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Life assessment and repair of fatigue damaged high strength aluminium alloy structure using a peening rework method

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Abstract

This paper discusses a holistic assessment and rework/repair process for fatigue damaged structure based on estimating the depth of any potential cracking, its removal and subsequently improving the fatigue resistance of the region by peening the area using tightly controlled conditions. The process was developed to be generally applicable to parts of the F/A-18 aircraft's aluminium alloy 7050 centre barrel bulkheads. An example problem and a proposed repair process for this area are presented. The method has been developed to allow repeated life extensions of fatigue-critical parts to enable fatigue tests to continue, or to provide fleet life extension when fatigue cracking is a surface initiating problem.

A crucial aspect of this rework method is the estimation of the current damage-state, which is controlled by the initial discontinuities in the component's surface, the local stress conditions and the load history experienced by individual aircraft. Cracks are often observed to form at discontinuities introduced during manufacture, such as intermetallic particles and micro-pores, and damage acquired during service, such as corrosion pits and mechanical scratches. However, the application of a chemical etching treatment to the 7050 aluminium alloy used on the F/A-18 aircraft, as a precursor to coating the structure with ion vapour deposited (IVD) aluminium which is used as a corrosion protection scheme, has introduced another source for fatigue crack initiation. This chemical etching, which is part of the cleaning process prior to IVD application, uniformly pits the surfaces with micro-pits that can act as fatigue crack initiators. Likewise, alternative corrosion prevention schemes commonly used on this material such as anodizing will produce similar discontinuities.

The results of small, low K_t coupon fatigue tests were used to establish relevant crack growth data and assess initial discontinuities sizes. These tests included etched, as-machined and peened coupons at several stress levels. Testing used F/A-18 wing root bending moment spectra and displayed very little scatter. Additional full-scale fatigue test results were also used. Quantitative fractography of many cracks allowed a better understanding of the importance of the initial discontinuities (flaw) sizes, and allowed estimates of the equivalent pre-crack sizes (EPS) for these cracks. Statistics on the EPSs from the coupon and full-scale fatigue test cracking, along with a simple exponential crack growth model provided an estimation of the expected depth of a potential crack in an example detail on service aircraft at the repair induction time. It was then possible to determine the amount of material containing potential cracks below the NDI threshold to be removed prior to peening. For the example area some of the data supporting this approach will be presented.

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1. Introduction

Research at the Defence Science and Technology Organisation (DSTO) into the material factors likely to affect the fatigue life of RAAF F/A-18 aircraft structure, highlighted the critical role played by the surface condition of this structure. Further research capitalised on this understanding by developing a procedure for localised life extension; the aim was to extend the life of structural regions which displayed unexpectedly rapid or early fatigue cracking during either full-scale fatigue testing or service. This method involves removing a specified amount of "fatigue damaged" material followed by inspection and a controlled peening operation. The process presented here is an update of that presented earlier [1], and includes new observations, clarifications and further details.

Much of the DSTO research in support of the F/A-18 program has focussed on the highly stressed and optimised 7050 aluminium alloy (AA) wing attachment bulkheads. The rework/repair method discussed here should be applicable to most of this critical structure, and the general process should be applicable to other structures and materials.

The critical areas for cracking found in full-scale fatigue tests on the F/A-18 airframe were mostly independent of fastener holes. Holes in highly stressed areas were fatigue enhanced and fitted with interference fit fasteners. This allowed bulk sections to be optimised thus creating uniform highly stressed sections. Consequently, most problem areas have been found to be associated with low K_t structural details. Most areas of the AA7050 primary structure are coated with the standard corrosion preventative system. This system includes ion vapour deposited (IVD) aluminium. The application of IVD aluminium is carried out on the pre-etched surfaces of the AA7050-T7451 F/A-18 aircraft bulkheads. The etching is part of the cleaning process and is used to promote adhesion of the IVD aluminium. Etching has introduced another source of fatigue crack initiation, which combined with the high uniform stresses that occur during loading, results in many areas becoming fatigue critical. Cracking in any problem location, may be spread over a significant area because of the uniformity of the stressing. The etching produces many very small pits at grain boundaries, surface breaking inclusion and other surface breaking discontinuities. These pits must be included with those discontinuities considered traditionally that initiate fatigue cracking (e.g. intermetallic particles, micro-pores and machining marks). Although the other discontinuities may exit, it has been observed that most cracks in the F/A-18's 7050 aluminium alloy structure initiate from these etch pits when present, in the absences of other more severe damage.

The results of three fatigue test programs using coupons made from 7050-T7451 aluminium alloy designed to be representative of critical F/A-18 structural details are used here. The surfaces of the coupons were typical of F/A-18 surface conditions, namely as-machined, etched, and shot-peened. During these tests quantitative fractographic (QF) results were used to characterise the equivalent pre-crack size (EPS) of the initiating discontinuities found at the start of fatigue cracks. The lead fatigue cracks all grew virtually from the beginning of fatigue cycling when loaded to simulate the conditions found in F/A-18 service.

Since RAAF F/A-18 aircraft are individually fatigue monitored, a repair that involves removal of any potential fatigue cracking can be optimised for each aircraft. This can be achieved by specifying an amount of material to be removed based on a calculation of the size of any potential fatigue cracks (at an acceptable level of risk) that may have grown in the detail of interest in an individual aircraft.

This paper presents the proposed method for the repair of critical regions to allow continued safe operation of the F/A-18 fleet and is considered applicable to other applications. Fatigue crack growth data, the statistical distribution of the calculated EPS values, established from the coupon and full-scale fatigue testing, along with a simple exponential crack growth model, allow an estimation of the expected depth of a potential crack in similar details in individual service aircraft based on historical usage. It will be shown that it is possible to determine the amount of material to be removed to ensure that cracks have been eliminated for the case where the non-destructive inspection (NDI) threshold amount cannot be removed due to structural repair limits.