DAMAGE TOLERANCE OF AIRCRAFT PANELS

P. M. S. T. de Castro[1], S. M. O. Tavares[1], V. Richter-Trummer[1], P. F. P. de Matos[2], P. M. G. P. Moreira[3], L. F. M. da Silva[1]

[1]Faculdade de Engenharia da Universidade do Porto, IDMEC-FEUP, Porto, Portugal
[2]IDMEC-FEUP, e ISVOUGA, Santa Maria da Feira, Portugal
[3]INEGI, Porto, Portugal






ABSTRACT

The basic concepts of damage tolerance criteria for civil aircraft are briefly reviewed. As a result of the traditional usage of riveted joints in the aluminium alloy fuselage of civil aircraft, one advantage of this type of joints is the existing experience concerning their design and maintenance. When subjected to cyclic loading, riveted joints suffer fatigue damage, including multiple site damage - MSD. Thus a second item of this presentation will concern modeling of MSD and of residual strength in riveted structures. Means to improve the strength of those joints will be presented and relevant results will be discussed. Alternatives to riveting are being considered aiming at economies in fabrication time, cost and weight. One such alternative is welding, particularly laser or friction stir welding (LBW or FSW). However, open issues concerning the use of integral structures in aeronautics include the damage tolerance problem, since the integral nature of the structure provides a continuous path for crack growth. The third topic of the presentation will address the fatigue behaviour of integral stiffened panels focusing on the influence of residual stress fields. Modelling fatigue crack growth requires the knowledge of stress intensity factor solutions for the relevant structural geometry, loading and crack configuration. Results obtained using the virtual crack closure technique - VCCT and ABAQUS finite element package are presented. Residual stresses play an important role in the behaviour of integral structures. Therefore, the fatigue crack growth behaviour of stiffened panels was modelled using modified stress intensity factor solutions that take into account the residual stresses and appropriate fatigue crack propagation laws. The presentation concludes with remarks on open issues requiring further research before a more widespread usage of integral structures is made by aircraft manufacturers.

1 - INTRODUCTION

After presenting current damage tolerance concepts, this paper discusses the behaviour of riveted and welded Al reinforced panels. Cold expansion as a means of improving fatigue strength in open hole specimens is presented, and recent efforts concerning the use of welded joints, including the EU project DATON, are discussed. Residual stresses will be a recurrent subject of this paper, firstly mentioned in the context of the Comet accidents, and later on discussed in the context of modelling the behaviour of cold expansion and of crack propagation in stiffened panels.

2 - DAMAGE TOLERANCE

As in many other engineering specialties, aircraft accidents may be sources of knowledge contributing to the advancement of the field. Organizations as the USA National Transportation Safety Board (NTSB) document in detail aircraft failure analyses. Concise presentations of some particularly exemplary cases are given by Wanhill (2003), namely the Comet losses and the Aloha Airlines Boeing 737-200 1988 failure, both mentioned here because of their importance in the advancement of design criteria. Comet I, developed by the UK Havilland Aircraft Company was the first commercial jet. Designed for high altitude service, it entered service in 1951. In 1954 tragic accidents occurred with planes with 1286 and 903 flights, originated by fatigue crack propagation leading to the disintegration of the pressurized fuselage. Failure analysis revealed that although designed and tested for the conditions found in service, the design was defective as concerns crack arrest capability. A second problem concerned the tests carried out by the manufacturer, (Schijve, 2009, Brot, 2008): due to economic reasons, a fuselage that was subjected to pressure testing up to ultimate design load, was subsequently fatigue tested. It is now understood that the results of the fatigue test were improved by compressive residual stresses originated by the previous pressure proof loading: indeed, compressive residual stresses retard crack propagation and as a consequence the manufacturer had no indication of the real structure weaknesses in service.

The Comet catastrophes indicated the need for a design with crack arrest capability and contributed decisively towards a new design philosophy based upon damage tolerance.

This paper deals with this topic linking these historical facts with the current research on alternative joining solutions, cheaper and faster than riveting, like FSW or LBW, and identifying possible fatigue crack propagation problems associated with these current developments.

Also yielding important lessons concerning fatigue behavior was the 1988 accident of the Aloha flight 243 (NTSB, 1989). At 7300*m* altitude, part of the fuselage of the Boeing 737 was suddenly detached from the aircraft, with sudden decompression of the cabin. Upper fuselage loss was up to 5,5*m* long extending from the aircraft main entrance door. Failure analyses identified multiple site damage along a substantial portion of a riveted joint as a key contribution to the accident.

Together with increasing recognition of fatigue phenomena, aircraft were initially designed with the intention of avoiding fatigue crack initiation during their life-time. The Comet accidents and progress of fatigue knowledge lead to a damage tolerant approach, where damage is considered unavoidable and measures are taken for its control along the aircraft life cycle. This leads to weight savings but also to increased maintenance costs, particularly those related with periodical inspections; however, the balance between the benefit of a lighter structure, and cost increase due to periodical inspection, is positive from the direct operating cost viewpoint, Boller and Buderath, 2007.

The adopted design philosophies are:

- *'fail-safe design'* where it is assumed that the component or structure may be safely operated even in the presence of some damage, that may grow up to a limit value before required component or structure replacement. In these structures a possible crack may grow in service, but will not reach critical size before its detection. Useful life is defined by a critical crack size (defined by the material toughness or other applicable criterion). Essential ingredients of these approaches are the knowledge of crack propagation as related with applied loading, and periodical inspections with a frequency ensuring that undetected damage in one inspection will not grow up to critical size before next inspection.
- *'safe-life design'* seeking to ensure that the component or structure will not develop fatigue cracks in service. This design philosophy is therefore based upon fatigue crack initiation avoidance during lifetime.

The concept of safety factor in safe life design or in damage tolerant design is different. The first implies evaluations of uncertainties associated with life estimation, whereas the second implies monitoring processes such as to ensure continued structural integrity, Bristow and Irving, 2007.

Safe life is used in components that cannot be duplicated, as landing gear; fail-safe is used for major structural components as fuselage. Fail-safe implies the definition of tolerable damage in the aircraft. The resistance of the structure may decrease below ultimate loads but not below design limit loads, defined as "JAR 25.301 loads" "*…(a) Strength requirements are specified in terms of loading conditions that give rise to limit loads (… maximum loads to be expected in service) and ultimate loads (… limit loads multiplied by prescribed factors of safety)*", (Bristow and Irving, 2007, JAR, undated). Figure 1 presents schematically this concept.

With damage tolerant aircraft, inspection intervals are defined on the basis of acceptable damage propagation. Starting from an initial defect size corresponding to the greater undetectable size using conventional inspection tools, the number of cycles up to critical dimensions is estimated.

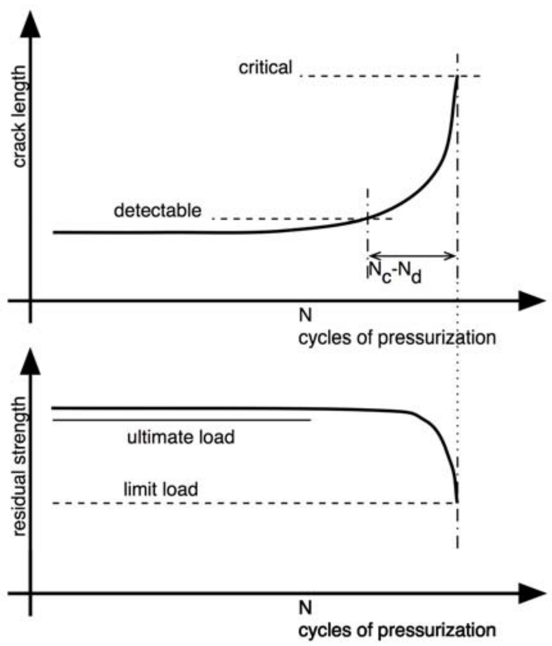


Fig 1 – Damage tolerance concept.

The corresponding time interval is then divided by a suitable safety factor, leading to the selected interval between inspections, see Figure 1, where the inspection interval would be $(N_c\text{-}N_d)/FS$, and FS is the chosen safety factor.

The already mentioned flight 243 accident underlined the need to take into account initiation in more than one location, *ie* multiple site damage (MSD), and lead to a revision of the procedure presented above, taking into account shorter inspection intervals, given the consequent lower number of cycles between damage detection and critical conditions, (Boller and Buderath, 2007, Schijve, 1995), see Figures 2 and 3.

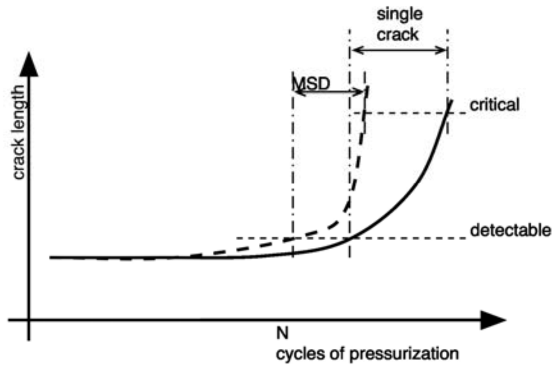


Fig 2 – Shorter life under MSD conditions.

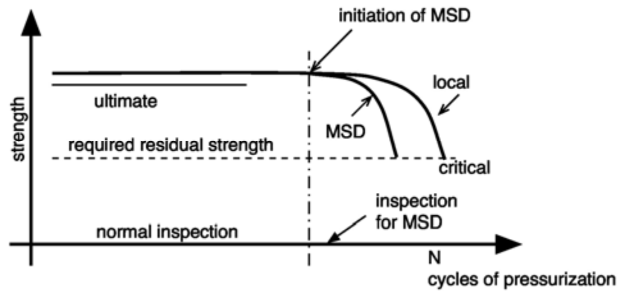


Fig 3 – Effect of MSD.

3 - RIVETTED JOINTS

Riveting is so far the standard method of joining Al fuselage parts. This is likely to change, given the current interest of aircraft manufacturers in composite fuselages for some aircraft sizes (*eg.*, Boeing 787), but also to efforts to find less costly Al solutions using welding technologies. Nevertheless, according to ATAG, 2008, some 2,000 airlines around the world operate a total fleet of 23,000 aircraft; these approximate figures underline the importance of riveting technology and associated maintenance practices, since most of these riveted aircraft are expected to still have long useful lives. As a very first approximation, the open hole specimen subjected to remote tensile cyclic loads gives relevant information on riveted joints behaviour. It allows, for ex., to identify the importance of fatigue improvement technologies as cold expansion, see Figure 4 taken from de Castro *et al*, 2007. Nominal cross section was 25mmx2mm, with nominal hole diameter 5mm.

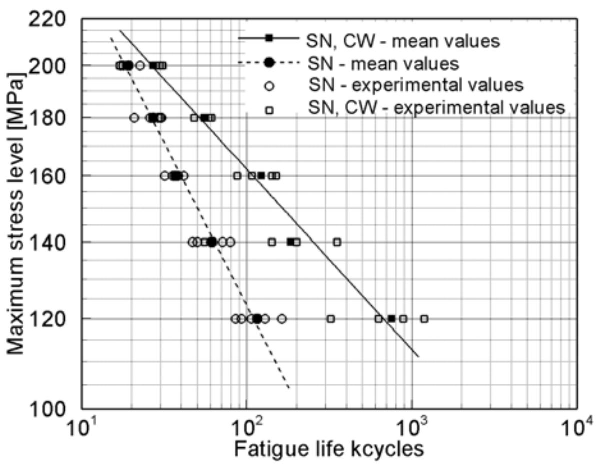


Fig 4 - AA2024-T3 Alclad: 24 specs. with and 21 specs. without residual stresses) tested at $R = \sigma min/\sigma max = 0.1$.

Cold expansion of rivet holes consists of expanding a hole (usually 3–5%) using a mandrel. This creates an annular region of plastically deformed material, which gives rise to a residual compressive stress field when the mandrel is removed. The resulting residual stress field improves fatigue performance by delaying nucleation and retarding crack propagation (de Matos *et al*, 2005, 2006, de Matos, 2008, de Matos and Nowell, 2008, and Pasta, 2007).

Since the pioneering work of Elber, 1970, it has been recognized that plasticity-induced crack closure plays an important role in fatigue crack propagation. This phenomenon is caused by plastically deformed material from the crack tip region which is left along the crack faces as the crack grows. As a consequence, the crack faces contact each other and the resulting contact stresses reduce the effective stress intensity factor at the crack tip and therefore the rate of propagation. As mentioned above, the cold-expansion process also creates a region of plastically deformed material. As a crack propagates through this region, the amount of plastically deformed material left along the crack faces is expected to differ from that for a crack growing in virgin material.

In his DPhil thesis supervised by D. Nowell at the Univ. of Oxford, de Matos, 2008, undertook analytical and numerical modelling of plasticity-induced fatigue crack closure in cold-expanded holes, comparing fatigue crack growth data from analytical predictions from a strip yield model (Figure 5) with experimental data.

Figure 6 shows typical fractures surfaces obtained after fatigue testing specimens with cold expanded holes.

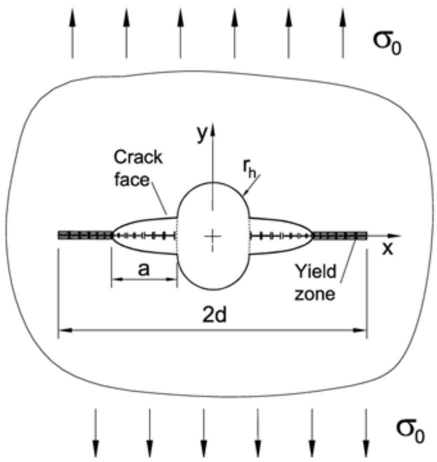


Fig 5 - Infinite plate with a centra hole and two symmetric cracks under uniform remote tension. Strip yield model, de Matos, 2008.

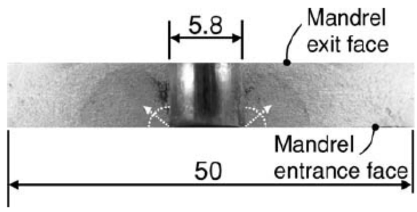


Fig 6 – Fracture surface of 4% cold expanded hole, material 5083-H321, after Pasta, 2007.

Figure 7 presents the circumferential residual stresses modelled using 2D FE analyses. By the residual stress field from Figure 7 nd the yield strip model from de

Matos and Nowell, 2008, the opening stresses are as shown in Figure 8.

Using the opening loads shown in Figure 8 and a fatigue crack propagation law, Paris law in the present work, fatigue lives were predicted based on the effective stress intensity factor range $da/dN = A(\Delta K_{eff})^m$. Figure 9 shows the comparison between experimental fatigue crack growth data and analytical prediction from the yield strip mo-del. It can be seen that model predictions are in good agreement with experimental data.

Fatigue crack propagation from cold expanded holes is clearly a 3D problem and 2D analyses will always be approximate. However, they may have a part to play in practical design calculations.

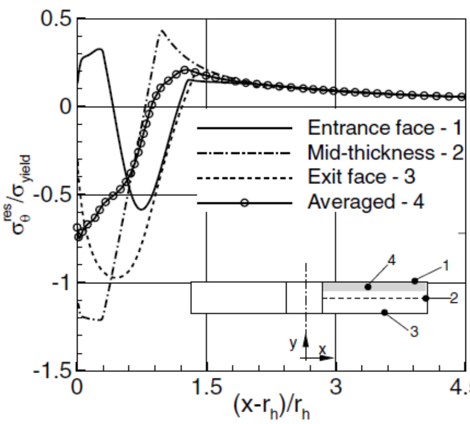


Fig 7 – Through thickness residual stress field, cold expansion 4%. Circumferential residual stresses.

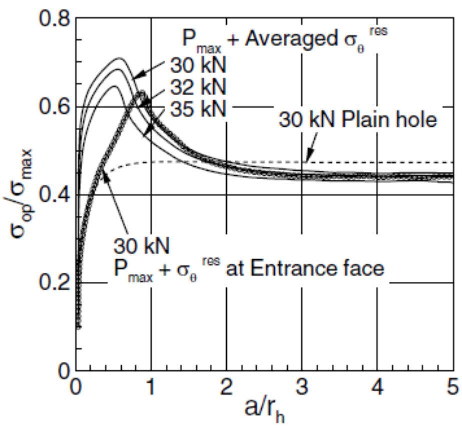


Fig 8 – Opening stresses for different levels of resi-dual stresses and remote applied load.

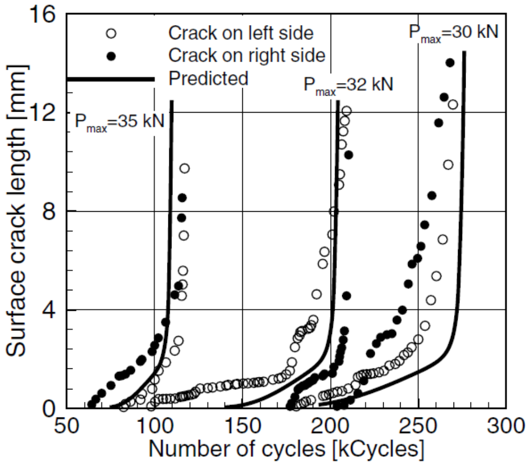
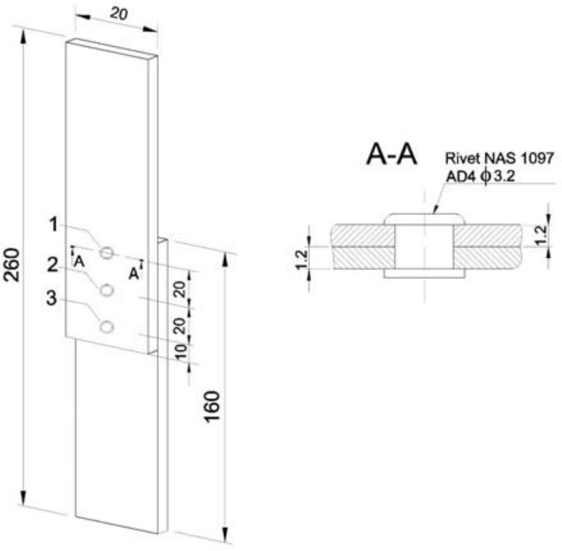


Fig 9 – Comparison of experimental results and fatigue lives prediction for different levels of max. applied load.

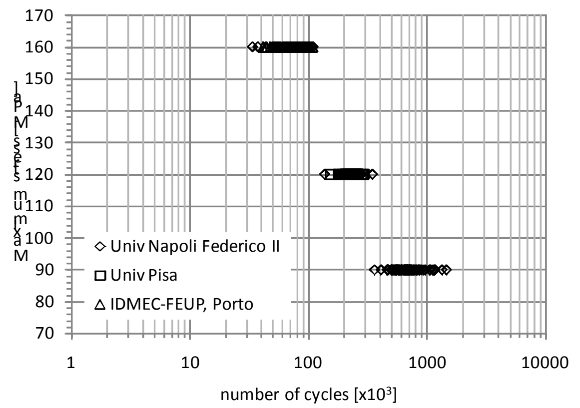
The riveted lap joint presents a complex behaviour including bending and contact between the rivet and the hole, and between sheets, leading to some specific features of fatigue cracking as the initiation of cracks above the hole and not through a plan containing the centre of the hole as in a first approximation might be expected (Silva *et al*, 2000, de Rijck *et al*, 2007, Skorupa *et al*, 2009).

The simplest tests of actual riveted lap joints concern one column specimens, as shown in Figure 10a), outcome of a round robin exercise intended to generate input data (crack length *a vs* number of cycles *N*) for the equivalent initial flaw size (EIFS) distribution determination and to generate a probabilistic stress-life (*P-S-N)* curve for safe-life analysis, Koolloos *et al*, 2003. The SN data of the 300 specimens tested is shown in Figure 10b.

It is to be noticed that the diameter of open hole specimens reported in Figure 4 is 5mm, and their thickness 2*mm*, whereas the specimens of Figure 10b are made of 1,2*mm* thick sheets and the rivet hole diameter is 3.2mm. Also, R = 0.1 in the case of the open hole specimens, and 0,05 in the case of the one rivet column lap joints. Because of these reasons, no direct comparisons can be made between the data of Figures 4 and 10b. However, the data suggests the increase in strength of the lap



a)



b)

Fig 10 – a) riveted lap joint: one column of 3 rivets. b) fatigue test data for 2024-T3 Alclad.

joint as compared with the normal non-cold worked open-hole specimen; and that should be related to the load distribution among the 3 rivets in the case of the lap joint specimens, a subject dealt with in detail in Moreira *et al*, 2007a. In that paper the distribution of load among the three rivets is discussed in the context of crack propagation. It suffices to mention here that for the un-cracked situation, the overall load is distributed approximately ~38% for the first and last rivet, and the remaining (~23%) for the middle one.

A serious form of damage in riveted Al fuselage is MSD, consisting of the simultaneous presence of several active fatigue cracks in a single joint. MSD lead to the widely discussed 1988 Aloha Boeing 737 accident. A discussion of this type of damage is given in Jones *et al*, 2008, Jones *et al*, 1999. The problem of MSD may be studied from the initiation and propagation viewpoints. As far as initiation is concerned, statistical techniques are used to model the initiation behaviour, as discussed in detail by Horst (Horst, 2006, Horst, to be published) Concerning propagation, test results are shown plotting number of cycles in vertical axis *vs.* crack growth at relevant locations, in horizontal axis, *eg* Figure 11 taken from Silva *et al*, 2000.

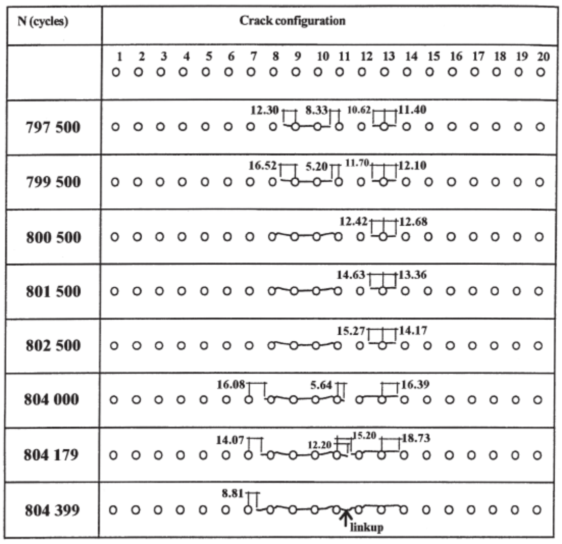


Fig 11 – MSD in a riveted lap joint, details in Silva *et al*, 2000.

4 - WELDED JOINTS

Part count reduction and possible economies in cost and fabrication time generated an interest in alternative joining processes for Al fuselage. Conventional fusion welding techniques applied to typical Al fuselage alloys generate weldments with unacceptable defects. This problem was overcome with laser beam welding (LBW) and friction stir welding (FSW), both already applied to parts of fuselages of Airbus aircrafts. FSW was adopted for the fabrication of the fuselage of the Eclipse 500.

FSW joints present sound metallurgical properties and using adequate welding parameters defects may be substantially avoided. Heat input and residual stresses are comparatively low. Moreira *et al*, 2008a, and Richter-Trummer *et al*, submitted 2009, present some relevant comparisons between MIG (metal inert gas), LBW and FSW applied to Al alloys.

FSW joints present a decrease in hardness in the thermo-mechanically affected zone (TMAZ, essentially the nugget and the region below the tool shoulder), see for example Figure 12, from Moreira *et al*, 2007b, concerning butt joints of AA 6082-T6.

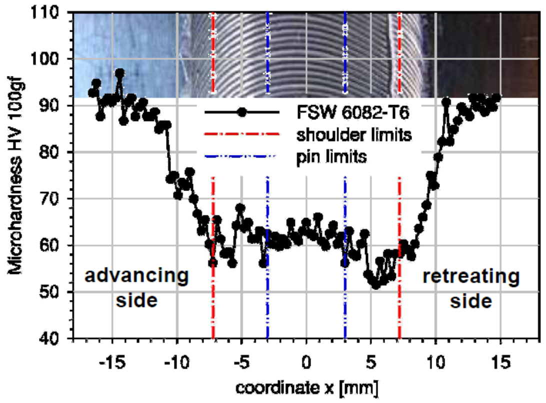


Fig 12 – Typical FSW hardness profile, Moreira *et al*, 2007b.

In most applications, particularly in aeronautics, fatigue behaviour is a critical concern. In case there is not a pre-existing crack, the fatigue process is composed of crack initiation, growth, and final rupture. It is therefore important to characterize the fatigue behaviour of FSW joints using a variety of tests. Initiation behaviour may be assessed using SN tests, since in this type of test once a crack initiates the specimen final rupture occurs very soon after. Basic SN data is reported in Moreira *et al*, 2008b, for AA alloys 6061 and 6082, showing the reduction of fatigue lives of as welded FSW joints compared with base material specimens, but also their improved behaviour when comparison with MIG welded specimens is made. For maximal stress corresponding to 10^5 cycles, reductions of approximately 45 % for LBW, 53% for FSW and 63% for MIG are found, see Figure 13.

These facts are not surprising; more noteworthy is what happens when there is a notch within the TMAZ. Moreira *et al*, 2008c, discuss load controlled SN tests of AA 6063 specimens loaded perpendicularly

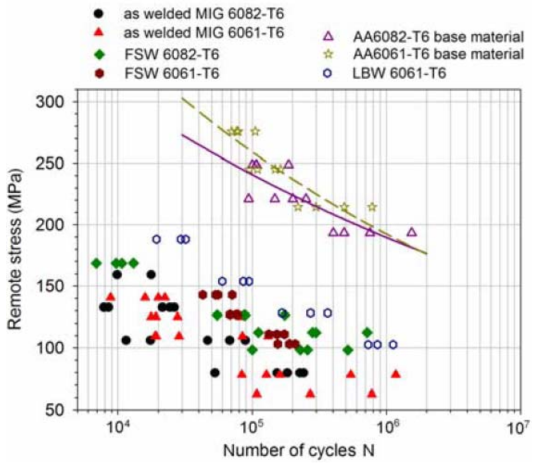


Fig 13 – SN behaviour, base and welded materials, Moreira *et al*, 2008b.

to the weldment, containing a small circular open hole in the TMAZ. In this case, it was verified that such specimens presented slightly higher fatigue lives than base material specimens with identical geometry, a fact possibly due to the microstructural changes undergone in the TMAZ. Another interesting feature of FSW fatigue behaviour is reported in Moreira *et al*, 2008d, where fatigue crack propagation tests were conducted on compact tension specimens with the crack presenting three different orientations *vis a vis* the weldment: crack growing along the heat affected zone, crack in the plane of symmetry of the weldment and crack growing perpendicularly to the weldment, see Figure 14. For comparison, base material specimens of identical geometry were also tested.

The results, discussed in detail in Moreira, 2008a and Moreira *et al*, 2008d [1], are presented in Figure 15.

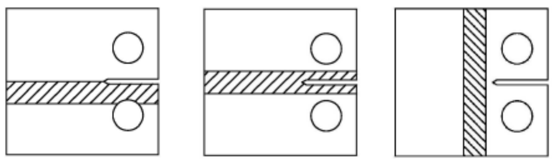


Fig 14 – Welded CT specimen types, Moreira *et al*, 2007b.

[1] In that paper, captions of Figures 7b and 9b should read as follows: "Fig.7b - Base material, FS weld material, HAZ and crack transverse to the weld seam for R=0.1"; "Fig.9b - FS weld material, and crack transverse to the weld seam for R=0.1".

The remarks made concerning the various types of tests and behaviours mentioned – SN tests on un-notched and notched specimens, fatigue crack propagation for several locations of the crack plane on CT specimens, - concern a few different base material aluminium alloys (6061, 6063, and 6082), and the trends identified should not be generalized to whole families of aluminium alloys. Nevertheless, for the materials and conditions studied, it is clear that FSW joints present good or very good fatigue behaviour, concerning initiation of cracks in notches located in the weldment and propagation of fatigue cracks in the weldment. Good fatigue behaviour was also found testing at INEGI-Porto FSW specimens of AA2195-T8X, an Al-Li alloy being considered for the cryogenic tanks of the next generation of the ARIANE launcher, Figure 16, Windisch, 2009, Eigen, 2009. In this case, when testing base material at R=0.1, maximal stress corresponding to 10^5 cycles is of the order of 350 to 400*MPa*, (Bonnafé *et al*, 2002)

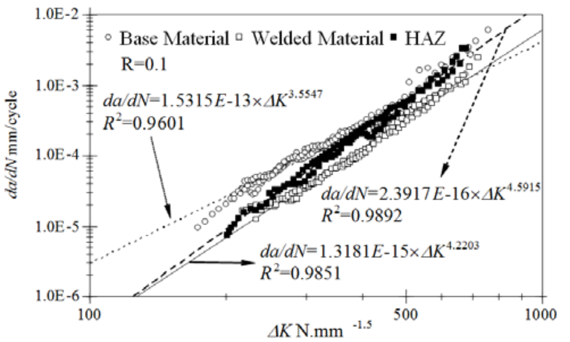


Fig15 – Comparison of base and FSW welded material, Moreira, 2008a, Moreira *et al*, 2008d[1].

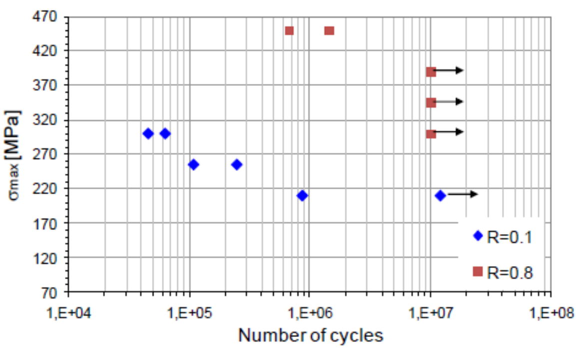


Fig 16 – FSW AA 2195-T8X, SN tests at room temperature, Windisch *et al*, 2009.

whereas for FSW joints, tested under R=0.1 with the weldment perpendicular to the load, maximal stress is of the order of 260 of 260 to 280*MPa* for the same number of cycles, a reduction of just approximately 30%.

5 - STIFFENED PANELS

Damage tolerant fuselage is supposed to sustain cracks safely until it is repaired or its economic service life has expired. Strength assessment of the structures is necessary for their in-service inspection, repair, retrofitting and health monitoring; therefore, damage tolerance analysis should provide information about the effect of cracks on the strength of the structure. Recently, studies are being conducted in both sides of the Atlantic to explore designs with equal or better performance than conventional designs with regard to weight and structural integrity, while achieving a significant reduction in manufacturing cost, *eg* the Boeing/NASA work, *eg* Pettit *et al*, 2000, or the EU DATON project which run from 2005 to 2008, largely unpublished so far (see synopsis in ref. European Communities, 2006).

The skin structure of a pressurized fuselage for transport aircraft is fatigue sensitive which is a problem often requiring frequent repairs. Under cyclic loading, a flaw can develop into a fatigue crack and propagate until fracture occurs. The residual strength concept permits the determination of the maximum crack length that can be safely sustained. With this information and the characterization of the crack growth behaviour of the material, the number of loading cycles that will be necessary for the crack to grow up to its critical length can be estimated in order to ensure safe operation. The development of numerical methodologies with the help of small laboratory coupon test results should be used to predict the residual strength of complex built-up aircraft fuselage structures.

Fracture mechanics concepts in conjunction with the Paris and other fatigue crack propagation laws are widely used to analyze and predict crack growth and fracture behaviour of aircraft panels.

The DATON project aimed to provide assessment tools for the damage tolerance of integrally stiffened structures produced by laser beam welding (LBW), friction stir welding (FSW) and high speed machining (HSM). There is an urgent pressure from the manufacturing side in the aerospace industry to apply these advanced structural concepts, since they promise considerable cost and production time benefits, together with a smaller number of fatigue and corrosion critical locations. The main drawback of integrally stiffened structures is the damage tolerance behaviour. Such an integrally stiffened design behaves totally different from the differential designs which are usually created by using riveted stiffeners. The prime problem is the crack arresting capability of the stiffeners both in fatigue crack growth as well as in residual strength. Fatigue and damage tolerance is therefore one of the main drivers in innovative aerospace structural designs as well as one of the main concerns on the safety of aging aircraft.

A test program which included fatigue crack growth rate of HSM, LBW, and FSW stiffened panels – see Figs 17, 18 - was performed. This test program was complemented by a scanning electron microscopy analysis of the fractured specimens.

Figure 19 gives details of the cross section and of the initial crack.

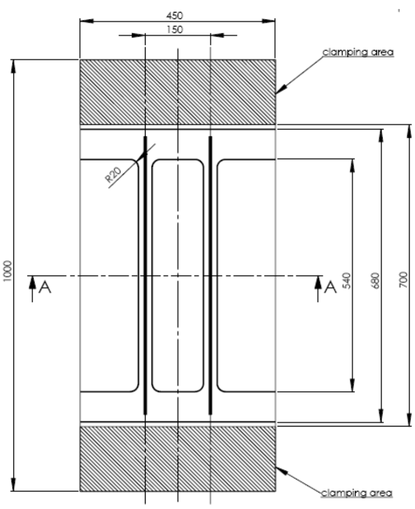


Fig 17 – DATON stiffened panel.

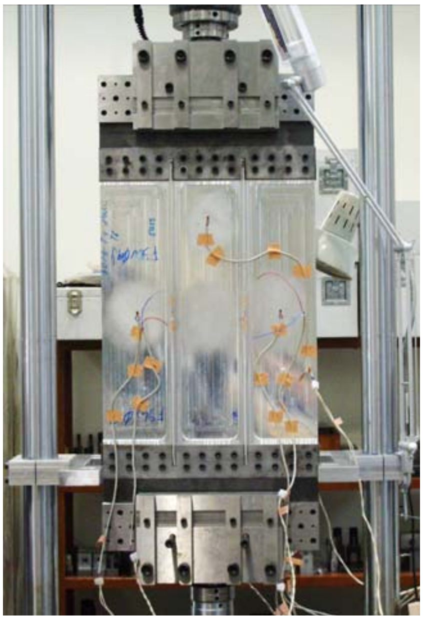


Fig 18 - Instrumented DATON panel (FSW) during fatigue crack growth testing at IDMEC-FEUP.

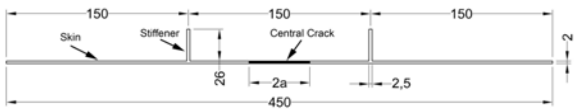


Fig19 – DATON panels' cross section.

When fatigue tested at R=0,1 the behaviour found for specimens of the 3 types studied - LBW, FSW and HSM - is shown in Figure 20, where welded panels always display greater fatigue lives than machined (HSM) panels.

Modelling these tests requires appropriate stress intensity factor solutions. This was obtained using VCCT (Tavares *et al*, 2009), see Figure 21.

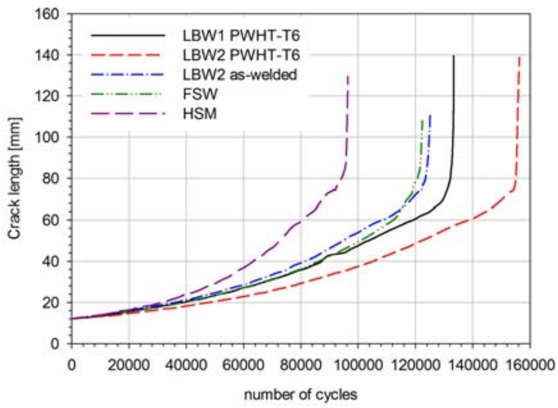


Fig 20 - Comparison of *a vs. N* for all specimens tested at R=0,1, Moreira, 2008a.

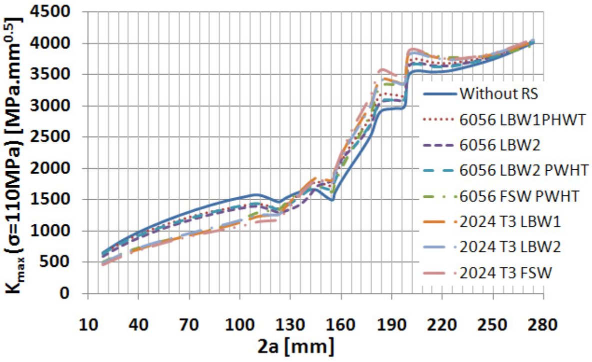


Fig 21 - SIFs for each welded configuration, and for panel without residual stress (HSM); remote load of 110MPa.

A particularly successful agreement between modelling and experimental data is shown in Figure 22, from Tavares *et al*, 2009.

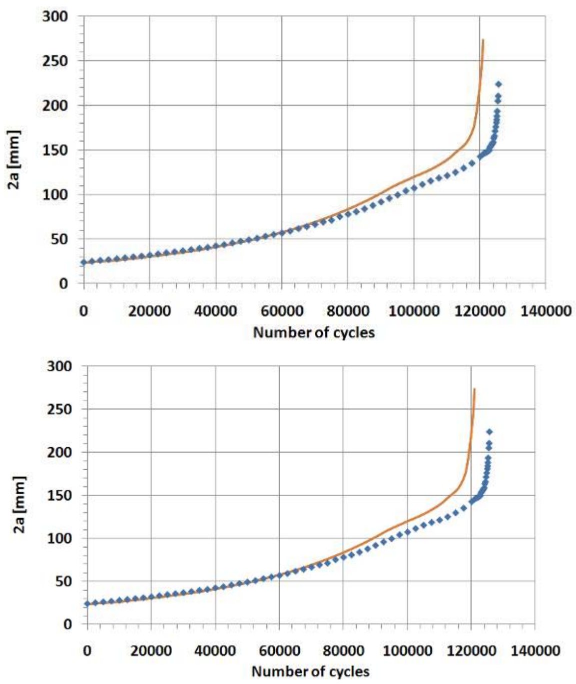


Fig 22 - Comparison between the numerical models (continuous line) and experimental results (markers). LBW2 AA6056 – T4 (as welded) - R=0.1 and σ_{max}=80*MPa*.

The work mentioned above illustrates again the importance of residual stresses in fatigue performance. In the case of DATON panels, because the initial crack was considered in the skin midway between stiffeners, the initial stages of fatigue crack propagation took place in a compressive residual stress field, thus retarding crack growth, which explains the behaviour presented in Figure 20.

Open issues concerning the use of monolithic integral structures in aeronautics include the damage tolerance problem, since the integral nature of the structure provides a continuous path for crack growth. Possible solutions under consideration include systematic thickness variations – crenellations – Uz *et al*, 2009, or specific surface treatments as supersonic particle deposition (SPD) desirably leading to some form of control of crack growth path. In parallel, alloys with improved strength but also good toughness and fatigue behaviour are considered, as the current generation of Al-Li alloys that overcomes the drawbacks of earlier generations characterized by high strength but poor toughness.

6 - CONCLUDING REMARKS

Open issues concerning the use of monolithic integral structures in aeronautics include the continuous path for crack growth. Possible solutions under consideration include crenellations or specific surface treatments as supersonic particle deposition (SPD) desirably leading to some form of control of crack growth path. In parallel, alloys with improved strength but also good toughness and fatigue behaviour are considered, as the current generation of Al-Li alloys that overcomes the drawbacks of earlier generations characterized by high strength but poor toughness.

REFERENCES

ATAG, April 2008, “The Economic and Social Benefits of Air Transport 2008”, Air Transport Action Group, Geneva.

Bristow, J. W. and Irving, P. E., 2007, “Safety factors in civil aircraft design requirements, Engineering Failure Analysis, vol.14, pp.459-470

Bonnafé, J.P., Gabard, D. and Grosjean, E., 2002, “Aluminium Lithium alloys use for reusable future launcher cryogenic metallic tanks”, 5th Conference on Aerospace Materials, Processes, and Environmeentl

Tecchnology, Sept. 2002, Huntsville, Alabama, NASA/CP-2003-212931

Brot, A, and Peleg-Wolfin, Y., 2008, “The damage-tolerance behavior of intregally stiffened metallic structures”, 48th Israel Annual Conference on Aerospace Sciences IACAS conference 2008

C. Boller, C., Buderath, M., 2007, “Fatigue in aerostructures—where structural health monitoring can contribute to a complex subject”, Philosophical Transactions of the Royal Society A, vol.365, pp.561-587

de Castro, P. M. S. T., de Matos, P. F. P., Moreira, P. M. G. P., and da Silva, L. F. M., 2007, “An overview on fatigue analysis of aeronautical structural details: Open hole, single rivet lap-joint, and lap-joint panel”, Materials Science and Engineering A, vol.468–470, pp.144–157

de Matos, P. F. P., McEvily, A., Moreira, P. M. G. P., de Castro, P. M. S. T., 2007, “Analysis of the effect of cold-working of rivet holes on the fatigue life of an aluminum alloy”, International Journal of Fatigue, vol. 29, pp.575–586

de Matos, P. F. P., “Plasticity induced fatigue crack closure : modelling and experiments”, DPhil thesis, University of Oxford, UK, 2008

de Matos, P. F. P., Nowell, D., 2008, “Analytical and numerical modelling of plasticity-induced crack closure in cold-expanded holes”, Fatigue and Fracture of Engineering Materials and Structures, vol. 31, pp.488–503

de Matos, P. F. P., Moreira, P. M. G. P., Nedbal, I., de Castro P. M. S. T., 2005, “Reconstitution of fatigue crack growth in Al-alloy 2024-T3 open-hole specimens using microfractographic techniques”, Engineering Fracture Mechanics, vol. 72, pp. 2232–2246

de Rijck, J. J. M., Homan, J. J., Schijve, J., Benedictus, R., 2007, “The driven rivet head dimensions as an indication of the fatigue performance of aircraft lap joints”, International Journal of Fatigue, vol. 29, pp.2208–2218

Eigen, N., Glaser, U., Sinnema, G., Moreira, P.M.G.P., Shneider, J., Sattler, E., 2009, “Microstructure and properties of AA 2195 base material and friction stir welds”, 11th European Spacecraft Structures, Materials and Mechanical Testing Conference (SSMMT), Toulouse, 15-17 Sept 2009

European Communities, 2006, “Aeronautics Research: 2003-2006 projects. Project synopses” vol. 1, ISBN 92-79-00643-6, p.167

Horst, P., 2009, “Assessment of multiple site damage in riveted aircraft joints”, chapter of a book in preparation.

Horst, P., 2006, “Widespread fatigue damage – an issue for aging and new aircraft”, Key Engineering Materials, vols.324-325, pp 1-8

J. Schijve, J., 1995, “Multiple site damage in aircraft fuselage structures”, Fatigue and Fracture of Engineering Materials and Structures, vol.18, (3), pp.329-244

Joint Aviation Authorities, JAR-25, Airworthiness, Large Aeroplanes, JAR 25.301 loads

Jones, R., Molent, L, and Pitt, S. 2008, “Understanding crack growth in fuselage lap joints”, Theoretical and Applied Fracture Mechanics, vol.49 p.38–50

Jones, R., Molent, L., and Pitt, S., 1999, “Study of multi-site damage of fuselage lap joints”, Theoretical and Applied Fracture Mechanics, vol.32 pp.81-100

Koolloos, M. F. J., de Castro, P. M. S. T., Esposito, R., Cavallini, G., 2003, “Fatigue testing of single-rivet lap joint specimens”, Technical Report ADMIRE-TR-3.1-06-3.1 / NLR CR-2003-281

Moreira, P.M.G.P., de Matos, P.F.P., Camanho, P.P., Pastrama, S. D., de Castro, P.M.S.T., 2007a, “Stress intensity factor and load transfer analysis of a cracked riveted lap joint”, Materials and Design, vol.28, pp.1263–1270

Moreira, P. M. G. P., de Jesus, A. M. P., Ribeiro, A. S., de Castro, P. M. S. T., 2007b, “Assessment of the fatigue behaviour of friction stir welded joints: aluminium alloy 6082-T6”, Key Engineering Materials, vols. 348-349, pp. 209-212

Moreira, P. M. G. P., 2008a, "Lightweight stiffened panels: Mechanical characterization of emerging fabrication technologies", PhD thesis, Faculty of Engineering of the University of Porto, Portugal

Moreira, P. M. G. P., Richter-Trummer, V., and de Castro, P. M. S. T., 2008b, “Fatigue behaviour of FS, LB and MIG welds of AA6061-T6 and AA6082-T6”, chapter in: G. C. Sih, ed., “Multiscale Fatigue Crack Initiation and Propagation of Engineering Materials – Structural Integrity and Microstructural Worthiness”, Springer Science + Business Media BV, pp.85-111

Moreira, P.M.G.P., de Oliveira, F.M.F. and de Castro, P.M.S.T., 2008c, "Fatigue behaviour of notched specimens of friction stir welded aluminium alloy 6063-T6", Journal of Materials Processing Technology, vol.207, (1-3), pp.283-292

Moreira, P. M. G. P., de Jesus, A. M. P., Ribeiro, A. S. and de Castro, P. M. S. T., 2008d, "Fatigue crack growth in friction stir welds of 6082-T6 and 6061-T6 aluminium alloys: a comparison", Theoretical and Applied Fracture Mechanics, vol.50, (2), pp. 81-91

National Transportation Safety Board – NTSB, 1989, Aircraft Accident Report, Aloha Airlines, Flight 243, Boeing 737-200, N73711, Near Maui, Hawaii, April 28, 1988, National Transport Safety Board, Washington, DC

Pasta, S., 2007,"Fatigue crack propagation from a cold-worked hole", Engineering Fracture Mechanics, vol. 74, pp. 1525-1538

Pettit, R., J. Wang, and C. Toh, 2000, "Validated feasibility study of integrally stiffened metallic fuselage panels for reducing manufacturing costs", NASA / CR-2000-209342

Rans, C. D., Alderliesten, R. C., and Straznicky, P. V., 2009, "Assessing the effects of riveting induced residual stresses on fatigue crack behaviour in lap joints by means of fractography", International Journal of Fatigue, vol.31 300–308

Richter-Trummer, V., et al, 2009, "Fibre Bragg grating sensors for monitoring metal inert gas and friction stir welding", submitted

Schijve, J., 2009, "Fatigue damage in aircraft structures, not wanted, but tolerated?", International Journal of Fatigue, vol.31, pp.998–1011

Silva, L. F. M., Gonçalves, J. P. M., Oliveira, F. M. F. , and de Castro, P. M. S. T., 2000, "Multiple-site damage in riveted lap-joints: experimental simulation and finite element prediction", International Journal of Fatigue, vol.22, pp. 319-338

Skorupa, M., Skorupa, A., Machniewicz, T., and Korbel, A., 2009, "Effect of production variables on the fatigue behaviour of riveted lap joints", International Journal of Fatigue, accepted for publication

Tavares, S.M.O., Richter-Trummer, V., Azevedo, P.C.M., Moreira, P.M.G.P., and de Castro, P.M.S.T., 2009, "Integral lightweight structures: fatigue and damage tolerance", AES – ATEMA' 2009 Third International Conference on Advances and Trends in Engineering Materials and their Applications, Montreal, July 06 -10

Uz, M.-V., Koçak, M., Lemaitre, F., Ehrström, J.-C., Kempa, S., and Bron, F., 2009, "Improvement of damage tolerance of laser beam welded stiffened panels for airframes via local engineering", International Journal of Fatigue, vol.31, pp.916–926

Wanhill, R. J. H., 2003, "Milestone case histories in aircraft structural integrity", in: I. Milne, R. O. Ritchie, B. Karihaloo, eds., "Comprehensive structural integrity", Elsevier, vol.1, pp.61-72, ISBN 0-08-044157-2

Windisch, M., Glaser, U, Trautmann, K.H., de Castro, P.M.S.T., Sattler, E., 2009, "Damage tolerance characterization of 2195 base material and friction stir welds", 11th European Spacecraft Structures, Materials and Mechanical Testing Conference (SSMMT), Toulouse, 15-17 Sept 2009