Efficient Airframe Management Using In-Situ Structural Health Monitoring

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ABSTRACT

The United States Air Force utilizes the Aircraft Structural Integrity Program (ASIP) to service and maintain its airframes. This schedule-based maintenance approach works well for ensuring system integrity; however, it is very costly, labor-intensive, and it reduces system availability. As a result, the Air Force intends to transition to a process that services aircraft based on their actual condition instead of the presumptive schedule-based approach. Structural health monitoring (SHM) technologies are being investigated to enable such real-time state awareness and decision-making. This paper provides a brief review of ASIP and the required inspections to investigate structural fatigue. The current ASIP process is demonstrated on a representative aircraft component which is fatigue loaded in the laboratory. A SHM system has been developed to estimate fatigue crack lengths in the representative component. The potential benefits of integrating advanced SHM techniques into the ASIP framework are highlighted.

INTRODUCTION

The Department of Defense (DoD) supports investigations across a variety of methodologies aimed at reducing operational cost, increasing availability, and maintaining safety of current and future weapon systems. In fact, operations and support (O&S) of DoD weapon systems accounts for 65-80% of the total lifecycle cost [1]. One of the principal contributors to O&S cost is the vehicle maintenance process. The current procedure, the Aircraft Structural Integrity Program (ASIP), requires vehicles to be removed from service for routine maintenance, including inspection, at predetermined times regardless of their actual conditions. This

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schedule-based maintenance approach works well for ensuring system integrity; however, it is very costly, labor-intensive, and it reduces system availability. The costs are also continuously rising due to the frequent inspections required to maintain safety in DoD aircraft which have an average age of roughly 24 years.

In December 2007, the DoD implemented a policy called “Condition-Based Maintenance Plus” (CBM+) intending to decrease the maintenance burden and increase aircraft availability. As the name indicates, the DoD intends to move toward a process that services weapon systems based on their actual condition instead of the presumptive schedule-based approach. As a result of this policy, an increased emphasis has been placed on the development of advanced health management technologies (for engines, structures, flight controls, etc.) within government agencies, industry, and academia over the past five years. The following section describes the current ASIP process used to insure safety of United States Air Force (USAF) airframes. Next, an experiment demonstrating the application of the ASIP process is presented. Finally, SHM technology is discussed and the potential benefits of integrating advanced SHM techniques into the ASIP framework are highlighted.

AIRCRAFT STRUCTURAL INTEGRITY PROGRAM

In 1958, the USAF established ASIP as a means for servicing and maintaining its airframes. ASIP’s goal is ensuring the desired level of structural safety, durability, and supportability with the least possible economic burden throughout the aircraft design service life. USAF aircraft structures are currently designed using a “damage tolerant” approach, where structures are designed to retain the required residual strength for a period of unrepaired usage after the structure has sustained specific levels of fatigue, corrosion, accidental, and/or discrete source damage. ASIP focuses on establishing predefined maintenance intervals (schedule-based maintenance) for inspecting for such damage and servicing the airframe. One of the key damage sources that ASIP must control is structural fatigue, which is discussed next.

Aircraft Structural Fatigue

Structural fatigue is a primary damage mechanism that ASIP must manage, since it can result in crack initiation and growth. When structural fatigue is a concern, crack growth predictions can be made using commercially available fracture mechanics software. These fracture mechanics tools predict crack growth chiefly based on the principles of Paris Law, which establishes a relationship between stress intensity factor range (ΔK) and sub-critical crack growth. Fatigue crack growth curves are typically divided into three regions. Region I is the fatigue threshold region, where the ΔK is too small to generate crack propagation. In Region II, the logarithm of the crack growth rate exhibits a linear relationship with the logarithm of ΔK, and reasonable predictions of crack growth rates can be made using Paris Law. Finally, in Region III, small changes in ΔK produce large crack growth rates, causing structural components to reach failure/fracture rapidly.

At present, ASIP manages fatigue cracking by assuming structural defects exist within the airframe upon entering the fleet. It is assumed that crack growth rates from these defects will increase steadily (Region II) during operation depending on the loading conditions. Additionally, the inspection intervals are established based on the predicted amount of crack growth required to reach the critical size between inspections. The critical crack size is defined to be the size of damage for which the crack rate will increase unpredictably (Region III). This approach relies on the ability to define and inspect a specific flaw size at the beginning of each inspection interval. ASIP estimates the current state of a structure (i.e. crack size and location) for an individual aircraft from the assumed initial defect size, a representative design load, and an estimate of crack growth during a given period. Manual inspections performed by a certified operator at each inspection interval verify the actual state of the structure.

Structural Inspections

Inspection intervals for maintenance cycles are determined by the detection capabilities of the non-destructive inspection (NDI) techniques and the anticipated crack growth. The ASIP process requires at least two inspections be conducted before the assumed flaw size reaches the predicted critical crack length, at which point the component would fail catastrophically. The first inspection is typically scheduled at half the time required for the predicted crack to reach critical length, with the second inspection scheduled to occur as the crack approaches the estimated critical crack size. This process allows inspectors two opportunities to detect/locate structural flaws in the airframe prior to it reaching the point of fast fracture. A block diagram of the ASIP inspection process is shown in Figure 1. In most cases, inspections performed during this process do not find any damage, and the airframe inspection cycle iterates using another initial flaw size assumption. Conversely, when damage is actually confirmed, the airframe is repaired before returning to service, and then re-enters the inspection cycle still assuming an initial defect. The next section presents a laboratory test illustrating application of the ASIP inspection process.

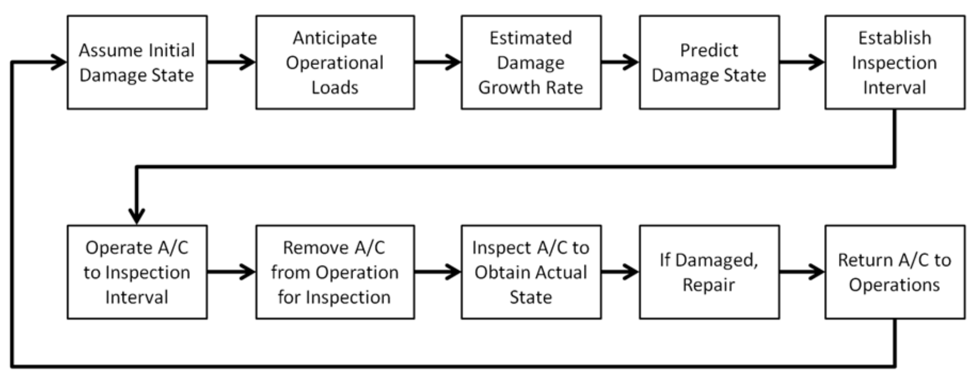
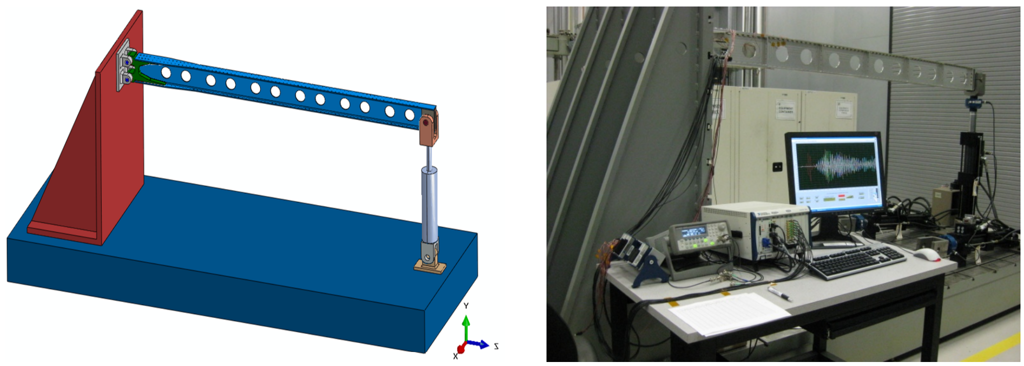


Figure 1. Block diagram of the ASIP inspection process.

LABORATORY EXPERIMENT

Laboratory testing was performed using a representative single wing spar assembly made of 6061-T6511 extruded aluminum that is subjected to flight-like fatigue loading. Although 2024 and 7075 are the most common alloys used in aircraft, 6061 was selected for this experiment because it is less expensive and readily available. One end of the spar was mounted to a test fixture representing the wing attachment to the fuselage. The opposite end of the spar was loaded in fatigue using a hydraulic actuator to emulate wing deflection during flight. This experiment used a constant cyclic load of 1,000 lbf (4.45 kN) and a minimum load of zero. Figure 2 shows a schematic and photograph of the test configuration. From a finite element analysis (FEA) performed on the test article, it was determined that the wing spar attachment clevis was the fatigue limiting element. Under cycling loading, corner cracks were predicted to initiate at the shoulders of the clevis and grow horizontally and vertically.



(a) (b)

Figure 2. (a) Schematic and (b) photograph of laboratory test configuration.

The commercially available AFGROW (Air Force Growth) fracture mechanics software was used to provide an estimate of crack initiation and growth. The loading profiles were assumed to be sinusoidal with constant peak load amplitude and a minimum load of zero (i.e., R=0 loading). The critical crack lengths in the A-direction (horizontal) and C-direction (vertical) were found to be 0.350 in (8.89 mm) and 0.700 in (17.78 mm), respectively. The predictions started from a crack size of 0.020 inches; however, if an initial crack or flaw is not present in the critical region, additional cycles are needed to initiate a crack. The number of cycles needed to initiate a crack can be approximated from fatigue testing performed on un-notched 6061-T6 specimens [2]. The estimated time for a 0.020 in (0.51 mm) crack to initiate was 10,000 cycles.

The ASIP process assumes that all critical airframe components have an initial flaw size to account for any damages that could have occurred during the manufacturing and maintenance processes. For this experiment, a 0.050 in (1.27 mm) flaw was assumed to exist in the clevis element. Using AFGROW with the loading profiles utilized for this testing, the estimated fatigue life (i.e. the time required for an initial crack of 0.050 in (1.27 mm) to grow to the critical crack length) for the wing spar assembly was approximately 8,700 cycles. Recall that the ASIP process establishes inspection intervals by performing the first manual inspection at half the estimated fatigue life, and the next inspection at the estimated fatigue life. Therefore, ASIP inspections for this test were conducted at or before every 4,350 cycles using penetrant to detect and measure cracks.

Experimental Results

During testing, cracks initiated from both shoulders of the clevis and propagated in both the A- and C-directions as expected. The first noticeable cracks were at

43,000 cycles in the C-direction, with lengths of 0.091 in (2.31 mm) and 0.080 in (2.03 mm) on the left and right shoulders, respectively. Cracks in the A-direction were not detected until 47,500 cycles with sizes of 0.058 in (1.47 mm) and 0.048 in (1.22 mm). As noted above, the estimated time for a 0.020 in (0.51 mm) crack to initiate was 10,000 cycles. Because the inspection technique used for this experiment could only detect flaws above 0.050 in (1.27 mm), the 0.020 in (0.51 mm) crack initiation assumption could not be verified. However, it is still interesting to compare the estimated and measured cycles required for crack initiation as shown in Table 1. The measured initiation cycles are those observed during the testing with the recorded crack length listed below. The estimated initiation cycles are based on the 10,000 cycles required for a 0.020 in (0.51 mm) crack to initiate plus additional cycles which are required to grow the 0.020 in (0.51 mm) crack to the recorded crack length. As shown in the table, the error in predicting crack initiation cycles was between 132% and 197% for all of the cracks. Table 1 also lists the estimated and measured cycles for cracks to grow from the initial crack size measured to the critical crack length. The crack propagation errors range by almost a factor of four, from 109% to 391%. These wide ranges are typical of crack propagation behavior, as it is not uncommon for fatigue crack growth predictions to vary by a factor of five.

Table 1. Error in crack initiation and growth predictions.

Crack Direction and Side	Cycles for Crack Initiation			Cycles for Crack Growth to Critical		
	Estimated	Measured in (mm)	Percent Error (%)	Estimated	Measured in (mm)	Percent Error (%)
A-dir left side	17,000	47,500 0.058 (1.47)	179	7,500	20,250 0.360 (9.14)	170
A-dir right side	16,000	47,500 0.048 (1.22)	197	8,500	17,750 0.360 (9.14)	109
C-dir left side	18,500	43,000 0.091 (2.31)	132	5,500	27,000 0.520 (13.21)	391
C-dir right side	18,000	43,000 0.080 (2.03)	139	6,500	27,000 0.680 (17.27)	315

Since the wing spar is a fracture critical component, ASIP would require periodic inspections to ensure fatigue cracks do not initiate and grow beyond the critical crack length before being repaired. Using the ASIP established inspection interval of every 4,350 cycles for this component, the clevis would be inspected approximately ten (43,000/4,350) times before any damage is detected for cracks in the C-direction and eleven (47,500/4,350) times for cracks in the A-direction. These inspections where no damage is detected are significant since the cost for inspecting similar components on fielded aircraft range from approximately $1,000 to $120,000 per inspection based on various factors (e.g. aircraft configuration, type of inspection, coating removal and restoration, etc.).

In summary, this experiment indicates that the current methods for estimating crack initiation and growth may contain substantial errors. Since the ASIP process requires manual inspections be performed based on these estimates, aircraft are repeatedly removed from operations to confirm structural integrity. Performing inspections at these predefined intervals can be very costly, time consuming, and labor-intensive. As a result, automated structural health assessment methods offer the potential to significantly reduce the inspection burden.

STRUCTURAL HEALTH MONITORING

SHM can be defined as automated methods for determining adverse changes in the integrity of mechanical systems. The ultimate goal of SHM is to provide an automated and real-time assessment of a structure's ability to serve its intended purpose. The need for, and benefits of, SHM systems for civil, military, and aerospace applications have been documented by many researchers [3-5]. Ideally, SHM would provide a diagnosis of the current state of a structure and a prognosis about the capability of the structure to perform its function in the future. The diagnosis should include the detection, localization, and assessment of any damage. The prognosis might be that the structure is as good as new; safe to operate for only a certain number of flight hours; or, that immediate repair is required. Many SHM experiments have been performed over the past decade, but few have transitioned to aerospace applications [6].

SHM systems are typically comprised of in-situ or embedded sensors and processing algorithms. The algorithms are used to interpret the sensors responses to discriminate between different damage types in order to provide an accurate damage assessment and corresponding prognosis. Various processing steps may be performed by the SHM system to transform the data into different forms which enhance the damage assessment ability. Most SHM systems process sensory data using pattern recognition as a means to classify structural states [7]. Development of SHM systems based on pattern recognition requires training data from all anticipated damage states and operational environments to be effective. The training data is used to design a classifier and the performance is evaluated by scoring the classification results from data not utilized during the design or training phases.

Applying SHM to ASIP

This section demonstrates using SHM crack length inspections within the ASIP framework. The representative wing spar attachment clevis was instrumented with piezoelectric actuators and sensors, as shown in Figure 3, to monitor fatigue cracks in the vertical direction. For demonstration purposes, a crack length estimation algorithm was developed using guided wave signals recorded as a crack initiated and grew in the wing spar attachment clevis.

At the beginning of the cyclic fatigue loading of the wing spar, the loading was paused every 1,000 cycles for collection of SHM signals and for visual crack length inspection. After a crack was visually detected, the loading was paused every 500 cycles for SHM signal collection and visual crack length measurements. The test was halted when the longest vertical crack length reached 0.700 in (17.78 mm).

The laboratory experiment involves only a single representative wing spar, so it was not possible to design a crack length estimation algorithm and then test the algorithm using independent data from a different component. Therefore, for this demonstration, a Monte Carlo approach was used to design and test crack length estimates from random partitions of the experimental data. The crack length

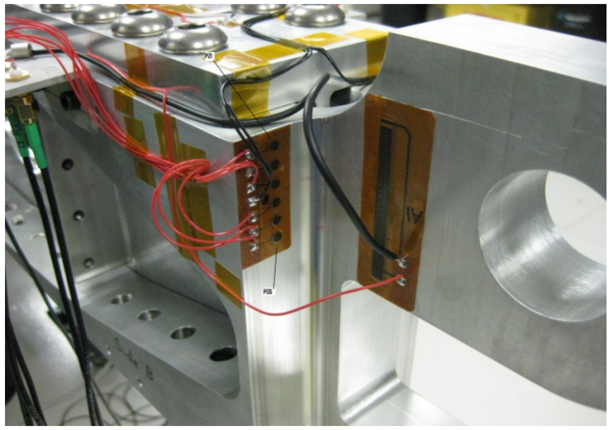


Figure 3. Piezoelectric guided wave SHM system installed on wing spar attachment clevis.

estimation algorithm is based on linear regression and is similar to the method described in [8]. In this work, 1,000 Monte Carlo trials were run, and the average crack length estimate and corresponding 95% confidence intervals were found from the Monte Carlo trials.

The upper 95% confidence interval limit is used to declare crack detection. Assume that the manual NDI inspections under ASIP are capable of detecting cracks equal to or greater than 0.100 in (2.54 mm) and the SHM system has a similar capability. A crack would be declared when the upper 95% confidence interval limit exceeds 0.100 in (2.54 mm). Figure 4a shows measured and estimated crack length, as well as the 95% confidence intervals (dashed lines), versus cycle count from 0 to 70,000 cycles. The figure shows that the upper confidence interval for estimated crack length typically has values well below 0.100 in (2.54 mm) when the clevis is uncracked. Figure 4b shows a zoomed view of the plot at higher cycles, and shows that the upper confidence interval first exceeds 0.100 in (2.54 mm) at cycle count 45,500 when the measured crack length is 0.098 in (2.49 mm).

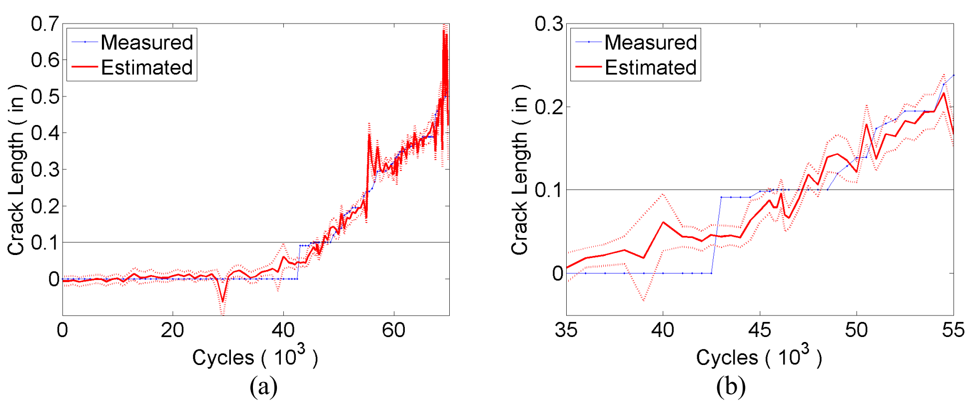


Figure 4. Measured and estimated crack lengths (a) over duration of test, and (b) zoomed.

Note that even in this laboratory demonstration problem, the SHM estimated crack length estimates are irregular. For example, at 55,000 cycles there is a step from under 0.200 in (5.08 mm) to over 0.350 in (8.89 mm). This step is followed by a decrease in estimated crack length. Crack length estimates based solely on sensor data may exhibit erratic behavior. Additional processing steps could be applied to incorporate contextual information, such as number of load cycles, material properties, and other environmental, operational, or loading factors. This additional information could lead to more regulated length estimates. However, even with these noisy estimates of crack length, the benefit of using SHM inspections to estimate crack length can be demonstrated. Under ASIP, manual inspections of the wing spar attachment clevis would be required every 4,350 cycles. If the manual NDI method was assumed to be capable of detecting cracks equal to or greater than 0.100 in (2.54 mm), 11 manual inspections would be performed before detection since the measured crack did not exceed 0.100 in (2.54 mm) until 45,700 cycles.

CONCLUSIONS

The potential benefits of integrating SHM methods into the ASIP inspection process have been demonstrated for a representative aircraft component which underwent fatigue loading in a laboratory. Under the current, schedule-based ASIP process, multiple manual inspections would be performed on the component before any damage could be confirmed. An accurate and robust SHM system could eliminate the need for the multiple manual inspections while declaring the presence of a crack near the confirmable size. Once validated, such SHM technology could permit a transition to condition-based maintenance from the current schedule-based approach, reducing the maintenance burden and increasing aircraft availability.

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