AUTOMATED ULTRASONIC INSPECTION FOR CRACK DETECTION AT F-111 LOWER WING SKIN FASTENER HOLES

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Abstract

The failure of an F-111 wing during a full-scale fatigue test had important implications for the structural integrity management of the RAAF F-111 fleet. This failure was due to a fatigue crack which initiated in the bore of a fastener hole. To assure structural integrity, an automated ultrasonic inspection has been developed which will be applied to up to 1200 fastener holes in each wing. The holes are inspected from the lower surface and definitive assessments of wing serviceability must be made without removing fasteners. The inspection system uses focussed immersion probes to perform a 45-degree angle-beam shear wave inspection, utilising full-waveform capture and post-processing. This style of complex, computer-based inspection system is new to the Australian Defence Organisation and challenged many existing engineering processes for performing NDT and interpreting the results. This paper reviews the application of automated ultrasonic inspection for the F-111 lower wing skin and outlines some of the significant challenges for both the science behind the inspection and the associated engineering processes. The lessons learnt will aid the successful integration of new technologies into existing NDT practices in the future.

1. Introduction

In 2002, an F-111 wing failed unexpectedly during full-scale fatigue testing due to a fatigue crack which initiated at a Taper-Lok fastener hole in the lower wing skin. Fractography demonstrated that poor hole surface finish and inadequate interference of the installed TaperLok fasteners were key factors contributing to the premature failure, whilst teardown of the failed wing revealed that poor surface finish and below specification interference were widespread across the whole lower wing skin of the failed wing [1, 2]. In addition, more than 80 fastener holes in the failed wing were shown to contain cracks, with 8 cracks between 1 and 11 mm in radial depth [3]. As a result of these observations, inspection of a large number of fastener holes per wing is likely to form an important part of the structural integrity management strategy for the F-111. The inspections must detect fatigue cracks initiating at any depth within the hole bores for skin thicknesses ranging from 5 mm to 33 mm.

A challenging feature of the inspection requirements for the F-111 lower wing skin is that definitive assessments of wing serviceability must be made without removing fasteners, preventing supplementary inspection of fastener holes by bolt-hole eddy-current (BHEC) techniques. This requirement presents a significant additional problem compared to previous applications of automated ultrasonic scanning for crack detection, which have normally allowed fastener removal to confirm indications by BHEC (and allow defect removal by oversizing).

2. Detection of Cracks at F-111 Lower Wing Skin Fastener Holes

A number of techniques were considered for inspection of the F-111 lower wing skin, including low-frequency eddy current with sliding probes, phased array ultrasonics, and both manual and automated contact probe angle-beam ultrasonics, Following an evaluation of the options, including trials on sections from F-111 wings containing artificial electric discharge machined (EDM) defects, inspections were implemented using a commercial automated ultrasonic scanning system[1] with focused immersion transducers and a dripless recirculating water-immersion scanning head to perform a 45º shear-wave inspections in the aluminium skin, refer to Figure 1 and Figure 2. This

1 Wesdyne AMDATA IntraSpectTM ultrasonic inspection system, Westinghouse Electric Company LLC.

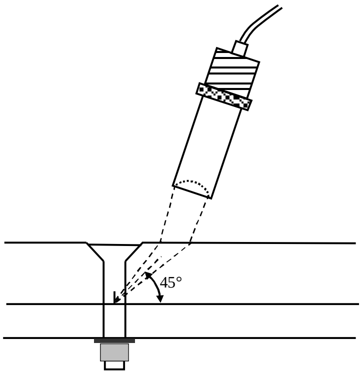

Figure 1 Schematic of 45º angle-beam shear-wave inspection using focussed immersion transducers

Figure 2 ACES™ dripless recirculating immersion scanning head[1].

system uses a pair of 15MHz immersion transducers oriented parallel to the wing spars to detect cracks oriented in the chordwise direction at the fastener holes.

The scanning system is computer based and stores the full-waveform ultrasonic A-scan (amplitude vs time) at each point over large area scans acquired at a rate of 150mm/s. It allows post-processing of the full-waveform data including A-scan, B-scan (vertical cross-section) and C-scan (plan view) presentations, Figure 3. The vertical cross-sections (B-scans) are particularly valuable for interpreting ultrasonic indications in complex wing structure and for discriminating possible cracks from indications due to geometric features within the component.

Using 45 ° shear wave ultrasonics to detect cracks at fastener holes is an established technique. Automated versions have been developed to detect corner cracks at fastener holes in C-130 and C-141 wings, the latter techniques having been used on USAF C-141 aircraft for some years [4].

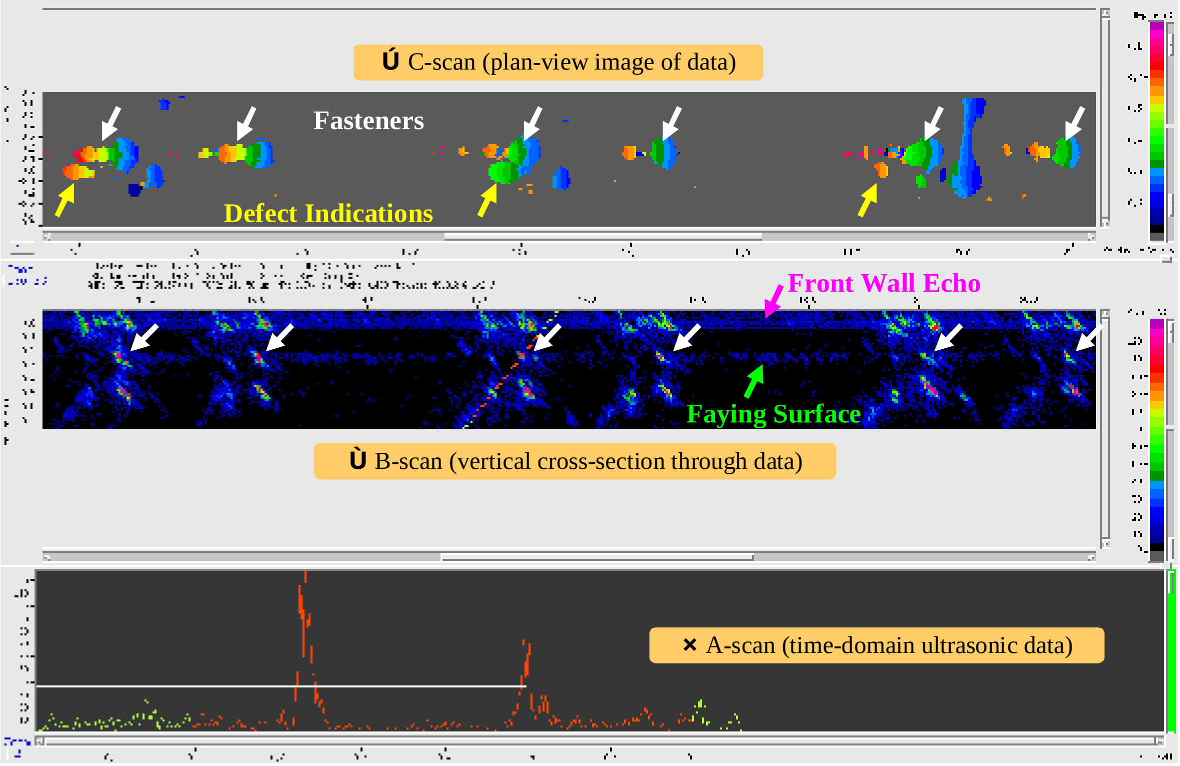

Figure 3. Data presentation for automated ultrasonic system.

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1 The ACES is a proprietary design of Science Applications International Corporation.

In the F-111 application, the requirement is to be able to detect cracks at any height up the bore of the fastener holes, with the fastener in place and in wing structure which is complex and includes sharply tapering skin thicknesses, web-stiffeners on the spar and fuel transfer grooves between the skin and the spar. The skin thickness within the required inspection region varies significantly from 6 to 33 mm over the span of the wing. Probably the most challenging aspect of the F-111 wing inspection is the need to make go/no-go decisions on wing serviceability from the ultrasonic data alone, without recourse to BHEC for confirmation of defects.

Development and validation of the inspection processes for the F-111 wings required both: (i) demonstration of reliable detection of simulated cracks in real wing structure, and (ii) validation of the sensitivity for detection of real fatigue cracks compared to the simulated defects (EDM notches) in the test wings. These requirements were dictated by the complexity and variability of the F-111 wing structure over the large inspection region, combined with the fact that it is completely unfeasible to generate fatigue cracks with suitable sizes distributed throughout a real wing.

For the first part of the validation, EDM notches were inserted in the bore of fastener holes in a decommissioned F-111 wing. This involved the temporary removal of a number of taper-lok fasteners and in-situ insertion of more than 60 EDM notches into the bores of the fastener holes without any dismantling of the structure other than fastener removal. The types of EDM defects inserted are shown in Figure 4. The test wing containing EDM notch defects was critical in establishing a viable analysis procedure for the acquired data which would reliably discriminate possible cracks from other benign structural features.

Initial trials conducted using 5MHz transducers (selected as a commonly used frequency for angle-beam shear-wave inspections) showed that whilst corner defects located at the faying surface[1] were easily detected, sensitivity for mid-bore defects part way up the bore of the fastener hole was relatively poor, particularly for 'mid-bore top' defects located just under the countersink. Additional trials were conducted using both 10 and 15MHz transducers. These trials demonstrated that 15MHz provided much better detection of mid-bore defects. With 15MHz, the geometry of the fastener hole could be clearly distinguished in the B-scan, Figure 5, and the crack indications exhibit a clear linear profile in the B-scan, see also Figure 8.

Two pairs of transducers with different focal lengths and element diameters were employed to cover the wide range of thickness to be scanned. The outboard regions of the wing, where the skin thickness ranges from 6 and 12 mm, are scanned with 38 mm (1.5 inch) focal length 15MHz transducers. For the inboard region with skin thicknesses between 12 and 33 mm, two scans are performed, using both 38 mm and 76 mm (3 inch) focal length 15MHz transducers, in order to provide adequate sensitivity to cracks initiating at any location within the hole bore.

2.1. Post-Process Analysis of Acquired Ultrasonic Data

The fact that the system stores the full A-scan (RF or rectified) data at every point in the scan area allows the data analysis to be performed as a post-processing operation, with full flexibility to change the gates to optimise detection of different types and locations of defects. Cracks at the faying surface give very strong (high amplitude) ultrasonic reflections due to the ultrasonic energy being reflected back towards the transducer by the specular reflection from the 90° corner formed by the intersection of the crack and the faying surface. Such faying-surface corner cracks are readily detected using an analysis gate which extends from just after the front wall echo to beyond the faying surface. However, detection of mid-bore cracks is reliant on non-specular (diffuse) scattering from the crack surface, Figure 8. These 'direct' reflections are much weaker in amplitude than the specular reflections from faying-surface corner cracks and may be weaker in amplitude than the scattering from the shot-peened faying surface.

The detection of mid-bore cracks is optimised by gating within the first layer to exclude both the front wall echo and the faying surface. This allows mid-bore cracks to be detected against a minimum noise threshold limited only by scattering from within the material, which has a much lower amplitude background signal than the faying surface.

C-scans generated using the gates described above can be presented in either peak amplitude (Figure 6) or depth display modes (Figure 7). Depth mode enhances ease of identifying cracks growing normal

[1] Faying surface is the interface between the wing skin (first layer) and spar cap (second layer).

to the holes because of their characteristic colour banding.

In the regions of the wing where the skin is thicker than 12 mm, the 38 mm focal length scan data is reviewed for mid-bore defects in the upper part of the fastener hole bore, using a short gate (within the first layer) where necessary to eliminate faying surface noise. The 76 mm focal length scan is reviewed with both a short gate to look for mid-bore defects in the lower part of the bore and also with a long gate to look for large defects in the second layer.

The analysis procedure makes extensive use of the B-scan data to support interpretation, particularly to unambiguously identify cracks based on the direct reflections observed in the B-scan, Figure 8 and Figure 9.

Indications are classified into class A or B. A class A indication exhibits a clear linear crack-like profile in the B-scan, with a measurable extent in the thickness direction. A class B indication is in the correct location to be a crack at a fastener hole and has sufficient amplitude or area to be notable. However, a class B indication may not be distinguishable from mechanical damage at the hole, excessive surface roughness or other non-crack sources of reflection.

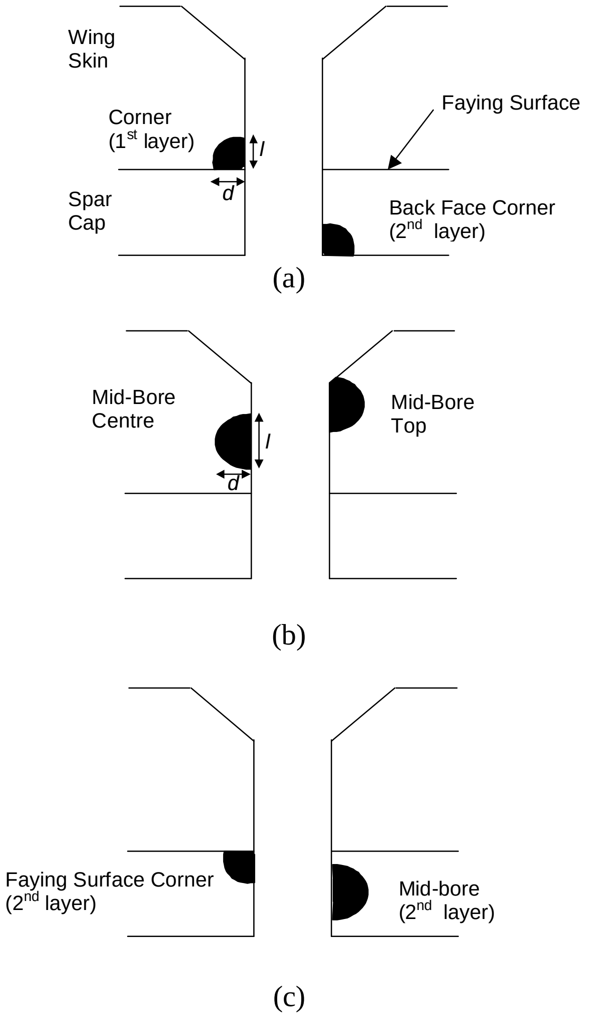


Figure 4 Schematics showing cross-section of fastener holes and the geometry of EDM notches inserted in test wing: (a) corner and back-face corner. (b) mid-bore centre and mid-bore top. These are distinguished according to their location at either the centre of the hole bore (centre) or immediately under the countersink (top). (c) faying surface corner and second-layer mid-bore. All EDM notch defects were nominally half- or quarter-penny in shape.

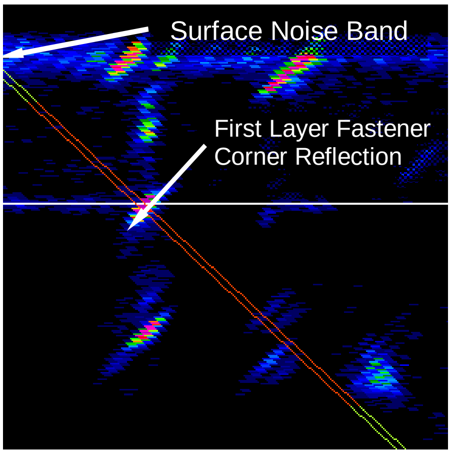


Figure 5 Fastener hole reflections in the B-scan

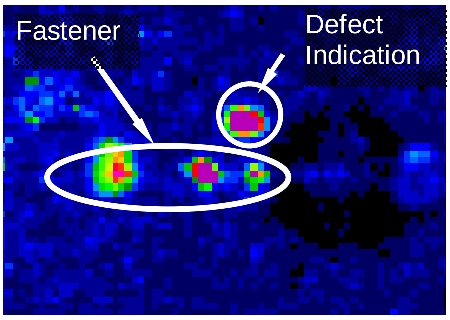


Figure 6 Typical corner defect indication in peak mode c-scan

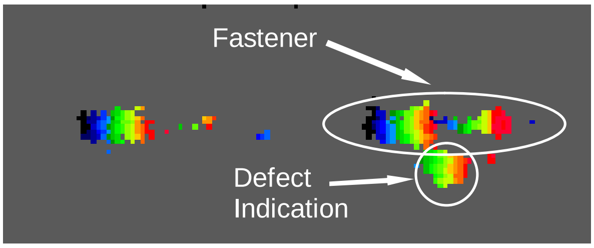


Figure 7 Typical mid-bore defect indication in depth mode C-scan.

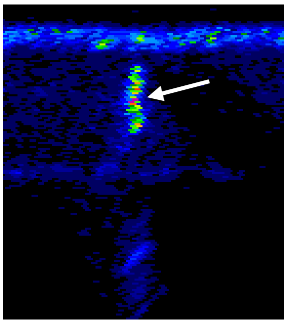


Figure 8 Direct reflection (arrowed) from a mid-bore crack in the B-scan

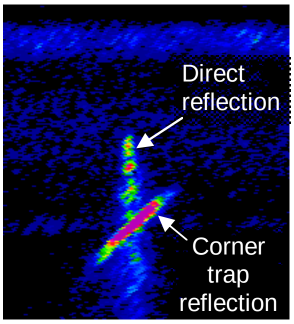


Figure 9 B-scan showing a typical large corner crack exhibiting both a corner trap reflection and a direct reflection. (Small corner cracks exhibit the corner trap reflection only.)

3. Studies Using Fatigue Crack Specimens

Specimens containing genuine fatigue cracks were fabricated to quantify the difference between the ultrasonic reflections from fatigue cracks compared to EDM notches. The specimens are 2024-T851 Al alloy 100 mm wide × 700 mm long and either 12 mm or 24 mm in thickness, with surfaces shot-peened to the F-111 wing specification. Each specimen contains either 11 or 16 fastener holes in a single line along the major axis of the specimen, Figure 10. Fatigue cracks have been initiated in undersize holes from small (0.5 mm) EDM notches. Once the cracks have initiated, the holes are taper-reamed to their final size to remove the starter notch, whilst leaving the fatigue crack. These specimens are then fatigued further to grow the cracks to the sizes required for validation of the F-111 wing automated ultrasonic procedure.

The specimens were fatigued using flight-by-flight load spectra, representative of the actual in-flight stresses at critical locations of the F-111 wings in RAAF service, in order to generate cracks as close as possible to those which might be present in in-service wings. It is expected that these spectra will generate cracks with a high residual compressive stress, i.e. tightly closed cracks.

Figure 10. Fatigue crack specimen

The automated ultrasonic inspection system was mounted onto the fatigue testing machine to enable ultrasonic scans to be performed whilst the specimens are under an applied tensile load. This allows the effects of crack closure on ultrasonic response to be quantified and compared to the ultrasonic response from other specimens containing EDM notches as simulated cracks.

3.1. Ultrasonic Response from Fatigue Cracks in Specimens

Figure 11 shows the comparison of the peak amplitude and area of indications observed in the ultrasonic C-scans for both (i) corner cracks in 12 mm thick specimens and (ii) corner EDM notch data. The sizes of the fatigue cracks have been estimated from ultrasonic data acquired with the specimen under a 150 kN applied load, which is sufficient to fully open the cracks. Fractography of two prototype specimens demonstrated good agreement between the true crack size and the crack size measured from the ultrasonic data. The sizes of cracks in other specimens will be confirmed by fractography at a later date.

The results indicate a dramatic decrease in the amplitude and areas of the C-scan indications for the laboratory grown fatigue cracks compared to EDM notches. Both the C-scan indication areas and peak amplitudes are consistently lower for the cracks than the EDM notches and much more scatter is observed in the crack data. For one crack

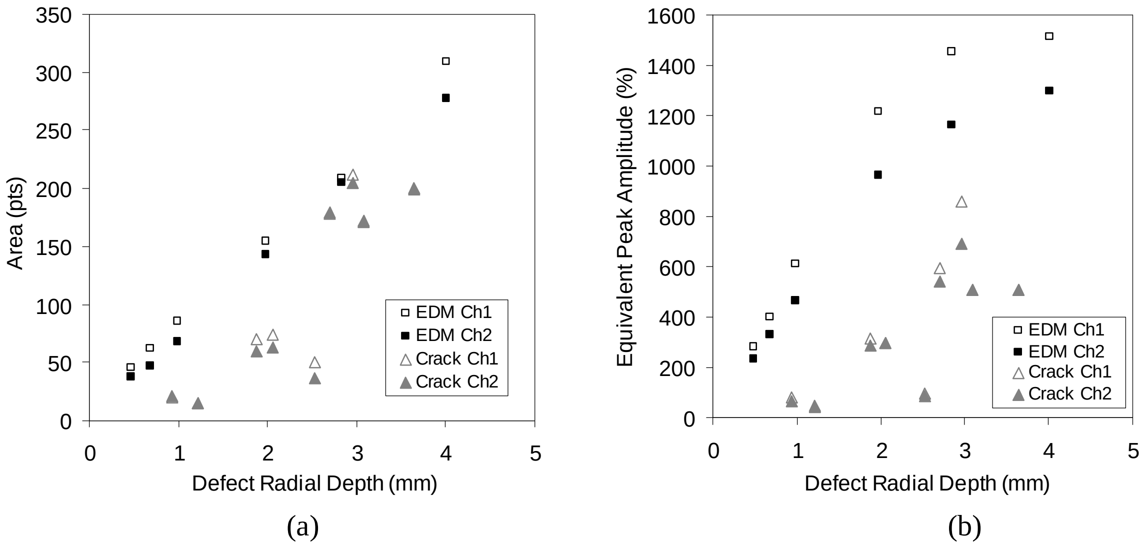


Figure 11. Ultrasonic response from corner fatigue cracks in specimens compared to corner EDM notch defects. (a) Area of indication above threshold, (b) Equivalent peak amplitude. [5]

of approximately 2.5 mm radial length, the amplitude was just 7% of the amplitude from a similar sized EDM notch. This means that any realistic assessment of the overall probability of detection for the automated ultrasonic inspection must take account of the differences between real cracks compared to EDM notches

3.2. Effect of Crack Closure on Ultrasonic Response

The differences between fatigue cracks and EDM notches shown in Figure 11 include contributions from a range of factors including surface roughness of the defects and crack closure. The effects of crack closure have been further examined by performing ultrasonic inspections on specimens under an applied tensile stress to progressively counter any residual compressive stress that may act to close the crack faces against each other. Such residual stresses are expected to arise from notch plasticity at the fastener holes occurring during peak loads in the fatigue spectrum. By performing the ultrasonic measurements at a series of applied tensile loads, the level of residual compressive stress acting on the cracks faces can be estimated and the effect of the residual stress on the crack indications can be characterised.

Figure 12 shows that the peak amplitude of the ultrasonic reflection increases with increasing applied tensile stress up to approximately 60 MPa, above which it tends to plateau. This is shown for a number of cracks of different sizes; those with larger amplitudes are the larger cracks. The response is consistent with the fatigue cracks initially being tightly closed. The amplitude increases with applied tensile stress until the crack is fully open with no significant contact between the crack faces. Thus 60 MPa of applied tensile stress is required to fully open the cracks in these specimens. Application of a compressive stress is observed to further reduce the amplitude of the ultrasonic indications from the cracks.

The influence of residual stress on the imaging of small cracks is shown in Figure 13. Under an

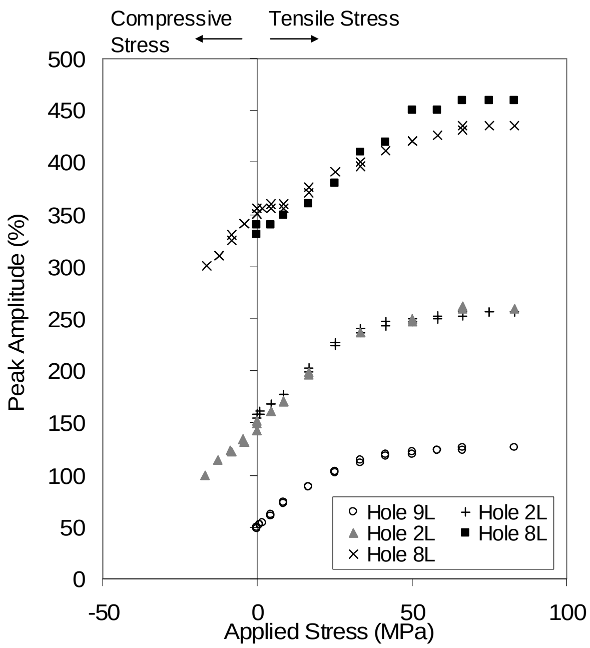
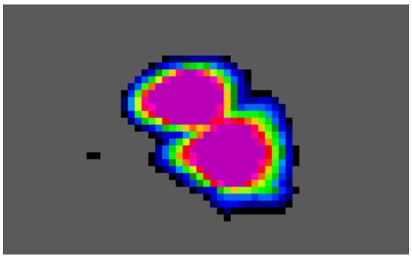
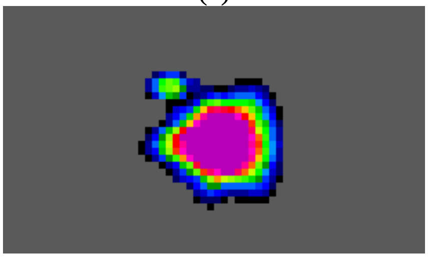


Figure 12 Ultrasonic response from fatigue cracks in specimens under an applied compressive and tensile stress.

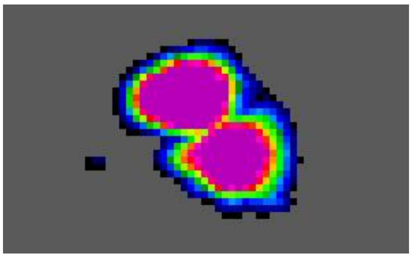


(a)

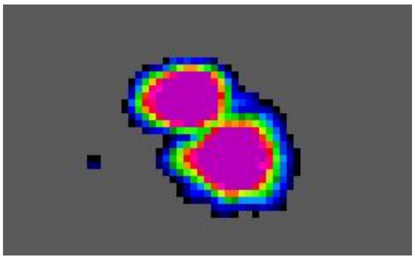


(b)

Figure 13 Change in ultrasonic indication with applied tensile stress for a small corner crack (estimated size < 2 mm) at a fastener hole: (a) Applied tensile stress of 80 MPa (b) No applied stress.



(a)



(b)

Figure 14 Change in ultrasonic indication with applied tensile stress for a larger corner crack (estimated size 3-4 mm) at a fastener hole: (a) Applied tensile stress of 80 MPa (b) No applied stress.

applied tensile stress of 80 MPa, Figure 13(a), the crack is fully open giving a clear ultrasonic indication, comparable to a similar sized EDM notch. However, with no applied load, Figure 13(b), both the amplitude and area of the ultrasonic C-scan indication are significantly reduced due to local residual compressive stresses acting to close the crack faces. For a larger fatigue crack, Figure 14, the effect of compressive residual stress is much less significant and the crack is clearly visible in the C-scan data with no applied tensile load.

4. Reliability Assessment

As part of the overall validation of the F-111 automated ultrasonic inspection procedures, DSTO is working with the Royal Australian Air Force (RAAF) to determine the probability of detection (POD) for the inspection. For this application, a conventional POD trial alone would be unable to capture the simultaneous effects of the complex wing structure, realistic defects and the human factors in inspecting a very large number of fastener locations. Ideally, a thorough validation of procedure reliability would involve testing on genuine fatigue cracks in representative structure. However, it is not feasible to introduce fatigue cracks into structure that would be sufficiently representative of the wing geometry. In order to overcome these limitations, DSTO is currently developing model-assisted methods to determine the overall POD for detection of fatigue cracks in wings based on a combination of data from field POD trials on artificial defects in retired wings and fatigue cracks in specimens. A key part of this work involves characterising the differences between the ultrasonic responses of genuine fatigue cracks compared to EDM notches in sufficient detail to be used as input to the POD modelling approach.

5. Implementation Challenges

The implementation of automated ultrasonic immersion scanning of fastener holes in the F-111 lower wing skin poses many challenges for both the science of the inspection and the surrounding engineering processes.

5.1. Training

The automated scanning system and analysis software uses PC-based equipment, which requires specialised training and experience to enable the inspector to feel confident and proficient to conduct the inspection. Consideration needs to be given to the possible variation in computer literacy between individual inspectors, which may depend on factors beyond formal NDT training.

It is likely that training for other advanced computer based NDT equipment, such as phased-array ultrasound and digital radiography, will encounter issues similar to those described above. In order to gain widespread acceptance and usage, all these technologies will require extensive training investments.

5.2. Classification, Reporting and Disposition of Indications

Due to the structure of the F-111 wing, definitive assessments of wing serviceability must be made without removing fasteners to allow supplementary inspection by bolt-hole eddy-current techniques. This means that the reporting thresholds and classification of indications is critical for providing an acceptable engineering solution. With over 1000 fastener holes per wing, a false call rate of even 0.1% would generate an engineering assessment workload to resolve indications that would be untenable.

As discussed under §2.1, indications are classified as class A or B according to the level of confidence that the indication represents a significant crack. A class A indication will render a wing unserviceable. Class B indications will be subjected to close assessment on subsequent inspections and the interval for re-inspection may be based on the measured size.

It is important that B-class indications be measured, recorded and assessed against strict criteria. This process contributes to the overall reliability of the inspection by ensuring that technicians remain vigilant about examining all possible indications.

6. Conclusion

The implementation of an automated ultrasonic scanning inspection for fastener holes in the F-111 lower wing skin has been a long and complex process. There is still ongoing work on the formal reliability assessment and also training and re-certification issues to be resolved. However, the inspection is in place and the greatest hurdles have already been overcome. This is likely to be the first of many complex, computer-based inspection systems to be used in the non-destructive inspection of defence airframes in the future. The lessons learnt from this application will aid the successful integration of new technologies into existing NDT practices in the future.

7. Acknowledgements

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