A History of Tracking Fatigue of the RAAF F/A-18 Hornet

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Abstract

The F/A-18 A/B Hornet has been in service for thirty years with the Royal Australian Air Force (RAAF). For most of this time the accrued fatigue damage of the aircraft has been calculated using a RAAF specific fatigue tracking system that has allowed the rates between different aircraft to be managed to produce a level distribution of fatigue damage amongst the fleet. Now that the aircraft is nearing retirement, decisions are needed based on accurate estimates of fatigue damage accrual. In this paper we review the history of the tracking system: how it was developed, and the differences between the original tracking methodology and new changes that have been proposed to improve its accuracy. These changes (which may or may not be implemented) cover the damage algorithm used in the damage calculations. It has been proposed that the existing damage algorithm based on the strain-life approach that was common at the time of the introduction of the aircraft into service, be replaced with a method based on crack growth calculations.

Keywords: F/A-18, tracking, fatigue damage, crack growth

Introduction

The F/A-18 A/B aircraft was introduced into service in the RAAF in 1983 to replace the ageing Mirage aircraft as the front-line fighter aircraft. The fleet consisted of 75 aircraft with 57 single seat aircraft and a further 18 dual seat aircraft used for training purposes. Since that time, four aircraft have been lost due to non-structural related circumstances.

To date, nearly 250,000 flights have been flown by the RAAF F/A-18 A/B aircraft giving a total duration of over 300,000 flying hours since its introduction. Most of the data from these flights have been recorded by the on-board Maintenance Signal and Data Recording System (MSDRS) for further post-flight processing. Flight data is periodically downloaded from the aircraft and processed using a collection of computer programs known as the Aircraft Service Life Monitoring Program (ASLMP) to estimate the fatigue damage accrual for each individual aircraft. A variety of fatigue, component and coupon tests have been used to establish the safe life of the various critical components in the airframe. The aircraft are then either retired or modified when the predicted fatigue damage accumulation reaches these safe life limits.

Over the service life of the RAAF F/A-18, the ASLMP has employed a number of computer programs to determine the fatigue life expended by individual aircraft. Each iteration was intended to overcome a shortfall of the previous program. When delivered, the RAAF was supplied with the Original Equipment Manufacturer (OEM) tracking software Structural Appraisal of Fatigue Effects (SAFE v112). A review of the performance of this software in service raised a number of concerns. The RAAF Mission Severity and Monitoring Program (MSMP2) was developed to address some of these concerns.

Recent observations of increases in the rate of fatigue life accrual in the F/A-18 A/B Hornet revealed some limitations with the existing strain-life damage calculation algorithm MSMP2 [1][2] used to determine the Fatigue Life Expended Index (FLEI) which represents the proportion of fatigue damage accrued in each aircraft. MSMP2 uses a unit damage matrix approach to calculate the damage produced in a flight. Since the development of MSMP2, numerous tests on the whole aircraft structure and individual components allowed the Defence Science and Technology Organisation (DSTO), the Directorate General of Technical Airworthiness (DGTA), the Defence Materiel Organisation (DMO) and the broader F/A-18 international users community to build up in-depth knowledge of the critical locations in the F/A-18 and how fatigue damage accumulates in the structure. These experiments have shown that fatigue damage occurs due to the growth of cracks initiating from discontinuities early in the life of the aircraft. The variability or scatter in fatigue lives has been found to be mainly due to variations in the size of these discontinuities or in crack growth rates. For the same material and loading, the variability in the crack growth rates of the lead cracks (i.e. those that would lead to failure in the service life) is generally small for the same location across different aircraft and most of the variability found has been due to differences in the initial discontinuity size [3].

Data Recording System

At the time of acquisition, the aircraft had two data recording systems. The manufacturer's MSDRS system and an indigenous system that had been previously used on F111 aircraft in the RAAF. This system known as the Airframe Fatigue Data Analysis System (AFDAS), was trialled on the F/A-18 in addition to the MSDRS. AFDAS had some advantages over the MSDRS installed on the aircraft since it was capable of higher sample rates making it possible to capture buffet on the empennage and could monitor a larger number of strain gauges over the aircraft including gauges on all three primary bulkheads.

AFDAS Mk3 was originally developed to record peak valley tables for each channel it monitored. For the RAAF F/A-18 a new version of AFDAS was developed, AFDAS Mk5, that allowed for the peak valley tables to be replaced with time stamped data for each channel, allowing for more robust error checking and fatigue monitoring. AFDAS Mk5 underwent limited flight trials in 1997. The Mk5 was a software update of the Mk3 running on the same Z80 based hardware. However, due to limitations in the hardware memory AFDAS Mk5 could fill its 60Kb of allocated memory during a flight, especially if that flight experienced a lot of buffet loading, resulting in the loss of data for the remainder of the flight. Due to this and other limitations when recording time stamped data, AFDAS was not used in the F/A-18 fatigue management program.

The OEM on-board MSDRS data acquisition system records system generated messages which cover the aircraft configuration, system state and measurement of aircraft parameters during the course of a flight. This system also provides the black box flight data that can be used for incident and accident investigation. One of the prime functions of the system is to record data on the flight and fatigue parameters that can be used for assessing the fatigue accrual of the aircraft and its engines.

The MSDRS data contains specific records for the strain responses from seven strain gauges. These seven gauges are located at positions that can be directly related to the main manoeuvre loading experienced by the structure. These gauges are located at the wing root, wing fold, forward fuselage and the aft fins and stabilators. Of these gauges, only the aircraft's wing root strain gauge is used by the RAAF in directly monitoring the fatigue usage of the aircraft. In addition, to prevent overwhelming the recording system, the strain gauge readings are filtered so that only the peak-valley turning points that occur outside of a dead-band and greater than a particular rise-fall are recorded. This has eliminated the recording of small cycles that generally do not contribute significantly to the overall fatigue damage. The rate of recording does not allow for the effect of buffeting to be directly monitored.

Aircraft Fatigue Life

The aircraft was manufactured by McDonnell Douglas (now Boeing) and was originally designed for US Navy use. Since its introduction into the RAAF, the usage of the aircraft has been monitored and found to be significantly different from both US Navy usage and the original design usage. It was found that the RAAF usage was typically more severe than the certification baseline and hence concerns were raised about the subsequent fatigue life of the aircraft. As a result of these concerns it was decided to perform an additional fatigue test of the aircraft under a usage spectrum that was more in line with RAAF usage.

A collaboration was formed with the Royal Canadian Air Force (RCAF), to jointly test the aircraft under a usage spectrum that was representative for both countries. This resulted in the joint International Follow-on Structural Test Project (IFOSTP) for the F/A-18 which reduced the individual cost, risk and effort for each of the countries. Canada tested the wing (FT245) and centre fuselage section (FT55) and Australia tested the aft fuselage and empennage structure (FT46) [4]. The US NAVAIR was also a participant in this testing, supplying previous fatigue and flight test data along with some test structure. In addition, flight trials were carried out by these countries to collect in-flight data from loads and accelerometer instruments recorded at a higher rate than used on the MSDRS system. This flight trial data allowed the fatigue test loads to be developed for the representative usage sequence of the aircraft as recorded by the MSDRS.

Airworthiness Standards

The original standard used to certify the F/A-18 Hornet for US Navy usage was the MIL-A-8800. Because the IFOSTP fatigue test was going to provide much new information on the crack locations and times throughout the tested structure, it was decided both by the RAAF and the RCAF to use a standard that provided more detailed guidance on the requirements and interpretation of a fatigue test. As a result, the certification basis for the fatigue life assessment was changed to be based on DEFSTAN 00-970. This standard required that the aircraft had a cumulative probability of failure of no greater than 1/1000 at the end of the life of the aircraft. Where the structure was un-monitored there was also the requirement for an additional safety factor of 1.5. However, the monitoring system of the aircraft was deemed adequate and this additional tracking factor has not been used for those locations responding to wing root loads.

Fatigue Critical Structure

During fatigue testing it became apparent that the critical life-limiting item in the aircraft was the centre-fuselage structure consisting of a built-up centre-barrel component (shown in Fig. 1). Failure of this structure in flight would result in loss of the aircraft. The centre-barrel joins the wing to the fuselage as well as supporting the undercarriage. Numerous small cracks have been detected in the centre-barrel during the course of fatigue testing.

The centre-barrel structure consists of three primary structural bulkheads that allow attachment of wing lugs through upper and lower pins on each of the bulkheads. Each of these bulkheads has a number of critical locations whose fatigue lives in RAAF service have been determined largely based on the FT55 full-scale fatigue test of the centre-fuselage structure [4]. The component lives determined from the FT55 test were used as the basis of the safe-life determination for the RAAF F/A-18 A/B aircraft prior to the completion of further RAAF centre-barrel testing. The centre-barrel in 10 high life aircraft have been replaced in order to extend the life of these aircraft. Additional testing and tear down inspections of these retired RAAF centre-barrels has extended the life of some critical areas [5][6].

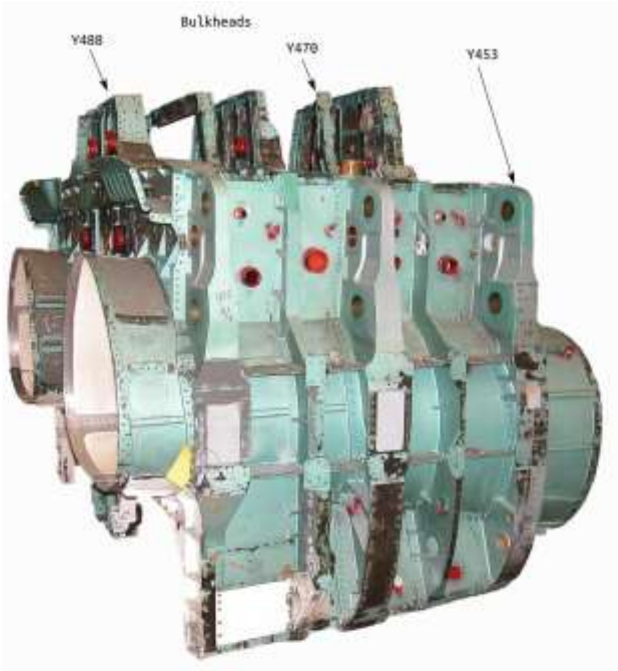

Fig. 1: Centre-barrel of the F/A-18 A/B Hornet

The centre-barrel bulkheads are of a monolithic construction each being machined out of a single 150 mm thick plate of aluminium alloy AA7050-T7451 material. At the time this was a relatively new allow that was developed to replace the 7075 series alloys and was specifically designed for thick section high strength components. The alloy was a mixture of aluminium, zinc and magnesium where the additional alloy components form precipitates during the age hardened process that restrict the flow of dislocations during plasticity thereby enhancing the static and fatigue strength. The heat treatment of the plate used in the bulkheads was (T7451) was in the over-aged condition which provides better resistance to stress corrosion cracking and exfoliation corrosion.

The main bulkheads were manufactured by machining pockets into the structure that generally left a thin central web surrounded by raised flanges, although the area around the wing attachment lugs was left largely at full thickness because of the high loads transferred by the attachment of the wings through a simple pin joint. However, because the structure was optimised elsewhere to remove material and reduce weight there are many high stress regions remaining, generally in the flange radii which initiate fatigue cracks. The fastener holes in the structure used interference fit fasteners that has resulted in these locations being highly resistant to fatigue cracking.

In addition, the processing and validation of the flight data collected requires a number of dedicated groups in various Defence organisations. The Tactical Fighter Systems Program Office (TFSPO) of the DMO provides oversight and direction to the contracted organisation

BAe Systems which writes, verifies and runs the tracking computer codes and performs the necessary checking to produce the fatigue life usage reports. The Defence Science and Technology Organisation (DSTO) and the RAAF DGTA Aircraft Structural Integrity branch (ASI) as the aircraft structural integrity managers, provide oversight, guidance and verification of the fatigue tracking methodology and how these results can be related to fatigue safe life limits.

Fatigue Tracking the Structure

Tracking Buffet Affected Structure

The empennage, outer wing and wing control surfaces, are subjected to aerodynamic buffet loads. The empennage buffet arises from vortices shed from the leading edge extension on the wing (Fig. 2), where the vortices burst over the fin and stabilators during high angles of attack creating large levels of aerodynamic buffet. Although strain gauges were installed over the aircraft to monitor the major loads through the structure, they were not recorded at a high enough rate to allow the fatigue damage from buffet on these structures to be directly determined from the strain gauge readings. In order to estimate the fatigue damage accumulation in the empennage the damage is calculated based on the accumulated time of flying at various Angle of Attack and dynamic pressure (AoA-Q) combinations. A damage matrix based on strain-life predictions of sample times of buffet measured during flight trials is used to calculate the fatigue in each region. Using data from the monitoring system for the time at each AoA-Q bin, an estimate of the fatigue damage accumulated in buffet regimes can be made. This approach is particularly useful for assessing the fatigue accrual in buffet affected structure such as the supporting structure for the vertical and horizontal stabilators.

Fig. 2: Vortices shed from the leading edge extension of the wing causing buffeting over the empennage

Tracking Nz Affected Structure

At the time the aircraft was put into service, the US Navy required that there should be essentially two fatigue lifetimes demonstrated by a fatigue test with no significant crack growth and one further lifetime to demonstrate that the crack growth durability life of the structure was acceptable. Ultimately these requirements provided a similar result as the DEFSTAN safety factor.

Initially, fatigue damage was calculated for the wing root loading dominated structure using the Palmgren-Miner approach based on the estimated life determined using a strain-life curve derived from constant amplitude coupon test data. In order to use this data to calculate the life for variable amplitude loading, it was necessary to determine an equivalent strain state at each notch or stress concentration in the structure. The local stress-strain response at an arbitrary stress concentration was obtained from the applied far field loading using Neuber's rule which assumes that the far-field stress is elastic and any plasticity is localised to the notch. However, because of the use of scaling factors to match safe lives, this method was found to introduce a significant history effect particularly where large overloads occurred during the aircraft flight. These overloads could be caused by actual manoeuvres, however occasionally they were also caused by erroneous data points.

Equivalent loading cycles were determined using a range-pair cycle counting technique that converted the variable amplitude sequence from the wing root strain gauge of each aircraft to a sequence of constant amplitude cycles. These cycles were at different stress ratios and were converted into equivalent fully reversed constant amplitude cycles for which there was coupon test data by an equivalent strain equation. Using the Palmgren-Miners rule (failure occurs when the sum of the damage equals one) a damage for each cycle could be calculated and the overall damage for each flight was then the sum of the damage of the individual cycles.

Where good strain data exist, the fatigue damage calculation for the centre-barrel and wing is based on the strain record of the wing root strain gauge. This gauge is mounted on the wing side of the wing attachment lugs which are made from titanium material which is a significantly more fatigue resistant than the aluminium alloy of the bulkheads. Where strain gauge data is unavailable additional fill-in techniques are used.

It was found that there was a sequence effect produced in processing the wing root strain data by the original version of the fatigue software SAFE v112. To remove this sequence effect an interim fatigue damage algorithm MSMP2 was developed that did not track the strain-life history but still used the strain response from the wing root gauge. This algorithm uses the range-pair counted strain cycles of the wing root gauge to determine the fatigue damage accrual. The damage matrix used in this program was derived from the combination of the equivalent strain equation used in the original system and the fatigue life obtained from constant amplitude fatigue tests. This approach notably did not include loading sequence effects and was found to give consistent fatigue life predictions. This algorithm became the permanent method replacing the original software.

Damage Calculation using Crack Growth Data

After years of conducting and examining numerous fatigue tests on the full size aircraft, it has been comprehensively demonstrated that the fatigue process for the F/A-18 loads and material combination consists almost entirely of fatigue crack growth that was initiated from intrinsic discontinuities in the structure. Many of these discontinuities were present as a result of the chemical etching of the structure for the application of the Ion Vapour Deposit corrosion protection scheme that was applied to the aircraft. Additional discontinuities present have also served to initiate cracks such as from machining marks, and to a lesser extent material porosity that existed in the bulkhead plates prior to machining. However, because the fatigue life improvement techniques that had been applied to the centre barrel holes were highly effective, cracking from defects typically associated with holes have not been a significant problem on the F/A-18.

Quantitative Fractography

During the 1980's and 1990's a number of other aircraft in the RAAF fleet had developed fatigue cracks that had been investigated by DSTO. This led DSTO to focus in on the inspection and identifying these and other fatigue cracks resulting in the development of quantitative fractography measurement techniques [7]. These techniques were later used to identify and measure the rate of fatigue crack growth from the characteristic marker bands (e.g. see Fig. 3) found on the fracture surfaces [8][9][10][11].

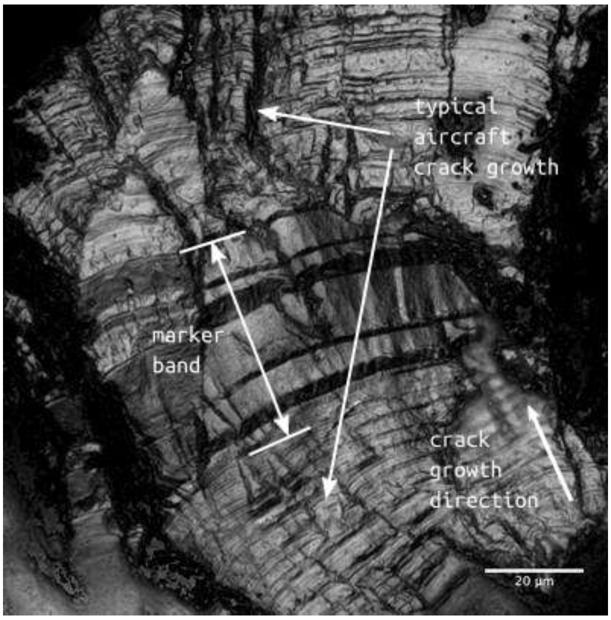

Fig. 3: Marker band sequence inserted into typical blocks of spectrum loading. This technique has been used to measure the rate of crack growth in typical blocks of F/A-18 wing root bending moment loading

As a result of this new interpretation and measurement capability at DSTO, it was found that all of the fatigue damage in the F/A-18 A/B aircraft can generally be attributed to crack growth [12]. This finding was evident through examination of crack fracture surfaces which generally show a distinctive pattern that allows the size of growth in each test loading block to be measured. Although the effect of any individual loading cycle is generally too small to be directly measured (which is generally known as a fatigue striation), when a repeating pattern of loading is applied, the rate of crack growth can be measured from the width of the segment and the average rate of growth per cycle calculated [13][14]. These marker bands consist of groups of striations that have similar loading conditions and in some cases can be correlated with flight records from fleet aircraft [7].

After developing this expertise these techniques have been used to measure the rate of fatigue crack growth from both spectrum loading sequences and simple sequences consisting of a mixture of spectrum loading and constant amplitude loading from small cracks. This has allowed the growth rates for small cracks and variable amplitude loading to be directly measured using quantitative fractographic techniques. The predicted lives from calculations using this data were found to be much closer to actual test lives allowing an improvement in the predictability of arbitrary sequences [15].

Predicting Crack Growth

Historically, the crack growth methodology used to calculate the size of fatigue cracks was developed at approximately the same time as the strain-life approach in the late 1950's and early 1960's. Paris et al. [16][17] was able to show that the growth of fatigue cracks could be characterised by a parameter known as the cyclic stress intensity factor ΔK which is used to describe the intensity of the stress field ahead of a crack tip. The application of cyclic loads with a stress range of $\Delta\sigma$ resulted in a stress intensity range ΔK which was found to correlate with the fatigue crack growth rate da/dN (where da is the increment of crack growth for a single cycle dN). This insight allowed the growth of cracks to be predicted on a cycle by cycle basis. If da/dN versus ΔK data can be represented by a straight line on log-log axes we obtain the Paris equation [17]:

$$\frac{da}{dN} = C\Delta K^m = C(\beta\Delta\sigma\sqrt{\pi a}\,)^m \tag{1}$$

where β is the geometry factor relating the boundary conditions to the crack tip, a is the critical dimension of the crack, and C and m are parameters characteristic of the material.

One drawback with the crack growth approach is that the increment of crack growth depends not only on the stress amplitude but also on the size of the crack. Therefore, using this approach directly made it difficult to develop a concept of fatigue damage where the damage of flights occurring at different times could be directly compared. The strain-life approach had the desirable character of using the concept of damage which could be compared at different times making it ideal for tracking the fatigue accumulation in the fleet. Whereas, although fatigue crack growth was shown to be measurable and accurate, it was considered more difficult to apply for managing the fleet.

A New Damage Metric

From fractographically measured crack growth data it was decided that better fatigue predictions could be made using fatigue crack growth prediction as the basis of a revised fatigue tracking algorithm. However, to become a drop-in replacement for the existing MSMP2 algorithm it was desirable for software compatibility reasons that there should be no sequence effects. As such, normal crack growth techniques could not be used as a drop-in replacement for the MSMP2 system. To implement crack growth methods without sequence effects it was decided to compromise and use fatigue crack growth data but at a fixed crack size. Thus each loading cycle would be independent of where it occurred in the sequence. The damage metric thus became the size of the crack growth for a cycle compared to the growth produced by the fatigue test baseline spectrum.

This new method was implemented in MSMP3 and uses the improved accuracy of the fatigue crack growth prediction and converts it to a damage metric as used in the strain-life method.

A typical crack size of interest in the F/A-18 of 1 mm, was taken as the reference point since this is typically the largest crack size that would be expected at the end of the life of the aircraft. The results however, are insensitive to the actual size taken. The damage is then the ratio of the sum of the crack growth for each cycle and the crack growth obtained from a reference sequence. Keeping this crack size fixed during this calculation creates an easy to interpret damage accumulation value. A cycle is determined from the matching of range-pair cycle pairs as is currently done.

The baseline reference sequence for the RAAF Hornet is the FT55 wing root bending moment sequence, identified as S_{Ref}. This sequence is used to calculate the length of the crack growth $G(S_{\mathrm{Ref}})$ that corresponds to the safe-life in terms of equivalent FT55 flying. The growth is calculated by converting the WRBM turning point sequence into range-pair cycles and interpolating the da/dN equation for each cycle of ΔK and R such as:

$$G(S_{\mathrm{Ref}}) = \sum \frac{da}{dN} = \sum f(\Delta K, R) \tag{2}$$

Hence, the relative damage D for any series of flights S_{flights} is simply:

$$D = \frac{G(S_{\text{flights}})}{G(S_{\text{Ref}})} \tag{3}$$

This method is strongly based on direct experimental evidence of the rate of crack growth which can indeed identify the crack growth increases due to bands of constant amplitude cycles with reasonable accuracy.

Comparison with Test Data

A range of coupon test data has been used to validate the fatigue life predictions based on the new algorithm. This data consists of variable amplitude tests using spectra that covered a range of F/A-18 usage in the fleet. Typically data was available for five coupon test result over three or four stress levels for each spectrum [18]. The results for one of the sequences used in the FT245 wing test which contains numerous small buffet cycles is shown in Fig. 4, where it can be seen that the proposed MSMP3 algorithm produces a closer match compared with the coupon test lives. These differences are much less pronounced when buffet cycles have been eliminated from the sequence.

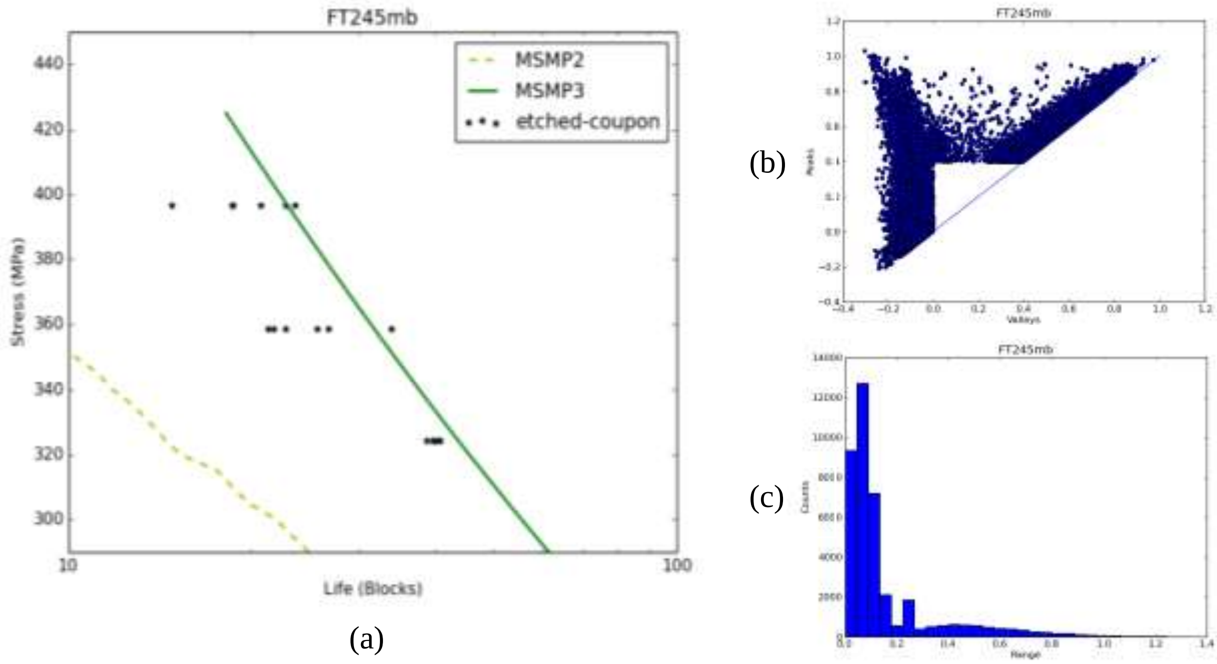

Fig. 4: (a) Comparison of predictions against etched coupon test lives for the wing test sequence FT245mb which contains numerous small buffet loads (b) peak-valley distribution of the cycles in the FT245mb sequence (c) histogram of the cycle ranges of the sequence. MSMP3 does a better job in correctly estimating the coupon lives than the MSMP2-relative version which is currently used to life aircraft

Conclusion

The fatigue damage consumed by flying the RAAF F/A-18 has be tracked and calculated over the life of the aircraft using a number of different algorithms based on tracking the fatigue damage due to loads responding to wing root bending moment. This process has involved a team of specialists across the RAAF, DSTO, DMO and commercial aircraft organisations. During this time many tests have been performed both of full scale aircraft components and of material coupons using a variety of test spectrums. This data has been combined and is used in assessing the safe life of the aircraft. Recently changes have been proposed to the damage calculation algorithm to base it on crack growth data instead of the original strain-life approach.

The new damage growth method used for predicting the fatigue damage accrual in the RAAF Hornet aircraft gives similar overall lives to the previous tracking algorithm. However, differences in the attribution of the fatigue damage means different flying regimes accumulate damage at different rates.

Overall the new MSMP3 program has proven simple to use as a drop-in replacement for the previous non-history effect tracking algorithm and has produced a satisfactory update to the fatigue tracking procedure used for the RAAF Hornet fleet.

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