SPD REPAIRS TO THIN ALUMINIUM STRUCTURES

R. Jones*, N. Matthews, J. Elston*, K. Cairns*, J. Baker*, B. Wadsley** and S. Pitt***
*** DSTO Centre of Expertise in Structural Mechanics, Department of Mechanical and Aeronautical Engineering, Monash University, Wellington Rd, Clayton, Vic 3800, Australia., ** Rosebank Engineering, 836 Mountain Highway, Bayswater, VIC 315**
rhys.jones@monash.edu, neil.matthews@rosebank-eng.com.au

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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 damage tolerance assessment of both repaired and unrepaired structure should allow for initial defects the size of which is documented in [9], i.e. typically 0.005 inch (1.27 mm) for thin metallic structures. To this end the present paper presents the results of experimental studies whereby SPD is used to repair thin aluminium alloy structures and fuselage lap joint specimens which contain representative initial flaws. We also show how a damage tolerance assessment of SPD repairs can be made using existing crack growth equations.

2 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.

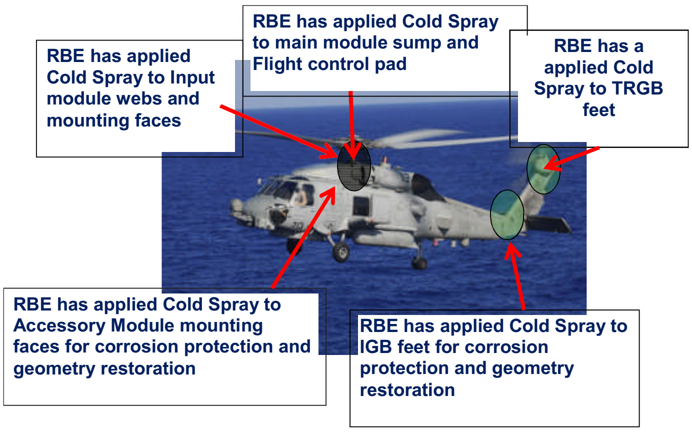

Figure 1 Recent Australian applications of SPD.

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 σ_{max} = 180 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.

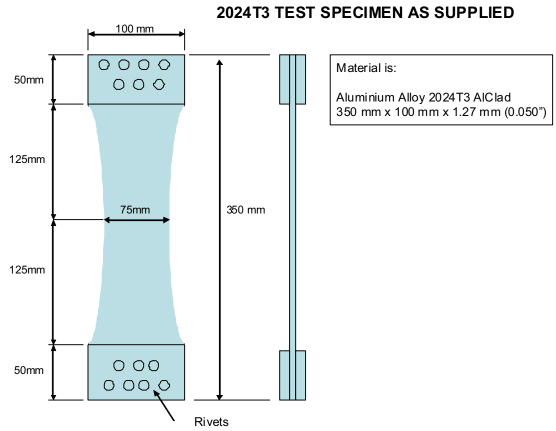

Figure 2: Geometry of the edge notch panel

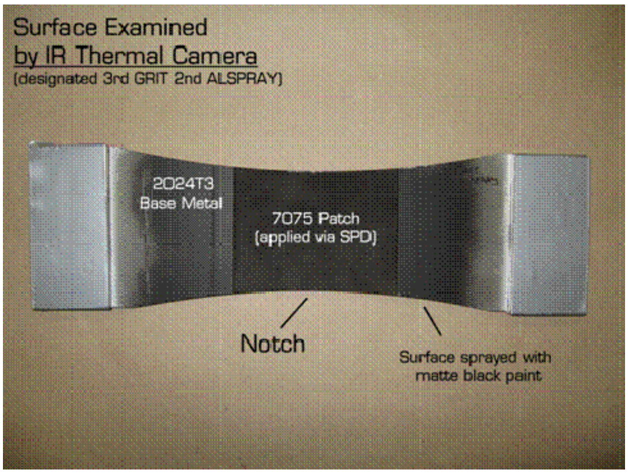

Figure 3: Plan view of the test panel and the SPD doubler

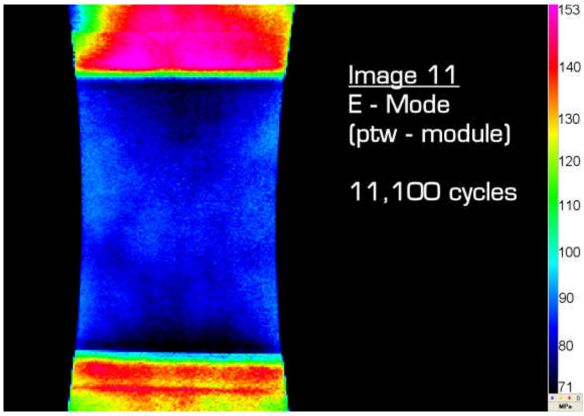

Figure 4 Stresses in the SPD doubler at 11,100 cycles, units are in MPa.

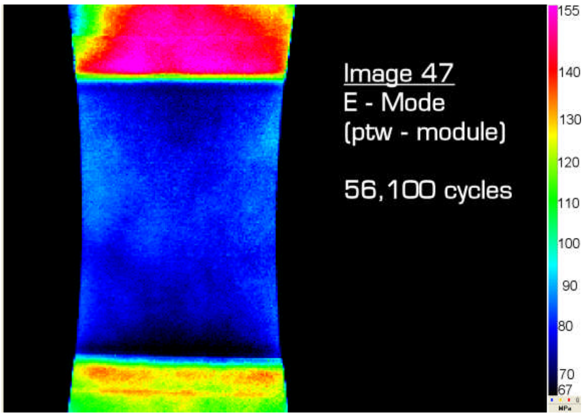

Figure 5: Stresses in the SPD doubler at 56,100 cycles, units are in MPa

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_{max} = 10 kN and R = 0.1 followed by constant amplitude loading with a maximum load P_{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.

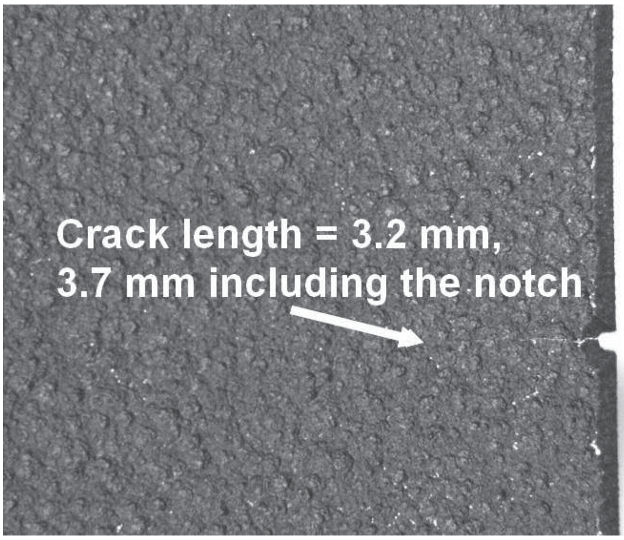

Figure 6 Photograph of the crack in the test.

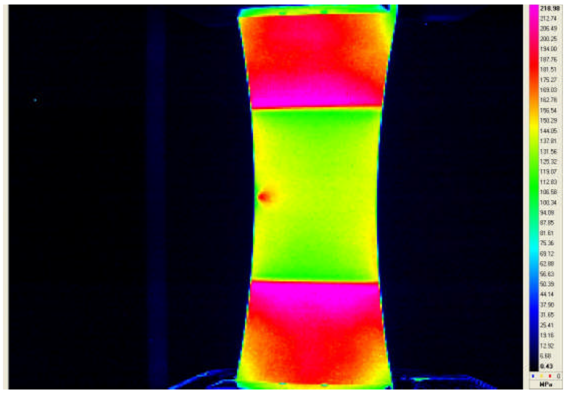
Figure 7: Infra-red picture of the stresses in the specimen at approximately 9,300 cycles

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 σ_{max} = 160 MPa and R = 0.1. In the second test we had a maximum stress σ_{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:

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

where A = 50 MPa √m, and ΔK_{thr} = 0 MPa √m. The value of D given in [11] for this alloy was $1.2\ 10^{-9}$. 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 predicted the crack length after 13,700 cycles for the SPD repaired specimen tested at σ_{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.

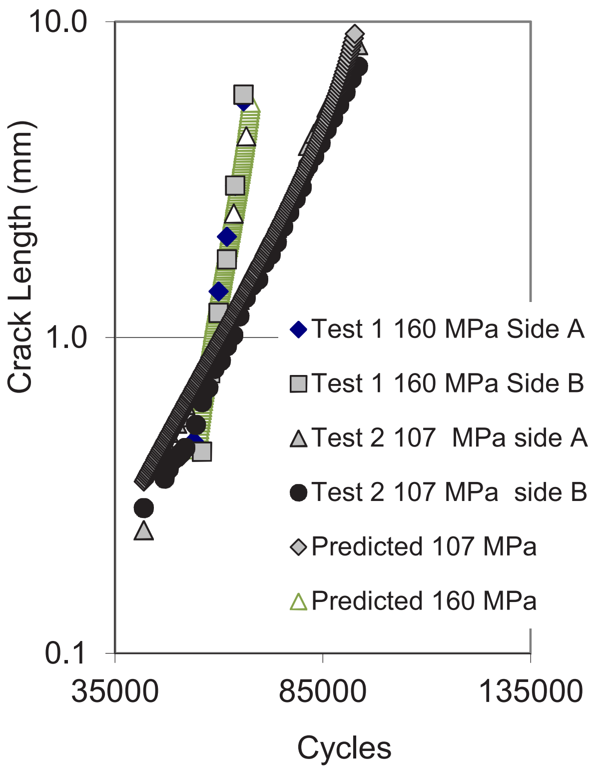

Figure 8 Measured and predicted crack length histories.

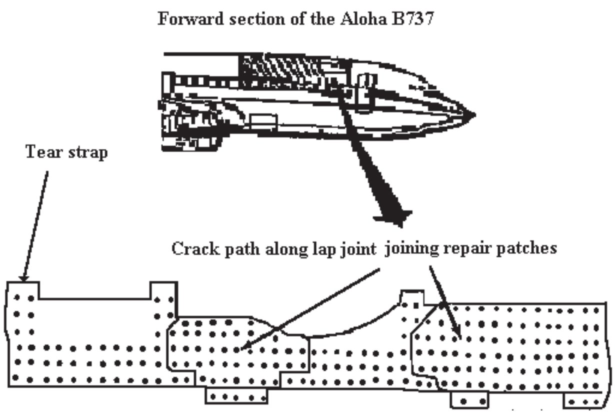

Figure 9 - The dangers of cracks linking from multiple repairs in fuselage lap joints, from [12].

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].

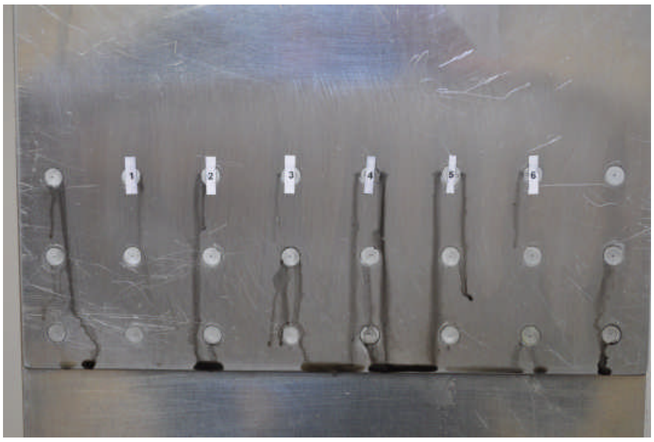

Figure 10 Bleeding of fluid from cracks and rivet heads, from [15].

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.

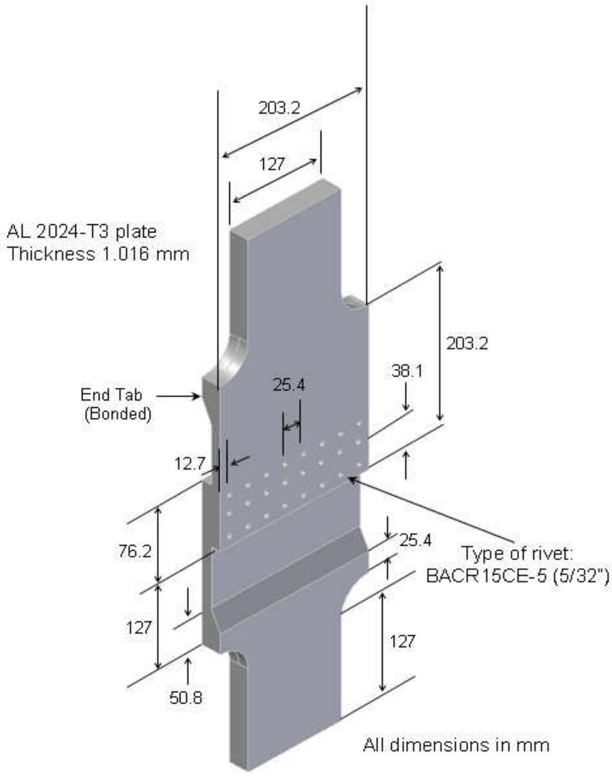

Figure 12. Schematic diagram of the fuselage lap joint specimen, all dimensions in mm, from [14].

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.

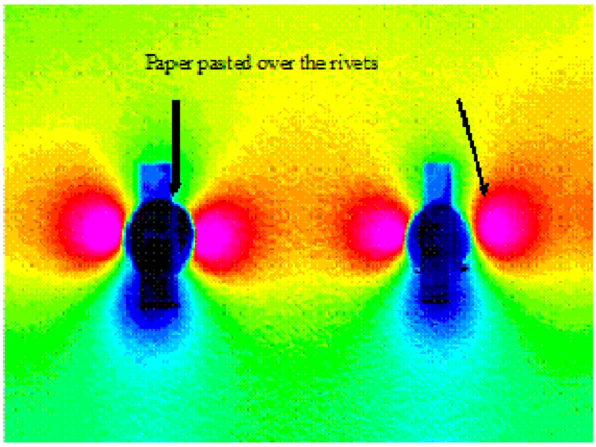

Figure 13 Stresses, in MPa, in the joint after ~ 6,500 cycles, from [15].

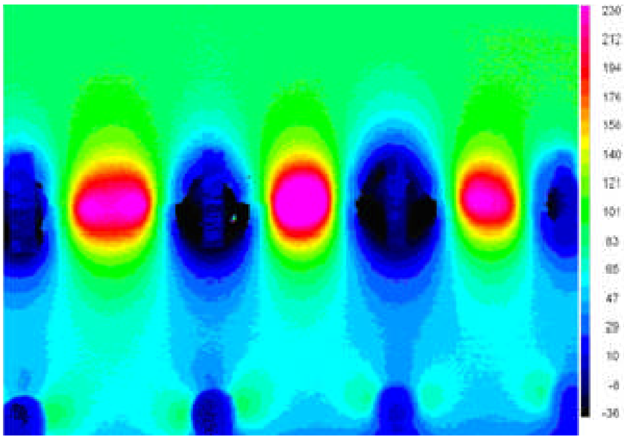

Figure 14 Stresses, in MPa, at approximately 29,000 cycles, from [15].

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.

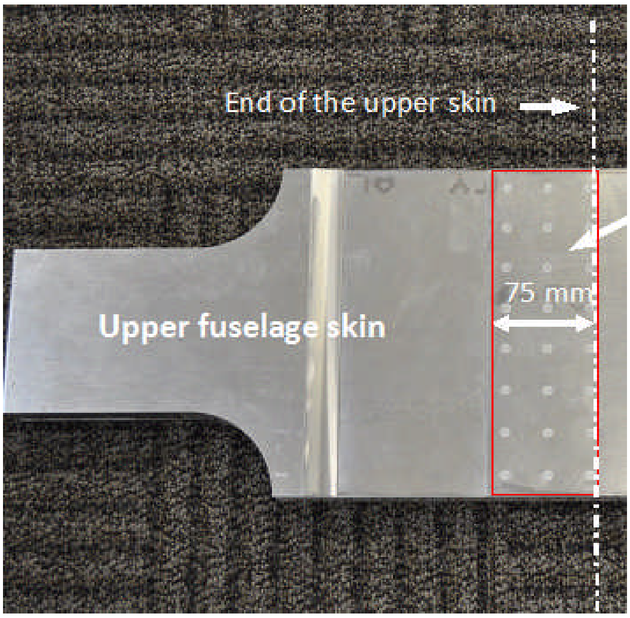

Figure 15. Geometry of the test specimen

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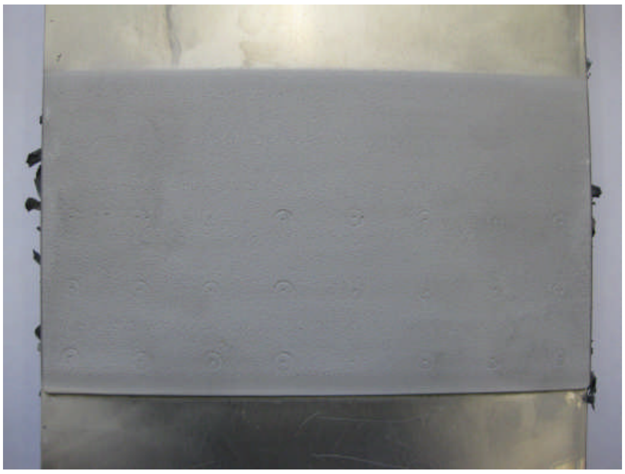

Figure 16. Close up view of theregion with the SPD covering the specimen.

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.

Acknowledgements

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References

[1] Efforts to Reduce Corrosion on the Military Equipment and Infrastructure of the Department of Defense, Office of the Secretary of Defense, USA. Department of Defense Report, June 2007.

[2] Jones R., Matthews N., Rodopoulos CA., Cairns K. and Pitt S., On The Use Of Supersonic Particle Deposition To Restore The Structural Integrity Of Damaged Aircraft Structures, Int. Journal of Fatigue, doi:10.1016/j.ijfatigue.2011.03.013.

[3] Villafuerte, J, Current and future applications of cold spray technology, Recent Trends in Cold Spray Technology: Potential Applications for Repair of Military Hardware, pp 1-14, NATO RTO-MP-AVT-163, 2010. Available via ftp://ftp.rta.nato.int.

[4] Matthews N., Supersonic Particle Deposition (SPD) Cutting Edge Technology for Corrosion Protection and Damaged Metallic Component Recover, Proceedings 2010 SDE Symposium, 24th-25th March 2010, RAAF Williams, Melbourne, Australia. www.defence.gov.au/dgta/WebPages/DAVCOMP/SDE/sde%20Symposium.html.

[5] Jones R., Pitt S., Matthews N., Baker J. and Cairns K., Cold Spray: A Means For Enhancing/Ensuring Aircraft Structural Integrity, Proceedings 1st Australasian Cold Spray Conference, Melbourne, 16-18th May, 2011.

[6] Mathews N., Latest Cold Spray Applications In Australian Aerospace Industries - N. Matthews, Proceedings 1st Australasian Cold Spray Conference, Melbourne, 16-18th May, 2011.

[7] Jones R, Krishnapillai M., Cairns K. and Matthews N., Application of infrared thermography to study crack growth and fatigue life extension procedures, Fatigue and Fracture of Engineering Materials and Structures, 33, 12, pp 871–884, December 2010.

[8] USAF Damage Tolerance Design Handbook: ASIST, Wright Patterson AFB, November 2002.

[9] Federal Aviation Administration Advisory Circular, 25.1529-1, “Instructions for Continued Airworthiness of Structural Repairs on Transport Airplanes”(1991).

[10] Federal Aviation Administration Advisory Circular, 25.571-1A, “Damage Tolerance and Fatigue Evaluation of Structure” (1986).

[11] Jones R., Molent L. and Walker K., Fatigue crack growth in a diverse range of materials, Int J. of Fatigue, Int J Fatigue (2012), doi:10.1016/j.ijfatigue.2012.01.004 .

[12] Jones R, Pitt S., Brunner AJ. And Hui D., Application of the Hartman-Schijve equation to represent Mode I and Mode II fatigue delamination growth in composites, Composite Struct (2011), doi:10.1016/j.compstruct.2011.11.030.

[13] Jones R., Tan J., Peng D. and Pitt S., Damage tolerance analysis of a helicopter structural component, Proceedings Australian Aircraft Airworthiness and Sustainment Conference 2011, Australia, Brisbane, Australia, 26th-28th July, 2011.. available on line at: www.ageingaircraft.com.au/files/.../Wednesday/P3/.../s7%20p3.ppt

[14] Jones R. and Molent L, Chapter 16, Repair of Multi-site Damage ,A. Baker, Advances in the Bonded Composite Repair of Metallic Aircraft Structure , Edited by L. R. F. Rose and Jones R., Elsevier Applied Science Publishers, 2002. ISBN 0-08-042699-9

[15] Jones R. Cairns K., Baker J., Krishnapillai K. and Hinton B. , A study of the effect of CPCs on fatigue crack propagation in a representative fuselage lap joint specimen, Engineering Fracture Mechanics, doi: 10.1016/j.engfracmech.2011.11.015.

[16] http://www.dailymail.co.uk/news/article-1374574/Southwest-Airlines-fully-operational-plane-cracks-repaired.html

[17] http://www.bbc.co.uk/news/world-us-canada-12954335

[18] FAA-2010-1280-001, Federal Register, Vol 76, 6, Monday January 2011, Rules and Regulations.

[19] Eastin RG, Sippel W., The "WFD rule" - have we come full circle?, Proceedings, 2011 USAF Aircraft Structural Integrity Conference, 29th November- 1st December 2011, San Antonio, Texas, USA,

[20] Molent L., Jones R., and Pitt S. Understanding crack growth in fuselage lap joints, Theoretical and Applied Fracture Mechanics, 49, (2008), pp 38-50.

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