Application of the AUSAM System for Fatigue Crack Monitoring in a Wing Skin: A Case Study

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Abstract

Acousto-ultrasonics offers a promising basis for broad-field detection of structural damage in aircraft and could provide a basis for the development of condition-based maintenance approaches for airframes. When applied in situ, the technique can offer both substantial cost savings and performance improvements over conventional ultrasonic nondestructive inspection (NDI). The application of the technique involves the insertion of piezoelectric elements and attendant electrical conductors in the host structure. The Defence Science and Technology Organisation (DSTO) has worked with a local industry partner to develop the Acousto Ultrasonic Structural health monitoring Array Module (AUSAM), a compact device for the control of embedded piezoelectric transducer networks. The module, which has the footprint of a small tissue box, provides autonomous control of two send and four receive elements, and can operate synchronously with other modules to accommodate larger transducer numbers. The efficacy of the system is demonstrated through a case study involving fatigue crack monitoring in an F-111C lower wing skin test coupon. The results indicate that the system can be successfully used for crack growth monitoring, however they also highlight several issues that need to be resolved before the approach can be applied routinely in practice.

Keywords: Acousto-Ultrasonics, Structural Health Monitoring, AUSAM, F-111, Wing Skin, FASS281.28, Piezoelectric Transducer, Crack Detection

Introduction

Acousto-ultrasonics (AU) offers a promising basis for the broad-field inspection of structural damage in aircraft. When applied in situ it could offer both substantial cost savings and performance improvements over conventional ultrasonic nondestructive inspection (NDI) and for this reason is a key structural health monitoring (SHM) technology. At the Defence Science and Technology Organisation (DSTO), development of this technology is currently focused on metallic and composite airframe structures with the aim of realising substantial savings in maintenance and life cycle costs over the current management practice. The technique involves the attachment of piezoelectric elements and attendant electrical conductors to the host structure to provide the requisite acoustic actuation and sensing functions for AU interrogation. The technique has been extensively investigated under laboratory conditions [1-4]. Although the results of these investigations are encouraging, several important factors are often omitted in laboratory investigations to make the problem manageable; including geometric complexity in the coupon, exposure to operational

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loading, Electromagnetic Interference (EMI), temperature extremes, and real damage. These are clearly relevant to the majority of real applications and therefore need to be considered as part of any process to implement the technology in practice.

The transition of SHM technology from the laboratory into practice is an ambitious step involving challenges on many levels, including scientific, engineering, logistical and arguably also cultural. The high level of complexity leads inevitably to an incremental process of development in which technology demonstrations on simplified applications play an important role, as already outlined.

An important step then in any transition process is to identify an appropriate platform for the development and demonstration of a technology that balances the need to capture the key scientific and engineering issues at the core of an application or a class of applications with the need to ensure the problem is stripped of superfluous complexity and is tractable. Specimen standards play an important role in this process for conventional nondestructive testing (NDT) and should also prove a useful approach for SHM applications, however the challenges for SHM run deeper as they include factors relating to the structural integration of sensors, including for instance exposure to operational loading as already remarked.

This paper outlines work on the development of a structural health monitoring capability for adhesively bonded repairs based on AU. It ostensibly focuses on the F-111C application as it affords what is considered an ideal developmental platform and springboard for other SHM applications. The work includes the development of a packaged piezoceramic transducer and dedicated instrumentation to enable autonomous interrogation of an array of installed transducers. The system is applied to the task of detecting crack initiation and growth in a structurally detailed test coupon representative of the Forward Auxiliary Spar Station (FASS) 281.28 structure, under representative F-111C aircraft spectrum loading.

AUSAM System

As mentioned previously, the AU technique involves the generation and reception of acoustic waves using permanently fixed piezoelectric elements. Control of the elements are affected through a device called the AUSAM (Acousto Ultrasonic Structural health monitoring Array Module), shown in Figure 1. The module, which has the footprint of a small tissue box, provides autonomous control of two send and four receive elements. However, this can be easily extended to two send and eight receive elements through the use of a multiplexer (MUX). The AUSAM system can also operate synchronously with other modules to accommodate larger transducer numbers. Communication with the controller is achieved through a USB link to a notebook computer, which normally provides sufficient power to operate the device; however external power is needed when the device is operated under a high duty-cycle. The MUX also requires external power for operation. The AUSAM can drive a typical piezoelectric capacitive load of 1 to 10 nF at near 100 V to frequencies beyond 1 MHz. The system basically emulates a signal generator, signal conditioner/amplifier and data acquisition, but obviously with a much smaller footprint (Figure 1).

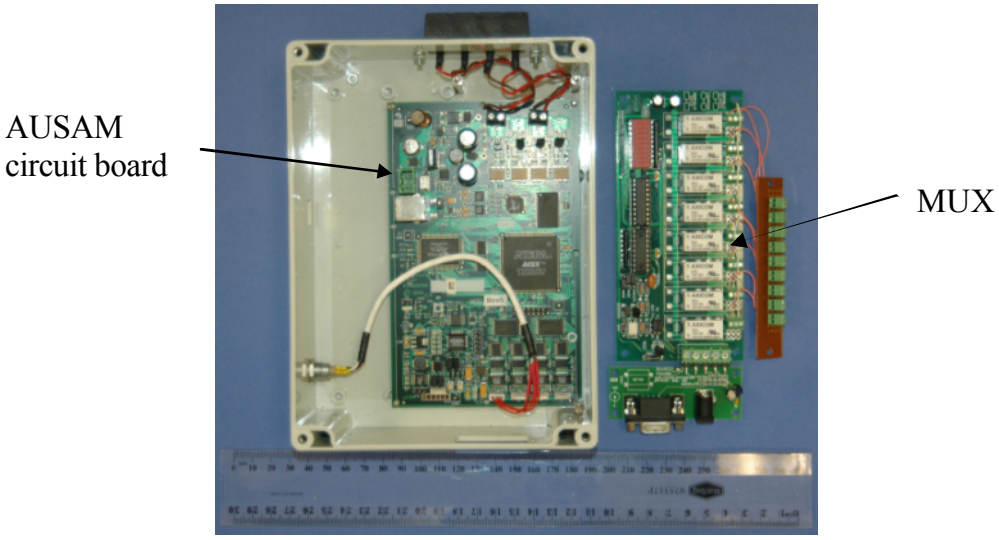


Fig.1: The AUSAM system

Case Study - Cracking in the F111 Lower Wing Skin

The many platform sustainment activities conducted by the DSTO in support of the ADF over the years provide a large range of applications that could serve as a possible focus for the development and demonstration of SHM technology. One particular metallic fatigue problem found in the F-111C aircraft in the mid 1990's was identified as an especially good candidate. It involved cracking in the lower wing skin, a problem first noticed because of fuel seepage from the wing. Subsequent inspections of that aircraft revealed a crack 48 mm long. The crack had initiated from a stiffener depression on the inside surface of the lower wing skin, approximately mid span along the wing at FASS 281.28. Calculations revealed that the crack was beyond the critical length at the design limit load and so posed an immediate threat to the structural integrity of the aircraft.

Figure 2(a) illustrates the geometry on the inside surface of the wing skin at the FASS281.28 location. The function of the stiffener depression is to allow for fuel flow and drainage between adjacent bays of the wing-box fuel tank and is referred to in this paper as the fuel transfer groove (FTG). Although it creates an obvious stress concentration, the FTG also leads to a local loss in span-wise stiffness and an eccentricity in the load path which produces out-of-plane bending, adding to the tensile stress in this region. As a result, cracks initiate on the inside surface of the wing skin, and tend to propagate chordwise along the FTG for some distance before penetrating the wing skin. It is an awkward problem for NDI, in part because the crack only penetrates the skin after growing a considerable length in the chordwise direction, but also because of crack closure. The closure stems from the loading of the wing when the aircraft is at rest on the ground, and from residual compressive stresses established by tensile plastic deformation in the FTG. A considerable effort was made by the DSTO and the RAAF to develop ultrasonic procedures to reliably detect cracks under these conditions.

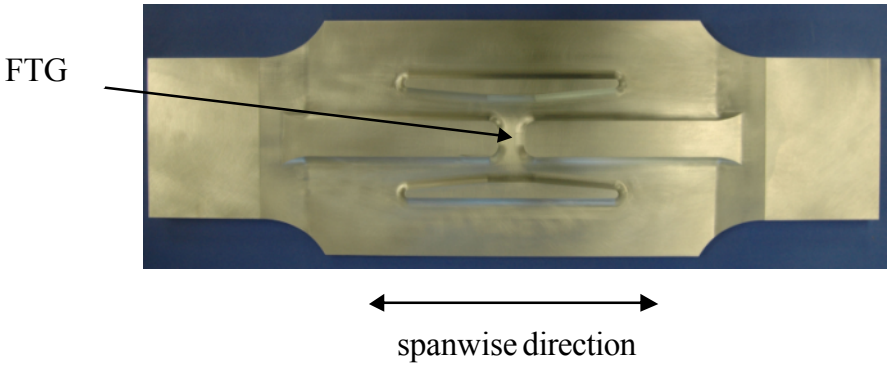
The application of an adhesively bonded repair to the wing skin makes the NDI problem even more challenging. Indeed, it has been argued that a composite patch should not be employed as a preventative reinforcement on the grounds that it obscures cracking in the host. Such concerns provide further motivation for the development of a diagnostic repair technology as it offers a basis for assessing the integrity of, not only the patch, but the host as well. The two diagnostic objectives are in fact linked since by definition a significant deterioration in the patch should lead to reduced patching efficiency and therefore an increase in the rate of crack growth. As a consequence measuring the rate of crack growth in the host provides a useful, albeit indirect, measure of patch performance and integrity.

A decision was made to consider the two diagnostic objectives separately and to initially pursue only the task of detecting cracking in the host. This was thought to be the technically simpler problem as the damage in the host is well-defined and localised, contrasting with the many possible modes and locations of damage in the patch. In addition, a staged approach to the development of diagnostic functionality in the repair was deemed essential given the limited resources available, and the need to concurrently address a suite of broader engineering issues that underpin a useful diagnostic capability. These include the factors remarked on earlier and reiterated here: transducer durability, structural complexity and real damage, all of which are represented in the FASS 281.28 example.

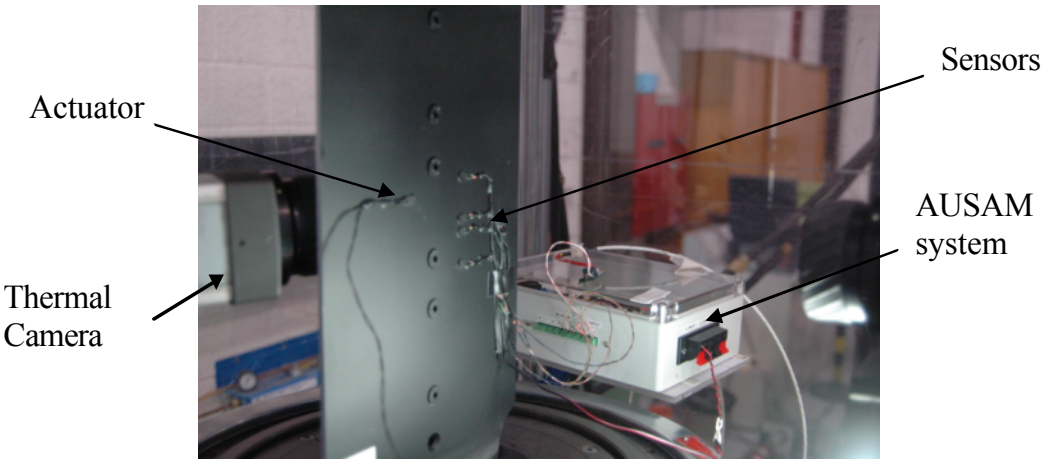
Piezoelectric transducers were located on the coupon based on a survey of the acoustic wave field using laser scanning vibrometry [5]. The frequency selection and excitation regime for the surveys and subsequent AU scans are described elsewhere [5]. Surveys were performed on a coupon in both a pristine and damaged condition. The damaged condition was simulated by machining into the FTG a semi-elliptical notch 20 mm long and 1.8 mm deep, with the major axis aligned in the chord-wise direction. Scattering from this notch was deduced by subtracting the incident wave-field measured for the pristine condition from the wave-field measured for the damaged condition. Piezoelectric transducers were attached in regions of relatively high scattering from the notch.

A total of three coupons (labelled A, B and C) were subjected to F-111C flight-spectrum loading. Each coupon was fitted with a metallic c-section member, bolted to the central stiffener, to provide a restraint to secondary bending under tensile loading, as well as to prevent buckling under compressive loading during the fatigue test [6]. Independent inspections of the FTG for cracking during the fatigue test were carried out using thermoelastic stress analysis (TSA), which requires a high infrared emissivity, so each coupon was coated with a high-emissivity black paint.

Figure 2(b) shows the test set-up for coupon A. Five piezoceramic transducer elements are shown fitted to the coupon surface. One serves as an actuator and the remaining four as sensors. The AUSAM system was located as close to the coupon as possible to minimise transducer lead lengths. Acousto-Ultrasonic interrogations were initially performed at intervals of 50 simulated flight hours (SFH) and TSA at 250 SFH. The interrogations were performed with the coupon in a load-free state – that is under prescribed zero static and dynamic loads.



(a)



(b)

Fig. 2: (a) Structurally detailed F-111C lower wing skin test coupon (internal view); (b) AUSAM system adjacent to test coupon in a 500 kN fatigue test rig with thermal camera to provide independent observations of crack growth.

Figure 3 shows the response for two transducers over a test duration equivalent to 2000 SFH. The presence of the crack was detected at around 1500 SFH as shown by the sharp change in signal gradient. The presence of a crack was corroborated by TSA. A similar trend was observed for coupon B, (Figure 4), however detection of the crack in this case occurred at approximately 2500 SFH for both AU scans and TSA. Crack closure is known to obscure crack growth in the FTG problem to conventional ultrasonic NDI. The closure stems mainly from compressive residual stresses established within the FTG [6] as a result of plastic deformation, however in practice the closure is further reinforced by the loading of the wing. To mitigate the effect of crack closure coupon C was interrogated under two separate loading conditions; firstly under a nominally zero static load, consistent with the first two tests and then under a 50 kN static load, equivalent to a net section stress (along the span-wise line of symmetry) of 60 MPa. The load level was determined from an experiment conducted on coupon B revealing a change in sensor response at this level, which was assumed to be caused by an opening of the crack.

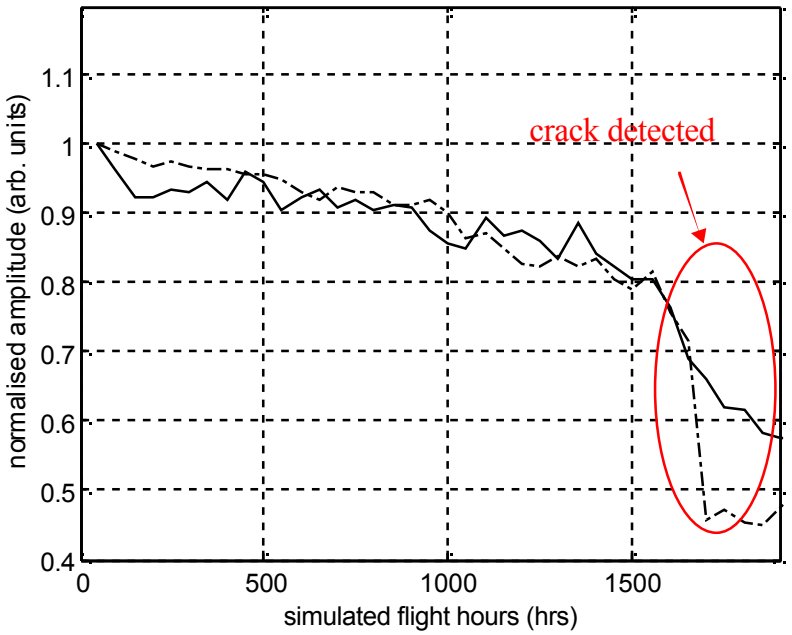


Fig. 3: Variation in peak amplitude of response signal at 1100 kHz for two transducers located near the FTG in coupon A.

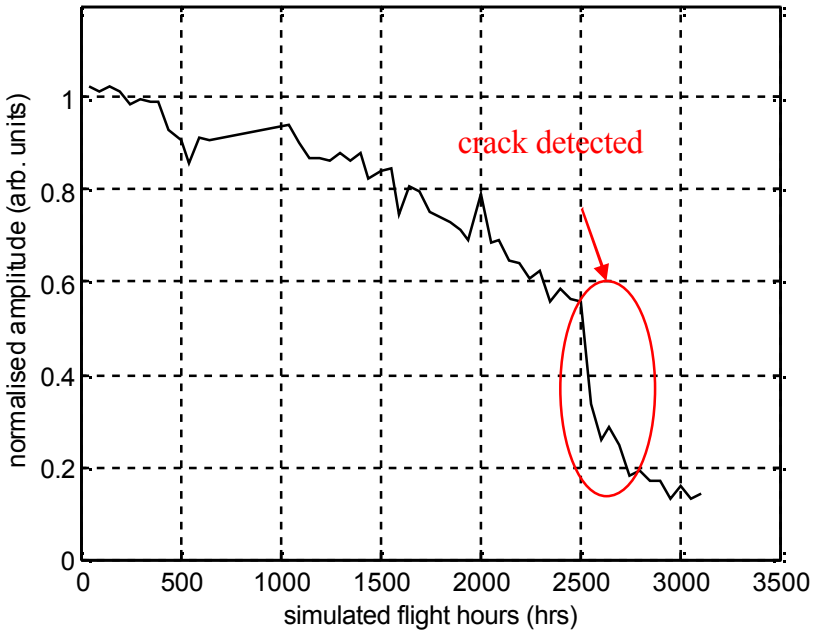


Fig. 4: Variation in peak amplitude of response signal at 1250 kHz for a transducer located near the FTG in coupon B.

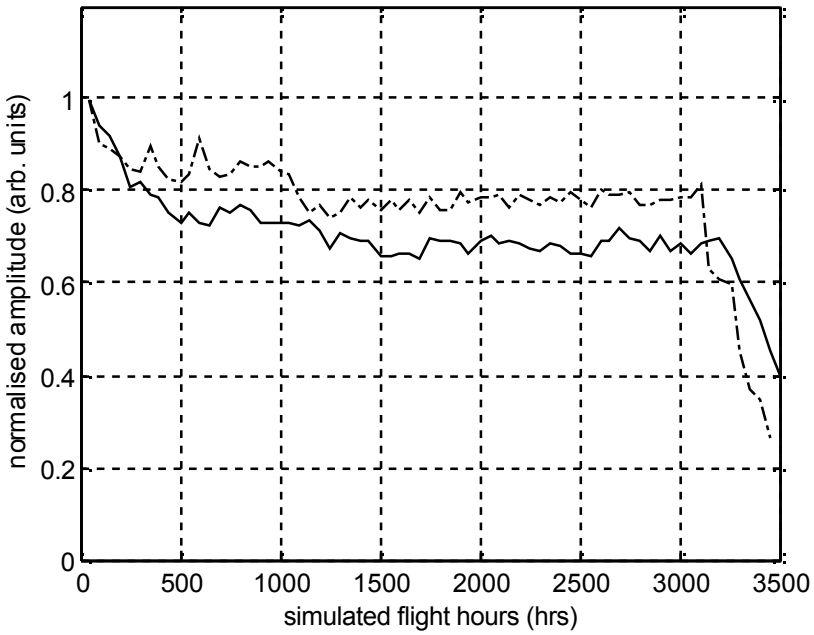


Fig. 5: Variation in peak amplitude of response signal at 1100 kHz for a transducer located near the FTG in coupon C. The solid and dotted lines correspond to applied static loads of 0 kN and 50 kN respectively.

Figure 5 shows a sensor response for the two static loading conditions. The two traces are different; however the outcome in terms of the crack detection threshold is marginal. In both cases, detection of the crack was judged to have occurred at approximately between 3100 and 3200 SFH, much later than in the previous tests. In fact, the TSA results showed presence of a crack at 2850 SFH. By 3200 SFH the crack had visibly grown beneath the actuator transducer element, raising the possibility that the change in sensor response may have occurred as a result of damage to the transducer itself.

Discussion

Some of the key findings drawn from the study include:

- Acousto-Ultrasonics provides a potentially useful basis for the detection of cracking in the FTG, based largely on the result from coupon A.
- The AUSAM performed reliably and had the requisite bandwidth, sensitivity and noise performance for the AU inspections.
- The repeatability between tests was poor. This is likely to have been caused, in part, by variations in the behaviour of the coupon, both in terms of fatigue properties and the elastic wave dynamics, as well as to variations in the performance of the AU system between installations.
- Variations in the performance of the AU system are attributable in part to a deterioration of the transducer caused by exposure to sustained mechanical loading [7,8].

Although the AUSAM performed well in the present study, deficiencies in both the form and functionality of the device were identified and as a result an improved device is currently being developed. The new module will have a smaller footprint, approximately half the size of the current device and the system power requirements are being lowered to ensure that a USB2.0 outlet on a typical notebook computer is able to provide adequate power for most applications. Looking further ahead, low energy consumption is vital if these systems are to be made completely autonomous by using power harvested from in-flight strains and vibrations in the airframe.

Conclusion

This paper has presented experimental work demonstrating the efficacy of an acousto-ultrasonic SHM approach to the detection of fatigue cracking in a geometrically complex F-111C wing skin structure. Although the results were encouraging, they underscore the need for substantial further development before the approach can be certified for use in real aircraft applications.

References

1. Alleyne, D. N. and Cawley, P., (1992), "The Interaction of Lamb Waves with Defects", *IEEE Trans. Ultrasonics, Ferroelectrics and Frequency Control*, Vol. 39, pp. 381-397.

2. Rajic, N., and Rosalie, S. C., (2008), "A Feasibility Study into the Active Smart Patch Concept for Composite Bonded Repairs", *DSTO Technical Report*, DSTO-TR-2247.

3. Viktorov, I. A., *Rayleigh and Lamb Waves - Physical Theory and Applications*, New York: Plenum Press, 1967.

4. Walley, A., and Rajic, N., (2004), "In situ Structural Health Monitoring of an Impact Damaged F/A-18 Horizontal Stabilator", *Proc. Second Australasian Workshop on Structural Health Monitoring*, Monash University, Melbourne, 16-17 Dec. 2004.

5. Rajic, N., Rosalie, S. C., and Tsoi, K. A., (2011), "In Situ Monitoring of Fatigue Cracking in a Wing Skin by Means of Acousto-Ultrasonics", *DSTO Technical Report*, In Preparation.

6. Liu, Q., Ryan, M. and Hugo, G., (2006), "Limitations of Manual Ultrasonic Inspection for Detection of Cracks in Aircraft Wing Skins", *International Conference on Structural Integrity and Failure (SIF2006)*, Sydney, published in CD.

7. Paget, C. A., Levin, K. and Delebarre, C., (2002), "Actuation Performance of Embedded Piezoceramic Transducer in Mechanically Loaded Composites", *Smart Mater. Struct.*, Vol. 11, pp. 886-891.

8. Tsoi, K., and Rajic, N., (2010), "Mechanical Durability of Piezoelectric Transducers for Structural Health Monitoring Applications", *Proc. of the Third Asia-Pacific Workshop on Structural Health Monitoring*, The University of Tokyo, Tokyo, Japan, 30 Nov. - 2 Dec. 2010.