FATIGUE TESTS WITH RANDOM FLIGHT-SIMULATION LOADING

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SUMMARY

Crack propagation was studied in a full-scale wing structure under different sim-ulated flight conditions. Omission of low-amplitude gust cycles had a small effect on the crack rate. Truncation of the infrequently occurring high-amplitude gust cycles to a lower level had a noticeably accelerating effect on crack growth. The application of fail-safe load (100 percent limit load) effectively stopped subsequent crack growth under resumed flight-simulation loading.

In another flight-simulation test series on sheet specimens, the variables studied are the design stress level and the cyclic frequency of the random gust loading. In-flight mean stresses vary from 5.5 to 10.0 kg/mm^2. The effect of the stress level is larger for the 2024 alloy than for the 7075 alloy. Three frequencies were employed: namely, 10 cps, 1 cps, and 0.1 cps. The frequency effect was small.

The advantages and limitations of flight-simulation tests are compared with those of alternative test procedures such as constant-amplitude tests, program tests, and random-load tests. Various testing purposes are considered. The variables of flight-simulation tests are listed and their effects are discussed.

A proposal is made for performing systematic flight-simulation tests in such a way that the compiled data may be used as a source of reference. The data could be used for estimating fatigue properties in the design stage of an aircraft and serve other purposes as well.

INTRODUCTION

The fatigue loads in a flight-simulation test are supposed to be a valid represen-tation of the loading conditions in service. In the present paper, an attempt is made to analyze the merits and the limitations of flight-simulation fatigue tests.

Recently at the National Aerospace Laboratory, extensive studies have been made regarding fatigue crack propagation under random flight-simulation loading in 2024-T3 and 7075-T6 aluminum alloys (refs. 1 to 3). Tests on crack propagation in a full-scale wing structure have just been completed and the results are summarized herein. Another program on the frequency effect under flight-simulation loading is halfway

finished. It also includes the effect of the design stress level. Preliminary data are given in a subsequent section.

A comparison is made between flight-simulation testing and alternative test procedures. The objectives of testing are obviously important for such a comparison. The variables of flight-simulation testing are listed and data on the significance of the variables are reviewed. The information is fairly limited as yet. Finally, a proposal is made for performing systematic flight-simulation tests in order to compile data that may be used as a source of reference.

TWO RECENT NATIONAL AEROSPACE LABORATORY TEST PROGRAMS

Crack Propagation in a Full-Scale Wing Structure Under Random Flight-Simulation Loading

The fatigue test and the fail-safe tests on the wing of the F-28 aircraft were successfully completed in April 1970. It was then decided to employ the test setup and the wing for a general crack propagation investigation. The aim of the investigation was to study the effects of

(1) omitting low-amplitude gust cycles
(2) truncating very-high-amplitude gust cycles to a lower level
(3) increasing the stress level by 25 percent
(4) preceding limit loads delaying subsequent crack growth.

The wing loads applied by 12 hydraulic cylinders included gust loads, ground-to-air cycles, flap loads, and undercarriage loads. The load sequence was a random flight simulation similar to the sequences applied in the certification tests on the wing and in tests on sheet specimens (refs. 1 to 3). A sample of a load record is given in figure 1, and the test setup is shown in figure 2. The gust load spectrum was approximated by a stepped function. (See fig. 3.)

Three test series (A, B, C) were carried out and data from the certification tests (series R) were also used for comparison. In the certification tests the low-amplitude cycles were not omitted. The omission of these cycles in test series A reduced testing time from 116 seconds to 46 seconds per flight; this implies a considerable time saving. The certification tests were completed by three fail-safe tests up to limit load, and this allowed observations on the effect of limit load application on subsequent crack growth during test series A, B, and C.

The numbers of simulated flights are indicated in figure 3. In each test series (A, B, C), groups of 2 to 4 artificial cracks were applied in the following elements: skin

between stringers, skin and stringer flange, stringer head (Z-stringer), and spar web. The cracks were applied at similar locations with approximately the same stress level in all test series. In the certification tests (series R), a large number of cracks were tested; however, for comparison with the present test series only a few cracks could be used in view of the similarity of location and stress level.

The complete results as given in reference 4 showed low scatter in each group of 2 to 4 similar cracks. A summary of the average results is presented herein. As an illustration, figure 4 shows average crack propagation curves for the skin when the stringer flange is also cracked. From similar curves, average crack rates were drawn and comparisons are made in this study.

Effect of omitting low-amplitude gust cycles.- The effect of omitting low-amplitude gust cycles from flight-simulation loading (comparison of results of test series A and R) is presented in the following table:

Cracked component	Material	Average effect on crack rate
Skin between stringers	2024-T3	Slightly slower
Skin at stringer	2024-T3	About equal
Stringer flange	7075-T6	1.5 times slower
Head of stringer	7075-T6	2 times faster
Spar web	7075-T6	1.5 times slower

The fourth result in the table (Head of stringer) is believed to be an erratic one which cannot be explained as yet. If this result is ignored, the table indicates a slightly slower or equal crack rate in the 2024-T3 skin material and a 1.5 times slower rate in the 7075-T6 material. These trends are in good agreement with previous tests employing similar load sequences (refs. 1 and 3) on 2024-T3 and 7075-T6 sheet specimens. In those tests, omitting gust cycles with the lowest amplitude implied that the average crack rates were about 1.2 times and 1.4 times slower for the 2024-T3 and 7075-T6 aluminum alloys, respectively.

Effect of truncating high-amplitude cycles.- The effect of truncating high-amplitude cycles is presented in figure 3 where it is seen that the three highest amplitudes were reduced to the next highest one. A summary of the results of truncating high-amplitude gust cycles of flight-simulation loading (comparison of results of test series B and A) is given in the following table:

Cracked component	Material	Average effect on crack rate
Skin between stringers	2024-T3	2.5 times faster
Skin at stringer	2024-T3	2 times faster
Stringer flange	7075-T6	2 times faster
Stringer head	2024-T3	4 times faster
Stringer head	7075-T6	2 times faster
Spar web	7075-T6	1.5 times faster

The last column of the table shows that truncation of the high-amplitude gust cycles to a lower level in all cases accelerated the crack growth. On the average the crack rate was about 2.3 times faster in test series B than in test series A. In reference 2 with flight-simulation tests on sheet specimens carried out at similar stress levels, the same truncation caused a 3 times faster crack rate. Although the factor is somewhat higher, it is still thought to be a fair agreement.

Effect of increasing the design stress level.- In test series C all loads applied were 25 percent higher than those in test series A. Obviously higher crack rates should then be expected which was confirmed by test series C. Unfortunately there was a fairly large variation between the accelerating effect of the various components. On the average the crack rate was 2.5 times faster in test series C. Sheet specimen data from reference 2 predicted a somewhat higher crack rate, whereas results in the following section of this paper predicted a lower crack rate.

Effect of fail-safe loads on subsequent crack propagation.- Several investigations (refs. 5 to 8) employing constant-amplitude loading have shown very long crack-growth delays if the test was interrupted for a high load. The wing test offered an opportunity to observe the effect of a high load on crack propagation under flight-simulation loading. Test series R was completed by applying limit load three times. Some 10 cracks that had shown a reasonable amount of crack growth during test series R were left unrepaired. Without exceptions, the limit load applications had a large delaying effect on the growth of these cracks. Some examples are shown in figure 5. The vertical bar in the graphs indicates the stable crack growth during the three fail-safe tests up to limit load. Several cracks came to a complete standstill with a tendency to resume crack growth during test series C (25 percent higher loads).

It was suggested now and then to apply periodically fail-safe loads in a full-scale test. The argument was that hidden cracks would then readily show up. Unfortunately, this might well be the best method to hide these cracks completely, since further growth will be drastically delayed.

Effects of Frequency and Design Stress Level on Crack Propagation

Under Random Flight-Simulation Loading

The test program is still in progress and preliminary results can be presented only. Tests are being carried out on 2024-T3 Alclad and 7075-T6 Alclad specimens. Specimen width is 160 mm and the thickness is 2 mm. The cracks are starting from a central saw-cut notch. Some specimens were precracked by constant-amplitude fatigue testing to a semicrack length l of 10 mm or of 20 mm. Others were given a saw cut to a semicrack length l of 8 mm and were tested without additional precracking.

The load sequences applied are the same as those applied to the wing, with one additional feature. In each flight each positive gust load was followed by a negative one and vice versa. However, the amplitude for each half-cycle was selected at random from the amplitudes to be applied in that flight.[a]

Truncation of the gust spectrum occurred at the same level as shown in figure 3 for test series B, whereas the low-amplitude cycles were not omitted. The first tests were started with the following values for the stress levels:

Mean stress in flight $\qquad$ S_m = 7.0 kg/mm^2

Gust amplitudes $\qquad$ S_a = 1.1, 2.2, 3.3, 4.4, 5.5, 6.6, 7.7 kg/mm^2

Minimum stress in GTAC $\quad$ S_{min} = -3.4 kg/mm^2

Hence, the truncation level is S_a = 7.7 kg/mm^2. The same values were applied in references 1 and 3.

Changing the design stress level implies that all stress levels should be multiplied by the same factor. With the mean stress in flight as the characteristic stress level, tests are carried out at S_m = 10.0, 8.5, 7.0, 5.5 kg/mm^2. The gust amplitude and the minimum stress in the ground-to-air cycles are amplified accordingly. Test results are available for a loading frequency of 10 cps. (See fig. 6.[b]) It turns out that the effect of the stress level is different for the two alloys. At S_m = 5.5 kg/mm^2, the 2024 alloy is still far superior to the 7075 alloy. At higher S_m values the difference is negligible. It is thought that this relatively good behavior of the 7075 alloy should be attributed to the fact that favorable residual stresses are better maintained in this alloy. It is noteworthy

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[a] In the terminology of Naumann (ref. 9) the wing loading employed "random cycles" and the specimens were loaded by "random half-cycles, restrained."

[b] Since the stress histories were the same in all tests, except for the intensity of the stress, it was hoped to correlate the data for different stress levels by employing the stress intensity factor K. The results were disappointing and apparently similar K values do not imply similar crack rates in this case. This is attributed to different K-histories in the crack tip area and predominant interaction effects of stress cycles with different amplitudes. This issue will be discussed in more detail in the final report of this investigation.

that a trend such as that illustrated by figure 6 cannot be predicted from constant-amplitude data.

The effect of the load cycling frequency is studied by carrying out tests at 10 cps, 1 cps, and 0.1 cps. Available data have been compiled in the following table:

	2024-T3 aluminum alloy		7075-T6 aluminum alloy	
Crack growth $2l$, mm . . .	24 to 60	40 to 60	24 to 60	40 to 60
S_m, kg/mm^2	10	8.5	10	8.5
Life in flights at –				
10 cps	1105	1235	1382	1520
1 cps	1262	1116	1265	1360
0.1 cps	1234	1200		1350

As the table shows, the effect is not fully systematic but it is small, as might have been expected from constant-amplitude data (refs. 10 and 11). Of course one should be careful about generalizing the present data, especially if corrosive environments are present.

FLIGHT-SIMULATION TESTING

In this section of the paper, various aspects of flight-simulation testing are discussed and a proposal is made for a systematic compilation of flight-simulation fatigue test data.

Development of Several Fatigue Testing Procedures

In the past attempts were made to predict fatigue strength and fatigue life from simple fatigue data. Several complicating factors were early recognized and extensively studied. Examples are the presence of a mean stress as opposed to zero mean stress, the presence of notches as opposed to unnotched material, and the existence of large components as opposed to small laboratory specimens. Other factors, such as the effect of surface finish and fretting corrosion, led to an overwhelming number of empirical investigations. Qualitatively, understanding of all these influencing factors has highly increased. Nevertheless, it is still generally believed necessary to perform fatigue tests on the real components and preferably on a full-scale structure. For full-scale testing there are additional arguments, such as a realistic representation of eccentricities and load transmission in the structure and the indication of accidentally poor design features.

Testing a component or a full-scale structure implies that one wants to simulate the structural configuration as realistically as possible. In great contrast with these efforts, an unrealistic simulation of service load history was usually adopted in fatigue tests.

Since the accidents with the Comets in the early fifties, several new designs were subjected to a full-scale fatigue test. Usually this was a flight-simulation test, that is, with a flight-by-flight loading. However, the gust loading was highly simplified in many cases.

A simulation of a complex load time history was obviously impossible in the early days. In 1939 Gassner (ref. 12) published his first paper on program loading. He presented program loading as an improved method for estimating fatigue life although at that time Gassner had already realized that program loading was still afflicted with important deviations from load sequences in service. However, available testing equipment at that time could not do a better job.

Around 1960 random load fatigue tests started to draw much attention, partly because it became possible to carry out this type of testing. On the other hand, several types of fatigue loads in service had a random character. Moreover an elegant mathematical framework was available for dealing with random variables. Kirkby and Edwards (ref. 13) proposed to adopt narrow-band random loading in a way similar to that proposed by Gassner for program loading. This type of loading could still be applied in a resonance-type fatigue machine.

A real breakthrough was the introduction of the closed-loop electrohydraulic system with servovalves for monitoring the hydraulic load. An arbitrary load time history can be obtained from a similar electric signal. This system is now being used for relatively slow full-scale testing as well as fast testing of components or small specimens. The system has been developed to a high degree of reliability. Actually the impossibility of simulating a complex load time history has been eliminated. In other words, realistic load sequences can now be simulated in fatigue testing. It appears rather natural that a realistic simulation of a component or a structure should then be combined with a realistic simulation of service loading.

Purpose of Fatigue Testing

For a discussion on testing methods it is useful to keep in mind the variety of test purposes and test articles. A broad summary is given in the following table:

Test article	Testing method	Test purpose
Laboratory specimen	Constant-amplitude test	(a) Basic data for fatigue life estimates
Component	Program test	(b) Comparative design studies
Full-scale structure	Random-load test	(c) Direct life estimates
	Flight-simulation test	(d) Indication of fatigue critical elements, crack rates, and inspection methods

It is thought that testing purposes (c) and (d) require a realistic simulation of both the test article and the service load time history. Consequently, some type of flight-simulation loading is necessary. With respect to test purposes (a) and (b) different opinions may be held.

The utilization of constant-amplitude test data from laboratory specimens as basic data for life estimates is a complex problem. A cumulative damage rule has to be adopted – for instance, the Palmgren-Miner rule. Then, the differences between laboratory specimens and the actual structure have to be considered. The conclusion has to be that only very rough life estimates can be made in this way.

Gassner and Schütz (ref. 14) have proposed to use data from program tests as basic data for making life estimates. A similar proposal was made by Kirkby and Edwards (ref. 13) for random loading. There are some indications that life estimates may be improved in this way. However, discrepancies between the fatigue lives obtained in random tests and equivalent program tests (refs. 15 to 19) are not encouraging in this respect. Moreover, uncertainties about the damage rule and the relevance of the specimens remain. Actually flight-simulation fatigue test data could well be used for this purpose.

Test purpose (b) in the foregoing table is concerned with the comparison between different components, materials, and surface treatments for the same application in an aircraft structure. Many people still feel that constant-amplitude tests are a good means for this purpose. However, the possibility of intersecting (or nonparallel) S-N curves is making this doubtful. In figure 7, test results at stress level $S_{a,1}$ would indicate design A to be superior to design B. However, at stress level $S_{a,2}$ design B appears to be superior.

As an illustration of different answers to the same question, a recent investigation with constant-amplitude loading (ref. 3) indicated that the crack propagation in 7075-T6 was 4 times faster than the crack propagation in 2024-T3. However, under flight-simulation loading it was only twice as fast. The data of figure 6 are also illustrative in this respect.

The numerous test series with program loading carried out by Gassner and his coworkers (ref. 20) indicate that the risk of a misjudgment would be much smaller if program loading were adopted for comparative testing. This applies also to random loading (ref. 21). Nevertheless, if flight-simulation loading can be adopted, it apparently is the most preferable solution. Real problems should be tackled with realistic testing methods if possible. Recently Branger and Ronay (ref. 22) adopted random flight-simulation loading for exploring the fatigue behavior of a high-strength steel. Imig and Illg (ref. 23) adopted this method for studying the effect of temperature on the endurance of notched

titanium-alloy specimens. At the National Aerospace Laboratory (NLR) as part of an ad-hoc problem random flight-simulation loading was used to compare two alternative types of joints.

Variables of Flight-Simulation Testing

A constant-amplitude loading is easily defined by its mean, amplitude, and cyclic frequency. For program loading, additional variables are (1) load spectrum, (2) amplitude sequence (the programing), and (3) the maximum and the minimum amplitude to be included in the test. For random tests, similar variables can be indicated. The sequence, however, has some random character rather than being programed.

For a flight-simulation test, the situation is still more complex because different types of flight loads have to be simulated, such as gusts, maneuver loads, and ground-to-air cycles.

The main variables of flight-simulation loading, with some comments on their significance, are as follows:

(1) Load spectrum

Obviously the fatigue life depends on the type of load spectrum. For instance, there are large differences between the load spectra for civil and for military aircraft. Usually gusts are important for civil aircraft, whereas maneuver loads are less important. For military aircraft, the reverse is true. A realistic flight-simulation test requires the adoption of an appropriate load spectrum.

(2) Load sequence

If a flight-by-flight simulation is adopted, the fatigue loads superimposed on the ground-air-ground transitions can still be applied in various sequences. The effect of sequence was studied by several authors (refs. 1, 3, 9, 19, 23, and 24) and the general impression is that in a flight-by-flight loading the sequence of the loads in each flight is of secondary importance. Although this is a convenient observation, it still appears to be advisable to adopt a realistic sequence, which generally implies a random sequence.

(3) Design stress level

It is clear that an increase of the design stress level reduces fatigue life. This is illustrated for crack propagation by the curves in figure 6. Similar data were found by Branger (ref. 25) with a hole notched specimen of 7075 aluminum alloy and maneuver spectrum, by Branger and Ronay (ref. 22) with a hole notched specimen of chromium-nickel steel and maneuver spectrum, and by Imig and Illg (ref. 23) with an elliptical hole specimen of titanium alloy and supersonic transport load spectrum.

Curves giving the fatigue life as a function of the stress level applied in flight-simulation tests easily indicate the gain or loss of fatigue life if the design stress level is readjusted. Actually such curves have some similarity with Gassner's "endurance curves" (Betriebsfestigkeitskurven) and the curves of Kirkby and Edwards, who plotted the random-load fatigue life as a function of the root-mean-square stress level. However, curves pertaining to flight-simulation data give more relevant information.

(4) Maximum load allowed in the test (truncation level)

The wing test results presented in a previous section have confirmed earlier NLR data, which indicates that the truncation level has a predominant effect on crack propagation. It is expected that this effect is more applicable to a gust spectrum than to a maneuver spectrum in view of the different shapes of the spectra. Nevertheless, the assessment of the maximum load allowed in the test is a delicate issue. This problem was discussed in reference 26; these results led to the recommendation to truncate load amplitudes expected less than 10 times in the target life of the aircraft. This proposal was made in view of the favorable effect of higher loads and the uncertainty that all aircraft of a fleet will meet those loads.

At the same time, it would also be unrealistic to truncate at a much lower level since that may also lead to unrepresentative life indications. As a consequence, a realistic simulation is not compatible with a single loading pattern applied in all flights. Obviously different flights should be simulated. (See fig. 1, for an example.)

(5) Minimum amplitudes to be simulated

The results presented earlier for test series A have indicated that low-amplitude gust cycles may be slightly damaging in a flight simulation test. A similar indication was obtained by Naumann (ref. 9) when testing edge-notched 7075-T6 specimens. Remarkably enough Branger (ref. 25) found a small life reduction when omitting low-amplitude cycles from flight-simulation tests on 7075-T6 notched specimens (maneuver spectrum). Anyhow, if accurate data are required, low-amplitude cycles have to be included.

The situation is different for low-amplitude taxiing load cycles. If the mean stress during taxiing is in compression, it was found in NLR tests (refs. 1 and 3) and by Gassner and Jacoby (ref. 24) and Imig and Illg (ref. 23) that omitting the taxiing loads did not have a noticeable effect on the fatigue life. It is expected that this trend is applicable only if the mean stress of the taxiing loads is either small or negative.

(6) Loading frequency

Preliminary results presented in a previous section indicated a very small frequency effect if any. More data from flight-simulation tests were not available in the literature. It is expected that the frequency effect will be small for those materials that show a small frequency effect under constant-amplitude loading.

Advantages and Limitations of Flight-Simulation Tests

The advantages of flight-simulation tests are clearly associated with the fact that the loading is a realistic simulation of the service load time history. For obtaining valid information on fatigue lives and crack propagation from testing components or a full-scale structure, a realistic flight-simulation test is a necessary condition.

For comparative fatigue tests on competing designs or materials, constant-amplitude tests, program tests, or random-load tests may be adopted. As explained previously, it is not certain whether the comparative result will also be valid under service loading. This uncertainty is eliminated by realistic flight-simulation loading.

With respect to estimating fatigue lives in the design stage, one could start from constant-amplitude data and calculate the fatigue life by employing a damage rule (Palmgren-Miner, for instance). This procedure implies a very large extrapolation with a doubtful extrapolation rule. By starting from relevant flight-simulation data, the extent of extrapolation and thus the uncertainty will be highly reduced. In the following section a proposal is made for compiling flight-simulation fatigue test data for this purpose.

A disadvantage is that flight-simulation testing requires a more expensive fatigue machine with more complex electrohydraulic systems. Actually the technical problems of this type of machine appear to be solved and the number of available machines is rapidly increasing. The closed-loop systems indeed allow a wide variation of testing procedures and it is sometimes surprising to see the limited utilization of the potentials of the machine in fatigue tests.

Another limitation which appears to be more serious is that the load spectrum in service will never be the same as the spectrum applied in the test. Although this is true, it is not a fair objection. If there are differences between assumed and actual load spectra, one might account for them by calculations or by testing. Unfortunately the Palmgren-Miner rule is unreliable for this purpose (ref. 3). It may even predict the wrong sign of life corrections. The only realistic approach is to rely on empirical trends as obtained in flight-simulation tests. The test program proposed in the following section may also be useful in this respect. For a particular aircraft, another solution is to conduct some comparative flight-simulation tests with the initial load spectrum and the actual service load spectrum. This could be done on components or relevant specimens. In this respect it was stimulating to see a good agreement between the crack propagation results of the wing and those of simple sheet specimens.

A Proposal for Systematic Flight-Simulation Tests

In the foregoing sections, flight-simulation testing was recommended as being a more realistic approach to the problem of estimating fatigue properties. It was also

pointed out that a compilation of systematic data from such tests would be helpful. A proposal for a compilation was made in reference 27. A test program for this purpose should include flight-simulation tests with the following variables:

(1) Type of specimen

Representative riveted and bolted joints should be tested.

(2) Shape of load spectrum

Some typical shapes should be adopted, for instance, representing gust and maneuver spectra. If the effect of the load spectrum is known, one might interpolate for intermediate spectrum shapes.

(3) Design stress level

Some values should be adopted in order to study the effect of the stress level in a way similar to that of Gassner for program tests.

(4) Ground-to-air cycles

The number and the magnitude of ground-to-air cycles may be varied.

Such a program, which could well be extended, may be considered as an exploration of the effect of several variables on the life under flight-simulation loading. On the other hand, it may serve some practical purposes. Firstly, the data could indeed be used in the design stage for making life estimates. Secondly, this type of information could also be useful for correcting data from full-scale tests if the service load spectrum deviates from the test load spectrum. Thirdly, without actually having to design a standardized test one could use the data as a standard for comparison when checking the fatigue quality of new components. A handbook with results from systematic flight-simulation tests could be updated from time to time.

In fact, tests according to the foregoing program could be considered as collecting service experience in the laboratory. In general, the experience from fatigue failures in service does not become available in a suitable form because of insufficient data on load spectra, stress level, and structural configuration in most of the failures.

SUMMARY OF RESULTS

Flight-simulation tests were made with a full-scale wing structure to study crack propagation under different loading conditions. The results are summarized as follows:

1. Omitting low-amplitude gust cycles from the flight-simulation test implied a considerable timesaving. However, it had a small but systematic effect on the crack propagation rate. The propagation rate was somewhat slower.

2. Truncation of infrequently occurring high-amplitude gust cycles to a lower level considerably accelerated crack growth.

264

3. Increasing the design stress level increased the crack propagation rate.

4. The application of fail-safe loads (100 percent limit load) caused a drastic delay of subsequent crack growth. Such loads increase the fatigue life.

5. Good agreement was found between wing test results and results from tests with simple sheet specimens. This illustrates that tests on laboratory specimens may indicate the effect of modifications of the load spectrum.

Flight-simulation tests were also made with 2024-T3 and 7075-T6 sheet specimens at loading frequencies of 0.1, 1, and 10 cps and at four design stress levels. Preliminary results are as follows:

6. The frequency effect was small and nonsystematic.

7. The design stress level had a considerable effect on the crack propagation rate which was different for the two alloys. The difference could not be predicted from constant-amplitude data.

A discussion was presented concerning the meaning of flight-simulation testing for various testing purposes. The variables of flight-simulation testing and their effects on the test results were discussed. The merits and the limitations of flight-simulation tests are summarized as follows:

8. As compared with constant-amplitude tests, program tests, and random-load tests, a flight-simulation test is a more realistic representation of service loading and gives more relevant information. For a full-scale test or a component test, a realistic flight-simulation loading is an essential requirement for estimating fatigue lives and crack propagation rates.

9. For comparison between competing designs, materials, or surface treatments, flight-simulation tests give more relevant indications than alternative testing methods.

10. If life estimates made in the design stage of an aircraft are based on flight-simulation test data, the extrapolation of the test results is smaller and, hence, more reliable than for alternative procedures employing data from constant-amplitude tests, program tests, or random load tests.

11. Since sufficient data from flight-simulation tests are not available as yet, a proposal has been made for a systematic compilation of such data. A handbook with this type of data would also be useful as a standard for comparison. Moreover, the data could be used for evaluating the significance of differences between the load spectrum of a test and the load spectrum in service. For this purpose, the Palmgren-Miner rule is unreliable.

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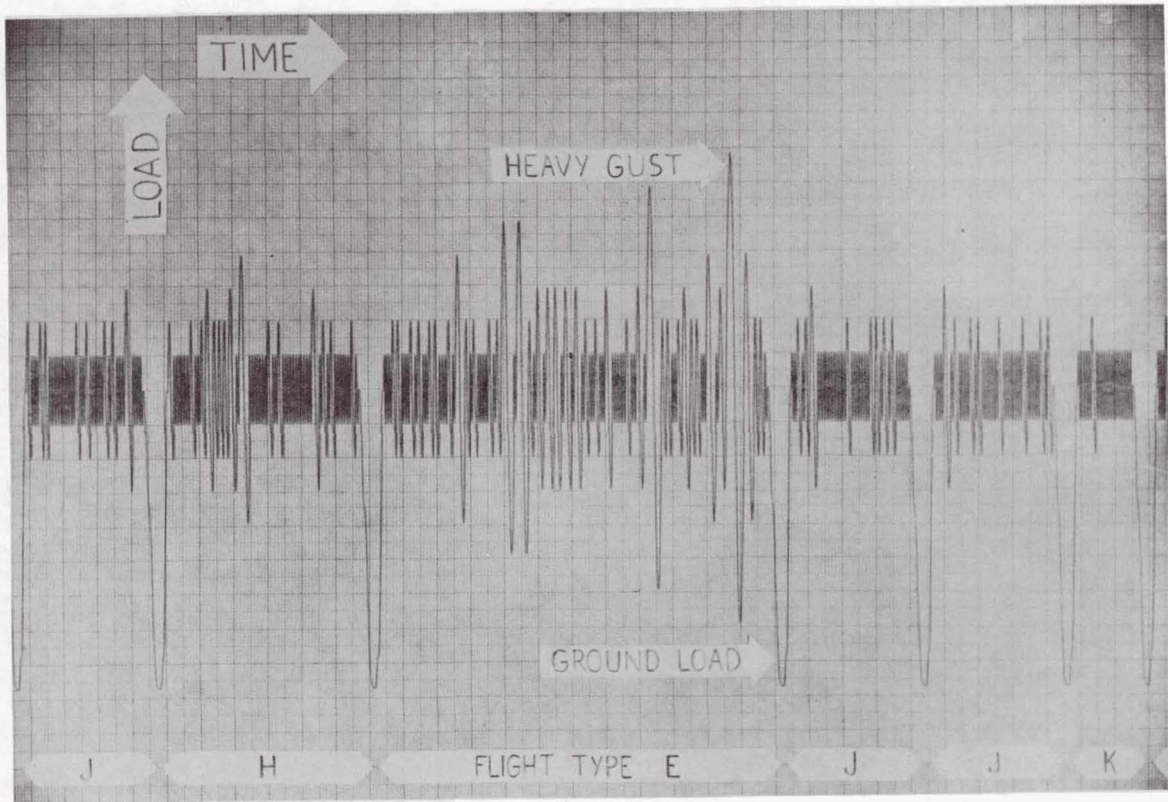
Figure 1.- Sample of a load record, illustrating the load sequence applied in the wing fatigue test. Ten different types of weather conditions are simulated; flight type E corresponds to fairly severe storm, whereas flight type K is good weather.
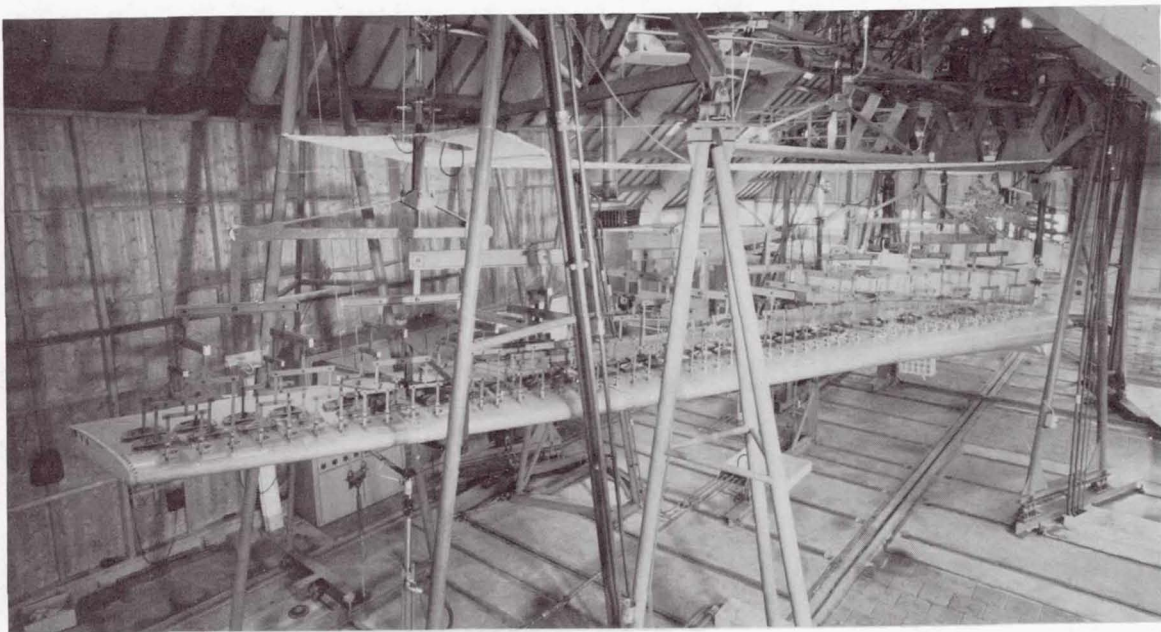
Figure 2.- Test setup with right wing and whiffle tree loading systems.
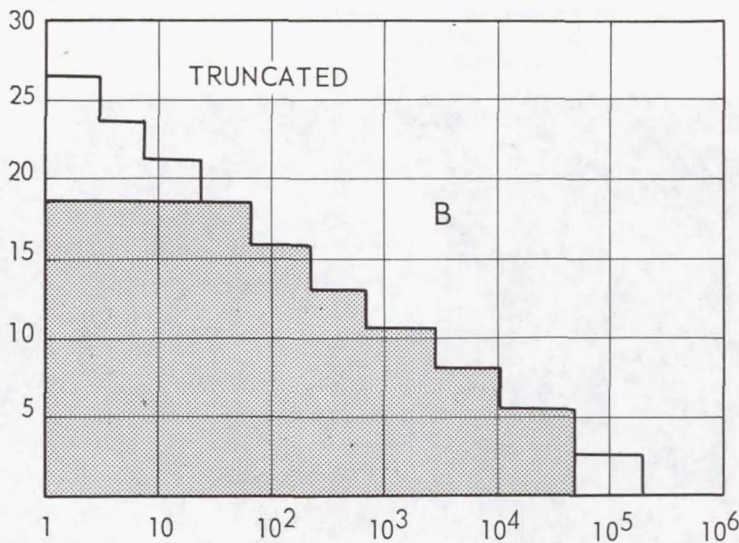
AMPLITUDE OF WING BENDING MOMENT,
% M_{ult}
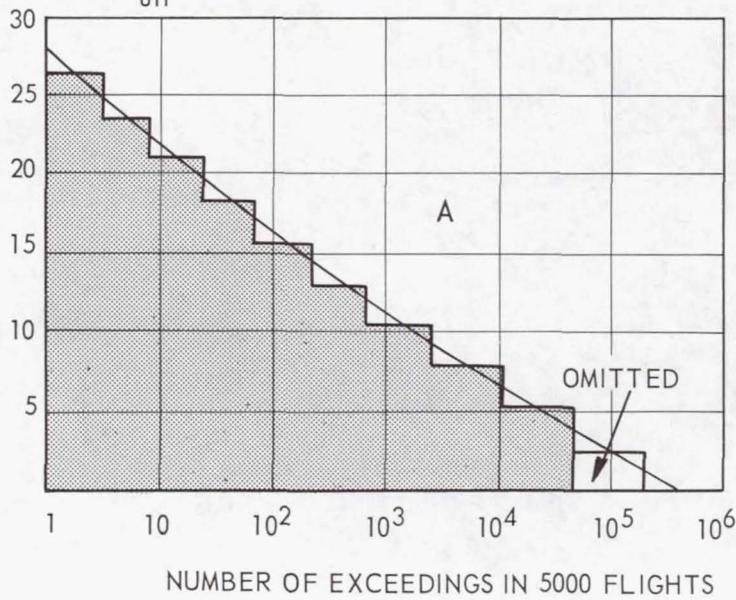
TEST SERIES

R

LOW – AMPLITUDE
GUST CYCLES INCLUDED;
47000 FLIGHTS

A

LOW – AMPLITUDE
GUST CYCLES OMITTED;
24000 FLIGHTS

B

HIGH – AMPLITUDE
GUST CYCLES TRUNCATED;
16000 FLIGHTS

C

SIMILAR TO A, BUT ALL
LOAD LEVELS INCREASED
25 %; 15000 FLIGHTS

NUMBER OF EXCEEDINGS IN 5000 FLIGHTS

Figure 3.- Gust load spectra in test series A, B, and C.

270

CRACK LENGTH, mm

NUMBER OF FLIGHTS
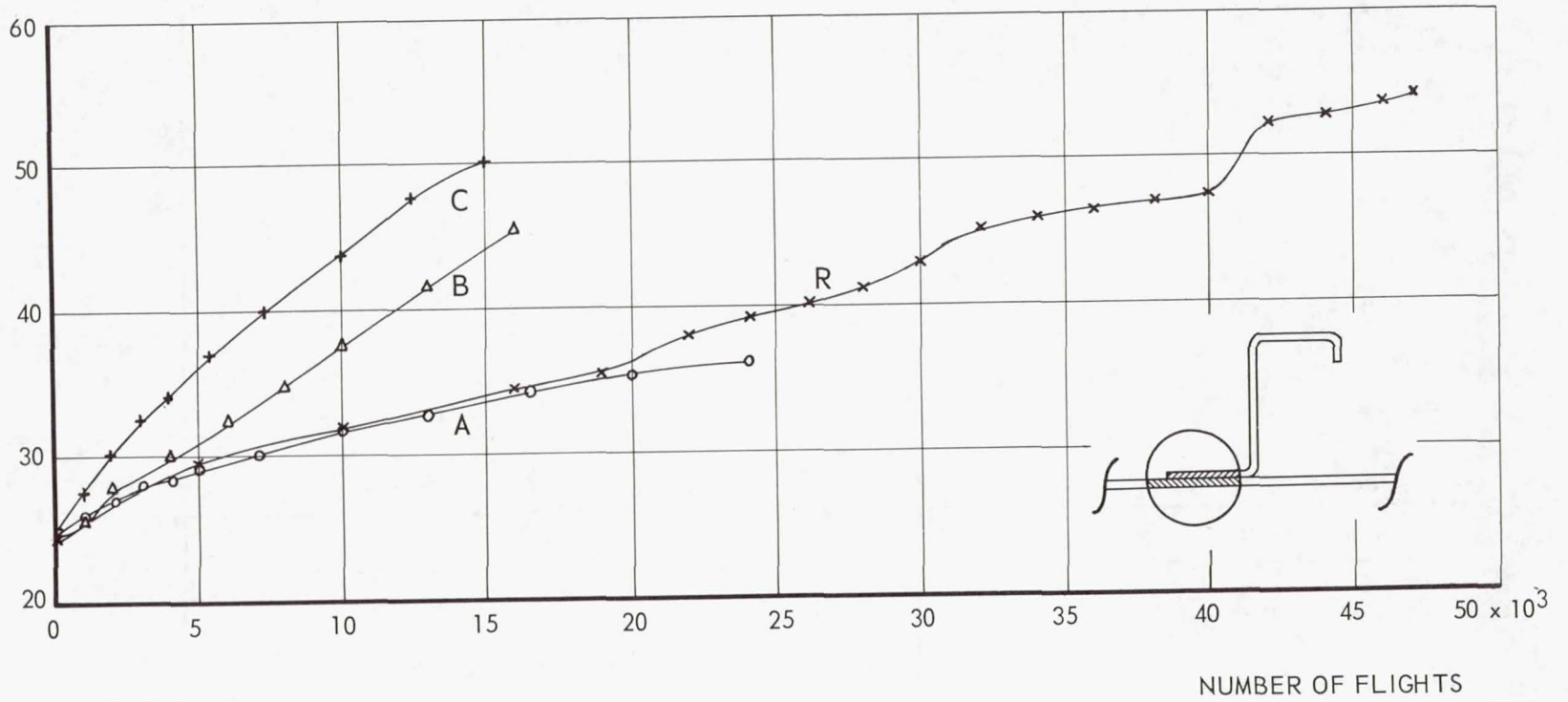
Figure 4.- Example of propagation curves for cracks in the skin. Comparison between the results of test series R, A, B, and C.
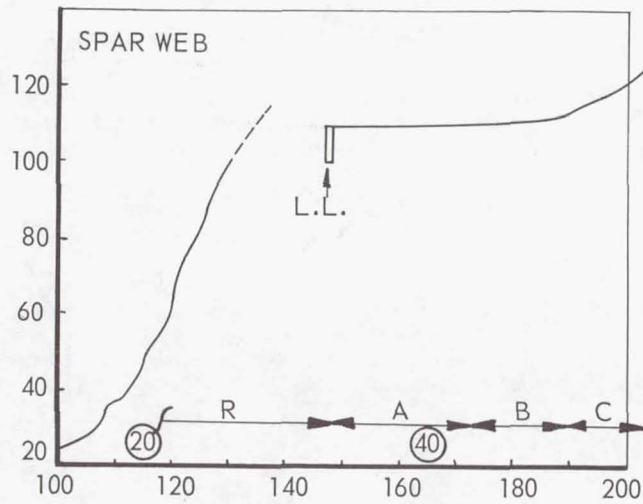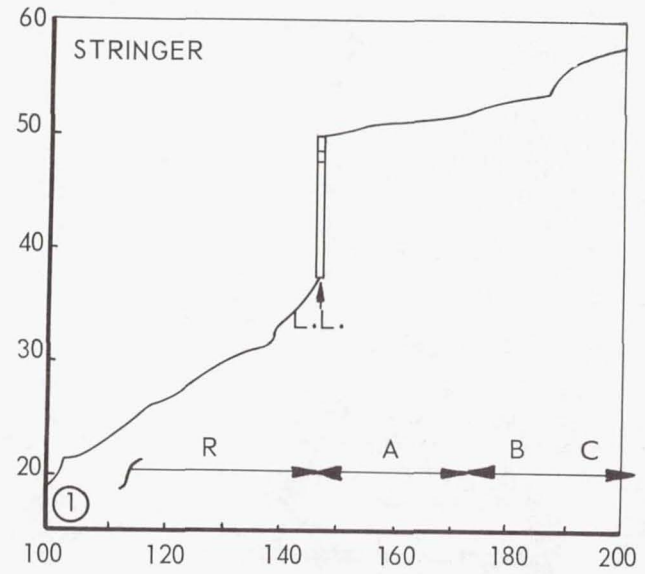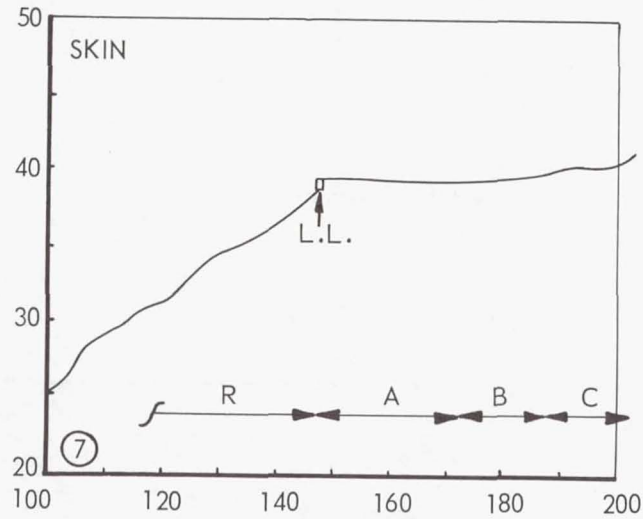
CRACK LENGTH, mm

SKIN

L.L.

R   A   B   C

⑦

100   120   140   160   180   200

STRINGER

L.L.

R   A   B   C

①

100   120   140   160   180   200

SPAR WEB

L.L.

R   A   B   C

⑳   ㊵

100   120   140   160   180   200

DOOR COAMING

L.L.

R   A   B   C

⑰

100   120   140   160   180   200

LIFE (1000 FLIGHTS)
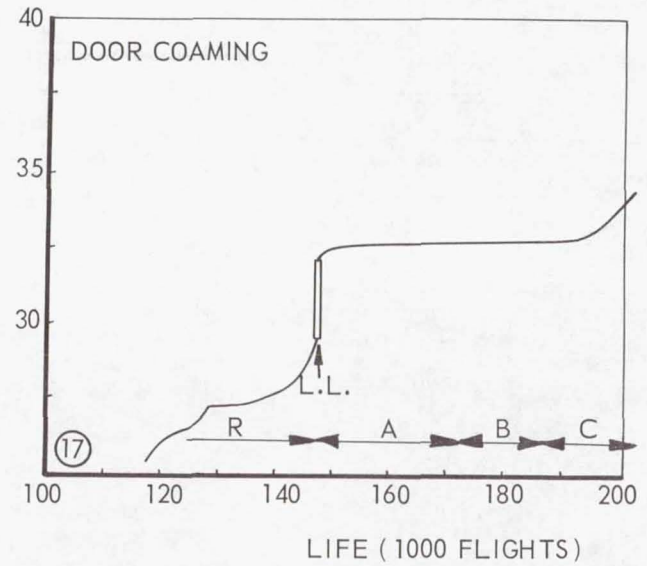
Figure 5.- The effect of limit load (L.L.) on crack propagation under flight-simulation loading.
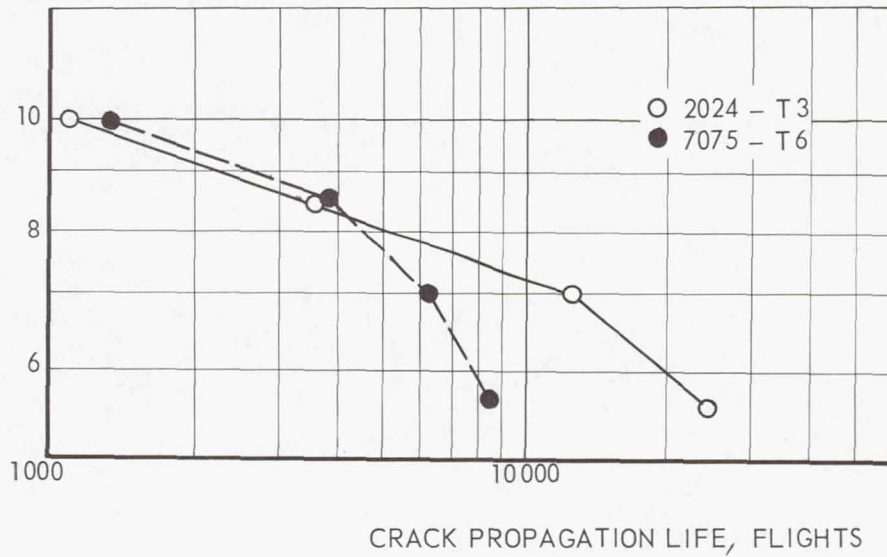
MEAN STRESS IN FLIGHT,
kg / mm^2

Figure 6.- Flight-simulation life for crack growth from $2l$ = 24 mm to $2l$ = 60 mm.
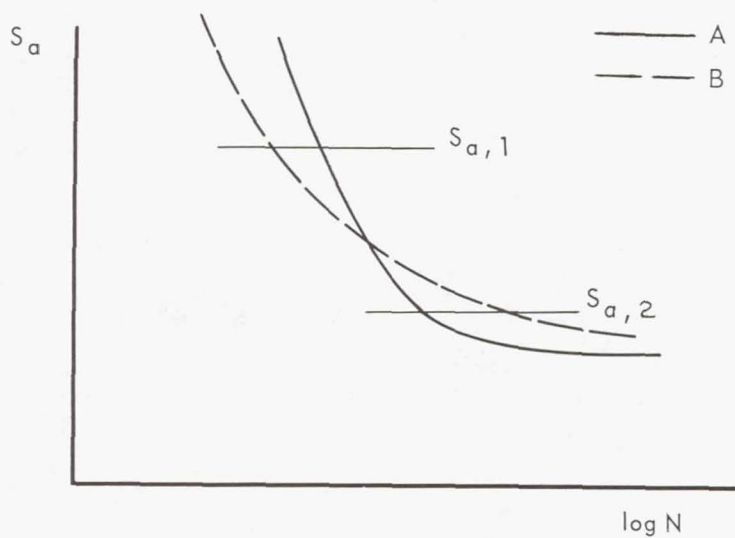
CRACK PROPAGATION LIFE, FLIGHTS

Figure 7.- Two intersecting S-N curves.