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Application of supersonic particle deposition to enhance the structural integrity of aircraft structures

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Aircraft metal components and structures are susceptible to environmental degradation throughout their original design life and in many cases their extended lives. This paper summarizes the results of an experimental program to evaluate the ability of Supersonic Particle Deposition (SPD), also known as cold spray, to extend the limit of validity (LOV) of aircraft structural components and to restore the structural integrity of corroded panels. In this study the potential for the SPD to seal the mechanically fastened joints and for this seal to remain intact even in the presence of multi-site damage (MSD) has been evaluated. By sealing the joint the onset of corrosion damage in the joint can be significantly retarded, possibly even eliminated, thereby dramatically extending the LOV of mechanically fastened joints. The study also shows that SPD can dramatically increase the damage tolerance of badly corroded wing skins.

cold spray, fuselage joints, corrosion, LOV

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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 the 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 neighbouring locations, a phenomenon which 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 [2–5]

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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 [6–8] 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 [7] and AC No: 25.571-1A [8], Mil-HDBK 130 and the USAF Damage Tolerant Design Handbook [6] 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 refs. [6–8] the damage tolerance assessment of both repaired and unrepaired structure should allow for initial defects, the size of which is documented in ref. [6], 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/seal fuselage lap joint specimens which contain representative initial flaws.

This paper also examines the potential of how SPD can be used to restore the structural integrity of badly corroded aircraft structures and how this approach can be used to overcome the need to repair corrosion by using mechanically fastened repairs which involve drilling holes, which act as stress concentrators, in the base structure and which also act as sites at which corrosion pits can develop and subsequently crack.

2 What is SPD

SPD is an additive process in which metal particles in a supersonic jet of an expanded gas impact a solid surface with sufficient energy to cause plastic deformation and bonding with the underlying surface so that the powder is reconstituted into a solid without the creation of heat affected zones which are undesirable in many structural applications. SPD is an approved Military Standard process for Powder Deposition [9] and has been accepted by Original Equipment Manufacturers, Military Regulatory Authorities and FAA for limited applications.

SPD has been applied experimentally and demonstrated its ability to enhance fatigue life of thin skins with a pre-existing defect [2]. In one test SPD was applied over centrally notched 1.27 mm thick 2024-T3 clad aluminium alloy dogbone specimen subjected to constant amplitude loading with δ_{max} = 180 MPa and $R = \sigma_{min}/\sigma_{max} = 0.1$. The baseline specimen, i.e. without a doubler, lasted approximately 35000 cycles. In contrast the SPD patched panel test was stopped after approximately 60000 cycles with little, i.e. no evident, damage in the 7075 SPD or crack growth in the 2024-T3 skin. In another test the 2024-T3 the specimen was first loaded so as to grow a sharp crack. This first phase of the test was stopped at 18886 cycles when the crack length was approximately 3.2 mm. A 10 mm wide and 1 mm thick SPD strip was then deposited and the test was continued. SPD strip significantly reduced the crack growth rate (By a factor of approx 3). Similar results were obtained on 7050-T7451 aluminium alloy specimens.

With the ability of SPD to enhance the structural integrity of thin skins illustrated, the ability of an SPD doubler to extend the fatigue life of mechanically fastened joints and in particular fuselage lap joints was evaluated. The 1988 Aloha accident, where cracking in the joint ran from one repair to another, (see Figure 1 from ref. [10]), 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 ref. [11] this does not stop the environment entering the joint through the fasteners. 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 [11].

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 (1.524 m) hole in the roof (see Figure 2). This incident led to the grounding of 79 of its older Boeing 737 aircraft [12] and to the cancelation of almost 700 flights. Subsequent inspections, which found cracks in a total of four Southwest aircraft [12], led to the US FAA mandating the inspection of 175 Boeing 737 aircraft that had seen more than 35000 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 [13]. This aircraft had experienced less than 23000 cycles. This led to

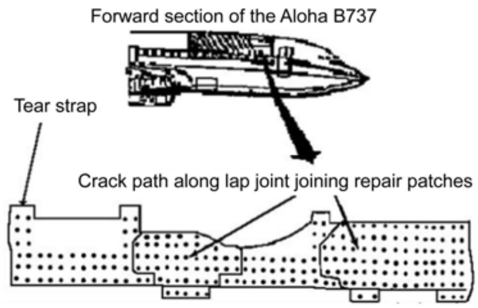

Figure 1 The linking from multiple repairs in the Aloha fuselage lap joint, from ref. [10].

Figure 2 Tarpaulin covering the five-foot-hole that ripped open roof, from ref. [12].

the discovery of cracking in other 757 aircraft and a subsequent January 2011 FAA Airworthiness directive [13] mandating the inspection of all 757-200 and 757-300 aircraft.

As a result of these incidents the FAA has introduced the concept of a limit of viability (LOV), defined as the onset of multi-site and/or multi-element damage [14,15], which the FAA now uses to define (limit) the operational life of civil transport aircraft [14,15]. 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 seal the joint and thereby alleviate corrosion damage and consequently extend the time to crack initiation at the joint so that the LOV is significantly increased.

The lapjoint specimen geometry used in this study is shown in Figure 3. This specimen geometry was developed 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 [10,11]. 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 (see Figure 3). 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 [10,11]. This test configuration was crucial in ensuring that the specimens reproduced fleet behaviour [10]. As in ref. [10] 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. Of the eight fastener holes in the specimen only the inner six were cracked [10,11]. The specimens were tested under constant amplitude loading, with the maximum and minimum loads as detailed in Table 1, i.e. P_{max}=40 kN and P_{min}=2 kN. These loads were determined from operational data obtained for the US DoT MSD Committee Review Board for the B-737 aircraft [10,11], and a stress picture showing the stresses in the baseline specimens is presented in Figure 4 and a stress picture just prior to link up of MSD is shown in Figure 5 where an increase in the stress in the remaining ligament and that just prior to link up, the high stress regions associated with each crack link can be seen. The latter feature, i.e. the link up of the high stress regions, and the shape of the stress field shown in Figure 5 is an important feature that will be used to assess link up throughout this paper.

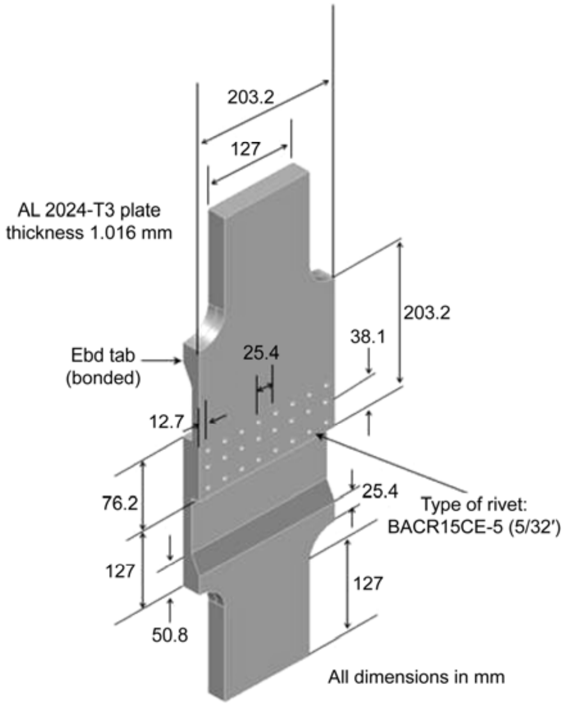

Figure 3 Schematic of a fuselage lap joint skin, all dimensions in mm.

In illustrating how SPD can increase the LOV a 1 mm thick 7075 SPD doubler was deposited over the fasteners (see Figure 6) and a total of four lap joints were tested. The test was subsequently repeated with a thin, (nominally) 0.3 mm thick, SPD doubler/coating located as per Figure 6 and

Table 1 Test conditions

P_{max} (kN)	P_{min} (kN)	P_{mean} (kN)	Test frequency (Hz)
40	2	21	5

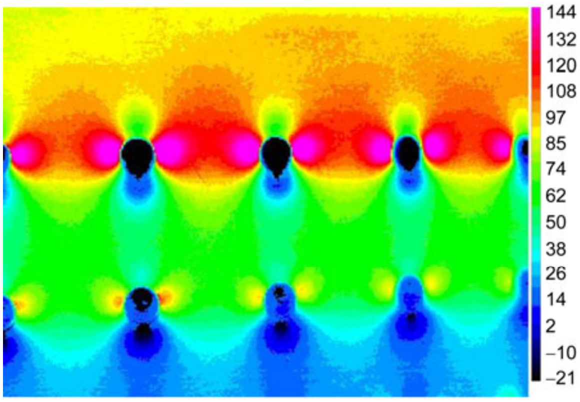

Figure 4 (Color online) The stresses, in MPa, at the critical rows of fasteners.

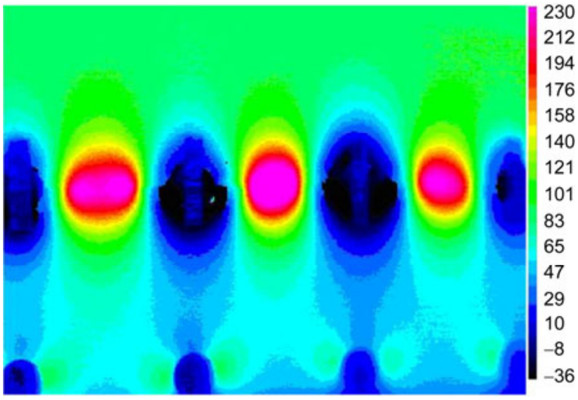

Figure 5 (Color online) Stresses, in MPa, prior to link up, [11].

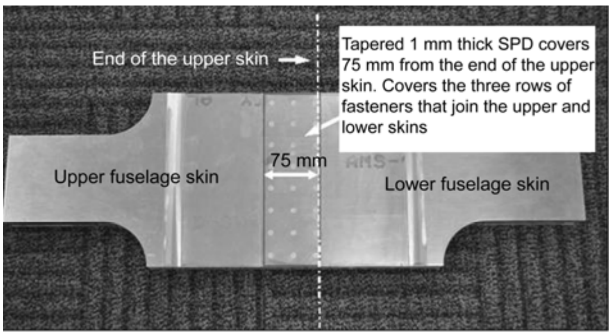

Figure 6 Fuselage lap joint specimen showing location of the SPD.

as previously a total of four lap joints were tested. In all cases the fasteners could be clearly seen through the SPD (see Figure 7). The stress picture obtained at the beginning of the tests (Figure 8) was almost identical to that seen for panels without an SPD protective layer. The first panel failed at approximately 110000 cycles. There was no failure in the SPD until after the link up of multi-site damage in the lap joint to produce a large crack that was approximately 50 mm tip to tip. The SPD over this large crack failed just prior to complete failure of the specimen. Following failure the thickness of the SPD coating was measured and found to lie between 0.2 and 0.3 mm.

By 91000 cycles the cracks at rivets 4 and 5 had almost joined, (see Figure 9) where (near) joining of the crack tip stress fields can be seen. However, at this point in the test there was no sign of cracking in the SPD. By 103000 cycles the crack between rivets 4 and 5 had grown significantly and the stresses in the SPD covering this region had increased significantly. By 104000 cycles the cracks had

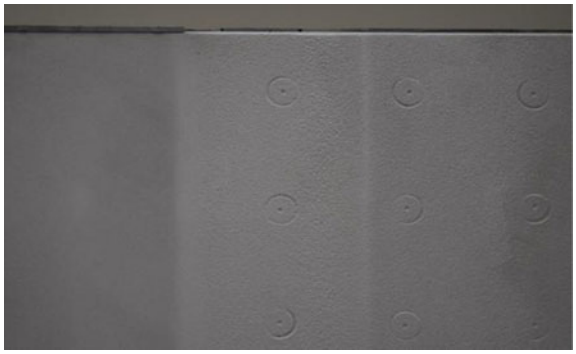

Figure 7 SPD doubler over the fasteners.

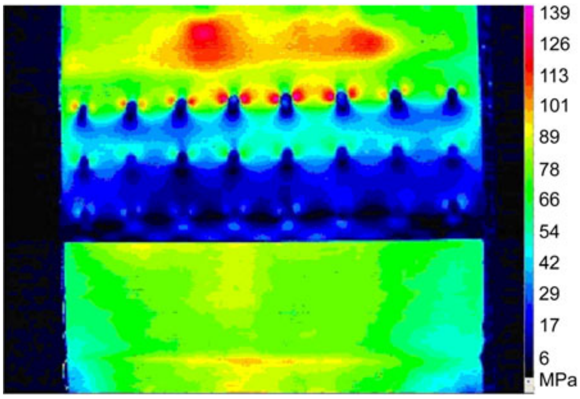

Figure 8 (Color online) Initial stresses in the SPD.

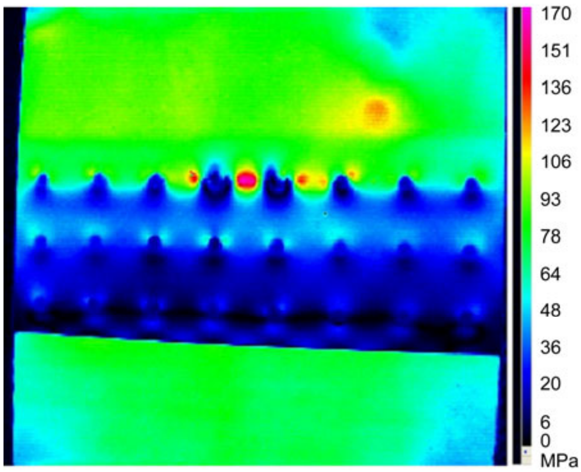

Figure 9 (Color online) Stresses in the SPD at 91000 cycles.

joined and the SPD over this (now) large crack had failed, (see Figure 10). Figure 10 also shows stress concentrations associated with the two (new) crack tips and low stresses between these two tips as a result of load flowing around the cracks in the SPD and the lap joint. Final failure of the panel occurred at approximately 113000 cycles.

Similar results were obtained for the other lap joints. This test program again reveals that there is no cracking in the SPD until after there is significant level of MSD. As such the SPD can be seen to have effectively sealed the fasteners up to the onset of linkup. As previously mentioned the test specimens were fabricated from two fuselage lap joints that were tested back to back. As such it was found that failure generally only occurred on one of the panels. It was also found that there was never any cracking associated with the SPD on the (non-failed) panel. This empowered the panel that had not failed to be disassembled and checked for cracks. As a result it was found that cracks of up to 6 mm could be tolerated in the fuselage skin without crack of the SPD (see Figure 11).

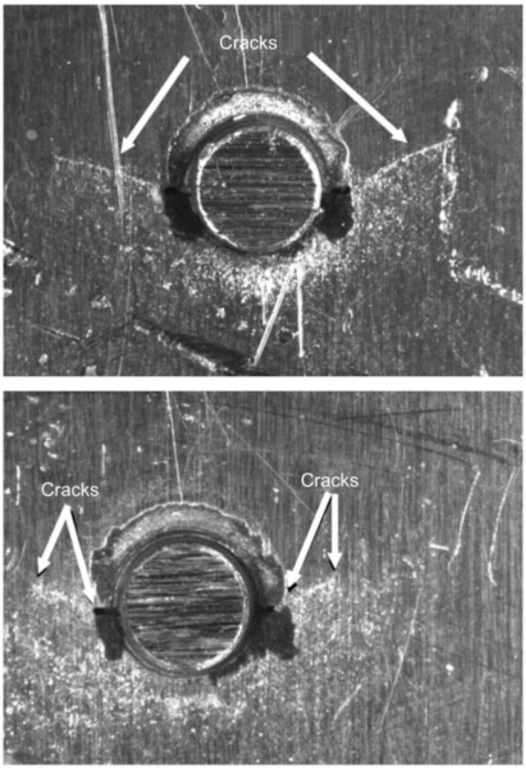

Figure 11 Examples of cracking in the fuselage skins which did not result in cracking of the SPD.

3 Repairs to corrosion reworks

The Alohoa accident also highlighted the problem of multiple interacting repairs, (see Figure 1) which reveals that despite the presence of MSD in the fuselage lap joint the failure ran from corrosion repair to corrosion repair. Indeed, for Boeing 727 aircraft [16,17] there are numerous instances when there was no crack growth until after a corrosion repair had been installed. The problem of multiple interacting repairs to corrosion damage is not confined to corrosion in fuselage lap joints (see Figure 12). The common approach to corrosion damage in operational aircraft is to cut out the corrosion and rivet a mechanical doubler over the region (see Figures 1 and 12). Unfortunately if the aircraft is operated in an aggressive environment, then corrosion is likely to occur over a (relatively) broad area and this can lead to a number of mechanical repairs that lie in relatively close proximity, (see Figures 1 and 12). This repair process

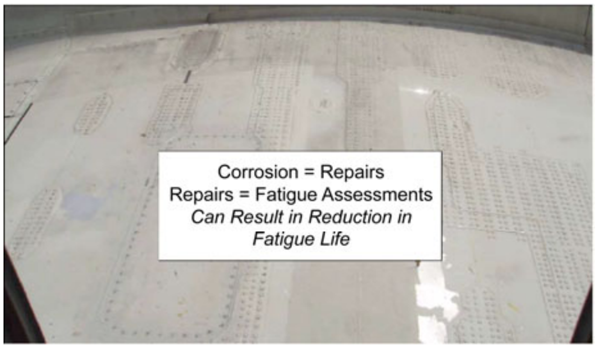

Figure 12 Multiple corrosion repairs on a RAAF P3C aircraft. This slide is by courtesy of Sqn Ldr Dorman RAAF DGTA.

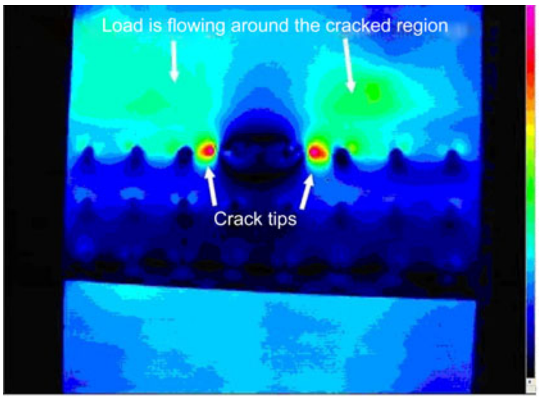

Figure 10 (Color online) Stresses in the SPD at 104000 cycles.

involves drilling holes, which act as stress concentrators, in the base structure and unless the operational environment changes these holes now provide sites at which corrosion pits can develop and subsequently crack as was seen in the Aloha accident (see Figure 1).

As such a repair methodology is needed whereby the structure need not be further damaged and new sites for pitting and subsequent cracking are created. As such the use of SPD to repair corrosion damage is one possible alternative. To study this approach we tested 2 mm thick, 400 mm long and 42 mm wide 7075-T6 aluminium alloy specimens. The specimens contained 0.3 mm deep by 50 mm long by 42 mm wide corrosion cutout, also commonly referred to as a corrosion blendout (see Figure 13 (a)). For simulating a small corrosion pit that was not removed by the corrosion blendouts containing a small 0.2 mm deep notch that ran the full width of the specimen, several specimens were repaired using SPD and a crossection of the geometry of the repaired specimens is shown in Figure 13 (b). The unrepaired specimens were subjected to constant amplitude tests at a peak stress of 140 MPa and R = 0.1 at a test frequency of 5 Hz whilst SPD repaired specimens were tested at both 140 and 160 MPa at a test frequency of 5 Hz, at R = 0.1. The tests were performed at room temperature in laboratory conditions.

The three baseline specimens, i.e. without an SPD repair, when tested at 140 MPa lasted approximately 36800 cycles. This contrasts with that in excess of 15000000 cycles for the three SPD repaired specimens when tested at the same (i.e. 140 MPa) stress level. (The SPD repaired specimens did not fail and there was no evidence of cracking in the SPD or in the baseline specimen, (see Figure 14). Hence the tests were stopped after 15000000 cycles. Two SPD repaired specimens were subsequently tested at 160 MPa. These specimens failed at approximately 640000 cycles and 1330000 cycles respectively due to small initial defects induced during removal of the surface material in attempting to simulate a corrosion blendout. In all cases the fatigue lives of the SPD repaired specimens were dramatically greater than those seen for the baseline corrosion blendout specimens. All tests were performed in room temperature and laboratory conditions.

4 Conclusion

This test program has established the damage tolerance of SPD coatings to fuselage lapjoints in that the underlying thin 2024-T3 skin will experience crack growth prior to crack growth/failure in the SPD. Furthermore, the presence of small, up to 6.5 mm long, cracks in the underlying fuselage joint skin did not result in cracking in the SPD. This is

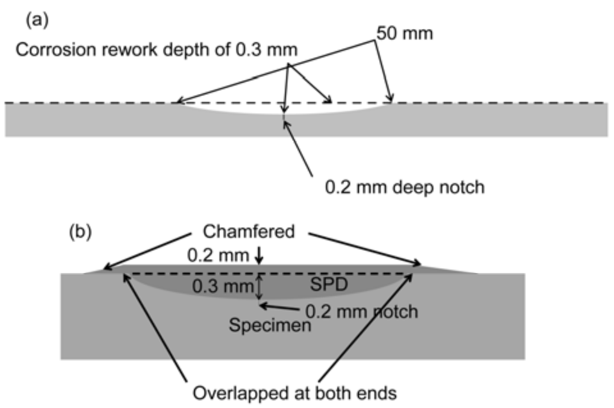

Figure 13 (a) Crossection of the simulated corrosion test specimen; (b) crossection of the SPD repair.

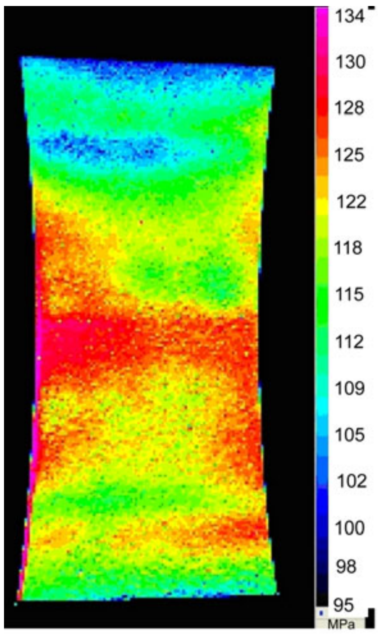

Figure 14 (Color online) Stress distribution on the surface of the SPD repair after approximately 10000.000 cycles.

important because it establishes the potential of the SPD to seal the joint, i.e. not to crack, and for this seal to remain intact even in the presence of MSD. By sealing the joint we have the potential to significantly retard, possibly even eliminate, the onset of corrosion damage in the joint and thereby dramatically extend the LOV of fuselage lap joints.

It is now a requirement that all structural modifications/repairs be assessed for their effect on the damage tolerance of the structure. In this process it must be assumed that the structure contains a small initial defect, which at a fastener hole in a thin skin is mandated to be 1.27 mm (0.05 inch) [6]. This test program has shown that the growth of such small defects, (the lap joint contained small 1.25 mm initial defects at the critical row of fasteners), does not compromise the integrity of the SPD and as such addresses questions as to the damage tolerance of the SPD itself.

This study has also shown that SPD can be used to repair corrosion damage and that the resultant repaired structure has a dramatically increased fatigue life. In this fashion repair to corrosion damage can be achieved without the need to introduce new holes, which act as potential sites for both corrosion and cracking, and without the need to parasitically stiffen (i.e. locally over stiffen) the structure and thereby change the load paths as a result of increasing the local stiffness of the region and as a result of adding a mechanical patch.

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