

Design Framework Validation for a Hot Spot on Complex Aircraft Structures

J.-B. IHN, M. S. LEONARD, S. E. OLSON AND M. P. DESIMIO

ABSTRACT

Aircraft structural components may have known “hot spots” where any initial damage is anticipated to occur or has consistently been observed in the field. Automated inspection of these areas, or hot spot monitoring, may offer significant time and cost savings for aircraft maintainers, particularly when the hot spots exist in areas that are difficult to access or where traditional NDE inspection methods will not work. This paper discusses the development of hot spot monitoring techniques for a metallic lug component using piezo-generated elastic waves. The development process utilizes the recently created SHM system design framework and uses a multi-step approach progressing from simple coupon tests to the full scale component for system validation. Initial testing has been performed on titanium dogbone coupons. This testing has demonstrated the potential to detect relatively small cracks. However, actual crack detection has been complicated by issues of sensor system robustness and the reliability of “truth” data. Subsequent testing has been performed using titanium cantilever beam specimens. Sensor robustness and the reliability of “truth” data have been improved, but additional testing is required to further refine the techniques as only limited data is available from the beam testing. Recent experiments include fatigue testing of lug subcomponents with geometry and material properties very similar to the full scale component. Preliminary work demonstrates that damage indices can be mapped to crack length, although further studies are needed to combine the readings of all the piezoelectric sensors into a single crack length estimate. Building on the results from all of the earlier testing, SHM system development is underway for a full scale lug component to be fatigue tested under spectrum loading. Modeling, experimentation, and signal analysis performed at various steps of the development are discussed.

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14. ABSTRACT Aircraft structural components may have known hot spots where any initial damage is anticipated to occur or has consistently been observed in the field. Automated inspection of these areas, or hot spot monitoring, may offer significant time and cost savings for aircraft maintainers, particularly when the hot spots exist in areas that are difficult to access or where traditional NDE inspection methods will not work. This paper discusses the development of hot spot monitoring techniques for a metallic lug component using piezo-generated elastic waves. The development process utilizes the recently created SHM system design framework and uses a multi-step approach progressing from simple coupon tests to the full scale component for system validation. Initial testing has been performed on titanium dogbone coupons. This testing has demonstrated the potential to detect relatively small cracks. However, actual crack detection has been complicated by issues of sensor system robustness and the reliability of truth data. Subsequent testing has been performed using titanium cantilever beam specimens. Sensor robustness and the reliability of truth data have been improved, but additional testing is required to further refine the techniques as only limited data is available from the beam testing. Recent experiments include fatigue testing of lug subcomponents with geometry and material properties very similar to the full scale component. Preliminary work demonstrates that damage indices can be mapped to crack length, although further studies are needed to combine the readings of all the piezoelectric sensors into a single crack length estimate. Building on the results from all of the earlier testing, SHM system development is underway for a full scale lug component to be fatigue tested under spectrum loading. Modeling, experimentation, and signal analysis performed at various steps of the development are discussed.		

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INTRODUCTION

Boeing and the Air Force Research Laboratory (AFRL) are developing a framework that enables efficient and defensible design of a SHM system for a structural hot spot. A defensible design is a design in which each step is supported by decisions based on the requirements for the system. Aircraft structural components may have known “hot spots” where a particular type of damage is anticipated to occur or has consistently been observed in the field. By exercising the newly developed framework for a specific aircraft application, Boeing and researchers under an Air Force program are working to design a SHM system to automatically monitor such hot spots to detect, assess, and act on the presence of damage. The development process has followed a multi-step approach progressing from simple coupon tests to the full scale component. One advantage of this approach is that it provides opportunities to continuously reevaluate and enhance the framework process based on the lessons learned from each step. The test articles include dogbone coupons, cantilever beams, lug subcomponents, and a full-scale lug component. All of these studies use surface-bonded piezoelectric disks for elastic wave excitation and sensing. The SHM system design framework, along with modeling, experimentation, and signal analysis performed at various steps of the development, are discussed in the following sections.

DESIGN FRAMEWORK

The latest version of the SHM system design framework for a structural hot spot is shown in Figure 1. The framework continues to be updated and matured as lessons learned from earlier implementations are incorporated. Two major additions from the last reported framework are: 1) the design framework is applicable not only for existing structures, but also for new structures where the SHM system is considered as a key design element during any design optimization processes; and, 2) virtual SHM system design and damage detection processes have been incorporated which enables the optimization of SHM system design parameters and performance in virtual domains prior to actual system construction. The framework, which includes understanding system level requirements, has been exercised for a hot spot application on a metallic lug component. Based on the system level requirements, wave propagation methods using piezoelectric materials have been selected as the best design approach. The following sections discuss modeling, experimentation, and signal analysis performed at various steps in the multi-step development approach.

CANTILEVER BEAMS

Results from earlier dogbone specimen testing [1] demonstrated the potential use of piezo-generated elastic waves to detect damage. To build on these results, additional studies have been performed using six cantilever beam specimens. The cantilever beams have been fabricated by Boeing using the same titanium alloy used for the dogbone coupon testing.

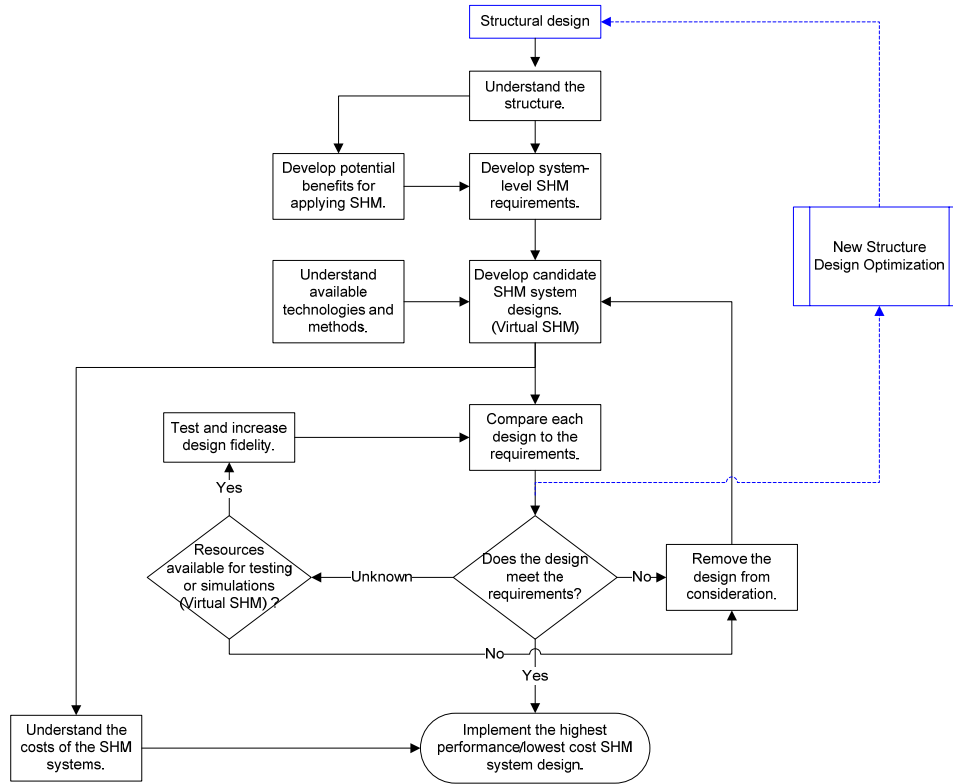


Figure 1. The flow of the SHM system design framework for a structural hot spot

In order to determine the optimal transducer and receiver configurations for detecting required damage on a cantilever specimen, three-dimensional finite element simulations of healthy and cracked cantilever beams with various transducer configurations (including different actuator and sensor shapes, sizes, and locations) have been performed. Figure 2 shows displacement contour results near the root of a healthy beam and in a beam with a 0.120 inch wide by 0.080 inch deep corner crack for excitation signals at 400 kHz. Crack detection is based on the difference formed by subtracting the baseline healthy response from the damaged response. As seen in the figure, the crack reflections are readily detectable. A similar process is used for the experimental data, where scatter signals are formed by subtracting the baseline healthy signal from the current measurement.

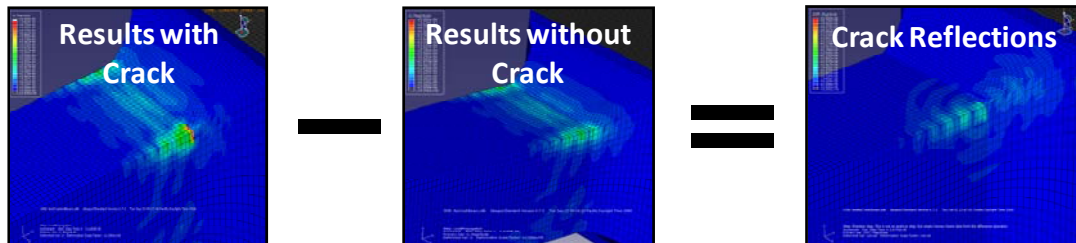


Figure 2. Displacement contour plots of healthy and damaged beams and procedure used to detect cracks by subtracting the baseline healthy response from the damage response

Figure 3a shows the basic geometry of the cantilever beams. The larger base of the beam is rigidly mounted and, as shown in Figure 3b, the smaller end of the beam is driven in the vertical direction using an MTS machine. As shown in the figures, each beam is instrumented with two piezoelectric transducer packages, one on the top surface and one on the bottom surface of the specimens. These sensor packages consist of a relatively large rectangular actuator and six sensing disks. Windowed sine burst excitation signals are sent to the actuator to generate elastic waves in the beam. The sensing disks capture energy that is reflected from cracks expected to grow near the root of the beams.

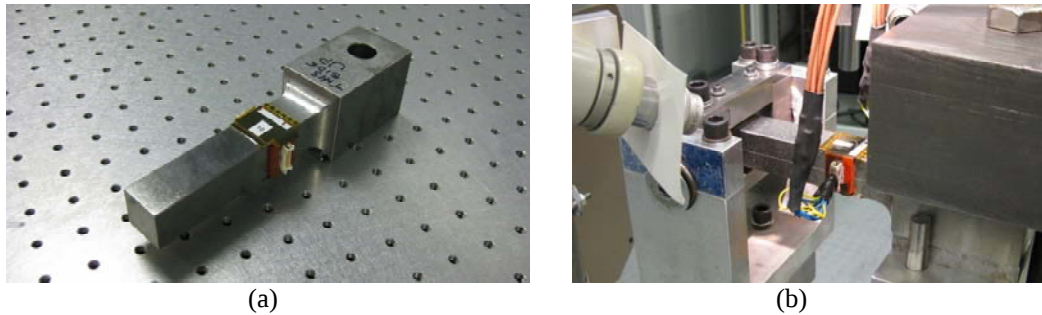


Figure 3. (a) Basic geometry of instrumented cantilever beam test article and (b) cantilever beam test article mounted in test fixture

Experimental fatigue testing has been performed at Boeing and AFRL facilities. Testing proceeds until the beam develops a crack across approximately half the width of the specimen. Boeing has utilized aircraft-specific spectrum loading for three specimen tests, whereas testing at AFRL has utilized simple fatigue loading for the remaining three specimens. The first AFRL beam test has been performed using a peak load of 4,400 lbf and a stress ratio, R , of -0.30. However, this beam developed a full-width crack prior to significant piezoelectric data being collected. The peak load has been adjusted in subsequent tests at AFRL such that the beam fatigues in a reasonable amount of time, yet sufficient crack growth data can be collected. The elastic wave excitation and response signals are recorded using an Acellent ScanWizard system. The “truth” data collection utilizes visual crack length estimation, as shown in Figure 4a, as well as fluorescent dye penetrant inspections, as shown in Figure 4b, for more accurate crack length determination. To assure the sensors are functioning correctly, impedance and capacitance measurements are made for each sensor at various times during the tests.

Similar to the dogbone specimen, the issues of sensor system robustness and the reliability of the “truth” data remain. However, in both cases the reasons are different than those for the dogbone specimens. The sensor system robustness issue relates to problems with the connectors that are rigidly attached to the specimen. As a result, the connectors are exposed to some loading and degradation in the response signals is observed. Regarding the issue of reliable “truth” data, the fluorescent dye penetrant provides a reasonable approximation of the surface crack lengths. However, as seen in Figure 4b, multiple cracks occur and originate on both sides of the beam. The presence of multiple cracks makes it difficult to quantify the various damage states.

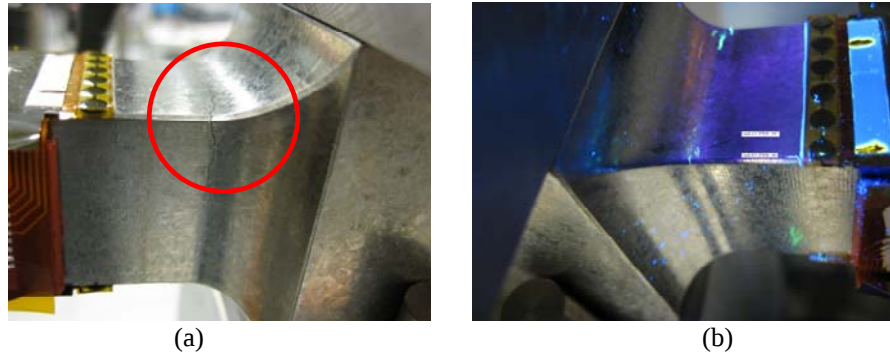


Figure 4. (a) Visible crack on cantilever beam specimen and (b) example of fluorescent dye penetrant inspections used for improved crack length estimations

Regardless of the data analysis issues, the experimental data still provide a good indication of the presence of damage. Boeing has led the development of general methods for calibration and crack length estimation using the signal data. Using collected measurements and “truth” data from the three cantilever beam tests at Boeing and the first beam test at AFRL, several damage indices have been computed. The damage index calculations are based on differential measurements between consecutive data sets and utilize a field of view defined for each sensor on the beam. The damage indices are converted to estimated crack lengths based on calibration data from one test article. Figure 5 shows that the damage index calculations accurately predict crack lengths in the second beam tested at AFRL.

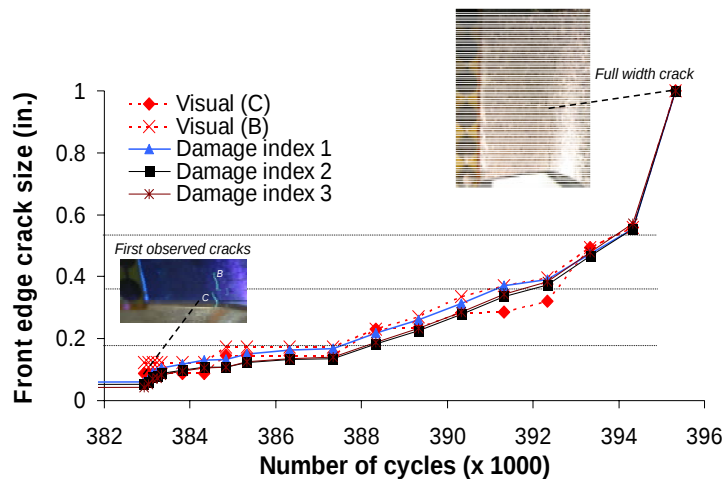


Figure 5. Crack length estimations in second beam tested at AFRL

LUG SUBCOMPONENTS

Results from the cantilever beam testing show the potential to detect crack damage, as well as to estimate the current crack length. However, the geometry of the cantilever beams is much simpler than the actual lug component. Therefore, the next step in the SHM development uses a titanium alloy lug subcomponent which

has geometry very similar to that of the actual full-scale lug component. Figure 6 shows the basic geometry of the lug subcomponents. As shown in Figure 6a, each lug subcomponent is instrumented with four piezoelectric transducer packages, one on the lug portion (Layer A), one on the wall of the structure (Layer B), and two on the sides of the subcomponent (Layers C and D). The sensor packages on the lug and wall of the structure include a relatively large actuator and various smaller disks. Windowed sine burst excitation signals are sent to the actuator to generate elastic waves in the beam. The sensing disks capture energy that is reflected from cracks expected to grow near the root of the lugs. Data has also been collected from the transducer packages on the sides of the subcomponent, but is anticipated to provide limited information and has not been investigated in detail.

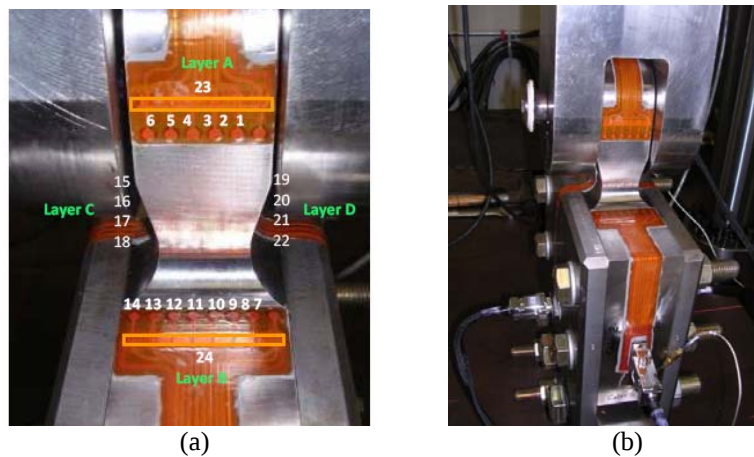


Figure 6. (a) Basic geometry of instrumented lug subcomponent test article and (b) lug subcomponent mounted in test fixture

Three-dimensional finite element simulations of healthy and cracked lug subcomponents have been performed. The entire subcomponent is not modeled due to processing time constraints. Model detail, waveform fidelity, and processing time tradeoffs are considered, leading to selection of model parameters to support simulations of excitation and response signals. The model has been run to simulate the healthy condition and for corner cracks up to 0.50 inch long. A windowed sine burst excitation signal at 400 kHz is applied on the lug and responses at various sensors on the wall of the subcomponent are simulated. The model has been used to aid in sensor size and placement. A damage index has been created based on scatter signals formed as the difference between the healthy and damaged responses. The simulated damage indices from sensors on the same edge as the crack provide a good indication of damage, particularly for larger cracks. A similar process is used for the experimental data analysis, where scatter signals are formed from consecutive data sets and utilize a field of view defined for each sensor on the lug subcomponent.

Experimental testing has been performed at Metcut Research, a commercial mechanical test facility. After instrumentation, each subcomponent is mounted in an MTS machine and cycled with aircraft-specific spectrum loading. After each half spectrum, the cycling is halted and elastic wave measurements are taken.

Excitation signals are applied at the actuators, Sensors 23 and 24 in Figure 6, and the responses are recorded at Sensors 1 through 14 using an Acellent ScanWizard system. Excitation frequencies range from 200 to 500 kHz in 50 kHz increments. The experimental data are stored on a laptop computer over the duration of the tests. Four of the five test articles have grown cracks. The “truth” data is based on visual crack length measurements.

The same calibration and crack estimation methods used for the cantilever beams have been applied to the lug subcomponent data. Using collected measurements and “truth” data from the lug subcomponent testing, as well as the results from the cantilever beam testing, several damage indices have been computed. The damage index calculations are based on differential measurements between consecutive data sets and utilize a field of view defined for each sensor on the lug subcomponent. In order to minimize potential false positives from the current crack estimation method, a new algorithm has been developed for detecting crack initiation that would be followed by subsequent crack growth algorithms should any cracks be detected. The preliminary results of the crack initiation and subsequent crack growth estimates are shown in Figure 7a and 7b, respectively.

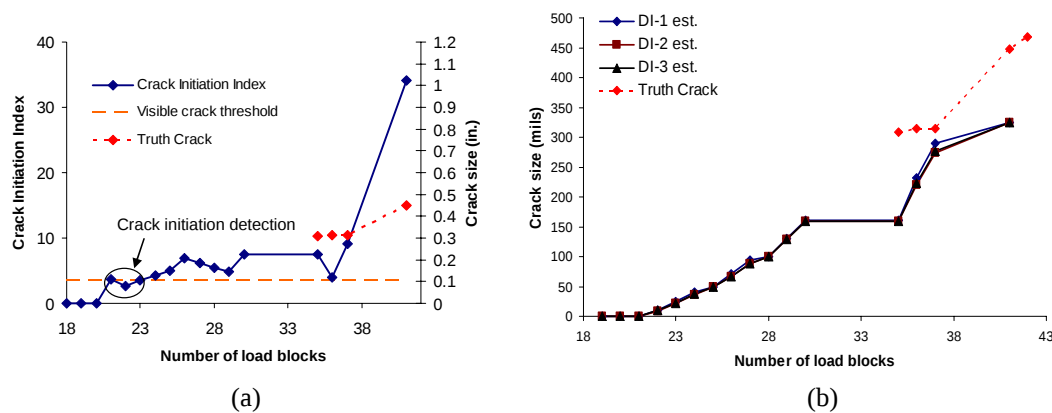


Figure 7. Preliminary results from (a) crack initiation detection algorithm and (b) subsequent crack growth estimation algorithm

The lug subcomponent testing further highlights the critical issue of sensor system robustness. At specific times during the subcomponent testing, sensors show instances of near zero energy levels which are believed to have been caused by poor coupling of the signal connector to the sensor package. This data must be discarded from further analysis. In other cases, sensors show increased signal energy which is due to actuator cables being switched and connected to the wrong sensor packages. The switch significantly decreases the distance between certain actuators and sensors. For the case of misconnected cables, it is often possible to rearrange the recorded waveforms and salvage the measurements.

FULL-SCALE LUG COMPONENTS

Experimental testing of a full-scale lug component is planned for the near future. Analytical modeling is currently underway to investigate wave propagation and aid in sensor placement. The same calibration method and the final algorithm developed from earlier steps will be applied to detect cracks and track crack propagation during the testing. Considerable attention will be given to sensor system robustness, which has complicated all of the previous steps of the SHM system development.

CONCLUSIONS

Hot spot monitoring may offer significant time and cost savings for aircraft maintainers, particularly when the hot spots exist in areas that are difficult to access or where traditional NDE inspection methods will not work. The development process of a hot spot monitoring system has utilized the recently developed framework and followed a multi-step approach progressing from simple coupon tests to the full-scale component. These studies have focused on the development of hot spot monitoring techniques for a metallic lug component using piezo-generated elastic waves. Additional studies are required, but the potential to both detect cracks and track crack growth has been demonstrated. However, sensor system robustness remains a critical issue and must be addressed. The upcoming full-scale lug component testing provides a good opportunity to demonstrate elastic wave crack detection techniques on an actual aircraft component.

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