Effects of Compressive Overloads on Fatigue Crack Growth

T. M. Hsu* and L. W. Lassiter†
Lockheed-Georgia Company, Marietta, Ga.

A phenomenological examination of acceleration in the rate of fatigue crack growth on Ti-6AL-4V beta annealed and 7050-T73 plate materials produced by compressive overloads was conducted. The effects of various magnitudes of compressive overloads and the sequence of tension-compression overload cycle with applied load ratio $R = -1.0$ on fatigue crack growth were investigated. The effects of truncation of compressive loads in a realistic flight-by-flight spectrum of the C-141 aircraft were also examined. Specimens used in the test had center cracks and the through-the-thickness cracks emanating from one side of a ¼ in. hole. When the overload cycle was changed from $R = 0$ to $R = -1.0$, with the constant amplitude loads at $R = 0$ and peak values of overload held constant, fatigue crack growth rate was increased by 5 to 40%. The rate increases with the increase of the overload ratio. The influence of sequence of tension-compression overload cycle with $R = -1.0$ on crack growth was less than 10%. The fatigue crack propagation times were shortened less than 15% when the compressive overloads were applied in blocked constant amplitude fatigue tests. Increasing the magnitude of compressive overload did not change the crack growth rate significantly. If the compressive loads in a realistic flight-by-flight spectrum of the C-141 aircraft are truncated, the predicted fatigue crack propagation time will be unconservative by as much as 50%.

Introduction

It has been known for some time that normal crack growth rate under constant amplitude loading changes if the load application is preceded by a different amplitude loading. The tensile overload causes permanent plastic deformation at the crack tip which in turn delays the crack growth of subsequent low load cycles,[1,2] while the compressive overload may accelerate the normal crack growth.[3,4] The importance of delay in the rate of fatigue cracked growth, as produced by tensile overload, on the accurate prediction of fatigue lives of structures has been recognized and quite a few investigations have been stimulated in this area.[5-12] Three most prominent theories[2,13,14] have been proposed to explain the fatigue crack delay phenomenon caused by tensile overload, and several models[15-17] have been suggested to account for the effects of delay on the prediction of fatigue crack growth. The compressive portion of a fully reversed tension-compression cycle contributed substantially to crack growth in constant amplitude fatigue test for high strength alloys,[2] but did not significantly affect crack growth for sheet aluminum alloys.[18-20] Crack growth under constant amplitude cyclic compression has also been observed.[21,22] However there are practically no systematic investigations that have ever been conducted on the effect of crack growth due to the presence of compressive overload.

There are limited data which indicate that the compressive overload will increase the rate of normal constant amplitude crack growth. The effects of compressive loading on fatigue crack growth rate are thus important in the life prediction of aircraft structures. If the compressive overload cycles are neglected, as in current practice in aircraft industry, the life predictions of fatigue crack growth could well be unconservative.

The objective of this investigation was to develop the needed understanding of fatigue crack growth rate as it is affected by the application of compressive overloads when applied in blocked constant amplitude fatigue tests. The effects of the truncation of compressive loads in a realistic flight-by-flight spectrum of the C-141 aircraft were also examined.

Experimental Procedure

To clarify the effects of compressive loads on crack growth during constant amplitude loading, an experimental study was conducted on Ti-6AL-4V beta anneal titanium and 7050-T73 aluminum plate materials. The material ultimate tensile strength, yield strength, and elongation (2 in.), respectively, are 133.3 ksi, 121.3 ksi, and 14.2% for titanium, and 75.0 ksi, 64.1 ksi, and 11.5% for aluminum. Specimens measuring 0.15 X 4 X 16 in. were cut from 1/2 in. plate. Loads were applied in the direction of rolling. Two types of crack geometries as shown in Fig. 1 were considered, namely the center-through-the-thickness cracks (Type A) and through-the-thickness cracks initiated from one side of a 1/4 in. hole (Type B).

An electro discharge machine was used to introduce the initial notch. The specimens were then pre-cracked under fatigue cycling until the initial crack lengths were 0.44 in. for Type A specimens and 0.068 in. for Type B specimens. The test area on all specimens was polished to enhance crack observation and a set of reference lines was marked on the surface of the specimen at about 0.01 in. spacing in the path of the crack. These lines were used to reference crack growth when observed through a variable power binocular

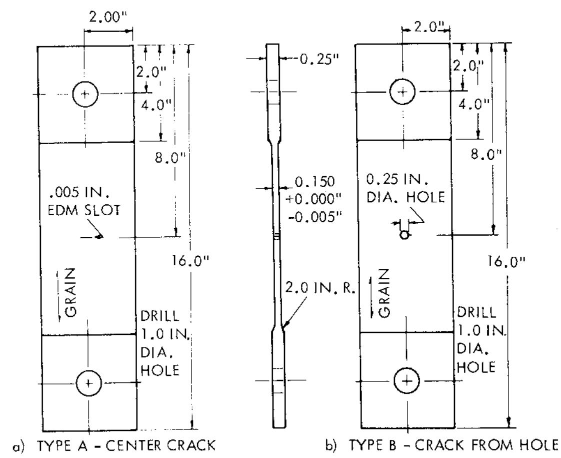

Fig. 1 Test specimen configurations.

Presented as Paper 74-365 at the AIAA/ASME/SAE 15th Structures, Structural Dynamics and Materials Conference; Las Vegas, Nevada, April 17-19, 1974; submitted May 6, 1974; revision received August 12, 1974. Experimental support for this work by W. M. McGee and H. R. Michael is gratefully acknowledged.

Index categories: Aircraft Structural Material; Materials, Properties of.

*Research, Design and Development Engineer—Associate, Advanced Structures Department.

†Assistant to Chief Engineer, Research and Technology.

microscope. A chamber of clear acrylic was placed around the specimen to be tested in room temperature air at 95-100% relative humidity. Humid air was produced by a wash bottle containing distilled water and bled through the chamber at low pressure. Buckling restraint was used when the test involved compressive load.

All tests were performed on an MTS systems closed-loop electro-hydraulic testing machine operated principally at 10 cps. Deviations from this loading frequency were made when an overload or compressive cycle was to be applied. The six basic load spectra used in the tests are shown in Fig. 2. Types A through D spectra were blocked loading. Each block consisted of one overload cycle and 49 cycles of constant amplitude stress at 20 ksi for Ti-6AL-4V specimens and 10 ksi for 7050-T73 specimens.

The overload ratios, which are defined as the ratio of the tensile stress overload and the constant amplitude stress load (σ_2/σ_1), in the Type A spectrum were 1.0, 1.2, 1.5, and 2.0. The overload ratios in the Type B spectrum were 1.2 and 1.5 with $R=-1.0$ for the tension-compression overload cycle, where R is the applied stress ratio and is defined as $\sigma_{\min}/\sigma_{\max}$. The sequence of tension-compression cycles used in Type C loading was fully reversed from Type B. The compressive overload ratio, applied in the Type D spectrum, were -1.2 and -1.5. Type E was a realistic flight-by-flight spectrum of the C-141 aircraft while Type F was a Type E spectrum with all compressive loads truncated.

The typical flight-by-flight spectrum of the C-141 aircraft, tabulated in Table 1, consists of the full and partial loading cycles per average flight. The full-cycle portion occurs in every flight while the partial cycle (or cycles) occurs only after every certain number of flights as shown in Table 1. For example in the test, load layer number 1 was applied 56 cycles for four consecutive flights and then was applied 58 cycles in every fifth flight; while layer number 14 was applied only once in every 18 flights. The entire spectrum was fully repeated after every 270 flights. The basic spectrum given in Fig. 2E and 2F are the complete run showing all load levels. During each test, the basic spectrum was repeated until the specimen failed or was stopped by the operator when the crack extended underneath the buckling support.

The Type A spectrum was used to determine the effects of the amount of tensile overload on the fatigue crack growth of the constant amplitude loads. The results of these tests provided the baseline data for studying the effects of the compressive overload on fatigue crack growth. Spectrum Type B

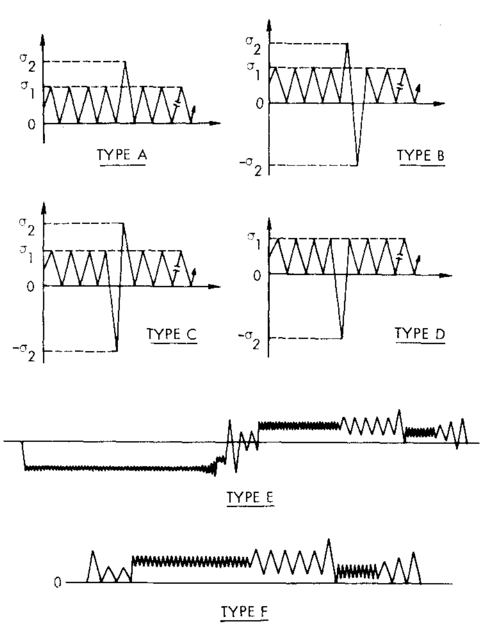

Fig. 2 Load spectra used in test program.

Table 1 Typical flight-by-flight spectrum of C-141 aircraft

Layer no.	Applied Stress (psi) maximum	Applied Stress (psi) minimum	Full cycles per average flight	Partial cycles: cycles	Partial cycles: /flight
1	-13070	-15920	56	2	5
2	-12060	-16940	2	1	9
3	-10880	-18110	. . .	1	10
4	- 9610	-19380	. . .	1	270
5	- 7220	-11390	2	. . .	. . .
6	- 6220	-12400	. . .	1	15
7	13570	-16990	1	. . .	. . .
8	6520	-4400	. . .	2	3
9	12420	7290	24	4	10
10	14940	4760	. . .	5	18
11	19550	150	. . .	1	270
12	9030	2980	10	. . .	. . .
13	10350	1650	1	. . .	. . .
14	12310	- 310	. . .	1	18
15	14930	-2920	. . .	1	270

was designed to investigate the effects of the compressive portion of a tension-compression cycle while spectrum Type C was used to investigate the influences of tension-compression cycle sequence on the fatigue crack growth. The type D spectrum was used to study the change of fatigue crack growth rate by the application of compressive overloads applied in blocked constant amplitude fatigue tests. One of the most common aspects of complex loading which has occasionally been recognized, but which has not received adequate attention, is the contribution of the compression-compression cycles and the compression portion of the tension-compression cycles to the fatigue crack growth. Spectra Types E and F were chosen to examine such effects because of the truncation of the compressive loads of a realistic flight-by-flight spectrum of the C-141 aircraft.

Results and Discussions

The test data from the program are presented in Figs. 3-15. Figure 3 shows the crack growth rate data at $R=0$ obtained from the constant amplitude tests by using both Type A and Type B specimens. Data obtained from the compact tension specimen test at $R=0.1$ were also shown in Fig. 3 for comparison purposes. As can be seen from Fig. 3, the results obtained from three different types of specimens showed good correlation. The results of tests under blocked loading (Types A-D spectra) are presented in Figs. 4-13 while the results of tests under flight-by-flight spectrum loading are given in Figs. 14 and 15. Each curve in Figs. 4-13 is identified by a letter and a number combination. The letter stands for type of blocked spectrum while the number represented the ratio of σ_2/σ_1 as shown in Fig. 2. A cross symbol appearing at the end of a curve denotes failure of the test specimen, while an arrow sign indicates that the crack has grown underneath the specimen's buckling support. Figures 4 and 5 show fatigue crack growth by using Type A and Type B specimens, respectively, and subjected to Spectrum Type A loading with overload ratio equals 1.0, 1.2, 1.5, and 2.0, for Ti-6AL-4V beta annealed material. As can be seen from these two figures, the tensile overload retards the fatigue crack growth, and as expected, retardation increases with higher values of overload for constant values of lower stress level. For the same amount of overload ratio, the difference in retardation effect obtained by using Type A and Type B specimens is negligible. Figures 6 and 7 show the influence of change of overload cycle from $R=0$ to $R=-1.0$ and the influence of tension-compression cycle sequence on fatigue crack growth for Ti-6AL-4V beta annealed material with Type A and Type B crack geometries, respectively.

The overload ratios used in those tests are 1.2 and 1.5. Curves A and B in Figs. 6 and 7 show that when the overload cycle was changed from $R=0$ to $R=-1.0$, with the constant amplitude load and peak value of overload held constant,

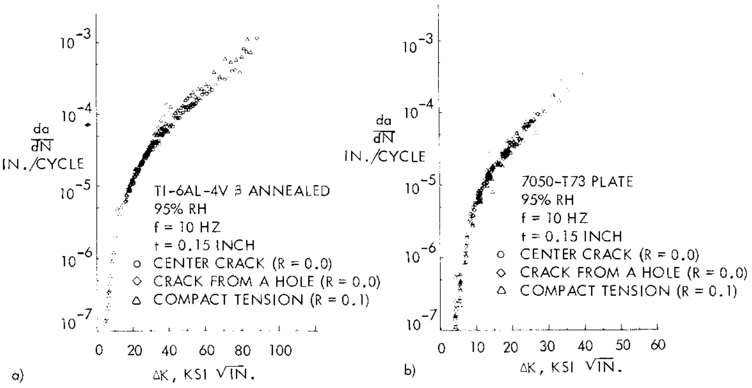

Fig. 3 Crack growth rate for a) Ti-6AL-4V Beta annealed, b) 7050- T73 plate.

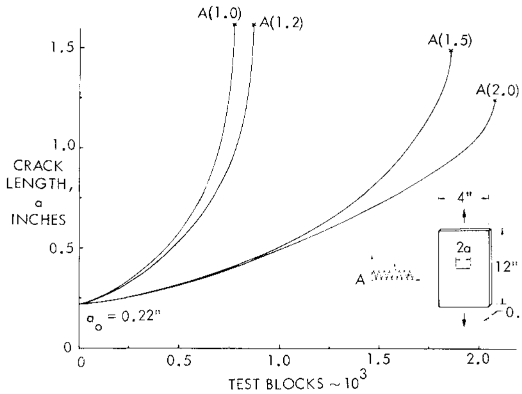

Fig. 4 Effects of tensile overload on crack growth for Ti-6AL-4V Beta annealed plate with center crack.

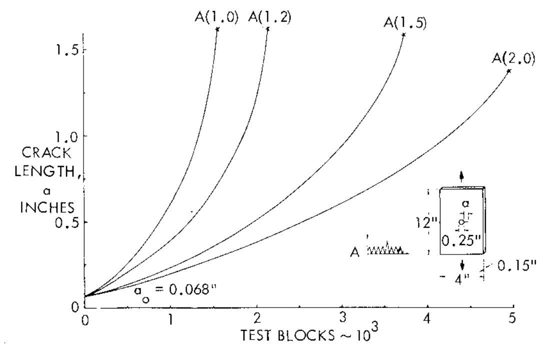

Fig. 5 Effects of tensile overload on crack growth for Ti-6AL-4V Beta annealed plate with crack emanating from hole.

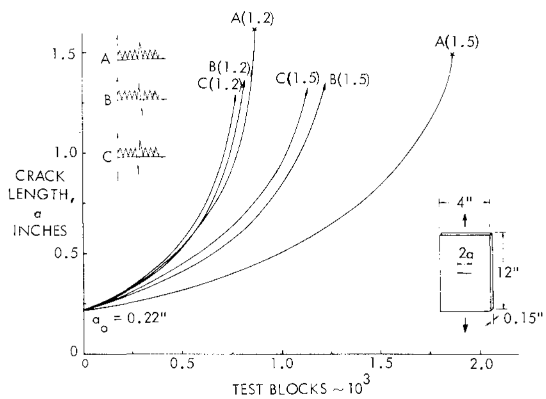

Fig. 6 Effects of change of overload cycle (from $R=0$ to $R=-1.0$) on crack growth for Ti-6AL-4V Beta annealed plate with center crack.

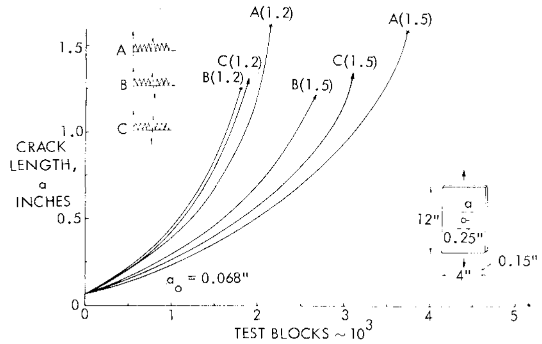

Fig. 7 Effects of change of overload cycle (from $R=0$ to $R=-1.0$) on crack growth for Ti-6AL-4V Beta annealed plate with crack emanating from hole.

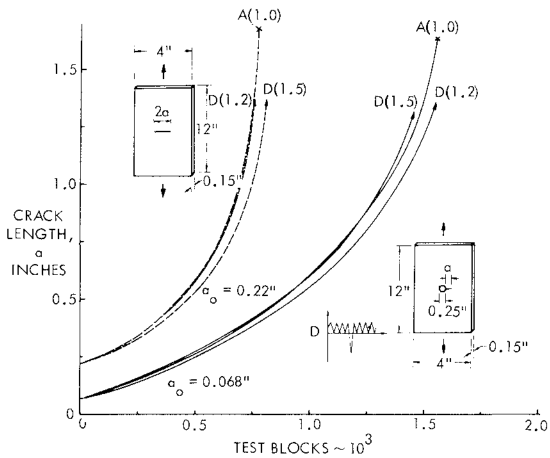

Fig. 8 Effects of compressive overload on crack growth for Ti-6AL-4V Beta annealed plate.

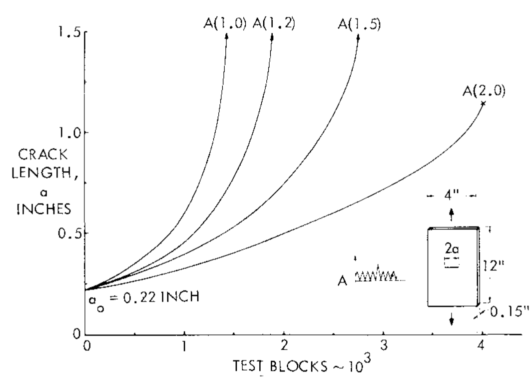

Fig. 9 Effects of tensile overload on crack growth for 7050-T73 plate with center crack.

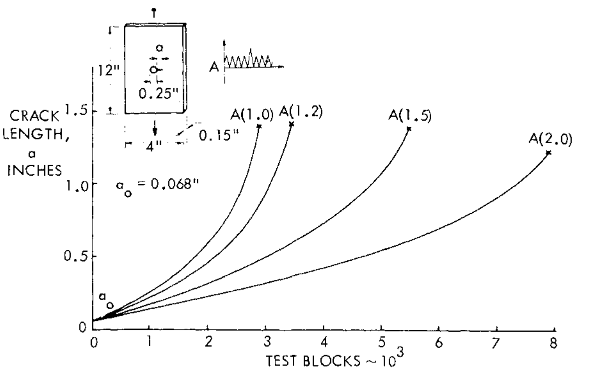

Fig. 10 Effects of tensile overload on crack growth for 7050-T73 plate with crack emanating from hole.

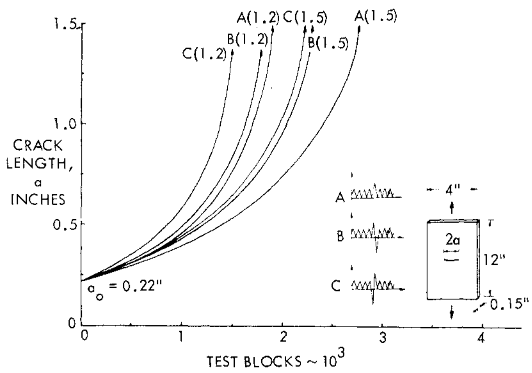

Fig. 11 Effects of change of overload cycle (from $R = 0$ to $R = -1.0$) on crack growth for 7050-T73 plate with center crack.

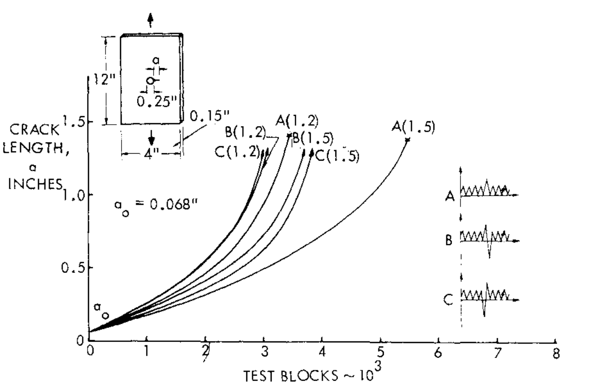

Fig. 12 Effects of change of overload cycle (from $R = 0$ to $R = -1.0$) on crack growth for 7050-T73 plate with crack emanating from hole.

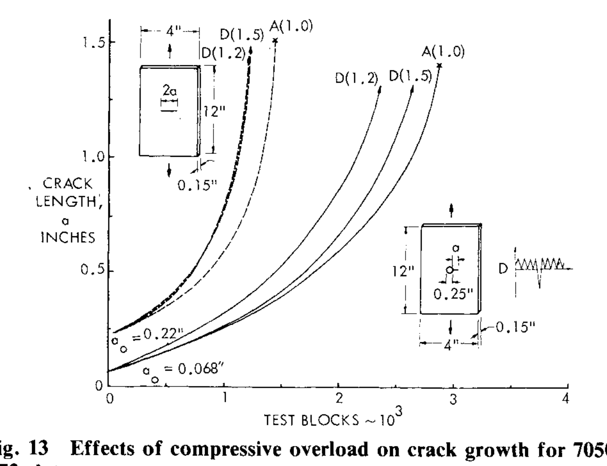

Fig. 13 Effects of compressive overload on crack growth for 7050-T73 plate.

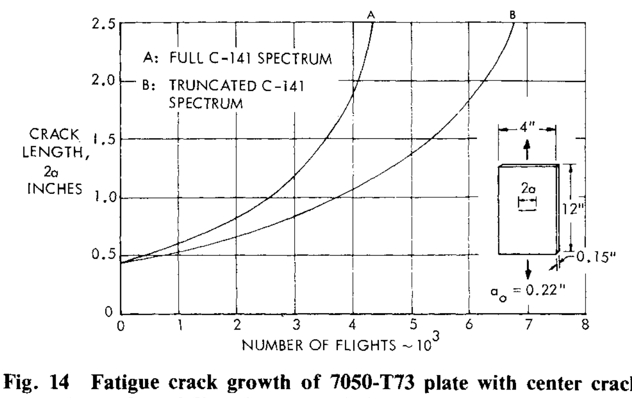

Fig. 14 Fatigue crack growth of 7050-T73 plate with center crack and subjected to full and truncated C-141 aircraft flight-by-flight spectrum.

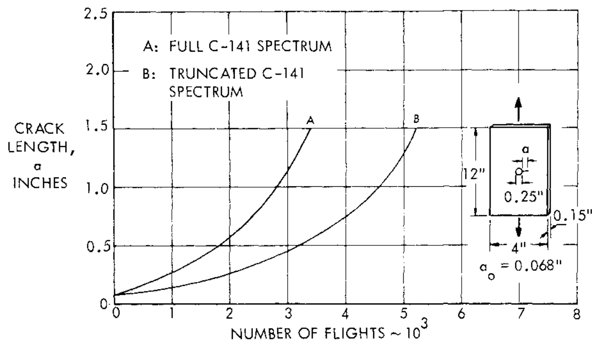

Fig. 15 Fatigue crack growth of 7075-T6 plate with crack emanating from one side of a hole and subjected to full and truncated C-141 aircraft flight-by-flight spectrum.

fatigue crack propagation rate was increased by 5 to 40%, and the rate increased with the increase of the overload ratio. Physically, this means that the compressive portion of tension-compression overload cycles tends to cancel the retardation effect created by a peak tensile overload, and the larger the magnitude of compressive load, the greater the degree of cancellation.

The effect of a tension-compression overload cycle sequence on fatigue crack growth can be observed from comparison of Curve B and Curve C having same overload ratio in Figs. 6 and 7. Normally, if the overload cycle was arranged in a minimum-maximum sequence, one would expect the retardation effect on crack growth to be much stronger than when it was arranged in a maximum-minimum order. However, the results in Figs. 6 and 7 indicate that the influence of sequence of a tension-compression overload cycle with $R = -1.0$ was small (less than 10%). For an overload cycle having an overload ratio $\sigma_2/\sigma_1 = 1.5$ and $R = 0.03$, Porter[17] obtained the difference in crack growth due to change of overload load cycle sequence was about 25% for Ti-6AL-4V mill annealed. From these results, we can see that for constant value of lower stress level and the same amount of overload ratio, the influence of overload cycle sequence increases with the increase of overload applied load ratio R.

Figure 8 shows the effects of various magnitudes of compressive overload on fatigue crack growth for Ti-6AL-4V beta anneal material. The results in Fig. 8 suggested that the effect of application of compressive overload in blocked constant amplitude test on fatigue crack growth was negligible. Increasing the magnitude of compressive overload does not change the crack growth rate significantly.

The results of similar tests on 7050-T73 plate material are presented in Figs. 9-13. Figures 9 and 10 show crack growth for no overload and tensile overloads with overload ratio $\sigma_2/\sigma_1 = 1.2$, 1.5, and 2.0, obtained from the test of Types A and B specimens, respectively. Figures 11 and 12 show results of change of overload cycle from $R = 0$ to $R = -1.0$ for overload ratio $\sigma_2/\sigma_1 = 1.2$ and 1.5. The effects of tensile overload, compressive portion of tension-compression overload cycle and tension-compression overload cycle sequence on crack growth for 7050-T73 plate material were, in general, similar to that of Ti-6AL-4V beta anneal material which has been discussed earlier. However, when compressive overloads were applied in a blocked constant amplitude test, the average fatigue crack growth rate was increased by about 15% as shown in Fig. 13. The increase of the magnitude of compressive overload did not show significant change in crack growth rate.

Figures 14 and 15 show fatigue crack growth under the full and truncated C-141 aircraft flight-by-flight spectrum loading for 7050-T73 plate with center crack and 7075-T6 plate with crack emanating from one side of hole, respectively. Curve A, in Figs. 14 and 15, is the test result using the full flight-by-flight spectrum of C-141 aircraft as shown in Table 1, while Curve B is the result using the same spectrum with all compression loadings were set equal to zero. As can be seen from these two figures, with all the compressive loads truncated, crack propagation time has lengthened by as much as 50% for both materials and both types A and B crack geometries. Hence, if compressive loads are neglected, as is current practice in the aircraft industry without proper adjustments, life prediction of fatigue crack growth could well be unconservative.

Conclusion

Based on the experimental program conducted, some of the conclusions reached may be summarized as follows:

1) A tensile overload retards fatigue crack growth and as expected, retardation increases with higher values of overload for constant values of lower stress level. For the same amount of overload ratio, the difference in retardation effect obtained by using Type A and Type B specimens is negligible.

2) When the overload cycle was changed from $R=0$ to $R=-1.0$, with the constant amplitude loads and peak values of overload held constant, fatigue crack propagation rate was increased by 5 to 40 . The rate increases with the increase of the overload ratio.

3) The influence of sequence of a tension-compression overload cycle with $R=-1.0$ was less than 10%. However, for constant value of lower stress level and the same amount of overload ratio, the influence of overload cycle sequence will increase, when the minimum stress of overload cycle increases.

4) For Ti-6AL-4V beta annealed titanium material, the compressive overload may be neglected if they are preceded by tension load cycles of nominal value and have essentially constant maxima.

5) For 7050-T73 plate material, the fatigue crack propagation times were shortened by about 15% when the compressive overloads were applied in blocked constant amplitude fatigue tests. Increasing the magnitude of compressive overload did not change the crack growth rate significantly.

6) If the compressive loads in a realistic flight-by-flight spectrum of aircraft are truncated without proper adjustment, the predicted fatigue crack propagation time will be unconservative.

References

[1]Schijve, J., "Fatigue Crack Propagation in Light Alloy Sheet Material and Structures," NLR MP 195, Aug. 1960, Amsterdam, Netherlands.

[2]Hudson, C. M. and Hardrath, H. F., "Effects of Changing Stress Amplitude on the Rate of Fatigue Crack Propagation of Two Aluminum Alloys," TN D-960, Sept. 1961, NASA.

[3]Crooker, T. W., "Effect of Tension-Compression Cycling on Fatigue Crack Growth in High Strength Alloys," Rept. 7220, Jan. 1971, U. S. Naval Research Lab., Washington, D.C.

[4]Rice, R. C. and Stephens, R. I., "The Influence of Intermittent Overloads on Subcritical Crack Growth," *Progress in Flaw Growth and Fracture Toughness Testing,* STP 536, July,1973, American Society for Testing Materials, Philadelphia, Pa.

[5]McMillan, J.C. and Pelloux, R.M.N., "Fatigue Crack Propagation Under Programmed and Random Loads," STP 415, June, 1967, American Society for Testing Materials, Philadelphia, Pa.

[6]Hudson, C. M. and Raju, K. W., "Investigation of Fatigue Crack Growth Under Simple Variable-Amplitude Loading," TN D-5702, March 1970, NASA.

[7]Von Euw, E.F.J., "Effect of Overload Cycle on Subsequent Fatigue Crack Propagation in 2024-T3 Aluminum Alloy," Ph. D. thesis, March,1971, Department of Mechanical Engineering, Lehigh University, Bethlehem, Pa.

[8]Von Euw, E.F.J., Hertzberg, R. W., and Roberts, R., "Delay Effects in Fatigue Crack Propagation," Stress Analysis and Growth of Cracks, STP 513, Sept.,1972, American Society for Testing Materials, Philadelphia, Pa.

[9]Hsu, T.M., "Investigation of Fatigue Crack Growth Under Simple Variable-Amplitude Loading," Rep., SMN 372, Feb., 1973, Lockheed-Georgia Company, Marietta, Ga.

[10]Corbly, D.M. and Packman, P.F., "On the Influence of Single and Multiple Peak Overloads on Fatigue Crack Propagation in 7075-T6511 Aluminum," *Engineering Fracture Mechanics,* Vol. 5, June, 1973, pp. 479-497.

[11]Trebules, V.W., Jr., Roberts, R., and Hertzberg, R.W., "Effect of Multiple Overloads on Fatigue Crack Propagation in 2024-T3 Aluminum Alloy," *Progress in Flaw Growth and Fracture Toughness Testing,* STP 536, July, 1973, American Society for Testing Materials, Philadelphia, Pa.

[12]Wei, R.P., Shih, T.T., and Fitzgerald, J.H., "Load Interaction Effects on Fatigue Crack Growth in Ti-6AL-4V Alloy," CR-2239, April, 1973, NASA.

[13]Hudson, C.M. and Hardrath, H.F., "Investigation of Variable Loadings of Fatigue Crack Propagation Patterns," TN D-1803, Aug., 1963, NASA.

[14]Elber, W., "The Significance of Crack Closure," *Presented at 1970 Annual Meeting of the American Society for Testing Materials,* Toronto, Canada, June 1970.

[15]Wheeler, O.E., "Spectrum Loading and Crack Growth," *ASME Transactions: Journal of Basic Engineering,* March 1972, pp. 181-186.

[16]Willenborg, J., Engle, R.M., and Wood, H.A., "A Crack Growth Retardation Model Using an Effective Stress Concepts," TM 71-1-FER, Jan. 1971, Air Force Flight Dynamics Lab, Wright Patterson Air Force Base, Ohio.

[17]Porter, T.R., "Method of Analysis and Prediction for Variable Amplitude Fatigue Crack Growth," *Engineering Fracture Mechanics,* Vol. 4, Dec., 1972, pp. 717-736.

[18]Illg, W. and McEvily, A.J., Jr., "The Rate of Fatigue Crack Propagation for Two Aluminum Alloys Under Completely Reversed Loading," TN D-52, Oct., 1959, NASA.

[19]Donaldson, D.R. and Anderson, W.E., "Crack Propagation Behavior of Some Airframe Materials," *Proceedings, Crack Propagation Symposium,* Vol. 2, Cranfield, England, Sept., 1961, p. 375.

[20]Hudson, C.M. and Scardina, J.T., "Effect of Stress Ratio on Fatigue-Crack Growth in 7075-T6 Aluminum Alloy Sheet," Engineering Fracture Mechanics, April, 1969, pp. 429-446.

[21]Hubbard, R. P., "Crack Growth Under Cyclic Compression," *ASME Transactions: Journal of Basic Engineering,* Dec., 1969, pp. 625-631.

[22]Saal, H., "Fatigue Crack Growth in Notched Parts with Compressive Mean Load," *ASME Transactions: Journal of Basic Engineering,* March 1972, pp. 243-247.