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USE OF SURFACE-TREATMENT LIFE EXTENSION METHODS FOR AIRCRAFT COMPONENTS

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ABSTRACT

Surface treatment methods such as shot peening or glass bead peening are well established as methods for extending the fatigue lives of structural details in aircraft. These methods introduce residual stresses in the surface layers that retard the growth of fatigue cracks, similar to the use of cold-expansion methods for fatigue enhancement of fastener holes. While these techniques have been used extensively to achieve satisfactory fatigue lives in individual components that exhibit unexpected fatigue cracking, the difficulty of guaranteeing full coverage and full implementation of the process make it difficult to factor the expected life extension fully into the component's service fatigue life. This paper discusses the development of a mid-life rework method, which can be applied to fatigue-critical aircraft structure. The process involves removing a layer of material from the surface to remove any damage in the form of cracks or flaws from earlier surface treatments. This approach alone will restore fatigue life, and a further life enhancement is achieved by application of an optimised glass-bead peening process. Two key elements are the use of life prediction methods to determine the size of any fatigue cracks developed during earlier service, and the development of a technique for the confident removal of the damaged material. The paper discusses the development of this approach for possible application to a fighter aircraft structural component.

KEYWORDS

Fatigue, peening, cracks, structural integrity, residual stress; 7050 Aluminium alloy; crack growth, life extension.

INTRODUCTION

Mechanical treatments such as radial cold expansion of holes and peening with glass or ceramic beads are used widely for fatigue life-enhancement of aircraft components, usually as part of the manufacturing process. New technologies such as laser shock peening and low plasticity burnishing provide alternative means of fatigue life enhancement and are being considered for application in the manufacture and repair of next generation aircraft. Peening is one of the most effective surface modification processes. It will improve the service life of structural components subjected to dynamic stresses by retarding fatigue crack growth in the early stages of a components service life. This occurs through the development of residual compressive stresses in the surface layers, generally, to a depth of 200μm to 300μm. The beneficial effects of peening surface treatments are dependent not only on the material and the pre and post-peening surface quality, but more importantly on the technique used and close control of the peening parameters. Further, the use of these methods in critical aerospace applications is highly dependent on the reliability of the process.

Research by the Defence Science and Technology Organisation (DSTO) examined the influence of material factors and surface condition on the fatigue life of aircraft structural components in the Royal Australian Air Force (RAAF) F/A-18 aircraft [1,2]. The work, which focussed on aluminium alloy 7050, used in the manufacture of F/A-18 wing carry-through bulkheads and other structural components re-emphasised the critical influence of the surface condition on the fatigue life of the component.

While surface modification technologies applied at manufacture are aimed to provide an enhanced fatigue life, there is a strong need for a viable repair/rework technique to restore and, where feasible, extend the fatigue life of components partially damaged by processes such as cracking, corrosion, or mechanical damage during service. The rationale behind the development of a life extension method for partially damaged components is to provide a means of extending the life of any local area that has displayed unexpectedly rapid or early fatigue cracking during service. To address this need, DSTO investigated the life extension potential of a re-work method involving removal of a specified amount of "prior fatigue damaged" material, including any surface damage remaining from manufacture, followed by an optional, controlled peening process using glass beads [3].

This paper discusses the potential application of this rework/repair method to a fatigue-critical region of a component in the RAAF F/A-18 fleet, in which fatigue cracking could lead to unacceptably high levels of inspection and maintenance. The approach is based on estimating the depth of any current cracking developed in service, using life prediction methods, and the development of a technique for confident removal of the damaged material completely, prior to rework by peening or re-peening, using tightly controlled conditions. The method has been developed to allow repeated life extensions of fatigue-critical parts reliably, with reduced maintenance costs being the expected benefits.

INVESTIGATION OF F/A-18 PEENING PROCESS

The fatigue critical areas in the F/A-18 aircraft, made of aluminium alloy 7050-T74511 were peened by the original equipment manufacturer (OEM), to provide additional resistance to fatigue cracking. However, the beneficial effect of peening was not accounted for in the original aircraft life modelling, rather, it was regarded as an added "insurance". At the time of aircraft purchase, local experience with peening high strength aluminium alloys such as AL7050 was very limited. Having recognised that the surface condition of airframe alloys in modern aircraft was critical [4], the DSTO research program focussed initially on assessing the Life Improvement Factor (LIF) associated with the peening treatments used on the F/A-18, under local service loading conditions.

The initial work [5], was aimed at understanding the effectiveness of the peening process used on the aircraft, and the influence of surface condition on a components fatigue life. Based on experiments with various peening conditions, it was shown that any life extension from peening resulted from the sum of two competing effects: the first; a decrease in fatigue crack growth rate associated with sub-surface residual compressive stresses, and the second; life reduction associated with the introduction of laps and folds in the surface during the peening treatment.

Improved Peening Method

During the course of the investigation, a number of refinements to the original (OEM) F/A-18 peening process were suggested to ensure that the maximum realistic LIF could be obtained from the modified peening treatment [6,7], the intention being to use the modified, improved treatment during any modification work performed on the fleet. The main refinements to the peening method recommended by DSTO were:

a. using a lower Almen intensity (peening intensity is measured using the bowing of small test strips);
b. better control of the bead impact angle;
c. better control of nozzle stand-off distance;
d. use of non-recycled glass beads, to minimise the impacts from broken or damaged beads.

The refined peening process led to substantially improved fatigue lives and reduced scatter in the fatigue lives due to the reduction in the level of surface damage in comparison with surface damage observed in OEM peened surface [6] ,(Figure. 1). The depth of the damaged layer in the OEM peened surface was 150μm, compared to 80 μm in the DSTO optimised (refined) peening process.

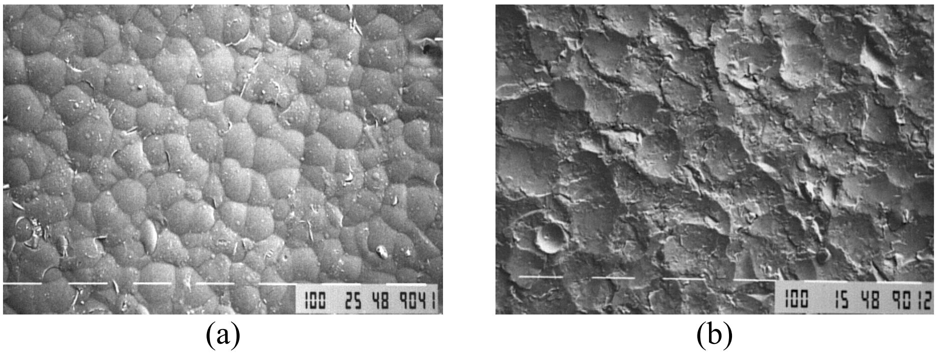

Figure 1: The improved surface condition resulting from the DSTO-optimised peening method (a), compared with the surface damage observed in fleet aircraft component produced by OEM peening (b).

PEENING AS A LIFE EXTENSION AND RE-WORK TREATMENT

The F/A-18 aircraft has several critical regions, some of which were peened during manufacture. Any mid-life re-work or life extension method must take into account the differences in surface finish and expected fatigue lives in the peened and un-peened regions. Since the original (OEM) peened surface was found to have peening associated surface damage [2], and one proposal was to rework the area by simply peening over the damaged area part-way through life, DSTO compared the laboratory fatigue lives of different peening rework processes performed on the aircraft, using 7050 aluminium alloy test coupons and a spectrum (variable amplitude) loading sequence representative of loads experienced by the aircraft structure during RAAF service [6]. In these experiments, fatigue lives of specimens hand polished to 800# finish and peened to simulate OEM peening were compared with direct over peen of OEM peening, and polish to remove previous surface and cracking damage and peen (DSTO method) treatments. Using a peak spectrum stress level of 410MPa the improvement in the fatigue life using the DSTO method was 50% over the OEM's over-peen approach, ie., peening over previous and possibly damaged peened surface. The results are shown in Figure 2. Based on the findings of these comparison experiments, it was concluded that all traces of the OEM peening should be removed prior to rework peening [6].

Further research established that peening over a surface which contained growing fatigue cracks or embedded damage (which could be a source of cracking), might not be beneficial for life extension. This is illustrated in Figure 3, in which the specimens were fatigue tested to a percentage of total life of the un-peened specimens and then peened before continuing the test until failure; peening of a specimen that has seen less than 50% of its failure life (fatigue damage) can be beneficial. However, if the damage exceeds ~60% of original life, the peening re-work is ineffective in restoring the fatigue life, since the residual stress distribution is less effective once the crack has progressed beyond the compressive residual stress layer (250 to 300μm deep for glass bead peening). Accordingly, if an effective life recovery procedure is to be developed, it will be necessary to remove both any cracking along with other surface damage.

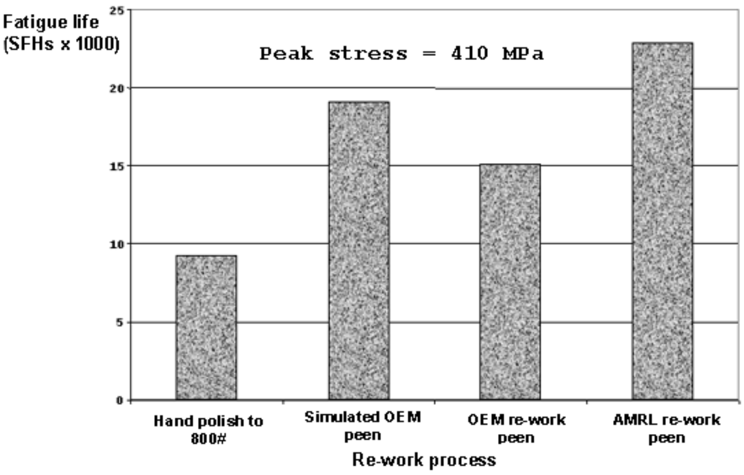

Figure 2. Comparison of fatigue lives (averaged on a log basis) for various re-work peening processes

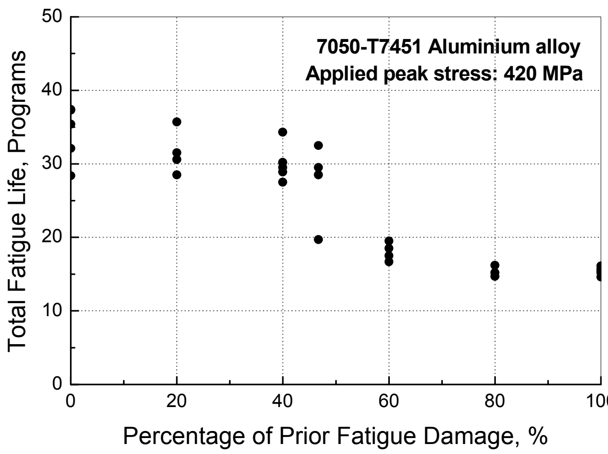

Figure 3: Fatigue life of specimens peened after various levels of prior fatigue damage (% of unpeened life).

Recovery and Extension of Fatigue Life

The first phase of the life-extension method is the effective surface removal of previous surface damage, including any cracking that may be present. Generally, the surface removal in any highly stressed fatigue critical airframe component is considered undesirable, since the amounts of material must be very carefully controlled to avoid major damage, and yet there must be a high level of confidence that the damaged material has in fact been removed.

Recognising that the controlled material removal is a critical aspect of the repair process, DSTO developed a surface removal method [8] to remove both the fatigue cracking which was developing, but was as yet undetectable, and any damaged surface layer. The steps involved in the DSTO surface removal method are shown in Figure. 4.

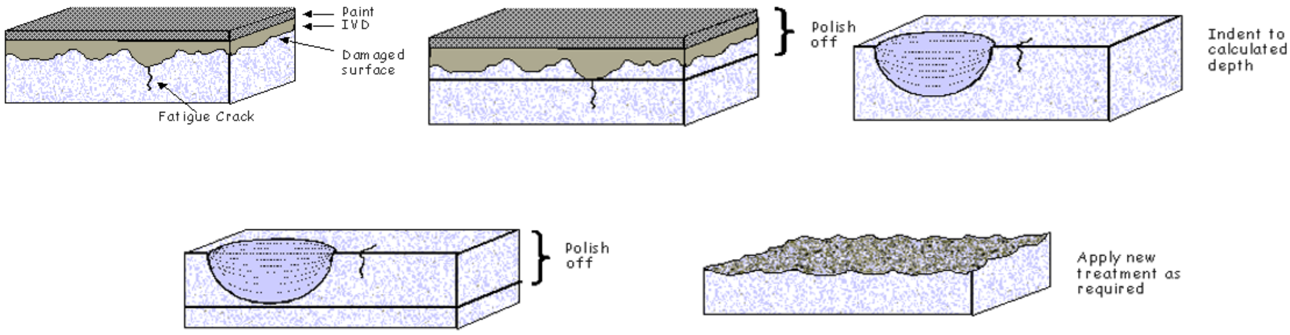

Figure 4: A step-by-step description of the complete repair process, using indentation to mark the damaged layer and the possible application of a new surface treatment

Rework method summary

Two local-area re-work processes have been identified and developed;

1. Remove the layer of material containing any original surface damage and a further quantity of material which will accommodate any fatigue cracking growing from it, making allowance for variability in the depth of this cracking. This would restore the original un-peened fatigue life. The DSTO surface removal method provides a means of removing thin surface layers on critical parts without risking either excessive or insufficient material removal from the surface.
2. If required, apply the DSTO optimised-peening process over the damage-removed or undamaged surface, as a supplementary life-extension method. Decision-making as to whether this process is desirable needs to balance the life extension against the possibility that introducing a peened surface finish may compromise component inspection.

APPLICATION OF LIFE EXTENSION METHOD TO AIRCRAFT COMPONENTS

The rework procedure was trialed by developing the method for a proposed repair to a critical area; the X-19 pocket one of the wing carry-through bulkheads of the F/A-18, shown in Figure 5 [3,10]. This bulkhead is made from 7050-T7451 aluminium alloy plate. The X-19 pocket area encounters high service stresses and is thought to be developing cracks at a rate faster than desired.

The manufacturer's original process specification does not require peening in the X-19 area (an Engineering Change Procedure implemented later introduced peening for this area). However, the location has a surface coating of Ion Vapour Deposition (IVD) aluminium for corrosion protection. Prior to applying this coating the surface of the component is acid etched. Laboratory fatigue tests and detailed fractographic analysis showed that this etching treatment produced etch-pits, which initiate fatigue cracks [10]. The average fatigue lives of etched specimens were 30% shorter than those of as-machined specimens.

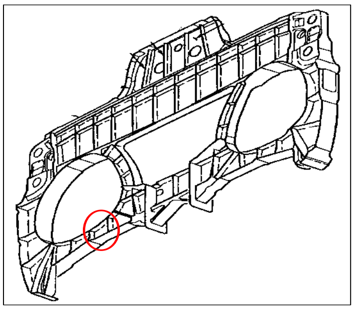

Figure 5 Figure showing the location of the X-19 pocket area (arrowed) of Y470 bulkhead

The amount of material to be removed is dependent on the previous surface treatments, and the extent of cracking, which, in turn, is dependent on the period the component, has been in service and the loading experienced. The two variables to be considered in the material removal are: (i) the initial defect starter size or Initial Discontinuity State (IDS) and (ii) the crack growth rate derived from quantitative fractography on cracked components from full-scale fatigue tests and laboratory etched and peened coupon tests. The IDS distributions – in this case, the distributions of initiating defect sizes for the dominant fatigue cracks were determined from quantitative fractography on etched fatigue loaded specimens backed up by full-scale test examples. The crack growth rate and therefore the depth of fatigue damaged material requiring removal on the X-19 area was also determined from these coupon tests and examples. The depth of a fatigue crack will be the sum of the deepest flaw and the amount of cracking possible at a given life. To determine a (1/1000) crack depth, three approaches are possible:

1. Use the 1/1000 defect as the initial crack depth ($3\log\sigma$) and mean crack growth rate;
2. Use the 1/1000 crack growth rate ($3\log\sigma$) and mean initial defect starter size; or
3. Use a combination of $1\log\sigma$ initial defect starter size coupled with a 1σ crack growth rate.

Graphical representation of the above [3], showed that Case 1 material removal was the most conservative (it provided an upper bound to the 1/1000 cases), for the fatigue lives expended (FLE) by the aircraft.

For an area in the component material that has been peened previously (by OEM), the 1/1000 IDS value was found to be 138μm. For the IVD etched material the 1/1000 IDS was 91μm. To return a AL7050-T7451 surface to its "as new" condition 138μm for a OEM peened surface and 91mm for a IVD etched surface would need to be removed without regard of service fatigue cracking. This is the amount of material that would need to be removed once any surface peening indications (ie dimples) or obvious etch pits have been removed, to ensure t no remnants of the original surface damage (including peening debris, grain boundary etching). A further allowance for service induced cracking would be removed estimated by considering the fatigue crack growth rate for this material under the aircraft-loading spectrum representative of fleet service.

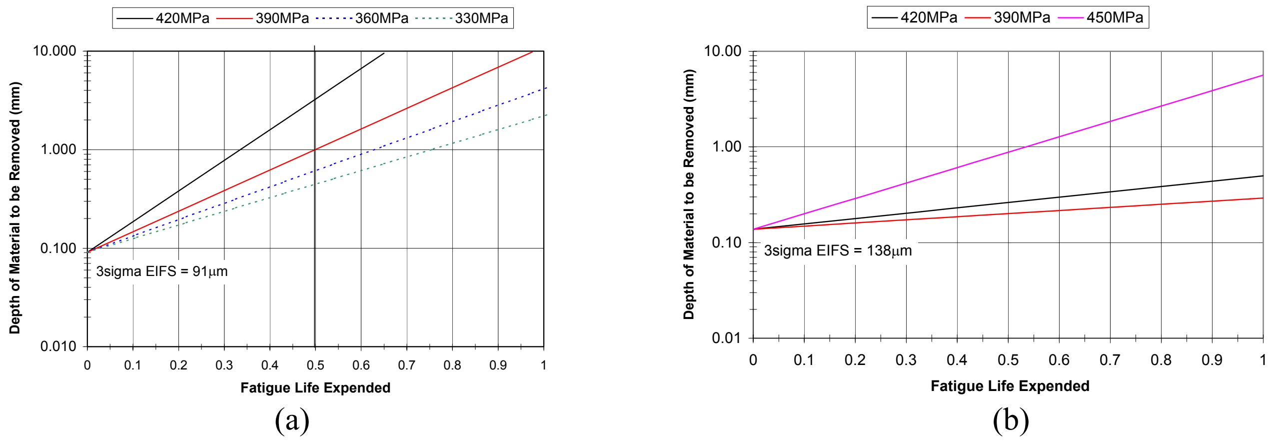

***Figure 6**: Material removal versus aircraft FLE ((a) not previously peened and (b) previously peened surface)*

For the proposed repair method, the material to be removed from un-peened and peened surfaces of a fleet aircraft structural component as a function of the FLE with a loading spectra and stress level representative of the area being considered in the aircraft are obtained from the curves [3,10], shown in Figure. 6. These curves were based on an exponential crack growth relationship drawn from the quantitative fractography of numerations coupon and full-scale fatigue test cracks at several stress levels, with the appropriate curve being used for the stress measured in the item being repaired.

Multiple Re-works

In considering the possible application of the rework method to aircraft, it is necessary to consider the possibility of further reworks later in service life, and to ask whether there is a limit to the number of times an area could be repaired, while still maintaining the required strength. While multiple reworks may protect the surface, they offer no protection against the increasing risk of failure initiating from sub-surface defects. The number of re-works is being examined to determine how many can be applied before sub-surface flaws cause failure.

CONCLUSIONS

This paper has presented the development of a rework/repair method developed at DSTO for application in fatigue-critical aircraft parts in the RAAF F/A-18 fleet. The approach is based on estimating the depth of any current cracking, and removing the damaged layer, prior to rework by peening or re-peening, using tightly controlled conditions. The method has been developed to allow repeated life extensions of fatigue-critical parts reliably and by enhancing the fatigue resistance of the aircraft components; reduced maintenance costs are the expected benefits.

The method involves removal of a layer of material containing any original surface damage and any fatigue cracks growing from it, using the surface removal method developed by DSTO. This should enable restoration of most of the original un-peened fatigue life. The second approach involves the use of DSTO optimised peening process, which could be used on relatively undamaged surfaces, previously peened surfaces or as a supplementary life-extension approach after surface damage removal, to provide additional fatigue life improvement.

REFERENCES

1. Barter, S.A., Bishop, B.C. and Clark, G. (1991). "Assessment of Defects in F/A-18 Bulkhead Test Article", Defence Science and Technology Organisation, Aeronautical Research Laboratory, Aircraft Materials Report, ARL-MAT-R-125, Commonwealth of Australia, August 1991, 26p.
2. Sharp P.K., Athiniotis N., Byrnes R., Barter S.A., Clayton J.Q. and Clark G. (1997). "Assessment of RAAF F/A-18 FS488 Bulkhead Offcuts; Microstructure and Surface Condition", Defence Science and Technology Organisation, Aeronautical and Maritime Research Laboratory Technical Report, DSTO-TR-0326, Commonwealth of Australia, September 1997, 93p.
3. Sharp, P.K, Barter, S.A and Clark, G. (2000). "Localised Life Extension Specification for the F/A-18 Y470 X-19 Pocket", DSTO-TN-0279, Defence Science and Technology Organisation, Aeronautical and Maritime Research Laboratory, 22p.
4. Clark, G. (1990). Defect Assessment: Changing Requirements in Military Aviation, *Materials Australasia* 22 (4) 18-21.
5. Clayton, J.Q. and Clark, G. (1988). The effect of steel shot and glass bead peening treatments on the fatigue resistance of 7050-T76351 aluminium alloy, *Proc. Aust. Fract. Group, "Fracture Mechanics in Engineering Practice*", Melbourne.
6. Sharp P.K., Clayton J.Q. and Clark, G. (1994). "The Effect of Peening 7050-T7451 Aluminium Alloy on Fatigue Life", Fatigue and Fracture of Engineering Materials and Structures, Vol. 17 No.3, pp 243-52, March 1994.
7. Sharp, P.K. and Clark, G. (1997). "The Effect of Peening on the Fatigue life of Aluminium Alloys", Advances in Fracture Research, Proceedings, ICF 9, Sydney, Vol. 3, 1997, pp 1491-99.
8. Barter, S.A. and Houston, M.I. (1997). Examination of methods to control the depth of surface removal on the F/A-18 488 bulkhead, DSTO-DDP-0192, Department of Defence, Defence Science and Technology Organisation, Aeronautical and Maritime Research Laboratory.
9. Barter, S.A., Molent, L., Sharp, P.K. and Clark, G. (2000). Repair and life assessment of critical fatigue damaged aluminium alloy structure using a peening rework method, Proc Conference USAF Aircraft Structural Integrity Program – ASIP 2000, San Antonio, USA, 5-7th December 2000.
10. Barter, S. A. and Price, J. (2000). "Effect of Surface Preparation Treatments on Fatigue Life of 7050 Aluminium Alloy", Structural Integrity and Fracture 2000, G. Heness (Ed.), Australian Fracture Group Inc. and University of Technology, Sydney, Australia, pp.140-53.