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Effect of Airframe Materials on Spectrum Modification for Full Scale Fatigue Testing of an Ageing Aircraft

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Abstract

In order to accelerate full scale fatigue testing, the original flight load spectrum of a combat aircraft was modified by eliminating small amplitude load excursions lower than a prefixed filter value. Different levels of filter value were used to derive several different test load spectrum having lesser number of fatigue cycles than the flight load spectrum. The fatigue crack growth behavior under all these spectrum load sequences was predicted in two different airframe materials viz., D16 aluminum and Ti6Al4V alloys using a fatigue crack growth model derived from constant amplitude fatigue crack growth tests, which incorporated crack closure effects. Results show that the basis of spectrum modification should not only be the elimination of small amplitude load cycles but also on the fatigue crack growth behavior of the structural materials used in airframe construction.

Key words: Full scale fatigue test, Spectrum load, Airframe materials.

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 a fleet of aging aircraft is loaded under spectrum sequence for a specified number of flight hours and the airframe is inspected intermittently 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 selected fleet 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 employed 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 sequences, generally containing a very large number of load reversals. Typical load sequence for a combat aircraft may contain in excess of 500 load reversals per flight and a typical block of loading may represent 200 to 250 flight hours of loading.

Owing to large displacements of loading members during full scale fatigue test, the average test frequency for loading is usually kept very low of the order of 0.1-0.5Hz [1]. Therefore, the total number of load reversals in the spectrum sequence is important and controls the testing time required for full scale fatigue test. Omission of a large number of small amplitude load cycles from the flight spectrum that may not cause considerable fatigue damage is a common method to accelerate the test [1-4]. The maximum range of small amplitude load cycles to be eliminated from the flight spectrum depends on the fatigue characteristics of airframe materials.

In this study, several different test load spectra sequences were derived from a flight load spectrum by eliminating small amplitude load excursions below a prefixed filter value. Fatigue

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crack growth behavior was predicted in 2024-T351 aluminum alloy and 4130 structural steel, under all these load spectra, employing crack growth model derived from constant amplitude fatigue crack growth tests. Based on the predicted results, it is shown that the fatigue crack growth behavior of airframe structural materials do significantly influence the derivation of test load spectrum for full scale fatigue test.

2. Modification of Flight Spectrum

Typical flight load spectrum 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 [5]. The total number of load reversals in this spectrum was 85082. The maximum stress was 186MPa and the minimum stress was –47.2MPa.

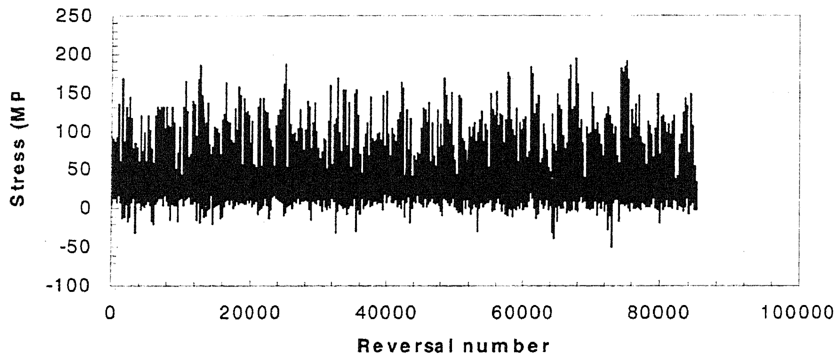

Fig. 1. Original flight load spectrum sequence used in this investigation

Several different test load spectra were derived from the flight load spectrum 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 test load spectrum reduces drastically with elimination of small amplitude load cycles.

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

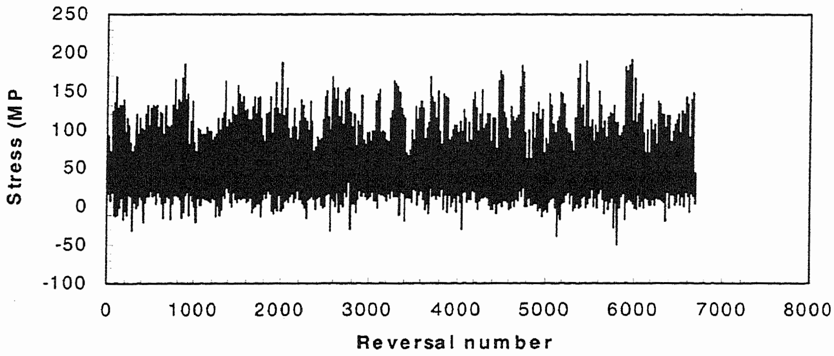

Fig. 2. Modified spectrum load sequence, M2

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, two different airframe materials viz., D16 aluminum alloy and Ti6Al4V Titanium alloy were chosen. The mechanical properties of these materials can be found elsewhere [7-8].

4. Results and Discussion

Constant amplitude fatigue crack growth test data for both materials investigated are shown in Fig. 3. [7,9]. The data is presented in terms of effective crack driving force. Constant amplitude FCGR data shown in Fig. 3 was approximated in the form of Newman's [10] equation as,

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

The constants in equation (1) were determined by fitting a curve to the experimental data in Fig. 3. The solid lines in these figures shows the entire sigmoidal FCGR data using eq. (1).

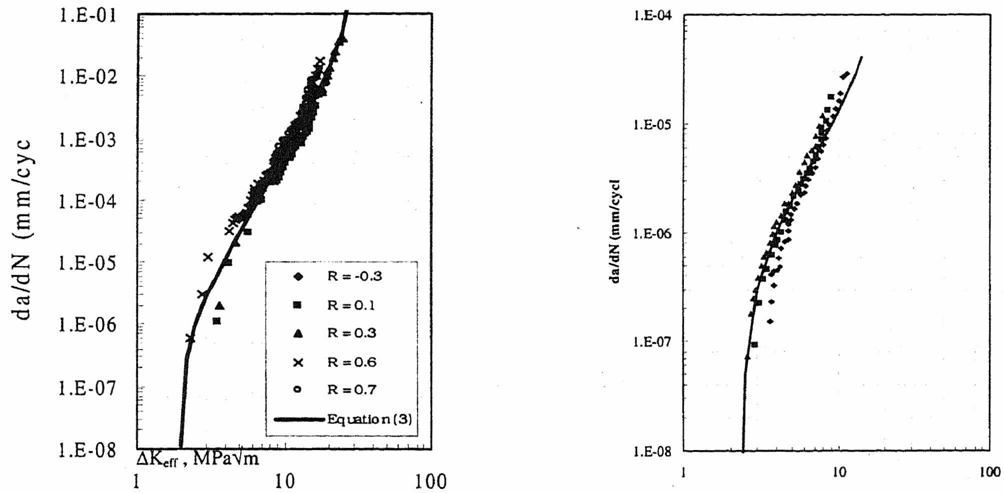

Fig. 3. Constant amplitude FCGR curves for (a) D16 aluminum [7] (b) Titanium alloy [9]

5. Prediction of Fatigue Crack Growth Behaviour

Though crack closure level varies cycle-by-cycle in spectrum loading, use of an average K_{op} provides a reasonable accuracy in life predictions [11]. This average K_{op} depends on the minimum stress ratio (K_{min}/K_{max}) existing in the spectrum sequence [12]. Thus, in the present spectrum, minimum R = -0.2538, and the corresponding closure level was assumed to be 0.39 and 0.37 in aluminum [13] and titanium alloys [8], respectively. Since the stress amplitudes below σ_{op} do not contribute to the crack tip driving force, spectrum was truncated at closure level for these materials. The amplitudes of all the load cycles above this truncation level were obtained by rain flow counting method [14]. The ΔK_{eff} thus calculated was used in eqn. (1) for determining the crack advance for each cycle in spectrum sequence to obtain *a vs.* N_b data. A 45mm wide and 1.5mm thick SET specimen with an initial crack length of 4mm was assumed for predictions. Fig. 4 shows the predicted crack growth behavior in an SET specimen under all the spectrum load sequences in both the materials.

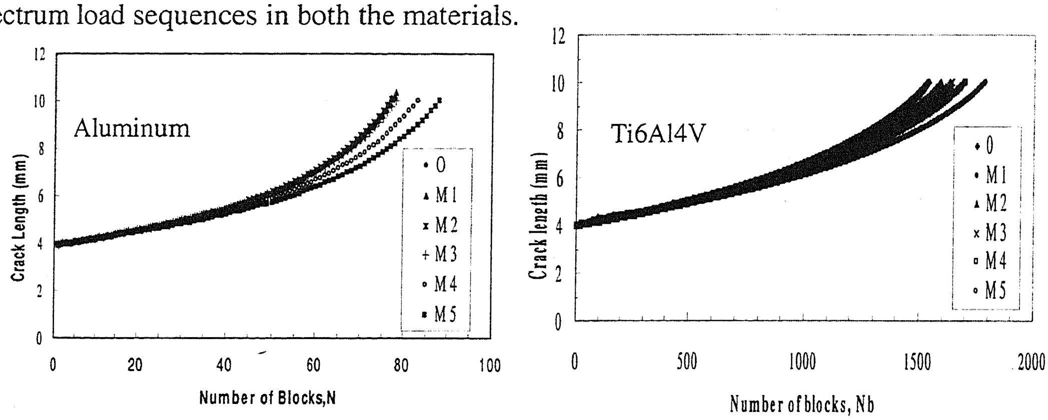

Fig. 4. Predicted crack growth behaviour under spectrum loading in airframe materials.

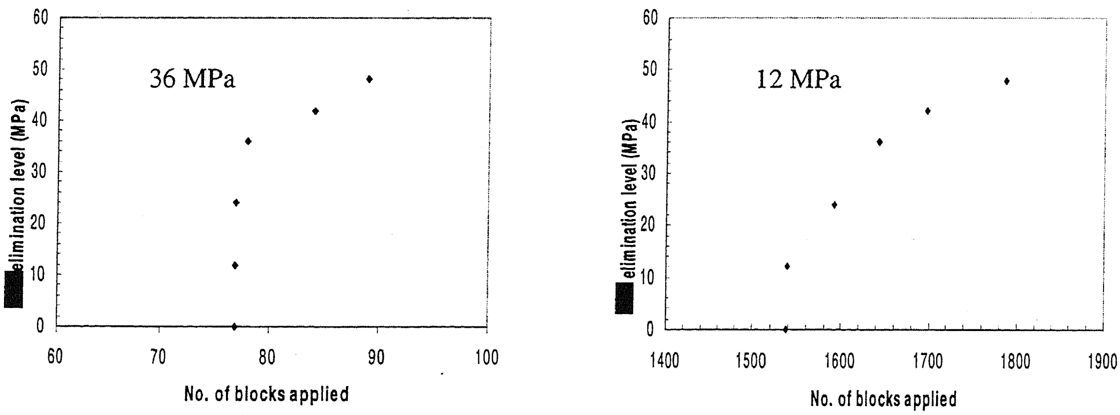

Fig. 5. Total life required to grow the crack from 4 to 10mm in an SET specimen

The total number of load blocks required to grow a crack from 4 to 10mm in an SET specimen of aluminum and Ti alloy is shown in Fig. 5. In aluminum, there appears to be no difference in fatigue rack growth behavior under 'O', M1, M2 and M3. However, a delayed crack growth is noticeable under M4 and M5 spectrum. In titanium, crack growth delay is observed from M3 itself. These results suggest that the small amplitude fatigue cycles up to $\Delta\sigma$ of 12MPa do not have any significant effect on fatigue damage in Ti alloy but the same limit is raised to 36MPa for aluminum alloy. If one were to consider modified spectrum based on studies from aluminum alloys only, the number of reversals would be 2827 (sequence M3). However, when one looks at the other material, Ti alloy, it can be noticed that the sequence M3 has delayed effect on crack growth predictions. Hence, sequence M1 containing 24073 load reversals, even at the cost of increase in testing time during FSFT is advisable to be used. It can be observed from this investigation that it is required to eliminate small amplitude fatigue cycles based on properties of all the materials used in airframe construction.

6. Conclusion

Optimizing the number of load reversals in test load spectrum for full scale fatigue testing of an aircraft should be based not only on the amplitude of small cycles but also on the fatigue crack growth behavior of materials employed in airframe construction.

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