Abstract

This paper discusses repairs and modifications to thin load bearing aircraft structures using Supersonic Particle Deposition technology, a topic which is currently under consideration by both the Australian Naval Aircraft Systems Project Office (NASPO) and the RAAF Directorate General Technical Airworthiness (DGTA). Ensuring continued airworthiness is of paramount importance and it is essential that (aircraft) structural integrity be maintained after repairs have been installed. To this end the present paper summarises the results of a series of experimental studies into the ability of SPD doublers to extend the fatigue life of thin aluminium structural components and the limit of viability (LOV) of fuselage lap joints.

1 Introduction

The high acquisition costs associated with the purchase of modern civilian and military aircraft coupled with the existing economic and market forces have resulted in utilization of aircraft beyond their original design life. This trend coupled with a number of high visibility aviation accidents has served as a trigger for government and industry action. In this context the April 1988 Aloha accident revealed a number of fundamental weaknesses both in structural design and maintenance. In this incident failure was due to the presence of multiple cracks in neighboring locations, a phenomena this is referred to as Multi Site Damage (MSD), coupled with corrosion damage and a less than complete maintenance system. Although in isolation each event was acceptable the overall effect was to compromise the structural integrity of the aircraft. It was also found that multiple mechanical repairs, in close proximity, can compromise structural integrity. In the military sphere the June 2007 Report to Congress by the Under Secretary of the Department of Defense (Acquisition, Technology and Logistics) [1] estimated the cost of corrosion associated with US DoD systems to be between $10 billion and $20 billion annually. This report outlined the need for research into four primary areas one of which was: Repair processes that restore materials to an acceptable level of structural integrity and functionality. It has recently been shown [2, 7] that supersonic particle deposition (SPD) technology has the potential to meet this challenge and it is in this context that the present paper discusses SPD repairs and modifications to thin load bearing aircraft structures and fuselage lap joints.

In line with current FAA and USAF US Defense [8, 9] guidelines all structural repairs carried out to aircraft must be approved by a competent airworthiness authority. In accordance with FAA AC’s No: 25.1529-1 [9] and AC No: 25.571-1A [10], Mil-HDBK 130 and the USAF Damage Tolerant Design Handbook [8] the damage tolerance evaluation of the repair is intended to ensure that should serious fatigue, corrosion, environmental degradation, impact damage, disbonding, delamination or accidental damage, occur to the repair then the remaining structure can withstand reasonable loads, without failure or excessive structural deformation, until the damage is detected. Furthermore, in accordance with the guidelines outlined in [8-10] the
2 Damage Tolerance Assessment of Both Repaired and Unrepaired Structure

In this study, it was shown that crack growth in 1.27 mm thick 2024-T3 clad aluminium alloy single edge notch tension (SENT) specimens, see Figures 2 and 3, tested under constant amplitude loading with \( \sigma_{max} = 180 \text{ MPa} \) and \( R = \sigma_{min}/\sigma_{max} = 0.1 \) could be eliminated using a 1.0 mm thick SPD doubler. (This stress level was chosen since it represents a realistically upper bound on the stresses that can be expected in a thin wing skin/fuselage skin.) In this test program the baseline specimen, i.e. without an SPD doubler lasted approximately 35,000 cycles. In contrast the 7075 SPD patched panel test was stopped after approximately 60,000 cycles with little, i.e. no evident, damage in the 7075 SPD or crack growth in the 2024-T3 skin. To illustrate this Figures 4 and 5, from [2], present infrared pictures of the stress field at 11,100 and 56,100 cycles respectively associated with the first SPD specimen. These figures show that the stresses in the SPD doubler remained essentially unchanged throughout the test. Examination of the specimen revealed no evidence of crack growth or damage in the SPD or in the skin under the SPD.

2.1 Fatigue Life Enhancement Using SPD

Whilst SPD is now widely used on Australian Seahawk helicopters [4, 6], see Figure 1, the paper by Jones et. al. [2] was the first to reveal the potential of SPD to enhance the structural integrity of thin aluminium structures.
Because thin skinned structures often contain fasteners and other stress concentrators this test program was repeated at a higher stress level, viz: with 4,300 constant amplitude cycles at a maximum load $P_{\text{max}} = 10$ kN and $R = 0.1$ followed by constant amplitude loading with a maximum load $P_{\text{max}} = 25$ kN, which corresponds to a peak stress in the working section of 275 MPa, and $R = 0.1$. (Note that the yield stress for this material is approximately 320 MPa.) In these tests each specimen had a small (nominally) 0.5 mm long edm starter crack. For the SPD specimen a crack was cut into both the 2024-T3 skin and the SPD. In the case of the baseline specimen, i.e. no SPD patch, the specimen failed catastrophically at approximately 1,800 cycles. The SPD repaired panel had a 0.5 mm thick SPD patch on either side. The test was stopped at approximately 13,700 cycles at which stage the crack was approximately 3.7 mm, see Figure 6. An infra-red picture of the specimen at approximately 9,300 cycles is shown in Figure 7 where the elevation in the stress around the crack tip is clearly evident.

As the number of cycles seen by the repaired specimen was greater than 6.5 times the life of the unrepaired panel the test was stopped at ~ 13,700 cycles. The results of these test when taken in conjunction with the result presented in [2], that a thin SPD strip located just ahead of a 2 mm long edge crack in a 1.27 mm thick 2024-T3 aluminium alloy specimen stopped all crack growth, reveal the potential for SPD to significantly enhance the structural integrity of thin aluminium alloy wing and fuselage skins.
2.1 Predicting crack growth in SPD repaired structures

The in service assessment of SPD repairs and structural modifications requires a damage tolerance analysis of both the unrepaired and the repaired structure. However, before we attempt to predict the fatigue performance of an SPD repaired panel we first need to establish that the methodology used can predict the growth of small cracks in an unpatched panel.

To evaluate this we tested two 1.27 mm (thick) x 76 mm (wide) 2024-T3 aluminium alloy SENT (single edge notch tension) specimens. The specimens, which were tested in laboratory air at a frequency of 5 Hz, had a small 0.5 mm semi-circular notch from which the cracks grew. The first test had a maximum stress $\sigma_{\text{max}} = 160$ MPa and $R = 0.1$. In the second test we had a maximum stress $\sigma_{\text{max}} = 107$ MPa and $R = 0.1$. The smallest through-the-thickness crack analysed in this study was approximately 0.29 mm. The resultant crack growth histories are shown in Figure 8.

A variant of the Hartman-Schijve crack growth equation presented in [11] for 2024-T3 was then used to predict crack growth, viz:

$$\frac{da}{dN} = D(\Delta K - \Delta K_{thr})^2/(1- K_{\text{max}}/A)$$  \hspace{1cm} (1)

where $A = 50$ MPa $\sqrt{\text{m}}$, and $\Delta K_{thr} = 0$ MPa $\sqrt{\text{m}}$. The value of $D$ given in [11] for this alloy was $1.2 \times 10^{-9}$. This formulation was chosen because:

i) It has been shown to hold for a wide range of aerospace aluminium alloys [11].

ii) It has been shown to hold for both long and short cracks [11, 12].

The resultant predicted crack length histories are shown in Figure 8 where good agreement between the measured and predicted crack length histories.

Having established the ability of this formulation to predict crack growth in the baseline specimens we subsequently used equation (1), with the values given above, to predict the crack length after 13,700 cycles for the SPD repaired specimen tested at $\sigma_{\text{max}} = 275$ MPa and $R = 0.1$. This gave a predicted crack length, including the length of the starter crack, of 3.1 mm which is in good agreement with the measured length of 3.7 mm which was obtained using digital cameras, see Figure 6.

![Figure 8](image_url) Measured and predicted crack length histories.

![Figure 9](image_url) The dangers of cracks linking from multiple repairs in fuselage lap joints, from [12].
2.1 Application to Mechanically Fastened Joints

Having illustrated the ability of SPD to enhance the structural integrity of thin skins let us next evaluate the ability of an SPD doubler to extend the fatigue life of mechanically fastened joints and in particular fuselage lap joints. The 1988 Aloha accident, where cracking in the joint ran from one repair to another, see Figure 9 from [14], revealed that the problem of cracking in fuselage lap joints can be exacerbated by the existence of multiple corrosion repairs in the joint.

As a result it is now relatively common practice to seal the edges of the mating surfaces. However, as shown in [15] this does not stop the environment entering the joint through the fasteners, see Figure 10 where we show fluid bleeding from a cracked fastener hole. In this particular example the fasteners had been exposed to a few drops of fluid prior to testing. The fluid dramatically increased the crack growth rate and bled from the (resulting) cracks [15].

Figure 10 Bleeding of fluid from cracks and rivet heads, from [15].

The extent of the problems associated with fuselage lap joints is aptly illustrated by the April 2011 incident whereby cracking in the fuselage lap joint in a Southwest Airlines Boeing 737-300 aircraft resulted in a large 5 foot hole in the roof, see Figure 11.

This incident led to the grounding of 79 of its older Boeing 737 aircraft [16] and to the cancelation of almost 700 flights. Subsequent inspections, which found cracks in a total of four Southwest aircraft, [16] led to the US FAA mandating the inspection of 175 Boeing 737 aircraft that had seen more than 35,000 cycles. The problem of cracking in fuselage lap joints is not confined to Boeing 737 and 727 aircraft. On 26th October 2010 an American Airlines 757-200 aircraft was forced to land at Miami International Airport due to a sudden decompression arising from cracking in a fuselage joint [17]. This aircraft had experienced less than 23,000 cycles. This led to the discovery of cracking in other 757 aircraft and a subsequent January 2011 FAA Airworthiness directive [17] mandating the inspection of all 757-200 and 757-300 aircraft.

Figure 11 Tarpaulin covering the five-foot-hole that ripped open in the roof, from [16].

As a result of these incidents the FAA have introduced the concept of a limit of viability (LOV), defined as the onset of multi-site and/or multi-element damage [18, 19], which the FAA now uses to define (limit) the operational life of civil transport aircraft [18, 19]. The challenge addressed in this paper is to develop a SPD application that, when used in conjunction with the standard practice of using a sealant to stop the environment entering the joint via the gap between the (mating) upper and lower fuselage skins, can both seal the joint and thereby stop
corrosion damage and consequently extend the
time to crack initiation at the joint, and also
reduce the crack growth rate so that the LOV is
significantly increased.

The specimen geometry used in this study
to investigate the use of a SPD doubler to
increase the LOV of the joint is shown in Figure
12. This specimen geometry was developed in
[14], as part of the FAA Aging Aircraft
Program, where it was shown to reproduce the
crack length history seen in Boeing 727 and 737
fleet data [14, 20]. The basic specimen used
consisted of two 2024-T3 clad aluminium alloy
sheets 1.016 mm (0.04 inch) thick, fastened
with three rows of BACR15CE-5, 1000 shear
head counter-sunk rivets, 3.968 mm (5/32 inch)
diameter. The width of the specimen was
chosen to coincide with the typical distance
between tear straps of a B-737 aircraft. Since
the amount of out-of-plane bending in a typical
fuselage joint is an important factor in the
fatigue performance of the joint, the amount of
local bending in the specimen was made similar
to that seen in a typical fuselage joint by testing
the specimens bonded back-to-back and
separated by a 25 mm thick honeycomb core.
This test configuration was crucial in ensuring
that the specimens reproduced fleet behaviour,
see [14]. As in [14] the upper row of rivet holes
contained crack initiation sites, induced prior to
assembly of the joint by means of an electrical
spark erosion technique, on either side of the
rivet holes. These initial cracks were (each)
nominally 1.25 mm long. This crack length was
chosen so that the (initial) defect was obscured
by the fastener head and as such was
representative of largest possible undetectable
flaw size.

![Figure 12. Schematic diagram of the fuselage lap joint specimen, all dimensions in mm, from [14].](image)

As mentioned above the FAA now defines
the fatigue limit of fuselage lap joints, which
they define as the limit of viability or LOV, as
the number of cycles to MSD or MED. The
fatigue performance of the baseline (no SPD)
specimens is documented in [15]. Here it was
found that for specimens without an SPD
modification the number of cycles to first link
up of cracks from adjacent holes occurs at
approximately 30,000 cycles. To illustrate this
and to show the stresses in the baseline joint
Figures 13 and 14 present the stresses in a
(baseline) joint at approximately 6,500 and
29,00 cycles respectively.
Figure 13 Stresses, in MPa, in the joint after ~
6,500 cycles, from [15].

Figure 14 Stresses, in MPa, at approximately
29,000 cycles, from [15].

To illustrate increase the LOV of the joint
a 1 mm thick 7075 SPD doubler was deposited
over the three rows of fasteners, see Figures 15
and 16, and the specimens tested as above. This
test program revealed that after approximately
110,000 cycles the SPD doubler was still intact.
Furthermore, there was no apparent crack
growth at any of the fasteners in the lap joint,
cracking in the SPD or damage to the bond
between the SPD and the skin/fasteners. If we
take the LOV to be the time to first linkup then
this corresponds to more than a 3.3 fold increase
in the LOV, depending on how the LOV is
defined.

Figure 15. Geometry of the test specimen

Figure 16. Close up view of the region with the
SPD covering the specimen.

3 Conclusion

The experimental test program outlined in
this paper has confirmed the potential of SPD
doublers to enhance the damage tolerance of
structural components. We also have established
that fatigue crack growth in SPD repaired
structures can be analysed using existing crack
growth equations and how for fuselage lap
joints an SPD doubler bonded over the fasteners
remains intact with no cracking in the SPD or
degradation to the bond between the SPD and
the structure after more than three times the
LOV of the joint. This finding suggests that a
SPD doubler has the potential to effectively seal
the joint and thereby protect against the onset of corrosion damage. The experimental results also reveal that this approach has the added advantage that it significantly retards crack growth.

Although this study has focused on fuselage lap joints the ability of an SPD doubler to form a durable bond to both the skin and the fasteners means that this approach may well be applicable to other problem areas.

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References

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