XIV NASAS: **Fatigue, Fracture and Ageing Structures**; Eds.: CM Manjunatha, VR Ranganath, RK Paretkar and DR Peshwe, 30-31 January 2006, Visvesvaraya National Institute of Technology, Nagpur, pp. 93-98.

Spectrum Modification for Full Scale Fatigue Testing of an Ageing Aircraft

CM Manjunatha* and VR Ranganath
Structural Integrity Division, National Aerospace Laboratories, Bangalore- 560 017, India.

Abstract

The flight load sequence of a combat aircraft was modified by eliminating different levels of small amplitude load excursions to derive several different test load sequences. The fatigue crack growth behavior under all these spectrum load sequence was predicted in a single edge notched tension specimen of an airframe grade D16 aluminum alloy. Crack growth behavior was predicted using a fatigue crack growth law derived from the constant amplitude fatigue crack growth tests, incorporating crack closure effects. It was observed that full scale fatigue testing time can be reduced significantly by using of one of the derived test load spectrum without compromising, in general, on the fatigue damage caused by eliminated load cycles.

Keywords: Fatigue crack growth, Spectrum Loading, Aircraft, Aluminum alloy

1. Introduction

Full scale fatigue testing is one of the important steps in life extension program of an aging airframe. A representative airframe selected from the fleet of aging aircraft is loaded under spectrum sequence for a specified number of flight hours and the airframe is checked thoroughly for fatigue damages through visual and NDT methods. Repair and/or replace schemes for damaged components, if any, are then evolved based on these test results to enhance the Total Technical Life (TTL) for the feet of airframe.

The spectrum load sequence to be applied on the airframe during full scale fatigue testing is generally obtained from the flight data records of previous flights. Considering fatigue loads in various types of missions that the aircraft is used for and a statistical mix of these missions, a representative flight load spectrum block is derived. Higher the number of flight data used for spectrum derivation, more realistic the spectrum sequence would be. Such an exercise results in a block of flight load sequence, generally, containing a very large number of load reversals. Since average test frequency for loading in full scale fatigue test is kept low, the total number of reversals in the load sequence determines the testing time. One of the methods to accelerate the test is to eliminate large number of small amplitude load cycles that may not cause considerable fatigue damage [1-4]. The modified load sequence should have as many less number of fatigue cycles as possible but has almost similar fatigue damaging effect on the airframe materials as that of the original load sequence.

In this study, several different test load spectrum sequences were derived from the flight load spectrum by eliminating small amplitude load excursions below a prefixed filter value. Fatigue crack growth behavior was predicted in D16 aluminum alloy, under all these spectrum load sequence using crack growth model derived from constant amplitude fatigue crack growth tests. Based on the prediction results, it is shown that full scale fatigue testing time can be reduced significantly by using of one of the derived test load spectrum without compromising, in general, on the fatigue damage caused by eliminated load cycles.

* Corresponding author: Tel./Fax.:080-2508 6316, e-mail: manjucm@mailcity.com

2. Modification of Flight Spectrum

The original flight load sequence block of a combat aircraft considered in this investigation is shown in Fig. 1. This one block of load sequence represents about 300 flight hours, which is approximately about 1/10th of the designed TTL of the aircraft. A repetitive block of one tenth of the design life is an arbitrary choice but is widely accepted and used [5]. The total number of load reversals in this spectrum was 85082. The maximum stress was 186 MPa and the minimum stress was –47.2 MPa.

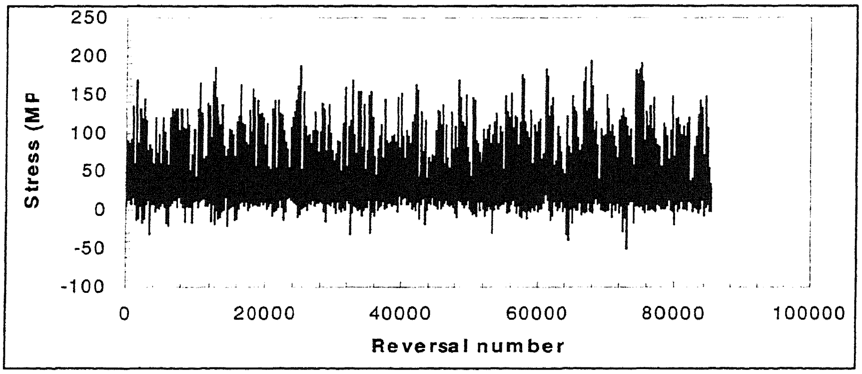

Fig. 1. Original flight load sequence used in this investigation

Several different test load sequences were derived from the flight load sequence by eliminating small amplitude load excursions using a simple computer program. The algorithm used for elimination was that an upward (downward) range, smaller than the prefixed filter value was eliminated only if it is followed by a larger downward (upward) range. This procedure is applied iteratively. At first, all the stress ranges in one direction are eliminated and then, iteratively, all the ranges in the other direction and so on, until no stress ranges lower than the filter value are present in the sequence. In this way, various load sequences were derived, the details of which are shown in Table 1. Typical M2 load sequence is shown in Fig. 2.

It may be noted that the total number of load reversals in the derived test load sequence drastically reduces with elimination of small amplitude load cycles (see Table 1). It is required to determine which of this derived test load sequence would simulate the fatigue damaging effects as that of the original load sequence. The fatigue crack growth analysis was performed to determine the suitable load sequence for full scale fatigue testing which would accelerate the test significantly.

Table 1. Details of the original and modified load sequences

Sl. No.	Load sequence ID	Elimination level of $\Delta\sigma$ (MPa)	No. of load reversals in one block	% reduction in total no. of load reversals
1	O	None	85082	--
2	M1	≤ 12.0	24073	71.70
3	M2	≤ 24.0	6685	92.14
4	M3	≤ 36.0	2827	95.00
5	M4	≤ 42.0	2117	97.51
6	M5	≤ 48.0	1553	98.17

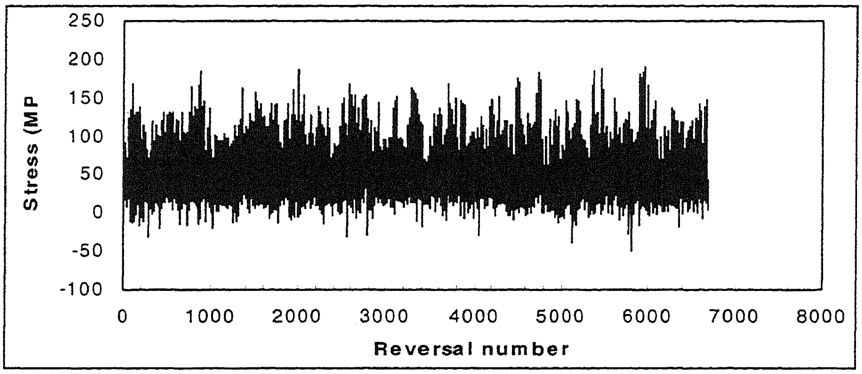

Fig. 2. Modified spectrum load sequence, M2

3. Experimental

In order to predict fatigue crack growth behavior under spectrum loading, it is required to deduce the crack growth law from constant amplitude fatigue crack growth tests. For this purpose, D16 aluminum alloy was chosen which is widely used in airframe construction. The mechanical properties of the material were as follows: σ_{UTS} = 460 MPa, σ_{YS} = 347 MPa and elongation = 12%. Constant amplitude fatigue crack growth tests were performed using Single Edge Notched Tension (SE (T)) specimens of W = 45 mm and t = 1.5 mm having a central U-notch of length 2.5 mm at one edge. Tests were performed in a computer controlled, 100 kN servo-hydraulic test machine, as per ASTM E647 standard [6] specifications at various stress ratios, R = - 0.3 to R= 0.7, with a sinusoidal waveform at a frequency of 10 Hz. Further details of the tests can be found elsewhere [7].

4. Results and Discussion

4.1. Fatigue Crack Growth Law

Results of constant amplitude fatigue crack growth tests at various stress ratios in D16 aluminum alloy is shown in Fig. 3 [7]. These results are in general agreement with observed FCGR behavior in this material [8].

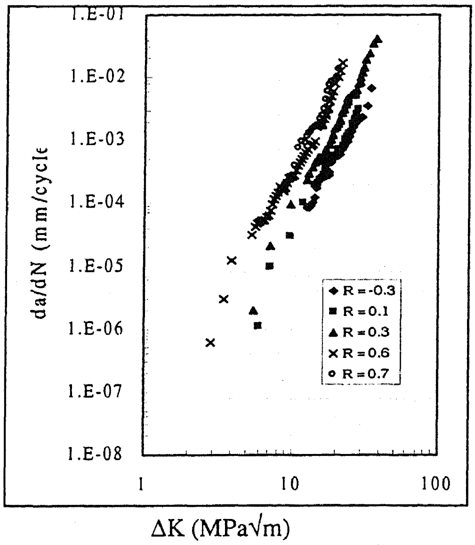

Fig. 3. Constant amplitude FCGR curves [7]

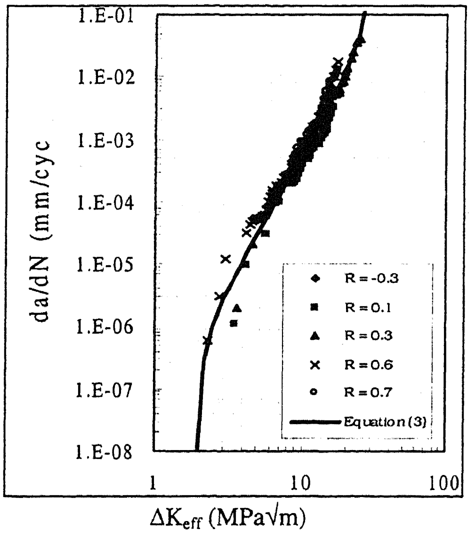

Fig. 4. Plot of da/dN Vs ΔK_{eff}

It has been shown [9] that for 2024-T3 aluminum alloys,

$$\Delta K_{eff} = U(R)\,\Delta K \tag{1}$$

$$U(R) = 0.55 + 0.33\,R + 0.12\,R^2 \tag{2}$$

Using the above equations, the FCGR data in Fig. 3 was re-plotted as a function of ΔK_{eff} in Fig. 4. Constant amplitude FCGR data shown in Fig. 4 was approximated in the form of Newman's [10] equation as,

$$\frac{da}{dN} = C_1\left(\Delta K_{eff}\right)^{C_2} \frac{\left[1-\left(\dfrac{\Delta K_{eff}^{th}}{\Delta K_{eff}}\right)^2\right]}{\left[1-\left(\dfrac{\Delta K_{eff}}{C_3}\right)^2\right]} \tag{3}$$

The constants in equation (3), determined by fitting the experimental data were as follows: ΔK_{eff}^{th} = 2.0, C_1 = 7.8239e-8, C_2 = 3.86, C_3 = 30.25. Solid line in Fig. 4 shows the plot of eqn. (3) with these constants, which describes the entire 'sigmoidal' FCGR curve quite well.

4.2. Prediction of FCG Behavior

Though crack opening load varies cycle-by-cycle in a spectrum load sequence [11], a constant K_{op} can be assumed during each program block of load sequence and it is fixed by the K_{max} and K_{min} existing in the spectrum load sequence [12,13]. Hence, assuming a constant K_{op} for the load sequence in this investigation, the σ_{op} for was calculated as [9],

$$\gamma = K_{op}/K_{max} = \sigma_{op}/\sigma_{max} = 1-(1\text{-}R)\,U(R) \tag{4}$$

In the spectrum load sequence, the minimum stress ratio, R = -47.2/186 = -0.2538. Substituting this R value in eqn. (4), and using equation (2) for evaluating U(R), σ_{op} = 75.46 MPa. Thus a constant crack opening stress of 75.46 MPa was used in truncating the spectrum load sequence. The amplitudes of all the load cycles above this truncation level was obtained by rain flow counting method [14]. The ΔK_{eff} thus calculated was used in eqn. (3) for determining the crack advance for each cycle to finally obtain 'a' Vs. N_b data.

The predicted fatigue crack growth behavior in an SET specimen of D16 aluminum alloy under all the load sequences are shown in Fig. 5. There appears to be no difference in fatigue rack growth behavior under 'O', M1, M2, and M3. However, the crack growth is delayed under M4 and M5 load sequence. Thus, it may be noted that the fatigue damage caused is not changed in load sequence up to elimination of cycles having stress amplitudes lower than 36 MPa, beyond which the fatigue life is extended. The above results suggest that the small amplitude fatigue cycles (up to $\Delta\sigma$ of 36 MPa) do not have any significant effect on fatigue damage in this material.

If suppose load sequence M3 is used for full scale fatigue testing instead of load sequence O, then there is significant reduction in testing time due to very less number of load reversals in M3 compared to O. However, it may be noted that use of this damage tolerance evaluation approach should be associated with similar studies on crack initiation. The test load sequence should then be decided based on both crack initiation and crack propagation studies.

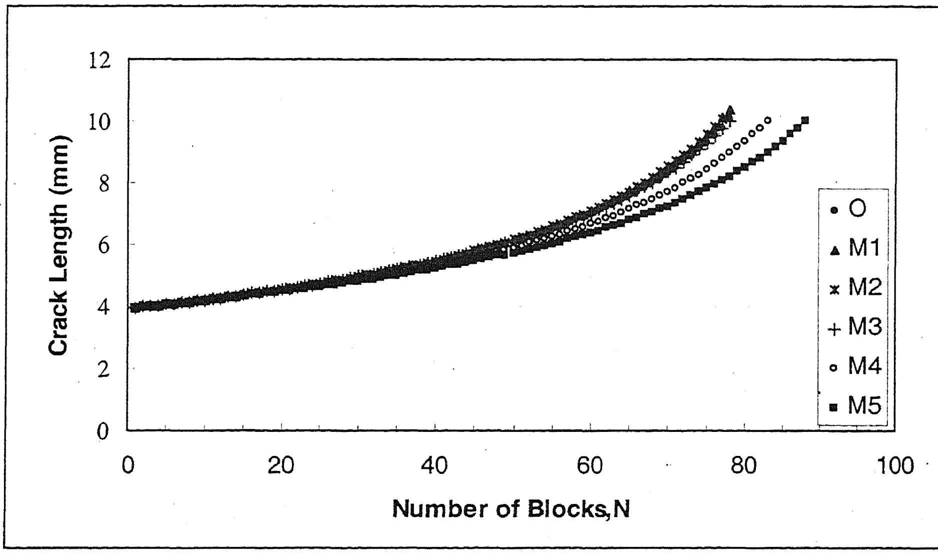

Fig. 5. Predicted crack growth behavior under spectrum loading in D16 aluminum alloy

5. Concluding Remarks

The original flight load sequence of a combat aircraft was modified to yield several different test load sequences. The fatigue crack growth behavior was investigated under all these load sequences. It is shown that full scale fatigue testing time could be significantly reduced by using one of the derived test load sequence which has very less number of load reversals but has almost similar fatigue damaging effect as that of original load sequence. It is suggested that similar studies in fatigue crack initiation should also be carried out before deciding on the required test load sequence.

Acknowledgements

The authors are thankful to Dr. A.R. Upadhya, Director, NAL and Dr. P.K. Dash, Head, SID, NAL for their support and encouragement.

References

1. Lanciotti, A. and Lazzeri, L., "Effects of spectrum variations on fatigue crack growth", Int. J Fat., 1992,14(5), pp. 319-324.

2. Schon, J. and Blom, A., "Fatigue life prediction and load cycle elimination during spectrum loading of composites" Int. J Fat., 2002, 24, pp. 361-367.

3. Nyman, T. Ansell, H. and Blom, A., "Effects of truncation and elimination on composite fatigue life", Compos. Struct., 2000,48, pp. 275-286.

4. Wang, S.S. and Chu, R.C., "Full scale fatigue crack growth test of advanced jet trainer AT-3", Theor. Appl. Fract. Mech., 1989, 11, pp. 71-91.

5. Swift, T., " Verification of methods for damage tolerance evaluation of aircraft structures to FAA requirements", Proceedings of 12th ICAF symposium, Toulouse, France, 1983, Paper 1.1

6. ASTM E647-95, "Standard test method for measurement of fatigue crack growth rates", American Society for Testing and Materials, PA, 1997, Vol. 3.01.

7. Manjunatha, C.M. and Parida, B.K., "Prediction of fatigue crack growth after single overload in an aluminum alloy", AIAA Journal, 2004, 42(8), pp. 1536-1542.

8. Schijve, J., Skorupa, M., Skorupa, A., Machniewicz, T. and Gruszczynski, P., "Fatigue crack growth in the aluminum alloy D16 under constant and variable amplitude loading", Int. J Fat., 2004, 26, pp. 1-15.

9. Schijve, J., "Some formulas for the crack opening stress level", Engg. Fract. Mech., 1981, 14, pp. 461-465.

10. Newman, J.C. Jr., "A crack-closure model for predicting fatigue crack growth under aircraft spectrum loading. In: Methods and Models for Predicting Fatigue Crack Growth Under Random Loading. ASTM STP 748, Philadelphia, PA, 1981, pp. 53-84.

11. Schijve, J., "Prediction methods for fatigue crack growth in aircraft material", In: Fracture Mechanics: Twelfth Conference, ASTM STP 700, PA, 1980, pp. 3-34.

12. Khalil, M. and Topper, T.H., "Prediction of crack opening stress levels for 1045 as-received steel under service loading spectra", Int. J Fat., 2003, 25, pp. 149-157.

13. Fleck, N.A. and Smith, R.A., "Fatigue life prediction of a structural steel under service loading", Int. J Fat., 1984, 6(4), pp. 203-210.

14. ASTM E-1049 "Standard practices for cycle counting in fatigue analysis", American Society for Testing and Materials, PA, 1997, Vol. 3.01.