THE SECOND F.J. PLANTEMA MEMORIAL LECTURE

# CUMULATIVE DAMAGE PROBLEMS IN AIRCRAFT STRUCTURES AND MATERIALS

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Amsterdam

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Summary

As an introduction to the main theme of this paper a brief analysis is given of the various phases of designing, testing and utilizing an aircraft as far as fatigue considerations are applicable. Fatigue damage, damage accumulation and interactions between cyclic loads of different magnitudes are defined. An extensive test series on crack propagation in 2024-T3 and 7075-T6 sheet material under flight-simulation loading was recently completed at the NLR and the results will be presented. Aspects investigated were: (1) Truncation of high-amplitude gust cycles, (2) omission of low-amplitude gust cycles, (3) omission of taxiing loads, (4) omission of the ground-to-air cycles, (5) application of a single gust cycle per flight, (6) programming the gust cycles within each flight in comparison to the random sequence, (7) reversion of the up- and downward gust cycle sequence. The analysis of the empirical trends includes a survey of relevant literature. In the discussion the merits and problems of flight-simulation tests are examined and a number of recommendations are given.

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## Foreword

In November of 1966 Dr. Plantema suddenly and fully unexpected passed away. We at the Structures and Materials Department of NLR lost a leader which left us with deep feelings of sadness.
It would be superfluous to describe here in any detail how Plantema initiated in 1951 the International Committee on Aeronautical Fatigue and how he led its activities to the benefit of so many of us. This was extensively and very well done by Mr. Branger in the first Memorial Lecture. However, I would like to make some more personal remarks.

Dr. Plantema had been leading the department for more than 20 years and had done it in such a personal way that it was difficult to accept the fact we would not have him with us any longer. The first impression Plantema made on an outsider was probably that of an intelligent and quiet man who modestly, yet at the same time carefully formulated his ideas. However, working in his department had a few extra dimensions. Dr. Plantema had the willingness and the ability to listen to others whether they were young or old. He enjoyed seeing new ideas and always stimulated further explorations. He never threw away suggestions at an early stage, but when conclusions needed to be drawn Plantema had the ability to reduce statements to their proper significance. In this way he created for everyone an atmosphere in which to go ahead while at the same time remaining our safeguard against going wrong.

Plantema joined the NLR in 1934 and since then had been working on airworthiness problems, aircraft structures, stress analysis and related problems. A large number of papers and reports could easily illustrate the variety of subjects. He wrote a book in Dutch on the stress analysis of aircraft structures while shortly before his death his book "Sandwich Construction, The Bending and Buckling of Sandwich Beams, Plates and Shells" was published in the United States. Another field of his interest was concerned with loads on aircraft and the response of the structure to these loads. It was the integration of these issues and his general interest in airworthiness problems that may explain why he focussed so much effort on fatigue. In 1949 he had completed an extensive study of information available at that time regarding fatigue of structures and structural components. From then on the fatigue activities of the NLR were steadily increased and cumulative fatigue damage became one of the prominent subjects. In 1955 Dr. Plantema presented the first NLR paper on this subject at the I.U.T.A.M. Colloquium on Fatigue, held at Stockholm, and he had stimulated investigations of this type until his death.

My personal relation to Dr. Plantema dates from January 1953 when I came to NLR to work in his department. In the first place I owe him the ample opportunity to study this most intrigueing fatigue problem. He stimulated me throughout, and it was his suggestion that I also should prepare a thesis on fatigue. Besides his encouragement I also had the benefit of his most alert criticisms of my work.
What I really want to say is that Plantema to me was a good friend, from whom I learned to appreciate the relative significance of all our work and the prime importance of good human relations.

Of course everyone will agree that we lost Dr. Plantema much too early. Since we have to accept this you will understand that I appreciate the opportunity to present this lecture as a late tribute to my previous chief.

# 1. INTRODUCTION

Fatigue in aircraft structures is a problem for which quantitative and generally accepted solutions are not available as yet. Even in recent years significant fatigue failures in aircraft structures have occurred. Some of these failures can be attributed to either poor design or underestimating of the fatigue environment. Unfortunately such statements are of little help since there are a confusingly large number of aspects involved. For a designer considering fatigue problems experience, qualitative understanding and a good intuition are therefore essential.

Although our qualitative understanding of many fatigue problems is steadily increasing the designer is in difficulty as soon as quantitative predictions are required. In general terms the leading question is: what is the quantitative fatigue response in a specified fatigue environment? Physically this question implies the prediction of the initiation and accumulation of fatigue damage. This question is the main theme of the present lecture. In an attempt to avoid a confusing discussion the problem of fatigue in aircraft structures will first be outlined more or less diagrammatically. The discussion will then center around the question:

How to perform fatigue tests for both design studies and for proving the fatigue quality of a new structure?

In view of the discussion on this question results of a recent investigation on crack propagation under flight-simulation loading will be presented in chapter 5. Secondly fatigue damage and interaction effects are defined in chapter 6. This background is useful for understanding the empirical trends and related data from the literature (section 7.1). The discussion in chapter 7 is followed by a summary of conclusions.

Acknowledgement: The flight-simulation tests and the analysis of crack propagation data were most carefully performed by Messrs F.A. Jacobs and P.J. Tromp.

## 2. SYMBOLS AND ABBREVIATIONS

| | |
|---|---|
| GTAC | ground to air cycle (in the literature also: GAC = ground-air-ground transition) |
| TL | taxiing loads |
| Crack propagation life: | number of flights for crack growth from l = 10 mm to complete failure of the specimen, if not specified otherwise |
| l | semi crack length, see fig.4 |
| n | number of flights (or cycles) |
| dl/dn | crack propagation rate |
| N | crack propagation life, or fatigue life |
| $S_a$ | stress amplitude |
| $S_m$ | mean stress |
| $S_{min}$ | minimum stress |
| $S_{max}$ | maximum stress |
| $S_{a,min}$ | minimum $S_a$ of the gust cycles |
| $S_{a,max}$ | maximum $S_a$ of the gust cycles |
| 1 mm | = $10^{-3}$ meter = 0.04 inch; 1 inch = 25.4 mm |
| 1 $kg/mm^2$ | = 1,422 psi; 1000 psi = 0.703 $kg/mm^2$ |
| 1 kc | = 1 kilocycle = 1000 cycles |
| 1 $\mu$/fl. | = crack rate of 1 micron ($10^{-6}$ meter) per flight |

## 3. SURVEY OF AIRCRAFT FATIGUE PROBLEMS

In table 1 a survey has been given of the various aspects of tbe aeronautical fatigue problem. The list is not necessarily complete, but it is probably complete enough for the present analysis. It is important to note that there are three phases in the history of an aircraft, namely (1) the design phase, (2) the construction of the first aircraft and the performance of test flights, and (3) finally the utilisation of the aircraft in service.

In the first phase the designer has to start with the actual design work, including general lay-out of the structure, joints, detail design, specification of the materials, etc. He then has to estimate the fatigue performance of the structure and, broadly outlined, this involves three steps as indicated in table 2.

The description of the fatigue environment is a complex problem, not only because it involves a good deal of guessing but also in view of the large variety of aspects. This is illustrated by table 3. Several topics in the table will be briefly touched upon when discussing the merits of life calculations and fatigue tests.

The second step of table 2 includes the dynamic response of the structure in order to arrive at the fatigue loads in the structure. A modern trend in this area is the application of power spectral density (PSD) methods which require a transfer function for the behaviour of the aircraft. The calculation of the fatigue loads in the structure is beyond the scope of this paper, but it has to be said that the aircraft response may be a significant source of uncertainties. It can be partly circumvented by direct load measurements in flight.

The last step in table 2 is concerned with estimating fatigue lives and crack propagation. Several aspects of this problem are mentioned in tables 4 and 5. A discussion of these tables is presented in the following chapter.

Limited fatigue testing may be done already in the design phase in view of substantiating certain theoretical estimates or for comparative design purposes. The aims of testing a complete structure are much broader. Obviously a major goal then is to prove that the fatigue performance of the structure is meeting expectations and airworthiness requirements. In addition there are some more specific goals.

A schematic survey of several methods for conducting fatigue tests and the main variables to be selected is given in table 6. Some comments on this issue are also given in chapter 4 but a more extensive discussion is presented in chapter 7.

## 4. SOME COMMENTS ON LIFE PREDICTIONS AND TESTING

### 4.1 Life predictions

The estimation of fatigue properties of a structure is obviously an important question in the design phase of a new aircraft. Several procedures may be adopted for this purpose. The main lines of three solutions are indicated diagrammatically in table 4. It is difficult to say which method has been most widely used, but the first has received major attention in the literature.

The first method in table 4 implies that basic fatigue data have to be obtained. With the aid of a fatigue damage theory fatigue lives and crack propagation rates can then be calculated for a specified fatigue load spectrum. There are a corresponding number of weaknesses in this procedure, namely:

(1) the relevance of the basic fatigue data

(2) the validity of the damage theory

(3) the appropriate choice of the fatigue load spectrum.

Regarding the basic fatigue data table 5 shows a number of alternative starting points. If S-N data are going to be used it is almost certain that a complete set of curves for a certain component of a structure will not be available. The curves can be estimated in several ways employing $K_t$ conceptions or available data for similar parts, and even a limited amount of testing could be done. Nevertheless the accuracy (or better to say the relevance) of such data will be fairly limited.

The meaning of the latter statement should be considered in conjunction with the validity of a damage rule to be applied to this type of data. Whether we adopt the Palmgren-Miner rule [1,2] or the Smith theory [3] or some other damage accumulation formula, there will be another source adding to the uncertainty of the life estimate obtained.

Damage theories so far have been unable to include interactions between fatigue loads of different magnitudes (sequence effects), whereas laboratory tests have clearly indicated that such interactions do occur and can be qualitatively understood. [4,5] This shortcoming also applies to the Smith theory although it takes into account the effect of the maximum load applied to the structure. Therefore we must accept that life predictions starting from estimated S-N data and employing some cumulative damage rule are in fact rough estimates only and it will be difficult to decide whether the S-N curves or the cumulative damage rule will be the major source of inaccuracy. It will also be difficult to say whether the prediction will be safe or unsafe.

Instead of using S-N curves a more advanced approach begins with results of programme tests [6,7] or random tests [8] (table 5). Programme test data for notched specimens are available mainly through the work of professor Gassner and his co-workers (for a survey see Ref.9). By such an approach it could be argued that part of the uncertainty regarding the cumulative damage theory and the irrelevance of S-N data is eliminated. This is probably true, but it is thought difficult to evaluate quantitatively the improvement obtained. Moreover the extrapolation from specimens to a complex structure remains and therefore the result obtained should still be labelled as a rough estimate. In a similar way this applies to results of random load tests if used as basic data for calculations.

In table 4 two alternative methods for estimating fatigue lives were indicated. The methods will certainly be utilized to some extent by firms that have a good tradition in structural design. Starting from an older structure with a good service fatigue record one may be able to design a new structure at least to the same standard of quality. In the third method of table 4 allowable stress levels are adopted based on past experience that has shown them to be allowable from the fatigue point of view, provided the structure is properly designed. This method is in fact not too much different from the previous one.

The designer employing the second or third method may have some more confidence in his estimates since to some extent he also eliminated environmental and frequency effects. Even then one cannot afford somewhat questionable estimates if it concerns safe-life components. In fact both methods give lower limits of the fatigue life only.

In conclusion: Rough life estimates can be made in several ways. However, it is extremely difficult to provide life estimates for a complex structure with an accuracy that will reasonably satisfy the designer. If accurate predictions are required realistic testing appears to be indispensable.

## 4.2 Testing

Since the above conclusion on the possibilities of making accurate life predictions was fairly pessimistic we may ask how testing could improve the situation. In order to eliminate the extrapolation from simple specimens to a complex structural element the first recommendation, though very trivial, should be that the test article has to be the real structural component itself.
The second question is concerned with the type of loading. In table 6 a survey of different types of testing has been given. A more extensive survey including

variants of programme and random tests can be found in Ref.10. Table 6 shows an increasing complexity of loading histories going from constant-amplitude tests, through programme tests and random tests to flight-simulation tests. For all types of testing there are several variables to be selected, which do affect the result and this makes it difficult to make straight forward recommendations on testing. Although it is more or less accepted that ultimately a full-scale test with a flight-simulation type of loading should be carried out on each newly designed aircraft, it is not generally realized that even then a number of variables have to be selected that may have a significant effect on the test results. Unfortunately fully rational criteria for making such a selection are not available.

In order to study the effect of a number of the variables of flight-simulation testing a test programme was recently carried out at the NLR. The fatigue life of a structure includes crack initiation and propagation. The present programme as a first step was concerned with propagation only, because it was expected that several factors to be studied might have a larger influence on the crack propagation as compared to the nucleation period. This programme and its results are described in the following chapter, while the merits of testing are reconsidered in chapter 7.

## 5. A TEST PROGRAMME ON CRACK PROPAGATION UNDER FLIGHT-SIMULATION LOADING

### 5.1 Scope

Flight-simulation tests with various load sequences were carried out on sheet specimens of 7075-T6 and 2024-T3 clad material. A gust spectrum with a mean stress of 7.0 $kg/mm^2$ was adopted and in each test 10 different flights were simulated varying from good to bad weather conditions. The flight simulation was made in several ways, and the table in fig.1 gives a survey of the variables of the test programme. The main aspects studied were the omission of taxiing loads and small gust loads, the truncation of high-amplitude gust cycles and the sequence of gust cycles within a flight. This is further illustrated by figure 3, while figure 2 shows the load record of the most severe flight. In the following sections the investigation and the results are summarized while full details are to be found in Ref.11.

5.2 Materials, specimens, experimental procedures

Specimens were cut from 2024-T3 Alclad and 7075-T6 Clad sheet material with the following static properties:

| | $S_u$ (kg/mm$^2$) | $S_{0.2}$ (kg/mm$^2$) | Elongation (50 mm gage) |
|---|---|---|---|
| 2024-T3 Alclad | 47.4 | 36.0 | 18 % |
| 7075-T6 Clad | 53.9 | 48.5 | 13 % |

The thickness of the sheets was 2 mm and the specimens were cut to a width of 160 mm, its length between the clamping also being 160 mm (see fig.4). The specimens with a sharp central notch were precracked until a crack length $l = 10$ mm by cycling between 0 and 10 kg/mm$^2$, a stress cycle well below the flight-simulation loading.

Since the specimens were also loaded in compression anti-buckling guides were used as shown in figures 4, 5 and 6. Strain gage measurements could not indicate any load transmission through the guide plates provided the bolts were loosely tightened.

In general two specimens were tested in series, see fig.6, which halved the testing time of the programme. Tests were continued until one of the two specimens fractured completely. Since scatter was low the crack in the second specimen had already traversed the major part of the width.

All tests were carried out in an MTS fatigue machine with a maximum capacity of 25 tons. The electro-hydraulic closed loop system for load control was fed by an electric signal representing the required load-time history. The generation of this signal was controlled by a punched binary digit tape. Load frequencies adopted in the tests were 10 cps for the taxiing loads and the lower gust loads ($S_a$ = 1.1-4.4 kg/mm$^2$) while for the higher gust loads the frequency was inversely proportional to the amplitude.

During the tests the crack propagation was observed with a magnifying glass or a stereo-microscope (30 x). Crack propagation records were made by noting the number of flights each time that the tip of a crack reached a scribe-line marking on the specimen, see fig.4. An example of crack propagation curves from two specimens tested in series is shown in fig.7.

## 5.3 Fatigue loads

### The gust loads

A gust spectrum was recently derived in the Netherlands from flight data obtained in England, Australia and the USA. The shape of the spectrum is shown in fig.1. It was converted into a stress spectrum, by using a conversion factor 1 ft/sec $\hat{=}$ 0.3 $kg/mm^2$, a value frequently adopted by the NLR for programme tests. As a mean stress a value, $S_m = 7.0\ kg/mm^2$ was selected.

The load spectrum as given in fig.1 had to be distributed over a number of different flights. It will be clear that the load spectrum cannot be the same for all flights since the more severe gusts on the average occur less than once in a flight. Ten different types of flight were devised, each characterized by its own load spectrum varying from "good weather" conditions to "storm" conditions. This was done in such a way that the shape of the load spectrum (statistically speaking: the distribution function) is approximately the same for all flights except for the severity which is different. Justifications for this procedure are found in gust load measurements evaluated by Bullen [12], and in the modern power spectral density conception indicating that the shape of the spectral density function of the gust is invariable, but the intensity is depending on weather conditions and flying height. [13] Starting from the stepped function in fig.1 the numbers of gust cycles for the flights A-K were obtained as shown in table 7.

The sequence of the gust cycles in the flight was one of the variables to be studied. This means a random sequence had to be compared with a programmed sequence. It should be noted that each positive gust amplitude was immediately followed by a negative one of equal magnitude. In other words gust cycles were applied as complete cycles around a mean load. This applies to both the random and the programmed sequence, see fig.3. For the random gust loads this is a restriction on the randomness, which is thought to be slightly conservative. [14] Tests with the reversed sequence were also carried out, see fig.3C.

The random sequences of gust cycles in each flight were produced by a computer, see for an example fig.2. The sequence of the flights was also random, with the exception of the very severe flights. Since the severe flights might have a predominant effect on crack growth it was thought undesirable that these flights had a chance to cluster together, which is the risk of a random selection. The most severe flights were therefore uniformly distributed over a block of 5000 flights. In the tests such a block of 5000 flights was repeated periodically. Since a block contains approximately 200,000 gust cycles in a random sequence the repetition of the block is thought to be irrelevant with respect to the randomness of the load-time history.

It was recommended in Ref.15 that the maximum load in a full-scale flight simulation test should not exceed the load level anticipated 10 times in the desired life time in view of the predominant and favourable effect of higher loads on the fatigue life. A similar recommendation was made in Ref.15 for crack propagation. Assuming an inspection period of 500 flights the stress amplitude that is equalled or exceeded 10 times in 500 flights (or 100 times in 5000 flights) according to fig.1 is about 6.6 kg/mm$^2$. This truncation level was used in several series, but in addition higher and lower truncation levels were employed. The test results clearly confirmed the slower crack propagation at higher truncation levels.

The ground-to-air cycles and the taxiing loads

In some preliminary tests the mean stress of the ground-to-air cycles (GTAC) was more or less arbitrarily assessed at $S_m = 0$. On this mean stress 20 taxiing load cycles were superimposed with an amplitude of $S_a = 1.4$ kg/mm$^2$, the stress range 2.8 kg/mm$^2$ thus being 40 % of the $S_m$-value of the gust cycles. A similar pattern for the taxiing loads was adopted previously by Gassner and Jacoby.[16] A somewhat more severe air-ground-air transition was adopted for the major part of the tests by taking $S_m = -2.0$ kg/mm$^2$ for the taxiing loads, and hence $S_{min} = -3.4$ kg/mm$^2$. Since it was expected that the damaging effect of the taxiing loads would be negligible (the tests have confirmed this view) it was thought unnecessary to refine the GTAC by varying both the number and the amplitude of these load cycles, although that could have easily been done. In the test series from which the taxiing loads were omitted the same $S_{min}$-value was adopted, i.e. -3.4 kg/mm$^2$.

5.4 Test results

In general crack propagation records started at a semi-crack length $l = 10$ mm while the semi width of the specimens was 80 mm. The complete propagation records are given in Ref.11. Examples of the crack rate as a function of the crack length are shown in figures 8 and 9 of the present paper. For the illustration of several trends it is sufficient to consider the crack propagation life only ($l = 10$ mm to $l = 80$ mm). Data of this type are contained in figures 10, 11 and 12. In most cases the results are the mean of four identical tests. Scatter was low as usual for macro-crack propagation.

The trends emerging from the data are summarized below:

1. Omitting the taxiing loads from the GTAC had an insignificant effect on the crack propagation. Although figures 10 and 11 were not prepared to illustrate this observation they contain comparable results of tests with and without taxiing loads.

2. Crack propagation lives were practically the same for both random and programmed gust sequences. Comparisons can again be made in figures 10 and 11. Another insignificant effect was due to reversing the sequence positive/negative gust loads in each cycle (fig.12). The two results suggest that changing the sequence of the gust cycles in a flight did not have a noticeable effect on the crack rate.

3. The effect of $S_{min}$ on the GTAC has not been studied systematically. The few data available suggest that the differences between the results for $S_{min} = -1.4$ and $S_{min} = -3.4$ kg/mm$^2$ was negligible for the 7075 specimens, whereas for the 2024 specimens $S_{min} = -1.4$ has given somewhat longer fatigue lives.

4. Systematic increases of the crack propagation lives were obtained when small gust loads were omitted from the flights. Omitting only the cycles with the smallest amplitude ($S_a = 1.1$ kg/mm$^2$) increased the life by almost 20 and 40 percent for the 2024 and 7075 specimens respectively. The increase was, however, much larger if the two smallest amplitudes ($S_a = 1.1$ and $2.2$ kg/mm$^2$) were omitted, see figures 10 and 12. On the average the life was just slightly less than doubles (1.9 x). If all gust loads were omitted, except for the largest positive gust in each flight the life was tripled, see figure 12. The above influences are illustrated in more detail by figure 9. Apparently the effects are larger in the first part of the crack propagation, i.e. for lower stress intensity factor. For increasing values of the crack length, involving higher stress intensities at the tip of the crack, the effect of omitting small gust loads becomes smaller.

The effect on machine time is illustrated by the testing periods required for 5000 flights (taxiing loads omitted):

| | |
|---|---|
| All gusts included | 356 minutes |
| Gusts with $S_a = 1,1$ kg/mm$^2$ omitted | 106 minutes |
| Gusts with $S_a = 1.1$ and $2.2$ kg/mm$^2$ omitted | 40 minutes |
| Only one gust per flight | 28 minutes |

5. A most predominant effect was exerted by the truncation level of the gust load spectrum. This is clearly illustrated by figure 11. Increasing the maximum amplitude from 4.4 to 8.8 kg/mm$^2$ increased the crack propagation life six times for the 7075 specimens and four times for the 2024 specimens. The effect is again shown in more detail by plotting the crack rate as a function of crack length, see figure 8. For the 7075 specimens tested with $S_{a,max} = 8.8$ kg/mm$^2$

the crack rate initially decreases. For a single specimen tested with the load spectrum from fig.1 fully untruncated ($S_{a,max} = 12.1$ kg/mm$^2$) the crack rate decreased so drastically (fig.8C) that the test had to be stopped in view of testing time.

6. The omission of the GTAC increased the fatigue life approximately 50 percent for the 7075 specimens and 75 percent for the 2024 specimens, see fig.12.
7. An average ratio of approximately 2 was obtained when comparing the crack propagation lives of the 2024 and the 7075 specimens. However, as shown by fig.11 the ratio is depending on the truncation level of the gust load spectrum.

### 5.5 Some additional tests

#### Crack-nucleation period in open-hole specimens

Some tests were carried out on sheet specimens with a central hole instead of a sharp notch. The aim of these tests was to see whether the significant effect of truncation as found for crack propagation also applies to crack nucleation. These tests were carried out on specimens of 2024-T3 material only.
The average crack propagation curves are shown in figure 13. Crack nucleation occurred at the edge of the hole and the nucleation period was arbitrarily defined as the number of flights to create a crack with a length of 2 mm ($l' = 2$ mm or $l = 12$ mm, see fig.13). The variable studied was the truncation level and fig.13 shows that this level had a large effect on the crack propagation life, similar to the results as found in the normal crack propagation tests. For the crack-nucleation period the truncation effect was smaller as illustrated by the life ratios in fig.13. In these tests the yield stress was theoretically exceeded at the edge of the hole ($K_t = 2.43$) only at the highest truncation level ($S_{a,max} = 8.8$ kg/mm$^2$). Hence larger truncation effects for the nucleation period might occur if still higher truncation levels are adopted.

#### Flight-simulation tests at a lower stress level

Some tests were carried out on 2024-T3 Alclad specimens with the stress levels reduced 22 percent ($S_m = 5.4$ kg/mm$^2$) and the amplitudes reduced accordingly (detailed results still to be published). Since crack propagation was much slower recording of the propagation was started at $l = 20$ mm. The crack propagation life for several truncation levels has been plotted in figure 14. The large effect of the truncation level as found already in fig.11 is clearly confirmed by these additional tests at lower stress levels.

Constant-amplitude tests and damage calculations

At the end of the investigation a small number of specimens were still left, which were then used for constant-amplitude tests. Crack propagation lives obtained have been plotted in figure 15. It may be noted from this figure that the ratios between the fatigue lives for the 2024 and 7075 specimens are approximately 4-5, whereas for the flight-simulation tests an average value of 2 has been mentioned before. This confirms that constant-amplitude tests adopted for comparative purposes may give misleading results.

Damage calculations could be made for the random tests without GTAC on 2024 specimens. The result was $\sum n/N = 3.4$.
The data from figure 15 further allow damage calculations to predict the increase of life due to omitting small gust loads. This was illusory, since in most tests $\sum n/N$ only for the small gust cycles was already larger than one. On the other hand starting from the data for the tests without small gust cycles (cycles with $S_a = 1.1$ and $2.2$ kg/mm$^2$ omitted) the life reductions to be expected by including these small cycles could be calculated. The reductions were in the order of 70 and 80 percent for the 2024 and 7075 specimens respectively. The tests, however, indicated reductions in the order of 50 percent. The Palmgren-Miner rule apparently overestimates the damage done by these low-amplitude cycles, at least in these flight-simulation tests.

5.6 Fractographic observations

Macro-fatigue bands, see figure 16, were observed on almost all fracture surfaces. The visibility of the bands was dependent of the testing conditions. Two-stage carbon replicas examined in the electron microscope easily revealed the well-known fatigue striations. Two examples are shown in figure 17. In general the striations were more clearly observed on the 7075 specimens. It would have been promising if the GTAC could be indicated in the electrographs, but unfortunately this was impossible for the fractures produced under random flight-simulation loading. For programmed flight simulation certain groups of gust cycles of equal magnitude could easily be indicated, see for instance the lower picture in fig.17. From this information the striations corresponding to the GTAC could be indicated in some cases, although in general this still remained difficult.

## 6. FATIGUE DAMAGE ACCUMULATION AND INTERACTION

Before discussing the previous test results and related information from the literature some physical arguments on fatigue damage will be presented. This should allow a better appreciation of the various trends observed.

### Fatigue damage

In Ref.17 fatigue damage was defined in a general way as the change of the state of the material caused by cyclic loading. More specifically this change was described as:

(a) the amount of cracking
(b) the (cyclic) strain hardening of the material
(c) the residual stresses introduced.

It was realized that the latter two properties were not homogeneous through the material, but would show large gradients especially near the tip of the crack. As a consequence an accurate description of fatigue damage is a highly complex problem, and qualitative conceptions have to be employed.
The amount of cracking obviously is the most important damage parameter since a crack constitutes already a partial failure. Nevertheless strain hardening and residual stress are significant damage parameters in view of the controlling influence on crack growth. A fatigue model developed in Ref.18 illustrates the latter point (see also Ref.19).
Crack growth was depicted as a geometrical consequence of cyclic slip at the tip of the crack either by dislocations flowing into the crack or dislocations generated by the tip of the crack. The amount of cyclic slip will depend on the local shear stress amplitude ($\tau_a$) and the local (cyclic) strain hardening. The efficiency of converting cyclic slip into crack extension will depend on the maximum tensile stress at the tip of the crack in view of energy release. This local stress should include the residual stress built up by the preceding load history.
The above aspects are summarized in fig.18, which also includes the two following features regarding the crack geometry. In recent American publications [20,21] it has been suggested that crack tip blunting and sharpening may be of more than just secondary importance. The argument is that local stresses will depend on the crack tip radius. The second aspect [11] is related to the state of stress. Low stress amplitudes are associated with slow crack propagation and plane strain at the tip of the crack (tensile mode fracture, macroscopically), while high stress amplitudes will induce fast crack propagation with predominantly plane stress at the tip of the crack (shear mode fracture). Changing from a low amplitude to a high amplitude may then imply that the crack front has not the spatial configuration associated with the

high amplitude. The same applies to the reversed amplitude change (see loading histories B and C in fig.18).

Interaction between load cycles of different magnitudes

If the fatigue load is changed from one level to a second level (by either changing $S_a$ or $S_m$ or both) the fatigue crack propagation at the second level will initially be different from the propagation occurring when the second level had been applied from the beginning of the test. Such an effect will be referred to as an interaction effect between load cycles of different magnitudes.

Crack propagation tests on 2024 material (Refs 22-24) with loading histories A-D shown in figure 18 have indicated the following trends. A single positive peak load (load history D) induced extremely drastic crack growth delays. It was thought that residual compressive stresses were mainly responsible for the delays, although crack blunting and strain hardening may have exerted additive influences. Smaller but still rather significant crack growth delays were found when the pre-history occurred at a higher cyclic load level (load history B). The delays were easily indicated by macroscopic observations. The results presented in the previous chapter are also favouring the residual stress conception rather than the crack blunting interpretation. High peak loads had a larger beneficial effect for the 7075 specimens than for the 2024 specimens despite crack blunting being more effective in the more ductile 2024 alloy.

If the preceding load history occurs at a lower amplitude (see load history C in fig.18), one would not expect an interaction effect on the subsequent crack growth if considering residual stresses only. An interaction effect on the crack growth could indeed not be observed by macroscopic means. However, assuming that the crack growth was accelerated during a few cycles only this will escape such observations. It is noteworthy that McMillan and Pelloux (Ref.21) on the basis of electron fractography came to the conclusion that an interaction effect of the pre-history could hardly be observed on the fracture surface, except perhaps for the first cycle applied after changing the fatigue load. They obtained some indications that interactions might be active then. When increasing the load amplitude (history C) the effect may be a consequence of crack sharpening and cyclic strain hardening (embrittling?). It is important to know that changing the fatigue load may introduce an interaction that is only significant for the first cycle following that change. The implication is that interaction effects could be highly significant for random load sequences, since the amplitude is changing from cycle to cycle.

For tests with a programmed load sequence, however, such interaction effects may remain almost unnoticed since changing the stress amplitude is a relatively infrequent occurrence.
In conclusion it has to be admitted that with the exception of the influence of residual stresses the qualitative understanding of the other interaction effects is still partly speculative and requires a further systematic study.

## 7. DISCUSSION

The discussion will deal with the following questions.
(1) The trends of the test programme described in chapter 4 and a comparison with similar evidence from the literature.
(2) A comparison between the various fatigue testing methods.
(3) The merits of flight-simulation testing.
(4) Comparison with service experience.
In the discussion several aspects of fatigue damage accumulation as indicated by table 1 will be touched upon. Conclusions are summarized in chapter 8.

### 7.1 Flight-simulation test data. Comparison with the literature

Low-amplitude cycles

The test results in chapter 4 have shown that omission of taxiing loads (TL) ($S_a$ = 1.4 kg/mm$^2$ and $S_m$ = -2.0 kg/mm$^2$) did not noticeably affect the crack propagation life. On the other hand omission of small gust loads ($S_a$ = 1.1 kg/mm$^2$ and $S_m$ = 7.0 kg/mm$^2$) systematically increased the crack propagation life although the increase was not large.
As pointed out in the previous chapter the value of $S_{max}$ of a stress cycle is important for the conversion of cyclic slip into crack extension. Since $S_{max}$ for the TL was still compressive it should be expected that these cycles did not contribute to crack growth.
In agreement with the present results Gassner and Jacoby (Ref.16) also found that the omission of TL from the GTAC did not affect the fatigue life in flight-simulation tests on notched bars ($K_t$ = 3.1) of 2024-T3 material.

The omission of small gust cycles from the flight-simulation loading induced a systematic influence on the crack propagation life as shown by figure 10. For an explanation two lines of thoughts may be considered:
(a) During the small gust cycles there will be some crack extension. In other words these cycles give some direct contribution to the crack propagation.

(b) Secondly the small gust cycles may induce an unfavourable interaction effect on the crack extension during larger amplitude cycles, see the discussion in the previous chapter. It is difficult to separate the contributions of the two possibilities (a) and (b).
Flight-simulation tests on notched elements, involving the effect of omitting small gust cycles were reported by Naumann (Ref.14) and by Gassner and Jacoby (Ref.16). Naumann employing random flight-simulation loading found a small life increase when omitting gust cycles with $S_a = 1.05$ kg/mm$^2$, namely 16 and 7 percent depending of $S_{min}$ in the GTAC (7075 edge notched specimens, $K_t = 4.0$, $S_m = 14$ kg/mm$^2$). Gassner and Jacoby reported a 2.5 times longer fatigue life in programmed flight-simulation tests if the cycles with the smallest amplitude ($S_a = 1.3$ kg/mm$^2$) were omitted (2024 central-notch specimens, $K_t = 3.1$, $S_m = 9.5$ kg/mm$^2$).

High-amplitude cycles

The truncation of the gust spectrum (see fig.1), implies that the amplitude of the more severe gust cycles are reduced to a common $S_{a,max}$-value. The present results have shown that this value has a predominant effect on the crack propagation life, see figures 11 and 14. The first figure clearly illustrates that the effect is large, irrespective of random or programmed gust sequences being adopted and taxiing loads being applied or not. It also shows that the effect is larger for the 7075 alloy than for the 2024 material.
For an explanation the interaction effects mentioned in chapter 6 have to be considered. Since the trends were the same for programmed and random gust sequences and also for random sequences with and without small gust cycles it is thought that residual stresses were indeed the main agent responsible for the effect of the truncation level (see also the discussion in the previous chapter).
The tests on the open-hole specimens (see fig.13) have revealed the same effect for the fatigue life until visible cracking although the effect was smaller. For the nucleation period, however, the truncation levels were relatively low when considering a target life of 50000 flights for instance.
In the literature similar results have been published only by Gassner and Jacoby (Ref.16). For a notched bar (2024-T3, $K_t = 3.1$, $S_m = 9.5$ and 11.0 kg/mm$^2$) with programmed flight simulation loading they reported 30 and 10 percent life reductions when $S_{a,max}$ was reduced from 2.1 $S_m$ to 1.55 $S_m$. Qualitatively it is the same trend as in the present investigation, but the effect is smaller. More tests on notched elements appear to be desirable.

Sequence of gust cycles in a flight

It was a remarkable result of the crack propagation tests that the sequence of the gust cycles in a flight did hardly affect the crack propagation life. Similar results were found for a random and a programmed sequence (figs 10 and 11) while a reversion of positive and negative gust loads also had a negligible effect (fig.12). Apparently the balance of favourable and unfavourable interaction effects remained the same, but it is difficult to explain this even qualitatively. It may well be that the flight-by-flight loading pattern and the maximum gust load of each flight had a predominant influence on the fatigue damage accumulation.

Similar information for notched specimens were published by Naumann (Ref.14) and Gassner and Jacoby (Ref.25). Naumann performed tests on an edge notched specimen ($K_t = 4$) of 7075 material with a random gust loading with and without GTAC. Three types of randomness were adopted, indicated by Naumann as:

(1) Random cycle: Each positive half cycle was followed by a negative half cycle of the same magnitude.

(2) Random half cycle, restrained: Each positive half cycle was followed by a negative half cycle, the magnitude of which was selected at random from the load spectrum and which therefore was generally not equal to that of the preceding positive half cycle.

(3) Random half cycle, unrestrained: Positive and negative half cycles were randomly selected with no restrictions on the sequence of positive and negative. The results are summarized in the table below.

| Randomness | Fatigue life in flights | | Fatigue life ratio[a] | |
|---|---|---|---|---|
| | No GTAC | GTAC | No GTAC | GTAC |
| (1) Random cycle | 5815 | 1334 | 0.66 | 0.84 |
| (2) Random half cycle, restrained | 7358 | 1515 | 0.84 | 0.95 |
| (3) Random half cycle, unrestrained | 8798 | 1588 | 1 | 1 |

(a) Ratio = 1 for case (3)

Gassner and Jacoby (Ref.25) performed flight-simulation tests with a random gust sequence and with two different programmed sequences. The tests on 2024-T3 specimens ($K_t = 3.1$) yielded fatigue lives of 2500, 2800 and 5800 flights respectively. There were approximately 400 gust cycles per flight programmed in a high-low-high amplitude sequence (life = 2800 flights) or in a low-high-low sequence (5800 flights). With such a large number of gust cycles per flight different programming techniques apparently may cause significantly different fatigue lives. Hence a realistic sequence should be preferred.

7.2 Testing methods and testing purposes

A schematic survey of testing methods and testing purposes has been given in table 6. Some comments on making life estimates were already made in section 4.1, but we will consider this issue once again in this section. Attention will first be given to comparative design studies.

Comparative design studies

In the past constant-amplitude tests (conventional fatigue tests) were frequently adopted for component testing when different designs had to be compared. The view that a constant-amplitude loading is good enough for this purpose is probably held by too many people. If design A is superior to design B at a load amplitude $S_{a1}$ it may be inferior at another amplitude $S_{a2}$ (intersecting S-N curves). An example of misleading information based on constant-amplitude loading was given by the crack propagation data in section 5.5.

A large improvement for comparative design tests is the programme test proposed by Gassner [26] as early as 1939, because it includes the various load cycles occurring in service. Since the development of the electro-hydraulic fatigue machines with closed-loop load control the random load test has become another feasible alternative [27]. An extensive and instructive survey on random load testing has recently been prepared by Swanson [27].

The question should be raised whether the random test is again an improvement on the programme test. From a theoretical point of view the answer should be yes. It was argued in chapter 6 that various mechanisms (see fig.18) could lead to interaction effects and thus have an influence on the fatigue life. As a consequence we should not only duplicate the variety of load amplitudes but also the possibilities for interaction, that means the random load sequence. Comparative tests with random and programmed loads were recently reviewed [10,11] and it turned out that there was no systematic relation between the results of the two types of loading. On the average the programme fatigue life was somewhat longer than the random load fatigue life. A surprising result was found by Jacoby [10]. Testing notched bars ($K_t$ = 3.1, 2024-T3 material) with a gust spectrum he found programme fatigue lives approximately 6 times as high as for random loading. In other words the balance of interaction effects must have been largely different for the two types of loading. This certainly deserves a further study.

If random tests are an improvement relative to programme test how significant is the improvement and is it really worth the price? Another skeptical question is whether a random load test is good enough for comparative design studies. If the service loads are mainly of a random nature a random load test should certainly be preferred to a programme test. However, the improvement may be questionable if service loads are a mixture of random and nonrandom loads, such as the GTAC. In fact correct answers then require some sort of flight simulation loading. Obviously this answer is more easily given at this time since fatigue machines for this purpose are now commercially available.

Life estimates

Several limitations of calculating fatigue lives in the design phase of an aircraft have been discussed in section 4.1 and will not be repeated here. An additional problem is indicated by the test data reported in chapter 5, since $S_{a,max}$ apparently has an important influence on crack growth. If life calculations are based on test data from programme tests or random tests one has to realize that the associated $S_{a,max}$ or clipping ratios are not of secondary importance and one should try to understand whether conservative or unconservative data are going to be used.

With respect to results from random tests it should be borne in mind that the spectral shape and the band width of the random load may also affect the endurances. Fortunately as far as data are available [11,27] it appears that the spectral shape is of minor importance. It is unknown, however, whether this is still valid for narrow band random load. Since the band width is then related to the modulation rate of the load amplitude the balance of fatigue interaction effects may be influenced in a similar way as for programme tests. Hence it is thought that random loads applied in tests should have a good deal of irregularity if service loads have a similar nature. This appears to be applicable to gust loads.

7.3 Flight-simulation testing

As said before correct life predictions require flight-simulation tests. Although this appears to be a trivial conclusion and hence a simple solution we are still left with a number of problems when carrying out flight-simulation tests.

Let us assume that the test is carried out on a full-scale component or structure and we thus need not bother about notch effect, size effect and fretting corrosion because the real part is tested. Differences between the test and service conditions may be related to the following aspects (see also table 3).

(a) Load spectrum
(b) Maximum load applied in the test
(c) Loading rate
(d) Corrosive influences
(e) Scatter of properties.
Some comments on these topics will be made below.

Load spectrum

Load measurements in service may indicate that the service load history is significantly deviating from the history applied in the test. Suggestions were heard in the past that the test result could be corrected for such deviations by calculation employing the Palmgren-Miner rule and some S-N curves. However, it is the opinion of the author that this rule is highly inaccurate for this purpose. It is well known that the Palmgren-Miner rule cannot account for including or omitting low-amplitude fatigue loads (see for instance section 5.5). Secondly adding or omitting very high loads in general hardly affects $\sum n/N$ while these loads have a considerable effect on the fatigue life. As a third example the damage predicted by the Palmgren-Miner rule for adding or omitting GTAC is highly incorrect.[15,28]
For a fail-safe structure the question of deviating load spectra in service is probably less important if the structure has good fail-safe properties. For a safe-life component, however, it appears that additional testing is indispensable if one feels that the service loading could be more severe than the test loading.

Maximum load applied in the test

The importance of the maximum load applied in a flight-simulation test was emphasized in Ref.15. It was recommended that the high load cycles should be truncated to the load level that on the average would be reached or exceeded 10 times in the target life of the aircraft. A similar recommendation was made for crack growth studies, but then instead of the aircraft life one has to consider the inspection period. The predominant influence of high loads on crack growth was dramatically illustrated by the results presented in chapter 5. If a full-scale structure with cracks is tested in order to study the crack rate, high loads will considerably delay the crack growth (see for instance in figure 8C the data for $S_{a,max} = 12.1$ kg/mm$^2$). The application of high loads may considerably flatter the test result. Therefore a truncation is necessary in order to avoid unsafe predictions for those aircraft of the fleet that will never see the very high loads.

Sometimes fail-safe loads are applied at regular intervals during a full-scale fatigue test to demonstrate that the aircraft is still capable of carrying the fail-safe load. If this load exceeds the highest load of the fatigue test the result may well be that a number of cracks that escaped detection so far will never be found because of crack growth delay. In other words this procedure could eliminate the possibility of obtaining the information for which the fatigue test is actually carried out.

Loading rate

A full-scale test on a structure is an accelerated flight simulation that may last for 1/2 - 1 year, while representing 10 or more years of service experience. A flight-simulation test on a component in a modern fatigue testing machine may take no more than a few weeks.
Considering loading rate effects one should not simply compare testing time with flying time, but rather the times that the structure is exposed to the high loads. This argument is speculating that any effect of the loading rate is a matter of some time-dependent dislocation mechanisms occurring at high stresses. It might imply that the effect is relatively small for a full-scale test but could have some importance for a component test in a fatigue machine running at a relatively high load frequency. No speculation will be made here on the quantitative magnitude of such a frequency effect under flight-simulation loading.

Corrosive influences

Differences between testing time and service life imply different times of exposure to corrosive attack. Therefore if corrosion is important for crack nucleation (corrosion fatigue) one certainly should consider this aspect.
In practice cracks frequently originate from bolt holes and rivet holes where the accessibility of the environment usually is poor and probably the corrosion influence is not very significant. However, as soon as macro-cracks are present the environment will penetrate into the crack and the effect on crack growth should be considered. Empirical data are not very abundant. In a comparative investigation of our laboratory [29] on 2024 and 7075 sheet material crack propagation was simultaneously studied in an indoor and an outdoor environment. Programme loading and random flight-simulation loading were employed. The results indicated a negligible effect for the 2024 material, but for the 7075 alloys the crack growth outdoors was 1.5 to 2 times faster than indoors. Figge and Hudson [30] in a recent study found a similar trend. It will be clear that allowances have to be made for this effect.

Scatter

Testing a symmetric structure generally implies that at least two similar parts are being tested. However, fatigue properties may vary from aircraft to aircraft because the quality of production techniques and materials will not remain exactly constant from year to year. I will not try to speculate on the magnitude of the scatter although some interesting data are available in the literature.

It will be recognized that the shortest fatigue lives in a large fleet of aircraft in general will be shorter than the result of the full-scale test.

## 7.4 Comparison between test and service experience

After having summarized several limitations of a full-scale flight-simulation test the proof of the pudding actually is the comparison between service experience and test results. Papers on this issue have been presented at previous ICAF symposia [31,32,33] and some comments will be made below.

As far as data are available the service life is usually shorter than the test life, although there are some cases where the agreement is reasonable. However, if the service life is from 2 to 4 times shorter than the test life further clarification is obviously needed.

There are a number of reasons why discrepancies between test results and service experience may occur. Several of them were just listed: scatter, environmental effects and loading rate. Secondly a fair comparison requires that the test was a realistic simulation of the service load history and this probably is a severe restriction on the comparisons that could be made in the past. Thirdly if a test reveals a serious fatigue failure it is likely that the aircraft firm will modify the structure, thus eliminating the possibility of a comparison.

If we now try to answer the question whether a full-scale flight-simulation test is really worthwhile as advocated by Branger [34] the following arguments should be mentioned.

1. If there are unintentional structural deficiencies the test will reveal these items.
2. The fatigue safety largely depends on the fail-safe properties of a structure [35,36] and for proving adequate fail-safe properties full-scale tests are to be recommended. In addition airworthiness requirements are stimulating realistic testing. [37]
3. The test may reveal several cracks which are insignificant for the safety but which are important for the economic operation of the aircraft [38,39]. Harpur and Troughton [33] and also Rhomberg [40] evaluating this argument came

to the conclusion that full-scale testing is fully justified from the economic point of view. In many cases the economic argument will not allow an easy and relevant evaluation, but we should recognize that a flight-simulation test on a full-scale structure is equivalent to "flying the aircraft in the test hall" and valuable structural experience may be gained.

It has to be said explicitely that the validity of the above arguments requires that the flight load simulation has a realistic nature. All significant fatigue loads should be included in the test while truncation of the load spectra is another important consideration. Fortunately from an experimental point of view we are in a much better position than we were in the past decades.

If we finally consider the question whether a full-scale test could also give quantitative indications on fatigue lives it is my opinion that this is possible provided the testing is a realistic flight simulation (see also Ref.15). The limitations on the accurary as discussed before need not be too severe, and to some extent allowances can be made. Some people may feel that this is an art rather than a science, but even then I think that intuition and a profound knowledge and understanding of revelant information is justifying some optimism.

## 8. CONCLUSIONS

In the present paper various aspects of estimating fatigue lives and conducting fatigue tests have been dicussed. Secondly a survey has been given of the results of a recent investigation on fatigue crack propagation under flight-simulation loading. The main findings of this test programme will be summarized first, while some conclusions of the general discussion will follow afterwards.

### Crack propagation in 2024-T3 and 7075-T6 clad material under flight-simulation loading

1. Omission of taxiing loads from the ground-to-air cycles did not affect the crack propagation.
2. Omission of small gust cycles ($S_a = 1.1$ kg/mm$^2$) slightly increased the crack propagation life. Omission of additional small gust cycles ($S_a = 2.2$ kg/mm$^2$) doubled the life.
3. The predominant effect on the crack propagation life was exerted by the maximum gust amplitude (truncation level). Increasing this amplitude from 4.4 to 8.8 kg/mm$^2$ linearly increased the crack propagation life from 2500 to 15000 flights and from 6000 to 25000 flights for the 7075 and 2024 specimens respectively (fig.11). The effect is larger for the 7075 alloy.

4. The sequence of the gust loads within each flight had an insignificant effect on the crack propagation life.
More details on the tests and the results are given in chapter 5.

Fatigue damage accumulation

5. Fatigue damage cannot be fully defined by a single parameter such as crack length. In addition crack blunting, crack orientation, strain hardening and residual stress may be significant aspects of the fatigue damage. This implies that the extension of the crack during a certain load cycle will depend on the pre-history of the fatigue loading (fig.18).
6. As a consequence of the previous conclusion we must not exclude the fact that the fatigue lives under random loading and programme loading can be significantly different.
7. Residual stresses are thought to be responsible for the large effect of the maximum gust amplitude as mentioned in conclusion 3.

Estimating fatigue lives

8. In the design phase of an aircraft accurate life estimates cannot be obtained by calculation only. Rough estimates, however, can be made in several ways and the damage rule adopted is not necessarily the major source of inaccuracy.

Fatigue testing procedures

9. Constant-amplitude fatigue tests may give misleading information in comparative design studies. Programme testing is a considerable improvement while random load tests in many cases will be a further improvement. It is even more realistic to adopt flight-simulation tests for this purpose.
10. A full-scale fatigue test on a newly designed aircraft has to be recommended for several reasons. In order to obtain relevant quantitative information on the fatigue quality of the structure the application of a realistic flight-simulation loading is required. Particular attention should be paid to truncating high-amplitude fatigue load cycles in view of the favourable effect of these loads on the fatigue life and the crack propagation.
11. The fatigue properties of an aircraft in service may deviate from the results in a full-scale flight-simulation test because of such aspects as a deviating load spectrum, scatter of fatigue properties and influences of environment and loading rate. Allowances can be made in a qualitative way depending on skilful judgement.

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TABLE 1 SURVEY OF AIRCRAFT FATIGUE PROBLEMS

| | | |
|---|---|---|
| Design phase | • Type of structure, fail – safe characteristics | |
| | • Joints | |
| | • Detail design | |
| | • Materials selection | |
| | • Surface treatments | |
| | • Production techniques | |
| | • Airworthiness requirements | |
| | • Prediction of fatigue environment | ■ |
| | • Dynamic response of the structure | |
| | • Estimation of fatigue properties : | |
| | fatigue lives | ■ |
| | crack propagation | ■ |
| | fail – safe strength | |
| | • Exploratory fatigue tests for : | |
| | design studies | ■ |
| | support of life estimations | ■ |
| Construction of aircraft prototypes<br>Test flights | • Load measurements in flight | |
| | • Proof of satisfactory fatigue properties by testing components or full structure | ■ |
| | • Allowances for service environment | ■ |
| | • Structural modifications | |
| | • Inspection procedures for use in service | |
| Aircraft in service | • Load measurements in service | ■ |
| | • Corrections on predicted fatigue properties | ■ |
| | • Cracks in service, relation to prediction | ■ |
| | • Structural modifications | |

Problems involving aspects of fatigue damage accumulation are indicated by ■

TABLE 2 THREE MAJOR PHASES OF ESTIMATING FATIGUE PROPERTIES

FATIGUE ENVIRONMENT → FATIGUE LOADS IN THE STRUCTURE → FATIGUE RESPONSE OF THE STRUCTURE

TABLE 3 VARIOUS ASPECTS OF THE AIRCRAFT FATIGUE ENVIRONMENT.

| | |
|---|---|
| Load – time history | ● Mission analysis, flight profiles |
| | ● Fatigue loads |
| | gusts |
| | manoeuvres |
| | GTAC |
| | ground loads |
| | acoustic loading |
| | etc. |
| | ● Statistical description of fatigue loads |
| | Counting of peaks, ranges, etc. |
| | PSD approach |
| | Unstationary character of environment |
| | Scatter of environmental conditions |
| | ● Sequence of fatigue loads |
| | ● Loading rate |
| | Time – frequency |
| | Wave shape |
| | Rest periods |
| Temperature – time history | ● Fatigue at low and high temperature |
| | ● Thermal stresses |
| | ● Interaction creep – fatigue |
| Chemical environment | ● Corrosion, influence on |
| | crack initiation |
| | crack propagation |
| | ● Interaction stress corrosion – fatigue |

TABLE 4 THREE PROCEDURES FOR PREDICTING FATIGUE LIVES IN THE DESIGN PHASE.

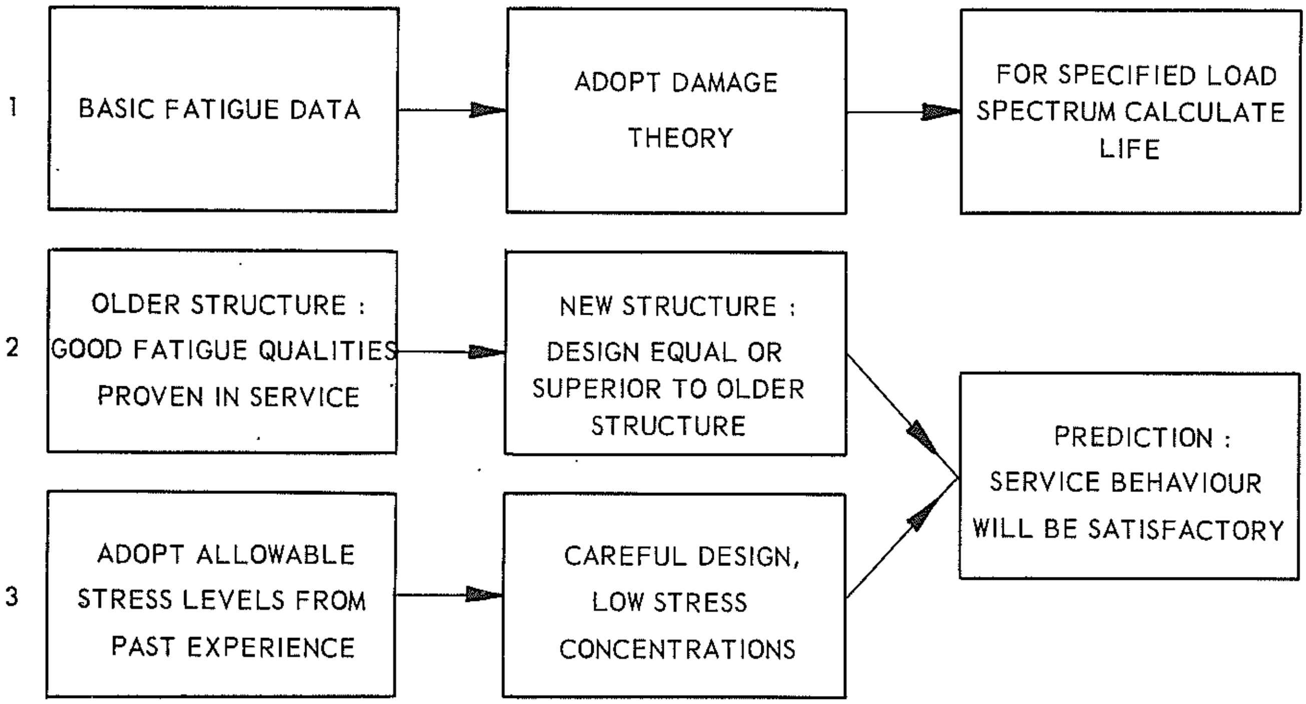



TABLE 5 ALTERNATIVES FOR BASIC FATIGUE DATA TO BE USED FOR LIFE ESTIMATIONS.

| | |
|---|---|
| S – N curves | ● Available S – N curves for simular components<br>● Calculate S – N curves, starting from $K_t$<br>● Calculate S – N curves, assuming an effective stress concentration factor<br>● Exploratory tests and estimate S – N curves from test data |
| Programme test results | |
| Random load test results | |

## TABLE 6 SEVERAL FATIGUE TEST LOADINGS, MAIN VARIABLES AND AIMS OF TESTING.

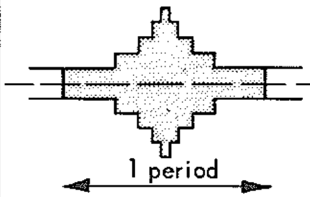
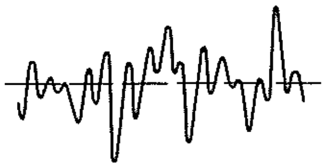
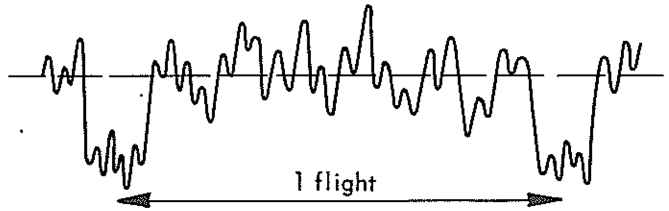
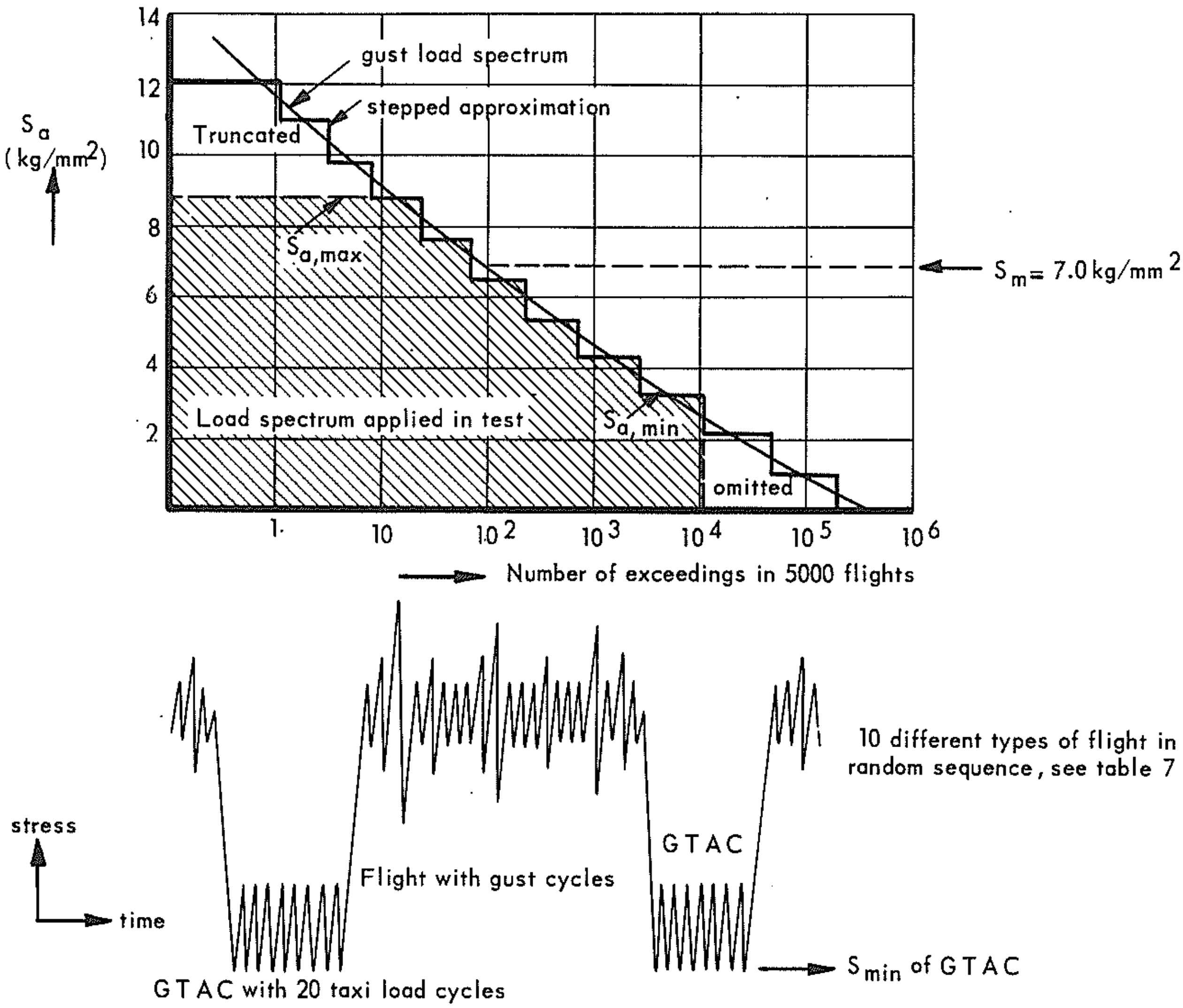
| Test | Conventional fatigue test | Programme test | Random test | Flight – simulation test: on specimens | Flight – simulation test: on structure |
|---|---|---|---|---|---|
| Loading |  |  1 period |  |  1 flight | |
| Main variables of the load | $S_m$<br>$S_a$ | $S_m$<br>Load spectrum<br>Length of period<br>$S_{a,max}$ | $S_m$<br>$(S_a)_{rms}$<br>Irragularity factor<br>Clipping ratio | Assumed service load conditions | |
| Aims of testing | Basic data for life estimates | Basic data for life estimates | Basic data for life estimates | | |
| | Comparative design studies | Comparative design studies | Comparative design studies | Comparative design studies | |
| | | | | Direct life estimates | Direct life estimates |
| | | | | | Indication of fatigue critical elements. Crack rates, inspection, repairs etc. |

Table 7 Gust load occurrences in the 10 different types of flights

| Flight type | Number of flights in 5000 flights | Number of gust cycles with amplitude $S_a$ ($kg/mm^2$) | | | | | | | | | | | Total number of cycles per flight |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | $S_a$=12.1 | $S_a$=11.0 | $S_a$=9.9 | $S_a$=8.8 | $S_a$=7.7 | $S_a$=6.6 | $S_a$=5.5 | $S_a$=4.4 | $S_a$=3.3 | $S_a$=2.2 | $S_a$=1.1 | |
| A | 1 | 1 | 0 | 1 | 1 | 2 | 3 | 5 | 9 | 15 | 27 | 43 | 107 |
| B | 2 | | 1 | 1 | 1 | 1 | 2 | 4 | 8 | 14 | 26 | 43 | 101 |
| C | 2 | | | 1 | 1 | 1 | 2 | 3 | 7 | 12 | 25 | 43 | 95 |
| D | 10 | | | | 1 | 1 | 1 | 3 | 5 | 11 | 24 | 43 | 89 |
| E | 27 | | | | | 1 | 1 | 2 | 3 | 9 | 22 | 43 | 81 |
| F | 91 | | | | | | 1 | 1 | 3 | 7 | 18 | 43 | 73 |
| G | 301 | | | | | | | 1 | 2 | 4 | 15 | 42 | 64 |
| H | 858 | | | | | | | | 1 | 3 | 11 | 38 | 53 |
| J | 3165 | | | | | | | | | 1 | 7 | 26 | 36 |
| K | 543 | | | | | | | | | | 1 | 19 | 20 |
| Total number of cycles in all flights | | 1 | 2 | 5 | 15 | 43 | 139 | 495 | 1903 | 8000 | 39252 | 149902 | |
| Number of exceedings, see fig.1 | | 1 | 3 | 8 | 23 | 66 | 205 | 700 | 2603 | 10603 | 49855 | 199757 | |




Note : Each gust cycle consisted of a positive gust followed by a negative gust of equal amplitude (exept for tests with reversed gust cycles)

| | Variables of test program (see also fig. 3) |
|---|---|
| Gust load spectrum | $S_{a,\,max}$ (truncation)<br>$S_{a,\,min}$ (omission of many small cycles) |
| GTAC | $S_{min}$ (2 values) |
| Taxiing loads | Omission of taxiing loads (same $S_{min}$) |
| Flight profile | Omission of GTAC<br>Only one gust cycle per flight |
| Sequence | Random<br>Gust cycles in reversed sequence<br>Programmed per flight |
| Material | 2 Aℓ – alloys , 2024 – T3 and 7075 – T6 |

FIG. 1 SURVEY OF VARIABLES STUDIED IN THE PRESENT TEST SERIES.

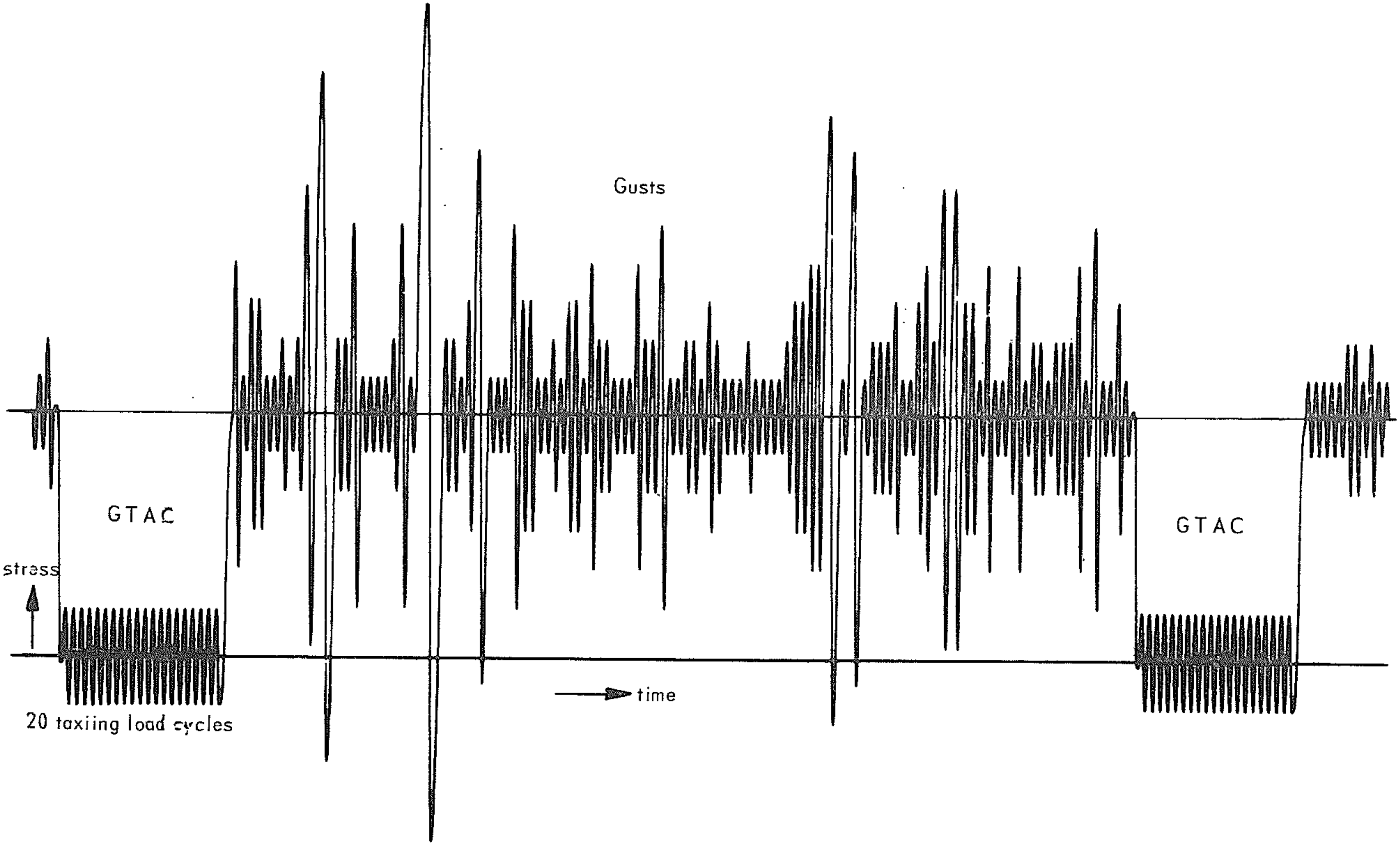
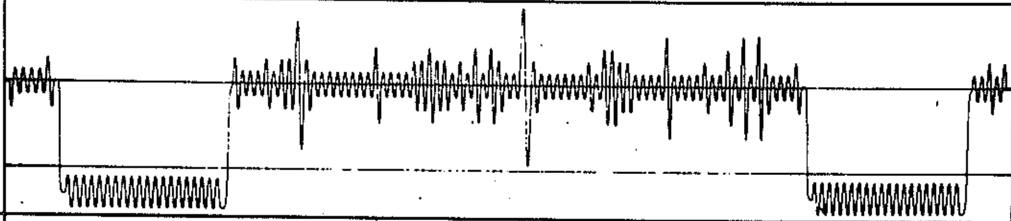
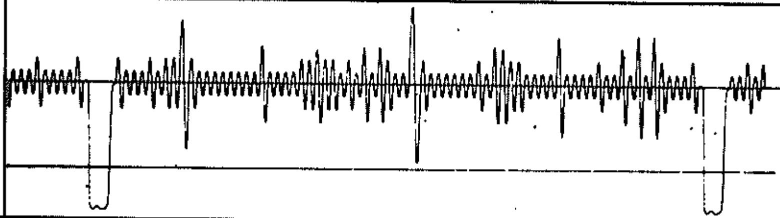
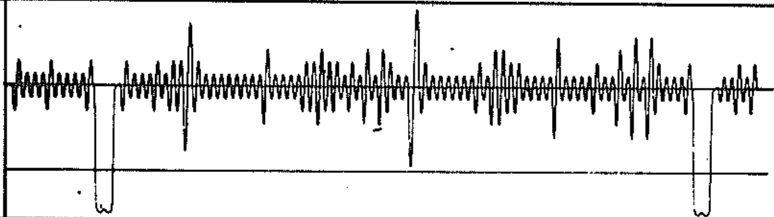
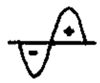
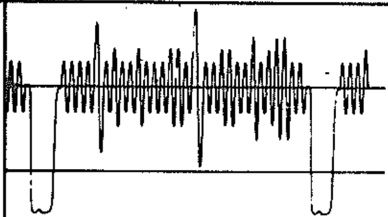
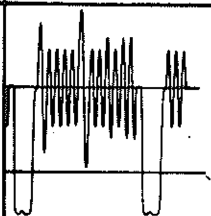
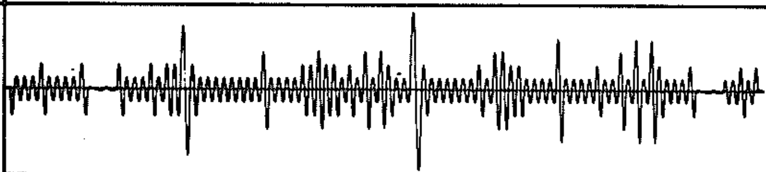
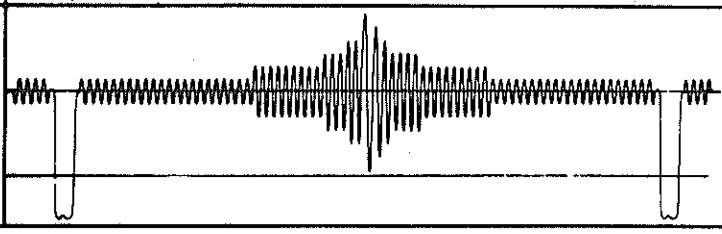
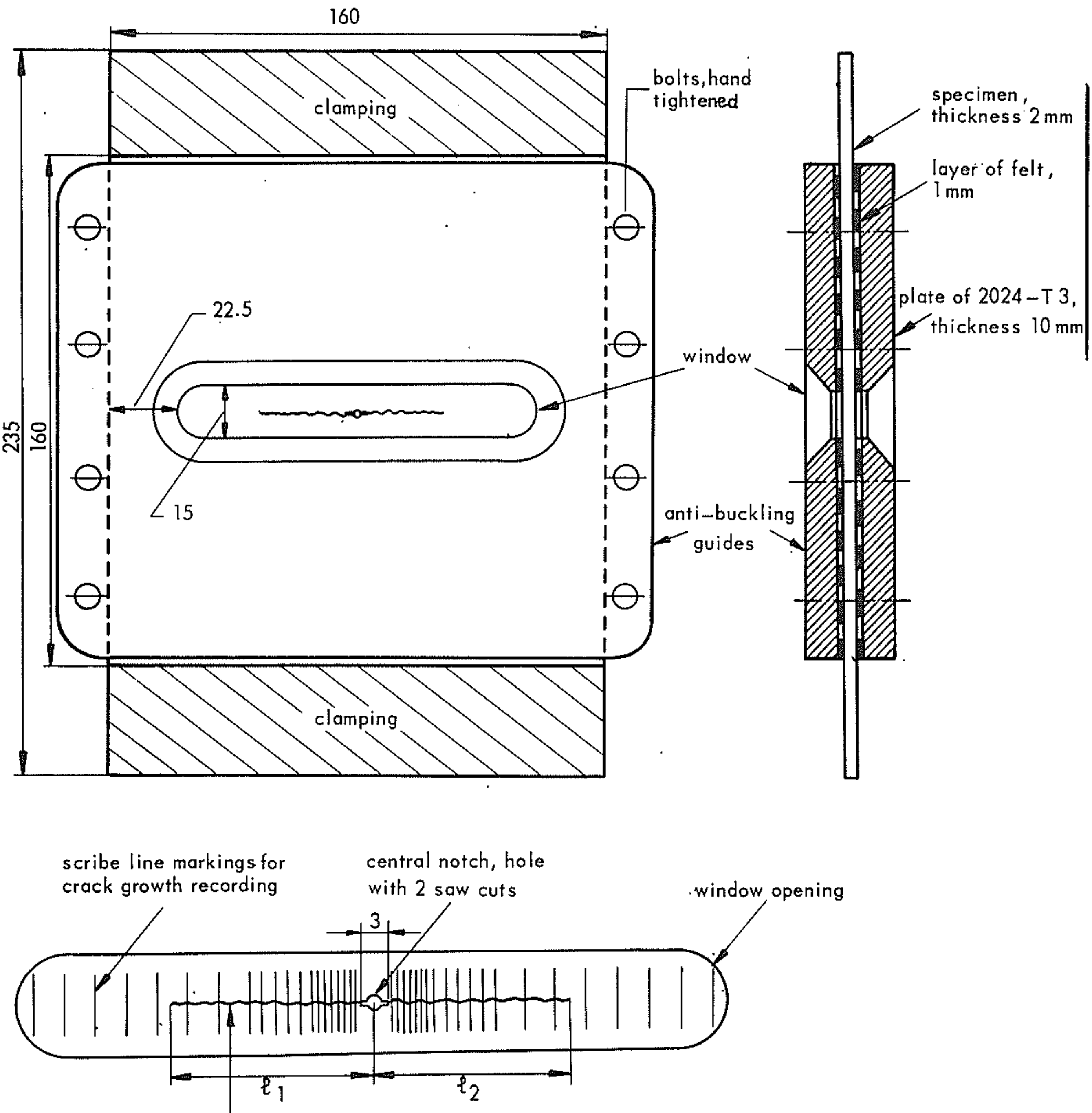



FIG. 2 THE LOAD SEQUENCE IN THE MOST SEVERE FLIGHT (TYPE A)

| | FIG. 3 LOAD RECORDS OF FLIGHT No. 19 (TYPE F) FOR DIFFERENT TYPES OF FLIGHT SIMULATION | |
|---|---|---|
| A |  | RANDOM FLIGHT SIMULATION |
| B |  | TAXIING LOADS OMITTED |
| C |  | GUST CYCLES IN REVERSED SEQUENCE  |
| D |  | SMALL GUST CYCLES OMITTED ( $S_a = 1.1$ kg/mm$^2$) |
| E |  | MORE SMALL GUST CYCLES OMITTED $S_a = 1.1$ AND $2.2$ kg/mm$^2$ |
| F |  | ONE GUST LOAD PER FLIGHT ONLY ( THE LARGEST ONE ) |
| G |  | GTAC OMITTED |
| H |  | PROGRAMMED SEQUENCE OF GUST CYCLES |




Definition of crack length :

$$\ell = \frac{\ell_1 + \ell_2}{2}$$

All dimensions in millimeters
1 mm = 0.04 in

FIG. 4 DIMENSIONS OF THE SPECIMEN AND ANTI–BUCKLING GUIDES

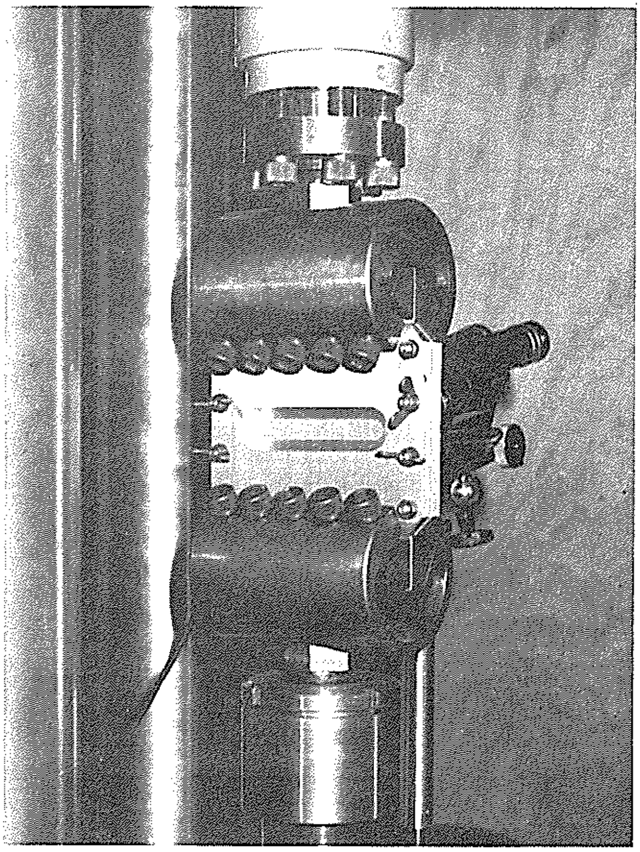


FIG. 5

PICTURE OF THE SPECIMEN, ANTI-BUCKLING GUIDES WITH WINDOW AND CLAMPINGS. STEREO-MICROSCOPE (30 x) FOR CRACK OBSERVATION IN THE BACKGROUND.

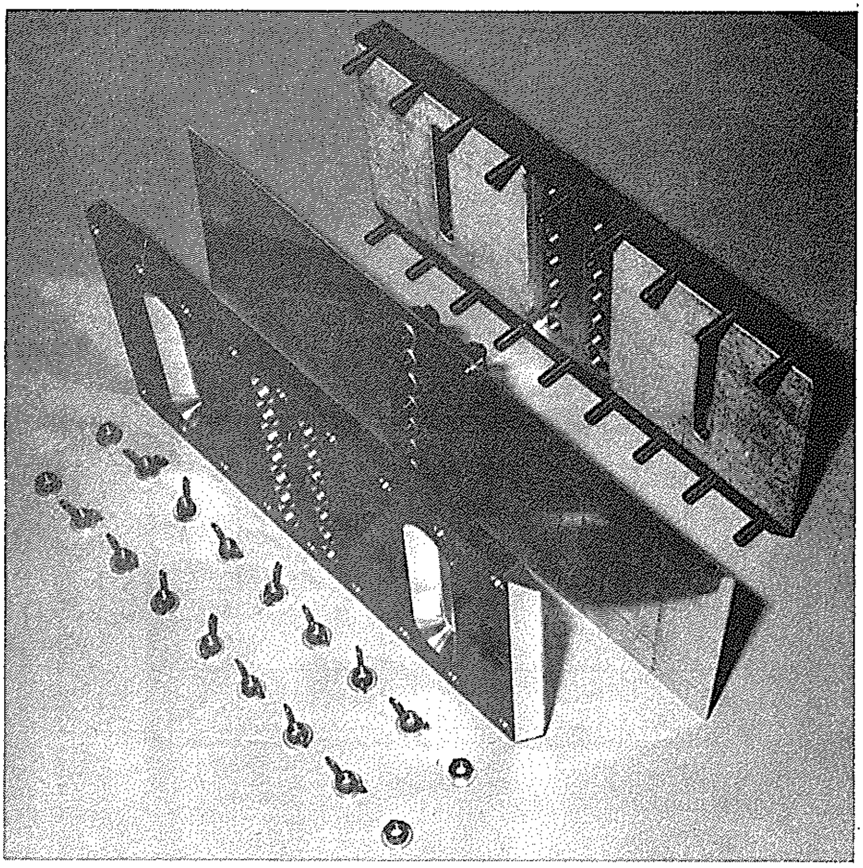


FIG. 6 TWO SPECIMENS CONNECTED BY A DOUBLE STRAP JOINT, ANTI-BUCKLING GUIDES COVERED BY FELT AT THE INNER SIDE AND PROVIDED WITH TWO WINDOWS EACH.

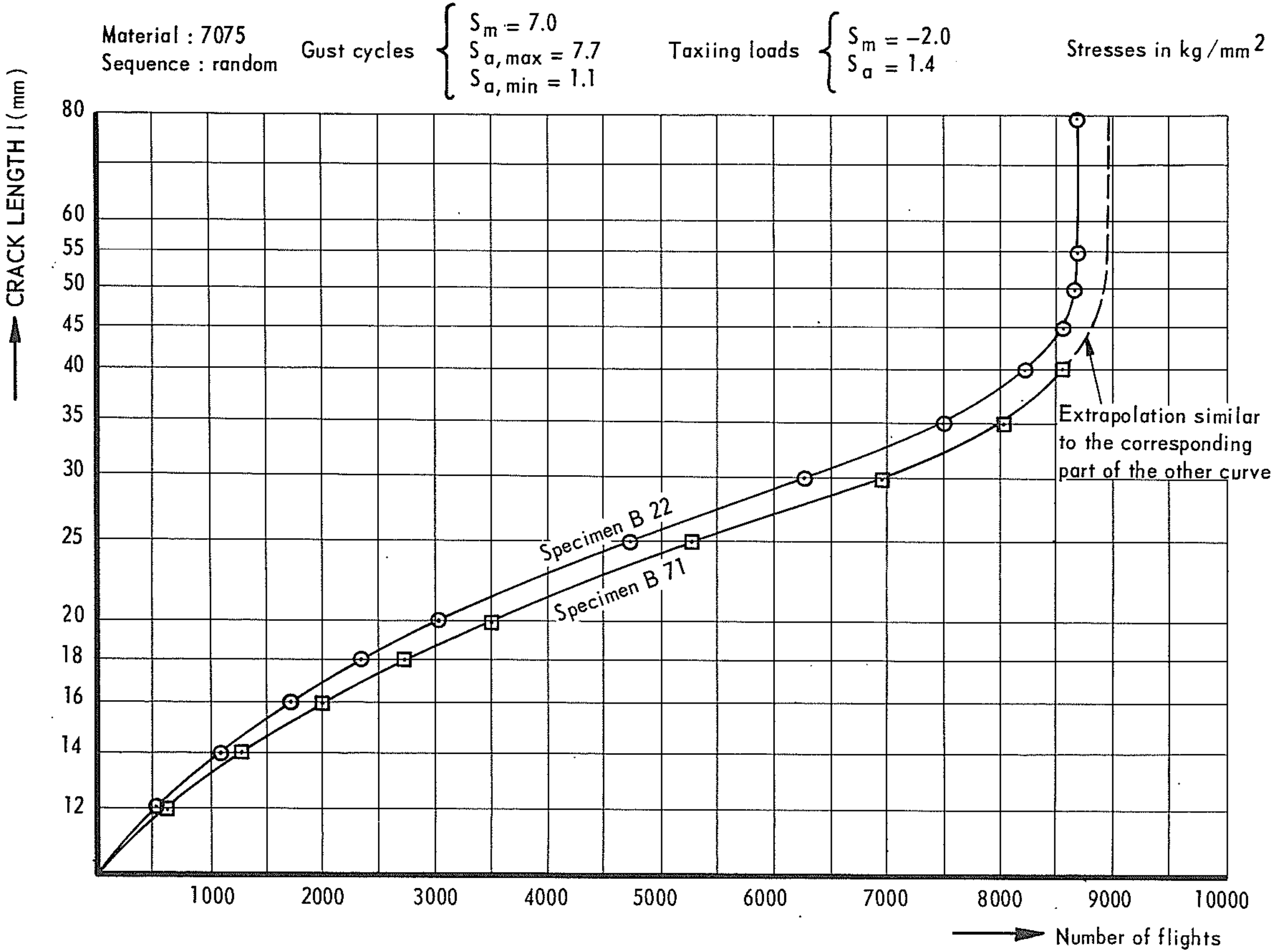



FIG. 7 EXAMPLE OF TWO CRACK PROPAGATION CURVES FOR TWO SPEÇIMENS SIMULTANEOUSLY TESTED IN SERIES

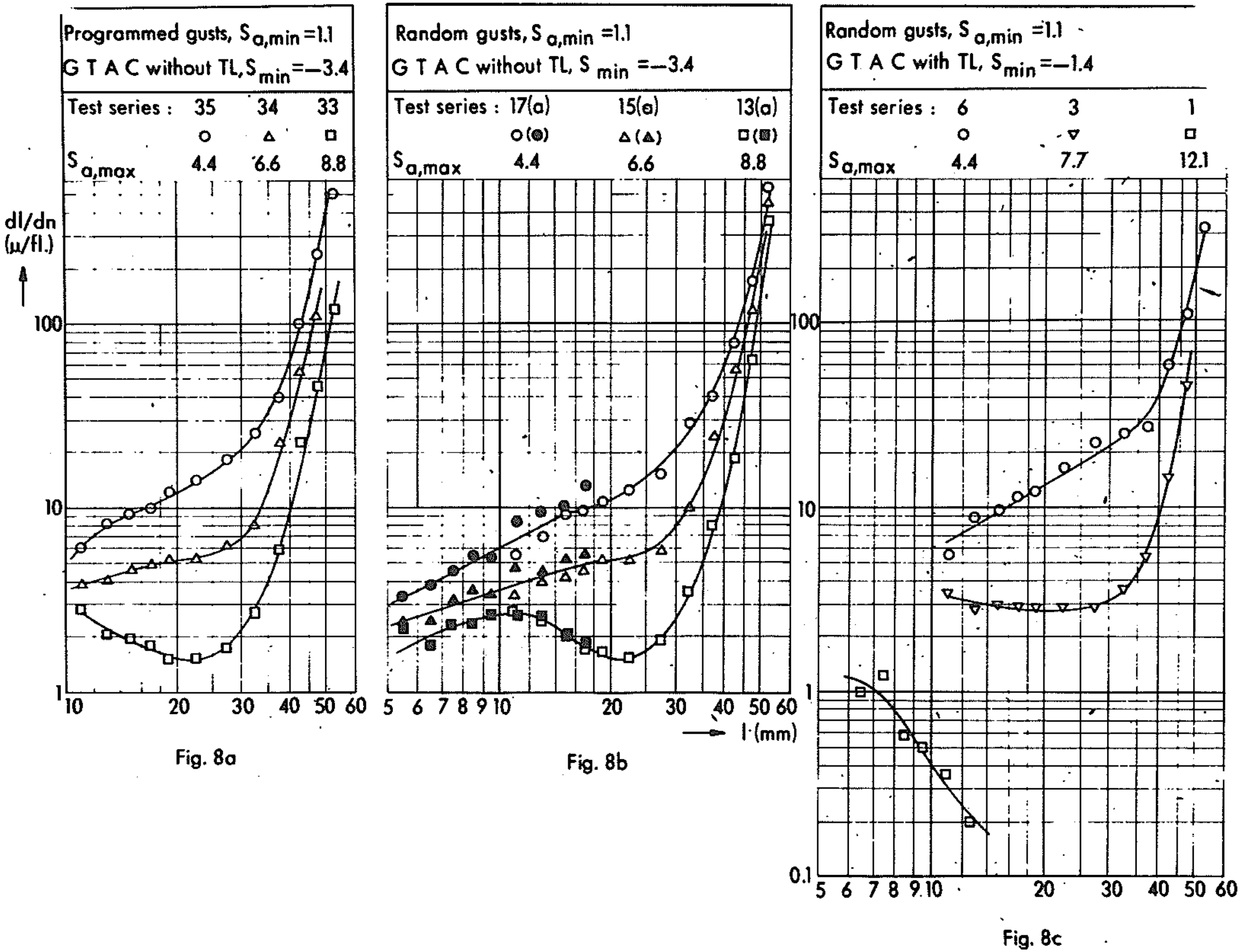



FIG. 8 EFFECT OF TRUNCATION ( $S_{a,max}$ ) ON THE CRACK PROPAGATION RATE

MATERIAL 7075—T6

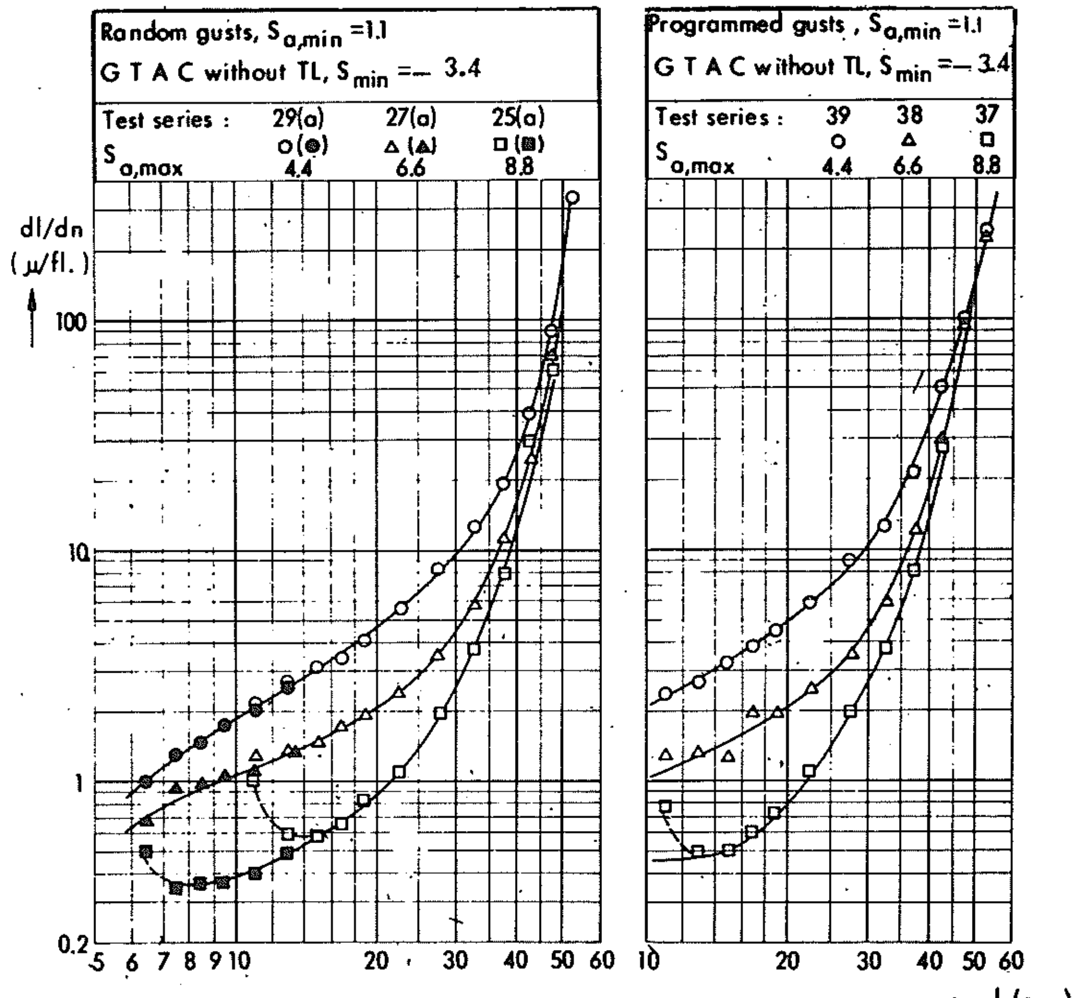



Fig. 8d

Fig. 8e

FIG. 8 , CONTINUED. MATERIAL 2024—T3

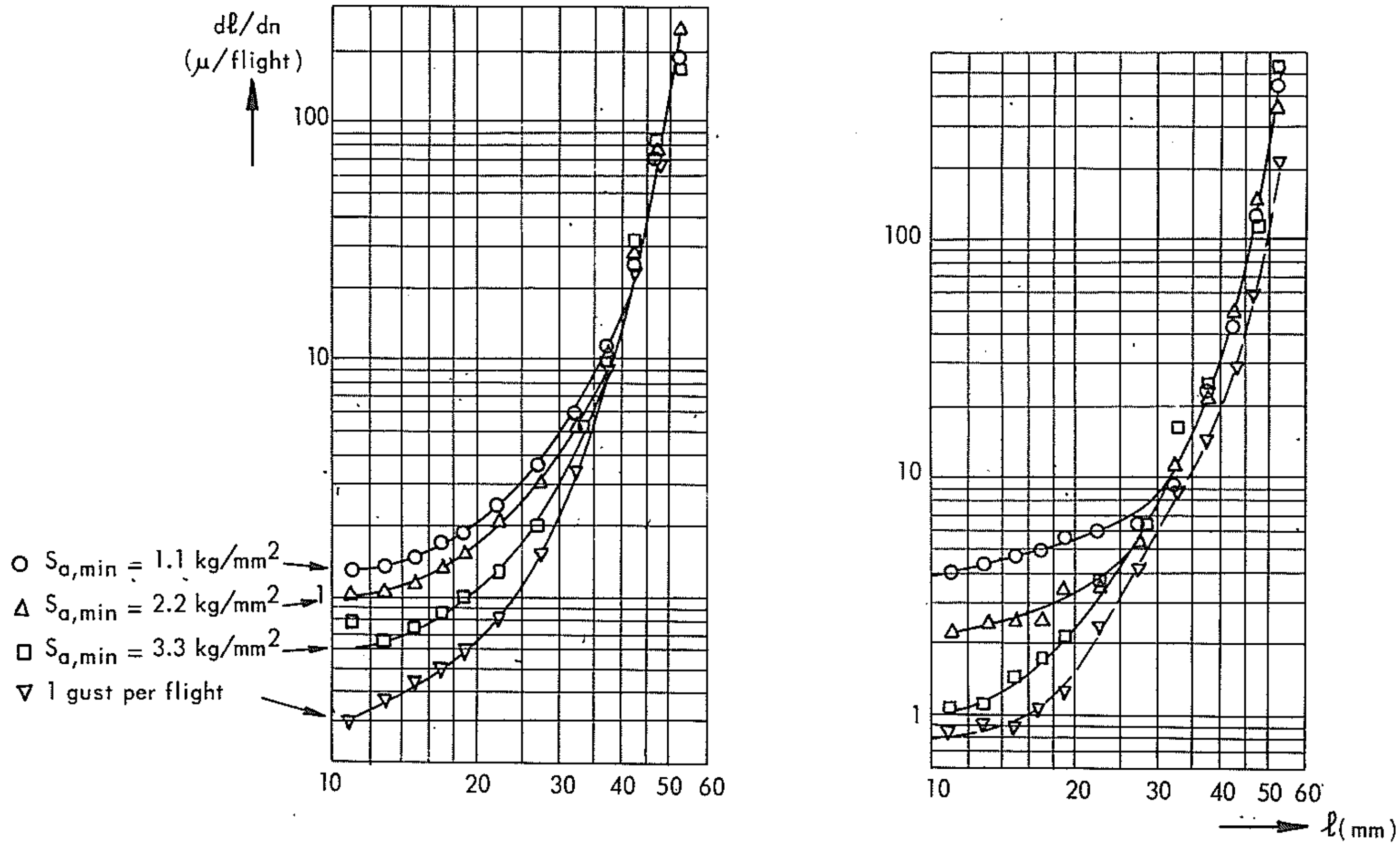



FIG. 9 THE EFFECT OF OMITTING SMALL GUST LOADS ON THE CRACK PROPAGATION RATE IN RANDOM FLIGHT – SIMULATION TESTS ($S_{a,max} = 6.6$ kg/mm²).

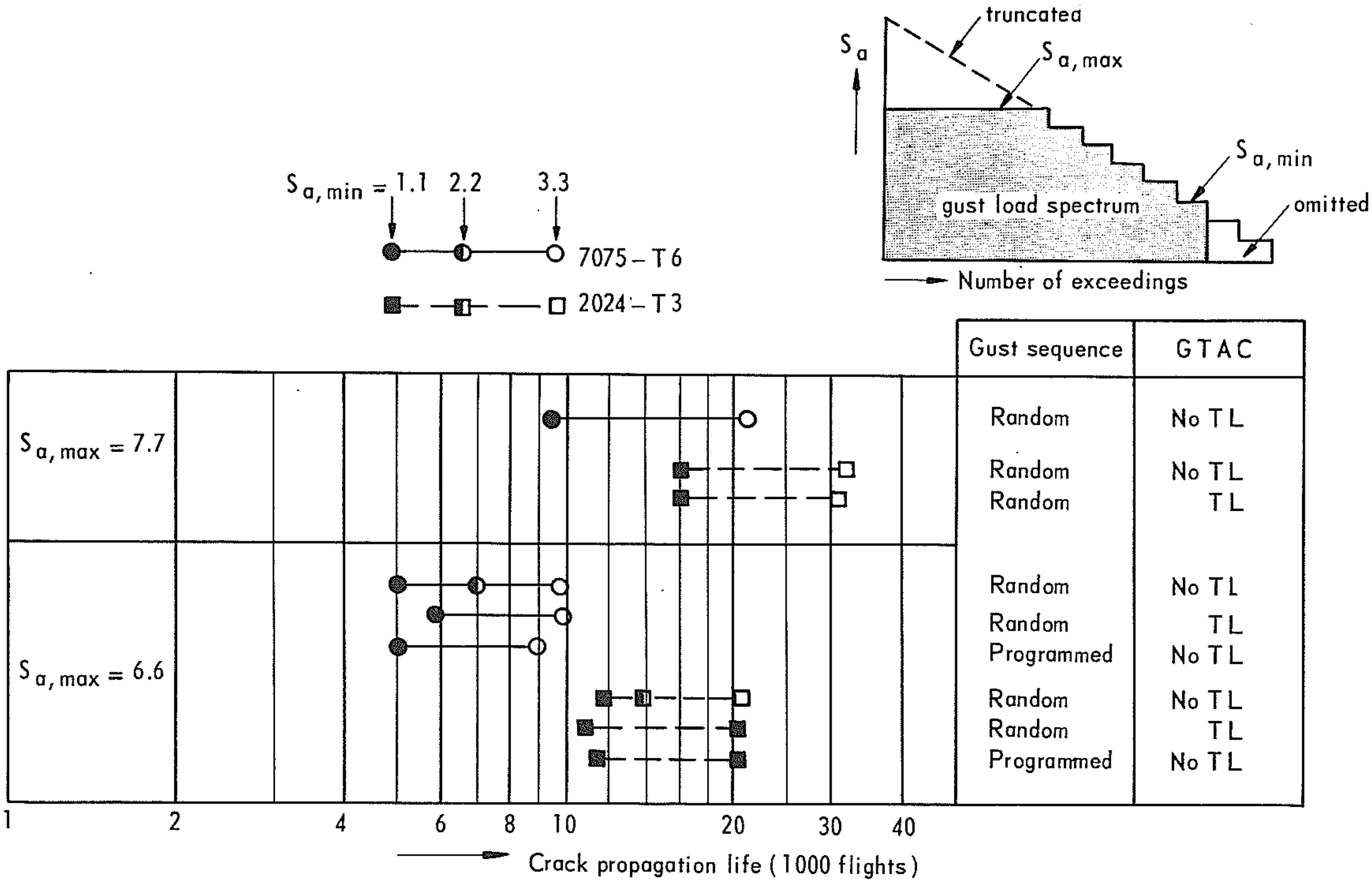
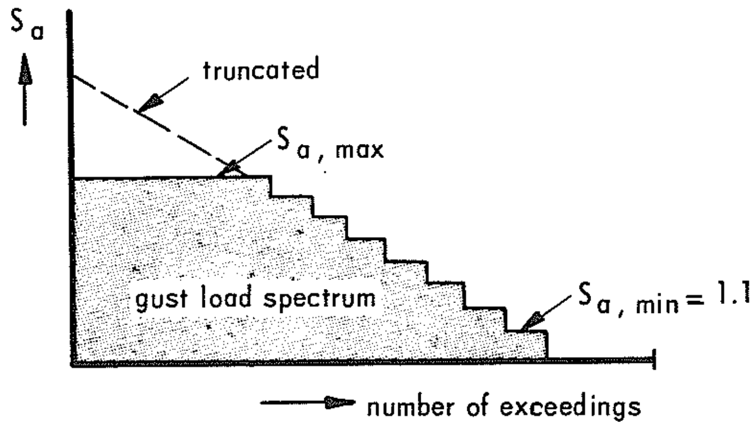



FIG. 10 THE EFFECT OF OMITTING SMALL GUST LOADS ON THE CRACK PROPAGATION LIFE.

| 7075-T6 | 2024-T3 | GTAC $S_{min}$ | TL | SEQUENCE |
|---|---|---|---|---|
| ○ | ● | −3.4 | no | random |
| □ | ■ | −3.4 | no | programmed |
| △ | ▲ | −3.4 | yes | random |
| ▽ | ▼ | −1.1 | yes | random |



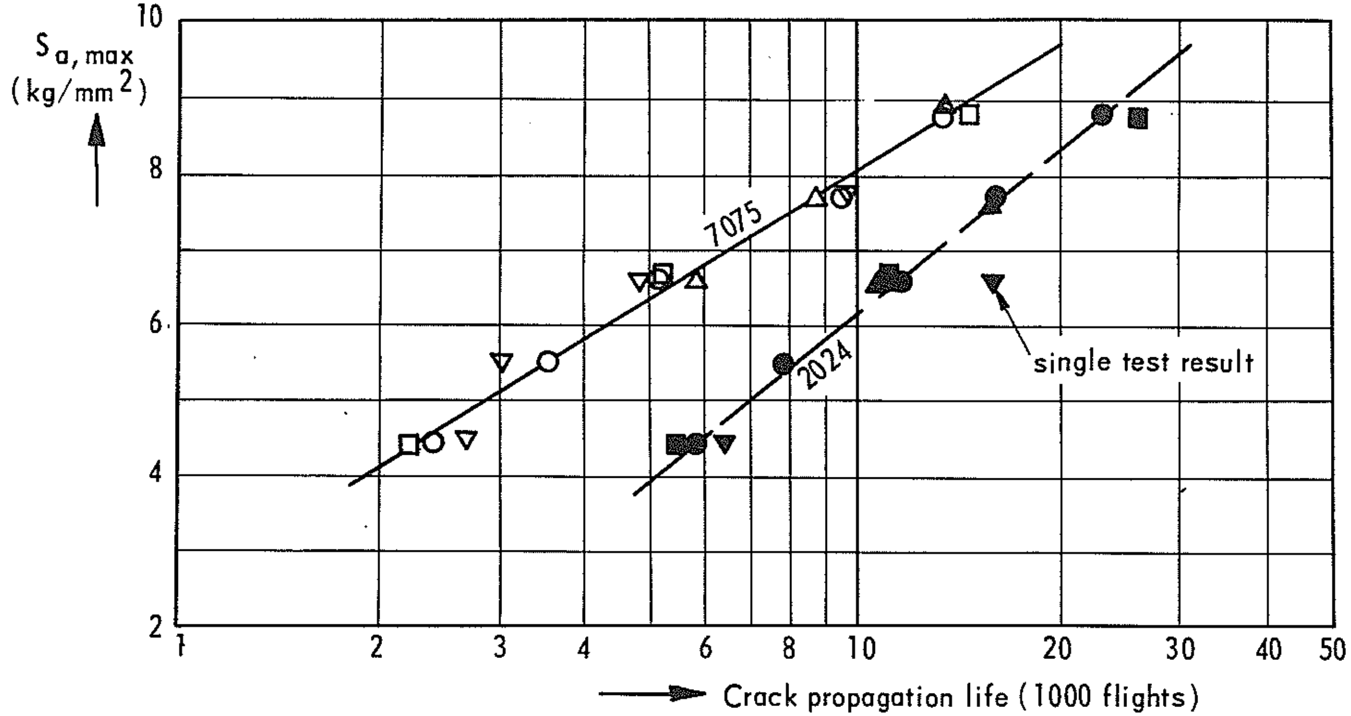




FIG. 11 THE EFFECT OF TRUNCATING THE GUST LOAD SPECTRUM ON THE CRACK PROPAGATION LIFE

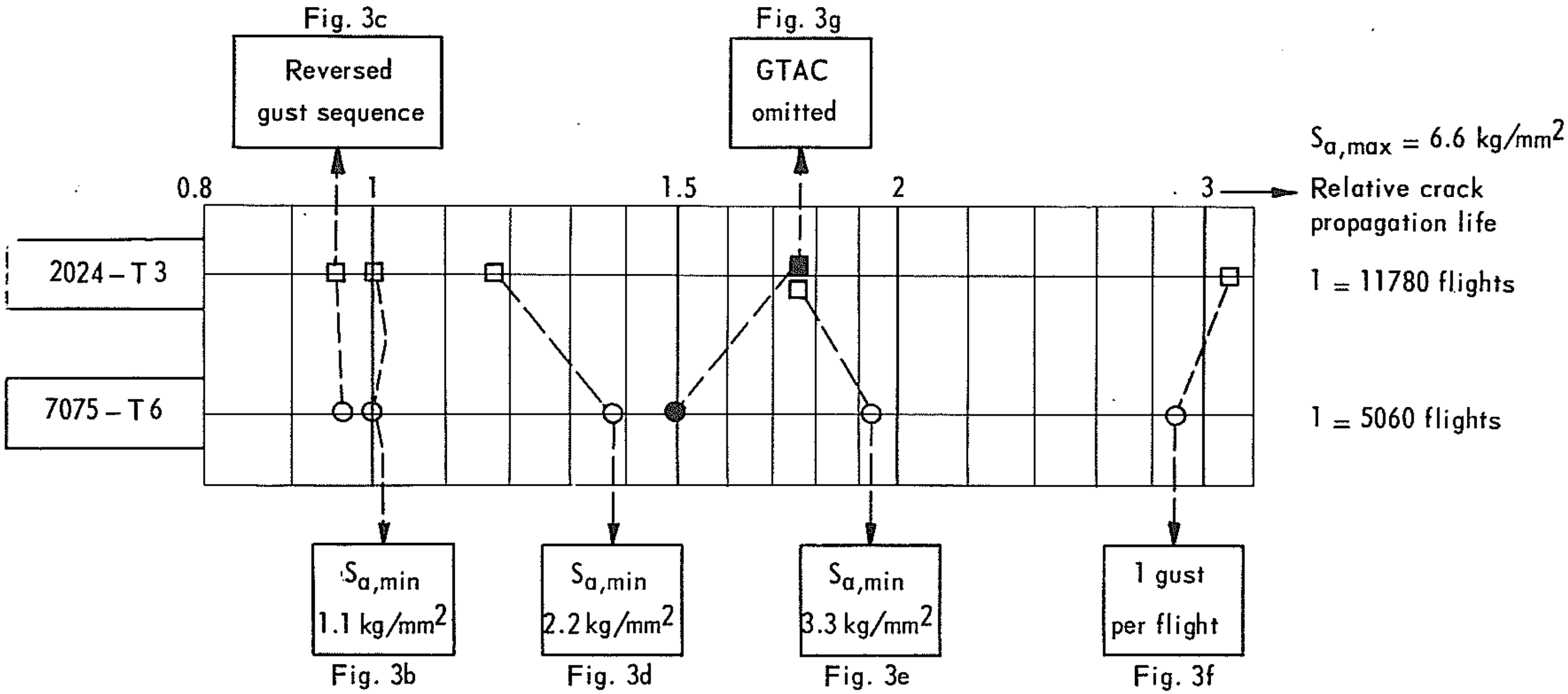



FIG. 12 RELATIVE CRACK PROPAGATION LIVES FOR RANDOM FLIGHT – SIMULATION TESTS.
EFFECT OF OMITTING SMALL GUST CYCLES, APPLICATION OF ONE GUST LOAD PER FLIGHT, OMITTING THE GTAC AN APPLICATION GUST CYCLES IN REVERSED SEQUENCE (NEG – POS).

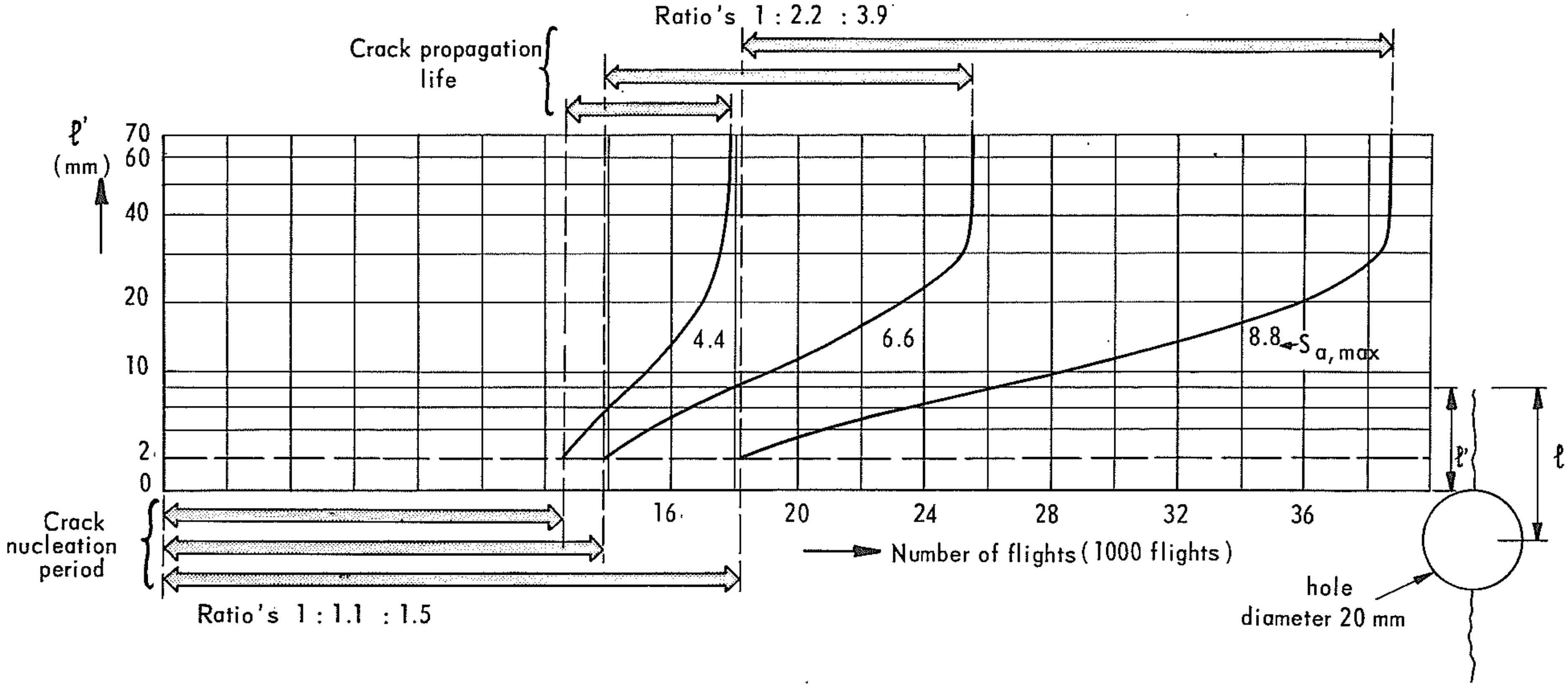



FIG. 13 CRACK PROPAGATION CURVES FOR THE 2024–T3 SPECIMENS WITH A CENTRAL HOLE. EFFECT OF TRUNCATION LEVEL ($S_{a,max}$) ON THE CRACK–NUCLEATION PERIOD (TO $\ell' = 2$ mm) AND THE CRACK PROPAGATION LIFE.

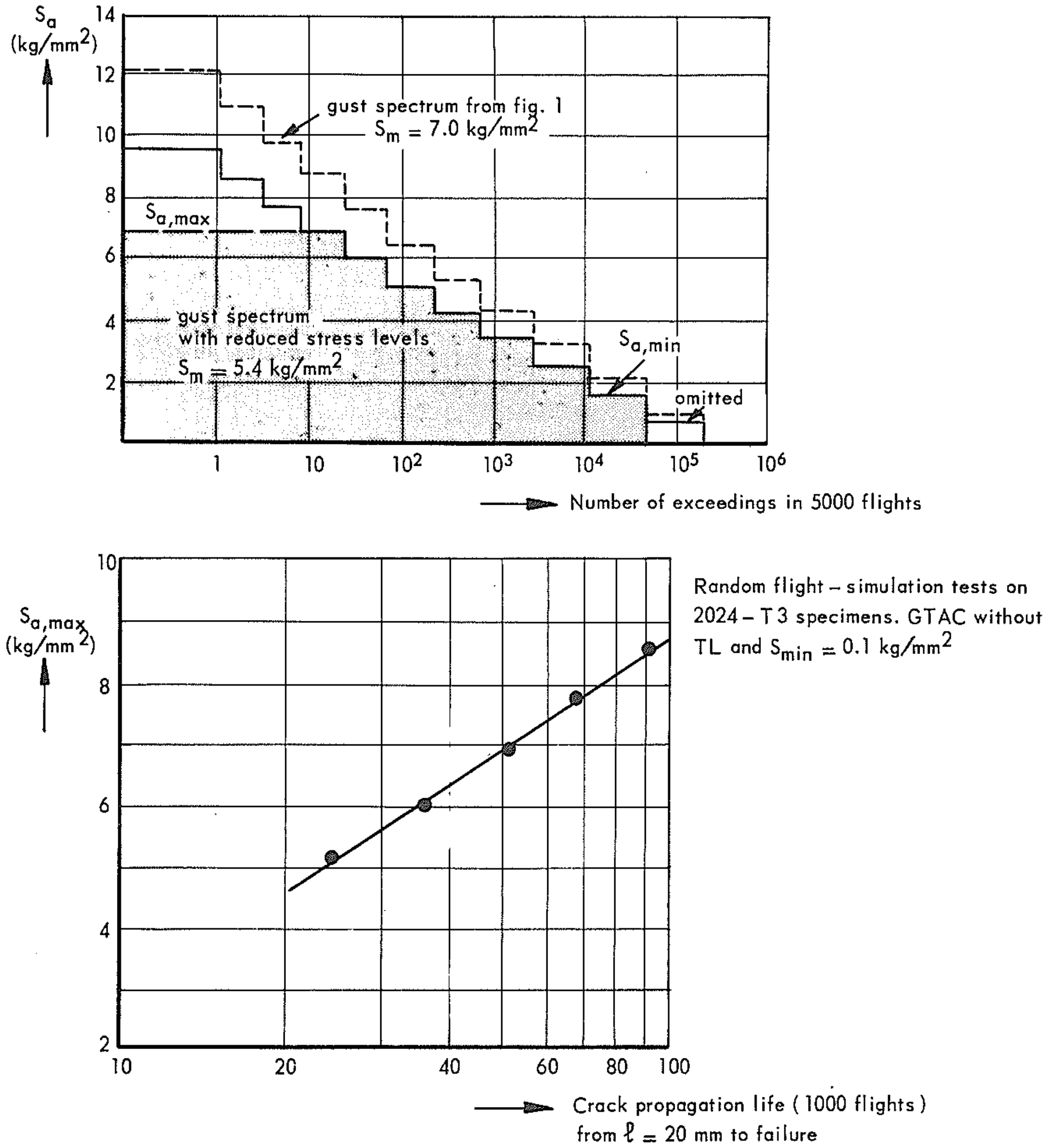



FIG. 14 THE EFFECT OF TRUNCATING THE GUST LOAD SPECTRUM, ADDITIONAL TESTS AT REDUCED STRESS LEVELS.

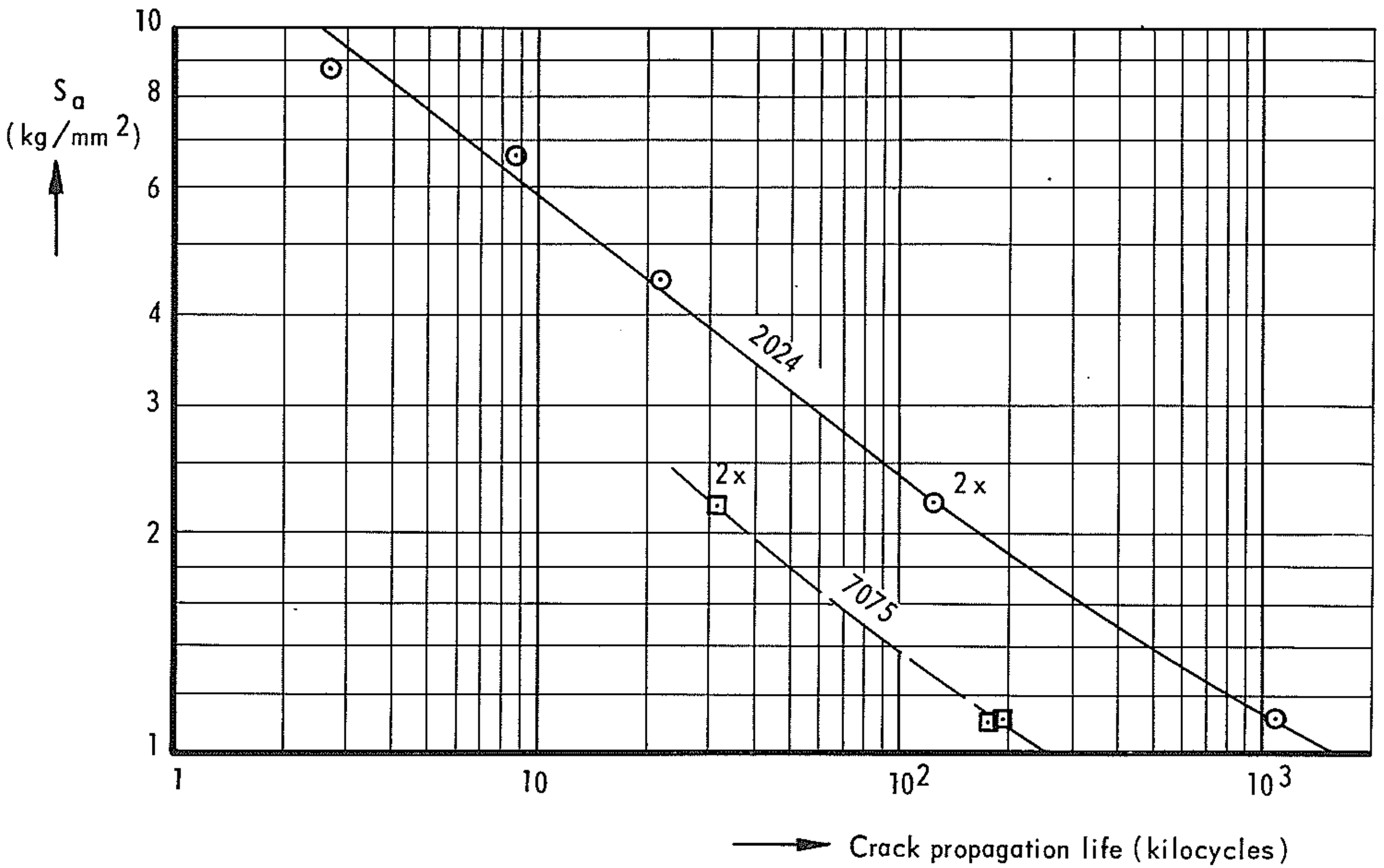
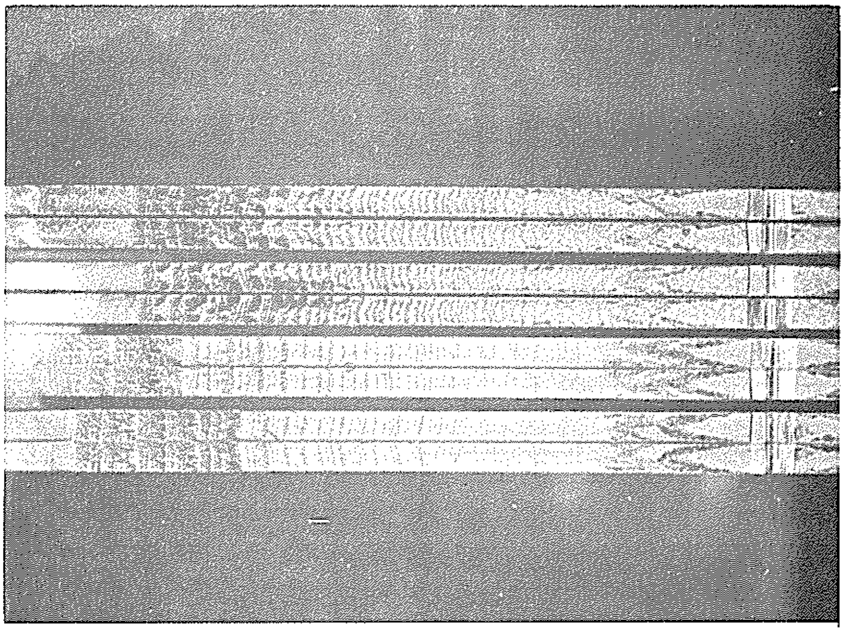



FIG. 15 THE CONSTANT-AMPLITUDE TEST DATA PLOTTED AS S-N CURVES



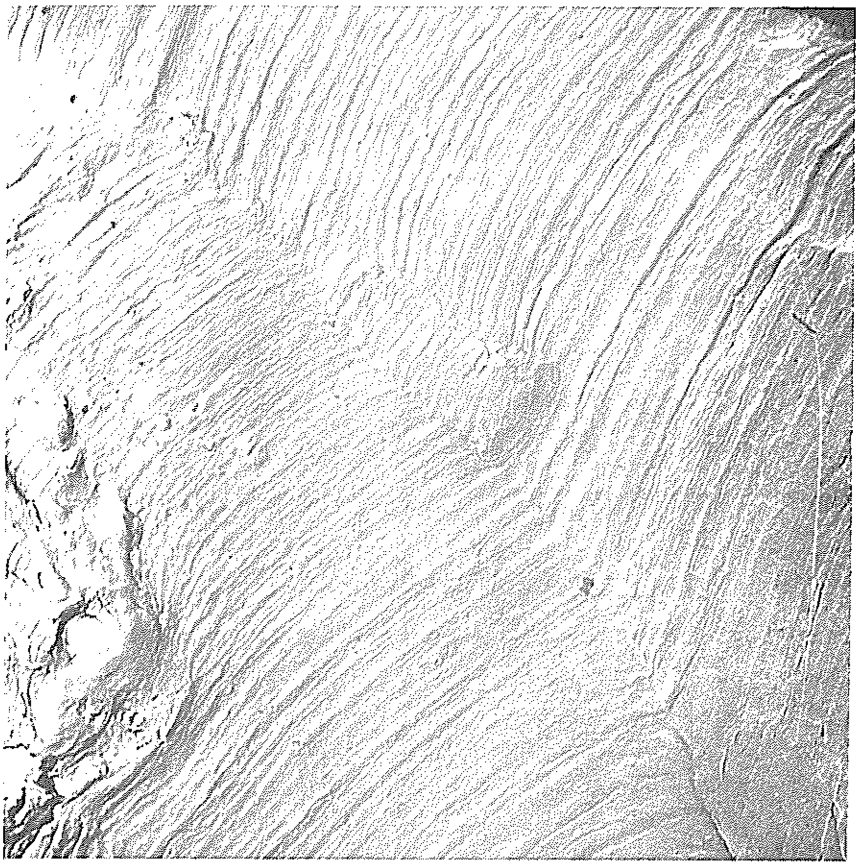
Random } 7075 – T 6
Programmed

Random } 2024 – T 3
Programmed

$S_{a,max} = 8.8$ kg/mm$^2$
$S_{a,min} = 1.1$ kg/mm$^2$
GTAC without TL,
$S_{min} = -3.4$ kg/mm$^2$

Magnification 2 x
Central notch at right side of picture

FIG. 16 FRACTURE SURFACES OF 4 SPECIMENS SHOWING MACRO FATIGUE BANDS.

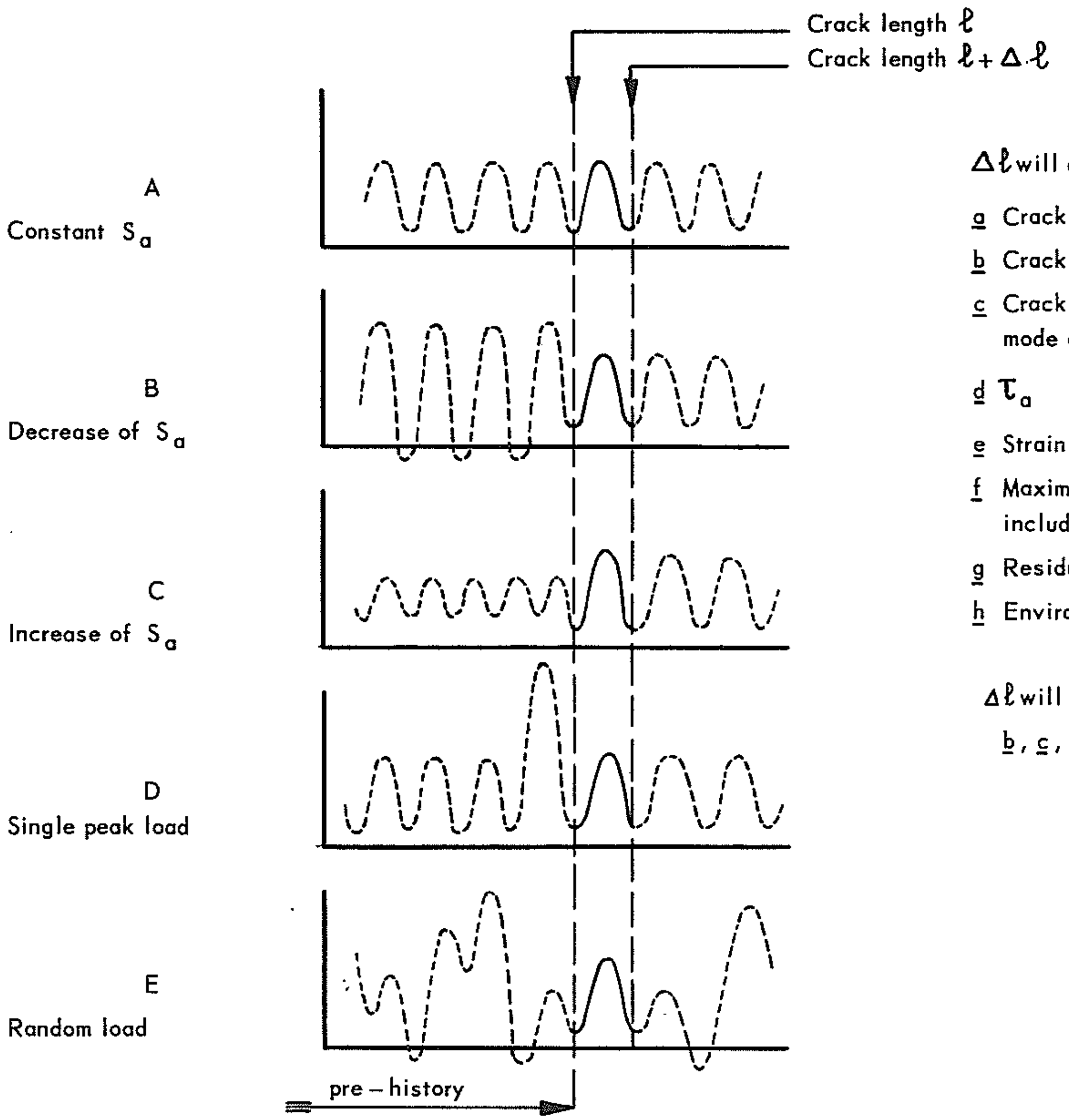


Specimen B47, 7075 – T6
Random flight simulation.
$S_{a,max} = 6.6 \ kg/mm^2$
$S_{a,min} = 3.3 \ kg/mm^2$
GTAC with TL
$\ell = 14$ mm   $d\ell/dn =$
1.3 $\mu$/flight
Magnification 5000 x

Specimen B18, 7075 – T6
Programmed flight simulation
$S_{a,max} = 4.4 \ kg/mm^2$
$S_{a,min} = 1.1 \ kg/mm^2$
GTAC without TL
$\ell = 20$ mm  $d\ell/dn =$
13 $\mu$/flight
Magnification 5000 x

FIG. 17 TWO EXAMPLES OF FATIGUE STRIATIONS AS OBSERVED BY THE ELECTRON MICROSCOPE.

FIG. 18 CRACK EXTENSION AS AFFECTED BY THE LOAD HISTORY




$\Delta\ell$ will depend on :

- a Crack length $\ell$
- b Crack blunting
- c Crack orientation, tensile mode or shear mode

} CRACK GEOMETRY

- d $\tau_a$
- e Strain – hardening
- f Maximum tensile stress including g
- g Residual stress
- h Environment

} CONDITIONS AT TIP OF CRACK

$\Delta\ell$ will be different for A, B, C, D and E because b, c, e, and g are dependent on the pre – history