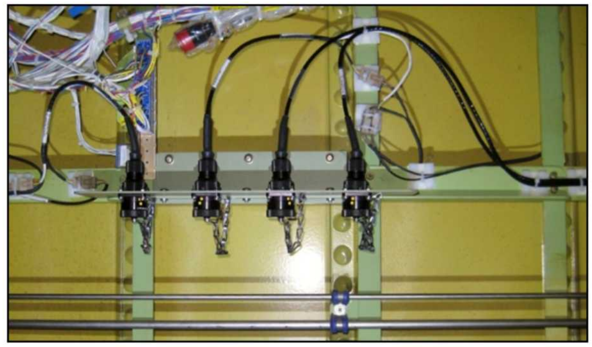


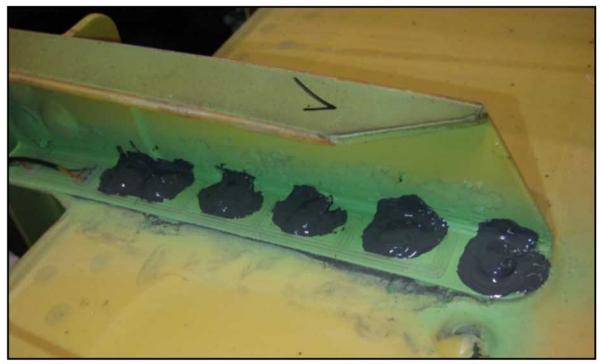
Figure 8-19: 737 Wing Box Fitting Installation Kit – Left Wing Side

Parts List for Installation Kit

Quantity	Description	Part #
5	Multisite RC 1x5, Int 0x6, Right	191-01-004
3	Multisite RC 1x5, Int 0x6, Left	191-01-001
2	Multisite RC 1x5, Int 0x6, Left	191-01-007
1	SLS, Lead, SCM=450mm SCF=320mm	108-00-001SMF
1	SLS, Lead, SCM=830mm SCF=960mm	108-00-002SMF
1	SLS, Lead, SC, L=300mm	108-10-000300SMF
1	SLS, Lead, SC, L=1370mm	108-10-001370SMF
1	Interconnect Lead, SC, L=480mm	108-11-000480MF
1	Interconnect Lead, SC, L=500mm	108-11-000500MF
1	Mounting Bracket, FSSF SLS	150-08-006
2	Mounting Bracket, Clamp	150-08-008
6	Screw Panhead Phillips C/S	125-10-011
75	Mount, Cable Tie Anchor, Size 3	125-99-006
75	Peek, Thermoplastic, Zip Ties, Size 3	156-30-002
16	Nut S/S Non Locking	125-00-004
16	Screw Panhead Phillips	125-35-008
16	Nut Locking Jam nut S/S	125-02-004
6	Washer Split Locking C/S	125-21-002
6	Washer Flat C/S	125-20-005



Installed bracket and SLS connectors



Installed Sensor, prior to FVB overcoat

**Figure 8-20: Comparative Vacuum Monitoring Installation Kit Contents
Part Number: 737NG-FSSF-IKCVM**

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8.8 SMS Technical Document, "dCVM Operations Manual – Part B PM200 Software," Doc. 000187MN Part B/3.3.

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CHAPTER 9

9.0 Summary - Validation of CVM Sensors for SHM Crack Detection

Background and Motivation for CVM Usage

Recent advances in on-board structural health monitoring sensors have proven that distributed and autonomous health monitoring systems can be applied to reliably detect incipient damage. Such systems have wide use in aerospace, automotive, civil infrastructure and other industrial applications. This study and associated report presents data that proves the viability of the Comparative Vacuum Monitoring (CVM) system for implementation on commercial aircraft. The application of SHM systems for monitoring the structural integrity of aircraft provides alternatives to invasive inspections. The success of SHM solutions and the decision to implement them ultimately hinges on the capability of the system to reduce the risk of structural failure while providing economic benefit in terms of maintenance cost savings and aircraft availability.

Comparative Vacuum Monitoring (CVM) has been developed on the principle that a small volume maintained at a low vacuum is extremely sensitive to any ingress of air and is thus sensitive to any leakage. When the sensors are adhered to a structure under load, the fine channels and the structure itself form a manifold of galleries alternately at low vacuum and atmospheric pressure. When a crack develops, it forms a leakage path between the atmospheric and vacuum galleries, producing a measurable change in the vacuum level. Thus, Comparative Vacuum Monitoring is a SHM technology that can monitor the onset and growth of structural cracking. These sensors can be attached to a structure in areas where crack growth is known to occur. On a pre-established engineering interval, a reading will be taken from an easily accessible point on the structure. Each time a reading is taken, the system performs a self-test. This inherent fail-safe property ensures the sensor is attached to the structure and working properly prior to any data acquisition.

Through the use of in-situ CVM sensors, it is possible to quickly, routinely, and remotely monitor the integrity of a structure in service [9.1 – 9.3]. Prevention of unexpected flaw growth and structural failure can be improved if on-board health monitoring systems are used continuously assess structural integrity and signal the need for human intervention [9.4 – 9.6]. Recent events have demonstrated the need to address critical infrastructure surety needs [9.7]. The applications for CVM sensors can include such diverse structures as: buildings, bridges, trains and subway vehicles, mining structures, railroad cars, trucks and other heavy machinery, pressure vessels, oil recovery equipment, pipelines, steel transmission towers, ships, tanks and a wide array of military structures. This report focuses on the application of CVM technology to aircraft structure.

The application of Structural Health Monitoring (SHM) systems using distributed sensor networks can reduce maintenance costs by facilitating rapid and global assessments of structural integrity. These systems also allow for condition-based maintenance practices to be substituted for the current time- or cycle-based maintenance approach thus optimizing maintenance labor.

Other advantages of on-board distributed sensor systems are that they can eliminate costly, and potentially damaging, disassembly, improve sensitivity by producing optimum placement of sensors with minimized human factors concerns in deployment and decrease maintenance costs by eliminating more time-consuming manual inspections. Current maintenance operations and integrity checks on a wide array of structures require personnel entry into normally-inaccessible or hazardous areas to perform necessary nondestructive inspections. To gain access for these inspections, structure must be removed, sealant must be removed, disassembly processes must be completed, or personnel must be transported to remote locations. The use of in-situ sensors, coupled with remote interrogation, can be employed to overcome a myriad of inspection impediments stemming from accessibility limitations, complex geometries, the location and depth of hidden damage, and the isolated location of the structure.

Application of CVM for a Specific Aircraft Application

A research program was completed to develop and validate Comparative Vacuum Monitoring (CVM) Sensors for surface crack detection. The program addressed formal SHM technology validation and certification issues so that the full spectrum of concerns, including design, deployment, performance and certification were appropriately considered. The goal was to move beyond the traditional prototype field testing completed in the first decade of 2000 and move into mainstream, industry-wide adoption of SHM. Towards that end, the Airworthiness Assurance NDI Validation Center (AANC) at Sandia Labs, in conjunction with Boeing, Delta Air Lines, Structural Monitoring Systems Ltd., Anodyne Electronics Manufacturing Corp. and the Federal Aviation Administration (FAA) carried out a certification program to formally introduce Comparative Vacuum Monitoring (CVM) as a structural health monitoring solution to a specific aircraft wing box application. Validation tasks were designed to address the SHM equipment, the health monitoring task, the resolution required, the sensor interrogation procedures, the conditions under which the monitoring will occur, the potential inspector population, adoption of CVM into an airline maintenance program and the document revisions necessary to allow for routine use of CVM as an alternate means of performing periodic structural inspects.

All factors that affect SHM sensitivity were included in this program: flaw size, shape, orientation and location relative to the sensors, as well as operational and environmental variables. The test matrix studied the effects of surface coating, skin thickness, and material type on the performance of the CVM sensors. Statistical methods were applied to performance data to derive Probability of Detection (POD) values for CVM sensors in a manner that agrees with current nondestructive inspection (NDI) validation requirements and also is acceptable to both the aviation industry and regulatory bodies. The result is a series of flaw detection curves that can be used to propose CVM sensors for crack detection. Complimentary, multi-year field tests were also conducted to study the deployment and long-term operation of CVM sensors on aircraft. This report presents the quantitative crack detection capabilities of the CVM sensor, its performance in actual flight environments, and the prospects for structural health monitoring applications on commercial aircraft and other civil structures.

The activities conducted in this program demonstrated the feasibility of routine CVM usage for the application selected. They also helped establish an optimum OEM-airline-regulator process and determined how to safely adopt SHM solutions. This formal SHM validation will allow aircraft manufacturers and airlines to confidently make informed decisions about the proper utilization of CVM technology. It will also streamline the regulatory actions and formal certification measures needed to assure the safe application of SHM solutions.

SHM Validation and Verification

The SHM Validation Program used a multi-phased approach that included controlled, representative laboratory testing that evolved into on-aircraft flight tests. Each phase successfully addressed various aspects of the four critical factors: damage detection capability, durability, installation/supportability, and safety. Validation testing used CVM sensors mounted to representative specimens which were cyclically loaded to generate and grow fatigue damage. The loading spectrum used for fatigue propagation was based on the anticipated on-aircraft load environment. These tests demonstrated the capability of the CVM system to detect and reliably identify relevant damage on the application structures in a representative environment. The validation process considered the numerous factors that affect the reliability of an inspection methodology including the individual inspector/operator, the equipment, the procedures and the environment in which the inspector is working. It also evaluated the viability of the SHM approach within an airline's maintenance program.

The validation plan was developed to properly: 1) provide a vehicle in which skills, automation of instrumentation and human error can be evaluated in an objective and quantitative manner, 2) produce a comprehensive, quantitative performance assessment of the SHM system and utilization procedure in a systematic manner, 3) provide an independent comparison between SHM solutions and alternate maintenance and monitoring methodologies, 4) optimize SHM utilization methodologies through a systematic evaluation of results obtained in laboratory and field test beds, 5) produce the necessary teaming between the airlines, aircraft manufacturers, regulators, and related SHM developers.

Probability of Detection Using CVM SHM System

Initial CVM performance validation testing, summarized in this report, provided Boeing Commercial Aircraft with sufficient data to place CVM sensor technology into their Nondestructive Testing Standard Practices Manual. The testing establish the ability of CVM sensors to detect cracks in fuselage skin structure and to determine the limits on skin thickness applications such that a crack of 0.10" length could be reliably detected. The allowable skin thickness where CVM sensors achieved the necessary $POD_{(90/95)}$ level of 0.1" was determined for both 2024-T3 and 7075-T6 materials. In the 2005-2006 time frame, Boeing's NDT Standard Practices Manual was revised to include CVM sensors as a possible structural monitoring option.

The follow-on effort validated the application of CVM monitoring solutions to a particular aircraft application: crack detection in the Wing Box fitting of Boeing 737 aircraft. The test specimens were wing box fittings from a Boeing 737, some pristine and some that were removed from operating aircraft. All crack detections are for the most conservative unloaded state, fasteners loose and the entire part sealed with Fuel Vapor Barrier (crack, sensor & fitting). CVM Crack detection lengths produced POD levels that were lower than the required crack detection level. Thus, the CVM sensors were deemed as good, or better than, the current inspection requirement. There were no False Calls (CVM sensor indicated the presence of a crack when actually none was present) associated with these tests.

CVM Sensor Durability

In addition to the crack detection performance data, this program also conducted tests to evaluate the long-term environmental durability of the CVM system. All operating conditions that may affect SHM system response were included in the test program. Data was acquired to show that the sensors maintained consistent and proper dCVM and continuity readings before and after representative, and in some cases extreme, flight conditions. In addition, after the application of the Corrosion Inhibiting Compounds (CIC) to the cracked specimens, it was observed that no CIC was drawn into CVM galleries. Related to this, the galleries did not experience any blockage during the CIC testing. Since the variation in POD levels for with and without CIC present was within experimental deviations, the conclusion is that there is no appreciable difference in CVM crack detection performance (POD) with or without the presence of CIC. Thus, it was observed that CIC did not affect normal CVM operation. Other tests were conducted to determine if a CIC could possibly wick under a CVM sensor or otherwise affect the adhesive layer between the sensor and the surface to which it is applied. Microscopic inspections confirmed that no wicking took place under the sensor FEP material. No discernible degradation was observed visually under the FEP or exposed during bond pull testing. All specimens had acceptable sensor bond strength indicating that the CIC did not adversely affect the sensor installation.

The CVM sensor includes a fail-safe feature which is critical to the application of SHM systems. This prevents the unknowing acquisition of faulty data that might result in a missed detection of damaged structure. With a fail-safe feature ensured, the durability tests primarily evaluated the nuisance factor that might be inflicted on an airline that uses such SHM systems. It is an undesirable scenario for airlines to revisit SHM sensor network installation sites in order to address, and possibly replace, failed sensors. Thus, durability of SHM systems is an important consideration related to the value and long-term use of SHM solutions over the life of an aircraft. With respect to reliability and repeatability of the sensor operation, it was noted that, in over 200 fatigue tests conducted using CVM sensors on various aircraft structures, there were no false calls produced by the sensors.

CVM Flight Tests

Data from two different flight test programs were compiled to use as proof of successful sensor function in an actual aircraft operating environment. The first flight test series placed CVM sensors in regions that were not expected to experience any cracking. For this reason, the flight tests were considered CVM installations in “decal mode” (i.e. no damage). The purpose of this initial test series was to explore general installation, operation and monitoring of CVM sensors by airline personnel while also assessing the durability of the sensors when exposed to real flight conditions. Different sensor designs were installed in various aircraft regions without any particular application in mind. This initial flight test series demonstrated the ability of CVM sensors to: 1) operate successfully on operating aircraft over long periods of time, 2) produce consistent data and 3) be properly installed and monitored by airline personnel.

The second flight test series was conducted in association with the 737 wing box fitting program. CVM sensors, designed to monitor the actual wing box fitting, were installed on the set of ten wing box fittings, on seven different 737 aircraft that were operating in the Delta Air Lines fleet. A set of 68 sensors were mounted on wing box fittings in seven different B-737 aircraft in the Delta Air Lines fleet. The sensors were monitored every 90 days for over four years, producing over 2,400 sensor response data points. These flight tests demonstrated the successful, long-term operation of the CVM sensors in actual operating environments. Other important topics that were studied and validated during the flight tests were: 1) SHM indoctrination and training for Delta personnel, 2) complete formal modifications to integrate SHM into airline maintenance programs, and 3) assessment of an aircraft maintenance depots' ability to safely adopt SHM and the FAA support needed to ensure airworthiness.

Overall, both flight tests allowed for the accumulation of over 1.5 million successful flight hours of CVM operation. Data from the monitored sensors showed that, in all cases, the continuity numbers maintained the desired high levels while the dCVM levels remained in the low numbers associated with no crack detection. In addition to the general high continuity (no flow blockage) and low dCVM (no cracks present) readings, the data was observed to be repeatable and consistent during the monitoring period. Results from the flight test series demonstrated the ability of CVM sensors to: 1) operate successfully on operating aircraft over long periods of time, 2) produce consistent data and 3) be properly installed and monitored by airline personnel. Flight testing is a key element, in combination with controlled laboratory tests, in establishing proper inspections (or maintenance actions) needed to safely detect damage onset during the operational life of the airplane.

Adoption of CVM by Delta Air Lines and SHM Path Forward

The maintenance program instituted by each air carrier must be modified to accommodate the unique operation and use of SHM systems. This program resulted in modifications to Delta's maintenance program to: 1) produce hardware specifications, installation procedures, operation processes, continued airworthiness instructions, 2) complete SHM indoctrination and training for Delta personnel, 3) complete the financial, technical and logistical internal signatures necessary to adopt CVM, and 4) determine their ability to adopt SHM and the FAA support needed to ensure airworthiness.

As a result of the CVM Wing Box Fitting program and the compiled results from completed lab/flight testing, CVM was added to the Boeing Nondestructive Testing (NDT) Manual for the 737 aircraft platform. Also as a result of the CVM Wing Box Fitting program, Boeing Service Bulletin 737-57-1309, was changed to include CVM technology as an alternate inspection method to the previously-specified visual and eddy current inspections. Embedded within this SB revision, was a series of approvals from the appropriate set of Designated Engineering Representatives (DER) or, as they are also known, Airplane Representatives (AR).

The activities conducted in this program facilitated the evolution of an SHM certification process including the development of regulatory guidelines and advisory materials for the implementation of SHM systems via reliable certification programs [9.8]. It successfully demonstrated an OEM-airline-regulator approach for safely adopting SHM solutions. The formal SHM validation approach developed in this program will allow the aviation industry to confidently make informed decisions about the proper utilization of SHM solutions.

Long-term SHM applications promise to include flight monitoring tasks which lead to prognostic health monitoring. Flight loads, mechanical functions, and service problems can all be identified and algorithms can be applied to anticipate maintenance needs. Similarly, SHM sensors can be used to predict structural integrity problems or track trends in specific regions, mechanical systems, electrical systems, or pressure systems such that condition-based maintenance can be used. Thus, SHM systems could further improve maintenance programs by allowing for streamlined, advanced planning based on a more complete picture of an aircraft's structural integrity and operational performance.

Conclusions

The effect of structural aging and the dangerous combination of fatigue and corrosion has produced a greater emphasis on the application of sophisticated health monitoring systems. In addition, the costs associated with the increasing maintenance and surveillance needs of aging structures are rising. Corrective repairs initiated by early detection of structural damage are more cost effective since they reduce the need for subsequent major repairs and may avert a structural failure. Aerospace structures have one of the highest payoffs for SHM applications since damage can quickly lead to expensive repairs and aircraft routinely undergo regular, costly inspections.

An overall summary of the performance assessment and validation of CVM technology is:

- CVM sensors detect cracks in the component they are adhered to
- Inspection process and diagnosis is fully automated and can be conducted remotely
- Early detection = less costly repairs
- CVM systems are fail-safe (inert sensors produce an alarm)
- Multiple sets of lab performance and multi-year flight test programs have been completed
- CVM technology has been formally adopted via its integration into OEM NDT Standard Practices Manuals and approved for routine use through a modification to an aircraft Service Bulletin.
- Certification and regulatory framework has been established to safely and comprehensively validate SHM

Global SHM, achieved through the use of sensor networks, can be used to assess overall performance (or deviations from optimum performance) of large structures such as aircraft, bridges, pipelines, large vehicles, and buildings. The ease of monitoring an entire network of distributed sensors means that structural health assessments can occur more often, allowing operators to be even more vigilant with respect to flaw onset.

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