25

FACTORS IN EVALUATING FATIGUE LIFE OF STRUCTURAL PARTS

By Walter Illg

Langley Research Center

SUMMARY

L
1
4
3
4

Three facets of fatigue testing are discussed in relation to problems involved in evaluating the fatigue life of structural parts. These facets are variable-amplitude loading, fatigue-crack propagation, and equivalent fatigue loading. Experimental test results are included to support conclusions.

INTRODUCTION

In the interest of safety and according to regulations, it must be shown that an aircraft structure has a practically infinite life or a safe life must be established. At present, theory alone is not sufficient to evaluate the fatigue characteristics of a structural part. The only alternative is testing. Simple laboratory specimens are unsuitable because of the many factors which must be simulated from the full-scale parts. At the present stage of fatigue knowledge, testing the actual part is the only sure way to evaluate the fatigue life of a piece of aircraft structure. In this paper some recent NASA research, which sheds light on a number of factors that are involved in any evaluation of the fatigue life of structural parts, is discussed.

SERVICE LOADING

The first problem in testing is how to load a specimen to simulate service loading. A record of actual stresses encountered in flight can be obtained with the use of strain gages at critical points. The record will depend a great deal upon how the aircraft is used. In the event the same vehicle is to be used for more than one type of mission, the flight record to be used in testing should be from that mission which provides the severest load conditions. Each type of mission will include a variety of maneuvers, such as taxiing, take-off, and climb, which have characteristic load histories. The record should include sufficient samples of all phases of the aircraft's mission to give an accurate picture of the extremes and probabilities of occurrence of all loads.

It may or may not be possible to combine the loads from different phases, depending on the mean load. Schedules for phases having widely different mean loads should not be combined. In the case of the helicopter, the mean load in the blade for all phases will be practically identical since it is dependent upon rotor angular velocity which is essentially constant. The parts in other aircraft, however, will generally be subjected to varying mean loads.

It would be very satisfying if the exact flight-loading patterns could be reapplied to the part in question in the laboratory; however,
the cost of such a procedure would be prohibitively high. On the other L
extreme, cycling continuously at a single load range has been shown to 1
be incapable of reproducing all the effects of the varied flight loadings. 4
For instance, fatigue tests of transport wings (ref. 1) indicated that 3
more cracks were initiated under variable amplitude than under constant 4
amplitude, and the crack which finally caused failure was in a different location for each type of loading.

VARIABLE-AMPLITUDE TESTING

Since constant-amplitude loading is unrealistic and exact flight-loading duplication is impractical, a compromise must be adopted. One solution is to translate the flight record into a load program composed of discrete steps which adequately represent the actual loads and which can be handled in the laboratory. Figure 1 illustrates how a step-loading schedule may be used to simulate a continuous spectrum. The smooth curve is a representative stress spectrum found from actual service conditions during one major phase of operations. The discrete levels shown are chosen so that they represent a number of equal stress intervals. The best number of steps to use has not been definitely established. In NASA work at the Langley Research Center, eight steps are generally used, and possibly no fewer than six should ever be used.

As figure 1 implies, all loads are assumed to cause fatigue damage. Even though the lowest load step lies below the fatigue limit, it is nonetheless important. One reason is that, after a crack appears, the stress-concentration factor increases, and the lowest step may contribute to crack propagation. For this reason, the test schedule should include loads below the fatigue limit. As mentioned before, a similar schedule will be obtained for each major phase of operation of the aircraft. The various test schedules should be applied in the same proportion that is expected to occur in service.

Once the magnitude and number of load steps have been determined, the sequence of application must be considered. Tests on simple specimens under variable-amplitude loading have been conducted, and one discovery has been the effect of the sequence of loading on the life. Figure 2 shows the relative effect of three loading sequences on the fatigue life of the specimens tested (ref. 2). Each of these patterns depicts a different sequence of applying the same eight loads the same number of times, each on the same type of simple specimen. The loading pattern was repeated until failure occurred. The life for the random sequence is given a value of 1 for purposes of comparison. Note that the lo-hi sequence gave the shortest life while the hi-lo sequence gave the longest life. The life obtained with the random sequence fell between the other two. Thus, by change of sequence the results changed by a factor of 4.

In general, service loading of aircraft is of a random nature, in which case, the laboratory test loads should be applied in random order. Special cases may require other sequences to simulate the loading. The number of cycles in each block (fig. 2) should be chosen so that the specimen will survive at least 10 blocks.

FATIGUE-CRACK PROPAGATION

Under the application of any load schedule, the failure will take place in two stages; the first stage is the period before a crack is initiated, and the second stage is the period during which the crack is propagated to failure. The interval between crack initiation and part failure can be very important to an aircraft operator. Fatigue damage is first visible during this interval in the life of a part. During this time, the strength of the part decreases in a manner as shown in figure 3. The dashed line represents the maximum repeated load encountered in service. The solid line indicates the load which the cracked part can still carry. As the crack grows, the strength decreases until the part can no longer withstand the applied load, and failure occurs. This interval between crack initiation and failure provides an opportunity to inspect a part for fatigue damage. Two properties of the part must be known to make it possible to set up a realistic inspection interval. The first property is the rate of crack propagation under expected stresses, and the second property is the crack length at which the residual strength equals the load and the part fails.

Prediction of the rates of crack propagation in simple specimens at constant-amplitude loading is possible. Figure 4 shows some results for 2024-T3 sheet specimens tested with the ratio of minimum stress to maximum stress in a cycle R always at zero (ref. 3). The data for many tests fall on a continuous curve when the rate of crack propagation is plotted against a stress parameter which involves the local stress at the tip of the crack. The dashed line represents a semiempirical expression used to correlate the data shown. The same curve may be used to estimate the rate of crack propagation at other stresses and for other size specimens. Crack propagation under variable-amplitude loading is currently being investigated to understand more fully crack growth under service-type loading.

The second property of the part which must be known (the residual strength as a function of crack length) has also been investigated for simple specimens. Some results are shown in figure 5 for 2024-T3 material (ref. 4). Note the sharp decrease in strength for small crack lengths. The solid line represents a theory for predicting strengths for any crack length. This theory is based on a stress concentration at the tip of the crack. When the local stress at the tip of the crack reaches the ultimate strength of the material, the specimen will fail. When the part is reasonably simple, this kind of data from simple specimens can provide a fairly good estimate of the number of cycles which the part can endure after a crack appears. For complicated structures, the part itself would have to be tested to obtain all crack-rate data and residual-strength data. In any case, the part should be tested to check the estimates made. From this information, an inspection interval could be calculated which would make it reasonably certain that a crack would be discovered before failure occurred. For the method to be practical, the probable locations of cracks must be known and easily accessible. Of course, in the event that the inspection interval is found to be too short, the parts must be discarded before a crack appears.

L 1 4 3 4

EQUIVALENT FATIGUE LOADING

In the course of an investigation of crack propagation, it may sometimes be desirable to substitute one loading ratio R for another, while maintaining identical rates of crack propagation. This procedure might be desired for practical reasons such as testing machine capability. Extreme care must be exercised in choosing an equivalent load. The following discussion illustrates how conventional methods for finding equivalent loads can result in invalid crack-propagation test data. The illustrative problem involves changing from a loading at $R = 0$ to a loading at $R = -1$.

The conventional method of finding equivalent fatigue loads is indicated in figure 6. The fatigue failure curves for unnotched specimens are plotted for the two load ratios $R = 0$ and $R = -1$. The ordinate is the maximum cyclic load and the abscissa is the total life. The conventional method requires the determination of the load level at $R = -1$ which gives the same life as the original load level at $R = 0$. However, the conventional method is invalid when dealing with the crack-propagation portion of fatigue tests.

Figure 7 shows the rates of fatigue-crack propagation for the two load ratios $R = 0$ and $R = -1$ (ref. 3). The material is 7075-T6 aluminum-alloy sheet. One set of tests was run with the minimum cyclic load held at zero ($R = 0$), and the other set of tests was run with the mean load held at zero ($R = -1$). These data show that, for the same maximum cyclic stress, the rates of crack propagation were the same even though the load range for $R = -1$ was twice that for $R = 0$.

In a recent investigation of the crack propagation in a full-scale helicopter rotor blade, the conventional method was used to change from a service loading at $R = 0$ to a test loading at $R = -1$. Since figure 7 indicated that in simple specimens the maximum cyclic stresses for similar cracking rates at $R = 0$ and $R = -1$ are the same, axial-load fatigue tests on small portions of the blade were performed to check the results of the full-scale test. One specimen was subjected to service loading and another specimen was subjected to the stresses that were used in the full-scale test. The results of these tests are shown in figure 8. The loading at $R = -1$, the so-called equivalent loading, gave a much smaller rate of crack length than did the service loading.

The result, then, of using an equivalent load obtained in the conventional manner, was that the full-scale test not only did not yield the answer that was sought but also gave a misleading indication of comparatively slow crack growth.

CONCLUDING REMARKS

It has been shown that many factors enter into the determination of the safe life of a structural component subjected to fatigue loading.

It is preferable to test actual parts because of the multitude of items which must be duplicated in simple specimens to make a valid test.

Variable-amplitude testing is preferred to constant-amplitude testing. The number of load steps for a given schedule should probably not be less than six. The sequence of loading should be random for most cases. The number of cycles per block should be such that at least 10 blocks will be survived by the part.

Loads below the fatigue limit should be included since they can affect the life of the part.

Inspection for cracks during the service life of an aircraft can improve its safety. When the procedure discussed in this paper for determining an inspection interval is used, the following properties must be ascertained:

(1) The probable location of fatigue cracks

(2) The rate of growth of all fatigue cracks

(3) The length of the crack at which the residual strength is no longer greater than expected loads

When the rates of crack propagation are investigated, great care should be taken in determining equivalent loads if they are to be used.

L
1
4
3
4

REFERENCES

1. Whaley, Richard E.: Fatigue Investigation of Full-Scale Transport-Airplane Wings - Variable-Amplitude Tests With a Gust-Loads Spectrum. NACA TN 4132, 1957.

2. Naumann, Eugene C., Hardrath, Herbert F., and Guthrie, David E.: Axial-Load Fatigue Tests of 2024-T3 and 7075-T6 Aluminum-Alloy Sheet Specimens Under Constant- and Variable-Amplitude Loads. NACA TN D-212, 1959.

3. McEvily, Arthur J., Jr., and Illg, Walter: The Rate of Fatigue-Crack Propagation in Two Aluminum Alloys. NACA TN 4394, 1958.

4. McEvily, Arthur J., Jr., Illg, Walter, and Hardrath, Herbert F.: Static Strength of Aluminum-Alloy Specimens Containing Fatigue Cracks. NACA TN 3816, 1956.

SIMULATION OF CONTINUOUS STRESS SPECTRUM BY USE OF REPRESENTATIVE STRESS STEPS

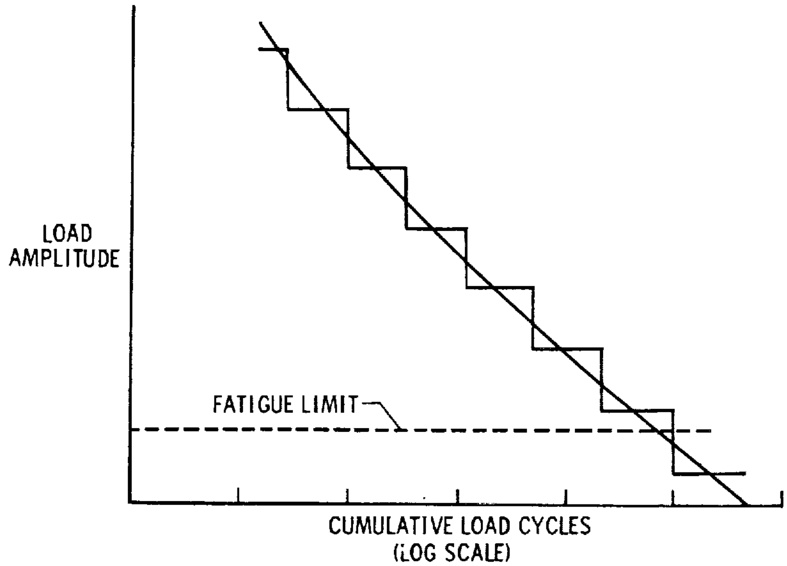

Figure 1

EFFECT OF TEST SEQUENCE ON FATIGUE LIFE

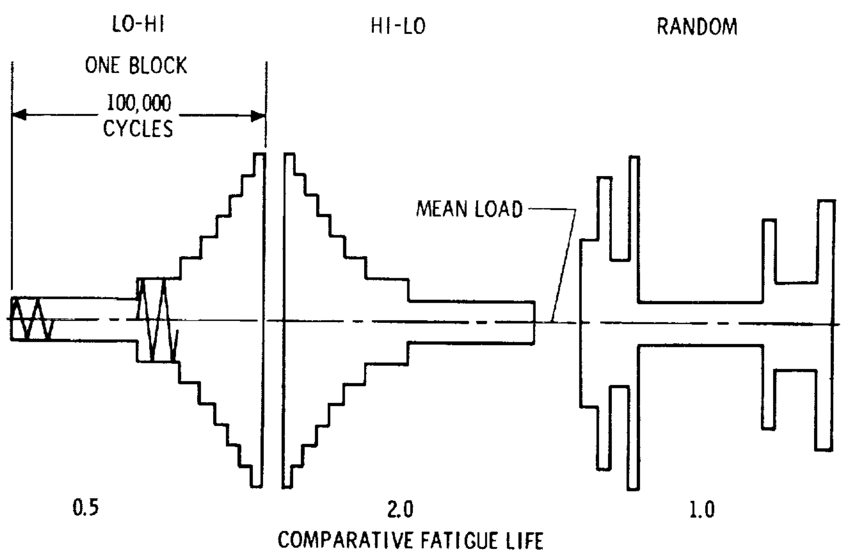

Figure 2

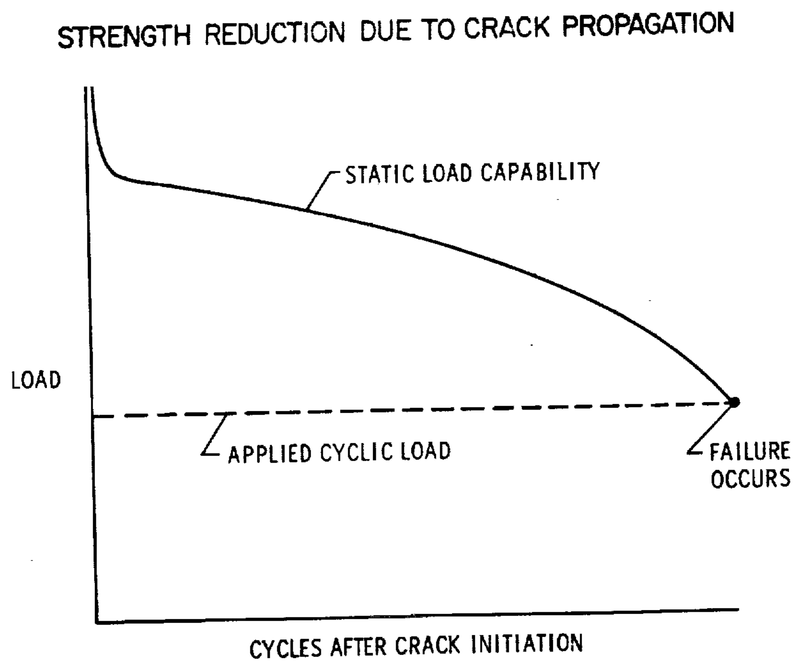

Figure 3

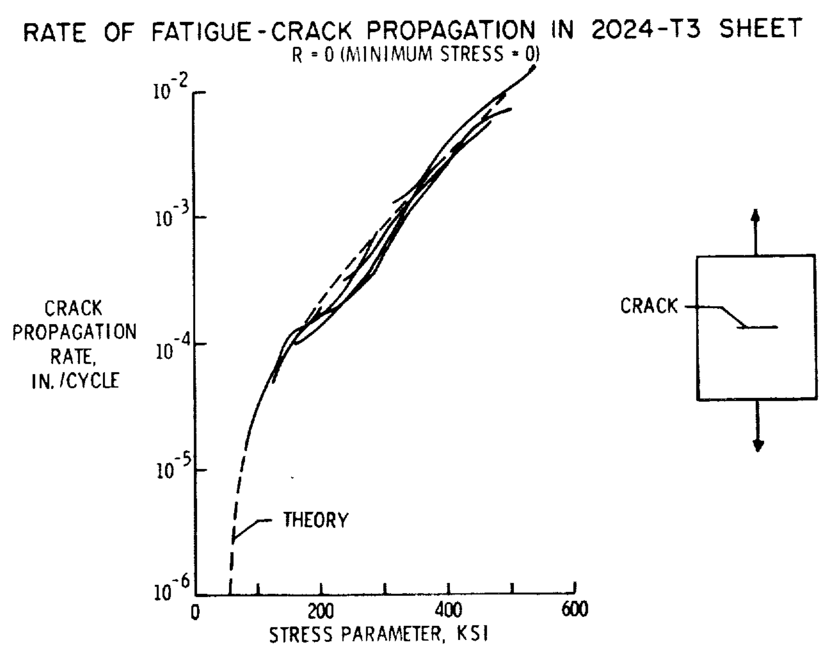

Figure 4

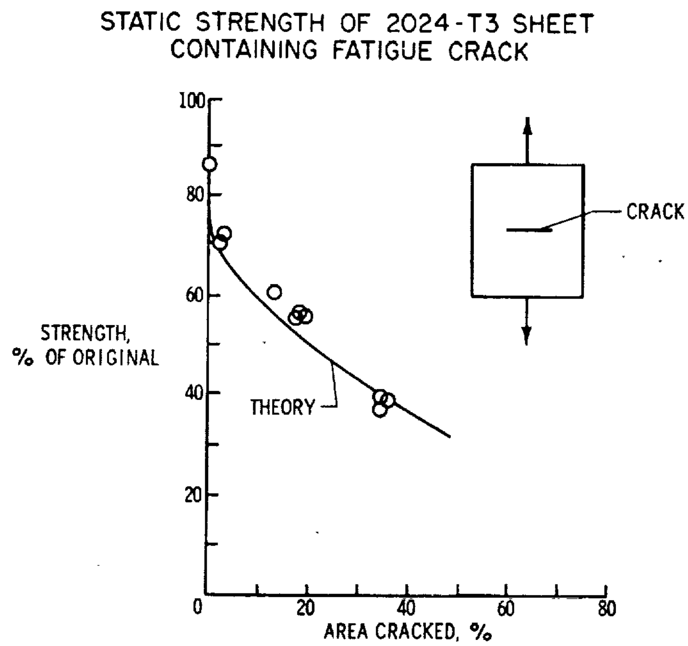

Figure 5

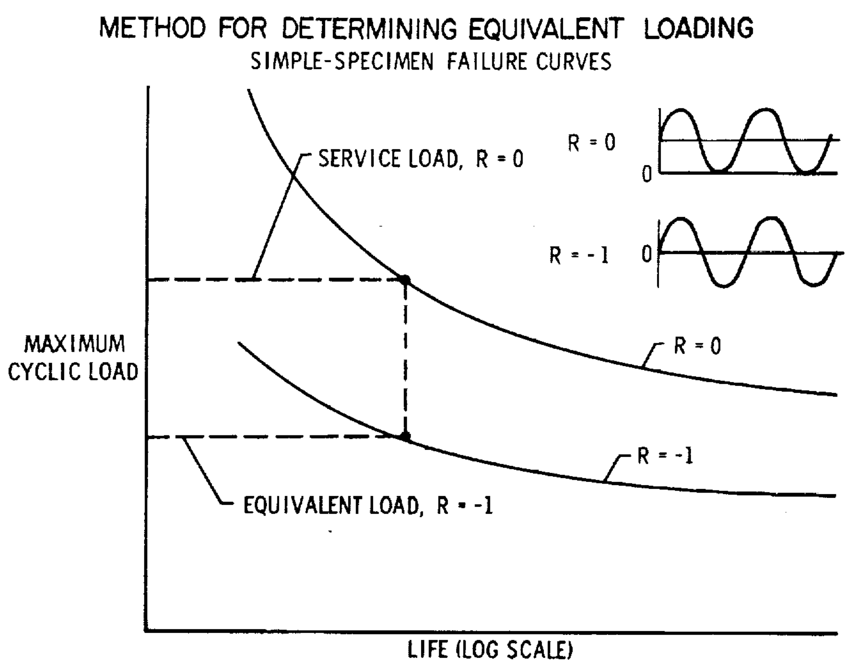

Figure 6

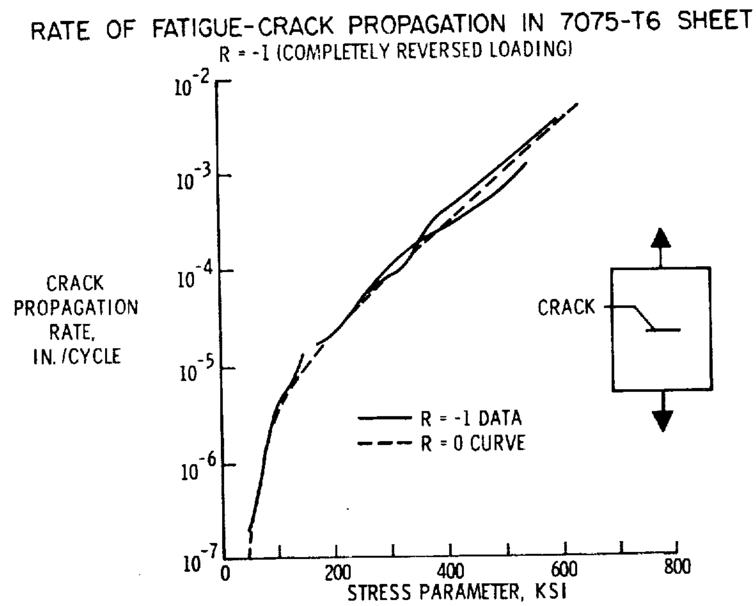

Figure 7

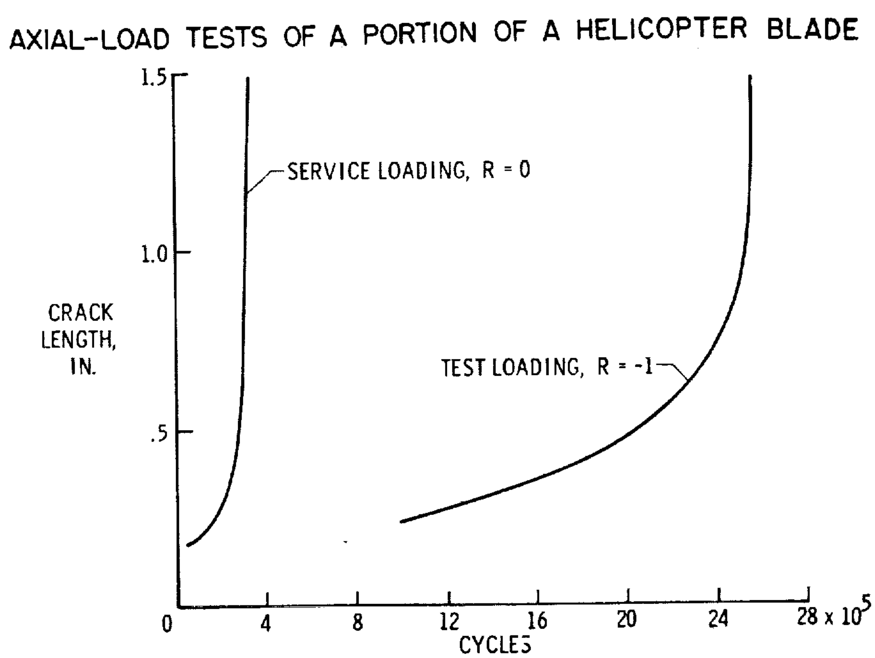

Figure 8