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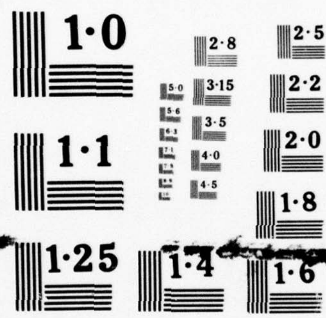

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FATIGUE DESIGN OF FIGHTERS .

Guidelines for Obtaining and Maintaining Adequate Fatigue Performance of Tactical Aircraft

Compiled by

11 Jan 78

The Working Group on Fatigue Life Prediction of
The Structures and Materials Panel of AGARD

12 140p.

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– Exchanging of scientific and technical information;

– Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;

– Improving the co-operation among member nations in aerospace research and development;

– Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;

– Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field;

– Providing assistance to member nations for the purpose of increasing their scientific and technical potential;

– Recommending effective ways for the member nations to use their research and development capabilities for the common benefit of the NATO community.

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PREFACE

In the past fatigue has not been a particularly important aspect in the design of fighter aircraft structures. Being required to sustain high manoeuvring loads and being of relatively short life expectancy, these structures were generally designed primarily by static strength considerations. More recently, the greater complexity and cost of new weapon systems, together with the general economic pressures to control defence expenditure in the NATO countries, has required that the operational lives of fighters be increased bringing in its train an increased probability of fatigue defects and failures. Apart from the safety aspects, these fatigue defects can cause a reduction in the total state of readiness of the NATO air forces and, with the more sophisticated materials and structural forms now being employed, can result in expensive repair bills.

Recognising this situation, the Structures and Materials Panel of AGARD have explored this area of concern and have proposed, wherever possible, generally accepted procedures for its solution. The AGARDograph on Fatigue Design of Fighters provides such guidelines for obtaining and monitoring adequate fatigue performance of fighter aircraft. It is commended to the structural designers, the procurement agencies, the safety agencies and the air forces in the expectation that the wider adoption of the procedures described will result in overall improvements in the cost-effectiveness of new fighter aircraft structures.

N.F.HARPUR
Chairman, Structures and
Materials Panel

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CHAPTER 1

INTRODUCTION

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1.1 BACKGROUND

Structural fatigue has been recognized as a problem in aviation since the early fifties, when a number of serious accidents occurred that were caused by fatigue. Originally, fatigue was considered as the province of the designer of commercial transport aircraft. However, when in the late 1950s different airforces were confronted with fatigue problems, it became apparent that fatigue should also be considered in the design of tactical aircraft.

In the last decades, a tremendous amount of research has been done and service experience has been obtained in the field of aeronautical fatigue.

In accordance with its Mission, the Structures and Materials Panel of AGARD has been contributing to the dissemination and exchange of knowledge on fatigue, e.g. by preparing a "Manual on Fatigue", organising Symposia and Specialist Meetings, and by sponsoring Lecture Series[1,2,3,4].

It cannot be said that, due to these efforts, fatigue has been ruled out as a structural problem: every day aviation pays its toll to fatigue by means of unscheduled repairs, modifications and early replacements. However, it may be stated that at present it is within our power to design, build and operate aircraft that are reasonably free from fatigue throughout their service life.

Reaching this goal is not a matter of applying a set of simple rules (e.g. "avoid stress raisers"): adequate fatigue performance can only be obtained by a careful fatigue-conscious approach throughout the design process. Fatigue considerations will have to play a role in each phase of the aircraft development, starting in the initial design phase and continuing in the aircraft operational life. The present "guideline" is intended to establish recommended procedures for such fatigue conscious design, with special reference to tactical aircraft.

1.2 SCOPE

The design process of an aircraft contains a number of successive steps in which fatigue aspects should be considered. The basic purpose of these "guidelines" is to outline these steps and to recommend for each step those procedures, based on current knowledge, most likely to provide adequate fatigue performance. It should be stressed that these recommendations are of an advisory nature and are by no means intended as a "requirement". It is hoped that these guidelines will set a generally accepted methodology for obtaining adequate fatigue integrity. Apart from providing information to the actual structural designer these guidelines may provide a better understanding to those not directly involved in the structural design but e.g. in procurement, operations or maintenance, for the way in which fatigue performance is obtained, what factors must be considered and what are its limitations.

1.3 LAY-OUT

These "Guidelines" are divided in seven Sections. In the following a short description of the different sections is given.

Section 2: General Survey

This section gives a general description of the different phases in the fatigue design process and indicates the specific aspects that should be considered in each phase. Items of special importance that will be treated in more detail in other sections will be indicated.

Section 3: The Development of Fatigue/Crack Growth Analysis Loading Spectra

This section describes methods to develop realistic aircraft loading spectra, considering the aircraft mission usage, mission profiles and representative mission segment spectra. Methods are illustrated by means of worked-out examples.

Section 4: Calculation Methods for Fatigue Life and Crack Propagation

This section gives a survey of existing methods for the calculation of fatigue life and crack growth. Attention is paid to the degree of accuracy that can be obtained. Possible improvements are discussed.

Section 5: Tests on Details and Components

Throughout the design of an aircraft system, tests are required to assess the relative fatigue properties of various structural details and components. This section gives a survey of the techniques to be used in such tests.

Section 6: Current Standards of Fatigue Test on Strike Aircraft

In the final assessment of the fatigue properties of a new aircraft, the Major Fatigue Test plays a central role. This section presents an outline of current standards of the major fatigue test for Tactical Aircraft.

Section 7: Fatigue Load Monitoring

The service life established for a new aircraft refers to an assumed "design load experience". For various reasons, the actual loads experienced in operational use can deviate appreciably from the orginal load assumptions. For re-assessing the actual operational service life, monitoring of operational load experience is indispensable. This section gives a description of different load monitoring techniques. Advantages and disadvantages of various methods are discussed. Finally, methods for interpreting measured load experience in terms of fatigue life are analysed.

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CHAPTER 2

GENERAL SURVEY

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2. GENERAL SURVEY

In the structural design process of a new aircraft, a number of successive stages can be defined. For the purpose of the present survey, four successive phases will be distinguished, viz.:

(a) *The definition phase*

In this phase, the basic structural lay-out, including the type of structure and materials to be used is determined.

(b) *The development phase*

In this phase the detail design of the structure takes place.

(c) *The prototype and production phase*

This phase is characterised by the assessment of the performance of the new aircraft and its certification.

(d) *The service phase*

The aircraft has entered service and is being subjected to its actual operational environment.

In the following, the impact of fatigue on the structural design and the considerations with regard to the fatigue phenomenon in each of these successive phases will be discussed.

2.1 THE DEFINITION PHASE

In this phase, that may also be described as early development phase, the basis is laid for the potential fatigue performance of the aircraft: when the structural type is poorly chosen, when materials with unfavourable fatigue properties are used and stress levels are too high, even the best detail design can only lead to mediocre fatigue properties. Consequently, careful consideration of fatigue aspects in this design phase is of eminent importance.

Figure 2.1 gives a schematic presentation of the fatigue-related aspects that are involved in the early design process. In the following, these aspects will be further discussed.

Design philosophy

Modern fighter aircraft structures have a limited fatigue life. The provision of adequate fatigue strength is in the first place a matter of economy: insufficient fatigue performance will necessitate costly repairs and modifications and/or untimely aircraft-replacement. In the second place, fatigue has an obvious safety aspect: even minute fatigue cracks may reduce the ultimate strength of the aircraft to an extent as to endanger the airworthiness of the aircraft. Experience has shown that even in very well proven designs occasional fatigue cracks can occur, initiated by e.g. material flaws, tool marks, etc. The recognition of this fact has led to the concept of so-called damage tolerance: a structure is said to be damage tolerant when in a damaged state it can still sustain acceptably high loads. Damage tolerance may be obtained in different ways, such as:

(a) Fail-safety:
 multiple load-path structures,
 crack-arrest structures,

(b) Slow-crack growth structures.

Essential for obtaining damage tolerance by means of fail safety is that cracks can be found sufficiently early. Hence, inspectability is an important design consideration. The provision of damage tolerance has become more and more an integral part of the general structural design philosophy. Recently, the USAF introduced a stringent Requirement on Damage Tolerance[1] that must be met in all their future aircraft. There is no doubt that the concept of damage tolerance has a major influence on the basic lay-out of the aircraft structure, the type of structure and the choice of materials to be used.

Material properties

In selecting structural materials a large number of aspects other than fatigue must be considered such as

- static strength,
- producability,
- cost,
- corrosive resistance,
- sensitivity to stress-corrosion.

There is no doubt, however, that increasing attention is being paid to fatigue and fatigue-related properties such as fatigue resistance, crack propagation and residual strength. The demand for fatigue resistant materials is reflected by current efforts of Alloy Designers to specifically design alloys for good fatigue and fracture resistance[2].

Usage definition

Usually the basic aircraft design specification contains a general service life requirement, which may range from 3000 to 6000 flying hours for tactical aircraft. One of the first tasks in the definition phase must be the specification of the utilization of the aircraft during its service life. This specification should include:

(a) The types of mission in which the aircraft will be operated.
(b) The flight profiles pertaining to these missions.
(c) Configurations and stores to be carried.
(d) Numbers of landings.

If the aircraft is to be used in a number of different roles (different mission-mixtures) it should be specified whether the required life refers to an average mixture of roles or to the relatively most severe role.

It will be clear that the estimation of the usage of a non-existing aircraft that will not enter service within at least four years is a very difficult task. It is, however, a very important one, as the whole fatigue design will be based on the associated utilization. It should be stressed that in this task a close cooperation between the procuring agency, the future operator and the aircraft designer is indispensable.

Load spectra estimation

As a next step, load spectra will be estimated pertaining to the specified aircraft utilization. These load spectra will largely be based on data collected from previous aircraft. Specific performance characteristics (e.g. higher g-load capability) should be taken into account. Chapter 3 of this Handbook contains more detailed information about spectrum estimation techniques, including pertinent spectrum data and worked-out examples.

Development of stress spectra

When a preliminary lay-out of the structure is available, first stress calculations can be made and load spectra can be converted into stress spectra for each relevant item of the aircraft structure. The derivation of stress spectra is also discussed in Chapter 3.

Preliminary fatigue analysis

When stress spectra are available, the structure becomes amenable to a first fatigue analysis. The primary purposes of this analysis are:

(a) Identification of possible fatigue critical areas in the structure.
(b) Establishment of fatigue allowables.

For the analysis, the following "tools" are available:

(a) General fatigue data (S-N data or program test results on notched specimens and/or structural details).
(b) Damage calculation techniques.

Sometimes, e.g. when new materials are envisaged, relevant fatigue data are not available. In that case, fatigue tests are done on relatively simple specimens, to generate the desired information. Such tests are discussed in detail in Chapter 5.

Damage calculations can be done with different degrees of complexity. In Chapter 4, different damage calculation techniques are discussed in full detail.

It will be clear that in the early development stage, when structural details have not yet been fixed, it is usually not justified to apply highly sophisticated analysis methods. Usually, relatively simple "Miner"-type calculations are made, maintaining large safety factors. Those areas that show a marginal life in these calculations are labelled for a more thorough analysis and possibly for component testing in a later design stage.

To summarize this sub-section, it is concluded that indeed a number of important fatigue aspects are involved in the early design phase. It is of primary importance that the design requirements with regard to fatigue are well-defined. This includes a consistent specification of the aircraft loading environment. Here, a good cooperation between designer, procuring agency and future operator is indispensable.

2.2 THE DEVELOPMENT PHASE

The involvement of fatigue in this design phase will be discussed under the following three headings:

(a) The fatigue design of structural details.
(b) Fatigue life calculation.
(c) The testing of details and components.

The fatigue design of structural details

Many fatigue failures that have occurred in practice can be attributed to "poor detail design"; consequently the importance of "fatigue consciousness" of each designer in detail design is generally recognised. Apart from the basic rules like avoidance of stress raisers and excentricities, the designer must be aware of a large number of factors that influence the fatigue properties, such as:

- metallurgical factors
  As an example, the effect of grain directionality may be mentioned.
- manufacturing techniques
  Of great importance in this respect is surface roughness. As a possible means to improve fatigue performance, the introduction of residual compressive stresses by means of e.g. shot peening should be mentioned.

Generally it can be stated that the most critical areas in the structure are the joints between structural parts. Consequently, major attention is to be paid to the design of joints. Apart from the major variables:

- Type of joint (riveted, bolted, bonded, welded, etc.)
- Type of fasteners and material

there are a large number of other factors that influence the fatigue properties, such as:

- use of tapered bolts,
- interference fit,
- coining and rolling,
- fabrication methods.

A fairly large amount of fatigue data on joints is available in the literature, but in view of the numerous variables involved these data have a limited value for obtaining accurate life predictions. The designer will rely upon his own experience, obtained from previous aircraft with a similar structure. In case of new designs, development tests will be necessary, as will be discussed later.

Fatigue life calculation

When the structural design advances and details are fixed, the structure becomes more and more amenable to a rigorous fatigue analysis. For a detailed description of existing calculation techniques, reference is made to Chapter 4. Essentially, all calculation techniques used are based on the well known Miner Rule; the shortcomings of this Rule are sufficiently known. Fortunately results obtained with recent techniques, which calculate the "real" stress/strain history at the notch root are very promising. These methods can be fully computerised and are being used on a routine basis by aircraft designers[3,4]. The following remarks should be made.

(a) To apply advanced calculation techniques that take account of sequence effects, not only the spectrum of loads but also the sequence in which these loads occur must be defined.

(b) For a proper fatigue analysis, an accurate knowledge of stress distributions is needed. The development of finite-element techniques, in conjunction with large and fast computer systems have greatly increased the possibilities of stress-calculation. However, in complex structural parts, even modern stress analysis methods tend to fail, in predicting detail stress distributions. Here, experimental investigation, e.g. by means of strain-gages or photo stress-techniques can be very useful.

The testing of details and components

In spite of recent advances in fatigue analysis techniques, it is recognized by every designer that for a proper fatigue design detail and component tests are indispensable. This applies particularly for joints, where the many variables involved greatly reduce the amenability to analytical fatigue life prediction. The amount of testing needed depends of course on the "orthodoxy" of the structure. When advanced designs are introduced, including e.g. new methods of joining and new materials, large test programs may be necessary. Figure 2.2 illustrates the type of components that were tested in the development of the F-15. In this case a total number of more than 300 component tests were done.

In the past, component tests were usually done under either a constant amplitude loading or simple block-type programme loading. Experience has shown, however, that the results of such tests may lead to erroneous conclusions, even in a qualitative sense. It is therefore recommended to apply realistic load sequences, usually indicated as "flight simulation loading", also in development tests on components.

A further discussion of techniques for detail and component tests may be found in Chapter 5.

2.3 THE PROTOTYPE AND PRODUCTION PHASE

At this stage, the actual structural design is essentially completed. A number of prototypes are available and the performance of the aircraft can be evaluated. With regard to fatigue, this evaluation must include the following:

(a) Flight load measutrements

Flight load measurements will be made to check the calculated structural loads and load distributions in various specified loading conditions. The accurate knowledge of load distributions is an essential prerequisite for carrying out a representative fatigue test. In this context it may be mentioned that special attention should be paid to empennage and landing gear loads, since in these cases assumed load spectra may turn out to be exposed to substantial errors.

(b) Full-scale fatigue test

The full scale fatigue test plays an essential role in the assessment of the actual fatigue performance that has been obtained. Because of the large influence of manufacturing techniques on fatigue properties it is very important that the test article should be as nearly as possible a production item containing production design features and methods of manufacture and assembly. This implies that usually the full-scale fatigue test is done in a relatively late stage, when the aircraft is in full production.

Chapter 6 is devoted to the full-scale fatigue test. The objectives of the test are formulated and a concise description of recommended procedures to achieve these objectives is given.

It should be stressed that the results of the fatigue test should not only certify that the fatigue properties of the aircraft comply with the design requirements but also serve as a basis for the fatigue life reassessment throughout the aircraft operational service life. Experience has shown that for various reasons, such as changes in mission profiles or extended service lives, the fatigue life that is actually desired exceeds the original fatigue life requirements. Several cases are known where, in the framework of a "life extension program", the fatigue test that had been terminated after covering the factored design service life had to be re-activated or due to loss of the original test specimen had to be repeated. It is threfore strongly recommended to continue the fatigue test after covering the factored service life in order to really evaluate the ultimate fatigue performance of the aircraft.

2.4 THE SERVICE PHASE

During the actual operational life of an aircraft type a continuous reassessment of its fatigue performance takes place on the basis of service experience. Service findings may result in:

(i) Technical measures, e.g.: changed maintenance and inspection schedules, modifications, preventive repairs.

(ii) Operational measures, e.g.: changes in manoeuvring patterns, rotations of aircraft over different duties, decisions with regard to time of replacement.

Moreover, service experience will be incorporated in the design of new aircraft versions and in future designs.

In service experience, two different aspects should be distinguished, viz.:

(a) The load experience encountered in actual usage.
(b) The failure-experience in service.

These two aspects will be briefly discussed.

(a) The service life determined in the full-scale fatigue test is related to the load spectra applied in the test. For various reasons, the actual usage and load experience may differ considerably from the original design assumptions and, moreover, may change drastically with time. Monitoring of service load experience is indispensable in order to reassess the service life determined in the fatigue test, and to adapt maintenance and inspection schedules. A wide variety of fatigue load monitoring systems are currently being used, differing in monitoring technique, complexity and sophistication. Factors that determine the choice of monitoring systems for a specific aircraft system are e.g.:

- The structural configuration (e.g. variable sweep)
- The degree to which fatigue is expected to be potentially critical.
- The components that may be critical.
- The type of aircraft usage and the expected amount of variation in load experience.
- Operational and maintenance procedures.
- Fleet size.

In Chapter 7, principles of load monitoring and techniques to be used are discussed in more detail.

The object of service load monitoring is to relate service load spectra in terms of fatigue life to the fatigue life observed in the full scale test. When service spectra do not differ "too much" from test load spectra, such comparisons can be made on an analytical basis, using assumed S-N curves and Miner-type calculations. When test loads and service experience are widely different, additional comparative fatigue testing on representative components may be necessary for a reliable life-reassessment.

(b) Experience has shown that in actual service fatigue cracks do occur that were either not predicted at all in the full scale test or predicted to occur much later. Part of these unexpected cracks are due to more or less incidental deficiencies in the aircraft not present in the test specimen such as mounting stresses or initial damages like internal flaws or tool marks. Carefully controlled manufacturing processes are essential in preventing such unexpected failures as far as possible. In other cases, however, the fact that service cracks were not predicted in the fatigue test must be attributed to the inability to fully represent the actual service environment in a fatigue test. Examples are:

- Effect of a corrosive environment on fatigue crack initiation.
- Effects of e.g. buffeting loads on control surfaces.

It is certainly true that "the best fatigue test in the world is a fleet of aircraft in service", and it must be made sure that optimal use is made of service experience. The analysis of service failures should cover the following aspects:

- What loading mechanism did cause the failure.
- What has been the loading environment to which the cracked aircraft was subjected. (Based on load monitoring-information.)
- Is the observed crack to be considered as an isolated case, or is its occurrence to be expected in more aircraft.
- Should the fatigue test results be readjusted in the light of service data.

To end this section, it should be noted that for the interpretation of service experience and the definition of consequent remedial actions specialised knowledge on fatigue and on the specified aircraft structural design is needed. In this task, the aircraft designer should play a major role. The gathering of operational experience, both with regard to load monitoring and service failure experience is the responsibility of the aircraft operator. It is therefore vitally important that good cooperation and communication between aircraft designer and operator are maintained throughout the aircraft service life.

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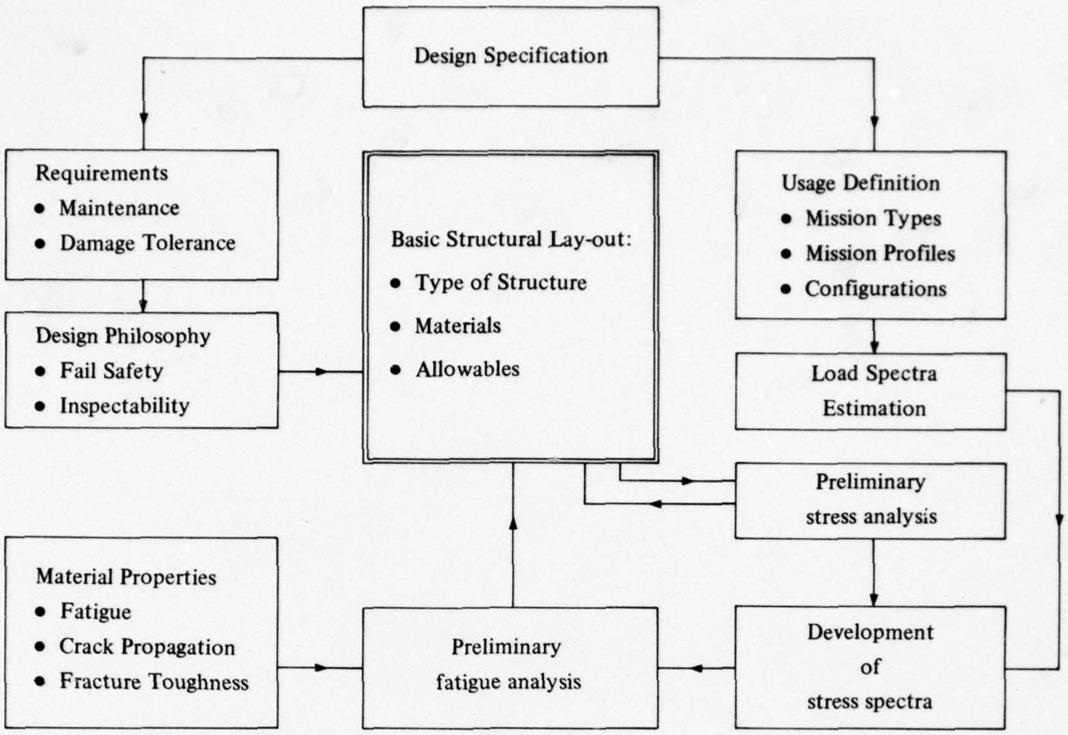

Fig.2.1 Fatigue in the definition phase

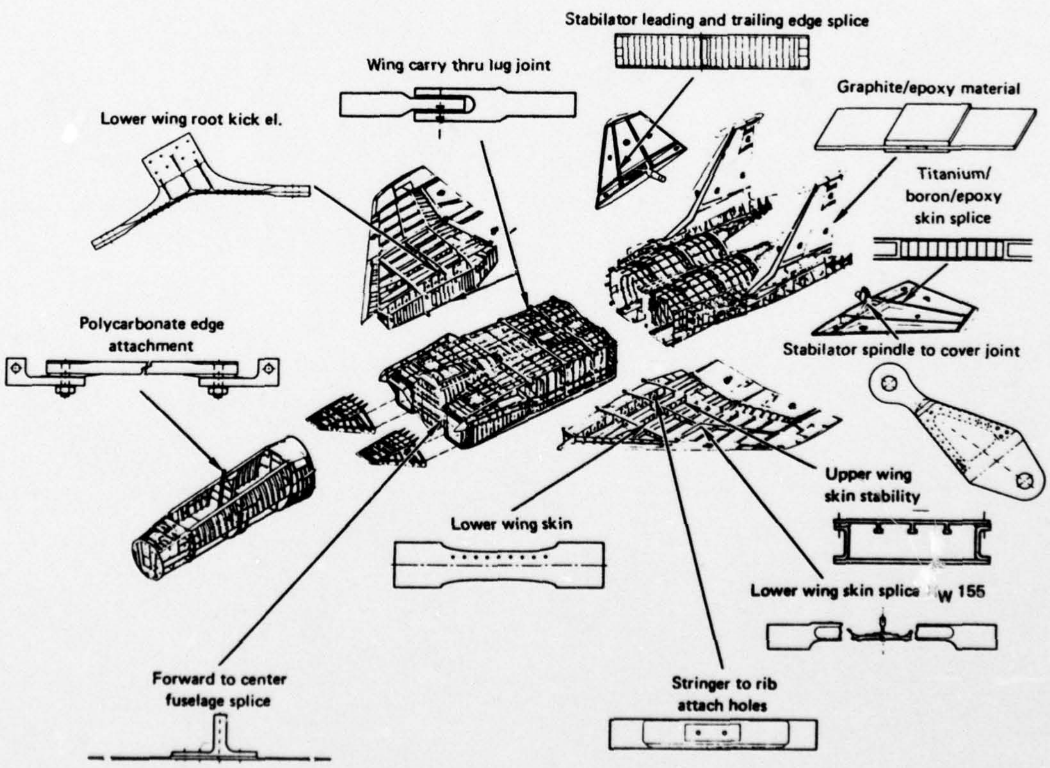

Fig.2.2 F-15 component tests (Ref.4)

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CHAPTER 3

THE DEVELOPMENT OF FATIGUE/CRACK GROWTH ANALYSIS LOADING SPECTRA

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3.1 INTRODUCTION

This chapter describes the processes involved in developing a realistic loading spectrum. The purpose of developing a realistic loading spectrum is to define a stress-time history that is representative of those stresses encountered by a component during actual usage. The spectrum is normally used as input to a fatigue or crack growth (fracture) analysis or test to predict the fatigue or crack growth life of a component. This chapter will confine itself to discussion of developing load and stress spectrums for aircraft structural (airframe) components, particularly fighter or strike aircraft.

The development of a realistic loading spectrum for an airframe component is a process that encompasses many disciplines, among them, loads, stress, fatigue, and crack growth analysis. It is a process that can vary in detail from simple to extremely complex. The realism of the spectrum will be determined by the accuracy of the input from the different disciplines and the degree of complexity that the analyst is willing, or able, to go to. The more realistic the spectrum is, the more realistic the results of the analysis or test that the spectrum is used for will be.

There are many external factors to be considered in the spectrum development process. Among these are time and money considerations, available data, degree of accuracy required, etc. These facotrs may require the use of simpler, less time consuming techniques than one would prefer to use for realism's sake.

This chapter discusses the processes in the order they occur. The steps include:

A. Mission profile definition,
B. Loading environment,
C. Loading conditions,
D. Structural loads analysis,
E. Stress analysis, and
F. Stress sequencing.

Whenever pertinent, there is some discussion of the simpler techniques available.

Also included is a detailed example representative of what has actually been done to develop a spectrum on a fighter/strike aircraft. The example is not meant to be taken as recommendation of an exact method. Due to the multiplicity of factors involved, a recommended method would be inappropriate.

There has been a wealth of literature written concerning spectrum development. The references herein are not intended, in any way, to be a comprehensive list. They can however provide a *starting point* for researching whatever aspects of the topic one desires. References 28–61 are not mentioned in the text but may provide information and other sources.

3.2 MISSION PROFILES

Mission profiles are the first items of information that must be obtained in the development of aircraft fatigue/fracture load spectra. This section will, (a) define mission profiles and specify the information that must be included and (b) discuss the sources of this information.

3.2.1 Definition

A mission profile is a sequential listing of mission segments or events which combine to produce a complete mission. This listing may be in various forms such as graphical (Figure 3.2.1.1) or tabular (Table 3.2.1.1). Regardless of the form, there are certain variables which each mission profile must include:

A. Type of mission
B. Mission utilization (in percent of total aircraft life and in total flight hours)
C. Configuration
D. Aispeed
E. Altitude
F. Gross Weight
G. Time

(D–G: for these variables the range within each segment must be specified)

H. Number of landings (full stop or touch and go)

Other variables that may be required if pertinent are:

I. Store drops
J. Other significant weight shifts
K. Mission radius
L. Types of runways used if other than standard
M. Any additional information that would be useful or helpful in defining representative missions.

3.2.2 Source of Information

The user is the primary source of information. This information may be in terms of the actual values mentioned above or may be expressed as normalized parameters such as power setting, optimum climb, optimum cruise, percent of maximum fuel or payload, etc. (Figures 3.4.1.1 through 3.4.1.5). By knowing the performance capabilities of the aircraft, the real values may be calculated from the normalized values. In most cases the information received from the user does not contain all of the required data. When this happens, the designer or engineer must use other sources. If the aircraft being evaluated is in the design and/or development stages, the only additional source of data is from similar aircraft flying similar missions. However, the data from other aircraft may have to be modified to reflect differences in aircraft capabilities. If the aircraft being analyzed is flying operationally, the best sources of additional data are aircraft flight logs, interviews with pilots, and measured data from flight recorders.

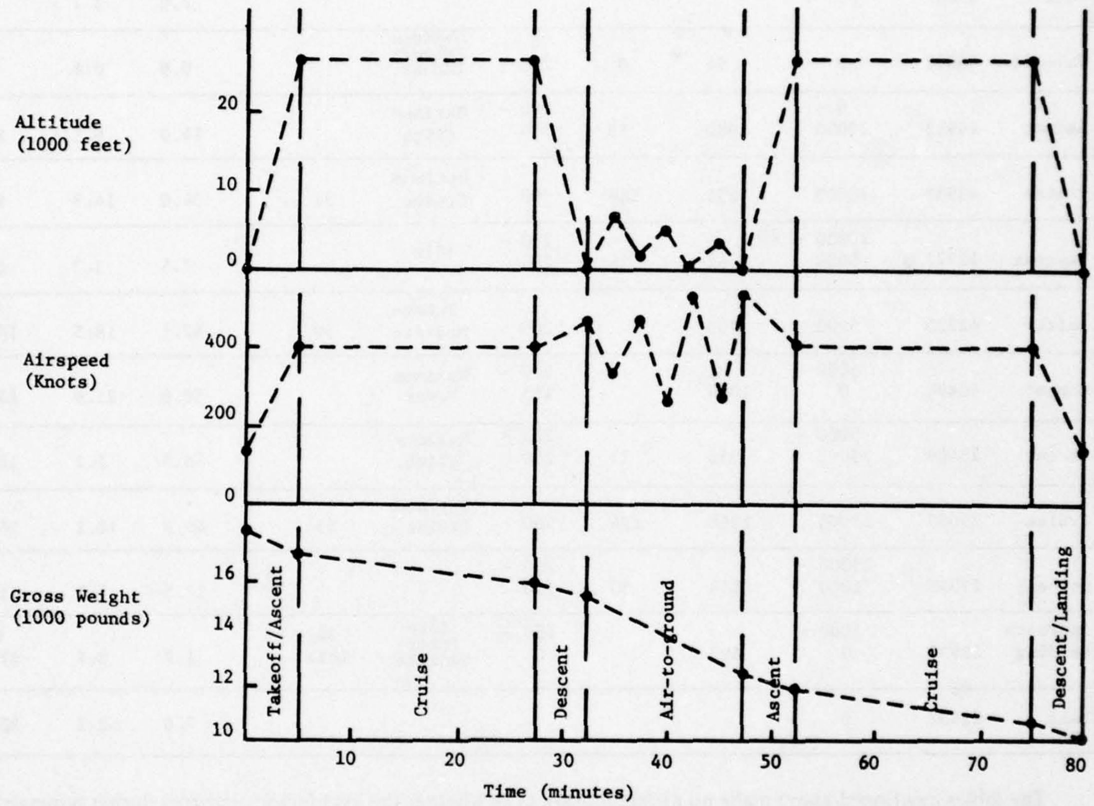


Fig.3.2.1.1 Graphical mission profile

3.3 STRUCTURAL ENVIRONMENT FOR MISSION SEGMENTS

For each mission segment, a load producing structural environment must be determined. This environment is usually expressed in terms of maneuver load factor spectra, gust spectra, landing sink rates, taxi spectra, etc. Methods of determining these spectra for flight loads and for ground loads are presented.

3.3.1 Flight Spectra

3.3.1.1 Maneuver

In Reference 1, Tables III through VIII contain load factor spectra for various mission segments and most types of aircraft. For example and information, Table III of Reference 1 is reproduced in Table 3.3.1.1. The spectra in these tables were obtained from load factor data measured during actual missions. It should be noted that in most cases the spectra represent an envelope of the measured data and, therefore, may be high when applied to a specific present day aircraft. When applied to future aircraft with improved load factor capabilities, the spectra may have to be modified upward to account for these increased capabilities. For aircraft in the design/development stage and for aircraft that have no recorded flight data, these tables have been and are still useful in developing initial load factor spectra. However, it is recommended that reliable and applicable recorded data be used to augment these spectra when possible to help produce a more representative usage.

TABLE 3.2.1.1

Tabular Mission Profile

MISSION - COMBAT EMPLOYMENT - CLOSE AIR SUPPORT

20% of Aircraft Life - 1200 hours

1000 lb Type A stores, 1000 rounds ammunition

	1	2	3	4	5	6	7	8	9	10
Segment	Initial Gross Weight (pounds)	Pressure Altitude (feet)	Fuel Used (pounds)	Distance Nautical (miles)	Speed (KEAS)	Power Setting	% of Maximum Continuous Thrust	Time (minutes)	% Total Time	Total Time (minutes)
Taxi	45007	0						7.0	3.1	7.0
Takeoff	45007	0	94	0	0 - 200	Maximum Takeoff Thrust		0.9	0.4	7.9
Ascent	44913	0 - 20000	980	53	200 - 250	Maximum Climb		14.0	6.1	21.9
Cruise	43933	20000	1656	169	300	Optimum Cruise	91	34.0	14.9	55.9
Descent	42277	20000 - 5000	62	28	250 - 200	Idle		7.5	3.3	63.4
Loiter	42215	5000	1735		200	Inter-mediate	50	42.1	18.5	105.5
Combat	40484	5000 - 0	4046		200 - 450	Maximum Power		50.0	21.9	155.5
Ascent	25409	5000 - 25000	350	23	225 - 260	Maximum Climb		5.0	2.2	160.5
Cruise	25059	25000	1066	174	260	Optimum Cruise	55	41.3	18.1	201.8
Descent	23993	25000 - 1000	158	50	200 - 150			17.5	7.7	219.3
Approach Landing	23835	1000 - 0	503		150 - 0	Idle Inter-mediate	34 - idle	1.7	0.7	221.0
Taxi	23332	0						7.0	3.1	228.0

The tables mentioned above make no differentiation as to whether the load factors occurred during symmetrical, unsymmetrical, abrupt, or smooth maneuvers. In Reference 1, paragraph 3.3. a states, "The spectra shall be proportioned between symmetrical, unsymmetrical, abrupt, and smooth maneuvers." However, there are no guidelines given as to how this should be done. In recent developments of fatigue/fracture loading spectra for various aircraft, the following methods have been used: (a) utilizing the data in Reference 2, if applicable (b) interview pilots to get reasonable approximations as to how they fly the aircraft (documented in Reference 3), and (c) fly an instrumented aircraft on simulated missions and record the types of maneuvers. No particular method or guidelines are recommended in this manual however, different types of maneuvers should be considered to insure not overlooking any significant loadings. Past experience has shown that considering these different types of maneuvers has little effect on wing loads spectra, but may have a significant effect on empennage loads spectra.

3.3.1.2 Gust

As stated in paragraph 3.4 of Reference 1, gust spectra may be developed from the continuous turbulence model specified in Reference 4. The gust loads spectrum should include the effects of combined vertical and lateral gusts. For terrain-following and contour-flying mission segments, the loads resulting from gust plus maneuver must be considered. There are no guidelines in Reference 1 showing how this should be done. An acceptable method would be to: (a) select a rational percentage of maneuvers to be combined with gusts, (b) determine the gust load factor spectra, (c) use two separate random number selector computer programs to randomly select a gust load factor and a maneuver load factor, (d) add the selected load factors and, (e) do this still all allocated maneuvers are used up. The resulting load factors will form the combined gust plus maneuver load factor spectrum. The same process may be done using loads instead of load factors.

It should be noted that past experience has shown that gust loads have little or no effect on the fatigue/fracture life of high load factor type aircraft.

3.3.1.3 Control Surface Usage

Control surfaces are defined as ailerons, rudder, flaps, slats, speed brake, etc. Reference 1 does not contain any spectra for control surfaces. This is primarily because control surface usage is different for every aircraft. Control surface spectra are determined by establishing the deflections that correspond with and produce the maneuver spectra discussed in paragraph 3.3.1.1. Control surface deflections used during takeoff and landing must also be included.

3.3.1.4 Fuselage Pressurization

The number of pressurization cycles used for design shall be determined by, and commensurate with, the mission profiles and design life requirements. Regulator valve nominal setting plus tolerance shall define maximum pressure. The resulting loads shall account for the interaction of appropriate flight loads (gust and maneuver) with the pressure loading[1].

3.3.1.5 Store Ejections

The number of cycles and types of store ejections may be obtained from the information contained in the mission profiles. However, since some aircraft are capable of carrying a large variety of stores and store combinations, it is sometimes advisable to add extra cycles representative of store ejection loads. This will account for possible store combinations other than those listed in the mission profiles.

3.3.2 Ground Spectra

Due to the lack of better data, Reference 1 will be used almost exclusively as the source for ground loads spectra. It is recognized that the spectra are stated in very general terms and when applied to a specific aircraft may not be an accurate representation of the actual usage. For this reason, it is recommended that reliable and applicable recorded data be used to augment these spectra when possible to help produce a more representative usage.

3.3.2.1 Taxi

The taxi ground loads shall include the vertical gear inputs resulting from taxiing on prepared runways as specified in Table 3.3.2.1. In lieu of the vertical-load cycles shown in Table 3.3.2.1, the airplane may be designed for the effect of takeoff, taxiing, and rollout on deterministic runway profiles having power-spectral-density characteristics as specified in Figures 3.3.2.1, 3.3.2.2, and 3.3.2.3 for runway requirements specified in the mission profiles. For this option, the analysis shall include all significant rigid body, gear dynamics, and elastic modes. Aerodynamic and propulsion forces shall also be included. The number of taxi operations for each of the levels of runway roughness and airfield types plus taxi times and speeds shall be as specified in the mission profiles[1].

3.3.2.2 Braking and Turning

Hard braking with maximum braking effects shall occur twice per full-stop landing and medium braking with half-maximum-braking effects shall occur an additional five times per full-stop landing, During a given mission, each full-stop landing that occurs shall be included. The effects of antiskid devices shall also be included. Pivoting, with half-limit torque load, shall occur every 10 landings. Turning with a side-load factor 0.4 at the airplane center of gravity, reacted by the landing gears alternately inboard and outboard, shall occur five times per landing[1].

3.3.2.3 Landing

The landing-loads spectra shall include consideration of the anticipated usage of the airplane, including such variables as sinking speed, forward speed, attitude, wing stores, and wing fuel distribution. The distribution of sinking speeds specified in Table 3.3.2.2 shall apply for the distributions of the number of landings indicated by the mission profiles. The table lists the relative frequency of occurrence of sinking speed per thousand lamdings for each type landing[1].

3.3.2.4 Miscellaneous

There are other ground operations which, for specific aircraft, may produce loads significant enough to be considered for loading spectra. These operations may include towing, engine run-up, etc. To develop these data, aircraft operations would have to be monitored and recorded if significant (see example in Section 3.4).

3.4 DETERMINATION OF LOADING CONDITIONS

With the mission profiles accurately defined and the structural environment specified, it is necessary to determine the loading conditions to be used in generating the fatigue/fracture load spectra. Determining the loading conditions is

TABLE 3.3.1.1

Maneuver-Load-Factor Spectra for Fighter and Fighter-Bomber Type Aircraft Cumulative Occurrences per 1000 Flight Hours by Mission Segment

N_z	Ascent	Cruise	Descent	Loiter	Air-Grnd	Spec Wpn	Air-Air
				Positive			
2.0	5000	10,000	20,000	15,000	175,000	70,000	300,000
3.0	90	2,500	5,500	2,200	100,000	25,000	150,000
4.0	1	400	500	250	40,000	7,500	50,000
5.0		1	1	25	10,000	2,000	13,000
6.0				1	1,500	250	2,500
7.0					200	15	900
8.0					15	1	180
9.0					1		60
10.0							15
				Negative			
0.5					10,000		44,000
0					350		4,000
-0.5					30		1,200
-1.0					7		350
-1.5					3		60
-2.0					1		8
-2.5							1

TABLE 3.3.2.1

Number of Cycles per Thousand Runway Landings that Load Factor N_z is Experienced at the Airplane CG

N_z	Number of cycles
1 ± 0.05	300,000
1 ± 0.15	165,000
1 ± 0.25	27,000
1 ± 0.35	2,000
1 ± 0.45	90
1 ± 0.55	4
1 ± 0.65	0.15
1 ± 0.75	0.005

TABLE 3.3.2.2

Occurrences of Sinking Speed/1000 Landings

Sinking Speed FPS	Trainer	All Other Classes
1	130	180
2	190	290
3	220	260
4	190	155
5	125	78
6	77	26
7	37	8
8	17	1.5
9	8	1.0
10	2.6	0.5
11	1.4	
12	0.9	
13	0.6	

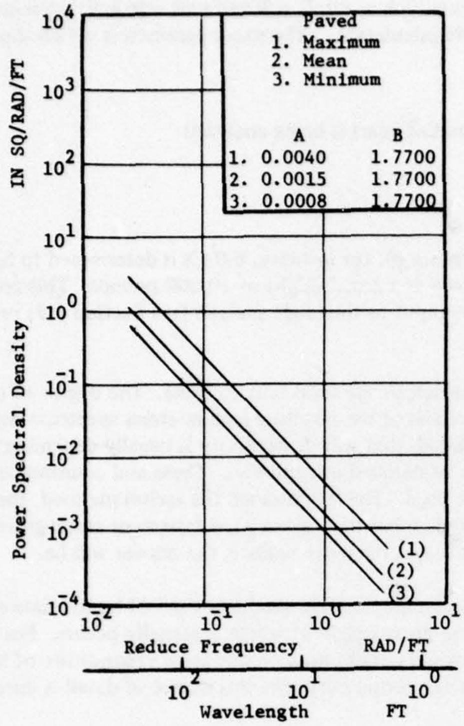


Fig.3.3.2.1 Power spectral density levels for paved airfields

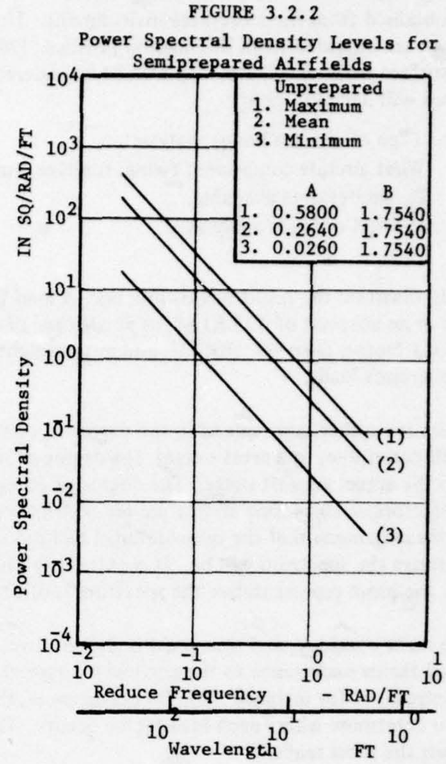


Fig.3.3.2.2 Power spectral density levels for semiprepared airfields

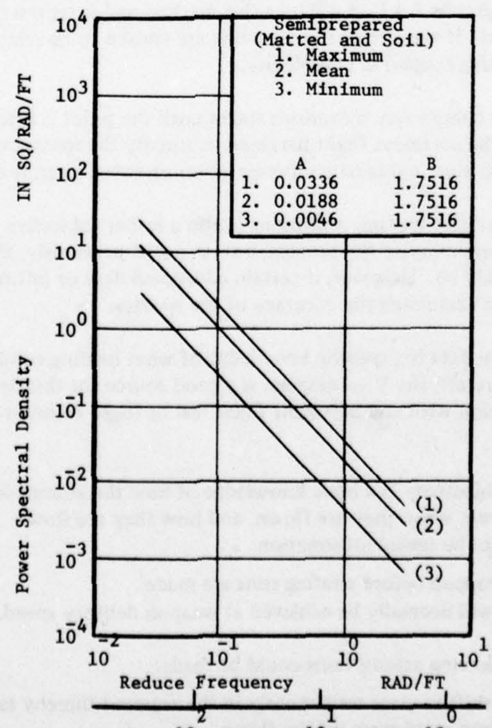


Fig.3.3.2.3 Power spectral density levels for unprepared airfields

that process by which various flight and ground maneuvers are coupled with certain parameters that are necessary for calculating loads. These maneuvers are defined, for the most part, in terms of a flight or ground load factor occurrence and are obtained from the structural environment. The primary parameters are normally airspeed, altitude, and gross weight and are obtained from the mission profiles. Other parameters such as pitch, roll and yaw rate and accelerations, control surface deflections, etc, may also be considered in the loads calculations. The exact parameters which should be considered will depend upon:

A. Type of aircraft being analyzed.
B. What aircraft component (wing, fuselage, empennage, landing gear) is being analyzed.
C. Parameter data available.
D. Method of loads analysis.
E. Etc.

To simply illustrate the result of this process: A load factor occurrence of, for instance, 6.0 g's is determined to have occurred at an airspeed of 475 KIAS, at an altitude of 1000 feet, and at a gross weight of 10,000 pounds. This combination of load factor, airspeed, altitude, and gross weight can then be input to the loads analysis (see Section 3.5) to calculate aircraft loads.

There is a rather large spread in the degree of complexity at which this process may be done. The degree of complexity will determine, to a great extent, the degree of representativeness of the resulting load or stress spectrum with regard to the actual aircraft usage. The degree of complexity, or detail, that will be necessary is usually dependent on external factors, such as time and/or money restrictions and a lack of detailed information. These and countless other restrictions may mean that the more detailed techniques cannot be used. The less detailed the technique used, the less representative the spectrum will be. It is extremely important to remember that, given good fatigue or crack growth analyses, the more representative the spectrum is of actual aircraft usage, the more realistic the answer will be.

The most complex, and thus most representative, method of selecting loading conditions would be to relate every single load factor occurrence to the particular airspeed, altitude, and gross weight at which it actually occurs. For a 4000 hour spectrum, of for instance 350,000 occurrences, this process would entail unreasonably large expenditure of time and money to determine where each load factor occurs. Thus, for all engineering purposes, this degree of detail is unreasonable, albeit the most realistic

The next level of detail entails relating each occurrence to a *range* of airspeed, altitudes, and gross weights. The ranges of the flight parameters are representative of those within each mission segment. This method reduces the task to a workable level and has, in fact, been used to generate realistic spectra[5,6].

By *averaging* the flight parameters within a segment, the process is reduced yet another level of complexity. The detailed example presented in paragraphs 3.4.1–3.4.9 uses this method and is representative of what has been done to develop spectra for analysis and test. If the size of the segments are chosen to be relatively small, this method starts to approach the preceding method (using *ranges*) in complexity.

The process can be reduced in complexity by various stages until the point is reached where all occurrences are related to a *single* combination of the pertinent flight parameters, usually the most severe flight condition possible. This method results, in most cases, in the most unrealistic representation of actual aircraft usage.

The determination of representative loading conditions is also a rather subjective process. The subjectivity can be greatly reduced by reducing the complexity of the process, but, as noted previously, the resulting spectrum may not be as representative as it could be or should be. However, if certain additional data or information can be obtained, the subjectivity can be reduced without decreasing the accuracy of the spectra.

An example of these additional data is a specific knowledge of what loading conditions the aircraft is capable of achieving. For never-been-flown aircraft, the V–n diagram is a good source for this data. For other aircraft, the V–n diagram should be used in conjunction with any pertinent flight test or flight recorder data to ascertain the aircraft's limitations.

Another way to reduce the subjectivity is a basic knowledge of how the aircraft is flown. That is, a general understanding of what maneuvers are flown, when they are flown, and how they are flown. For instance, in a ground support combat segment, the following might be useful information:

A. External stores will be dropped before strafing runs are made.
B. The highest load factors will normally be achieved at weapon delivery speed.

From these two items, the following assumptions could be made:

A. There will be a dramatic shift in gross weight early in the segment thereby establishing the gross weight range at which external store drop maneuvers will be flown,
B. The gross weight range will be established at which strafing maneuvers will be flown,

C. The high load factors should only be associated with weapon delivery speeds, and further,

D. Since, in this type of segment, weapon delivery is usually done at relatively low altitudes, the highest load factors should only be associated with low altitudes as well as only with weapon delivery speed.

This information is best obtained from first-hand sources such as pilots or weapons systems operators. Training manuals, technical manuals, and mission syllabuses may also be valuable sources.

Other information that might be obtained could take the form of operating restrictions for certain external store configurations, operating limitations due to engine restrictions, etc. Obviously any data that can establish a narrower bound on the number of loading conditions possible is useful.

In paragraphs 3.4.1–3.4.9 a detailed example is presented which:

A. Illustrates the processes described in Sections 3.3 and 3.4 and a method of combining these processes in selecting loading conditions.

B. Is representative of what has actually been done to develop spectra for a fighter type aircraft.

The example is not to be taken as a recommendation of an exact method. It is representative of what has been done. It should also be noted that the various distributions and definitions used in the example are for a specific aircraft and may not be applicable to or adequate for another aircraft. As stated previously many factors influence the degree of detail needed and the individual will have to decide based on these factors, what level of complexity to utilize.

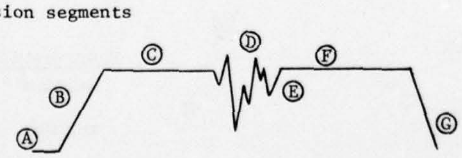
3.4.1 Initial Data

The mission profile information received from the user is as follows:

Design objective for service life	4000 flight hours
Design objective for landings	4000 landings
Aircraft utilization	85% air-to-air combat 15% air-to-ground combat
Number and types of missions	3 air-to-air missions 2 air-to-ground missions
3 configurations will be considered	"A" Basic aircraft with wing tip rockets "B" Conf. "A" plus centerline store "C" Conf. "B" plus inboard and outboard wing stores

Mission profiles for the 5 missions are presented in Figures 3.4.1.1 through 3.4.1.5.

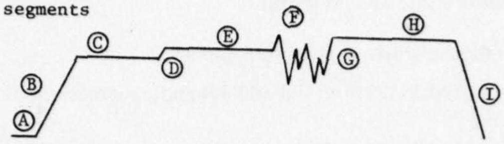
1. Average flight time - 53.7 minutes
2. Configuration "A"
3. Take-off gross weight - 15,470 lb.
4. Take-off fuel - 4360 lb.
5. Mission radius - 153 nautical miles
6. Air-to-Air combat averages 9.6 minutes at various airspeeds, altitudes, and load factors
7. Mission segments



		Power setting
A	Take-off	military
B	Climb	military
C	Cruise	optimum cruise
D	Air-to-Air Combat	maximum
E	Climb	military
F	Cruise	optimum cruise
G	Descent	penetration (80%)

Fig.3.4.1.1 Air combat Mission I

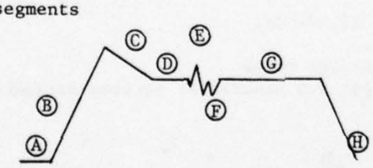
1. Average flight time 111.6 minutes
2. Configuration - "B"
3. Take-off gross weight - 17,564 lb.
4. Take-off fuel - 6148 lb.
5. Mission radius - 377 nautical miles
6. Air-to-Air Combat averages 10.4 minutes at various airspeeds altitudes and load factors
7. Mission segments



		Power setting
A	Take-off	military
B	Climb	military
C	Cruise	optimum cruise
D	Climb	military
E	Cruise	optimum cruise
F	Combat	maximum
G	Climb	military
H	Cruise	optimum cruise
I	Descent	penetration (80%)

Fig.3.4.1.2 Air combat mission II

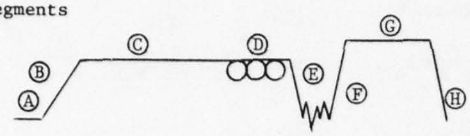
1. Average flight time - 31.0 minutes
2. Configuration - "A"
3. Take-off gross weight - 15,325 lb.
4. Take-off fuel - 4360 lb.
5. Mission radius - 103 nautical miles
6. Air-to-Air Combat averages 4.2 minutes at various airspeeds, altitudes, and load factors
7. Mission segments



		Power setting
A	Take-off	maximum
B	Climb	maximum
C	Dive	maximum
D	Cruise	maximum
E	Air-to-Air Combat	maximum
F	Climb	military
G	Cruise	optimum cruise
H	Descent	penetration (80%)

Fig.3.4.1.3 Point intercept mission

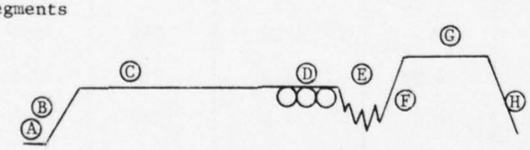
1. Average flight time - 78.4 minutes
2. Configuration - "B"
3. Take-off gross weight - 17,549 lb.
4. Take-off fuel - 4360 lb.
5. Mission radius - 100 nautical miles
6. Air-to-Ground Combat averages 1.8 minutes at various airspeeds, altitudes, and load factors
7. Mission segments



		Power setting
A	Take-off	military
B	Climb	military
C	Cruise	optimum cruise
D	Loiter	loiter
E	Combat	military
F	Climb	military
G	Cruise	optimum cruise
H	Descent	penetration (80%)

Fig.3.4.1.4 Close air support mission I

1. Average flight time - 88.3 minutes
2. Configuration - "C"
3. Take-off gross weight - 20,104 lb.
4. Take-off fuel - 6148 lb
5. Mission radius - 100 nautical miles
6. Air-to-Ground Combat averages 4.8 minutes at various airspeeds, altitudes, and load factors
7. Mission segments



		Power setting
A	Take-off	military
B	Climb	military
C	Cruise	optimum cruise
D	Loiter	loiter
E	Combat	military
F	Climb	military
G	Cruise	optimum cruise
H	Descent	penetration (80%)

Fig.3.4.1.5 Close air support mission II

3.4.2 Segmented Mission Profiles

By making certain assumptions and calculations the mission profile data in paragraph 3.4.1 is changed into a form that is more easily used in developing representative loading spectrums. A mission mix at 85% air-to-air and 15% air-to-gound is specified, but the individual mission utilization is not given. Hence, it is necessary to assume a distribution of the five missions. This assumed distribution is presented in Column 4 of Table 3.4.2.1 as percent type mission. The mission utilization is obtained as the product of the mission mix percent and the percent of type mission. The number of flights of each mission and mission duration are also shown in Table 3.4.2.1. The number of flights at each mission per lifetime is obtained in the following manner.

A. Average flight duration is calculated as

$$T_{ave} = \sum \frac{\text{Mission duration x mission utilization}}{100}$$

$$= 74 \text{ minutes} = 1.233 \text{ hours.}$$

B. Total number of flights per life is then

$$N = \frac{4000 \text{ hours}}{T_{ave}}$$

$$= \frac{4000}{1.233} = 3244 \text{ flights.}$$

C. Number of flights of each mission is the product of 3244 flights and the mission utilization.

TABLE 3.4.2.1

Mission Utilization

1	2	3	4	5	6	7	8	9
Type Mission	Mission Mix %	Mission	% of Type Mission (assumed)	Mission Utilization % 2 X 4	No. Flts Per Life 5 X 3244	Mission Duration Min	Mission Time Per Life Hr 6 X 7/60	Percent of Life-Time 8/4000
Air-to-Air		ACM I	40.0	34.0	1103	53.7	987	24.7
		ACM II	40.0	34.0	1103	111.6	2052	51.3
		PI	20.0	17.0	552	31.0	285	7.1
Air-to-Ground		CAS I	50.00	7.5	243	78.4	318	7.9
		CAS II	50.0	7.5	243	88.3	358	9.0
Total	100			100			4000	100

By using known performance capabilities and fuel consumption rates, the average gross weight, altitude, and Mach number for each mission segment plus the time spent in each segment may be determined. This information is presented in Table 3.4.2.2. Also included in this table are the percentages of total aircraft life time for each segment. The percentage of total aircraft life time for each segment is the product of the "percent of mission (% miss)" and the "percent of lifetime".

3.4.3 Maneuver Load Factor Spectra

This segmented mission profile data is now used with mission segment load factor spectra to obtain aircraft load factor spectra. The mission segment load factor spectra are obtained from Reference 1. It should be noted that the maneuver load factor spectra are truncated at the 9g level because experiencing load factors higher than 9g's for this aircraft is considered unrealistic. Figure 3.4.3.1 presents the positive maneuver load factor cumulative occurrences for each type of mission segment. Negative maneuver spectra are specified only for the air-to-air and air-to-ground combat segments and are plotted in Figure 3.4.3.2. Appropriate spectra are used to determine the load factor experience within each segment of the five mission profiles. The selection of spectra to be used with each segment is shown in Tables 3.4.3.1 through 3.4.3.5. Maneuver spectra for each mission are derived from the mission segment spectra by summing the products of the cumulative occurrences per 1000 hours at each n_z of the appropriate segment spectra and the related percent/100 of the mission. This procedure is expressed mathematically on facing page.

TABLE 3.4.2.2

Mission Segment Flight Parameters

Mission	Percent of Lifetime	Mission Segment	Segment Time (Min)	% Miss	% Mix	Configuration	Average Flt Parameters GW(LB)	ALT(FT)	Mach No.
ACM I	24.7	Take-off	0.7	1.3	0.3	"A"	15,470	S.L.	0.28
		Climb	6.5	12.1	3.0		14,933	18,500	0.73
		Cruise	12.9	24.0	5.9		14,464	37,300	0.82
		Combat	9.6	17.9	4.4		13,420	30,000	0.75
		Climb	2.3	4.3	1.1		12,525	35,200	0.89
		Cruise	17.1	31.8	7.9		12,246	40,750	0.82
		Descent	4.6	8.6	2.1		12,025	20,550	0.59
ACM II	51.3	Take-off	0.8	0.7	0.4	"B"	17,564	S.L.	0.27
		Climb	8.0	7.2	3.7		16,925	17,100	0.71
		Cruise	16.3	14.6	7.5		16,344	34,500	0.81
		Climb	0.2	0.2	0.1	Dropped Center-line Store	15,911	35,150	0.89
		Cruise	23.7	21.2	10.9		15,378	36,000	0.82
		Combat	10.4	9.3	4.8		14,122	30,000	0.75
		Climb	2.2	2.0	1.0		13,163	34,650	0.89
		Cruise	45.4	40.7	20.9		12,565	39,700	0.82
		Descent	4.6	4.1	2.1		12,025	20,550	0.59
P.I.M.	7.1	Take-off	1.1	3.55	0.3	"A"	15,325	S.L.	0.45
		Climb	2.2	7.1	0.5		14,608	18,750	0.92
		Dive	4.0	12.9	0.9		13,880	36,250	1.21
		Cruise	1.7	5.5	0.4		13,208	35,000	1.49
		Combat	4.2	13.5	1.0		12,671	35,000	0.78
		Climb	1.4	4.5	0.3		12,340	37,450	0.89
		Cruise	11.8	38.1	2.7		12,171	40,000	0.80
		Descent	4.6	14.8	1.1		12,035	20,000	0.58
CAS I	7.9	Take-off	0.8	1.0	0.1	"B"	17,549	L.S.	0.27
		Climb	2.5	3.2	0.3		17,103	7,500	0.63
		Cruise	13.0	16.6	1.3		16,724	15,000	0.62
		Loiter	43.0	54.8	4.3		15,707	15,000	0.55
		Combat	1.8	2.3	0.2	Dropped Center-line Store	13,846	5,000	0.60
		Climb	6.1	7.8	0.6		12,545	17,750	0.74
		Cruise	6.6	8.4	0.7		12,256	40,650	0.82
		Descent	4.6	5.9	0.5		12,180	20,400	0.59
CAS II	9.0	Take-off	1.0	1.1	0.1	"C"	20,104	S.L.	0.25
		Climb	4.4	5.0	0.4		19,491	7,500	0.59
		Cruise	10.0	11.3	1.0		18,881	15,000	0.64
		Cruise	0.7	0.8	0.1	Dropped Center-line Store	18,485	15,000	0.64
		Loiter	50.0	56.6	5.1		17,271	15,000	0.55
		Combat	2.7	3.1	0.3	Dropped Inboard Wing Stores	15,387	5,000	0.60
		Combat	2.1	2.4	0.2	Dropped Outboard Wing Stores	14,051	5,000	0.60
		Climb	7.8	8.8	0.8		13,136	17,050	0.72
		Cruise	5.1	5.8	0.5		12,804	39,250	0.80
		Descent	4.5	5.1	0.5		12,736	19,700	0.58

At each n_z

$$(co/1000)_{mission} = \sum_{j=1}^{j=N} \underbrace{(co/1000)_{segment\ spectra}}_{\text{Figs 3.4.3.1 \& 3.4.3.2}} \times \underbrace{P_p/100}_{\text{Table 3.4.2.2}}$$

where co/1000 = cumulative occurrences per 1000 hours

P_p = percent of mission = t/T × 100

t = segment time

T = mission duration

j = variable representing each mission segment

Plots of the resultant mission spectra are presented in Figure 3.4.3.3. The composite spectra are determined as the sum of products of the mission spectra and the mission "percent of lifetime".

At each n_z

$$(co/1000)_{composite} = \sum_{K=1}^{K=5} (co/1000)\ \text{mission} \times \frac{P}{100}$$

where P = percent of lifetime (Table 3.4.2.1)

K = variable representing each mission

A graph of the composite maneuver spectra is shown in Figure 3.4.3.4. In developing maneuver loads to be used in conjunction with the maneuver spectra, the gross weights, Mach numbers, and altitudes of the mission segments will be considered. The maneuver spectra of Figure 3.4.3.4 are only a convenient representation of the total maneuver environment, and by themselves do not define load levels. The first step toward a more complete definition is to associate the load factor occurrences with each mission segment. This was accomplished in part during the building processes of the composite spectra. On a per thousand hour basis the cumulative load factor occurrences in each segment are the product of the appropriate segment spectra and the segment time. In the actual fatigue analysis, a per lifetime basis is more appropriate. For a 4000 hour lifetime, the cumulative occurrences in each segment are obtained as follows:

At each n_z of a mission segment in a particular mission,
$(co/4000) = (co/1000)_{\text{segment spectra}} P_p/100 \times (P/100) \times 4$
where terms are as previously defined.

Load factor occurrences at discrete n_z levels during a lifetime may be obtained as the difference between the cumulative occurrences at the upper and lower boundaries of the interval.

$$\text{Occur. per life @ } n_{z_i} = (co/4000) \text{ @ } n_{z_i} - (co/4000) \text{ @ } n_{z_{i+1}}$$

where n_{z_i} = i th discrete n_z level.

These data are listed for each segment of the five missions in Tables 3.4.3.1 through 3.4.3.5. Only one mission, point intercept, contains supersonic flight segments. These segments, dive, cruise, and part of combat, contain the maneuvers which will be considered supersonic. Load factor occurrences in these segments are listed in Table 3.4.3.3. According to the mission profile, the air-to-air combat is initiated at subsonic speeds. However, in order to account for the transition from the supersonic cruise to the subsonic combat, ten percent of the occurrences at each position n_z in the combat segment shall be assumed to be supersonic, steady, symmetrical maneuvers. These supersonic maneuvers and the subsonic maneuvers are listed separately in Table 3.4.3.3.

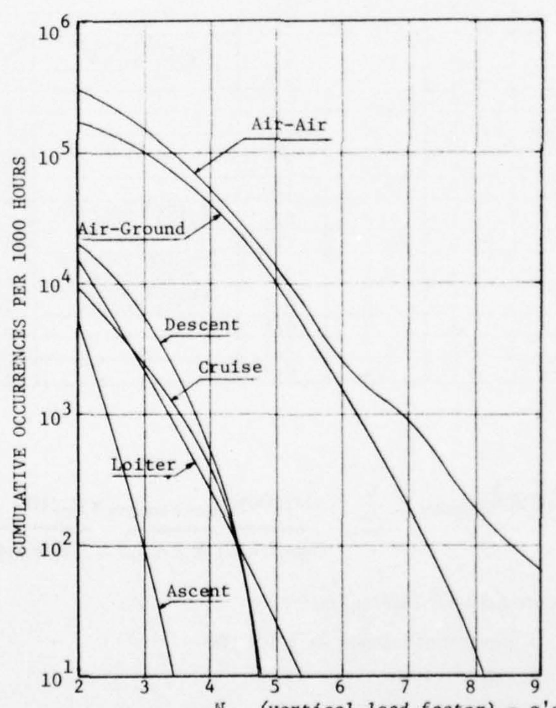


Fig.3.4.3.1 Mission segment positive maneuver spectra
(from Reference 1)

3.4.4 Abrupt and Unsymmetrical Maneuvers

Thus far, the maneuver load factor occurrences have been defined without regard to what type of maneuver they should be associated with. In this example three basic types of maneuvers shall be considered; symmetrical maneuvers, abrupt maneuvers, and unsymmetrical maneuvers. These maneuvers shall be defined as follows; an abrupt maneuver is

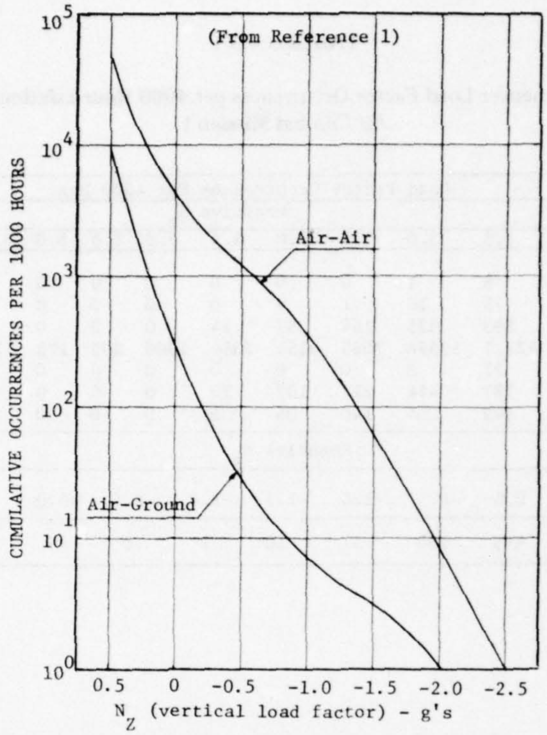


Fig.3.4.3.2 Mission segment negative maneuver spectra

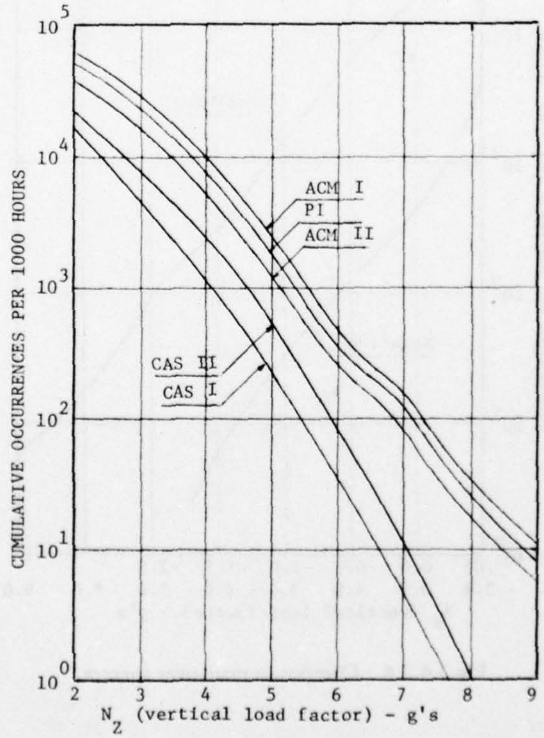


Fig.3.4.3.3 Maneuver spectra by mission

TABLE 3.4.3.1

Maneuver Load Factor Occurrences per 4000 Hour Lifetime
Air Combat Mission I

Mission Segment	Applicable n_z Spectrum	Load Factor Occurrences Per 4000 Hrs														
		Positive n_z														
		2.0	2.5	3.0	3.5	4.0	4.5	5.0	5.5	6.0	6.5	7.0	7.5	8.0	8.5	9.0
Take-off	Ascent	55	8	1	0	0	0	0	0	0	0	0	0	0	0	0
Climb	Ascent	512	75	10	1	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	1187	593	335	164	81	14	0	0	0	0	0	0	0	0	0
Combat	Air-Air	13777	12717	10598	7065	4151	2384	1360	495	173	110	85	42	15	7	11
Climb	Ascent	181	27	3	0	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	1573	787	444	217	107	19	0	0	0	0	0	0	0	0	0
Descent	Descent	779	449	284	140	36	6	0	0	0	0	0	0	0	0	0
		Negative n_z														
		0.5	0.0	-0.5	-1.0	-1.5	-2.0	-2.5	-3.0							
Combat	Air-Air	7074	495	150	51	10	1	0	0							

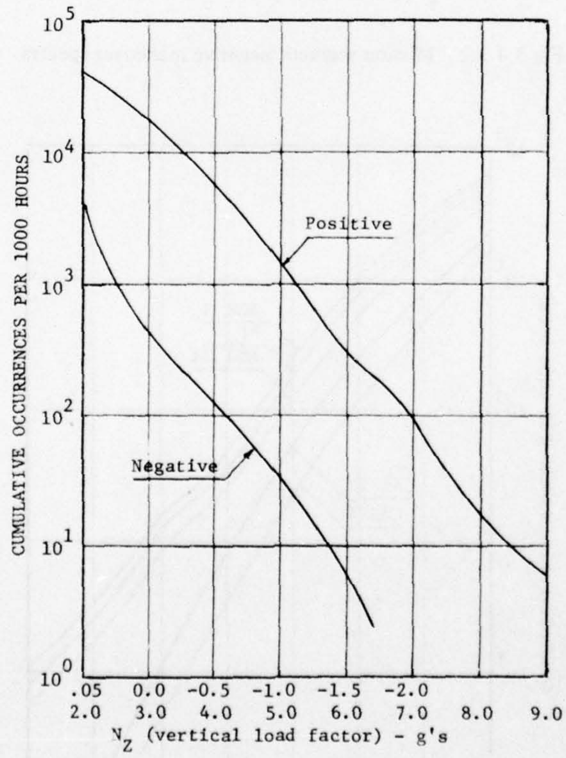


Fig.3.4.3.4 Composite maneuver spectra

TABLE 3.4.3.2

Maneuver Load Factor Occurrences per 4000 Hour Lifetime
Air Combat Mission II

Mission Segment	Applicable n_z Spectrum	Load Factor Occurrences Per 4000 Hrs														
		Positive n_z														
		2.0	2.5	3.0	3.5	4.0	4.5	5.0	5.5	6.0	6.5	7.0	7.5	8.0	8.5	9.0
Take-off	Ascent	63	9	1	0	0	0	0	0	0	0	0	0	0	0	0
Climb	Ascent	730	93	12	1	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	1499	749	423	207	102	18	0	0	0	0	0	0	0	0	0
Climb	Ascent	16	2	0	0	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	2179	1089	614	301	148	26	0	0	0	0	0	0	0	0	0
Combat	Air-Air	14916	13768	11474	7649	4494	2582	1472	535	187	119	92	46	16	7	11
Climb	Ascent	173	26	3	0	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	4174	2087	1177	576	284	49	1	0	0	0	0	0	0	0	0
Descent	Descent	778	448	283	140	36	6	0	0	0	0	0	0	0	0	0
		Negative n_z														
		0.5	0.0	-0.5	-1.0	-1.5	-2.0	-2.5	-3.0							
Combat	Air-Air	7634	534	162	56	9	2	0	0							

TABLE 3.4.3.3

Maneuver Load Factor Occurrences per 4000 Hour Lifetime
Point Intercept Mission

Mission Segment	Applicable n_z Spectrum	Load Factor Occurrences per 4000 Hrs														
		Positive n_z														
		2.0	2.5	3.0	3.5	4.0	4.5	5.0	5.5	6.0	6.5	7.0	7.5	8.0	8.5	9.0
Take-off ②	Ascent	43	6	1	0	0	0	0	0	0	0	0	0	0	0	0
Climb ②	Ascent	86	13	2	0	0	0	0	0	0	0	0	0	0	0	0
Dive ①	Cruise	183	92	52	25	13	2	0	0	0	0	0	0	0	0	0
Cruise ①	Cruise	78	39	22	11	5	1	0	0	0	0	0	0	0	0	0
Combat ①	Air-Air	300	277	231	154	90	52	30	11	4	2	2	2	1	1	1
Combat ②	Air-Air	2701	2493	2078	1385	814	467	266	97	34	22	17	7	2	0	1
Climb ②	Ascent	55	8	1	0	0	0	0	0	0	0	0	0	0	0	0
Cruise ②	Cruise	541	270	152	75	37	6	0	0	0	0	0	0	0	0	0
Descent ②	Descent	388	223	141	70	18	3	0	0	0	0	0	0	0	0	0
		Negative n_z														
		0.5	0.0	-0.5	-1.0	-1.5	-2.0	-2.5	-3.0							
Combat	Air-Air	1534	107	33	11	2	0	0	0							

① Supersonic
② Subsonic

TABLE 3.4.3.4

Maneuver Load Factor Occurrences per 4000 Hour Lifetime
Close Air Support Mission I

Mission Segment	Applicable n_z Spectrum	Load Factor Occurrences Per 4000 Hrs														
		Positive n_z														
		2.0	2.5	3.0	3.5	4.0	4.5	5.0	5.5	6.0	6.5	7.0	7.5	8.0	8.5	9.0
Take-off	Ascent	14	2	0	0	0	0	0	0	0	0	0	0	0	0	0
Climb	Ascent	43	6	1	0	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	262	131	74	36	18	3	0	0	0	0	0	0	0	0	0
Loiter	Loiter	1560	659	248	90	29	10	3	1	0	0	0	0	0	0	0
Combat	Air-Grnd	254	290	239	196	138	80	44	18	7	3	1	0	0	0	0
Climb	Ascent	105	16	2	0	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	133	67	38	18	9	2	0	0	0	0	0	0	0	0	0
Descent	Descent	171	98	62	31	8	1	0	0	0	0	0	0	0	0	0
		Negative n_z														
		0.5	0.0	-0.5	-1.0	-1.5	-2.0	-2.5	-3.0							
Combat	Air-Grnd	70	3	0	0	0	0	0	0							

TABLE 3.4.3.5

Maneuver Load Factor Occurrences per 4000 Hour Lifetime
Close Air Support Mission II

Mission Segment	Applicable n_z Spectrum	Load Factor Occurrences Per 4000 Hrs, Positive n_z: 2.0	2.5	3.0	3.5	4.0	4.5	5.0	5.5	6.0	6.5	7.0	7.5	8.0	8.5	9.0
Take-off	Ascent	17	3	0	0	0	0	0	0	0	0	0	0	0	0	0
Climb	Ascent	77	11	2	0	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	204	102	58	28	14	2	0	0	0	0	0	0	0	0	0
Cruise	Cruise	14	7	4	2	1	0	0	0	0	0	0	0	0	0	0
Loiter	Loiter	1835	775	292	106	34	12	4	1	0	0	0	0	0	0	0
Combat	Air-Grnd	385	440	363	297	209	121	66	28	10	4	2	1	0	0	0
Combat	Air-Grnd	300	343	283	231	163	94	51	21	8	3	1	0	0	0	0
Climb	Ascent	136	20	3	0	0	0	0	0	0	0	0	0	0	0	0
Cruise	Cruise	104	52	29	14	7	1	0	0	0	0	0	0	0	0	0
Descent	Descent	169	97	62	30	8	1	0	0	0	0	0	0	0	0	0

		Negative n_z: 0.5	0.0	-0.5	-1.0	-1.5	-2.0	-3.0
Combat	Air-Grnd	191	6	1	0	0	0	0

a maneuver with non-zero pitching or rolling acceleration: an unsymmetrical maneuver is one in which a roll velocity equal to or greater than 30°/sec occurs in combination with an incremental load factor, $n_z \geqslant \pm 1.0$ (i.e., all roll velocities considered were coupled with maneuvers with an incremental load factor, $n_z \geqslant \pm 1.0$. No roll velocities were considered to occur with incremental load factors less than $n_z = \pm 1.0$). All other maneuvers are considered to be symmetrical maneuvers. The type of maneuvers should be related to the mission segment in which they are most likely to occur. This is a relatively straightforward matter with the five mission profiles established for this aircraft. Since these profiles are combat type missions, normal pilot procedures would preclude most abrupt maneuvers and unsymmetrical maneuvers from the non-combat mission segments. Thus, all maneuvers associated with climb, cruise, loiter, and descent segments shall be considered symmetrical. Within the combat segment, the percentages of abrupt and unsymmetrical maneuvers must be established.

Since there was no statistical data available to establish these percentages, an instrumented aircraft from an earlier flight test program was obtained and flown on simulated air-to-air and air-to-ground combat missions. The resulting measured data was used to establish the following:

A. 25 percent of all maneuvers in the air-to-air and air-to-ground segments are abrupt.

B. 90 percent of all maneuvers in the air-to-air segments are unsymmetrical.

C. 10 percent of all maneuvers, in the air-to-ground segments are unsymmetrical.

The only abrupt maneuvers considered are abrupt pitch maneuvers. Based on the measured data a horizontal tail displacement rate of $S_H = 7$ degrees/second will be used to calculate loads for all abrupt pitch maneuvers.

Also based on the measured data, the rolling velocities to be considered for the unsymmetrical maneuvers were determined. These rolling velocities and their distributions are presented in Table 3.4.4.1.

TABLE 3.4.4.1

Rolling Velocities for Unsymmetrical Maneuvers

Percent of Unsymmetrical Maneuvers	Roll Velocity deg/sec
60	40
35	75
3	125
2	175

3.4.5 Gust Loads

Gust loads normally have little or no effect on the fatigue/fracture life of high load factor aircraft. However, gust loads were investigated for this example aircraft to insure that no possible significant loadings would be overlooked. The aircraft gust load spectra will be based on power spectral techniques which account for the mixture of gusts of all wavelengths and the higher gust velocities associated with the longer wavelengths. Power spectral techniques applicable to airplane gust loads are discussed in Reference 7. Also, aircraft structural responses to atmospheric turbulence as determined from instrumented aircraft flight tests will also be used. The final gust spectra will reflect the anticipated airplane usage that is defined in paragraph 3.4.2. The basic approach that will be employed is illustrated by the flow chart in Figure 3.4.5.1.

The first equation of Figure 3.4.5.1 gives a relation from which the frequency of exceedance, $N(y)$, of a given level of the response y can be obtained for a particular flight condition. In that equation, the P and b terms represent the likelihood of encountering turbulence and the probable intensity of the turbulence, with subscripts 1 and 2 denoting non-storm and storm turbulence. The values of these parameters along with the scale of turbulence L are specified by Reference 4 as a function of altitude in Table 3.4.5.1. The structural response is characterized by the terms $\bar{A}$ and N_o. $\bar{A}$ is an average of the response amplitude and N_o is a measure of the typical response frequency. The data was available at various flight conditions and include c.g. vertical acceleration and structural loads at various locations. This data can be used to represent the aircraft response for each of the mission segments by taking into account any deviation of altitude, Mach number or gross weight from the corresponding flight test condition. The use of flight test data to evaluate the aircraft structural response automatically includes complex effects such as aerodynamic induction, transient lift growth, stability augmentation and obviates the need for a time consuming analysis involving airplane flexible mode shapes determination. The second equation of Figure 3.4.5.1 illustrates how the total gust spectra can be obtained by summing up the frequencies of exceedance over one mission and then over the entire life. In that equation, t_j denotes the fraction of mission profile time in one segment and t_k the fraction of total airplane usage represented by a particular mission profile.

Figure 3.4.5.2 presents the aircraft spectrum of vertical load factor due to gust on a per thousand hour basis. In this case, the total spectrum is the sum of the five missions spectra which are also shown in the figure. While the close air support missions exhibit a higher percentage of mission time at low altitude than the air to air missions, their contributions are lower because their flight time utilization is less. Also, the close air support missions are flown most of the time in high-drag stored configurations at lower Mach numbers. These factors tend to reduce aircraft response and gust encounter frequency.

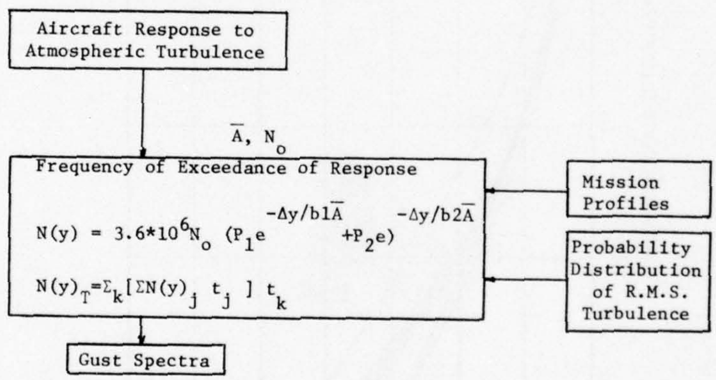


Fig.3.4.5.1 Basic approach to fatigue gust load determination

TABLE 3.4.5.1

Atmospheric Turbulence Parameters

Altitude	P_1	P_2	b_1	b_2	L
0 - 1,000 ft	1.0	0	3.9	--	500 ft
1 - 2,000	0.32	0.0004	4.6	9.4	1,000
2 - 10,000	0.08	0.00125	3.8	9.8	1,000
10 - 20,000	0.045	0.0015	3.7	10.4	1,000
20 - 30,000	0.06	0.0012	3.5	11.2	1,000
30 - 40,000	0.065	0.0006	3.4	11.1	1,000
40 - 50,000	0.023	0.0002	3.1	11.7	1,000
50 - 60,000	0.02	0.0001	2.8	12.5	1,000

3.4.6 Landing

As a design objective, the aircraft will be designed to withstand loads associated with 4000 landings. Since an aircraft service life of 4000 hours is attained during 3244 flights, many of the flights must contain two or more landings. It shall be assumed that the additional 756 landings are touch and go's which are randomly distributed among an equal number of flights. Consequently, these 756 flights will consist of one touch and go landing and one full-stop landing.

TABLE 3.4.5.2

Percentage of Flight Time in Altitude Ranges Applicable to Gust Analysis

Altitude (ft)	Percent of Lifetime	Average Mach Number
0 - 1,000	1.1	0.35
1 - 2,000	0	0
2 - 10,000	1.4	0.63
10 - 20,000	20.9	0.61
20 - 30,000	5.7	0.50
30 - 40,000	59.7	0.88
40 - 50,000	11.2	0.90

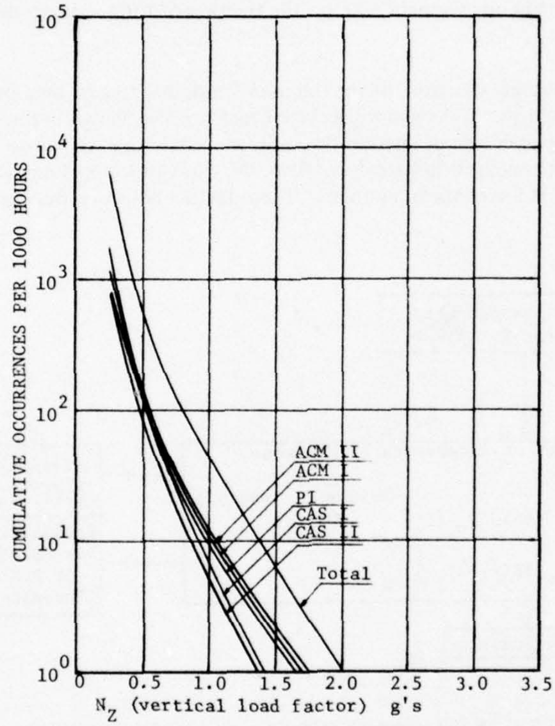


Fig.3.4.5.2 Gust spectra by mission and total up or down gust

The mission profiles being used are performance-oriented resulting in the airplane landing with only landing reserve fuel on board. All external stores except the tip missiles have been released prior to landing. Thus, the mission profiles include only one configuration and essentially only one gross weight during landing. This is unrealistic and would most likely result in unconservative landing loads spectra. Instead, a rational approach which will include consideration of aborted missions, hung stores, and heavy landing shall be adopted. The mission utilization and takeoff configuration of the mission profiles will serve as a basis for the rational approach, as will applicable data from other aircraft.

After reviewing landing gross weights recorded for actual combat flights and actual training flights from similar aircraft, the landing gross weights and configurations in Table 3.4.6.1 were specified as being rational and representative of actual usage.

A distribution of aircraft landing sink speeds is specified in Reference 1. These sink speeds are combined with the gross weight distribution to form the joint distribution presented in Table 3.4.6.2. Since the design limit sinking speed for gross weights above the maximum landing design weight of 15,000 pounds is 6.0 fps, all landings at higher sink speeds are considered to occur at gross weights below 15,000 pounds.

The gear loads at landing will be determined by analytical methods based on the gross weight – sink speed distribution of Table 3.4.6.2. In developing the landing loads spectra, it is intended that no loads greater than limit design loads will be considered. Greater loads will be reduced to limit load.

Side drift landings shall be considered to occur once out of every ten landings. Side loads will be assumed to be proportional to the vertical loads. A value of $\mu = 0.8$ acting inboard and $\mu = 0.6$ acting outboard shall be used.

TABLE 3.4.6.1

Landing Configuration and Weights

Gross Weight lbs.	Weight Range	Configuration	c.g.	Percent
12,000	11,000 - 12,999	"A"	Fwd Aft	37.0 37.0
14,000	13,000 - 14,999	"A"	Fwd Aft	10.5 10.5
16,000	15,000 - and over	"B" "C"	Fwd Aft Aft	2.5 2.4 0.1

TABLE 3.4.6.2

Landing Gross Weight and Sink Speed Distribution

Sink Speed FPS	Percent of Landings							
	12,000 lbs Config. A		14,000 lbs Config. A		16,000 lbs			Total
					Config. B		Config. C	
	Fwd c.g.	Aft c.g.	Fwd c.g.	Aft c.g.	Fwd c.g.	Aft c.g.	Aft c.g.	
1	6.660	6.660	1.890	1.890	0.450	0.432	0.018	18.000
2	10.730	10.730	3.045	3.045	0.725	0.696	0.029	29.000
3	9.620	9.620	2.730	2.730	0.650	0.624	0.026	26.000
4	5.735	5.735	1.628	1.628	0.388	0.372	0.016	15.500
5	2.886	2.886	0.819	0.819	0.195	0.187	0.008	7.800
6	0.962	0.962	0.270	0.270	0.070	0.060	0.003	2.600
7	0.312	0.312	0.088	0.088				0.800
8	0.058	0.058	0.017	0.017				0.150
9	0.039	0.039	0.011	0.011				0.100
10	0.019	0.019	0.006	0.006				0.050
Total								
	Number of Landings							
1	266	266	76	76	18	18	1	721
2	428	428	122	122	29	28	1	1158
3	385	385	109	109	27	25	1	1041
4	230	230	65	65	16	15	1	622
5	116	116	33	33	8	8		314
6	38	38	11	11	3	2		103
7	12	12	4	3				31
8	2	2	1	1				6
9	2	1						3
10	1	0						1
Total	1480	1478	421	420	101	96	4	4000

3.4.7 Ground Loads

The ground loads spectra are composed of taxi, braking, turning and other ground handling conditions. One-half of these loads will be considered to occur before takeoff while the other half will occur after landing. A distribution of takeoff gross weights and configurations based on the mission profiles is presented in Table 3.4.7.1. A similar table (Table 3.4.6.1) for landing appears in the previous section and as discussed in that section, the landing configurations and gross weights are based on a rational approach as opposed to the mission profile concept. In both tables, each gross weight is assumed equally represented by forward and aft c.g. locations. The taxi and ground handling spectra shall be determined for these takeoff and landing configurations and gross weights. In developing these loads spectra, it is intended that no loads greater than limit design loads will be considered. Greater loads will be reduced to limit load.

Taxi – A distribution of vertical load factors resulting from runway roughness is presented in Reference 1. It shall be assumed that one-half of these occur after landing. Taxi loads resulting from the taxi load factor spectra shall be determined for the takeoff configuration and gross weights as well as the landing configuration and gross weights. The latter are listed in Table 3.4.7.1, the former are shown in Table 3.4.6.1. The resulting joint distribution of taxi load factors and gross weights is presented in Tables 3.4.7.2 and 3.4.7.3. Although the taxi loads occur on a per flight or full

TABLE 3.4.7.1

Takeoff Configurations and Gross Weights

Configuration	Gross Weight	c.g.	Percent
"A"	15,745	Fwd Aft	25.50 25.50
"B"	17,860	Fwd Aft	20.75 20.75
"C"	20,486	Fwd Aft	3.75 3.75

stop landing basis, the number of applications is based on the total 4000 landings. This approach is consistent with previous analysis on other similar aircraft. In addition to the loads represented by Tables 3.4.7.2 and 3.4.7.3, at least one cycle of limit design taxi load shall be considered.

Braking – Reference 1 specifies that hard braking with maximum braking effect shall occur twice per full stop landing and medium braking with half maximum braking effect shall occur an additional five times per full stop landing. This amounts to 6488 cycles of hard braking and 16,220 cycles of medium braking per lifetime. The maximum braking effect shall be based on the brake system capability. Vertical loads shall be as required for airplane balance at 1.0g.

Turning – Based on the requirements of Reference 1, turning with a side load factor of .4 acting at the airplane c.g. shall be considered to occur five times per full stop landing. The side force shall be reacted by inboard acting loads and alternatively as outboard acting loads applied to the main gear. Vertical loads shall be as required for airplane balance. The total cycles of turning load is 16,220.

Pivoting – The pivoting loads specified in Reference 1 shall consist of one cycle, of 1/2 limit torque load applied once every 10 full stop landings. This load shall be combined with the static ground reaction.

Towing – One-half the design limit towing load combined with the static ground reaction, will be applied 3,000 times in the life of the airplane. This requirement is similar to that previously used on other similar aircraft.

Nose Wheel Steering – The maximum torque from the nose wheel steering actuator will be applied once every tenth full stop landing (324 times). One-half the maximum torque will be applied 4 times per full stop landing (12,976 times). These criteria reflect those previously used on other similar aircraft.

Engine Run-Up – Engine run-up shall be assumed to occur once per flight plus an additional 756 times for a total of 4,000 cycles. Drag loads resulting from brake application during run-up shall be balanced by incremental vertical gear loads.

TABLE 3.4.7.2

Occurrences of Taxi Load Factor Prior to Takeoff

Taxi	Config "A" 15,745#		Config "B" 17,860#		Config "C" 20,486#		Total
n_z	Fwd c.g.	Aft c.g.	Fwd c.g.	Aft c.g.	Fwd c.g.	Aft c.g.	
1 ± .05	153,000	153,000	124,500	124,500	22,500	22,500	600,000
1 ± .15	84,150	84,150	68,475	68,475	12,375	12,375	330,000
1 ± .25	13,770	13,770	11,205	11,205	2,025	2,025	54,000
1 ± .35	1,020	1,020	830	830	150	150	4,000
1 ± .45	46	46	37	37	7	7	180
1 ± .55	2	2	2	2	0	0	8
1 ± .65	1	0	0	0	0	0	1
1 ± .75	0	0	0	0	0	0	0

3.4.8 Control Surfaces

Control surface load spectra shall include considerations for ailerons, rudder, flaps and speed brake.

Aileron – The aileron loads spectra are consistent with the planned usage of the airplane and shall reflect the

TABLE 3.4.7.3

Occurrences of Taxi Load Factor After Landing

Taxi n_z	Config "A" 12,000#		Config "A" 14,000#		Config "B" 16,000#		Config "C" 16,000#	Total
	Fwd c.g.	Aft c.g.	Fwd c.g.	Aft c.g.	Fwd c.g.	Aft c.g.	Aft c.g.	
$1 \pm .05$	222,000	222,000	63,000	63,000	15,000	14,400	600	600,000
$1 \pm .15$	122,100	122,100	34,650	34,650	8,250	7,920	330	330,000
$1 \pm .25$	19,980	19,980	5,670	5,670	1,350	1,296	54	54,000
$1 \pm .35$	1,480	1,480	420	420	100	96	4	4,000
$1 \pm .45$	67	66	19	19	5	4	0	180
$1 \pm .55$	3	3	1	1	0	0	0	8
$1 \pm .65$	0	0	0	0	0	0	0	0
$1 \pm .75$	0	0	0	0	0	0	0	0

magnitudes and frequencies associated with the unsymmetrical maneuvers previously discussed in paragraph 3.4.4. In addition, consideration of aileron usage in other maneuvers such as in landing patterns shall be given.

Rudder – The rudder loads spectra are consistent with the planned usage of the airplane and shall reflect the various kinds of maneuvers in which the rudder is used. Particular attention shall be focused on rudder use during unsymmetrical maneuvers at high load factors.

Flaps – The flap loads spectra include consideration for actuating the flaps during maneuvers as well as during takeoff and landing. In formulating these spectra, extensive use was made of the data measured during the previously mentioned simulated air-to-air and air-to-ground combat missions. This data was used in defining the characteristics of typical flap actuation cycles. Definitions of representative flap cycles used in developing these spectra are shown in Figure 3.4.8.1.

Under nominal conditions the flap takes approximately 3 to 4 seconds to cycle from 0/0 (leading edge flap deflection in degrees/trailing edge flap deflection in degrees) setting to a 12/8 setting. Although this is fairly rapid for a flap system, the airplane response time, even for smooth maneuvers, is considerably less. Hence, if the flaps are actuated at the same time a maneuver is initiated, it is likely that the airplane would attain a steady-state load factor before the flaps reached the desired deflection. Review of the simulated combat data indicates that, for 100 recorded load factor occurrences, there were 23 complete extension/retraction cycles of the flap. For 17 out of 23 of these cycles, a normal load factor peak occurs either just before the flap began to extend from the 0/0 position or during the extension cycle before the flap reaches the 12/8 setting. For this reason, it shall be assumed that in a maneuver the flaps are actuated at the first peak load factor experienced. In those cases where the load factor peak exceeds the design flap actuation requirement of 6.0g, the flap shall be actuated at the 6.0g level.

Review of this simulated combat data also indicates that for 9 out of 23 of the flap cycles, retraction occurs at $n_z \geqslant 2.0$g, and for 14 out of 23 cycles retraction occurs at $n_z < 2.0$g. Therefore, it is assumed that 50% of the retractions occur at the preceding peak load factor and 50% of the retraction occurs at 1.0g.

Flap usage during each mission is determined by analyzing the probable normal flap setting for each mission phase. At the beginning of the climb phase, it will be assumed that the pilot fully retracts the flaps from the 24/20 takeoff setting. This operation is assumed to take place at 1.0 g, and the flap cycle is illustrated as cycle number 1 in Figure 3.4.8.1.

Ten percent of the maneuvers in the climb, cruise, and descent phases, and in the CAS-I and CAS-II combat phases, are with flaps extended to the 12/8 setting. Prior to these maneuvers, and after their completion, the flaps are considered to be fully retracted or in the 0/8 cruise setting. This type of flap cycle is shown as cycle number 2 in Figure 3.4.8.1.

Flap settings for the combat phase maneuver occurrences for the ACM-I, ACM-II, and PI missions are dictated primarily by the data recorded in the simulated air combat missions. This data showed that 23 flap cycles took place in conjunction with 100 maneuver occurrences. From this data it could be seen that there were 71 maneuvers performed with flap settings of 12/8, 18/16, and 24/20. This results in an average number of maneuvers per flap actuation of 71/23 = 3.1. An in-depth study of this recorded data revealed that the exact type of flap cycle experienced in air combat can be highly variable due to extensive auto cycling between various flap settings. To simplify the flap cycling criteria, three idealized typical flap cycles have been chosen based on 3 maneuvers per actuation, from the above calculation, and by inspection of the actual flap setting excursions that comprise the 23 noted flap cycles. These cycles are illustrated in Figure 3.4.8.1 as cycles number 3, 4, and 5.

When entering the landing pattern, the maneuvering flap is selected during the initial break over the field and the airspeed is above the higher autoshift speed of 250 KCAS. During the turn to the downwind leg, the flaps shift to the 18/16 position as the speed decays. As the aircraft turns base, it is assumed that the flap either shifts automatically to the 24/20 landing position, or the pilot manually selects this setting. This landing sequence is shown as cycle number 6 in Figure 3.4.8.1.

Speed Brakes – The speed brakes shall be extended and retracted 15,000 times per airplane lifetime. Actuation shall occur at 1.0g. This usage is based on probable deployment during air-to-ground weapons delivery, air-to-air combat, and descent. Since the speed brake loads are a function of airspeed, the loads spectra shall reflect the airspeeds normally associated with these events.

① Retraction @ 1.0g is illustrated

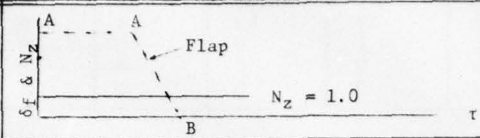
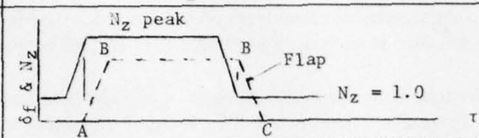
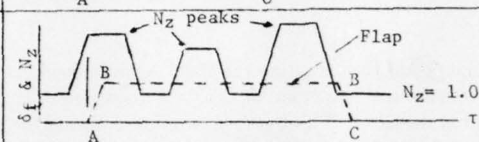
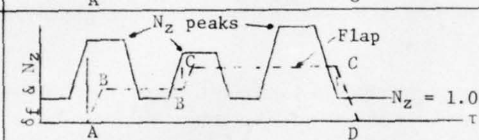
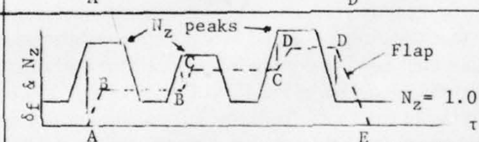
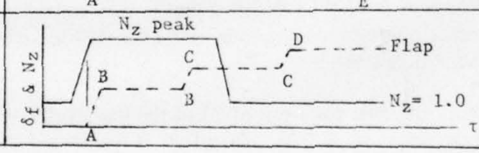
Cycle No.	Definition	Flap Setting	N_z@ Retraction
1		A= 24/20 B= 0/0	1.0
2		A= 0/0 or 0/8 B= {12/8, 18/16, or 24/20 C=0/0 or 0/8	50%@1.0 ① 50%@N_z act.
3		A= 0/0 B= 12/8 C= 0/0	50%@1.0 ① 50%@N_z last
4		A= 0/0 B= 12/8 C= 18/16 D= 0/0	50%@1.0 ① 50%@N_z last
5		A= 0/0 B= 12/8 C= 18/16 D= 24/20 E= 0/0	50%@1.0 ① 50%@N_z last
6		A= 0/0 or 0/8 B= 12/8 C= 18/16 D= 24/20	Retraction @ Zero Load

Fig.3.4.8.1 Flap cycle definition

3.4.9 Miscellaneous Fatigue Loads

Pressurization – The cockpit shall be pressurized to its maximum relief valve setting of 5.1 psi twice per flight. This is consistent with Reference 1 requirements of two pressurization cycles per landing, since the number of flights equals the number of full stop landings. Flight loads shall be combined with the cockpit pressurization.

Store Ejection – The store ejection load shall be based on the mission profiles, but consideration of additional store configurations and ejection sequences shall be included. In the mission profiles, only one mission, Close Air Support II, is configured with wing stores. There are 243 flights of this mission, each with two bombing passes. Thus, the mission profile dictated usage is satisfied with 243 ejections from each pylon. This definition of store release is felt to be too limited in both quantity and scope and another 243 ejections will be added. The additional ejections are selected to include the more severe configurations and sequences. Ejection conditions and associated frequencies are listed in Table 3.4.9.1. As may be noted, all ejections are considered to occur at 1.0g.

Landing Gear Extension/Retraction – The landing gear shall be extended 5,000 times and retracted an equal amount The retraction sequence shall be considered to occur shortly after liftoff at airspeeds which are compatible with the take-off speeds for the gross weights and configurations listed in Table 3.4.7.1. These criteria are reflected in the retraction speed distribution shown in Table 3.4.9.2. In this table the retractions associated with the takeoffs in the "B" and "C" configurations are grouped together.

TABLE 3.4.9.1

Fatigue Store Ejection Conditions

Cond No	Ejection Sequence	n_z @ Eject	Stores ① WS x	Stores ② WS y	Tip	Ejections	Comments
1	WS y	1.0	Store A	Store A	Rocket	122	
2	WS x	1.0	Store A	Store A	Launcher	121	Based on
3	WS x	1.0	Store A	-	Rocket	122	Mission
4	WS y	1.0	-	Store A	Launcher	121	Profiles
5	Ripple	1.0	Store B	Store B	Rocket	100	
6	WS y	1.0	-	Store B	Rocket	72	
7	WS y	1.0	Store B	Store B	Rocket	71	

① x = inboard wing station

② y = outboard wing station

TABLE 3.4.9.2

Gear Retraction Speed Distribution

Takeoff Config	Retraction Speed KEAS	% of Retractions	No of Retractions
"A"	165	51.0	2550
"B" & "C"	185	49.0	2450

All 5,000 gear extensions shall be considered to occur at the gear placard speed of 240 KEAS. This is consistent with the extension speeds used in fatigue analysis for other similar aircraft.

3.5 STRUCTURAL LOADS ANALYSIS

After determining the load conditions to be considered in developing the loading spectra, the next step is calculating the loads for these conditions. The external loads analysis used in the design of the aircraft is normally used for these calculations.

3.5.1 External Load Analysis

An external loads analysis is the determination of the magnitude and distributions of all loads acting on the exterior of the aircraft for various loading conditions. This analysis must include aerodynamic loads, inertia loads, ground loads, and dynamic loads, such as gust, landing, and taxi. The resulting loads are usually expressed in terms of shears, bending moments, and torsions for each component of the aircraft. Using this analysis, the loads for any point on the aircraft can be determined for any loading condition.

3.5.2 Analysis Verification Through Flight Testing

In developing fatigue/fracture loading spectra, it is very important to calculate loads accurately. Small errors in load calculations have been known to generate large errors in fatigue calculations. For this reason, the accuracy of the loads analysis should be verified. The best method to substantiate the analysis is to conduct a flight and ground loads survey to measure actual loads at various loading conditions. The detailed requirements for such a survey are contained in Reference 8 and are very briefly listed as follows: (a) sufficiently instrument the wing, fuselage, empennage, and landing gear to measure flight and ground loads, (b) calibrate this instrumentation to insure accuracy of the measured loads, (c) measure and record loads while flying within and to the limits of the structural boundaries of the aircraft. These loads are then used to verify and/or revise the loads analysis to establish the desired accuracy.

3.6 STRESS ANALYSIS

3.6.1 Internal Loads and Stress Analysis

In designing an aircraft, the external loads are used to conduct the internal loads and stress analysis. An internal loads analysis is performed using either classic hand calculation techniques or finite element solutions. The internal loads analysis will provide shears, bending moments, axial loads, and shear flows for the internal structural members of the structure. Since a more detailed and accurate internal loads analysis is required for fatigue and fracture calculations than for the normally conservative internal loads used for strength calculations, the use of large scale finite element models for calculation of internal loads is becoming increasingly predominate.

These internal loads are then used to perform a stress analysis of the area in question. The stress analysis will give the local axial, bending, shearing, bearing, and combined stresses for the structural component. It should be noted that the stress analysis which was performed to verify structural strength may be unacceptably conservative for fatigue and fracture spectrums and yield unrealistically short analytic lives. For this reason complicated parts such as fittings, joints, and splices are sometimes modeled in finite element programs to obtain internal load and stress distributions.

3.6.2 Analysis Verification Through Testing

For the same reasons as stated for the external loads analysis, it is important to calculate the internal loads and resulting stresses within a reasonable degree of accuracy. The accuracy of these analyses are verified by static tests. This involves applying known loads (usually design loads) to aircraft structural components and measuring the resulting stresses.

The test specimen should be extensive enough so that a realistic stress distribution can be expected in the local area of interest. The method of actual strain measurement depends upon the specimen, local area geometry, and the type of data required. Several strain gages are usually used for actual strain measurements, while brittle stress coats and photo-elastic coatings are often used for stress distributions. The results of these strain surveys are used to verify and/or modify the internal loads and stress analyses to establish the desired accuracy.

3.6.3 Load-to-Stress Ratio

The results of the stress analysis in conjunction with the internal and external loads analyses are used to develop a load-to-stress ratio for any area of interest on the aircraft. The load-to-stress ratio can be in several formats such as PSI/N_z, PSI/Body Bending Moment, PSI/Wing Bending Moment, or PSI/Body Shear, but the format chosen is usually one in which the critical combined stress varies directly with the primary external load.

3.7 DEVELOPING STRESS EXCEEDANCES

The next step in generating fatigue/fracture loading spectra is the development of stress exceedance data. This process will be illustrated by referring back to and combining Sections 3.2 through 3.6 in the following manner:

A. Obtain segmented mission profiles (Section 3.2).
B. Establish load producing structural environment (Section 3.3).
C. Determine loading conditions per segment by coupling flight and ground maneuvers from the load producing structural environment with appropriate parameters from the segmented mission profiles (Section 3.4).
D. Calculate external loads for all defined loading conditions (Section 3.5).
E. Determine the load-to-stress ratios for these areas (Section 3.6).
F. Calculate the stress exceedances for each segment by combining the external loads (D) with the load-to-stress ratios (E).

By following this generalized process, stress exceedance data are developed for each mission segment. This stress exceedance data will be for positive loads (usually $>$ 1g) and negative loads (usually $<$ 1g). At the same time, the stresses corresponding to 1g level flight conditions are calculated.

3.8 GENERAL CONSIDERATIONS IN STRESS SEQUENCING

Stress exceedance data has been developed for positive g (usually $>$ 1g), negative g (usually $<$ 1g), and level flight (= 1g) conditions. It is now necessary to combine these data to form a discrete point-by-point stress-time history, or stress spectrum. In this discussion, the terms *maximum* and *minimum* will be used in describing stress occurrences. Maximum stresses are those stresses associated with positive maneuvers. Minimum stresses are those stresses associated with negative and level flight maneuvers. To derive a spectrum, it is necessary to take a set of maximum and minimum stress exceedance curves and associate maximum stress occurrences with minimum stress occurrences to form stress cycles (minimum stress-maximum stress-minimum stress or maximum-minimum-maximum). This association can be done a number of ways; arbitrarily pairing the highest maximum with the lowest minimum, random association, etc.

The methods that can be used to sequence the stresses vary in complexity. In Section 3.4 it was stated, with regard to the methods used to select loading conditions, that the most complex method would produce the most realistic representation of the loads, and ultimately the stresses, that the aircraft might actually experience. The same statement applies to sequencing methods – the more complex methods will generally produce a more representative spectrum. If a spectrum that represents actual usage is used in a fatigue or crack growth analysis, the results will represent what will occur during actual usage.

There have been many publications pertaining to these topics and other related subjects; among them, References 5, 6, 18–27. All agree that random loading, ordered loading, spectrum severity, spectrum shape, and flight duration can have a great influence on test and analysis results.

Spectra generally fall into two categories: Block or flight-by-flight. A block spectrum consists of groups of stress cycles, with each group having the same maximum and minimum stress. The magnitude of the stress for each block are found by differentiating the stress exceedance curve at a given number of stress levels and using the resultant number of occurrences. The blocks so formed are normally ordered low-high-low, as illustrated in Figure 3.8.0.1 but can also be ordered low-high or high-low. Usually the highest maximum stress is paired with the lowest minimum stress, etc. The block formed represents 1/nth of the life and is repeated n times to make up the spectrum.

Early fatigue and crack growth spectra were all block-type. The test equipment available could not handle complex stress-time histories and the analytical methods in use did not recognize sequence effects. Unfortunately, experience proved these analytical and experimental techniques could not predict actual service problems. Thus it became necessary to define more complex spectra in order to get a better representation of actual usage. Test techniques and tools were developed that could handle the more complex spectra. Analytical methods were developed and recognized sequence effects[9–12]. The flight-by-flight spectrum was conceived to better represent the actual usage. It cannot be stated categorically that a block-type spectrum is not representative of actual aircraft usage, however, for virtually any component in an aircraft structure, the flight-by-flight spectrum is certainly more representative of actual usage. In fact, the USAF Aircraft Structural Integrity Program[13] and the recently written USAF specification for damage tolerance[14] require the use of a flight-by-flight spectrum for analysis and test. It is important to note that numerous studies have shown that analysis using a flight-by-flight spectrum will give more conservative results than analysis using a block-type spectrum when both are derived from the same exceedance data[5,6,15,16,17].

3.8.1 Flight-by-Flight Spectra

There are two categories of flight-by-flight spectra. The first, and more complex, is the mission segment-by-mission segment spectrum. The second is the flight-by-flight spectrum.

3.8.1.1 Mission Segment-by-Mission Segment

The most realistic representation of an aircraft stress-time history is a mission segment-by-mission segment ordering. The maximum and minimum stress exceedance data is broken down into mission segments as previously described (Sections 3.2, 3.3, 3.7) and the stress occurrences within each segment are established by differentiating the curves at a given number of stress levels. The number of stress levels chosen is dependent on the range of stresses and the capability of the analytical tools and test equipment being used. The greater the number of stress levels chosen, the finer the analysis and test.

The maximum and minimum stresses of each segment are paired to form cycles in generally three ways. The first is pairing them on a random basis, assuming that the aircraft returns to a level flight condition or a negative condition after every positive maneuver. A minimum will follow a maximum. The second method is to randomly sequence all the stress occurrences. That is, there may be two or more maximum stress occurrences in a row, or two or more minimum stress occurrences in a row. The third method is to order the stresses in a block-type spectrum fashion within each segment. This method, while less realistic than the others, may in certain cases be acceptable.

After the stress occurrences are ordered within each segment, the segments of each mission type are linked together to form a flight. Then all flights are linked together to form the mission segment-by-mission segment spectrum.

The flights, or missions, can be ordered in two ways. The first being a random sequence of missions; the second, some predetermined order. If the order in which the missions will be shown is known, then that order should be used. If not, the random sequence is the only alternative.

It is important to note that no matter what mission sequencing method is chosen, the mission segments will retain their order within each flight. That is, in each particular flight, a stress associated with a landing load cannot occur before a stress associated with a takeoff load. A representation of a typical mission is illustrated in Figure 3.8.1.1.

3.8.1.2 Flight-by-Flight

The flight-by-flight spectrum is constructed similarly to the mission segment-by-mission segment spectrum. The difference is that the flight-by-flight spectrum is broken down no finer than a mission basis. The stress exceedance data are developed for each mission and the stress occurrrences are then ordered within each flight. The stresses are paired to form cycles and then sequenced using processes identical to those used in deriving the mission segment-by-mission segment spectrum. The methods used to order the missions are also identical.

It is important to note that without breaking each mission into segments the general sequencing of the stresses within each flight is lost. That is, it is possible to have a stress associated with a landing load occur before a stress associated with a takeoff load.

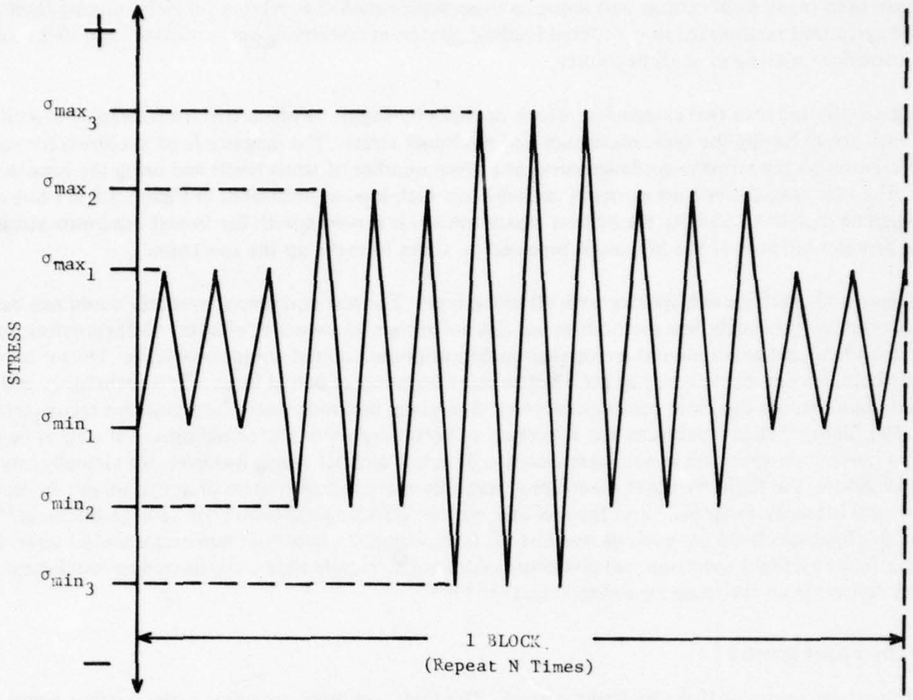


Fig.3.8.0.1 Typical block shape

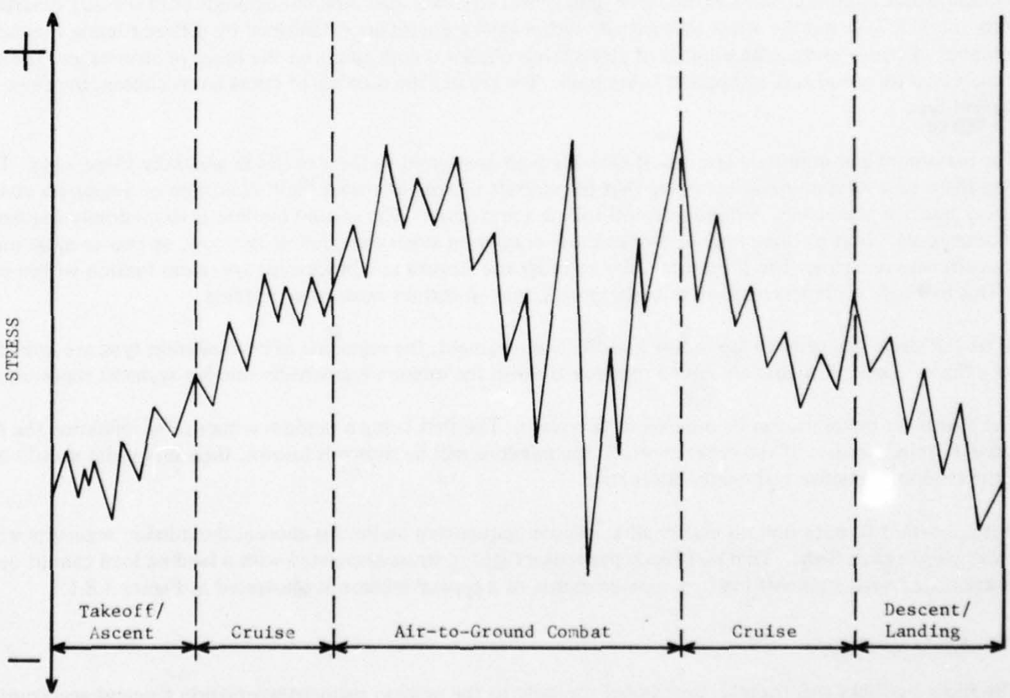


Fig.3.8.1.1 Representation of a mission segment-by-mission segment mission

3.9 CONCLUSIONS AND RECOMMENDATIONS

This chapter was written to be used as a guide in developing realistic fatigue/fracture loading spectra. The information presented is primarily oriented toward showing the designer or engineer the steps involved in deriving the spectra and the various factors that must be considered. There are no firm requirements or methods recommended since these could vary greatly depending upon the aircraft and component being investigated, data available, computer capability, funding, schedules, etc. It is intended that this report be helpful in selecting the most appropriate requirements and methods for application to a particular aircraft component. In doing this, the following general recommendations should be considered.

A. Obtain mission profiles that are as representative as possible of actual aircraft usage.

B. The flight and ground maneuver spectra presented in Reference 1 are based on recorded data from past aircraft. The applicability of these spectra to present day and future aircraft is sometimes questionable. Therefore, reliability and applicable data from other sources should be used to augment these spectra when possible.

C. Within external constraints such as time and/or funding restrictions, lack of detailed information, etc., the more complex, and thus more representative, method of selecting loading conditions should be used.

D. The loads and stress analysis method should be accurate and verified if possible.

E. Within external constraints such as time and/or funding restrictions, lack of detailed information, etc. the more complex, and thus more representative methods of stress sequencing should be used.

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CHAPTER 4

CALCULATION METHODS FOR FATIGUE LIFE AND CRACK PROPAGATION

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CONTENTS

4.1 INTRODUCTION

4.1.1 General Remarks

From an engineering standpoint the fatigue life of a component or structure consists of two periods:

- Crack initiation, which starts with the first load cycle and ends when a technically detectable crack is present.
- Crack propagation, which starts with a technically detectable crack and ends when the remaining cross section can no longer withstand the loads applied and fails statically.

Depending on the type of component or structure the two periods can consume widely different portions of the complete fatigue life. Usually it is assumed that massive, mildly notched components made of high-strength, brittle materials, stressed by severe load spectra at high mean stresses have a relatively short crack propagation period. For built-up structures, made of low strength, tough materials, the reverse is usually true.

In extreme cases, practically the whole fatigue life consists of crack propagation alone. This is obviously so if cracks or crack-like defects are present from the manufacturing stage onwards, as is sometimes assumed for welded joints.

In general, a calculation method for fatigue life prediction therefore would have to account for both periods; if one assumes a crack to be present from the start, the calculation method could be restricted to crack propagation alone. On the other hand, for particular components where the appearance of a crack signals imminent danger of failure, calculation of crack initiation alone would be sufficient.

4.1.2 Problems of Fatigue Life Prediction

Before discussing the various methods available at present it may be prudent to mention in general terms some of the problems facing the engineer who tries to predict fatigue life:

I. Use of specimen data

- The fatigue life is to be predicted for large components or complete structures, which may consist for example of integrally machined, single load path forgings or differentially stiffened curved shells made of thin sheet. These components are stressed by a complex mixture of stochastic and deterministic loads, including hold times, corrosion, high and low temperatures and other enviromental effects.
- The data available for the fatigue life calculation, however, have mostly been established using simple fatigue specimens loaded at best by a simplified flight-by-flight sequence without regard to environmental effects etc. If component- instead of specimen-data are employed, usually very few test results are available because of cost.

For these reasons alone it is somewhat astonishing that it has been possible to design aircraft which have in some cases reached their design fatigue life without giving too much trouble in this respect.

II. Determination of basic data

All the existing methods for the calculation of fatigue life and crack propagation require basic data which must be determined by experiments: For Miner-type calculations SN-curves are necessary for crack propagation calculations so-called "constants" (C, n and K_c for example in a Forman-type calculation). Even if these data are available with a large degree of confidence, i.e. if a large number of tests have been carried out, these basic data were certainly determined on different heats of material than those of which the aircraft in question is being built. Quite large differences in, for example, the SN-data of typical Al-alloys have been reported in the literature; the consequences for any fatigue life calculation are obvious.

III. Scatter

Any calculation method for fatigue life will at best give a statistically exact mean fatigue life. However, the fatigue life for a high probability of survival, say 99.9 percent, is required. For this the standard deviation of fatigue life is necessary which could only be reliably obtained by a large number of fatigue tests under realistic load sequences which are usually not available.

Since the scatter of fatigue life is different in constant amplitude tests than in realistic tests it is of no use in this respect.

IV. Proof of the accuracy of the prediction method

All of the methods available at present and in the foreseeable future are hypotheses. They can be shown to be correct only by experimental evidence, that, is, by a large number of realistic tests. It is much more helpful (but much more expensive) to test available methods by a judiciously selected experimental program than to add another, slightly different hypothesis to the large number already available. Only in this way we will be able to gather the evidence necessary for the correct use of any life calculation method and the experience with which to judge the results of the calculation.

The proof of any model lies in its ability to predict fatigue life of a specimen or component subjected to any arbitrary loading sequence. Therefore the test program mentioned above should contain many different, but realistic sequences, for example for

- tactical aircraft wing loads
- transport wing loads
- landing gear loads
- vertical and horizontal tail loads
- random loads etc.

V. The concept of "damage"

Any method of fatigue life calculation must make use of the concept of "fatigue damage". Although many papers have been written on this subject, fatigue damage cannot yet be described in a way which can be employed for fatigue life calculation.

It should be mentioned here, that this problem also occurs if the fatigue life is considered to consist of crack growth only. In other words: These problems are not lessened when one uses the damage tolerance philosophy, because the calculation of crack propagation under realistic load sequences also is a problem of damage accumulation. The hypotheses for the calculation of crack propagation life therefore suffer from similar deficiencies in this respect as the well known damage accumulation hypotheses.

VI. Estimation of the service fatigue loads

Although this parameter has nothing to do with the fatigue life calculation method as such the assumed load spectrum influences fatigue life at least in the same degree as the other problems mentioned above.

VII. Statistical evaluation of service load records

To obtain the load spectrum the service load records, if available, are evaluated statistically (see Chapter 3). A large number of different evaluation methods are possible,[1,2,3] and it is not all clear which method best reproduces the various "damping" portions of an irregular load sequence.

4.1.3 Requirements to be Met by a Method for the Calculation of Fatigue Life

- The accuracy and reliability of the method must be high, that is, the spread of the predicted fatigue lives around the actual ones should be small. This may need some explanaiton: The *only* criterion for a fatigue life prediction method is the spread (or scatter) of actual (test) fatigue lives around the predicted ones. "Unconservative" or "conservative" predictions are not a criterion, because, for example one can make Miner's rule always conservative by postulating $\Sigma n_i/N_i = 0.05$. This would, however, result in unnecessary overdesign for most components.
- The method should be generally applicable, that is, it should give the correct prediction for any
  - load spectrum or sequence
  - material
  - component
  - stress concentration factor or shape
  - type of loading etc.

  At least the lmits of the validity of the method should be known. For many practical applications a reliable method even for *one* type of load spectrum would be sufficient.
- The determination of the constants necessary for any calculation method must not require the tests which the method is supposed to replace, in other words: The method must not consist of just fitting some constants to fortuitous test results.
- The effort for such a calculation must be less than for a realistic fatigue test; otherwise industrial application is hompeless.
- The crack initiation *and* the crack propagation periods should be covered.

4.1.4 Available Solutions

As is well known there have been numerous attempts to meet any or all the above requirements, dating from 1924 (Palmgren) to the crack propagation hypotheses of the present decade. In fact there are so many new hypotheses almost every year, that it becomes very difficult to keep track of them and to be able to judge them on an equal basis.

All of them fall into one of the following four categories:

I. Simple methods which do not account for any of the complex events taking place in the most highly stressed area around the notch root or crack tip. Miner's rule for the crack initiation period and a simple linear addition of the crack extension due to each individual cycle using Forman's equation for the crack propagation period are examples.

II. Methods, which do account for some of the complex events taking place at the notch root or crack tip using *assumptions*, for example about the size and shape of the plastic zone and its effect on fatigue life or crack propagation. For crack propagation, the Willenborg model[4] is an example of this approach, while for crack initiation there are the ESDU-method[5] or the Impellizeri-analysis[6].

Methods 1 to 2 are based on constant amplitude tests.

III. Methods which try to measure what actually happens at the notch root or crack tip, for example the companion specimen method[7] for crack initiation or the crack closure concept[8] for crack propagation.

IV. Methods having a restricted general applicability using realistic sequence test data and assumptions on how to read across from these data to the actual load sequences.

The so-called "relative" Miner rule first defined by the author in Reference 9, but used in similar form for many years by the RAE[10], Lockheed[11] and others is an example for the crack initiation period, while for the crack propagation period there is the Wheeler model[12].

In Sections 4.2 and 4.3 a number of more or less well known methods to calculate fatigue life in the crack initiation and crack propagation phases will be discussed. Each section is divided into two sub-sctions:

- Present status,
- Potential improvements.

Methods will be classified as "present status" when they have been

- used widely in industry (not just by one firm),
- examined by systematical test programs or
- are simply outgrowths of older methods and no improvement can be expected for fundamental reasons.

They will be classified as "potential improvements" when they offer some hope for improvement compared to the present status but this has not been conclusively proved. Thus a method will be in the "potential improvements" section when it has not been thoroughly examined experimentally, although it may have been around for many years.

The extent to which the individual methods are discussed will depend on their importance. Some of these methods are so well known (e.g. Miner's rule) and have been described so extensively in the literature, that it is not considered necessary to explain their background in detail; rather results from test programs on their accuracy as well as details on how and where they are used in industry will be presented. Other methods, especially in the "potential improvements" section are not so well known and will therefore be explained in some detail. The theoretical background will not be presented, but the underlying basic assumptions will be covered.

4.2 METHODS TO CACULATE FATIGUE LIFE IN THE CRACK INITIATION PHASE

The methods described in this section have very often been used to predict the fatigue life *to failure* (crack initiation plus crack propagation); strictly speaking, however, they can only be used for prediction of the crack initiation life.

4.2.1 Present Status

4.2.1.1 Miner's Rule

4.2.1.1.1 Use of Miner's Rule in Industry

At the moment, Miner's rule[10]

$$\Sigma n_i/N_i = 1.0$$

is still the most widely used method. It is probably safe to say that *all* of the military and civilian aircraft flying today (except those in which fatigue life was not considered at all and possibly except the B-1) have been designed using Miner's rule. In the various NATO countries, there are differences in the way this method is employed, the basic (S-N) data used, etc. In the Federal Republic of Germany it is customary to use constant amplitude data in the form of Goodmann diagrams for notched specimens of the material in question, for example with a stress concentration factor of 3.6 or 2.5 (Refs 14–16), in other countries the corresponding MILHdbk 5b data, the well known NASA results or the ESDU[17] curves have been utilized. The procedure used in Germany is as follows:

Applying Miner's rule, assuming a damage sum of unity at failure and a safety factor of four, the life is calculated under the assumed load spectra, which envelops the ground to air cycle. This is considered to range from the minimum stress during taxiing to the maximum stress occurring once per (average) flight. If the required life, i.e. 4 000 hrs for a tactical aircraft, is reached or exceeded the corresponding 1–g stress is considered safe, otherwise the 1–g stress is lowered and the procedure repeated. As an added safety factor, the S-N curve is sometimes extended to a stress amplitude of zero with the slope of the finite life portion, so that *all* stresses in the spectrum are damaging.

There is a large number of implicit assumptions in this method, among them:

- The S-N curves of the structural detail in question and the notched specimen chosen are identical.
- The fatigue strength under service, e.g. flight-by-flight loading, is also identical.
- Miner's rule is valid, i.e. $\Sigma n_i/N_i = 1.0$.
- The load spectrum selected is accurate.

It goes without saying that all these assumptions are more or less incorrect and that Miner's rule is certainly not the weakest assumption. For tactical aircraft, probably the weakest assumption is the last one.

The safety factor of four mentioned above does not mean that $\Sigma n_i/N_i = 0.25$ is assumed; on the contrary, it is necessary for reasons of scatter: the SN curves used are mean or median curves; if $\Sigma n_i/N_i = 1.0$ were valid and all other assumptions correct the calculated life would still have a probability of survival of just 50 per cent; in order to reach $P_s = 99.9$ per cent, a safety factor of about 4 is needed at the standard deviation $s = 0.20$ advocated for this purpose by, for example, Lundberg and others[18,19].

4.2.1.1.2 Accuracy of Miner's Rule

Surprisingly few valid attempts* to check the accuracy of Miner's rule have been published in the literature – due to the enormous cost of such programs. Probably the first real effort was presented by Lockheed in 1962 (Ref.20) using realistic load sequences; the conclusion was that Miner's rule was at least as good, but much simpler, than any of the other methods available at that time.

During the last decade, Schijve[21], Jacoby[22], NASA[23–25], the LBF[29,30,59] and the author[26–28] carried out test programs with the following general results:

- Quite often $\Sigma n_i/N_i \approx 1.0$ was actually proved correct; however, there is a large amount of scatter in the ratios of actual predicted lives; in Reference 26, for example, the log mean of 29 random sequence test series actually was $\Sigma n_i/N_i = 1.05$ but there were individual results down to 0.3 and up to 3.0, see Figure 4.1.
- Flight-by-flight transport wing load sequences tend to give unconservative results,
  - In Reference 26, 57 test series were evaluated: The log mean was $\Sigma n_i/N_i \approx 0.6$; individual test series ranged from 0.15 to 3.0, see Figure 4.1.
  - In Reference 30, 42 test series with a standardized transport load sequence according to Reference 31 resulted in a log mean of ≈ 0.6 and individual series down to 0.025 and up to 3.0, see Figure 4.2.
- Another test series on titanium specimens with a transport load sequence[32,33] again gave ≈ 0.6 as the log mean, see Figure 4.3. If the tests were carried out at elevated temperatures, the log mean of the ratios actual/predicted life decreased to 0.32.
- For a standardized flight-by-flight load sequence of tactical aircraft ("Falstaff") the log mean was about 1.5 (Ref.29), see Figure 4.2, while individual test series ranged from 0.60 to 3.2.
- A random sequence of Gaussian type resulted in a mean $\Sigma n_i/N_i = 0.45$ at failure[28] with individual series down to 0.08 and up to 0.64, see Figure 4.4.

In the opinion of the author the main disadvantage of Miner's rule is: In any actual case it is generally not possible to predict if Miner's rule will be on the unconservative or on the conservative side. This is so because it is not known which parameters do influence the result of the prediction.

4.2.1.1.3 Improved Miner's Rules

Many authors[34–41] have attempted to improve on Miner's rule by adding the damage done by some or all of the stress amplitudes below the fatigue limit to the original damage sum of Miner. However, no improvement is to be expected from these hypotheses; one look at Figure 4.1 shows why: All of these methods add an incremental damage, however small, to the damage sum according to Miner. For a real improvement this would mean that this addition would have to be large for actual/predicted lives $\ll 1.0$, medium for < 1, nil for $= 1.0$ (and negative for > 1.0 !); in other words, the spread of the ratios actual/predicted life would have to become smaller. Obviously this will not be possible,

* Miner's rule (or any other similar method) should ideally be checked against the results of *realistic*, e.g. complex flight-by-flight or random sequences. Blocked program tests are a rather poor second choice, the well known two step or single overload tests are useless in this respect.

because the cases where actual/predicted life < 1.0 are not due to the omission of small amplitudes only, but to other unknown reasons. So a similar, but much simpler method would be to postulate $\Sigma n_i/N_i < 1.0$, say 0.3 and most of the test series shown in Figures 4.1 to 4.3 would then be conservative.

Nevertheless these methods are used, for example, by Boeing[38] and by some German firms, probably because they have one very practical advantage: The fatigue limit, which is very expensive to determine, especially with structural components, is no longer required; tests in the finite life range at two or three stress levels to fix the slope of the SN-curve are sufficient.

4.2.2 Potential Improvements

4.2.2.1 Methods Based on Local Elastoplastic Strains

In recent years, a large number of "improved" damage accumulation hypotheses based on local elastoplastic strains have appeared[3,5,6,7,42–48,57]. They are based on the assumption that the strain history at the notch root determines the life to crack initiation.

There are two problems in this approach:

– How to determine the elastoplastic strain history at the notch root?
– How to account for "damage" accumulation?

For the first problem, there are

– mathematical procedures, mainly finite element calculations[50]
– approximation procedures[51–54]

} based on assumptions about the cyclic stress-strain behaviour of the material in question

and

– experimental methods[55], i.e. measuring the actual strain by strain gage, laser, etc.

For the second problem, a linear accumulation of damage is assumed in all hypotheses known to the author. In Reference 56 a very good extract of how problems 1 and 2 are treated in some of these hypotheses was given. With the permission of the author of Reference 56, two figures of this paper are reproduced here in the form of Figures 4.5 and 4.6.

One advantage of all these methods is that (in contrast to Miner's rule) just one SN-curve for $K_t = 1.0$ and a mean stress of zero is required.

Some improvement of the scatter of the life calculation (compared to Miner's rule) has been shown[49] for the simplest of these, the ESDU I method[5] see Figure 4.7. However, only Al-alloys were examined and the load sequences used were unrealistic i.e. blocked program tests.

The ESDU has recently published a more modern but also more complex model in Item No.77004 (Ref.46a); the results of an evaluation are shown in Figure 4.8: The scatter of actual/predicted lives is still quite large; also, Item No. 77004 contains a certain amount of "fitting constants" as a number of the fatigue life calculations is done using $\Sigma n_i/N_i = 0.3$ as a failure criterion while a smaller percentage of the calculations uses $\Sigma n_i/N_i = 1.0$; when $\Sigma n_i/N_i = 1.0$ is assumed for all test series, then the left line applies (symbol ○) and the scatter is still larger.

Another of these methods was tested in Reference 45 against three different, but similar realistic random load sequences on three different steels used in the automobile industry; the results are shown in Figure 4.9, symbol ●: The log mean of the ratio actual/predicted life was about 0.9, but there was an extremely large scatter. Still another model was used in Reference 46, with the results also shown in Figure 4.9 (symbol ▲): Larger scatter, practically identical to that of Reference 45. Comparing this with Figures 4.1 to 4.3, no improvement compared to Miner's rule is apparent.

A similar investigation is reported in Reference 94, using the identical three load sequences: A similar (large) scatter of actual/predicted life was obtained for 4 different prediction methods all based on elastoplastic strains at the notch root. More results like those shown in Figures 4.8 and 4.9 and in Reference 94 are necessary to be able to judge the merits of these modern methods.

In the author's opinion, not too much improvement in the accuracy of the predictions should, however, be expected from any of these hypotheses, because they contain a large number of explicit and implicit assumptions and in the end a linear accumulation of damage is assumed.

The direct measurement of local strains at the notch root[55] avoids some of these assumptions and is therefore to be preferred. The measurement with strain gages is difficult due to the well known change in resistance of the grid wires themselves under repeated loading. There are good prospects that local notch strains can be measured by laser[58] in the near future, solving this problem.

A combination of the relative Miner rule (see paragraph 4.2.2.2) with the calculation of measurement of local strains at the notch root will probably be the optimum solution in the foreseeable future. First results, presented very recently[30], look quite encouraging.

4.2.2.2 *The Relative Miner Rule*

A large percentage of the many implicit and explicit assumptions necessary for any of the above-mentioned methods is no longer needed, if the results of *realistic*, i.e. variable amplitude tests are used as a basis. If the test spectrum is identical with the one for which the fatigue life prediction is required, no hypothesis is needed. Usually however, there will be a difference in spectrum shape and some sort of damage accumulation hypothesis must still be used.

A good example is the use of test results from standardized load sequences like "Falstaff"[29] or "Twist"[31] for the determination of fatigue allowables in the design stage. (Up to now, constant amplitude data and Miner's rule have usually been employed.) These standardized sequences have been obtained by averaging load measurements on different aircraft models of one basic type, tactical aircraft for "Falstaff", transport aircraft for "Twist". Therefore the load spectrum for the new aircraft to be designed will most probably differ more or less from the standard one and the success or failure of "Falstaff" or similar standarized sequences rests to a large degree on the availability of a damage accumulation hypothesis.

Employing Miner's rule only as a "transfer function", the life under the different spectrum can be calculated. It is not necessary in this case that the damage sum at failure be unity, as postulated by Miner, but only that the damage sum at failure be the same for the different spectra. This is the basic idea of the relative Miner Rule:

$$N_A = \frac{N_B \, (\Sigma n_i/N_i)_B}{(\Sigma n_i/N_i)_A}$$

where

N_A = fatigue life to be predicted for spectrum A
N_B = fatigue life in test under similar spectrum B
$(\Sigma n_i/N_i)_B$ = damage sum under spectrum B
$(\Sigma n_i/N_i)_A$ = damage sum under similar spectrum A.

As long as the spectra are similar this hypothesis certainly holds true; the problem boils down to the question: What, in this respect, is a similar spectrum?

The results of three relevant test programs[9,29,30] are summarized in Figure 4.10:

- The prediction is obviously quite good for the tactical aircraft load sequence[9] where two stress concentration factors and three spectra were employed. The spectra differed mainly in the region of medium-sized stress amplitudes, that is in the + 2g to + 5g range and not at all in the negative load factor range.
- The life under several modifications of the Falstaff spectrum again was predicted quite well[29], but hardly better than by Miner's rule, as can be seen by a comparison of Figures 4.2 and 4.10. The modifications consisted of a truncation to 80% of the maximum stress levels and another truncation to 38% of the minimum stress levels.
- The relative Miner rule obviously failed for the standardized transport load sequence[3], where again the influence of some modifications of the basic spectrum were investigated, among them: Truncation, omission and various minimum stress levels of the taxi loads. As might be expected, the predictions were especially inaccurate for the truncation series and for the series where the minimum stress level of the taxi loads was very low, i.e. where it went down far into compression.

From the (scarce) evidence available it therefore appears that the relative Miner rule will work for *tactical aircraft* fatigue life prediction, for example

- for the use of the standardized "Falstaff" sequence for fatigue life prediction purposes in the design stage,
- for typical spectrum variations which occur in service between different individual aircraft of one type, and
- for the typical variations between the spectrum used in the full scale test and later measured in service aircraft.

More experimental proof is certainly necessary as with all the other methods mentioned in this and the following chapter; considerable engineering judgment and a lot of experience will be necessary to avoid using this method in cases for which it is not suitable.

4.3 METHODS TO CALCULATE FATIGUE LIFE IN THE CRACK PROPAGATION PERIOD

Some of the problems of fatigue life prediction mentioned in the introduction are especially difficult to solve for the crack propagation phase. For example, the experimental verification of hypotheses is very time-consuming and expensive, because the necessary crack propagation tests with realistic load sequences are much more expensive than the corresponding fatigue tests. In addition the influence of component or specimen thickness on life seems to be larger for the crack propagation period than for the crack initiation period. This would mean that the basic and the verification data must be determined for many different material thicknesses. Despite the large engineering effort that has gone into crack propagation investigations in recent years – the ASTM alone has published at least seven books on crack propagation[60–66], it is therefore not surprising that

– a number of basic questions are unsolved,
– considerable basic data are still unavailable, and
– very few valid* attempts to check the accuracy of prediction methods are known.

Some examples: any method for prediction of crack propagation must contain two boundaries, one where the specimen fails due to static load (at K_c or K_{Ic}) and the other below which no crack proapagation occurs (ΔK_{th} or fracture mechanics fatigue limit[67]). At present there is no internationally agreed method how to determine K_c, let alone how to calculate K_c from tests at other specimen widths. As for ΔK_{th}, this depends on the stress ratio R present, and can only be determined by very time-consuming tests at different stress ratios, see paragraph 4.3.1.1.1.

Therefore the requirements for the classification into "present status" and "potential improvement" have to be relaxed somewhat; otherwise probably no method except Forman's equation for constant amplitudes could even be classified as "present status".

The methods mentioned in this chapter will not be described in detail, because an extremely thorough description can be found in another AGARD publication[86]. Also the problem of crack propagation in actual structure as opposed to flat unstiffened specimens will not be treated.

Not all the important parameters of a crack propagation calculation are sufficiently defined by simply stating that a certain method has been used. On the contrary, the result of such a calculation can depend much more on detail decisions than on the method employed. For example: Should one use the enveloping spectrum or the actual cycle-by-cycle sequence-for the calculation? This obviously influences the cost of the calculation. Another question: Are the portions of the individual cycles below zero stress to be omitted? If not, can the constants determined in constant amplitude tests at $R > 0$ be used or have new constants to be obtained by additional tests?

4.3.1 Present Status

4.3.1.1 The Forman Equation

4.3.1.1.1 For constant stress amplitude

The Forman equation[68]

$$\frac{da}{dn} = \frac{C \cdot \Delta K^n}{(1-R)K_c - \Delta K}$$

has been proved by several research laboratories to give a reasonable approximation to test results for many aircraft structural materials[69–73]. Figures 4.11 and 4.12 show some examples of results obtained at the IABG[69]. It was demonstrated by these tests that the constants C and n in the equation, determined by a few tests at one stress amplitude and mean stress can be used for other mean stresses and stress amplitudes with reasonable accuracy as long as $R \geqslant 0$. For $R < 0$, they must be derived from new tests at $R < 0$. It was also demonstrated that the portion of the stress amplitude below zero cannot be neglected for some medium strength materials (e.g. 2025-T3 or the steel 1.7734.5), while for high strength materials (e.g. 7075-T6 or maraging steel) it can; see Figure 4.13.

The original equation above does not contain ΔK_{th}. Of the two formulae[75,76] known to the author to correct for this deficiency the following gave the better fit to experimental results[74]:

$$\frac{da}{dn} = \frac{C \cdot (\Delta K^n - \Delta K_{th}{}^n)}{(1-R) \cdot K_c - \Delta K} .$$

An example is given in Figure 4.14. In any case, ΔK_{th} must be determined experimentally. In the international literature only very few data on ΔK_{th} have been published for aircraft materials[74,77,78], most of the data are for medium and low strength steels[79–84]. A very comprehensive survey of ΔK_{th} for many materials is given in Reference 85. Very recently Speidel[87] has found extremely low ΔK_{th}-values which are about half as large (!) as the values in Reference 85. He ascribes these low numbers to exacting test procedures and patience, because his tests were run down to a crack propagation rate of 10^{-8} mm/cycle.

In the Federal Republic of Germany the Forman equation is being used in its original form, i.e. without the correction for ΔK_{th} within the aircraft industry[89]. A commonly agreed computer program developed by MBB is used to determine the constants C and n from the test data. K_c is determined from the residual static strength of the specimens,

* Again, the validity of any method should be ideally checked against crack propagation tests under many different but *realistic* load sequences.

using the Feddersen concept[88]. Crack propagation under different stress ratios is calculated using the above C, n and K_c-values. This is obviously not quite correct, because K_c depends among other things on specimen width. In spite of this and other deficiencies, the author considers the Forman equation to be adequate for constant amplitude crack propagation calculation. Other formulae have appeared in large numbers[90–92], none however, has been thoroughly verified by experiments. For a comprehensive survey see Reference 95.

4.3.1.1.2 For variable stress amplitudes

The Forman equation can also be used for variable stress amplitudes by calculating the crack propagation per individual cycle, neglecting retardation by high tensile loads. This method has also been called a Miner-approach[86] because the damage increments in the form of crack growth increments are linearly added. This is the method used, for example, for the prediction of crack propagation life, inspection intervals etc. for the Panavia Tornado aircraft[93], where the calculation is carried out with the enveloping spectrum for 200 flights.

The Forman calculation has some very practical advantages:

- The calculation is simple (especially if the enveloping spectrum is used), many firms already have computer programs for the Forman equation.
- The result is usually conservative (see below).

The Forman prediction for variable loads has been checked against several test programs: the results are shown in Figures 4.17, 4.18 and 4.20:

- The IABG[96] has found in 11 test series with the 4 different realistic load sequences shown in Figures 4.15 and 4.16 that the result was always conservative with a log mean of actual/predicted life of about 4.0, see Figure 4.17. The actual cycle-by-cycle sequence was used, the constants C and n were determined by constant amplitude tests at $R > 0$, the portions of the cycles below zero stress where *not* omitted, ΔK_{th} was considered to be zero.

- MBB[97] have checked 10 test series with 4 different load sequences and 4 different materials, see Figure 4.18. They also found most of the calculated values to be conservative, with a log mean actual/predicted life of about 3.0 and a rather large scatter. The one unconservative test series was for a load sequence with many cycles in compression, shown in Figure 4.19. Crack propagation was again calculated cycle-by-cycle. C and n were taken from constant amplitude tests at $R > 0$; no omission of cycles below zero stress; ΔK_{th} was taken to be zero.

- Nelson and Fuchs[98] calculated crack propagation under the three load sequences also used in their[94] and Socie's[46] work on prediction of crack initiation life. They used two different values of ΔK_{th}; the constants C and n were determined at $R = 0$; compressive loads were assumed not to cause crack growth. The predictions are not directly comparable to those of MBB and the IABG mentioned above because of this assumption. The results are shown in Figure 4.20:
  - The influence of ΔK_{th} was quite small.
  - Most of the predictions are again conservative; however, there is a large percentage of unconservative results down to actual/predicted lives below 0.1 and the scatter is rather extreme.

  The highly conservative prediction occurred for a load sequence which fluctuated around zero stress and at very long lives, i.e. at low stress amplitudes. The highly unconservative results were obtained for both steels investigated at very short crack propagation lives for a load sequence which is for the most part in compression. This probably means that the assumption "compressive loads cause no crack growth" is not correct. The result of the prediction would certainly have been beter if the compressive part of the cycles had not been omitted.

- Some recent results of MBB[99] also point in this direction: Under constant amplitude loading there was no influence on crack propagation, whether $R = 0$ or $R < 0$ was used for the same maximum stress. A realistic load sequence with a large percentage of the cycles going partly into compression was then tested. A Forman calculation in which all these parts in compression were omitted was unconservative by a factor of 6!

Both of these results can probably be explained at least qualitatively by crack closure (see paragraph 4.3.2.6).

Therefore it is necessary to account for the compressive parts of realistic load sequences in the Forman calculation even when they have no influence under constant load amplitudes.

It is interesting to compare the predictions for the crack initiation phase in Figure 4.9 with those of the crack propagation phase in Figure 4.20, because they were for identical load sequences and materials: Both predictions gave an extremely large scatter. However, it is easier to understand and explain some of the very far-off predictions in the case of crack propagation.

Definitive conclusions on the Forman model are certainly not yet possible. For limited applications and with considerable experience it is certainly not worse than any of the other modesl described below, but much simpler.

4.3.1.2 *The RMS – Method*

The first use of this method[103] characteristically came from acoustical fatigue tests where the rms-value of the stress amplitudes applied plays a considerable role. The basic idea was to find a constant stress amplitude ΔK_{equ} which is equivalent (with regard to crack propagation) to the typically variable realistic stress amplitudes. The root mean square (rms) of the realistic sequence was thought to be that equivalent. It is clear that no interaction effect can be accounted for in this way. For example, rare high stress peaks will not influence the rms-value appreciably, but they do influence crack growth. No *valid* check on this approach is known to the author, the few results[105,106] available do not fulfill the necessary requirements. As early as 1960, Paris dropped the idea of an equivalent constant amplitude ΔK_{equ} and proposed a correction factor B (Ref.107) to be determined in realistic load sequences. In effect this is a relative damage accumulation hypothesis similar to the Wheeler model or the relative Miner rule. Paris himself[107] and some others[104] found a quite good correlation between tests and predictions. However, for flight-by-flight load sequences there was a large difference between reality and prediction[108].

4.3.2 Potential Improvements

4.3.2.1 *The Willenborg Model*

The Willenborg model[4] was developed by the USAF to account for interaction effects in realistic load sequences. By the way the USAF in its relevant Military Specification[100] expressly requires "interaction effects, such as variable loading and environment, shall be accounted for".

For a comprehensive description of the Willenborg model see References 86, 101, 102. The basic idea is:
– A stress peak develops a plastic zone in front of the crack tip.
– Crack growth is retarded as long as the crack tip is in that plastic zone.

In effect this means that the effective mean stress or stress ratio R is reduced for that period and, therefore, growth is retarded. The Forman equation is used, all stresses below zero are to be omitted in the calculation. The advantage of the Willenborg model is that no special test data are required: The constants C and N can be taken from constant amplitude tests; the yield strength σ_y is also necessary for the calculation of the size of the plastic zone.

To the author's knowledge only two valid test programs[96,97] are available to check the Willenborg model; their results are shown in Figures 4.17 and 4.18: Most predictions are unconservative, scatter is large especially for the MBB data (Fig.4.18).

For the IABG data three different load sequences (tactical aircraft, landing gear and F-104, see Figures 4.15 and 4.16) and four different materials were used in the 11 test series; the one conservative value was for the landing gear sequence and 2024-T3; the most unconservative results were for the tactical aircraft sequence and a maraging steel. C and n were determined from constant amplitude at $R > 0$. Crack propagation was calculated cycle-by-cycle; compressive stresses were omitted.

The MBB results were extremely conservative for the spectrum shown in Figure 4.19 which contains a large number of cycles in compression and for the material 7075-T351. This is not surprising since MBB omitted the stresses below zero. However, the two conservative points in Figure 4.18 are for a horizontal tail load spectrum also containing a large number of compressive cycles.

Thus the picture on the Willenborg model is somewhat hazy at the moment; it is certainly no better than the Forman model; the computer time is about twice as high as for the Forman model if a cycle-by-cycle calculation procedure is used.

4.3.2.2 *The Wheeler Model*

The Wheeler model[12] is a relative damage accumulation hypothesis, because tests under a realistic load sequence are used to determine a factor which is used to fit the prediction to the actual crack propagation life. Gray and Gallagher[109] call the original Wheeler retardation model "more of a data fitting technique than a predictive technique".

However, when it can be proved that the same factor can be used for different load sequences it is "a predictive technique", much like the Paris rms approach described in paragraph 4.3.1.2 or like the relative Miner rule. Originally Wheeler used the simple Paris equation

$$\frac{da}{dn} = C \cdot \Delta K^n$$

– which accounts neither for ΔK_{th}, nor for specimen failure at K_c, nor for stress ratio effects – and constant amplitude tests to determine C and n. Wheeler also used blocked program tests to find the data fitting factor mentioned above.

So the first improvement would be to use Forman's equation and realistic load sequences. This has been done by MBB and IABG[110,111], while the third comprehensive check known to the author, the one by Broek[112] at least incorporated complex blocked flight-by-flight sequences and Forman's equation. The IABG check results were obtained as follows:

- C and n were determined from constant amplitude tests at $R > 0$.
- The data fitting factor m was taken from Broek[112] for
  - Ti6Al4V as $m = 1.8$ and for
  - Al-alloys 2024-T3 and 7075-T7351 as $m = 1.4$.
- For maraging steel the factor was obtained by data-fitting as $m = 0.65$.
- The original cycle-by-cycle sequence was used, compressive stresses were *not* omitted.

The results of the check of Wheeler's model for the four different materials above and the four different load sequence shown in Figures 4.15 and 4.16 as well as the "Falstaff" sequence are given in Figure 4.17: A large percentage of the ratio actual/predicted lives is near 1.0; the most unconservative result is on 7075-T7351 for a truncated tactical aircraft load sequence. The results of Broek[112] are also shown in Figure 4.17: All predictions lie within ±30 per cent of the actual lives.

The IABG results look quite convincing, especially considering that the data were obtained for three materials using Broek's factors m and for four different sequences. Broek's very good results can be explained by the fact that he used only one basic spectrum type for a tactical aircraft; he then mainly modified this spectrum in the low-g range.

MBB used the same approach as the IABG and determined the factor m by data-fitting to the test results for the tactical aircraft load sequence shown in Figure 4.16; the numerical value of m used was

- 0.9 for 7075-T7351,
- 1.8 for 300 M,
- 1.25 for HP9-4-30, and
- 1.95 for Ti6Al4V.

Results are shown in Figure 4.18: Four predictions are exact (where m was found by data fitting); there is one highly uncoservative prediction (again for the spectrum shown in Figure 4.19 with a large percentage of compressive stresses). However, the highly conservative value was obtained for a horizontal tail spectrum also incorporating large numbers of stress cycles in compression.

It is interesting to compare the Willenborg-, the Wheeler- and the Forman-model predictions in Figure 4.18, because at least the most unconservative and the most conservative results shown apply to the same load sequences and materials.

As for the other crack growth prediction models, discussed in this chapter the data above do not allow a definitive verdict. However, it appears, as in the case of the relative Miner rule, that Wheeler's model will predict the crack propagation life quite well in the following circumstances:

- For the use of standardized load sequences for crack propagation prediction in the design stage.
- For typical spectrum variations in service between different aircraft of one type.
- For the typical variations between the full scale fatigue test spectrum and actual usage in the fleet.

Considerable engineering judgment is certainly necessary, especially if the Wheeler model is to be used for other purposes than mentioned above.

4.3.2.3 The Relative Forman Model

According to a suggestion of MBB[99] a much simpler but at least as good model as the Wheeler-model would be to use the result of realistic crack propagation tests, do a Forman calculation and then use the factor actual/prediction life as a correction factor for the prediction of crack growth under different, but similar spectra. As in the case of the relative Miner rule the question "what is a similar spectrum" can only be answered by experience. A few of the test series shown in Figures 4.18 and 4.19 can be analyzed in this way: On the assumption that a new correction factor has to be obtained by test for every material, the results are very good with the exception of the spectrum of Figure 4.19 and of a transport spectrum. So the same reservations apply to this model as to the Wheeler model or the relative Miner rule.

4.3.2.4 The Habibie Model

The Habibie model[113,114] also is a relative damage accumulation hypothesis; it requires realistic test data; in comparison with, for example, the Wheeler model, it in addition expressly accounts for retardation effects. In principle, therefore it should be superior to the other models explained so far. However, it is quite difficult to handle and requires a large computer effort. Therefore, it has not been checked extensively enough against test results to allow definitive conclusions. It has been compared with Schijve's wellknown data[115] on 2024-T3 and 7075-T6 and similar tests of the IABG on Ti6Al4V (Ref.114). In both programs, a flight-by-flight load sequence for a transport wing was used, and many variations like truncation, omission at several maximum stress levels, omission of the g.t.a. cycle etc. were investigated.

The prediction with the Habibie model was very good. However, when a tactical aircraft sequence was employed[117] it became clear that another retardation factor was necessary, which had again to be determined experimentally. So, as with all the other relative damage accumulation hypotheses, it requires experience and engineering judgement to safely apply the Habibie model.

4.3.2.5 The Nelson and Fuchs Model

Nelson and Fuchs, in addition to their check on the Forman model mentioned in paragraph 4.3.1.1.2 developed their own model[98]. It is discussed here only because they also compared it with the identical test results that they used for the check on the Forman model. Thus a direct, unambiguous comparison of both models is possible – a very rare occasion! The model uses an effective ΔK_{eff}

"to account for the following possible load sequence effects:

(a) crack retardation,
(b) the decrease of retardation by sufficiently large compressive overloads, and
(c) the acclerations of crack growth rate by cross yielding in compression (but not in tension)".

In both prediction models, it was assumed that compressive loads cause no growth.

The result is shown in Figure 4.21 and should be compared to Figure 4.20: Except for the highly unconservative predictions, which are somewhat better, but still far too much on the unconservative side, all other predictions are practically equal. This shows, as mentioned before, the obvious difficulty of finding a prediction model which is better than the simplest one (be it Forman or Miner) if one uses constant amplitude data as a basis.

4.3.2.6 Crack Closure

As early as 1968, Elber[8,118] found that following a high stress peak the crack *did* close before the load applied became zero and, when smaller amplitudes followed, that it did not open as soon as the load was above zero. Therefore he defined an effective stress range $\Delta\sigma_{eff}$, an effective stress ratio $U = \dfrac{\Delta\sigma_{eff}}{\Delta\sigma}$ and a corresponding effective stress intensity factor range ΔK_{eff} resulting in a crack growth "law" of the form

$$\frac{da}{dN} = C \cdot (\Delta K_{eff})^n .$$

For a comprehensive description see Reference 128.

An impressive amount of scientific and engineering effort has been spent in the meantime on crack closure[119–127]. Shih and Wei[119] showed that the effective stress ratio U was not constant, but depended on R and σ_{max}. Others[121] found that in thick specimens where plane strain conditons prevail in the interior the crack had already closed on the outside, but was still open inside the specimen so that crack closure was neither unambiguous nor could be measured reliably; U also depended on the crack depth, the stress amplitude, the type of loading (bending or tension), the environment and the material.

In one practical application of crack closure by Grumman[122] it is assumed that U decreases exponentially for a high-low sequence and that it increases linearly for a low-high sequence.

These many assumptions point out, in the author's opinion, the real difficulties to be surmounted before the crack closure model can be used with any assurance. To the author's knowledge there is only one attempt to check the crack closure model[129]. In it, Elber proposes an "equivalent constant amplitude concept" based on crack closure and compares the predicted results with the actual ones.

The equivalent constant amplitude is supposed to have the same maximum stress as the random sequence; the minimum stress of the constant amplitudes is to be the crack opening stress in the random sequence, which has to be measured. The equivalent number of cycles at constant amplitude also has to be assumed.

The tests were carried out under an extremely short pseudo-random sequence delivered by a noise generator. The predicted lives were within ± 20% of the actual ones. This test program certainly does not fulfil the requirements stated earlier and it would have been interesting to see predictions with other models, for example the rms-method.

The author agrees with Schijve[128] that a scientifically acceptable model for crack growth prediction under realistic load sequences must incorporate crack closure (and *many* other parameters). But any optimism on the early availability of such a model is certainly unfounded.

4.4 CONCLUSIONS

It will be very difficult to improve on the accuracy of Miner's rule and for Forman's equation for the prediction of fatigue life and crack propagation, if constant amplitude data (SN-curves) are used as a basis. None of the many new models proposed in recent years have reliably shown convincing improvements compared with Miner's rule or the Forman model.

A better approach, in the author's opinion, is to use data from realistic tests as a basis and predict the fatigue life and crack propagation employing a "relative" damage accumulation hypothesis.

However, in real design applications one should always bear in mind the large number of explicit and implicit assumptions contained in *any* of the prediction models described in this chapter. In other words: For the present and for the foreseeable future there is no way around realistic tests with real structures and components!

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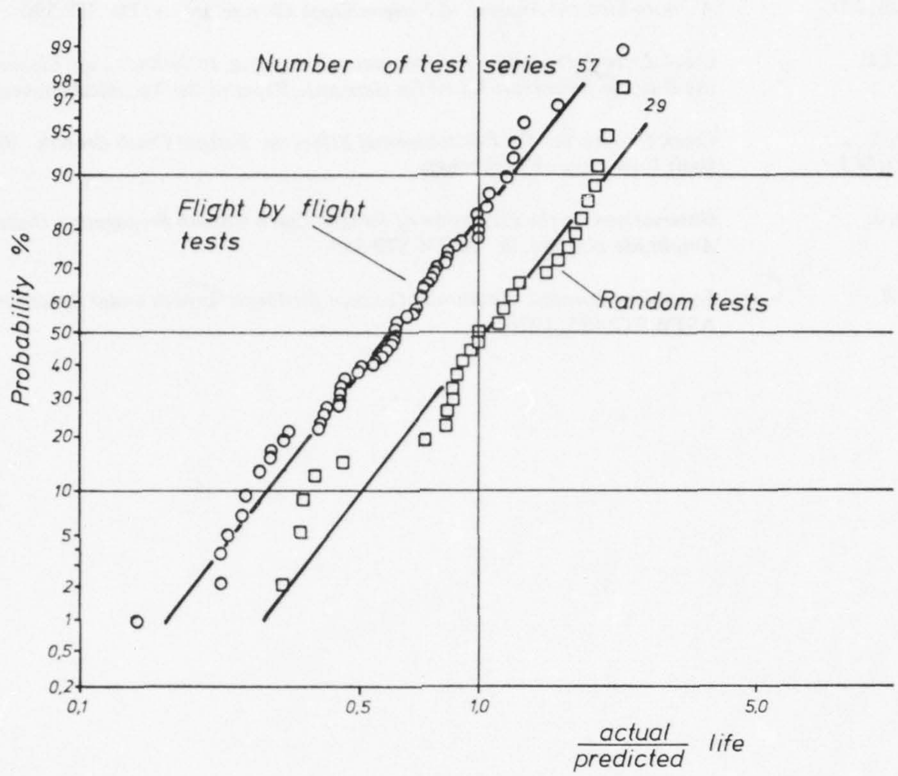


Figure 4.1

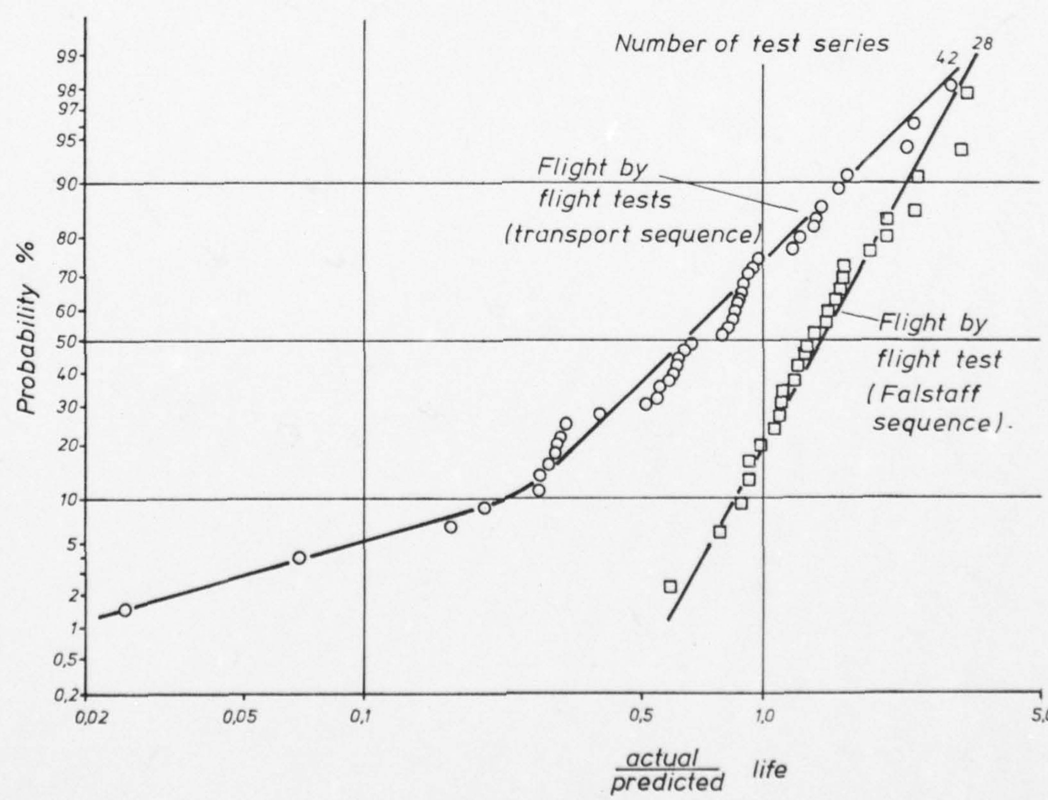
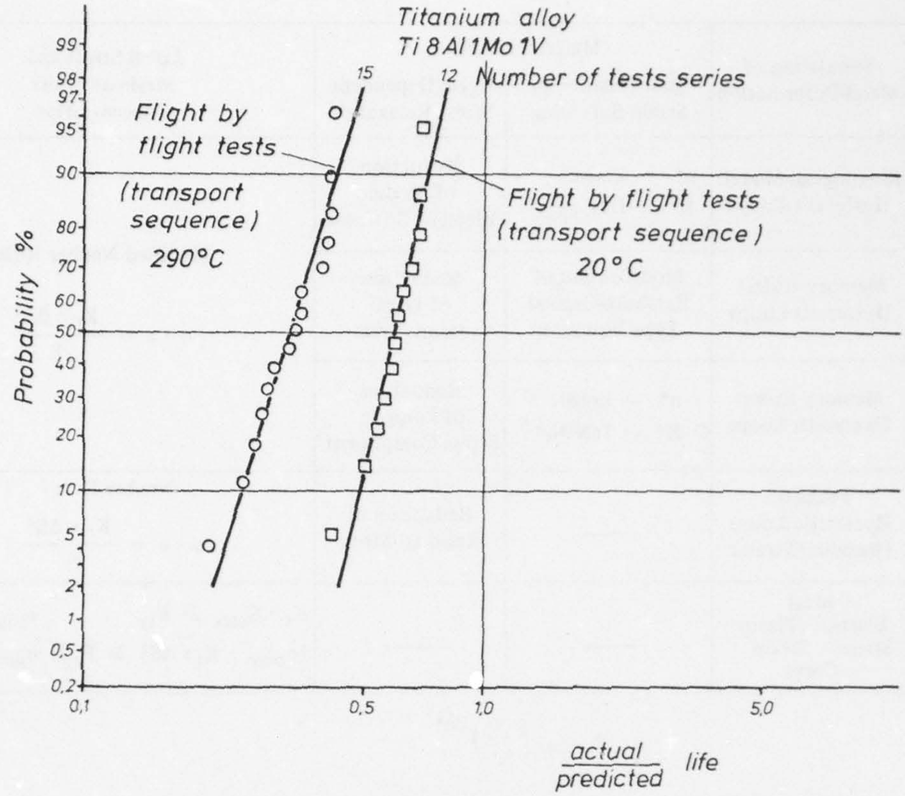


Figure 4.2



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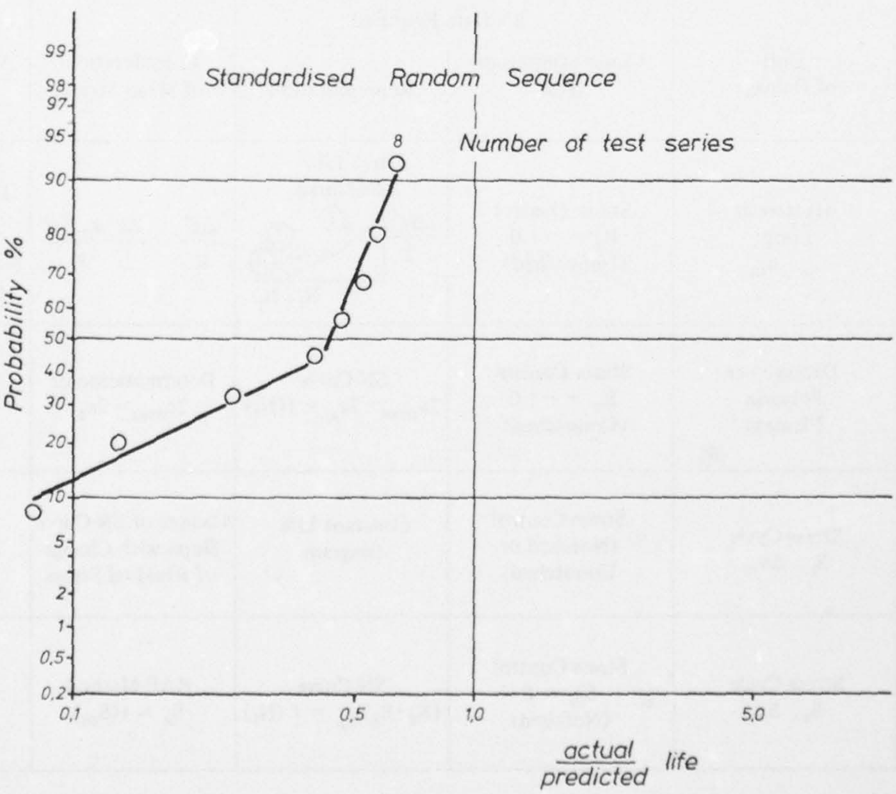


Figure 4.4

Author	Simulation of Metal-Deformation	Material Behavior: Strain Hardening Strain Softening	Material Behavior: Cycle Dependent Stress Relaxation	Local Stress and Strain at Stress Concentration
Martin, Topper Sinclair	Rheological Model: Hysteresis Loops	$n^{\bullet} = $ konst. $K^{\bullet} = f(N, \epsilon_{pa})^{\oplus}$	Reduction of Model Element Stiffness	Modified Neuber Rule $\sigma \cdot \epsilon = \frac{K_f \cdot \Delta S^2}{E}$
Jhansale, Topper	Memory Rules: Hysteresis Loops	Modification of Ramberg-Osgood Type Equation	Reduction of Local Mean Stress	
Wetzel	Memory Rules: Hysteresis Loops	$n^{\bullet} = $ konst. $K^{\bullet} = f(N/N_f)^{\oplus}$	Reduction of Polygon Stress Component	
Impellizeri	Peaks of Hysteresis Loops (Residual Stress)	——	Reduction of Residual Stress	Neuber Rule $\sigma \cdot \epsilon = \frac{K_t \cdot \Delta S^2}{E}$
RAE	Ideal Elastic – Plastic Stress – Strain – Curve	——	——	$K_t \cdot S_{max} \geqslant F_{ty}$: $\sigma_{max} = F_{ty}$; $[\sigma_{max} - K_t \cdot \Delta S] \geqslant F_{cy}$: $\sigma_{min} = F_{cy}$

$$\oplus \ \epsilon_{pa} \left(\frac{\sigma}{K^{\bullet}}\right)^{1/n^{\bullet}}$$

Fig.4.5 Simulation of local stress-strain behavior (taken from Ref.56)

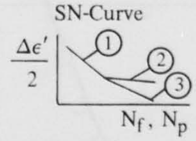
Author	Unit of Damage	SN Data Required: Const. Amplitude Test	SN Data Required: Representation	Consideration of Mean Stress	Accumulation
Martin, Topper, Sinclair / Jhansale, Topper	Hysteresis Loop: ϵ_a, σ_m	Strain Control $R_\epsilon = -1.0$ (Unnotched)	Three Line – SN-Curve 	$\frac{\Delta\epsilon'}{2} = \frac{\Delta\epsilon}{2} \frac{\sigma_m{}^{0.73}}{E}$	Linear: Initiation 2 + 3; Propagation 1 + 3
Wetzel	Damage per Polygon Element	Strain Control $R_\epsilon = -1.0$ (Unnotched)	SN-Curve $2\sigma_{max} \cdot 2\epsilon_a = f(N_f)$	Determination of $2\sigma_{max} \cdot 2\epsilon_a$	Linear
Impellizeri	Stress Cycle S_r, $\Delta\sigma_m$	Stress Control (Notched or Unnotched)	Constant Life Diagram	Change of SN-Curve Slope with Change of Residual Stress	Linear
RAE	Stress Cycle S_a, S_m	Stress Control $S_m = 0$ (Notched)	SN-Curve $(K_t \cdot S_a)_{S_m} = f(N_f)$	RAE-Method $S_a = f(S_m)$	Linear

Fig.4.6 Damage calculation (taken from Ref.56)

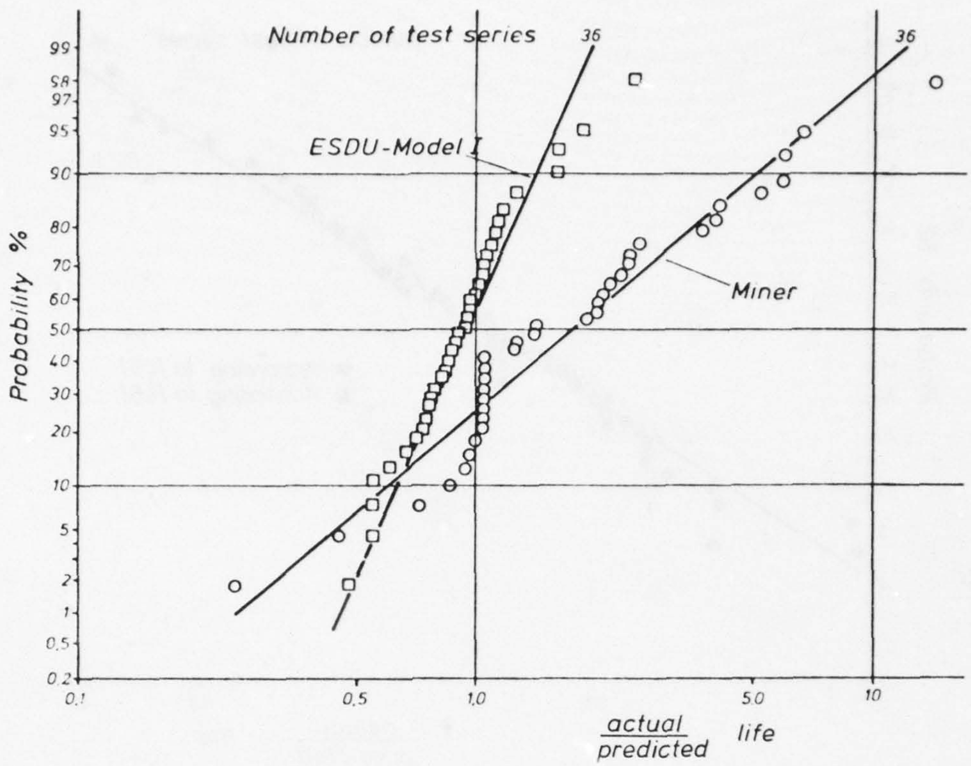
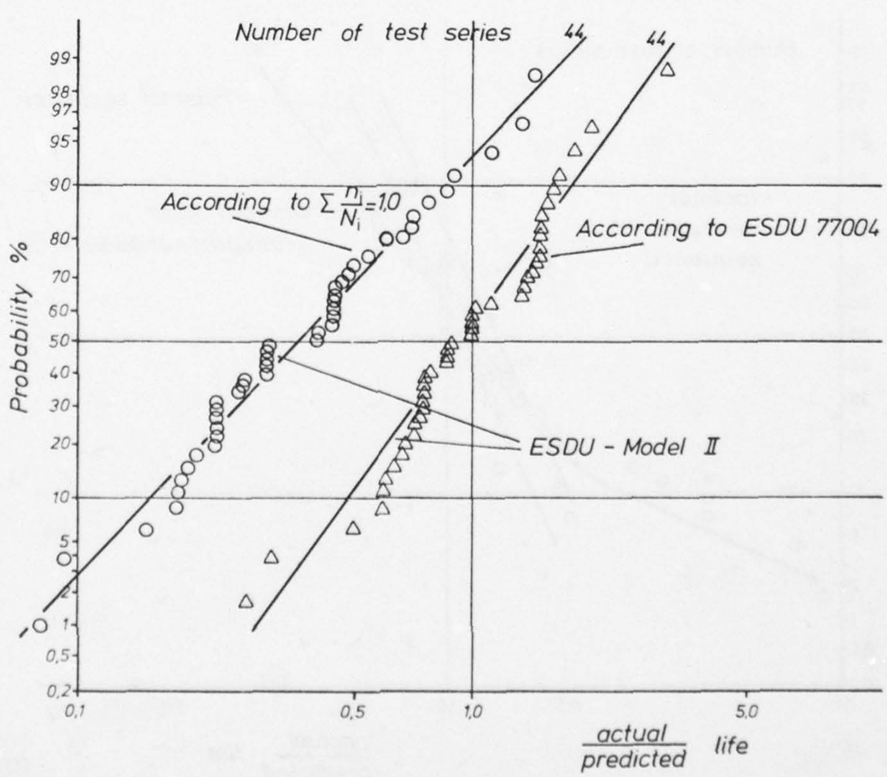


Figure 4.7



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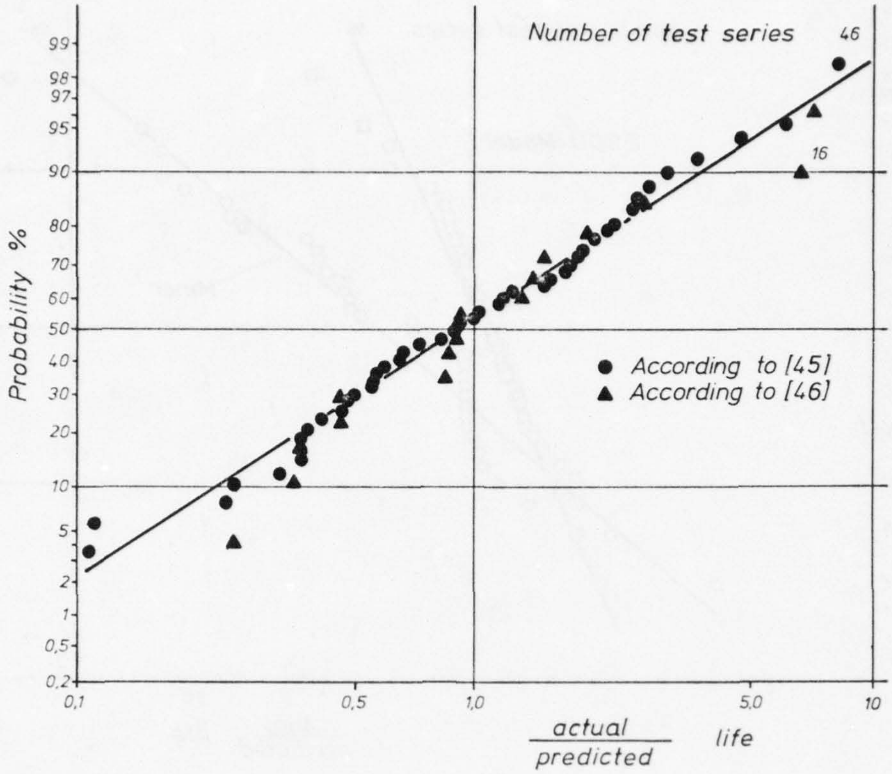

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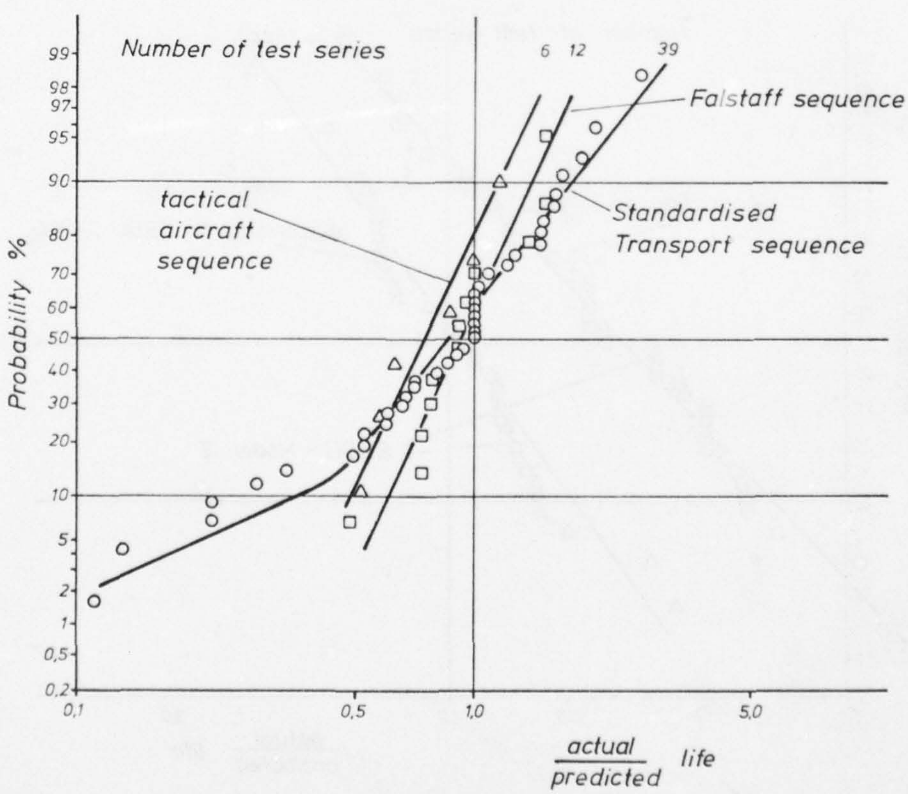


Figure 4.10

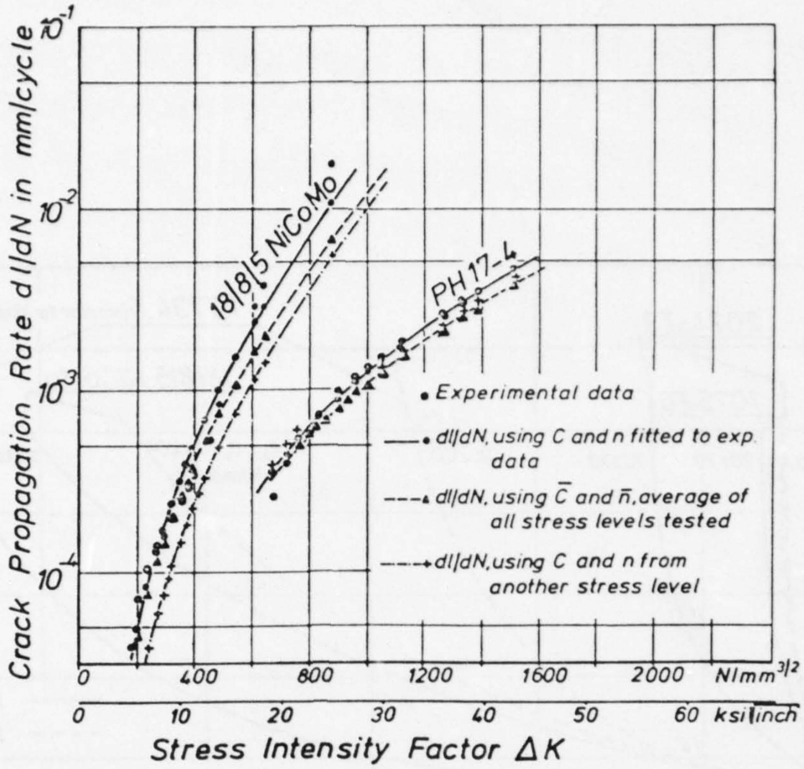


Figure 4.11

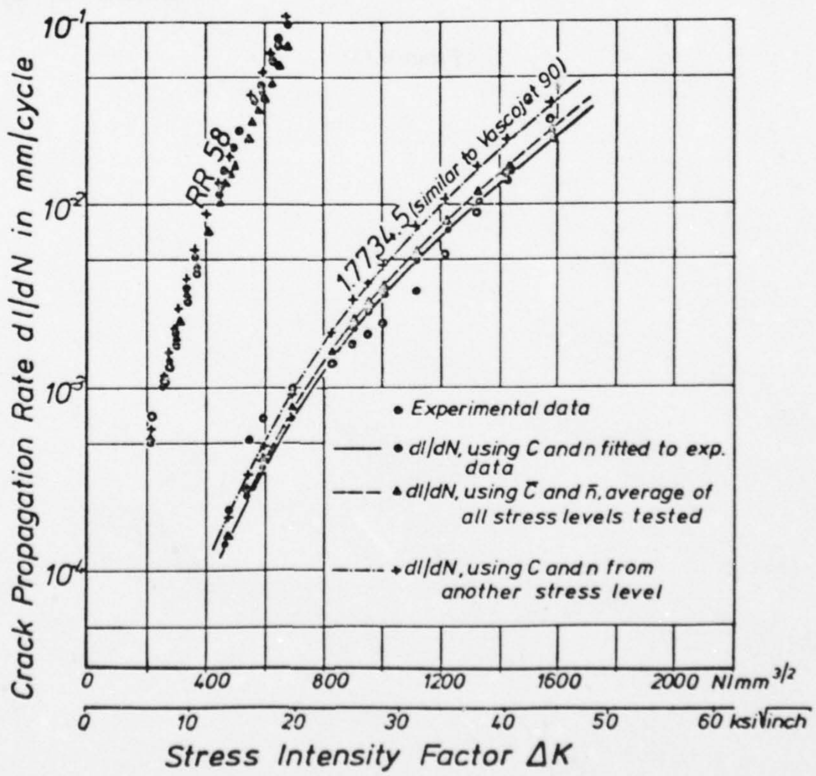


Figure 4.12

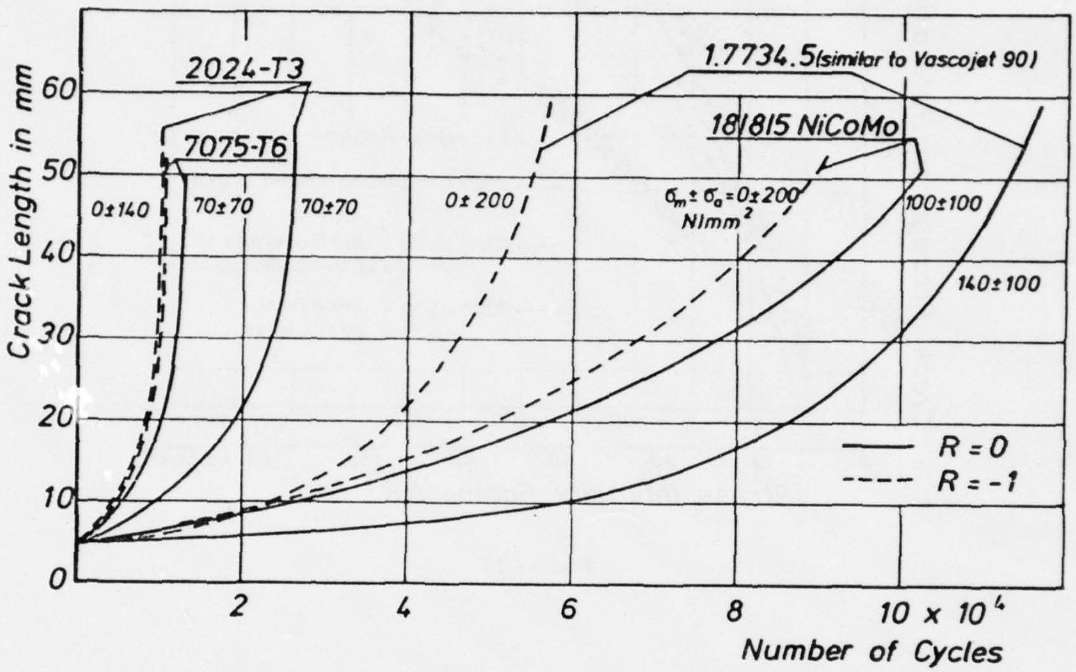
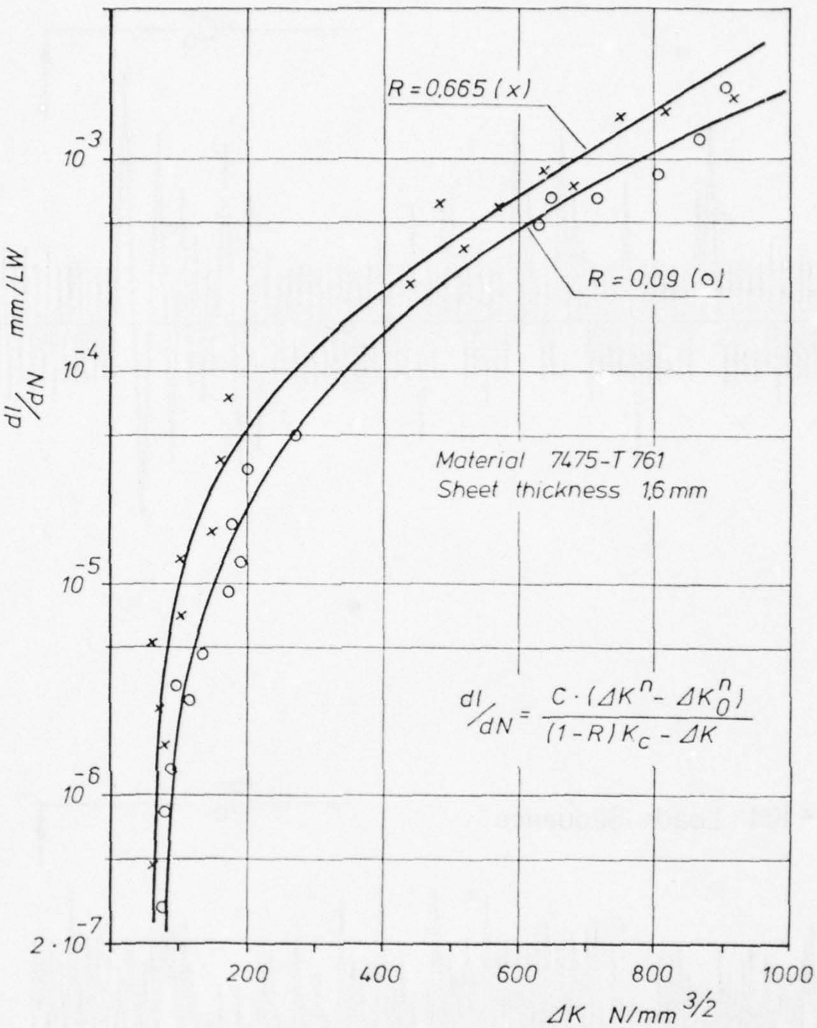

2024-T3
7075-T6
1.7734.5 (similar to Vascojet 90)
18/8/15 NiCoMo
0±140
70±70
70±70
0±200
$\sigma_m \pm \sigma_a = 0 \pm 200$ N/mm²
100±100
140±100
R = 0
R = -1
Crack Length in mm
0
10
20
30
40
50
60
2
4
6
8
10 x 10⁴
Number of Cycles

Figure 4.13



R = 0,665 (x)
R = 0,09 (o)
10^{-3}
10^{-4}
10^{-5}
10^{-6}
$2 \cdot 10^{-7}$
dl/dN mm/LW
Material 7475-T 761
Sheet thickness 1,6 mm
$\frac{dl}{dN} = \frac{C \cdot (\Delta K^n - \Delta K_0^n)}{(1-R) K_c - \Delta K}$
200
400
600
800
1000
ΔK N/mm^3/2

Figure 4.14

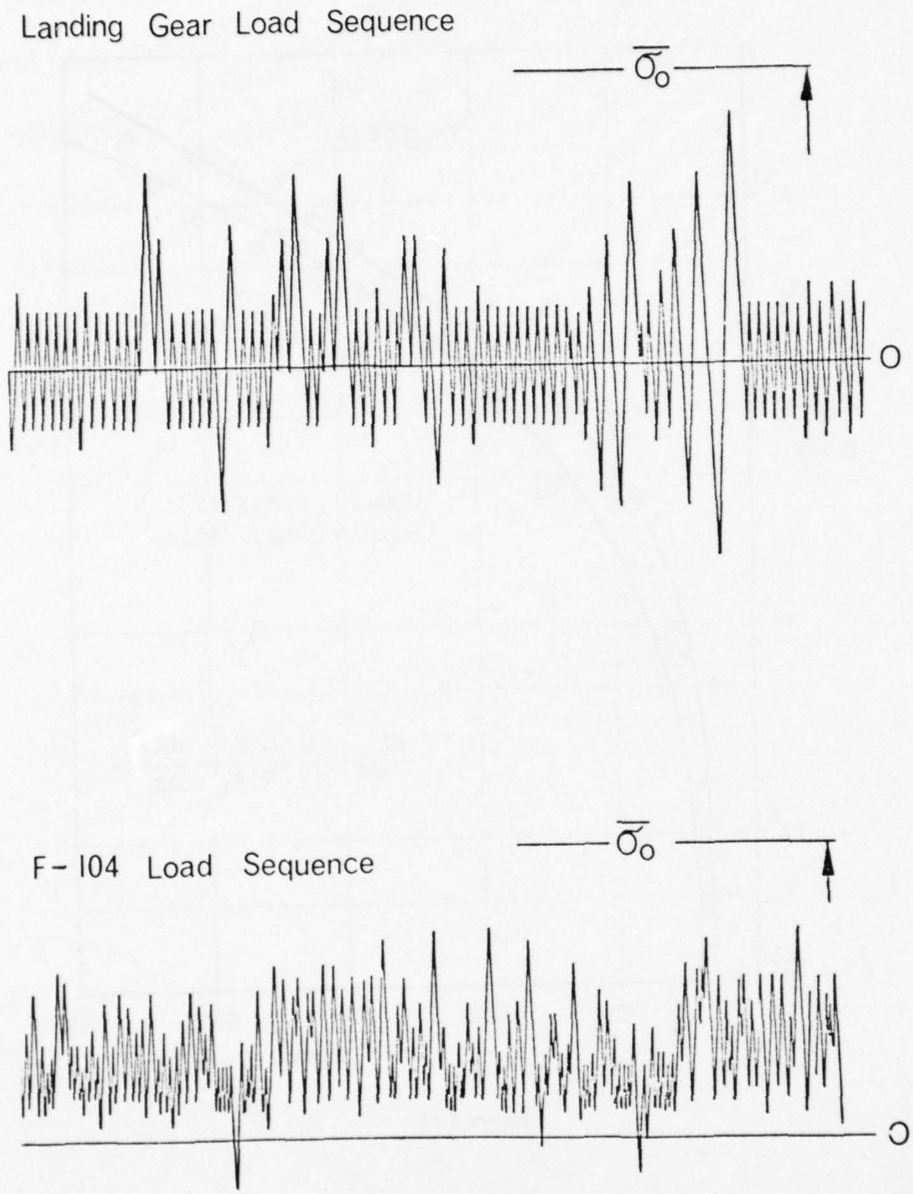
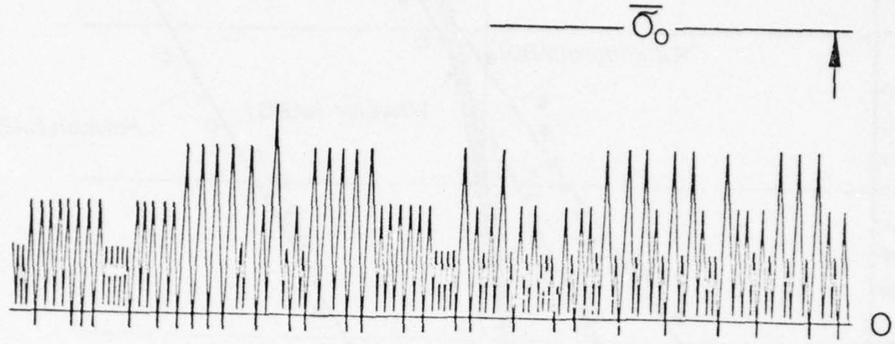
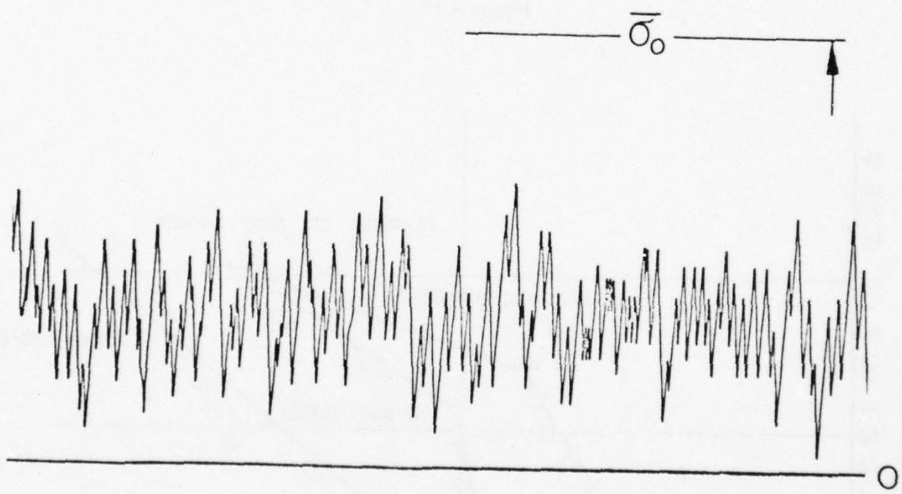

Landing Gear Load Sequence
$\overline{\sigma}_0$
O
F-104 Load Sequence
$\overline{\sigma}_0$
O

Figure 4.15

Tactical Aircraft Load Sequence



Random Load Sequence



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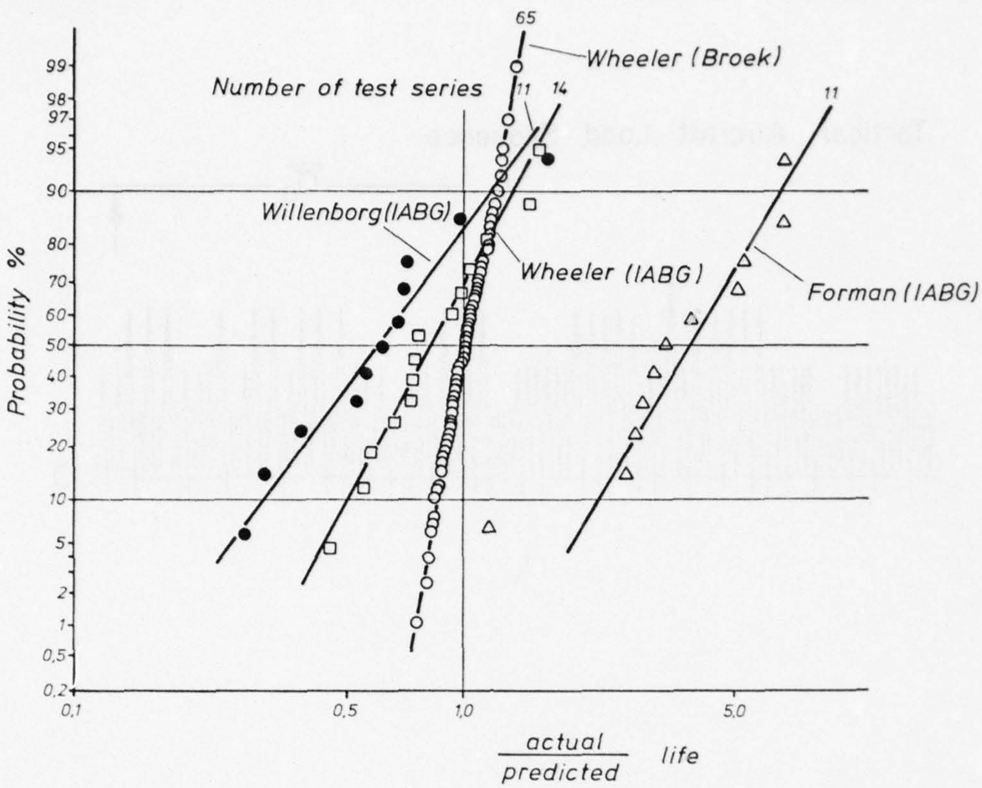

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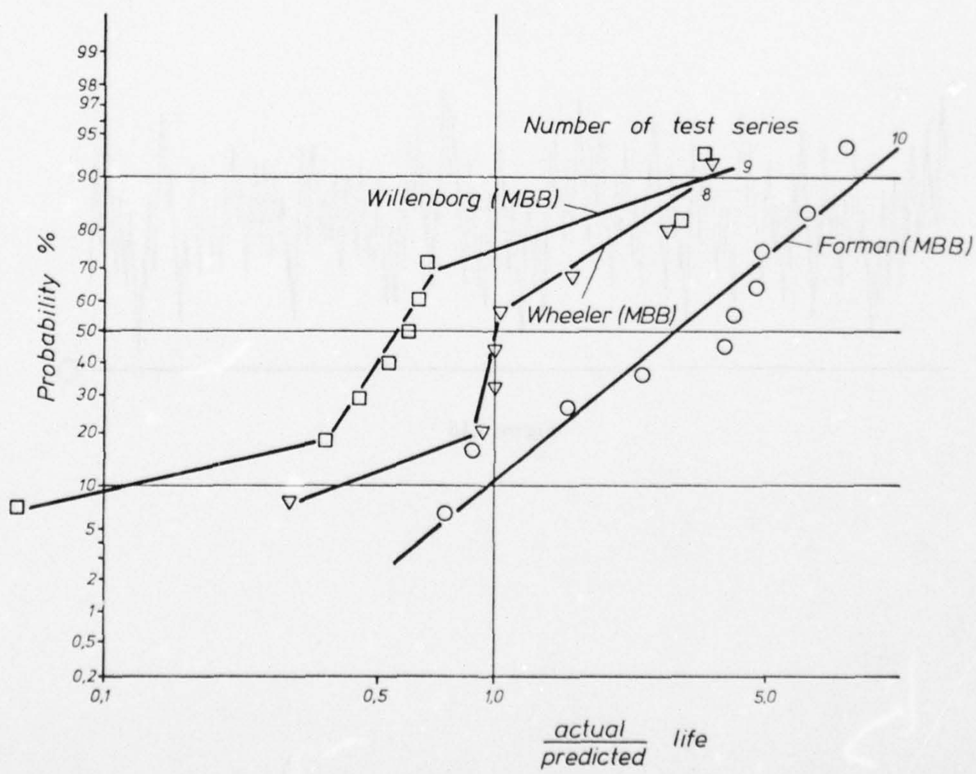


Figure 4.18

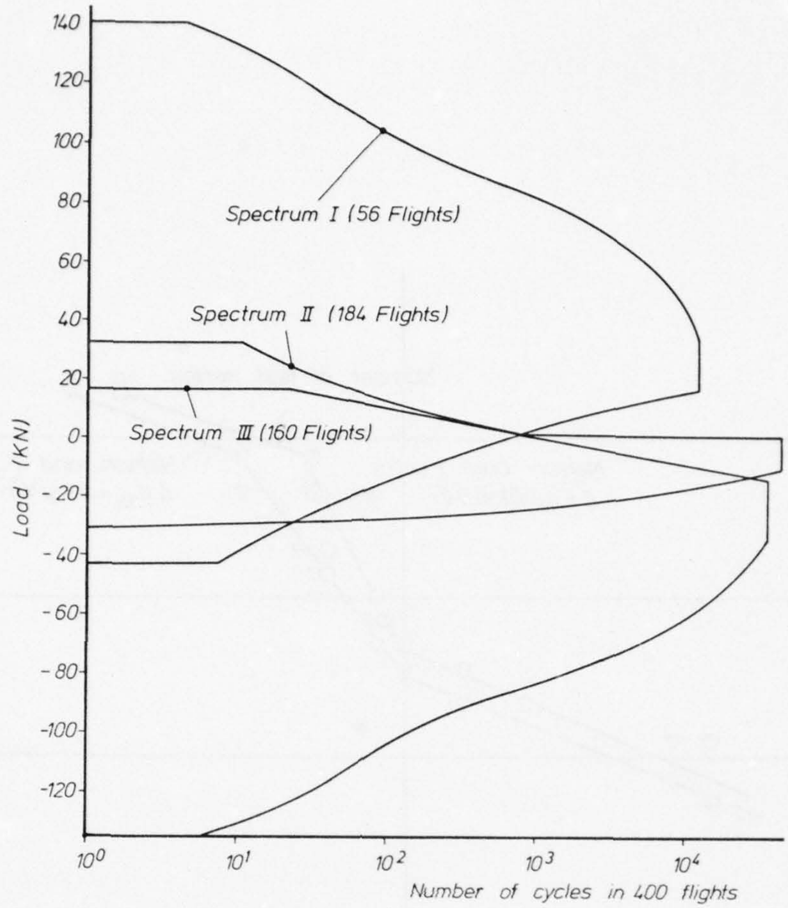


Figure 4.19

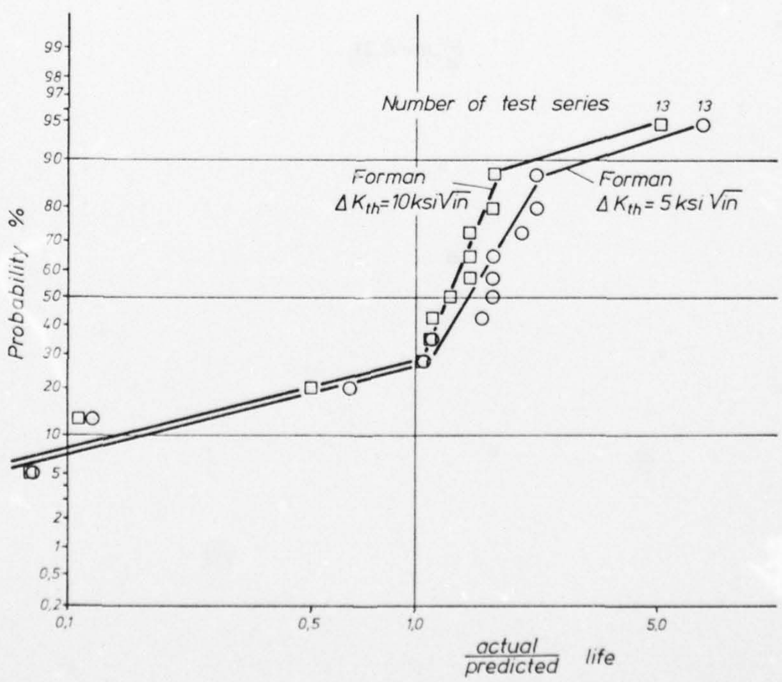
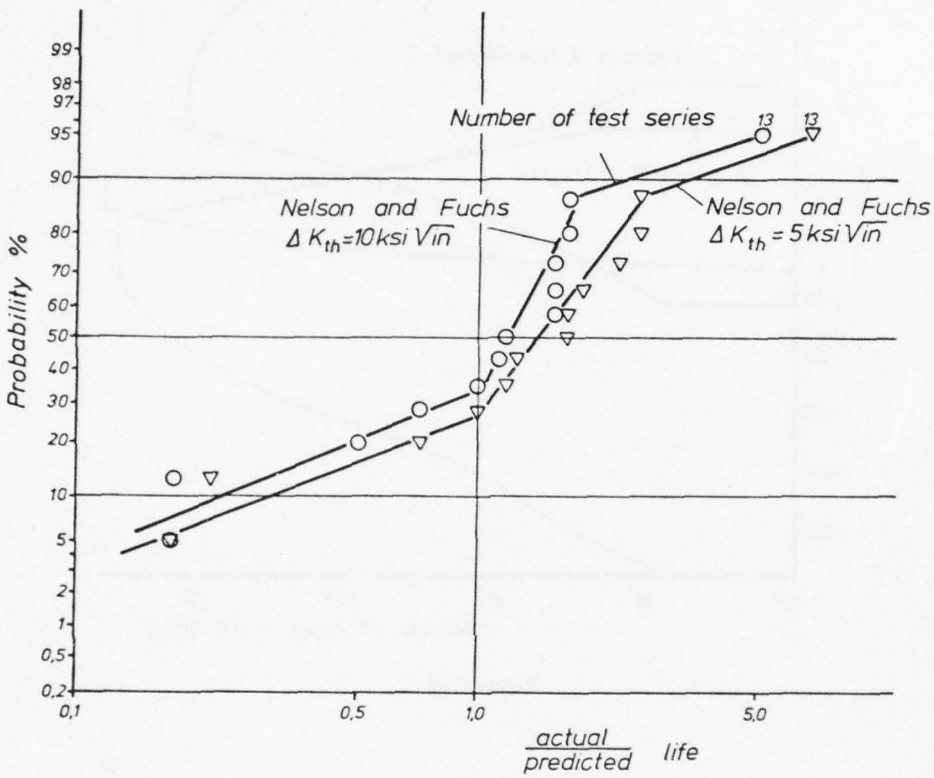


Figure 4.20



Probability %
99
98
97
95
90
80
70
60
50
40
30
20
10
5
2
1
0,5
0,2
0,1
0,5
1,0
5,0
actual / predicted life
Number of test series
13
13
Nelson and Fuchs
ΔK_{th}=10 ksi $\sqrt{in}$
Nelson and Fuchs
ΔK_{th} = 5 ksi $\sqrt{in}$

Figure 4.21

CHAPTER 5

TESTS ON DETAILS AND COMPONENTS

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CONTENTS

5.1 INTRODUCTION

This chapter will give a brief survey of the techniques to be used in detail and component tests. Since most of the conventional techniques are well-known, discussions will be limited to special techniques, and those points that have to be kept in mind if reliable results are to be obtained.

Usually, component tests are conducted from the start of a weapon system definition phase through its service phase. Table 5.1.1 is a graphical display of the activities covered by the term "Detail and Component Testing", shown in correlation to the time consumed from the development through the service phase of a weapon system.

During the project and definition phases, fatigue tests of specimens are necessary whenever:

A. new materials, or materials produced by new or modified manufacturing techniques, are used;

B. new manufacturing techniques are applied to structural components, or even more, to joining such components (e.g. EB-welding, new fasteners for critical joints);

C. the required usage of the weapon system will lead to loads or load sequences that are not covered by past experience, or whose theoretical analysis may produce unreliable results. This will also apply, if additional influences have to be considered, such as temperature effects.

Such tests will sometimes become necessary in order to establish – even in the definition phase – a firm structural layout including the choice of material, the type of structure, structural joints etc. If the fatigue layout is expected to have a major influence on the mass balance of the aircraft, well-found stress levels will already have to be available at this time. Sometimes even major component tests will be necessary in the definition phase, if new design principles and structural technologies are to be applied. Examples for such early investigations include:

D. the testing of the pivot area of sweep-wing aircraft to establish the data for the bearing pressure, necessary bearing diameter, and trade-off studies for complex and critical areas of the structure.

These activities are continued and intensified during the *development phase*. In addition, the fatigue life of critical components and/or the reaction of their structures to cracks (crack propagation, residual strength) must be determined to serve as a basis for fatigue life calculation and the final layout of fatigue-critical components. Such fatigue tests are often combined with detailed stress analyses. Also to be carried out in this phase are specimen tests to determine the crack propagation rates for the materials used and the crack growth in critical areas. This is required especially for all structures that have to be laid out strictly conforming with the damage-tolerance criteria of the US-Military Specification[1,2,3].

Specimen tests are employed to complement the component tests because even in this early phase, it is not only interesting to find out how the structure will react to a mission mix used for fatigue verification, but it is also necessary to investigate the influence of variations of mission mix. These trade-off studies give information about the limitations in life associated with different usage. A seemingly more severe load spectrum may yield a longer life. For example, it is known that a limited number of additional high loads contained in the load sequence may cause an increase in fatigue life, because of plastic effects at the notch root.

So, the specimen and component tests in the development phase may be characterized as follows:

E. continuing the type of testing listed under items A through D, possible confirmation of the results found in the definition phase;

F. determining the fatigue life, and structural reaction to cracks of critical components to be used as a basis for their final layout;

G. especially for damage-tolerance design, investigating crack growth in critical areas, including investigating the often very sensitive influences of modified missions/loads/load sequences.

During the verification of structure it is in some cases necessary to support the full scale fatigue test by component tests in order to:

H. provide a fatigue life for certain areas that have not been included in the FSFT, or areas for which the rate of testing in the Full-Scale Fatigue Test have not been representative;

I. in addition, it is often necessary and practical, to investigate necessary modifications in separate component tests.

During the time the aircraft is in service, further specimen and component tests become necessary, for example:

K. to assure the fatigue life in the case of modifications required e.g. because of service damage;

L. to determine the modified admissible fatigue life for missions differing from the layout missions (different mission mix, new configurations);

M. to allow the establishment of well-founded inspections and inspection intervals for critical areas. It should be mentioned that sufficient results can only be obtained from major component tests.

5.2 DEFINITION OF COMPONENT TESTS

5.2.1 Type of Test

To illustrate different types of testing, Figure 5.2.1 schematically shows crack growth during total life. During the first period of life cracks are not existing or non detectable. Usually, fatigue tests are carried out to investigate the total life of the structure, which is designated as L_G in Figure 5.2.1. Crack propagation data may be obtained either by close surveillance during the test or by analysis of the fracture surface after failure. This type of test will be called *Fatigue Life Test* in this paper.

If it is the intention to determine only the crack propagation after an initial crack, a_1 is present, the test will be called *Crack Growth Test*. The life of this period is indicated in Figure 5.2.1 by L_R .

There are some other fatigue tests used, for example in combination with proof loading. In this case, during fatigue testing, the structure will be proofed periodically with loads which are higher than the maximum load of the fatigue spectrum. If the structure is designed for fail safe criteria it is important to find the residual strength at a defined crack length or given life (life L_{GP} or L_{RP} in Figure 5.2.1).

5.2.2 Type of Structure Tested

Tests on details and components cover a wide field of fatigue testing. There are many different types of structures and specimens which are tested in this area, for example:

Minor Specimens

Investigations of materials and processes
- fatigue properties of materials
- processes of forging, pressing, cold working
- thickness and shape of materials
- heat treatment, surface treatments

Investigations of detailed design and manufacturing
- area of stress concentration, notches
- rivet- and bolt joints, bonding, welding etc.
- fasteners themselves

Components

Structure elements such as:
- fittings, load introduction parts
- bearings, lugs, rods
- cut outs of shells, skin panels
- complex joint parts of structures.

Major Components

For example:
- flaps, slats, spoiler, rudders
- landing gears, tanks, pylons
- vertical, horizontal tail
- wing or fuselage sections

The latter items may be also called "Full Scale Structures" in some cases.

5.3 STANDARD OF TEST SPECIMENS

5.3.1 Layout of Specimens

The test structure should resemble the real structure as much as possible in its critical areas. This is compulsory for qualification tests and highly desirable for component tests. For specimen tests, major or minor simplifications are permissible.

5.3.1.1 Layout of Minor Specimen Test Structures

If tests on small specimens are to provide only limited information, simplifications in comparison to the real structure are possible. This applies whenever less costly preliminary tests are used for comparative analysis, e.g. of different materials or manufacturing processes. The results of such pre-tests are only of limited use for real fatigue-life analysis, especially if the load sequences are additionally simplified (cf. section 5.4). An exception would be made, if for example, certain stress concentrations could be approximated by notched specimens that result in the same damage index for the same load sequences. This method, however, can only be used to a limited degree; it does not work when additional secondary effects occur, such as bending stresses, stress corrosion etc. Therefore, it is usually impossible to use representative notched specimens for joints and lugs. Also, designers are warned in this connection not to take the stress concentration factor K_T as a standard measure and to transfer results when or if equal K_T-values are observed for greatly differing structures.

Whenever simplified specimens are used it should be assured that:

- the specimens are taken from products made according to the same material manufacturers specifications, including heat treatment processes.
- temperatures that are likely to influence material properties do not occur when manufacturing the specimens.
- no readjusting or cold working is done, e.g. to correct specimens that have been deformed in manufacture.
- any finishing or surface treatment does not falsify the results.

5.3.1.2 Structural Layout of Components in Development Phase Tests

For fatigue component tests during aircraft development, structural details that do not affect fatigue life may be modified, e.g. to cut costs. The critical areas, however, must resemble those of the original structure even in the way they have been manufactured. The structure should be chosen such that the true stress distributions exist in the areas to be tested, and that no premature secondary damage, say by overloads, occurs in the remaining areas.

5.3.1.3 Structural Layout of Components for Verification Testing

It should be firmly ascertained that the standards of the structures to be tested correspond to that of the series-manufactured air frame structures, and that they have been made in accordance with the manufacturing specification. The structure should contain secondary details, such as attachment fittings, bars, holes for fairings or equipment.

5.3.1.4 Examples of Different Test Structure Layouts

Figure 5.3.1 shows a shell under internal pressure which has deformed at its critical place, the panel joint, as indicated in "Section A–B" of this figure. Various possible testing methods have been suggested and discussed in connection with this:

- testing a flat specimen, as indicated in Figure 5.3.2, will, for lack of support, lead to further deformations and bending stresses and is consequently bound to yield entirely wrong results.
- the test arrangement shown in Figure 5.3.3 will also produce results that are unsatisfactory because only one state of deformation can be approximately simulated via strain gauge calibration.
- the test rig shown in Figure 5.3.4 which allows the simulation of several states of deformation e.g. through an additional hydraulic jack, will provide acceptable results only if the critical area can be exactly located, and if the additional bending at this place can be exactly simulated by calibration.
- an exact simulation of the local deformation behavior (Sect.A-B) and, consequently, the true stress distribution can only be achieved by using a pressure chamber.

The same problem exists for the wing panel joint as shown in Figure 5.3.5. The support by the rib admittedly can be simulated in a large-size shell specimen, but it would be highly difficult to cover secondary effects, such as may result from the bending of the wings and possible shear and torsion loads. For this reason, it is often necessary, in order to simulate true stress conditions, to use real sections of a wing box.

5.3.2 Manufacture

As mentioned before, the structural parts of verification tests should be manufactured at the place where the parts for the series structure are to be manufactured. The manufacturing specification must be complied with exactly, and possible deviations must be noted. The manufacture of structures for component tests also should largely follow the series standard. In the case of major components, assembly as well as disassembly, for inspection and maintenance during the tests should be accomplished in line with the specifications for the respective series. This applies, for example, to assembly sequences, torque of bolts, wet assembly and lubrication.

5.3.3 Load Introduction System

The load introductions should be laid out so that the areas to be tested are not influenced. If no experience is available on this, it should be verified by checking the stress distributions with strain gauges or by other methods.

Another prerequisite for the test is, of course, that the structure to be tested is not destroyed during load introduction. To prevent this, suitable reinforcements will have to be installed especially in components that represent only one detail of an integral construction. It is not easy, in all cases, to arrive at a satisfactory compromise between these two requirements, that of sufficient reinforcement on the one hand, and that of keeping the test structure free from influence of these reinforcements on the other. For this reason, it often becomes necessary to precede major component tests by preliminary smaller tests in order to obtain sufficient information on the influence exercised by load introduction.

Another problem is the correct simulation of the stiffness in the attachment area. This problem occurs in almost any component test, except for components having statically determinate attachment points with low friction. It has been shown in Figures 5.3.1 to 5.3.4 how difficult it is to simulate local stiffness, displacements, and deformations in a shell structure.

While such local problems can be solved only to a limited degree by calculation, stiffness in major components has to be simulated on the basis of static calculation. It would be more reliable, however, to use stiffness parameters determined in previous static tests.

Two examples of stiffness simulation on major structural components are demonstrated here. Figure 5.3.6 shows the stiffness conditions as simulated on a multi-web wing. The stiffness simulation here involves the attachment of each beam in the wing root area (rotation from M_x). Figures 5.3.7 and 5.3.8 show the simulation of the displacement of the fuselage side walls during a fatigue test with a sweep-wing aircraft wing center box. The following recommendations should be noted when considering stiffness:

- when laying out the test structure for a component that is firmly integrated in a total system, its deformation behavior should be considered in addition to its stress distribution. For example, in the wing shell joint specimen shown in Figure 5.3.5 which is being primarily tension-tested, the secondary influence coming from the wing bending has a strong effect on its fatigue life.
- in minor components, such as fittings, the deformation behaviour of the counterstructure should be checked. Often it is advisable to include the counterfittings in the simulation.
- even if the attachment of a component to a rigid test rig is statically determinate, local falsifications may occur, e.g. as result of friction.
- whenever stiffness is not simulated, a prior estimate is necessary to show the influence of this simplification on test results.

For statically indeterminate attachments, it should also be examined how the real conditions can be simulated in the attachment area, to what extent an unrealistic attachment will affect stress and other conditions, and what reinforcements may become necessary. In major components, the real structure in the vicinity of the attachments is usually reinforced and then considered unrealistic, or the test structure is attached to a dummy structure whose stiffness is so established that the correct load distributions are achieved in the attachment area. Often, the two techniques are combined.

5.3.4 Number of Test Specimens

As a result of different materials, manufacturing tolerances, and other influences, the fatigue life values for several test sequences show varying degrees of dispersion.

In order to arrive at reliable values for fatigue life calculations, and to allow determining their respective significance for comparative analyses, the test results must be statistically evaluated. For this purpose, it is advisable to examine a number of specimens for certain stress levels and evaluate the results.

To obtain fairly reliable values, the number of specimens n should amount at least to five, better eight, per load level. In the case of smaller numbers of specimens, an additional safety factor is often used. It is multiplied with the mean value of fatigue life to achieve a better confidence and because both, the measurement of dispersion and the mean value, become less reliable as the number of specimens drops.

5.4 TYPE OF LOADING

5.4.1 Test with Constant Amplitudes

This type of testing gives only limited information for the life estimation of aircraft structures, because most of the loading sequences ocurring in service are much more complex than single-step loading. Also the weakness of using constant-amplitude life functions for fatigue calculation is well known. A lot of work showing this discrepancy has been done in this field to compare fatigue calculations based on the Miner rule or similar procedure with test results[5,6,7]. For this reason Miner calculation may be used only at very early stages of development.

On the other hand, constant amplitude testing has advantages for simple estimations, such as finding out differences in the fatigue behaviour of materials, processes, structure joints, details of design, etc. Examples of this are given in Section 5.2.2 for minor specimen tests. The advantage in this case is that simple S–N data can be better compared than flight-by-flight test data, especially if tests are done in different programs at different time periods. It should be noted that such kinds of tests can often produce misleading results, for example for joints in which fretting corrosion may occur.

5.4.2 Program Tests

During the development of fatigue testing techniques, the method of program testing was used as an early attempt. As known, in this test procedure loadings in a given load sequence which are equal in amplitude and mean value, are collected into blocks. During testing, the different blocks are repeatedly tested one after another. For this reason this kind of test is also called block testing. The recent and rapid development of very economic process computers including interfaced and also servo-hydraulic equipment makes it possible to simulate loading programs of high complexity in sequence. Program tests are more or less replaced by this technique, because the comparison of flight-by-flight test results with those obtained from simplified sequences, such as program testing, also shows considerable differences.

Program tests are only used in any case if special testing equipment is necessary, for example, in testing rotor blades of helicopters. In such kinds of tests, very high-frequency-loading sequences must be employed to perform the tests within an acceptable time period. On the other hand, mean loads, mostly resulting from centrifugal force, can be assumed constant for segments of loading program. Also, in fatigue tests in which environmental and temperature effects must be simulated, temperatures and atmospheric conditions are often blocked to obtain an acceptable testing time.

5.4.3 Flight-by-Flight Tests

The program of loads chosen for the test should aim to represent as many as possible of the real flight conditions. A high standard is necessary especially for test results that will be used for verification of aircraft structure. With the existing knowledge and techniques in this field – such as computer, electronic equipment, software experience, etc. – complex load sequences can now be simulated with a very high degree of accuracy.

In Chapter 2 of this Handbook the preparation and establishing of flight-by-flight programs are described in detail, mainly concerning flying conditions. In the following, only special aspects of the development of realistic loading programs will be discussed in detail.

For development of flight-by-flight test programs it is necessary to divide the typical missions or usage patterns into segments of typical operational conditions. The sequences of these conditions are normally fixed deterministically for equal missions, based on operational usage of aircraft. A typical example for this is a loading program of a fatigue test of an undercarriage structure. The operational segments, like landing, taxiing, braking are dictated by the usage of an A/c at different airfields. For a particular air force base, Figure 5.4.1 shows a typical breakdown of starting and landing cycles into segments[13].

Figure 5.4.2 and Figure 5.4.3 show the entire loading program for one typical flight, divided into the phase of starting and landing. The sequence of operational segments, which are designated on the top of Figures 5.4.2 and 5.4.3, is the same as that of Figure 5.4.1. The variable loading amplitudes, which are superimposed on every segment of the ground loads during starting and landing are based on the flight load measurements of load spectra[14]. The total loading sequence of this flight by flight program, which was repeated during testing, includes 200 different flights. Amplitudes of loads into the different segments, the number of cycles during different taxiing periods and also the loads into the 200 different flights were randomized by using a random generator. Only the correlation between P_z, P_x and P_y loads was assumed at a constant ratio in some conditions, for example as shown in Figure 5.4.4 during landing impact.

This loading program includes a very high number of cycles. An average of 300 load cycles per flight was simulated. Most of these cycles resulted from taxi loading – an average of 250 cycles per flight. In this case, the large number of cycles was necessary because the fatigue-critical parts of this test u/c are mainly damaged by loads occurring in the X-direction. On the other hand, loads in the vertical direction (P_z) induced P_y loads and also P_x loads which are relatively high.

Figure 5.4.5 in contrast shows a loading sequence typical of main u/c fatigue testing on another type of airplane. This loading program has only an average of 40 cycles per flight. Figure 5.4.6 (Ref.15) shows the test set up. Loads are applied by 4 hydraulic jacks. The examples discussed above show that even when testing similar components (in this case main u/c) loading programs will be very different. Therefore strongly detailed instructions are not possible for establishing flight-by-flight programs. The following general rules may be taken as a reference:

- For testing parts of wing panels, the number of cycles should be not less than 60 cycles per flight hour.
- If maneuver conditions are critical in fatigue for fuselage components, the same number of cycles used for the wing structure should be considered for fuselage components.
- The above given number as a minimum or a higher number of cycles shall be used for built up structures with rivet, bolt joints, etc. to cover fretting corrosion effects.
- Minimum load level should be on the order of 5 to 10% of maximum spectrum level.
- Minimum number of load cycles for tailplane fatigue testing should be in the order of 100 cycles per flight hour.

Other important influences, such as truncation, influence of negative loads, etc. will not be discussed here; they are explained in References 7, 16, 21, 22 and 23.

The flight by flight programs to be applied to the individual aircraft components are very different. Therefore, standardization on a broad basis cannot be achieved in a practical manner. A standardized individual flight program has been developed for the lower panels of fighter aircraft wings, which has been determined to be one of the most critical points of the entire structure[7,8,9,10,11]. This program, developed through the cooperative efforts of several European countries, is based on continuous flight recordings of 324 flights with 5 different aircraft types. The purpose of this standardization was to establish a loading program, suitable for conducting the following investigations on a uniform basis:

- Comparative examinations of effects for materials, joints, manufacturing processes etc. with minor specimens. The advantage over constant amplitude tests performed until now, lies for example, is that rivet and bolt joints are loaded in a representative manner. In this case effects such as fretting corrosion which depend very heavily on the loading level are included, and they representatively influence the life.
- Furthermore, this program can be implemented for comparison studies in early development phases.

Results obtained with this program may be applied to calculate the life under different load spectra, using the so-called relative Miner rule. The deviations are substantially smaller than by life-time calculation according to standard Miner' calculation based on S-N data.

5.4.4 Random Test

In many cases, particularly when no deterministic relationship of individual loading segments or no considerable changes in the mean load exist, the loading program can be purely randomized by customary methods, for example, using a random generator. There also exist standardized loading programs for certain loading spectra such as the Gaussian Distribution. It should be mentioned here that some portions of flight-by-flight loading programs generally are also randomized.

5.4.5 Boundaries for Loading Programs, Environmental Aspects

The main limitations of a loading program result from the loading frequency. The loading frequency is heavily influenced by the mounting structure, load introduction system and by the stiffness of the test article. Unallowable phase shifts cropping up when several hydraulic cylinders with grossly different displacements are used, can also be a basis for limiting the loading frequency. A portion of the above mentioned effect can be compensated through the use of modern process computers for controlling the test, for example by prior programming of anti-phase shifts for the individual cylinders or different loading speeds dependent upon the loading amplitude.

In determining the allowable loading frequency it should also be observed that there are no loading errors in the main damage area that are greater than a few percent.

Also, other influences such as temperature or humidity are to be considered. However, they will not be further explained here. Worth mentioning is the influence of corrosive media or also the normal environment of service aircraft in sea atmosphere. Concerning this latter influence, as much as a factor of two between laboratory test results and real damage in service has been observed. At the current level of knowledge, there is no factor that can be uniformly employed in determining this influence. However, it appears important to conduct tests in salt water atmosphere for crack propagation, that is, for damage tolerance considerations. This is necessary because a corrosive atmosphere evidently has considerable influence on the life span of structure portions in which cracks already exist.

5.5 CONDUCT OF TESTS

5.5.1 Test Set-Up, General Requirements

The layout of the test set-up must allow a simulation that is representative of the actual loading and stress distribution in the test structure.

The results of the fatigue test naturally depend on the layout used for the test set-up, the construction of the test specimen (see Section 5.3.1) and the load introduction (see Section 5.3.3). This layout exhibits a compromise since it must be weighed against the costs expended. The details of a test set up should not be entered into at this point, but it appears that the layout of the system which produces the loads in an appropriate manner is worth mentioning.

As already mentioned in Section 5.4, with today's computer technology it is possible to produce very detailed loading programs. In order to be able to perform these within a reasonable test time, the loading speed must be as high as possible. Therefore, appropriate load producing systems should be developed with the following characteristics:

- Displacements as samll as possible during the process, to minimize testing time.
- The minimizing of the inertias of all moving parts of the loading system.
- Stability, and prevention of secondary displacements, as well as minimization of bearing play during tensile and compressive loadings;
- An overall low friction set up.

The above mentioned requirements imply, as the rule, compact layouts which can influence the displacement restraints of the test article.

A typical example that considers the above mentioned requirements is the load distribution system for a wing box illustrated in Figure 5.5.1. With this system, loading frequencies from 60 to 120 load changes per minute can be reached for several similar wing box systems. This load introduction system is relatively compact in the loading direction, displacements in side direction caused by wing bending are compensated for by the use of rubber mounts. These mounts were especially developed for this purpose and tested and optimized during trial tests.

5.5.2 Load Control

The load control and the load checking system have to be such that all important test data i.e. applied loads, pressure data, etc. can be monitored continuously at all times. Deviations in load application have to stay within reasonable limits, i.e. the life of the structure must not be significantly affected by these deviations. When maximum loads or given tolerances are exceeded, or differences in the loading history occur, the loading system has to be switched off automatically. In this case no increasing of loads shall occur and the reducing of loads to zero shall be in a controlled fashion.

In order to comply with these requirements, the electrical, hydraulic or mechanical safety devices necessary must have clearly definable, adjustable limits. These devices have to be function tested periodically during the test. In addition to these load oriented safety devices, certain test data e.g. strain gauges and crack wires should be continuously controllable by way of safety switch arrangements.

At total or partial failure of the test structure, the installation must switch off automatically. Further laod increases or load applications at the failure load condition have to be avoided.

The expense, which is necessary for this system, will be influenced by questions such as: How complex is the test structure? For which purpose will the results be used? What is the total testing time? Also, the applicable facilities, their types and standards are, of course, important. If simple facilities are at hand, only simple safety devices are possible for example: limitation of hydraulic pressure or other direct boundaries of loading units. Complex and larger components will normally be tested with servo-hydraulic closed-loop systems controlled by computer, which is the current standard of fatigue testing. If using these, it is possible to realize quite varied loading sequences. Figure 5.5.2 shows the functional diagram of such a system, in which numerous safety devices are also included. Figure 5.5.3 shows a small period of loading sequences with tolerances and boundaries of different safety criteria. The tests are controlled and monitored during the entire testing time in the above mentioned manner. In Table 5.5.1 some basic data for establishing loading programs of component and full scale tests is pointed out[19], to show fullness of data and complexity of the loading program.

Sometimes, component testing will be used to support full-scale tests, for example: If structure modifications will be proofed very quickly. In this case stress history can be measured directly with strain gauges applied in critical areas. With this history it is possible to control and calibrate the facilities of the component test directly. Figure 5.5.4 shows such a kind of test including the strain gauges used for control[24]. This method can also be applied if stress history is monitored in flight. In this case all weaknesses of load assumption, load distribution and stress analysis of structure can be eliminated.

5.5.3 Additional Stress Analysis

Usually before starting fatigue testing, a more or less detailed stress analysis is carried out with strain gauges, photo stress, etc. There are various reasons for this, for example:

- compare stresses with theoretical analysis,
- find out stress distribution,
- find out critical stress concentrations for further instrumentation during testing,
- check load introduction, modification and reinforcements for loading at test structure,
- better interpretation if cracks occur,
- correlation with other tests such as full scale test, static tests or flight measurements.

In some cases it may be recommended to make periodical stress measurements during testing to find out differences, which can occur due to setting effects, wear etc.

5.5.4 Inspections

In order to make sure that initial cracks will be detected before they reach the critical crack length, previous theoretical considerations should be taken to select suitable inspection intervals and methods.

Generally it is necessary to subject the test specimen at a frequency e.g. daily visual inspection. For areas that are not accessible without partial disassembly of the structure, special inspections will have to be made at defined intervals. In order not to influence the crack initiation and the crack growth e.g. by chemical media, only NDT-techniques are generally used for special inspections. While optimizing the suitable methods (e.g. eddy current, ultra sonic) it is often necessary to test the effectiveness of these methods on geometrically similar structures with artificial cracks.

Structural components which are accessible only with considerable effort, i.e. only in large time intervals, should be controlled by special provisions which can be observed continuously during the test (e.g. crack wires, strain gauge instrumentation, etc.).

When a crack is detected, the crack propagation should be checked at relatively short intervals. The cracks should be compared with the results of theoretical analyses, if possible. Crack observation is one of the most important tasks within the test procedure because only through this can inspection intervals for aircraft in service be reliably defined.

Besides inspection to detect cracks, the structure will have to be checked for items such as fretting corrosion, wear, high friction etc., if these effects can be expected.

5.5.5 Duration of Test

The test should be continued at least until the minimum specified service life, multiplied by an appropriate safety factor, is reached. Normally a factor of 4 will be taken, the minimum factor specified in /n/ is 2. Using this safety factor, uncertainties based on scatter of material properties and manufacturing methods as well as the so-called test service factors are covered. The latter factor takes into account the usually longer life which is obtained by testing under laboratory conditions in comparison to service life under long time environmental conditions (corrosive influence on crack initiation and propagation etc.).

If no significant damage occurs before reaching the required test time, the following procedures are recommended:

- The test is to be continued with the normal loading program but in areas considered critical, artificial cracks should be applied which are continuously checked for crack propagation while the test is being carried out. This procedure normally yields the best information from all the ones discussed here. In this case it is normally possible to evaluate how critically a crack propagates, or which preventive inspection methods should be applied after a certain time of service life. The intervals of inspection should also be determined.
- Completion of test by performing a fail safe test in order to determine the residual strength or undetected cracks. With respect to the residual static strength, this procedure yields reliable and informative answers. The detection of hidden cracks which can't be found by usual inspection techniques is, however, using this procedure, more or less accidental.
- The test is to be carried on with 10 to 20% higher loads until damage or failure occurs, to find out the most damaged areas of structure.

The above paragraphs make it obvious that the actual testing method will be different from one test to the other: Different design philosophies for the layout of tested structure dictate the procedure to be used.

5.5.6 Test Results

When the fatigue test has been carried out the fractured surfaces should be examined very accurately to find out the actual reason for failure. This, for example, could be obtained from an analysis of its starting point. If, at the end of the tests, the structure shows only initiating flaws without any major failure (for example on screwed or riveted components) a complete tear down inspection will have to be conducted in order to gain necessary information on crack initiation and extension. To be able to examine the crack extension for crack propagation by fracture mechanical techniques, it is useful to prepare the loading program, so that typical marker lines will be induced at the fracture image by loading. Hereby the respective typical flights and loading sequences can be recognized and, consequently, crack propagation can be determined afterwards more accurately. In applying this procedure, nevertheless, it must be ensured by experience or by previous tests that modification of the loading program (concentrating big load cycles as a rule) does not influence the life endurance.

The final test report should contain all important results within the following categories:

- Definition of structure tests,
- Test set-up,
- Load sequence,
- Measurements test set-up, methods used,
- Documenation of defects and damage,
- Exploitation of the stress measurements,
- Evaluation of crack propagation data,
- Application of results to aircraft in service.

In particular, it ought to be specified which standard the test structure is to meet (development tests, standard of series), which simplifications are due to the test set up, to load introduction, etc. in spite of the true attitude, and on what data the loading program is based. Furthermore, the agreement with other results, for example through comparative stress measurements on full scale tests and measurements in flight, should be discussed.

5.6 CONCLUSIONS/RECOMMENDATIONS

Specimen tests and component tests are necessary in order to get the first life endurance data for the choice of material and for modified production procedures at an early stage of development. During the actual development, they serve to support the theoretical analysis of complex structure elements. In the phases of verification and utilization, they represent a valuable complement of the full scale investigations, and the results are quoted in order to judge structural modifications and changed usage of aircraft. The ability to interpret the results is usually dependent on the expenditure incurred for the design of the test structure, the load attachments and the complexity to find correlations to other investigations, and real conditions, it will be necessary, in many cases to make detailed measurements before starting the proper fatigue investigations. (For example, to check stress distributions realized.)

Also, the quality of the results depends upon the ability to simulate the true loadings in detail. Due to the progress in electronics and computer techniques in this field, sufficient expertise is now available to simulate, very accurately, the true loads through flight-by-flight programs and to supervise them during the test performance.

The apparatus and equipment available to supervise the test runs are highly automated. An important aspect is the periodic inspection of the test article which is performed with current standards such as NDI-methods. Through detailed records of all occurrences, damage and damage extensions during the test performance, well-founded conclusions can be drawn for complex component tests, in order to be able to determine modifications, inspection intervals, the time of exchange and so forth.

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TABLE 5.1.1

Component Testing During Definition, Development and Usage of A/C

SERVICE PROTOTYP • DEVOLP • DEFIN • PROJ •

A/C – DESIGN AND USAGE	FATIGUE ANALYSIS AND DESIGN	FATIGUE TESTING OF COMPONENTS
• AIR VEHICLE SPEC • STRUC. LAYOUT • STRUC. TEST PROG.	• FATIGUE REQUIREMENT • ESTIM. LOAD SPECTRA • FATIGUE CRITICAL PARTS • FATIGUE TEST PROGRAM	MINOR SPECIMEN: • MATERIAL SELECTION • MANUFACT. PROCESS • LIFE FUNCTIONS • CRACK PROPAGATION • RESIDUAL STRENGTH
• STRUC. DESIGN CRITERIA • DETAIL STRUC. DESIGN • QUALIFICATION TESTS • FLIGHT SURVEY	• FATIGUE DESIGN CRITERIA • DETAIL FATIGUE ANALYSIS • FULL SCALE TESTING • FATIGUE CONTROL PROGRAM	COMPONENTS: • FOR TRADE-OFF STUDIES • DESIGNING OF CRITIC. COMP. • SUPPORT FULL-SCALE FAT. TESTS • LIFE-, CRACK-GROWTH INVESTIGATION
• SERVICE EXPERIENCE • TRACKING PROGRAM • FLEET INSPECTION • STRUC. MOD'S	• FATIGUE LIFE MONITORING • FATIGUE FAILURE • FATIGUE ANALYSIS FOR DIFFERENT USAGE AND STRUC. MODIFICATIONS	MINOR SPECIMEN A. COMP.: LIFE- INVESTIG. FOR STRUC. MOD'S INVESTIG. LIFE IF USAGE DIFFERS FROM DESIGN CRITERIA

TABLE 5.5.1

Basic Data of Flight by Flight Loading Programs*

FULL-SCALE FATIGUE OR MAJOR COMPONENT TEST	TIME PERIOD	NUMBER OF			REPEATED LOADING SEQUENCE	
		DIFFERENT MISSIONS	DIFFERENT LOAD CONDITIONS	LOADING UNITS	DURATION	TOTAL CYCLES
F 104 – FSFT	1969–73	8	143	33	1000 FL. HRS	$85 \cdot 10^3$
F 104 – MAIN U/C	1972–73	–	150	5	200 FLIGHTS	$60 \cdot 10^3$
G 91 – WING TEST	1972–74	8	230	16	1000 FL. HRS	$65 \cdot 10^3$
A 300 B-AIRBUS, FSFT CENTER FUS./WING	1973–77	DIFF. TYP. FLIGHTS	1200	54	1000 FLIGHTS	$30 \cdot 10^3$
MULTI RULE A/C MAJOR COMP. WING/FUS.	1974–76	6	535	22	1000 FL. HRS	$76 \cdot 10^3$

* Tested by IABG.

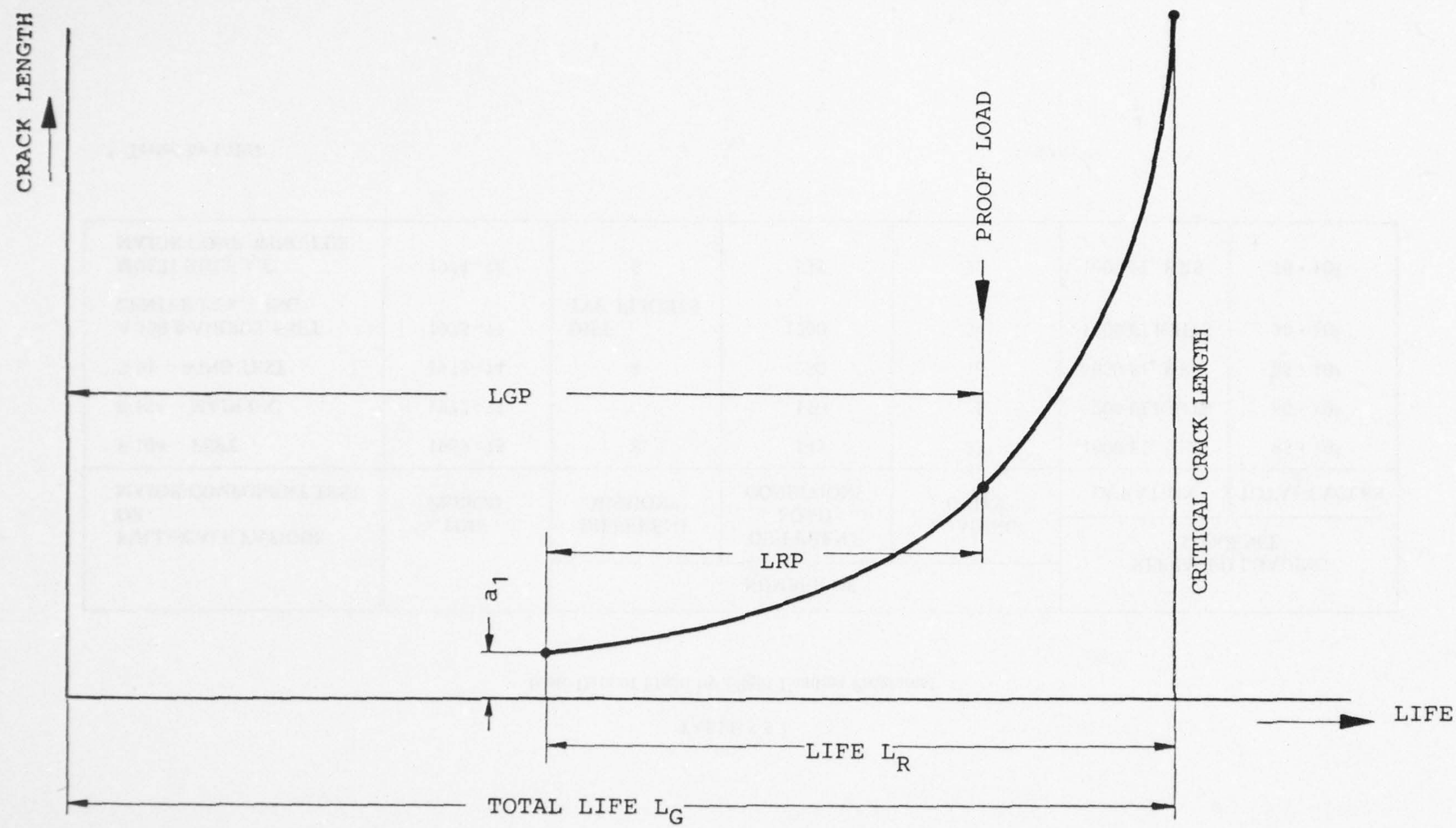

Fig.5.2.1 Fatigue life/crack growth

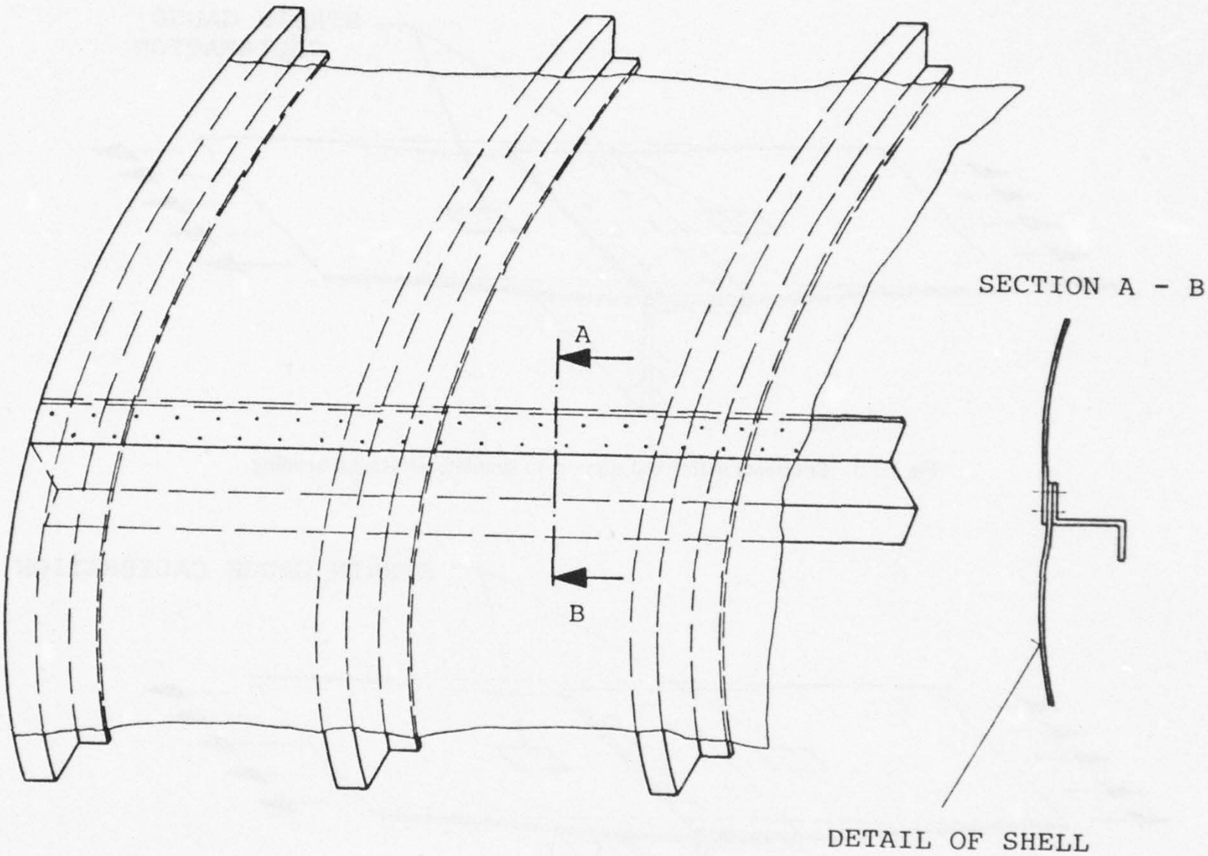

Fig.5.3.1 Pressure shell

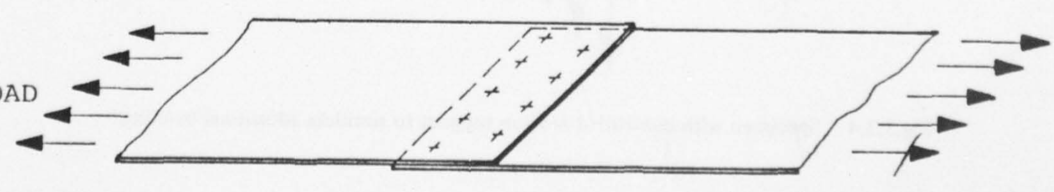

Fig.5.3.2 Flat specimen without support

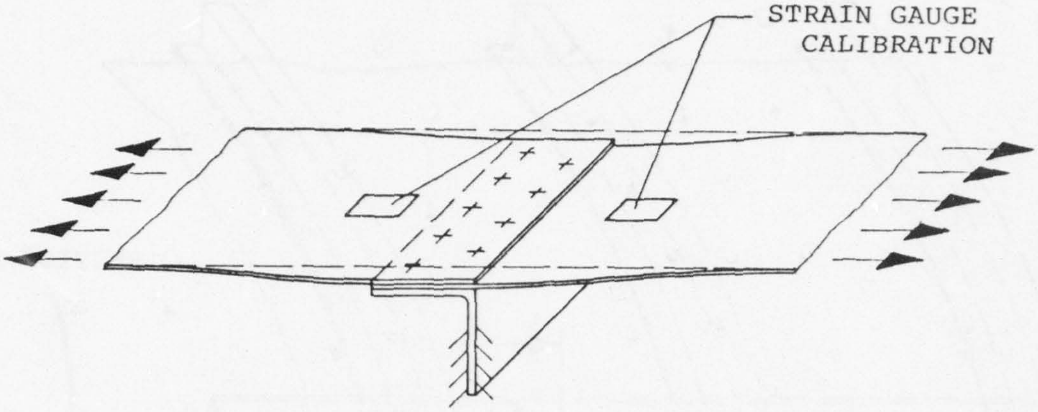

Fig.5.3.3 Specimen with fixed support to simulate additional bending

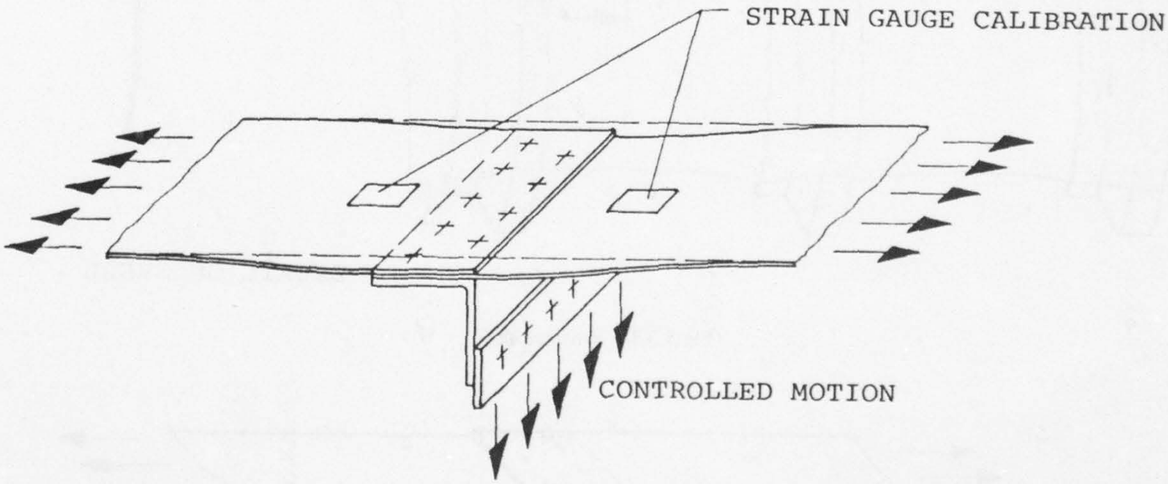

Fig.5.3.4 Specimen with controlled motion support to simulate additional bending

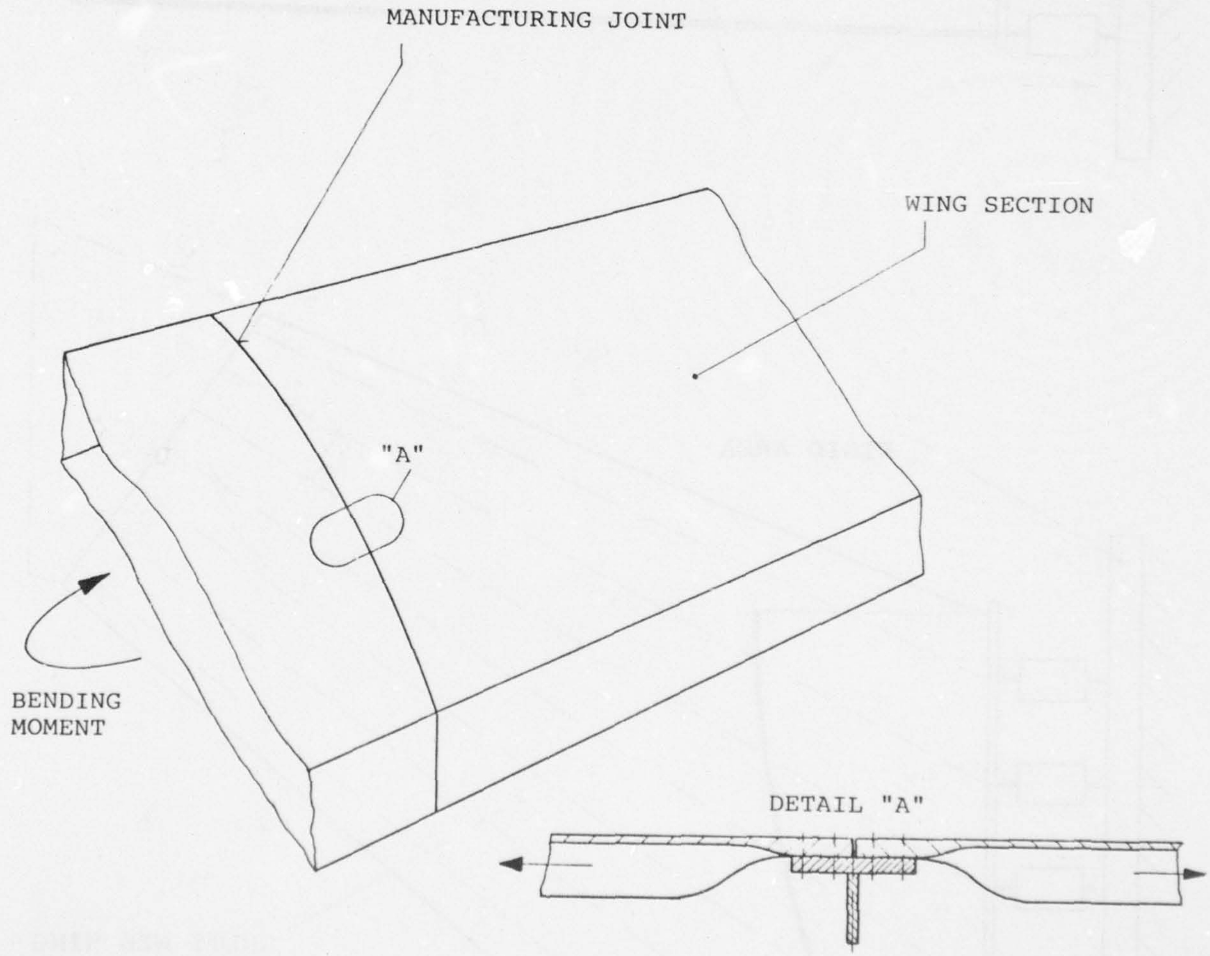

Fig.5.3.5 Joint of wing box

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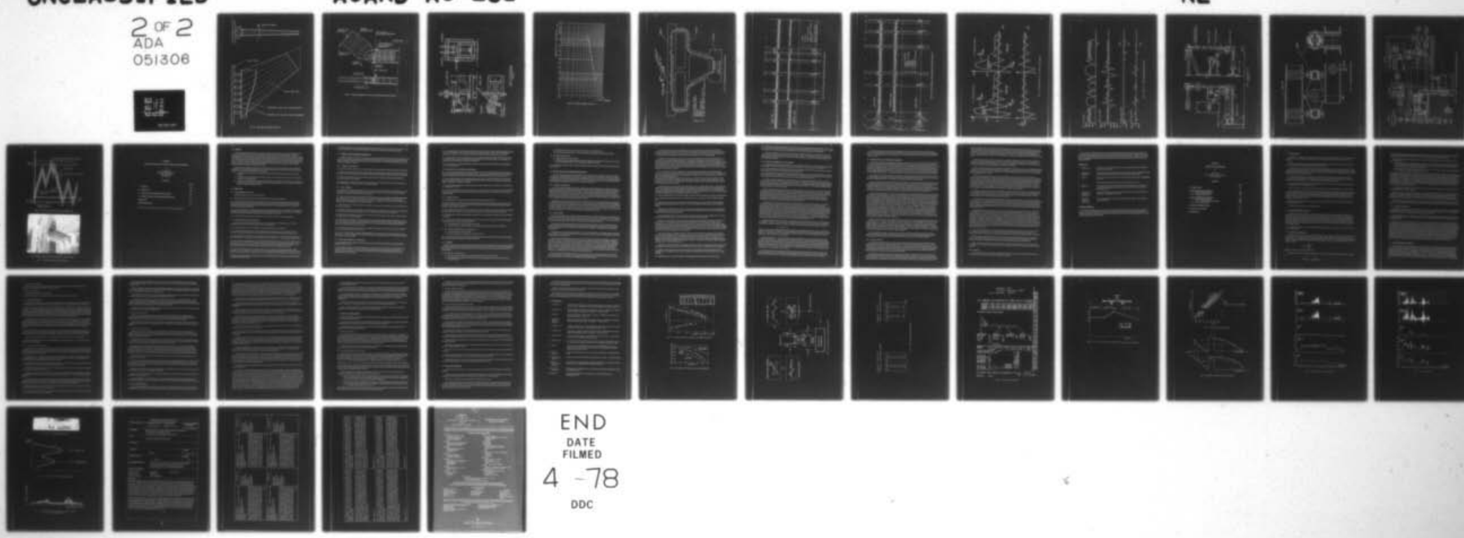

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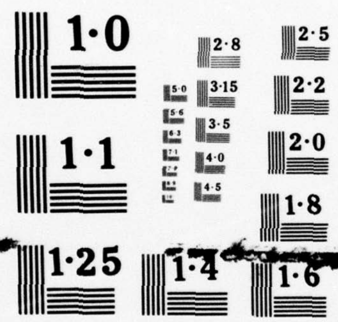


NATIONAL BUREAU OF STANDARDS
MICROCOPY RESOLUTION TEST CHART

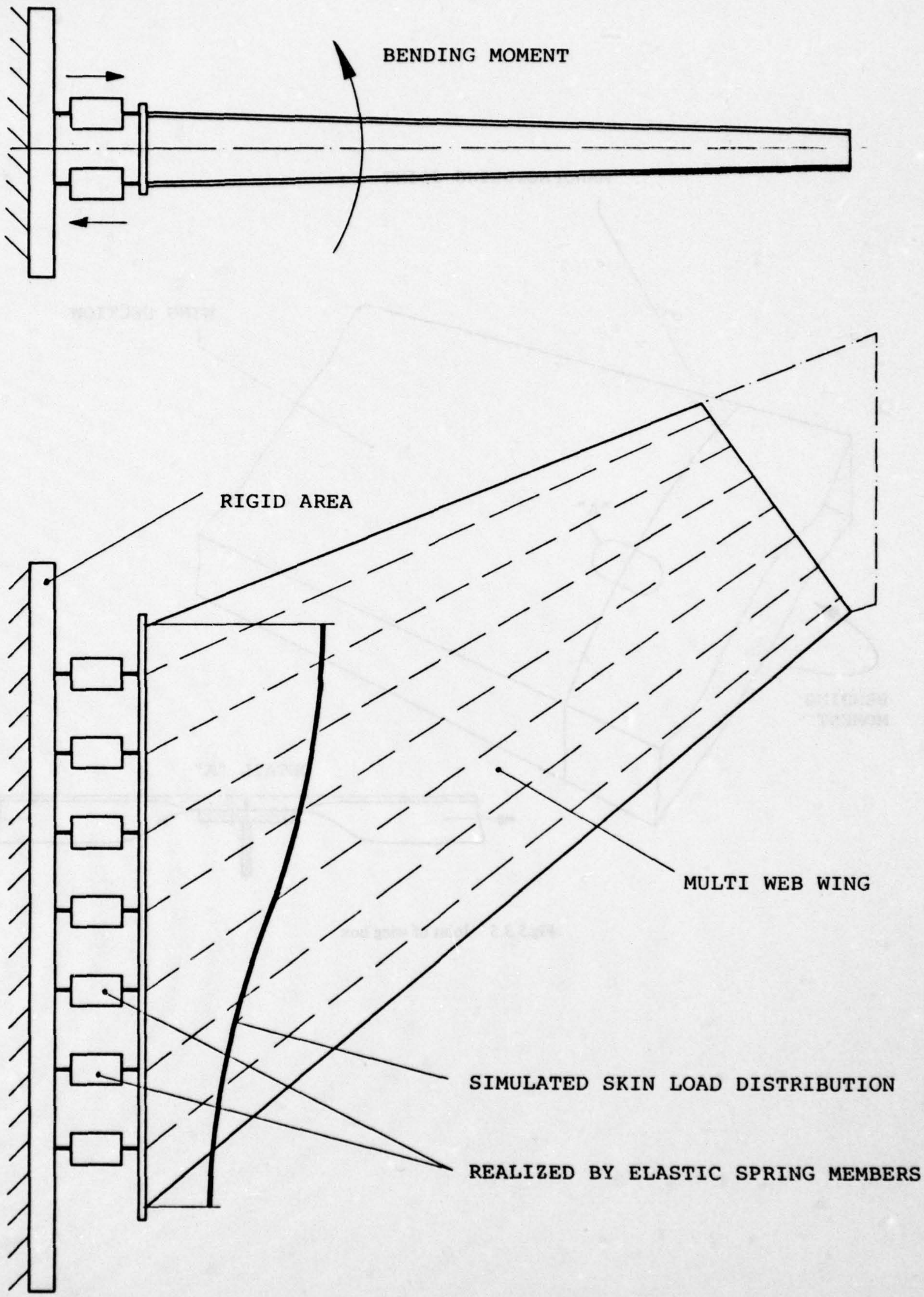


Fig.5.3.6 Wing fatigue test simulation of stiffness

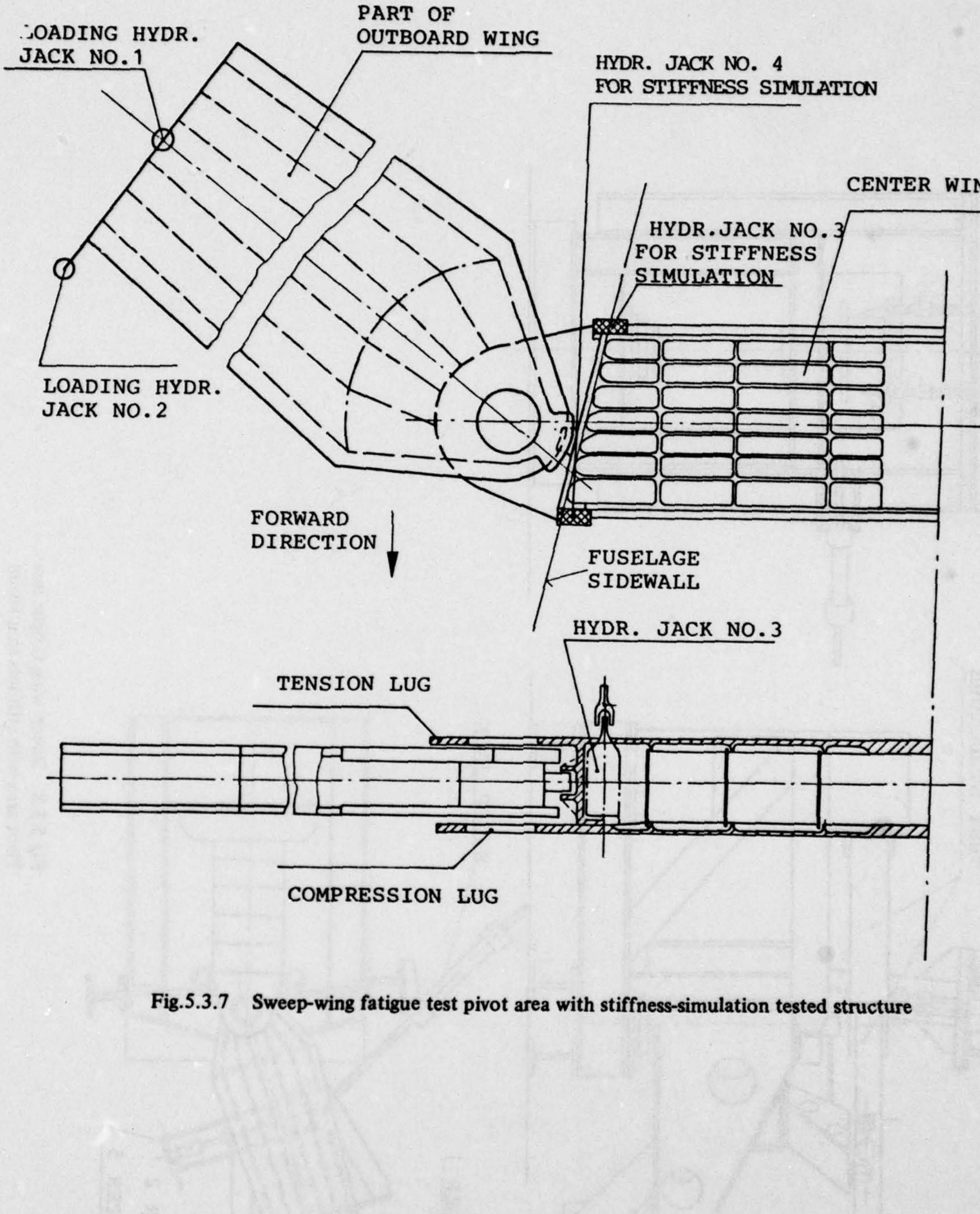


Fig.5.3.7 Sweep-wing fatigue test pivot area with stiffness-simulation tested structure

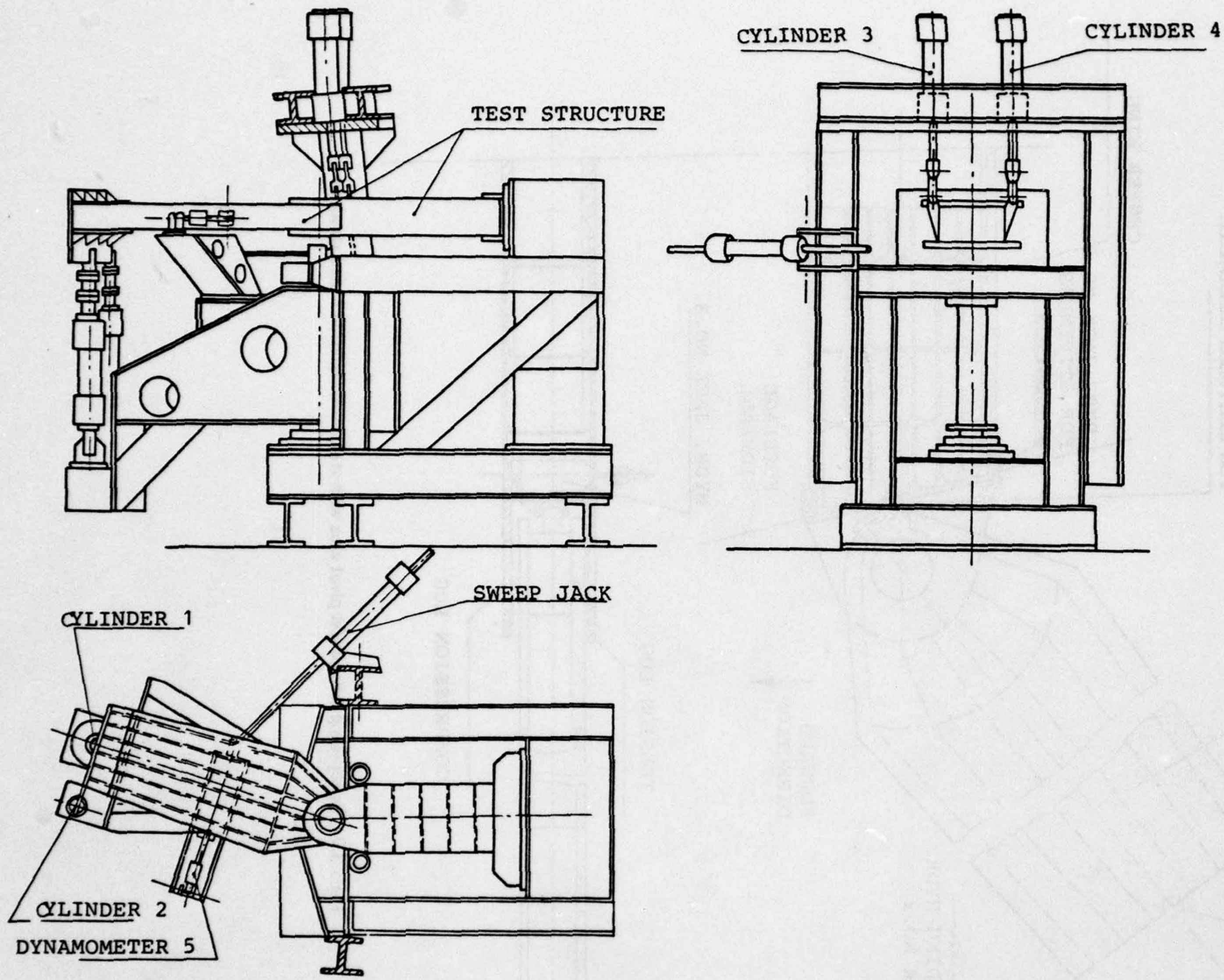


Fig.5.3.8 Sweep wing fatigue test
Pivot area with stiffness simulation
Test set-up

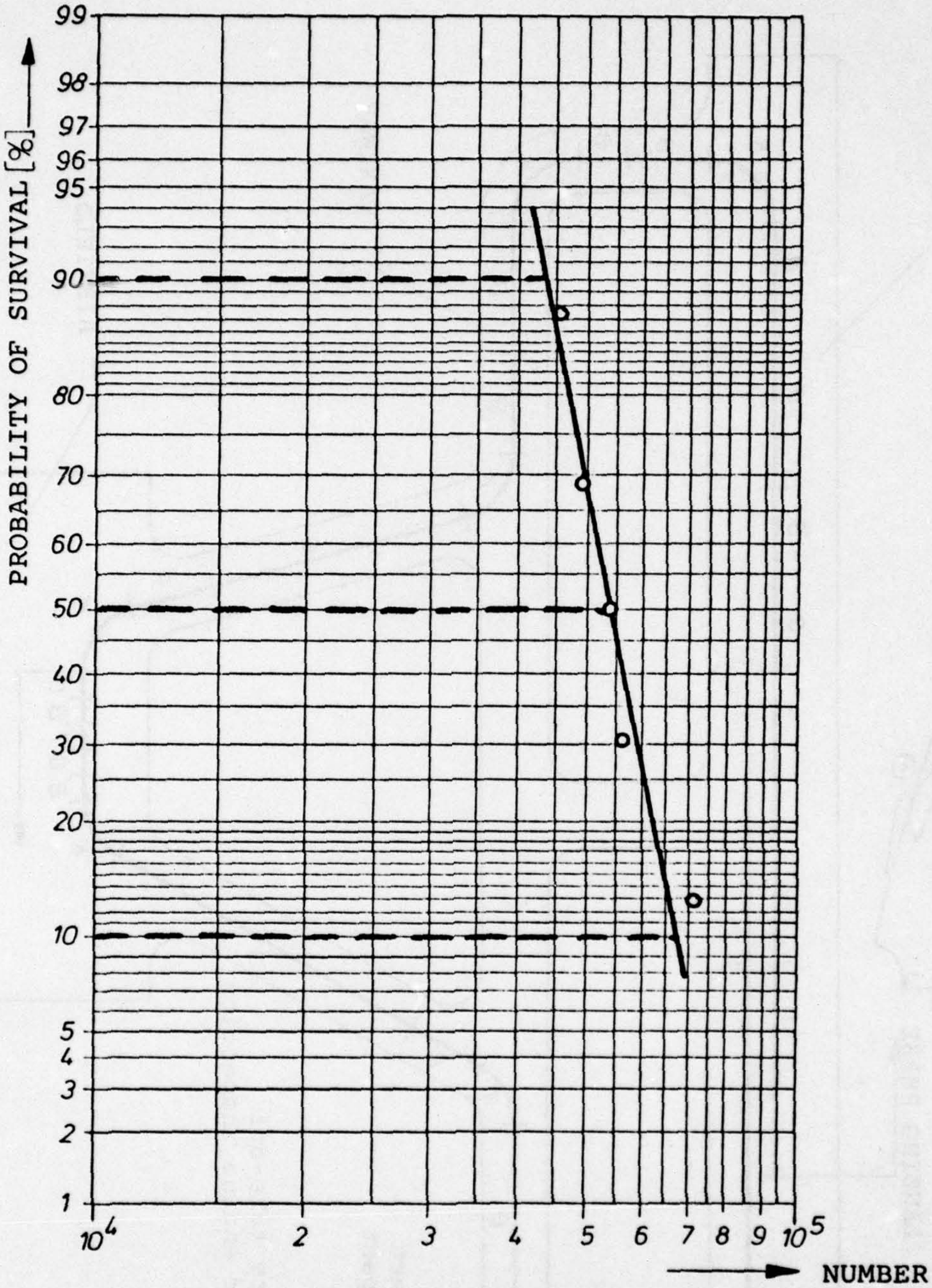


Fig.5.3.9 Statistical evaluation of test results

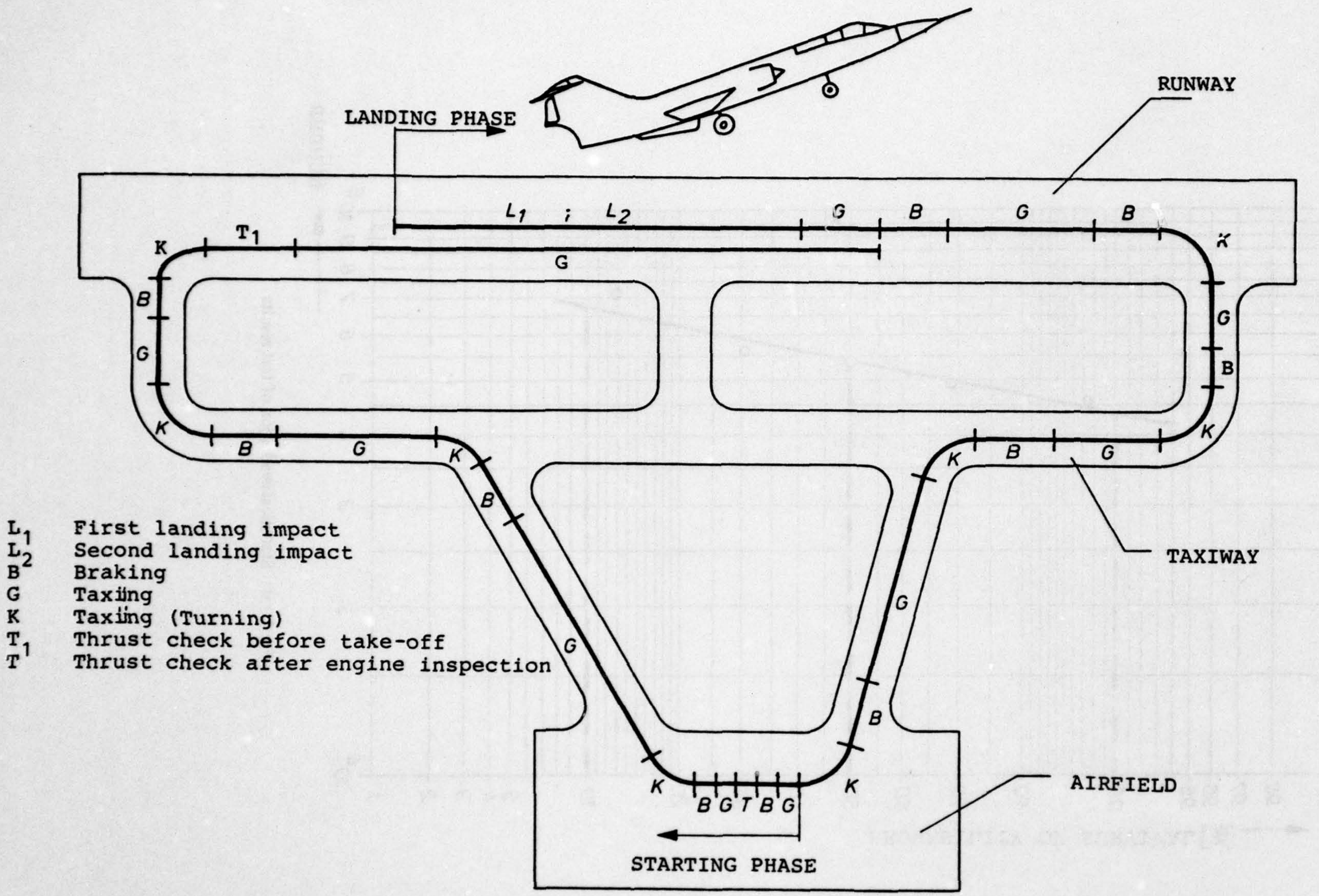


Fig.5.4.1 Starting-landing breakdown into typical segments

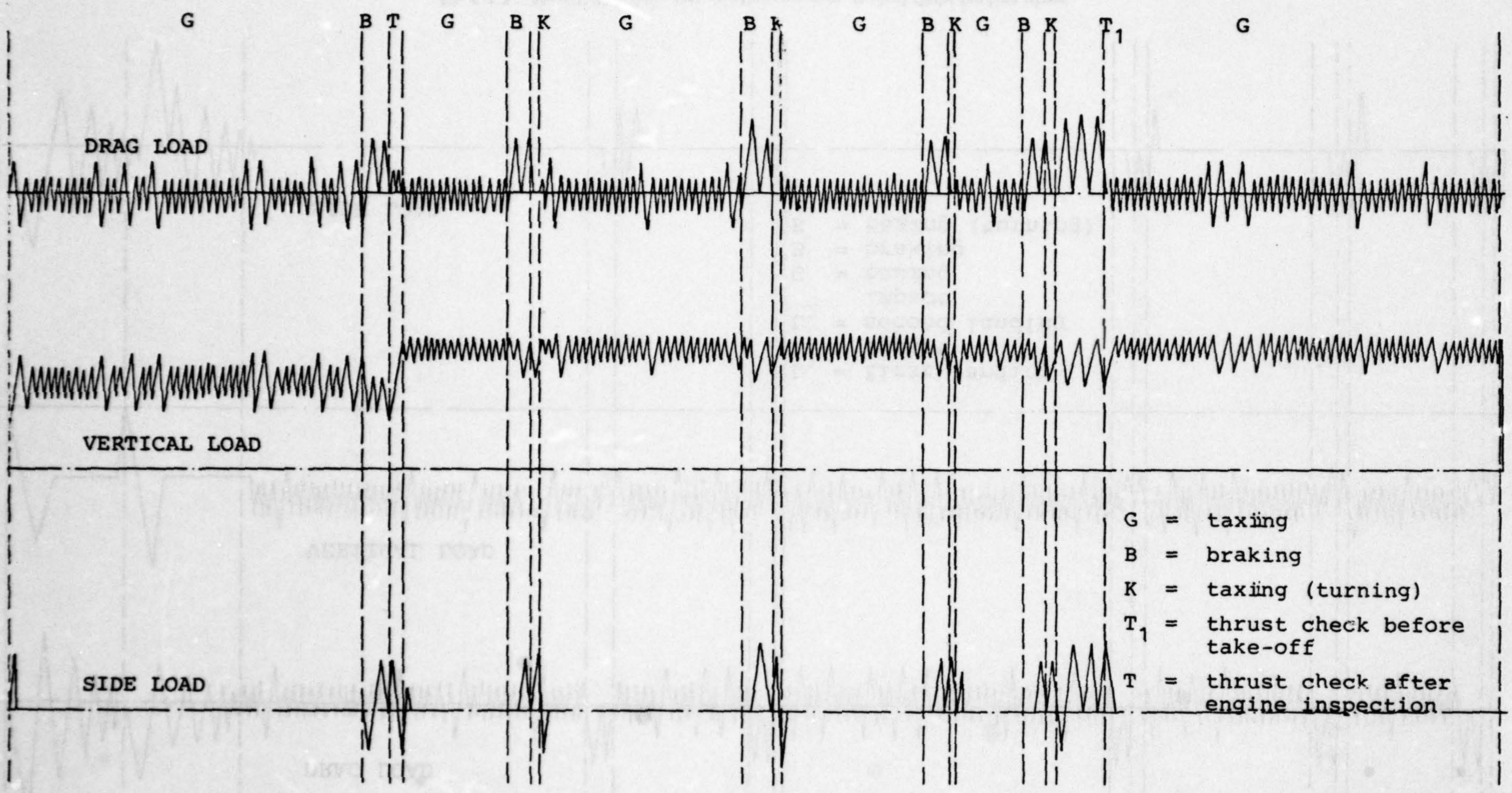


Fig.5.4.2 Main U/C fatigue test, loading program, typical flight, starting phase

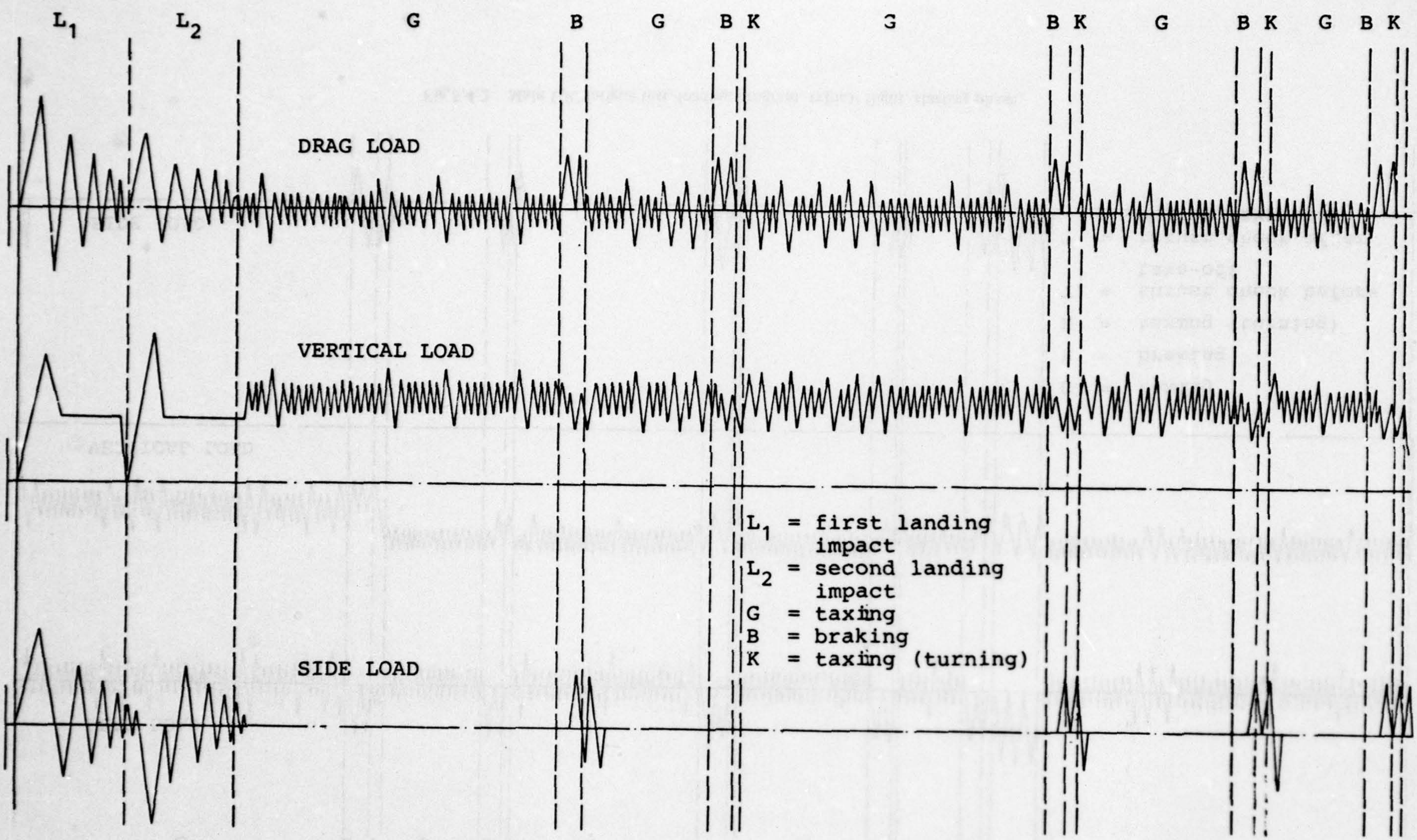


Fig.5.4.3 Main U/C fatigue test, loading program, typical flight landing phase

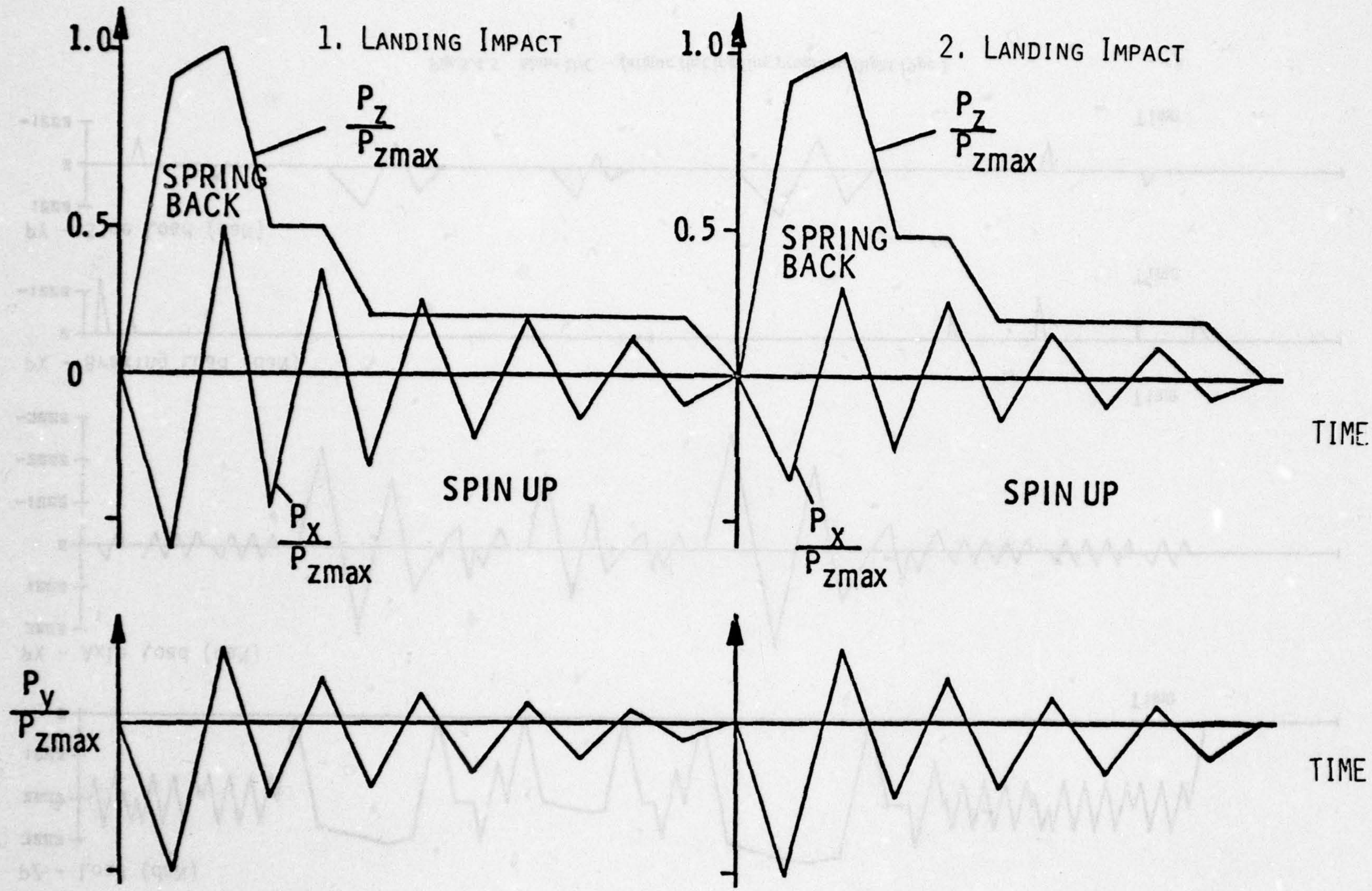


Fig.5.4.4 Correlation between P_x-, P_y- and P_z-loads during landing

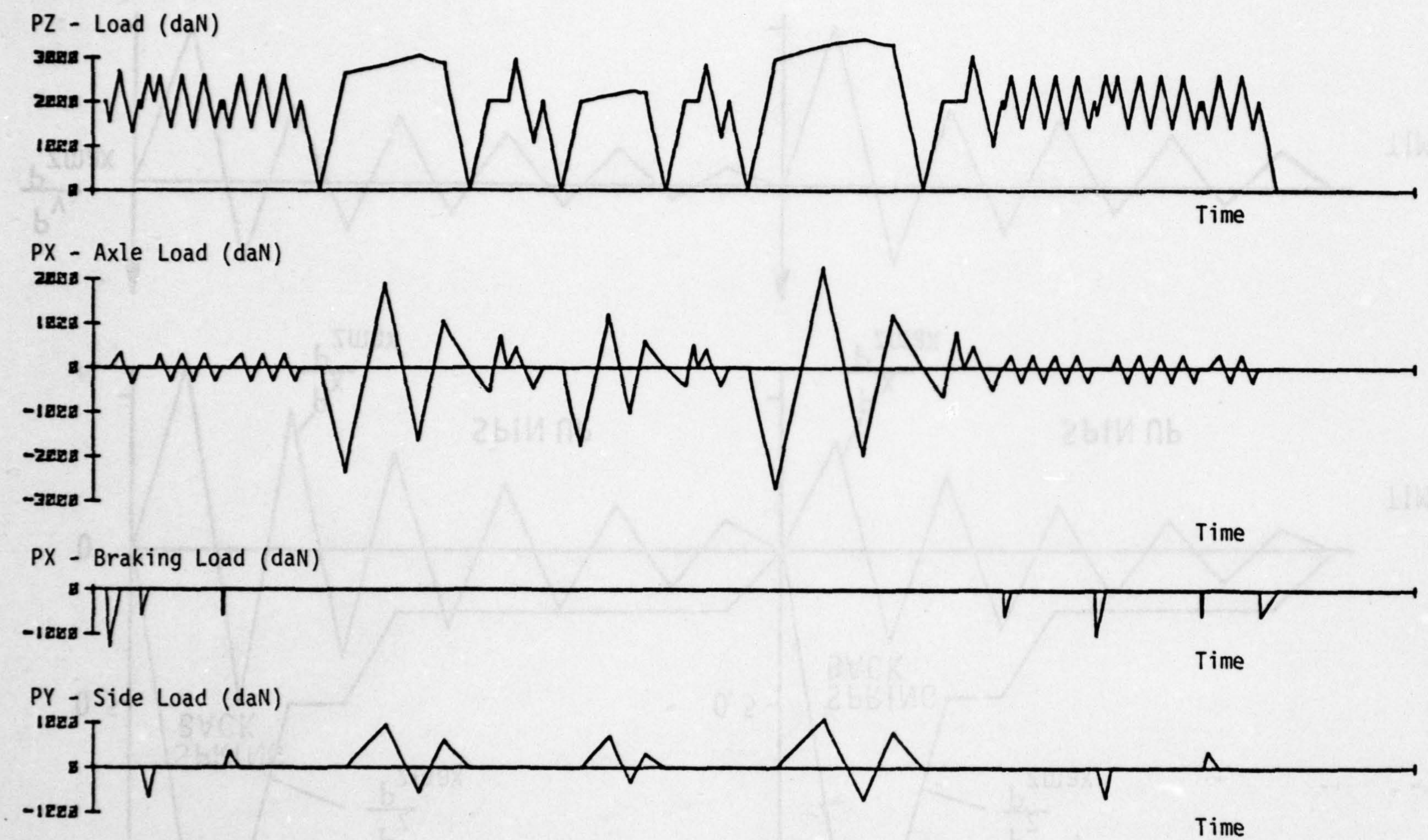


Fig.5.4.5 Main U/C – fatigue test loading program, flight type I

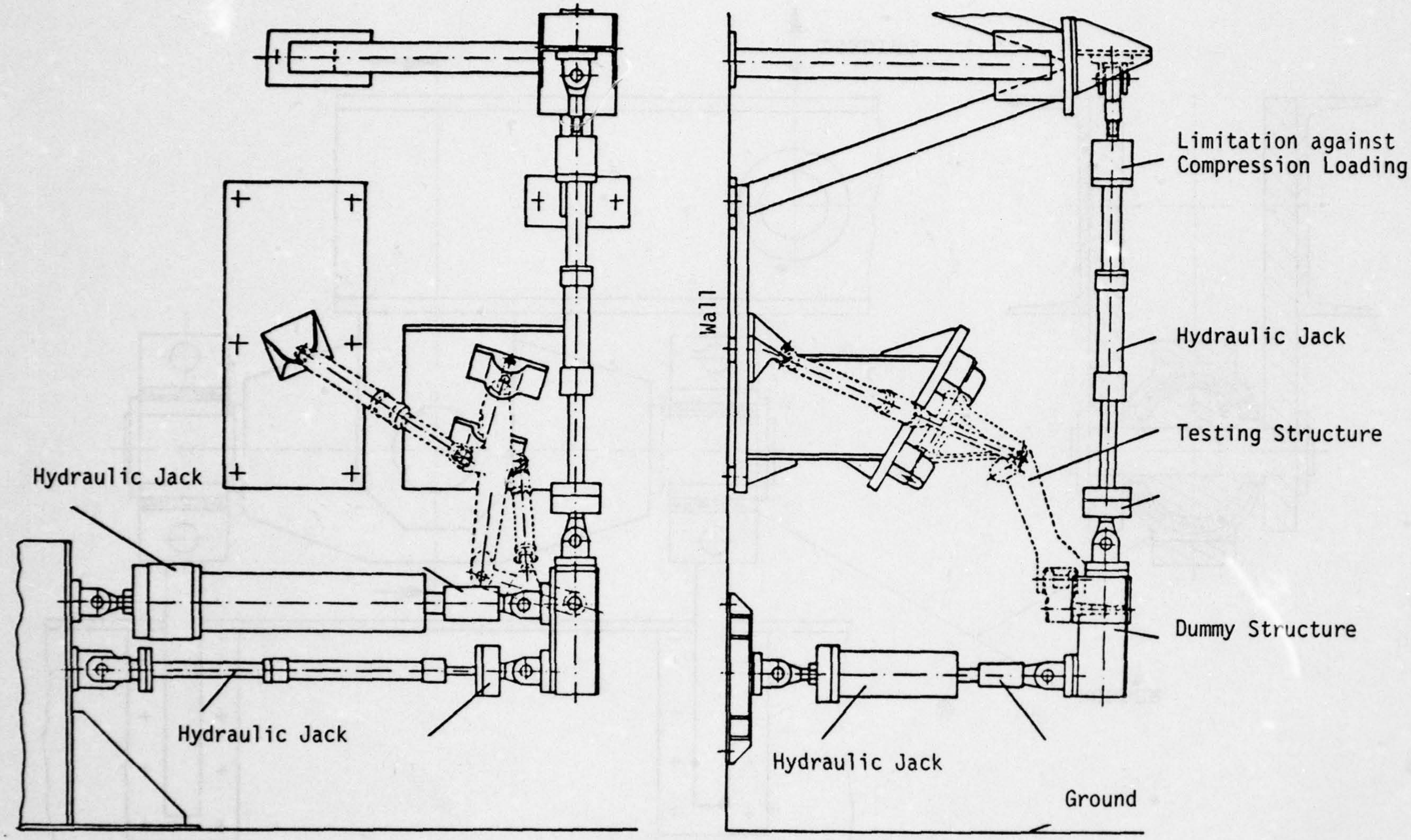


Fig.5.4.6 Main U/C fatigue test
Test set-up

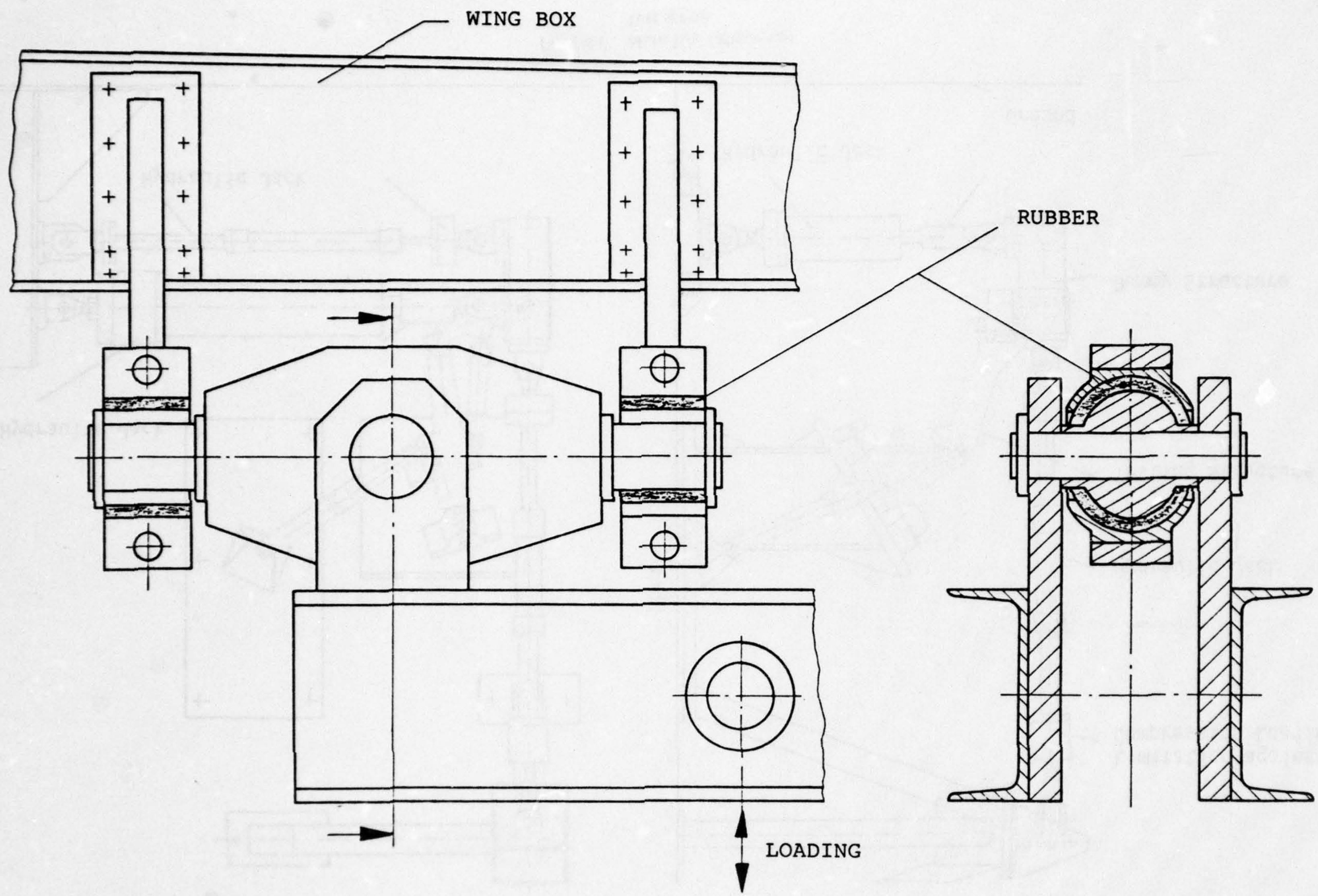


Fig.5.5.1 Load application system of fatigue test

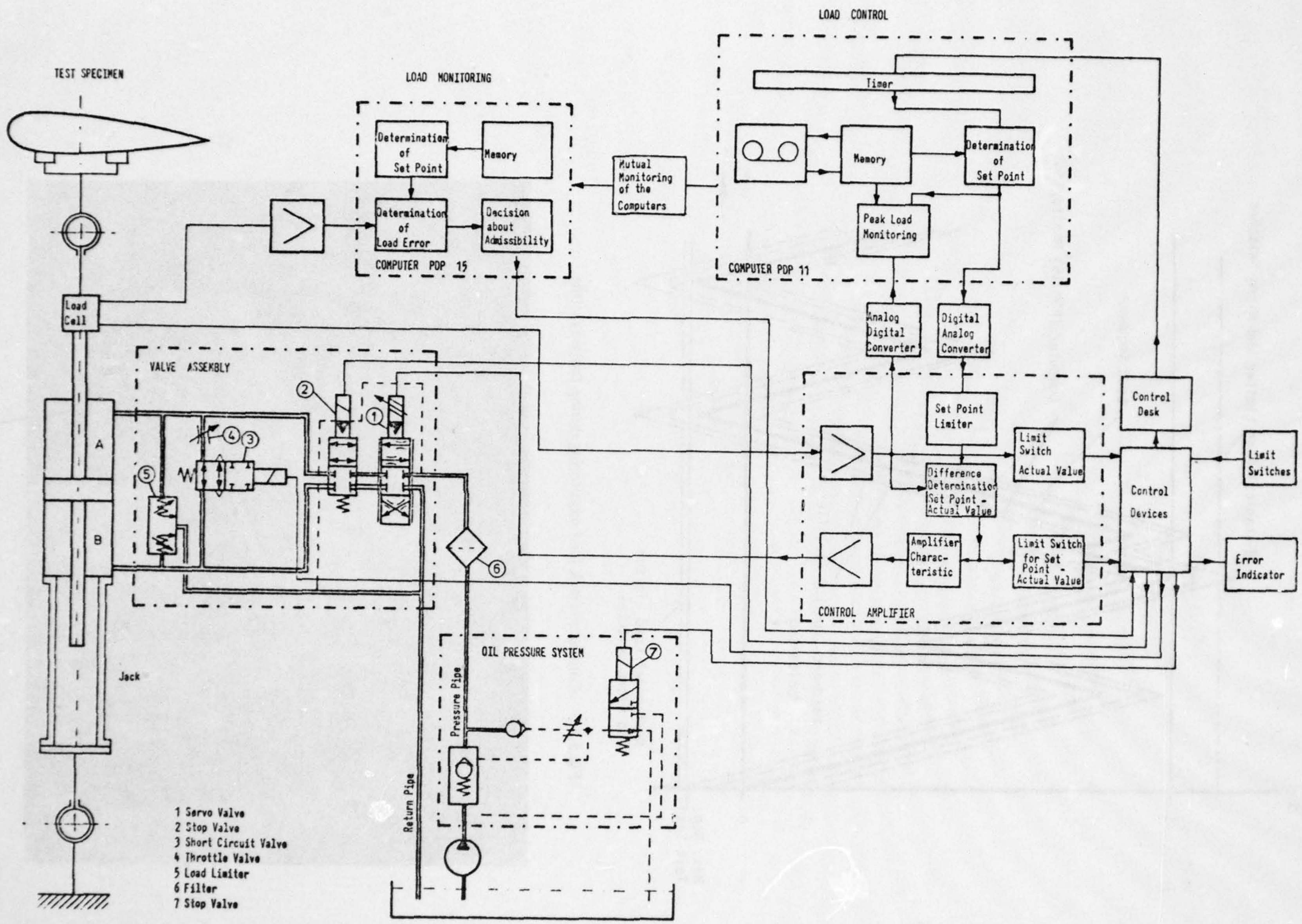


Fig.5.5.2 Fatigue test – load control and monitoring system including safety devices

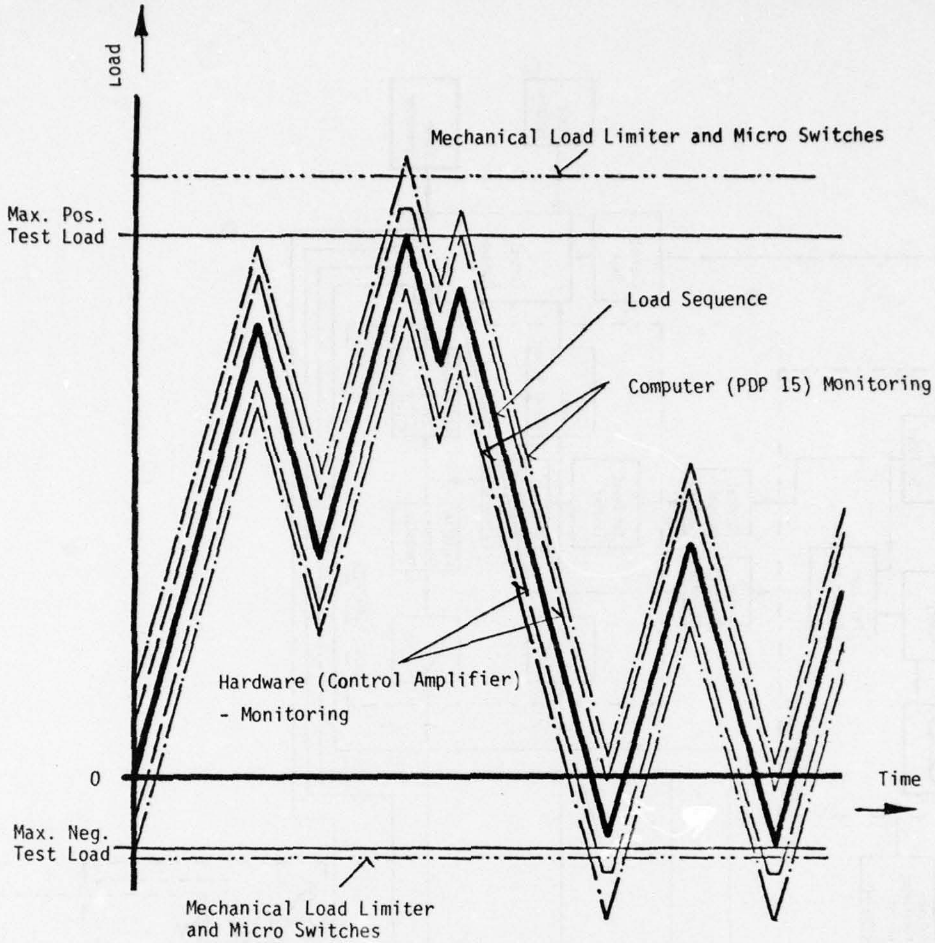
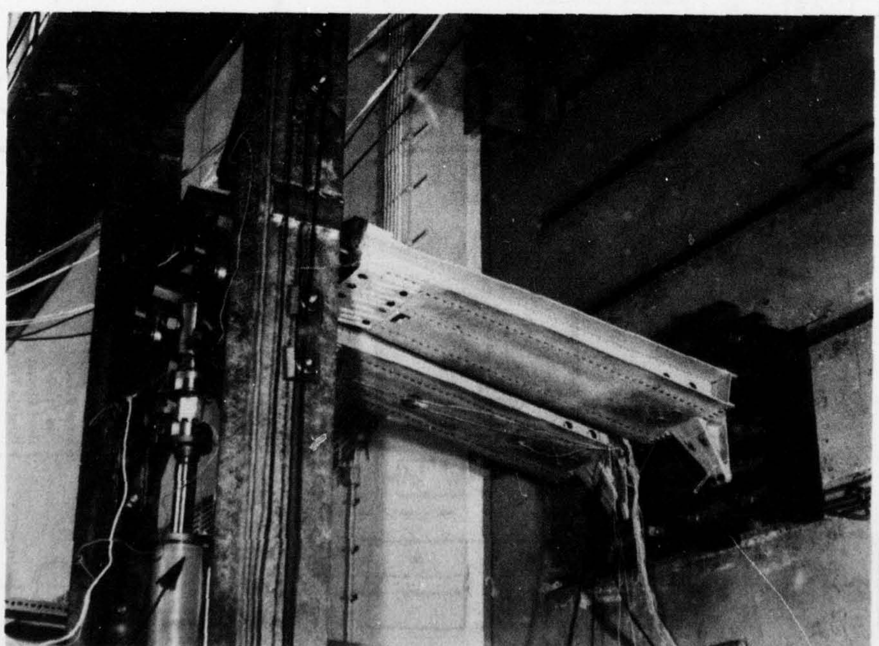


Fig.5.5.3 Continuous jack load monitoring during fatigue testing



TEST LOADING CALIBRATED AND CONTROLLED BY STRAIN GAUGES

Fig.5.5.4 Wing fuselage attachment fitting fatigue test

CHAPTER 6

CURRENT STANDARDS OF FATIGUE TEST ON STRIKE AIRCRAFT

by

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CONTENTS

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6.1 PURPOSE

Airworthiness regulations require that, in most cases, a fatigue test be carried out on the major structural items of any new type of strike aircraft. Such tests are expensive and it is essential that as much information as possible is obtained from them. In addition, many types of aircraft are operated by countries other than those in which the aircraft were designed and tested. It is important, therefore, that sufficient information on major tests is made available to enable the airworthiness authorities of all user countries to interpret the tests in terms of their own requirements and monitoring systems. In order to provide such information and to enable reassessment of the test results in the light of future service usage, it is important not only to achieve a high standard of testing but also to carry out sufficient supporting flight measurements and to present the necessary data in reports.

The purpose of this chapter is to outline, in an advisory manner, a list of the steps necessary to achieve the above objectives and to recommend those procedures, based on current knowledge, most likely to produce acceptable outputs at each step. Hence the chapter is divided into three sections:

(i) A brief statement of the objectives of a fatigue test and a list of the essential steps needed to achieve those objectives.

(ii) A summary of the recommendations of the way in which each of the steps should be carried out. This can be considered as a useful check-list.

(iii) A review of the background philosophy associated with the recommendations. This is intended to indicate the reasons for the recommended procedures and will allow individuals to assess the need to modify the recommendations in the light of subsequent research.

6.2 OBJECTIVES

6.2.1 Primary Objectives of the Test

The primary objectives of the major fatigue test are:–

(a) To expose the positions most likely to develop fatigue failures* due to design weaknesses.

(b) To provide the best estimate of the time of occurrence of such failures under a known load spectrum in order to allow monitoring in service. This implies a knowledge of local stress levels on the test specimens and at corresponding positions on an aircraft flying in all the conditions anticipated in service, to allow interpretation of the test results in terms of all types of operational flying.

(c) To provide information on crack propagation rates and residual strengths at failures. Although for aircraft designed to fail-safe or damage tolerant principles there will probably be separate tests to measure these quantities, the opportunity should be taken, whenever possible, to measure them for any naturally occurring crack.

(d) To check the behaviour of repairs; in particular to ensure that they provide an adequate remaining life.

6.2.2 Essential Steps to Achieve the Objectives

In order to achieve the objectives outlined above it is essential that each of the following steps is covered, as omission of any one of them would make interpretation extremely difficult:–

(a) The specimen should be chosen to represent as nearly as possible the production airframe.

(b) The test loading should represent as nearly as possible the loads anticipated in service. In addition, when it is considered that environmental conditions would significantly affect the result, these should be represented on test.

(c) Strain gauges should be applied at sufficient positions on the test specimen to provide reference stress points for all likely failure positions. These gauges must be calibrated against the test loads, and monitored regularly during the test.

(d) Stress or load measurements should be made during ground and flight operations on an aircraft of similar structure to that used in the test. Sufficient information should be obtained to allow the test results to be interpreted in terms of future operational usage and to provide a basis for monitoring individual aircraft fatigue damage accumulation. It is therefore desirable to apply strain gauges in positions similar to those used on the fatigue test specimen.

(e) The test should be allowd to run for sufficient time to cover the required life with an adequate safety margin making allowance for the possible evolution of the aircraft in service) including the application of the repairs. It should also provide as much information as possible on rates of crack propagation and residual strength in the presence of cracks.

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* Throughout this chapter failure means a detectable crack. It does not necessarily imply that the strength has been reduced at that time below the level acceptable for service.

(f) Reports should be written containing sufficient information to allow all the work in the previous steps to be understood and interpreted by any organisation responsible for defining safe operating and monitoring procedures.

6.3 SUMMARY OF RECOMMENDED PROCEDURES

Although, as stated in Section 6.2.2 it is necessary to cover each of the six major steps to allow the objectives of the fatigue test to be achieved, the standard of safety and reliability obtained will depend upon the quality of work at each stage. Accordingly, a summary of recommended procedures is given below, and the philosophy justifying these recommendations is detailed in Section 6.4.

6.3.1 Standard of Test Specimen

(a) All primary structure should be tested. Structure liable to produce expensive nuisance cracks should also be tested.

(b) The structure can be broken down into major components provided sufficient overlap is provided or the boundary conditions can be defined sufficiently accurately to ensure correct load distribution on all elements of primary structure.

(c) The test article should be as nearly as possible a production item containing production design features and methods of manufacture and assembly.

(d) Variations from the production structure (including differences in materials) should be recorded and the effect of the variation assessed – either theoretically or by component testing.

6.3.2 Types of Loading

(a) All loads likely to cause fatigue damage should be considered and represented. However, the effects of some forms of high frequency, low amplitude loading may have to be examined separately. Environmental conditions should be considered and represented if significant effect is anticipated.

(b) The spectrum of loads chosen for the test should aim to represent as many as possible of the flight conditions likely to be experienced in service. Where the aircraft is to be used for a wide range of operations the test should aim to represent either the most frequently expected type of operation or a combination of operations that will result in the minimum amount of interpretation to the extremes of usage.

(c) Where the spectrum of loads expected in service is wide, the maximum load level should be that occurring about ten times in one aircraft life.

(d) In defining the minimum load amplitude to be used consideration should be given to the inclusion of loads estimated to produce stress amplitudes below the fatigue limit of the component considered to be the most critical.

(e) Where the spectrum of loads expected in service is wide, these should be represented on test by as many load levels as possible with six as an absolute minimum.

(f) The load sequences should aim to reproduce those expected in the various flight conditions experienced in service. Hence the loads should be applied in a flight-by-flight sequence with realistic variations in the consecutive flights and with the various conditions within each flight in the most likely order. Load sequences within a flight condition should be random where this represents the service situation.

(g) The load distribution should represent the relevant flight conditions but should also aim to minimise the difference in rates of testing in various parts of the structure.

(h) As it is not always possible to represent all flight conditions on test, load distributions should be chosen to permit allowance to be made by calculation for such conditions that have not been represented, such as carriage of stores, changes from subsonic to supersonic flight.

6.3.3 Strain Gauge Application on Test Specimen

(a) Strain gauges should be applied in sufficient places to check the local stress distributions under the applied loads against the calculated values.

(b) Strain gauges should also be applied in sufficient places to provide reference stress levels for the assessment of any crack that occurs on the structure. These reference gauge positions should be chosen such that under the various operating conditions occurring in service the reference position stress retains a linear relationship to the stress at any failure positions within its local area.

(c) All test gauges should be calibrated under sufficient combinations of known loads to enable calculations to be made of the effects of changes of centre of pressure, all up weight, carriage of stores etc. Only if sufficient information is provided can interpretation of the test results be made for the inevitable changes of usage that occur in service.

(d) All gauges should be monitored throughout the test and significant deviations recorded. It should be noted that some changes in gauge outputs are inevitable as load redistribution occurs with joint bedding-down or slippage. It is important to note these changes for subsequent interpretation.

(e) It may be necessary to add gauges in specific areas as the test reveals unexpected cracks in order to investigate particular stressing problems.

6.3.4 Flight and Ground Load Measurement Programme

(a) In order to allow interpretation of the fatigue test result in terms of subsequent usage of the aircraft in service, sufficient strain gauges should be applied to a flight aircraft to determine the relationship between the loads experienced in various operating conditions and the stresses at the reference gauge positions used on the fatigue test specimen.

(b) All such gauges should be calibrated against known loads in order to facilitate correlation with corresponding gauges on the fatigue test specimen. If possible the calibration should be made under the application of ground loads similar to those used on the fatigue test specimen.

(c) Gauge measurements should be recorded and analysed in terms of loads under a sufficient variety of conditions to cover all regular service usage.

(d) Service load monitoring sensors should be included in the instrumentation of the aircraft to provide the relationship between the monitoring system and the fatigue test in order to allow the service monitoring data to be related to the fatigue test results.

6.3.5 Conduct of the Test

(a) The test should be continued until at least the fully factored safe-life has been proved.

(b) If the structure is capable of further testing, longer lives should be proved, possibly under increased loads, to give flexibility in service usage. In some cases it may be useful to introduce artificial cracks during this phase to evalute fail-safe or damage tolerant characteristics.

(c) Inspections should be carried out at least as frequently as those to be used in service. Where cracks occur, inspection techniques should be developed on test to ensure that the most sensitive method is available for use in service.

(d) Where repairs are necessary it is desirable that these should be to aircraft standard to enable the life of such repairs to be assessed.

(e) In order to allow a fail-safe or damage tolerant approach to be made whenever possible, the following data should be obtained provided the specimen is not endangered:

- (i) Crack propagation rate under the known test loading.
- (ii) The time for which a completely broken item was allowed to remain broken before being repaired.

(f) The test should be completed by a residual strength test:–

- (i) To expose significant undetected cracks.
- (ii) To check the strength in the presence of the fail-safe cracks.

(g) The specimen should be stored under suitable conditions for possible examination at a later date to help investigations of service problems, or subjected to a tear-down inspection.

6.3.6 Reporting

Reports on the fatigue tests and supporting work should be issued covering sufficient of the above data to enable life assessments to be made for all critical items under any load spectrum the item is likely to meet. The required information may vary depending upon the monitoring systems used by the various operators, but reports should cover at least the following items:–

(a) The methods of testing showing:–

- (i) The load sequences and distributions used and an indication of the accuracy achieved.
- (ii) The calibration data showing the relationship between loads and gauge outputs at the reference positions relevant to failures.

(iii) Details of failures, times of occurrence, form of repair, crack propagation data.

(iv) S-N data relevant to failed items, to help with subsequent interpretation in terms of service usage.

(b) The flight test data showing:–

(i) Method and results of the calibration.

(ii) Relationship between flight conditions flown, gauge readings and monitoring sensor outputs, (e.g. centre of gravity accelerometer outputs to aid interpretation of fatigue meter data).

(c) The production standard of the test specimen indicating variations in detail design and materials from the normal production item.

6.4 JUSTIFICATION OF RECOMMENDED PROCEDURES

The process of fatigue testing and applying the results to maintain safety and reliability in service is continually evolving as more understanding and new techniques of testing are developed. The purpose of this section is to explain the background to the recommendations under the six headings in Section 6.3 above. In this way, individuals will be able to assess whether any new piece of information can be used to justify deviations from these recommended procedures. The philosophy will be discussed under the same six headings as in Section 6.3.

6.4.1 Standard of Test Specimen

The current methods of design and estimation of fatigue life are such that calculations and component tests are liable to give errors both in time and location of failure[1,2]. Since initiation of failure depends critically on design detail, manufacturing standards and the local stresses, which are dependent on relative stiffnesses of the components of the built-up structure, it is essential that the test specimen is as near as possible to the production standard. If possible a specimen from the production line should be used. If this is not possible, all differences between the test specimen and the production item should be recorded and an estimate made of the effect such differences might have on fatigue performance. Preferably these estimates should be backed by detail tests.

Although the whole structure must be tested it is acceptable to break it down into large components provided that the boundary conditions can be reproduced accurately. In many cases this may mean that part of the structure adjacent to the specimen mounts will be wrongly loaded. That part of the structure cannot then be regarded as having been tested and must be included in some other specimen. Clearly this leads to the need for more than one complete airframe. This economic disadvantage may be offset by a reduction in test complexity and the fact that a failure in one component will not stop testing in the other components, which will give an overall saving in test time.

6.4.2 Type of Loading

In an extensive literature survey, Schijve[3] indicated and analysed the problems of life estimation of structures subjected to service-type loading. The general conclusion could be drawn that, in order to reproduce the various interaction effects that play a role in the fatigue accumulation process, fatigue test loading should reproduce as nearly as possible the spectrum of loads expected in service (which may be derived from a combination of roles).

In this respect, full-scale fatigue testing has undergone a tremendous development. In the earliest full-scale tests only one load condition (limit load pertaining to the critical design configuration) was applied a thousand times; in a contemporary fatigue test on the F-5E more than 2900 different load conditions are represented in a complex flight simulation test[4].

The importance of considering the representation on test of all the loads likely to cause fatigue damage is illustrated by the number of occasions on which failures have occurred in service in places not shown by the fatigue test because the critical loading was not represented[2].

However, as a modern fighter may be flown in a wide variety of flight conditions and configurations, and may be subjected to loads from many different sources such as gusts and various types of manoeuvres of different magnitude, it is clear that an infinitely large number of load conditions can be defined for each aircraft. Consequently, some simplification is necessary both for financial reasons and to reduce the duration of the test in order to have results sufficiently early to keep repair and maintenance costs to a minimum. On the other hand, over-simplification will endanger the validity of the test for its basic purpose, namely the accurate evaluation of the fatigue performance of the aircraft structure. The recommendations in Section 6.3.2 with regard to the type of loading are aimed at achieving a reasonable compromise. The background to the recommendations is discussed below.

As one aircraft type will probably be used in a variety of duties, it has to be recognised that one test may have to represent many different types of usage. This implies that for the interpretation of test results in terms of service life for any particular duty, a certain amount of extrapolation will generally be needed.

In order to minimise the amount of extrapolation, either towards 'light usage' or towards 'severe usage', it is recommended that the spectrum chosen to apply to the test correspond to an average expected usage making allowance for possible evolution of the aircraft in service. It should be kept in mind that in this way the test load spectrum need not correspond with one specific type of usage but with a combination or mixture of different usages.

Sometimes it has been proposed, in order to have a built-in 'extra safety factor' and to speed up the test, to apply test loads that are a certain percentage higher than the computed loads pertaining to the various load conditions. Experience has shown, however, that the influence of stress-increase on fatigue life under service-type loading is difficult to predict quantitatively. Moreover, the influence may be different from component to component, depending on configuration and stress level. In other words, the interpretation of test results with increased test loads in terms of actual service life may be erroneous. Consequently, it is recommended that the loads applied on test should reproduce as nearly as possible the actual service load levels.

It has been observed by many investigators that very high but infrequent loads have an important beneficial influence on the fatigue endurance due to build-up of residual compressive stresses at the notch root. Consequently, the choice of the highest load level to be applied in a fatigue test is of importance. The recommendation in Section 6.3.2 is based on the following consideration:–
The load experience of different aircraft operated in the same way will show a certain amount of variation. A load exceeded in one aircraft once in its lifetime is seen several times in another. However, on statistical grounds it may be assumed that the vast majority of aircraft in a fleet will encounter on at least a few occasions the load that is exceeded on the average ten times in an aircraft life.

In aircraft structures, load cycles with low amplitude usually occur in relatively large numbers. Consequently, the duration of the fatigue test is often largely defined by the amplitude of the smallest load cycle that is still represented in the test. In this respect, the choice of this smallest amplitude is of great practical importance. It is recognized that a quantitative recommendation is somewhat arbitrary. However, the important thing to note is that load cycles with amplitude below the fatigue limit can be damaging for more than one reason, e.g.:

(a) Due to the large numbers, they may induce fretting or fretting corrosion damage and thus enhance crack nucleation.

(b) Low-amplitude cycles may contribute to crack growth as soon as a crack has been created by higher-amplitude cycles.

(c) Stress levels omitted because they are below the fatigue limit based on an assumption of an S-N curve associated with good design or low loads may prove in practice to be above the fatigue limit when a failure occurs earlier than expected.

Consequently, low amplitude cycles should be included in the main fatigue test. In some cases of high frequency, low amplitude loading, such as acoustic excitation, where this is the predominant loading case it may be acceptable to examine the effects of such loading separately.

Having established the spectrum of loads that will be applied in a fatigue test, the test designer will have to decide upon the number of load levels to be used and the sequence in which they will be applied.

Reference to Stagg's paper[5] illustrates the problems of representing a continuous load spectrum by a number of discrete load levels. The difficulty arises because the relationship between the applied load and fatigue damage accumulated varies from station to station depending upon the local stress level and the S-N curve appropriate to each particular station. Consequently, although it is possible to define a number of discrete load levels to do the same damage as the continuous spectrum, based on some cumulative damage hypothesis, for one station, it is rarely possible to achieve this match at all stations in the structure. Analysis shows that the more levels that can be used to represent a continuous spectrum the better the representation that will be achieved.

All of the many fatigue investigations carried out in the past[3] have indicated the tremendous influence that the sequence of load application can have on the fatigue accumulation process. The cumulative effect of the various interaction effects of sequential loads with different amplitude is complex and difficult to predict. Consequently, it is generally recognized that the load sequences applied in a fatigue test should aim to reproduce those expected in actual operation; the load sequence applied in the test should be 'realistic'. In the first place this implies a flight-by-flight load sequence instead of a so-called block-type test.

In general, it should be stated that for establishing a realistic load sequence a careful mission analysis is needed. The following remarks may illustrate the type of considerations that should be made in establishing the test load sequence.

(a) Probably, in the actual usage, relatively severe missions (e.g. air-to-ground) and relatively light missions (e.g. navigation) can be distinguished. In such a case, it is 'realistic' to simulate relatively light and relatively severe 'fatigue test flights'.

(b) Each flight consists of a number of 'segments'. These segments should be simulated in the correct order.

(c) Loads within a segment may be applied in a random order where this represents the service situation. On the other hand, when in the actual situation a kind of deterministic order exists, this should be reproduced in the test. For example, in a landing case, the touch down, spin-up, spring-back and taxi-loads should be applied in the right order.

It must be recognised that the load distribution may be changed during various types of sortie due to use of fuel, carriage of stores, changing from subsonic to supersonic flight etc. Hence care must be taken to ensure that the design of test loads provides an adequate test of all parts of the structure likely to be subjected to fatigue loads during any of these variations of operation.

6.4.3 Strain Gauge Installations on the Test Specimen

The main purpose of strain gauge measurements on the fatigue test specimen is to provide the means for re-assessing the test results in terms of the various ways the aircraft is used in service. Secondary uses of strain gauges are:–

(a) To confirm calculations of stress distributions under the applied loading cases (differences may indicate unexpected secondary loads hence provide early warning of potential failure areas).

(b) To indicate changes of internal load or stress distribution during the test. These may indicate failures in the region of the gauge concerned or a bedding down of the structure. Such changes should clearly be recorded to provide a record of the cumulative loads experienced at all locations in the structure to aid subsequent analysis.

In order to permit the reassessment of a test failure for a service load spectrum different from that used on test it is first necessary to derive an S-N curve appropriate to that failure. The spectrum of loads to failure on test is known and a shape of S-N curve appropriate to the failed feature can be chosen. Provided the relationship between applied loads and local stress can be determined, a spectrum of local stresses to failure can be defined. In this context, local stress means the stress appropriate to the S-N curve chosen.

Using a cumulative damage hypothesis such as that derived by Miner, the S-N curve can be scaled in the S direction such that the correct time to failure can be obtained under the derived spectrum of local stresses. The same, scaled S-N curve and the same cumulative damage law can then be used to predict lives under any other load spectrum provided the relationships between those loads and the local stress are known. In general, the same approach can be made to determine crack propagation rates under various load spectra when the rate on test has been measured. Clearly, the relationship between applied load and local stress is extremely important, and the main purpose of the strain gauge measurements on test is to provide that relationship for the test conditions.

In general the stress value quoted in S-N curves for structural features such as notches or joints is the net section stress, so that ideally we would like to determine the net section stress as the local stress in the above discussion. However, at the start of a fatigue test the actual failure locations will be unknown and it is clearly impracticable to cover all the possible locations in this way. It is therefore considered that a reasonable compromise would be to instal gauges to monitor 'field stresses' at a reduced number of positions on the structure such that each gauge output retains a reasonably constant ratio to the local stresses of a number of potential fatigue failure positions under all the variations of load condition expected. Such gauges should be placed at positions of low stress gradient. The relationship between the measured 'field strain' (E) and the local stress (S) at the failure would become part of the scaling factor applied to the S-N curve. This relationship would then be retained for all subsequent reassessments. It might avoid confusion if the curve derived from the fatigue test were reported as an E-N curve rather than an S-N curve.

It should normally be possible to establish the relationship between the 'field strain' (E) and the overall loading on the component. The latter can be represented by a moment (M), torque (T) and shear (V) about axes at or adjacent to the location of the gauge. In general it is possible to estimate the strain (E) from a linear combination of M, T, V viz:

$$E = \alpha M + \beta T + \gamma V$$

where α, β, γ are coefficients obtained by regression analyses from a sample of strains (E) for a variety of distributed loadings. This sample can be obtained either by the applications of distributed loadings, including those selected for the fatigue test, to the test specimen or by superposition by using strain gauge data for successive applications for known single loads at selected locations on the component.

The above equation for E could also be used to estimate the significance of overall load distributions that were not applied in the fatigue test; in particular, with the associated E-N curve, it would then provide a method for reassessment of cumulative damage data using flight load data when they become available. It should be noted that the modern strike aircraft may be used in a large variety of roles involving carriage of stores, variations in fuel state throughout the flight and excursions from subsonic to supersonic flight so that reassessment of fatigue test results may have to cater for a wide range of load distributions. Hence a sufficient number of loadings should be applied to check the validity of the equation for E over a wide range of load conditions.

It may be advantageous to add gauges in specific areas when the test reveals the precise location of cracks. These will enable checks to be made of the constancy of the relationship between the 'field strain' in the area and the strain close to the crack initiation and also to check local stressing problems to help identify the reasons for early failure.

All 'field strain' gauges should be monitored regularly to ensure that the relationship between applied load and gauge output remains constant. Changes may indicate early failures or changes in local distribution due to bedding down of joints. All such changes should be recorded so that a correct total spectrum of local strain is available for analysis of failures in the area.

6.4.4 Flight and Ground Load Measurement Programmes

The main purpose of the strain gauge measurements on the flight aircraft is to provide information on the loads experienced in the various flight and ground operating conditions likely to cause fatigue damage. This is to help interpret the results of the fatigue test in terms of a service usage different from that assumed in deriving the test load spectrum. In order to reassess the fatigue test result it is necessary to know the spectrum of stress local to the fatigue failure experienced during the service usage under consideration. This can then be used with the S-N (or E-N) curve derived from the fatigue test to calculate the life required. In order to obtain this spectrum it is necessary to know the frequencies of occurrence of the operating conditions likely to cause fatigue damage and the local stresses associated with each of these conditions.

The collection of data on the frequencies of occurrence of operating conditions is clearly a long-term task and may be obtained either by the analysis of mission profiles and pilot questionnaires or preferably by instruments on board each aircraft recording parameters that can be related to the operating conditions under consideration.

The data on local stresses associated with the operating conditions can be obtained from the flight and ground load measurement programme on the development aircraft, and is a short-term exercise. This programme is usually concerned with the substantiation of the calculations made for the ultimate design strength clearance of the aircraft but the programme could be extended to investigate the loads in less severe conditions which, because of their higher frequencies of occurrence, are known to be significant for fatigue. It is important that the extended programme should provide load data for a reasonable sample of the typical operating conditions imposed by the pilot and the environment.

Clearly it would be advantageous to obtain the relationships between operating conditions and local stresses directly, but frequently the flight programme is completed before the fatigue test so no knowledge of the failure points is available. In these circumstances it is impossible to cover every possible failure origin. The next compromise would be to apply gauges to all the points at which 'field-strains (E)' are to be measured on the fatigue test (see Section 6.4.3). This would give a direct relationship to the scaled S-N (or E-N) curve derived from the fatigue test. However, even this may require too many measurement points on the flight test aeroplane. The more usual solution is to provide gauges to measure loads that can be compared with those applied to the fatigue test specimen. These would be obtained in the form of bending moment (M) torque (T) and shear (V) at selected sections of the structure. As explained in Section 6.4.3, the 'field-strains' can then be obtained from the equation $E = \alpha M + \beta T + \gamma V$, where M, T and V are the values appropriate to the section at which the 'field-strains' are measured. It is generally inadvisable to locate the gauges in regions of high stress gradients and it is possible that the chosen positions will not coincide with those selected for the monitoring of the 'field-strains' mentioned in Section 6.4.3. However, sufficient sections must be instrumented on the flight aircraft to enable interpretations to be made for all the positions on the fatigue test at which 'field-strains' may be required. Care must be taken to allow for the effects of changes of mass distribution that may occur in the various operating conditions due to use of fuel, carriage and dropping of stores etc.

Ideally strain gauge positions on the flight aircraft would be chosen so that each gauge responded to only one of the quantitities M, V or T required. This condition is not often achieved but it is usually possible to combine the outputs of selected gauges for the estimation of each quantity with reasonable accuracy. The coefficients for the combined gauges are obtained by regression analysis of the gauge outputs for a sample of calibration loadings applied to the flight aircraft[5,6]. Direct application of the calibration loads to the flight aircraft may be difficult and expensive. An alternative though clearly less accurate method would be to calibrate similar gauges at similar positions on the fatigue test specimens, and only then if the gauges are not in regions of high stress gradient.

Any instruments that are likely to be used on service aircraft to record parameters related to specific operating conditions should also be installed on the flight load measurement aircraft. This will enable the relationship between these parameters and the fatigue test result to be established via the loads measured in the flight programme.

6.4.5 Conduct of the Test

It is known that identical fatigue test specimens subjected to identical load patterns exhibit variations in time to crack initiation and in crack rate. Because of this scatter in performance, the major fatigue test, which is a test of one article, or at best two samples of a particular design feature – one on each side of the aircraft, must be carried on to a life which allows for the fact that some aircraft will crack earlier than the tested specimen. This is particularly important for aircraft designed to safe-life principles in which safety depends upon aircraft being retired or modified in critical areas before the probability of failure reaches an unacceptably high level.

For fail-safe or damage tolerant structures, which depend for safety upon the ability to detect cracks before they reduce the strength below an acceptable level, it is argued that the purpose of the fatigue test is to check that cracks do not occur so early or so frequently as to invoke an economic penalty. Consequently it is considered that for such aircraft, higher probabilities of cracking can be accepted and so the test duration can be lower (i.e. a lower scatter factor can be used). Nevertheless, it should be recognised that the fatigue test is aimed to expose unexpected weaknesses in design or unexpectedly high local stress conditions, either of which could lead to the invalidation of the substantiations used in the fail-safe concept. In addition, this philosophy normally considers the influence of a single crack, whereas towards the end of the economic life many origins may be present in one area which could modify the crack propagation and residual strength behaviour of the structure. There is therefore a strong case for testing all structures to the fully factored life.

Scatter factors vary from authority to authority, but, provided an average load spectrum has been used, the factor for safe-life aircraft is usually between three and four. If lives are to be attributed to a fleet in which the individual aircraft are not equipped with monitoring devices to determine the variation in load experienced, some allowance must also be made for this, so that some additional factor may need to be applied.

Finally it is recognised that the service environment, particularly a corrosive one, may reduce the life relative to that achieved on test. The normal effects of environment are difficult to quantify and simulate in the time scale of the fatigue test and unless severe, are not normally represented. It is usually considered that the scatter factors now in use include some unspecified allowance for this.

When the specimen has been tested to the factored design life it may still be in a relatively crack-free state. In general it will be advantageous to continue testing to provide information on the behaviour of the structure at longer lives. It is not uncommon for aircraft to be required to remain in service for longer than was originally specified, particularly if the structure is still sound. Developments in usage frequently occur in service which result in changes in stores-carrying capacity and the types of sortie flown. Hence the load spectrum changes, usually becoming more severe. It is clearly important for safe-life aircraft to continue the test as long as possible in order to provide the information on which to base extended lives. It is also advantageous in the case of fail-safe aircraft to examine the possible multi-crack behaviour and to provide some indication of likely repair frequencies to aid logistic support decisions.

During the extension of the test it may be reasonable to increase the loads by 10 to 20% in order to hasten failures. It can be argued that this is a valid procedure as extensions of life in service are often accompanied by load increases. On the other hand, as disucssed in Section 6.3.2, it is known that higher loads affect the subsequent damage, so that the pattern of failure may be changed and make interpretation difficult.

An alternative is to carry on with the original loading, introducing artificial cracks into some parts of the structure in order to collect data on crack propagation rates and so to determine the inspection frequencies necessary to apply a fail-safe approach (many aircraft designed on safe-life principles exhibit slow crack growth in some areas and can be treated as partly fail-safe). In general, since aircraft structures are for the most part symmetrical about the longitudinal centre line it is possible to examine fail-safe characteristics on one side whilst continuing to accumulate data on natural initiation of failure on the other.

Throughout the test, inspections should be made at frequent and regular intervals in order to detect cracks at the earliest possible time. This is firstly to catch the cracks before catastrophic failure occurs and so prevent the loss of an expensive specimen. Secondly it enables decisions to be made whether to repair immediately or to allow the crack to propagate to provide the data on crack propagation rates, which will be used to define the service inspection frequencies for that area. Care should be taken not to use inspection techniques, such as those using chemical applications, that may influence the subsequent crack initiation and propagation. If service inspection frequencies have been determined by calculations or component testing, the test inspections should be at least as frequent as those to be used in service to safe-guard the specimen. However, in order to obtain crack propagation rate data it is desirable to inspect more frequently on test. When cracks do occur on test, thus indicating the more likely locations of service failures, the test specimen can be used to develop special inspection techniques for those areas that will be more sensitive than those used for the normal total survey of service aircraft.

In order to continue the test after a crack has been detected and allowed to propagate as far as is considered safe, the structure should be repaired in the way in which the service aircraft will be repaired. This is to check that the repaired crack will not itself continue to propagate to catastrophe and to ensure that the repair introduces no new hazard.

When the fatigue testing is finished, the test should be completed by residual strength tests to the required level. These should check the residual strength of the structure in the cracked state, which may necessitate the removal of repairs, and will also expose undetected cracks that have reduced the strength below the acceptable level.

Finally, the specimen should be stored under suitable conditions for possible examination at a later date to help investigation of service problems, or preferably, subjected to a complete tear-down inspection to reveal undetected areas of damage.

6.4.6 Reporting

In order to facilitate the interpretation of the fatigue test in terms of service usage, sufficient information must be reported to enable reassessments to be made throughout the service life.

Hence all the information obtained in the previous steps should be presented in a form that can be readily understood by personnel working perhaps as long as twenty years after the test has been completed. This should include the S-N (or E-N) curves as discussed in section 6.4.3.that represent the failures obtained on test. In addition any other S-N data or crack propagation data obtained from component tests that may assist interpretation of the major fatigue test should be reported.

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CHAPTER 7

FATIGUE LOAD MONITORING

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CONTENTS

7.1 INTRODUCTION

The spectrum of loads assumed in fatigue analyses and applied in the full-scale fatigue test refers to an anticipated average aircraft use. The fatigue life established in such a test is obviously related to that average "design" load experience.

However, the actual load experience, and consequently service life, of individual aircraft or groups of aircraft can deviate appreciably from these design assumptions, for various reasons.

In the first place, modern fighter aircraft often have multi rôle capability and may be used in various rôles. Figure 7.1 illustrates the variation in load factor experience of German F-104G squadrons used for different duties. Successful fighter types are often being operated by many different Air Forces, each with its own geographical environment, operational requirements and tactical procedures.

Sometimes, individual aircraft are subjected to a loading environment that deviates dramatically from the fleet average; this is illustrated in Figure 7.2 for "Blue Angel" and "Thunderbird" demonstrating aircraft. These aircraft accumulated a fatigue damage equivalent to 4000 test hours during one single show season![2].

In the second place, the load experience may change drastically with time, either due to changes in aircraft rôle or due to changes in air tactics or training procedures.

Figure 7.2 shows how the average F-4 load spectra increased by an order or magnitude between 1965 and 1972.

For modern fighter aircraft with their usage versatility and desired long operational lives on the one hand and their finite fatigue endurance on the other, monitoring of operational load experience has become indispensable. Monitored loads are used for re-assessing the service life under operational conditions and the inspection intervals for fail-safe structures which are based on crack growth rates in conjunction with loading severity. Methods available for such re-assessment are discussed in Section 7.2.

In Section 7.3, various load monitoring techniques are described. General aspects of fatigue load monitoring are being discussed in Section 7.4.

The chapter ends with a number of concluding remarks.

7.2 LIFE RE-ASSESSMENT METHODS

Analytical and experimental means to establish the fatigue performance in the design stage were described in Chapters 4 and 5. It will be clear that basically these same methods may be applied to re-assess the fatigue performance under service conditions. It is essential, however, that for re-assessing the actual service life the fatigue performance as determined during the full-scale fatigue test is available as a basic "life-reference": critical areas are known and lives under the test loading environment are established. Consequently, the methods applied for re-assessment usually have a comparative rather than an absolute nature in as much as it is tried to compare the test load and service load spectra in terms of fatigue damage.

In the following, methods to make such fatigue damage comparisons, both on analytical and on experimental basis, will be briefly discussed.

7.2.1 Relative Damage Calculations

In Chapter 4 it was pointed out that all calculation methods for crack initiation currently in use are essentially based on Miner's rule. Such calculations can be done on a relative basis, in which case Miner's rule is used more or less as a "transfer function". Suppose the damage associated with the test spectrum, assumed to be the basic reference is calculated as D_{test}; the damage associated with the service spectrum is computed as $D_{service}$. A "Spectrum Severity Index" SSI can then be defined

$$SSI = \frac{D_{service}}{D_{test}},$$

relating the damages of the two spectra considered.

If the safe life under the test loading is established as $(Life)_{test}$, then the life under service conditions is re-assessed as:

$$(Life)_{service} = \frac{1}{SSI}(Life)_{test}.$$

The limitations of Miner type calculations were amply discussed in Chapter 4. Apart from the difficulty of obtaining appropriate S-N-data, the inability of Miner's rule to account for sequence-effects may be mentioned.

Reasons why the application of Miner's rule on a "relative" basis can give more accurate results than a "direct" Miner calculation are:

- When the spectra that are compared are similar and refer to the same type of flying, it can be expected that e.g. interaction effects are approximately the same in both cases. The errors in life calculations due to neglect of these effects will cross out in a comparative calculation.
- The calculation result is less dependent upon the S-N data used; the S-N curve used must have approximately the correct "shape", the absolute values will again cross out.

Satisfactory results with comparative calculations have been reported (e.g. Reference 3). However, it should be stressed that similarity of the load spectra to be compared is a necessary prerequisite for obtaining reliable results. In Reference 4, for example, a case is treated of a service aircraft with very few service hours showing a fatigue crack in a lower wing cover which was not encountered in the full scale test. Review of accelerometer data revealed that this aircraft had only been subjected to a maximum of 4.5 g. The design spectrum applied in the fatigue test included two exceedings of 7.8 g per 100 flight hours. Additional spectrum tests under a test load spectrum truncated at 4.5 g gave a reasonable duplication of the service cracks observed. Obviously, the omission of the high loads prevented the formation of beneficial residual compression stress in the notch root.

An "ordinary" Miner calculation would undoubtedly but erroneously have indicated the truncated load spectrum as less damaging than the complete spectrum.

Recently, "advanced" calculation techniques have been developed, which compute the "real" stress-strain history at the location of stress-concentrations ("notch-root"), accounting for such effects as plastic deformation and stress relaxation. These techniques are able to take interaction effects, such as build-up of residual stresses, into account. Promising results have been reported[5,6]. However, these results are usually obtained for relatively simple configurations.

For actual structural configurations of some complexity, where other effects like fretting etc. play a role, it is felt that, if the service loading deviates essentially from the design test spectra, reliable calculation results cannot be expected.

7.2.2 Comparative Fatigue Tests

Especially when the load experience encountered in service turns out to deviate appreciably from the loading applied in full-scale test, the analytical prediction of service life is bound to have a limited reliability. In such cases, additional testing will be necessary to assess the fatigue life under operational conditions.

An obvious solution is to do a fatigue test on an aircraft component, shown in the fatigue test to be critical, under load spectra encountered in service. A different solution, which is usually less expensive, may be described as comparative specimen testing.

Figure 7.3 gives a schematic representation of the procedure that may be followed. Specimens incorporating basic structural details and as such considered representative for certain structural areas, are fatigue tested both under the load spectrum applied in the full-scale test and under the monitored service load spectra. The stress level may be adjusted such that the specimen life under the full-scale test spectrum equals the full scale test results. Servo-hydraulic testing machines, in which specimen tests under complex flight load histories can be performed at relatively low cost, are now generally available. It should be stressed that comparative specimen tests can give highly accurate results. Consequently, it is felt that in many cases such comparative tests should be considered as an efficient alternative to tedious and often doubtful, analytical life calculations. However, care should be taken that the specimens used in such comparative tests are indeed representative for the structural areas considered. With reference to Barrois[14], it must be clear that a proper K_t-value and the same material alone are insufficient: appropriate stress gradients, the same surface treatments and, in case of joints, the correct fasteners and same techniques of fastening are required to obtain representative specimens.

7.3 LOAD MONITORING TECHNIQUES

Structural fatigue is caused by alternating *stresses*; obviously, an adequate fatigue load monitoring system should be able to provide information about the *stresses*, occurring in service in those structural areas that are prone to fatigue. Referring to paragraph 6.4.3, it may be recalled that the "field stresses" in critical areas rather than local stresses near stress concentrations are of interest. Paragraphs 6.4.3 and 6.4.4 described means to establish relations between these field stresses and overall loading expressed in moment M, torque T and shear S on the one hand and determination of the overall loading for various flight considerations on the other. Currently a wide variety of load monitoring equipment is being used. However, according to their working principle, two basic monitoring approaches can be distinguished, viz:

(a) *Load parameter monitoring*

Recording of load-related paramaters. Relations between these parameters and structural stresses are to be established by analysis and additional flight load measurements.

(b) *Direct stress monitoring*

Recording of stresses in critical structural areas.

In the following paragraphs both monitoring approaches will be discussed in some more detail.

7.3.1 Load Parameter Monitoring

Undoubtedly, monitoring systems based on the recording of one or more load-related parameters have found the widest application by far. Here, the expression "load related parameter" may be interpreted in a very wide sense.

For example, usually inspection intervals are being expressed in terms of "flight hours" or "flights". Obviously, a "flight hour" or a "flight" is considered as a measure, a "unit of damage". In this sense the number of flight hours may be considered as a load-related parameter. Obviously, the recording of flight-time, "keeping a flight log", may be considered as the simplest form of parameter monitoring. This type of monitoring, only involving a kind of "bookkeeping", is usually indicated as usage monitoring. More complex forms of usage monitoring include the noting down of aircraft take-off configuration, mission type flown, or even the times spent in various mission segments.

Very often, parameter monitoring takes the form of "event counting". Obvious examples of "events" descriptive for the fatigue damage to a certain structure are the number of pressure cycles for a pressurized cabin or the number of flap actuations for a manoeuvring flap mechanism. But also the widely used counting accelerometer, which will be described in detail in paragraph 7.3.1.1, counts "events", the events of interest being the exceedances of various acceleration levels. Only in very few cases, the monitoring will consist of a continuous recording of one or more physical quatities.

Essential for load parameter recording is the existance of a reasonably consistent relation between one or more "simple" parameters on the one hand and structural loads on the other. Reference 12 describing the development of a monitoring system for fin loads, illustrates the feasibility of using combinations of parameters as a loading measure.

In Reference 13, statistical methods to convert parametric data into load spectra are discussed in detail.

Means to establish relations between structural loading and one physical parameter, viz. c.g. acceleration, will be discussed in the next paragraph.

7.3.1.1 Counting Accelerometer Systems

Taking into account that according to Newton's law the external load acting on an aircraft is equal to the mass of that aircraft times the c.g. acceleration it will be clear that the c.g. acceleration is a parameter which is highly descriptive for the aircraft loading. Thus, it is not surprising that in nearly all load parameter monitoring systems c.g. acceleration is the primary parameter to be recorded.

Specifically one type of instrument, the so-called counting accelerometer, has found a very wide application as a load monitoring device. A counting accelerometer is a device which counts, for a number of acceleration levels, the number of times that each acceleration level is reached or exceeded.

Usually, a level exceedance count is not completed before the acceleration has dropped again to a secondary "reset" level which is closer to 1 g. Typical meter-setting are given in Figure 7.4.

The counting accelerometer consists of a counter unit and an acceleration sensor. The acceleration sensor is preferably mounted in the vicinity of the aircraft c.g.

Usually, the acceleration counters are read after each flight and readings filled out on a debriefing form, together with additional information about the type of flight, the aircraft configuration, etc. A typical debriefing form is shown in Figure 7.5

Counting accelerometers are relatively simple and inexpensive, and can easily be installed. Moreover, they need relatively little maintenance. However, their applicability as a load monitoring device is restricted to those cases where a reasonable relation can be found between c.g. acceleration and the loading of fatigue-critical areas.

Usually, the "stress per g" in these areas is not a constant but depends on a number of other parameters such as weight, weight distribution, Mach number and altitude.

In order to convert the acceleration spectra obtained with counting accelerometers into stress spectra, representative "average" stress/g ratios will have to be determined.

If the accelerometers are read after each sortie, as is usually the case with fighter aircraft, the g-records may be collected into groups of flights of similar mission type and the task is to determine average stress/g ratios pertaining to each mission type.

Figure 7.6 illustrates a typical variation in Wing Root Bending Moment per "g" during a flight due to fuel consumption. It will be clear that the value of the "average" load/g-ratio depends on the flight phase in which manoeuvring takes place, at what speed these manoeuvres are carried out, etc.

Undoubtedly the most accurate way of determining average stress/g ratios is by means of flight load measurements in which structural stresses and c.g. acceleration are simultaneously recorded.

It should be kept in mind that the "average" stress/g ratios to be determined are statistical quantities. For this reason it is essential that the measuring flights are representative for the actual operational usage in terms of mission profile and manoeuvring patterns flown. Whenever possible, measuring flights should be carried out as normal operational sorties, preferably flown by operational pilots instead of special test pilots.

A number of techniques may be applied to obtain the "average" stress-g relations from recorded stress-and acceleration histories. Of these, two will be briefly discussed.

(a) *Peak-correlation techniques*

The stress-records and acceleration records are searched for peaks. Stress peaks and acceleration peaks are considered to correspond when they occur at or approximately at the same instant. Corresponding peak-values are plotted against each other. The result may be presented in a matrix-format as illustrated in Figure 7.7.

Standard regression techniques can further be applied to obtain average stress/g values. Moreover, the amount of correlation between stress and g can be quantified as well as the variability of the stress/g ratio, using standard statistical methods.

(b) *Spectrum-correlation technique*

Sometimes, the stress or acceleration data are not available as complete time histories but only as spectra or exceedance-curves. In such a case, when peak-correlation techniques cannot be applied, average stress/g values can be obtained by relating those values of stress and acceleration which have the same exceedance frequency. The method is illustrated schematically in Figure 7.8.

It should be observed that this technique provides no information about the amount of correlation between stress and g; in fact the method presumes the existence of such a correlation. Hence, before applying this technique it should be made sure that the two quantities considered are indeed reasonably correlated.

Referring to the problem of variability of stress/g values it may be noted that in practice often a high degree of correlation between stress and acceleration is found despite the previously mentioned dependence of stress on instantaneous fuel weight, speed and altitude. The reason is that often the severe manoeuvring associated with a certain mission type is restricted to a limited period, e.g. at half mission time, and thus takes place at approximately constant fuel weight. Moreover, e.g. in Air to Ground attacks, all manoeuvres are made within a relatively narrow velocity band.

A typical example of such a mission type for which a high correlation between stresses and accelerations may be expected, is shown in Figure 7.9

On the other hand, the Air Combat Mission presented in Figure 7.10 exemplifies the opposite case. It may be noted that severe manoeuvres are performed in various mission phases, at various altitudes and different speeds. If the stress per g depends on fuel condition, altitude and Mach number, a weak correlation between stress and acceleration must be expected for this mission type.

In Reference 7, which gives an excellent review of the use of counting accelerometers, the use of multi-mode meters, using different sets of counters, is mentioned as a possibility.

However, it may be said that in case of poor stress-g-correlation, the counting accelerometer reaches the limits of its suitability as basic monitoring device.

So-called multi-parameter monitoring systems in which several parameters are simultaneously recorded lack the basic property of simplicity. In such cases, application of direct stress monitoring devices may be worthwhile to consider.

7.3.2 Direct Stress Monitoring

As said previously, fatigue is caused by alternating "stressing" or "straining" of the structure. The most direct way of fatigue load monitoring is undoubtedly the recording of stress (or strain) histories occurring in fatigue critical areas.

It should be repeated that this does not imply the measuring of stress at the exact "hot spot", usually a point of stress concentration where cracking is bound to start, but rather a "field strain" being a direct measure for the net section stress in that area. Direct stress monitoring presupposes the knowledge of the fatigue-critical areas. If a full-scale fatigue test has been carried out, this knowledge exists as the test must have revealed these areas. Unfortunately the timing of the major test can be such that results are not available before several aircraft have alreay entered service. Retrospective installation of monitoring devices may be very difficult. Undoubtedly, here lies one of the problems associated with direct stress monitoring. However, it should be kept in mind that usually the highly stressed structural areas are well-known and that in many cases the critical areas revealed by the major test don't come as a complete surprise.

Until now, direct stress monitoring has only found limited and incidental application. Yet, a relative large number of strain monitoring devices, using different measuring and recording techniques, have been and are being developed.

Undoubtedly, the simplest category of strain monitoring devices is formed by the so-called "Scratch gauges" or "Mechanical strain recorders". As a typical example, the Prewitt scratch strain gauge which is shown in Figure 7.11, will be described.

The gauge consists of two base plates which can be attached to a structural member. Under load the scribe arm attached to the smaller plate moves relative to the larger base plate. The stylus at the end of the scribe arm scratches in the target (disc) a mark, equal to the relative motion of travel between the two base plates.

Rotation of the target disc, caused by driver brushes, occurs only when the distance between the base plates decreases. The "scratch" corresponding with an alternating load, has a saw tooth character. With the Mechanical Strain Recorder developed by Technology Incorporated rotation of the target occurs both in the case of decreasing and increasing strain, resulting in "triangular" recorder pattern. The Mechanical Strain Recorder developed by Leigh uses a metal tape as recording medium instead of a disc.

The basic advantage of these devices are their extreme simplicity and associated very low cost. Their installation is simple: no wiring is needed, no power required, etc.

On the other hand, it should be recognized that the strain record obtained consists only of a tiny scratch on a piece of metal. The reading out of these traces is a tedious job and for applications on a larger scale, automatic equipment is indispensable.

Finally, it should be observed that the possibility to apply mechanical strain gauges is subject to a number of limitations. In the first place, to have an acceptable sensitivity, the mechanical strain gauges must be relatively large, with a measuring length of at least 3 inch. Consequently, for installation a flat surface with a length of at least 4 inches and 1 inch wide is required. Moreover, as the target disc must be replaced at regular intervals, the gauges can only be installed in readily accessible locations.

Electrical strain gauges, on the other hand, can be very small, down to a few millimetres. Experience has shown that, if carefully installed and properly sealed and protected, they may remain serviceable for practically unlimited periods of time. Consequently, accessibility is only required during installation.

In the last decade, electronics have shown a tremendous development; stable D.C. amplifiers, necessary for the strain signal conditioning, can now be produced in very small sizes and for a remarkably low price. Moreover, small size – low price recorder equipment is becoming available. Hence, a fatigue load monitoring system in which the output of one or more strain gauges is continuously recorded on magnetic tape might be considered as feasible. However, it is easy to see that such continuous recording, especially in the case of fleet-wide monitoring, would imply a very large and probably prohibitive amount of ground-based data processing.

For this reason, all monitoring systems based on strain gauges that have been envisaged so far include some form of in-flight data reduction.

A very effective and rigorous way of data reduction is "strain exceedance counting". Various "strain exceedance counters" have been or are being developed (see e.g. References 9, 10). Basically, these instruments are analogous to the previously discussed counting accelerometers. In other words, whenever the strain gauge signal exceeds prescribed values, an electromagnetic counter is activated. No additional tape recording is needed. The monitoring result has the nature of a stress exceedance spectrum and is read directly from the counters. Because of its simplicity, strain exceedance counting is very attractive. However, one should realize that level cross counting may yield results that are difficult to interpret if the signal analysed is not characterised by some constant "reference"-value such as the 1-g value in the case of accelerations. As an example, Figure 7.12 illustrates the type of level cross count result obtained for a signal with two different "reference" levels, viz. the n = 1-stabilizer bending stresses in flaps-up and flaps-down configuration respectively. It will be clear that the interpretation of this spectrum in terms of "load cycles" would already give rise to some problems. Hence, it is felt that "strain counters" are specifically suited for the reduction of strain signals that do have a well-defined average, for example the vertical tail bending moment stress, where the "average" stress is obviously zero.

A more sophisticated method of data reduction is adopted in the "Demon"-system, a prototype of which was recently built and flight-tested[11].

For conventional structural materials it may be stated that the fatigue damage associated with a certain load-history is defined by the peak and valley levels and that the influence of load duration and rate of load change are not important. Besides, it has been shown that very small load changes, smaller than some specified range-threshold are insignificant.

The reduction method of "Demon" is based on these considerations. The continuously measured strain signal is searched for peaks and valleys associated with a change of at least a specified threshold value. The values of successive peaks and valleys are transferred to the recording medium in their chronological sequence. The final result is a "load history".

The prototype system described in Reference 11 used a simple cassette type magnetic tape recorder as recording medium. As alternative, a hardware memory that is read out after each flight could be considered.

Figure 7.13 gives an illustration of the type of monitoring obtained. It should be observed that the load data obtained may be used directly as input for comparative component testing.

7.4 GENERAL CONSIDERATIONS

In the previous chapter various fatigue load monitoring techniques were discussed. Each of these turned out to have its own advantages and its limitations.

In fact, there is no single technique that can be labelled as "the best" for all applications: it depends on the properties of the aircraft and its expected usage what is the most suitable monitoring system.

In the following, various aspects that must be considered when defining a fatigue load monitoring system for a specific aircraft and operator will be discussed.

(a) *The criticality of fatigue*

The degree to which fatigue is expected to be critical for a certain aircraft type in a specific type of operation is defined on the one hand by the fatigue performance under the design load spectrum as established in the full-scale fatigue test and on the other hand by the expected severity of the operational use in relation to the design assumptions. In this respect it should be realized that especially the forces of smaller countries often merely buy their fighter aircraft more or less "off the shelf" in another country without having any influence on the design load assumptions. Obviously, the usage by such an operator can easily be very different from the "design usage".

It will be clear that the degree of criticality will have a major influence on the required accuracy with which loads have to be monitored and the complexity of the monitoring system.

(b) *The expected variability in load experience*

Very often, one aircraft type is used within one Air Force for various widely different duties, associated with very different load experience. In the second place the possible variation in load experience amongst aircraft flying nominally the same duty should be considered. As one "duty" includes various mission types of different severity, the "load experience per flight" may be considered as a stochastic variable, say $\underline{x}$, with mean $E[\underline{x}] = \bar{x}$ and variance $E[(\underline{x} - \bar{x})^2] = \sigma_x^2$. The average load experience over n flights is a stochastic $\underline{x}_n$ with mean $E[\underline{x}_n] = \bar{x}_n = \bar{x}$ and variance $E[(\underline{x}_n - \bar{x}_n)^2] = \sigma_{x_n}^2 = \frac{1}{n}\sigma_x^2$. In words, the variance of the average load experience over a number of flights decreases proportionally to the number of flights considered: hence, the statistical variation in load experience over longer periods, between aircraft flying the same duty, will be very small *provided* each aircraft flies a purely random selection out of the population of flights pertaining to that duty.

However, in actual service, unexpected large differences in average load experience are sometimes observed, indicating that flight-selection is not fully random. Causes may be e.g.:

(i) Different missions are flown in different configurations, both with regard to stores and avionics. If conversion from one configuration to the other is relatively complicated, aircraft may remain in the same configuration for a long time, and will fly the mission associated with that configuration.

(ii) Aircraft of the same type show differences in performance. If a specific tailnumber is known or just "said" to have a "good gun", the machine is chosen very frequently for Range Missions, International Shooting contests etc., accumulating fatigue damage at a very fast rate.

Other possible causes of systematic load experience variations can easily be imagined. Their presence depends heavily on the operational and maintenance procedures, applied by the specific operator.

It should be noted that the amount of loading-variability will define the number of aircraft that have to be equipped with monitoring devices. Moreover, as will be discussed further on, it may determine the choice between sample monitoring and individual aircraft monitoring.

(c) *Location of fatigue critical areas*

The full-scale fatigue test is supposed to reveal the fatigue critical spots, that means where the first fatigue cracks are expected to occur. Before this test, these areas might be predicted on the basis of stress analysis and measurement in conjunction with fatigue analysis. This will define the areas for which the structural loading will have to be monitored and from the previous chapter it will be clear that this may largely determine the type of monitoring equipment that should be used. For example, if the tail turns out to be the most critical item, it may become advantageous to apply direct stress monitoring techniques. Important in this respect is also, how many areas should be indicated as "critical", as this will influence the required complexity of the monitoring system.

(d) *Sample monitoring versus individual monitoring*

Fatigue Load Monitoring Systems can be divided into so-called Sample monitoring systems and Individual aircraft monitoring systems. In the case of sample monitoring, the purpose is to obtain an estimate of the average load experience pertaining to a certain type of operation by recording a representative sample of flights pertaining to that specific rôle.

The necessary size of that sample depends on the required accuracy of the estimate and on the variability of the loading experience per flight. As the load experience is not a stationary stochastic (usage changes with time), the sample must be drawn within a limited time period. These factors together determine the number of aircraft to be equipped with monitoring devices and sometimes this number may be relatively high.

However, the statistical nature of sample monitoring should be stressed: the monitoring result is an average load spectrum pertainint to a certain rôle; it is assumed that this spectrum is representative for the load spectra pertaining to each individual aircraft engaged in the rôle. In other words, it is presumed that a systematic variation between different aircraft, as discussed previously, does not occur.

Otherwise, the load experience of each individual aircraft is to be considered as unique: information about that experience can only be obtained by load monitoring in that particular aircraft.

This is the essence of individual aircraft monitoring: loads measured in a particular aircraft are used only for monitoring the fatigue life of that particular aircraft, and not to define an average spectrum pertaining to a certain rôle.

Individual aircraft monitoring may be described as being deterministic in nature as opposed to the statistical nature of sample monitoring.

In those cases where systematic variations in usage between different aircraft must be expected, it becomes highly advisable to apply individual monitoring.

However, in the author's opinion, individual monitoring should not be considered as superior to sample monitoring in all cases.

For the same amount of money needed to install a simple device, e.g. a counting accelerometer, in each aircraft of a fleet it may be possible to equip e.g. 15 percent with more sophisticated load recorders.

Apart from accurate service load spectra, the latter system might provide highly useful additional information about the type of loading and, in the case of spectrum changes, might indicate the causes.

7.5 CONCLUDING REMARKS

In the previous chapters, various load monitoring techniques were described and different aspects of monitoring discussed.

It turned out that no one method could be described as universally the best but that the optimum monitoring solution depends on the specific aircraft type, its fatigue properties and the way it is being operated.

With regard to expected future trends the following observations can be made.

Future generations of fighter aircraft will probably be designed according to damage tolerant or fail-safe design principles. The safety of flight for these aircraft may be guaranteed by this damage tolerance, but their structure will still have a finite fatigue life. Moreover, the inspection periods for fail-safe structures will be based on crack growth rates in conjunction with a certain usage-severity.

Considering the increasing labour cost associated with inspections, repair and modification it is believed that fatigue control by means of accurate load monitoring, both for service life-reassessment and for re-assessing inspection intervals, will become increasingly desirable.

Moreover, accurate load monitoring will yield a better knowledge of the loading environment of fighter aircraft, resulting in better design criteria for future aircraft systems.

Consequently, the use of sophisticated monitoring devices providing accurate load data will become increasingly justified. Further development of such devices, reflecting the current state of technology in electronics and recording techniques and providing a higher degree of automation in data handling and processing, should be promoted.

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Symbol	Squadron	Total Flight Time - hrs	No. of A/C Observed	Ext. Stores
o——o	A	1224	6	yes
•- - -•	B	481	4	no
□——□	C	650	4	yes
△——△	D	562	3	yes

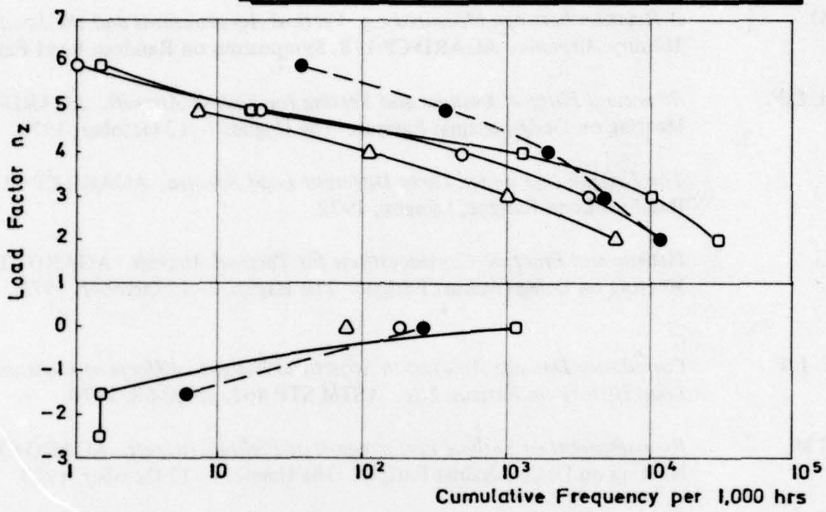


Fig.7.1 Load factor spectra of four squadrons with different duties (Ref.1)

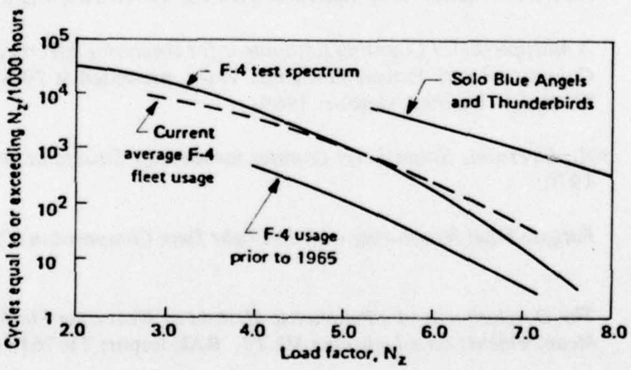


Fig.7.2 Comparison of F-4 test spectrum to service usage (Ref.2)

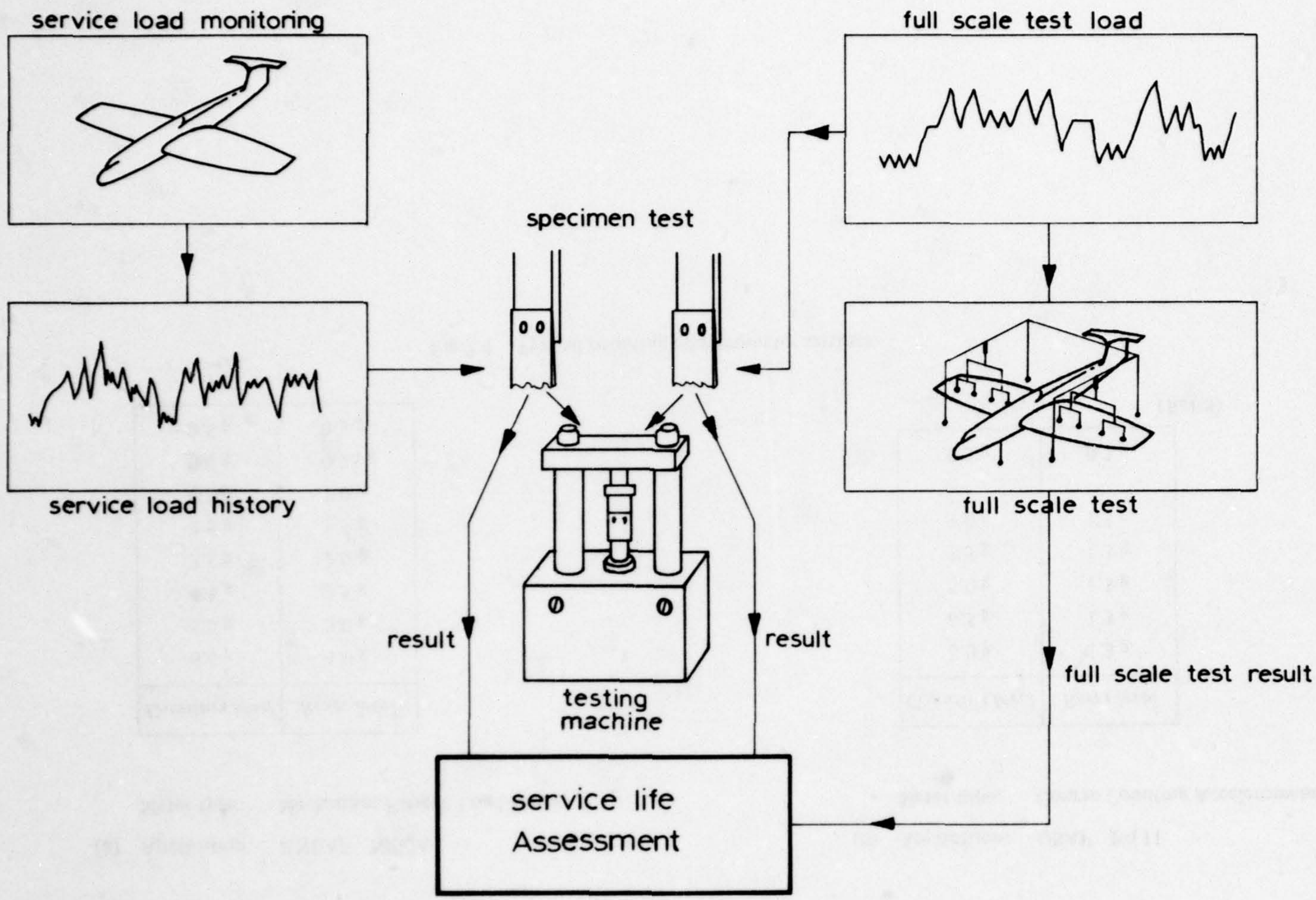


Fig.7.3 Life re-assessment by comparative spectrum tests

(a) Application: RNLAF NF-5A

Meter type: Mechanism Fatigue Load Meter

Counting level	*Reset level*
6.5 g	4.0 g
5.5 g	3.0 g
4.5 g	2.5 g
3.5 g	2.0 g
2.5 g	1.5 g
0.5 g	1.0 g
0.0 g	0.75 g
−0.5 g	0.5 g

(b) Application: USAF F-111

Meter type: Conrac Counting Accelerometer

Counting level	*Reset level*
8.0 g	1.3 g
6.5 g	1.3 g
5.0 g	1.3 g
3.5 g	1.3 g
3.0 g	1.3 g
0.5 g	0.7 g

(Ref.8)

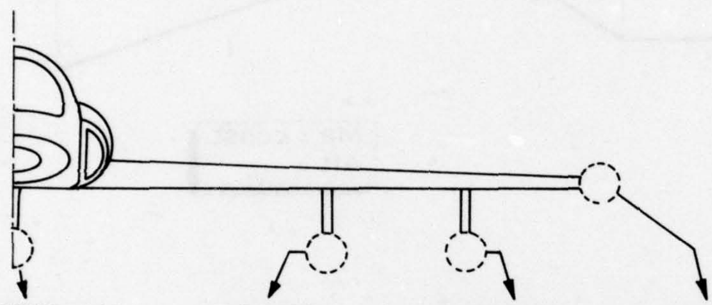
Fig.7.4 Typical counting accelerometer settings

SQUADRON : 617
LOCATION : Rotterdam AFB
TYPE OF AIRCRAFT : MIG 21

DATE 29/7/74 ACCELEROMETER NO. 1166 AIRCRAFT NO. D302

G-LEVEL	6.5	5.5	4.5	3.5	2.5	0.0	-1.5	FLAPS
landing	58	191	643	1934	4309	1552	200	720
take-off	58	190	640	1925	4289	1551	200	716
result	0	1	3	9	20	1	0	4

EXTERNAL STORES CONFIGURATION



CENTERLINE		INBOARD		OUTBOARD		WINGTIP	
pylon only	☐	pylon only	☐	pylon only	☒	dry 50	☐
dry 150	☐	dry 150	☐	....	☐	full 50	☒
full 150	☐	full 150	☒			....	☐
r.b.d.	☐	....	☐				
....	☐						

REMARKS: ....

PILOT

FLIGHT DURATION 59 MAX. G-LEVEL 6.1
T/O TIME MIN. G-LEVEL -0.5

MISSION DESCRIPTION: ACM
MISSION SEGMENTS FLOWN

NAVIGATION	☐
INTERCEPTION	☐
AIR COMBAT	☒
AEROBATICS	☒
SUPERSONIC	☐

		LIFE	SIMULATED	NUMB. OF PASSES
AIR TO AIR ☐	GUNNERY			
A/G RANGE ☐	ROCKETRY			
A/G TACTICAL ☐	BOMBING			

VOLGNO.	SQDR	AFB	AC	DATE	METER	A/C NO.	6.5	5.5	4.5	3.5	2.5	0.0	-1.5	FLAPS	CONFIGURATION CL	IB	OB	WT	FLIGHT DURATION	MISSION TYPE
1732	17	1	2	290774	66	1302	00	01	03	009	020	01	0	04	0	3	1	2	059	03

THE MISSION WAS CARRIED OUT ACCORDING TO THE A.T.P. ☒
O.C.C. ☐ → NO.....

REMARKS: None

Fig.7.5 Example of a debriefing form

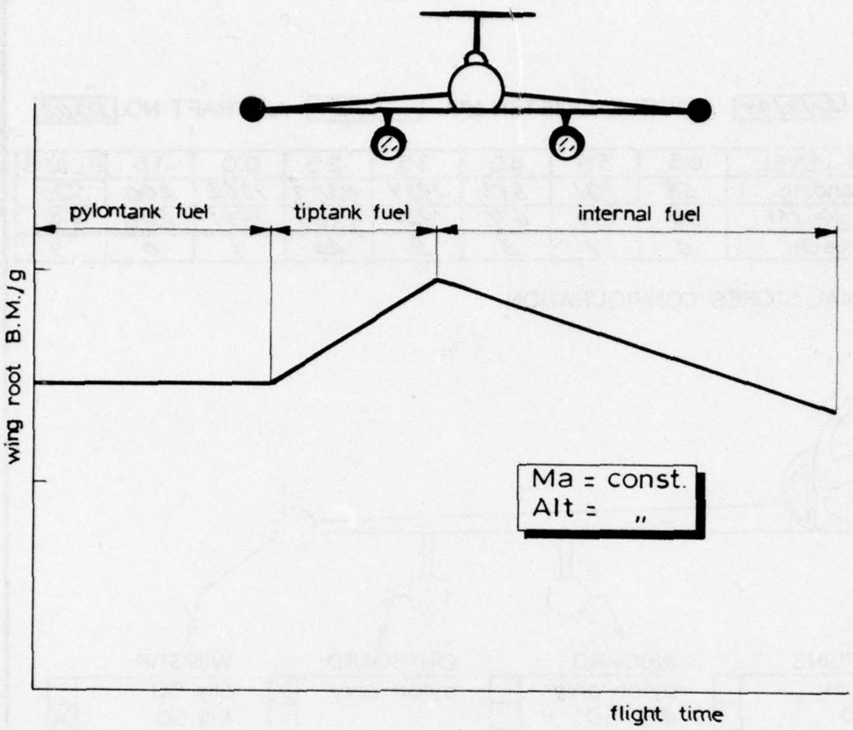


Fig.7.6 Typical variation of wing root bending moment per g due to fuel consumption

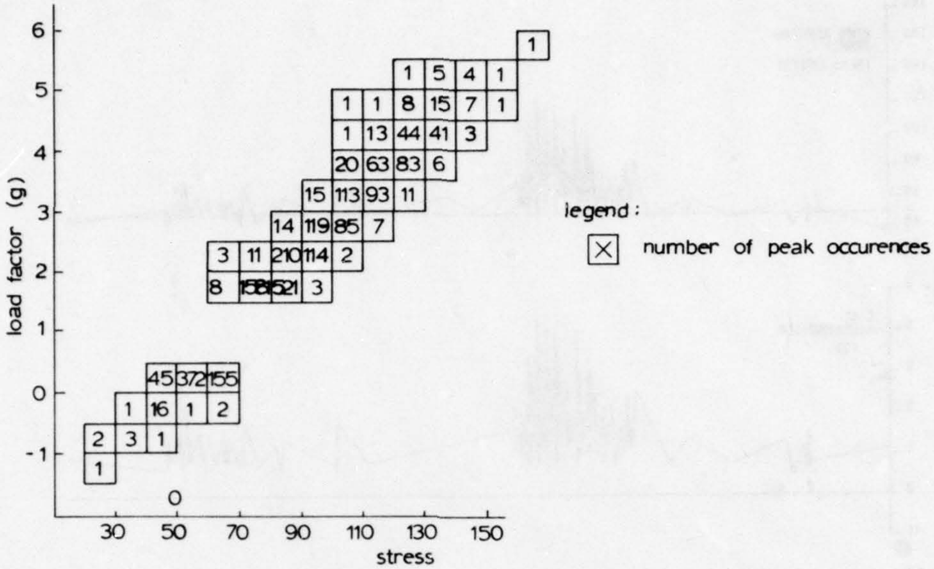


Fig.7.7 Example of peak-correlation plot

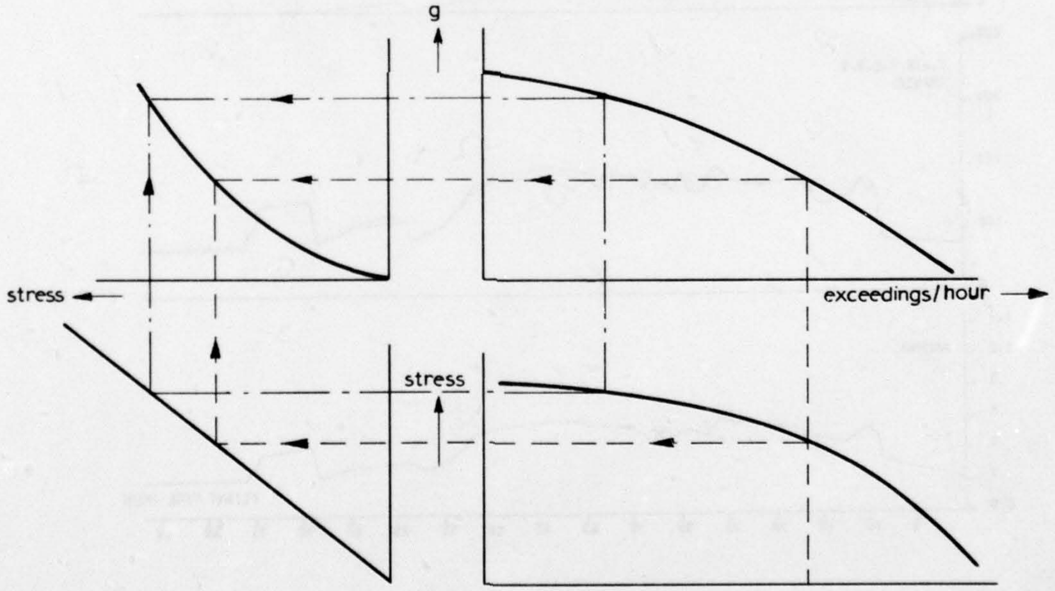


Fig.7.8 Illustration of the spectrum-correlation technique

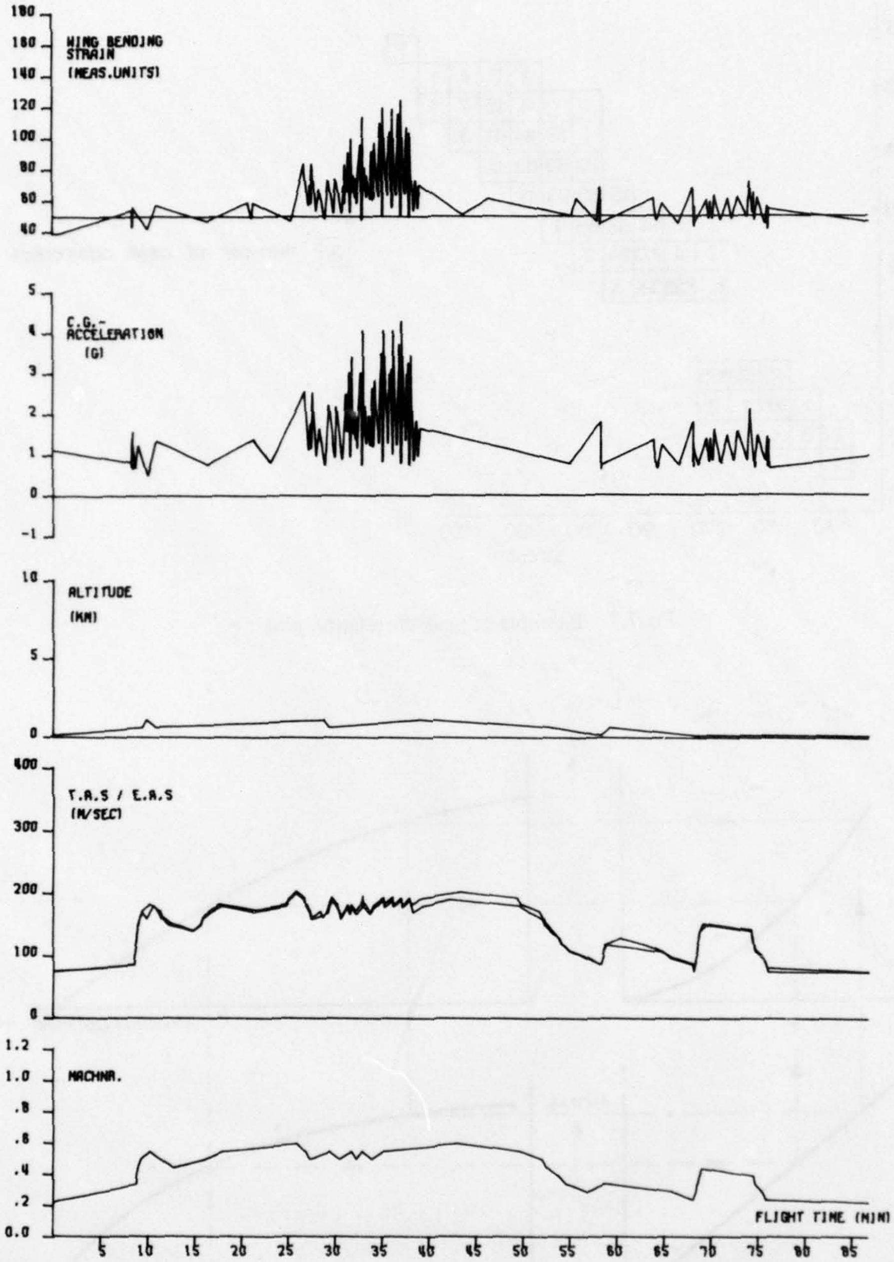


Fig.7.9 Example of air to ground mission profile

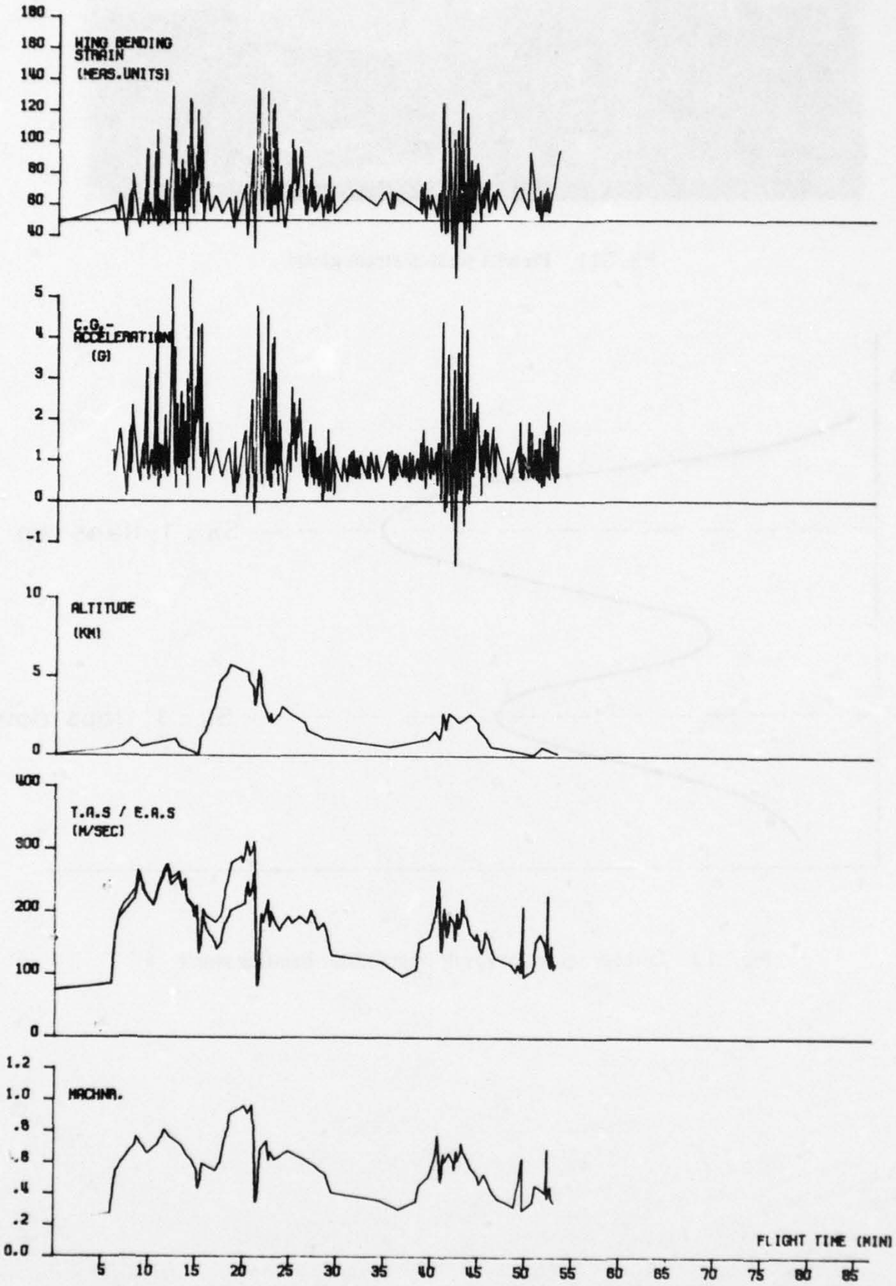


Fig.7.10 Air combat mission

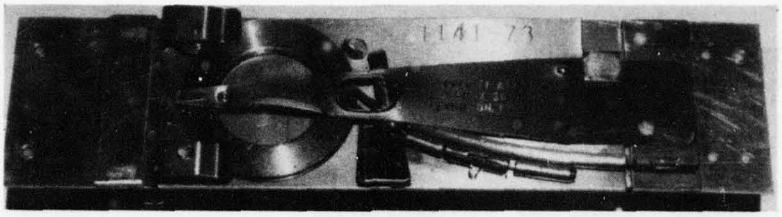


Fig.7.11 Prewitt scratch strain gauge

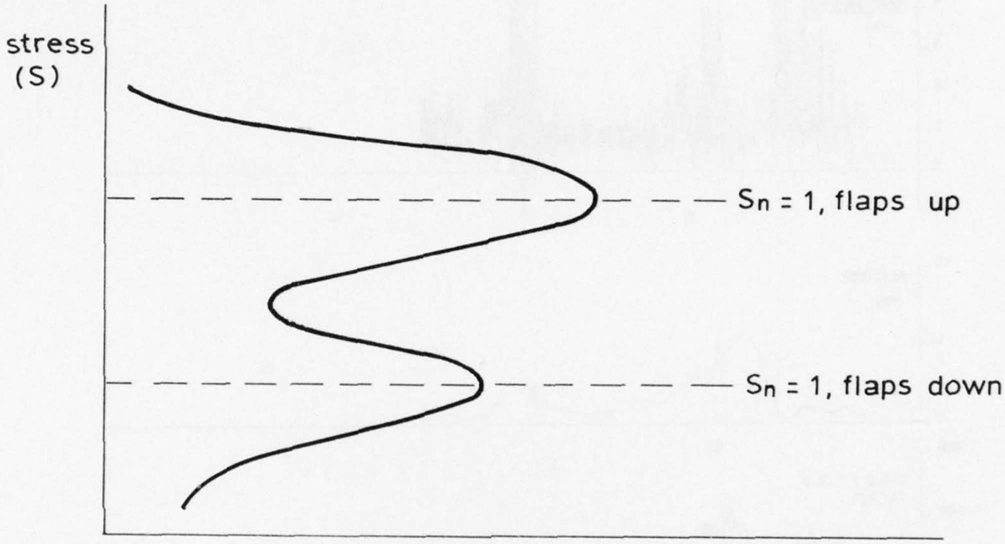


Fig.7.12 Level-cross count result of stabilizer bending stress

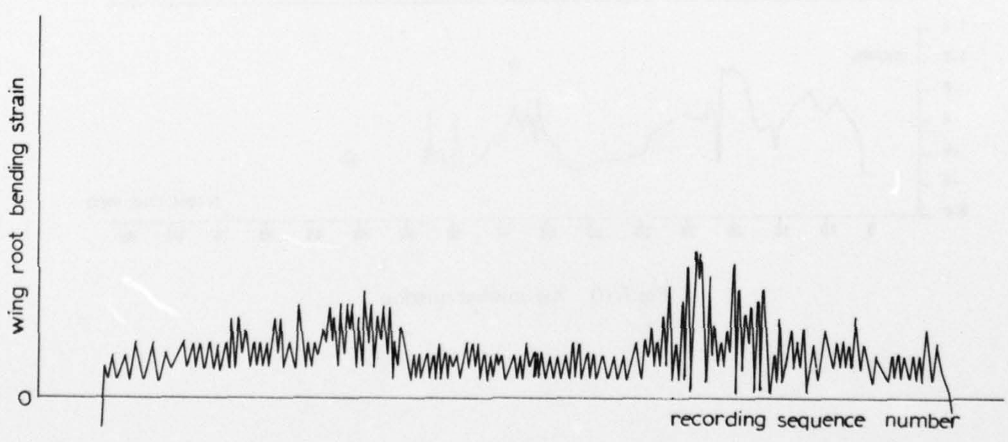


Fig.7.13 Example of load record pertaining to one flight, as obtained with "Demon"-equipment (Ref.11)

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14. Abstract

In the past fatigue has not been a particularly important aspect in the design of fighter aircraft structures. Being required to sustain high manoeuvring loads and being of relatively short life expectancy, these structures were generally designed primarily by static strength considerations. More recently, the greater complexity and cost of new weapon systems, together with the general economic pressures to control defence expenditure in the NATO countries, has required that the operational lives of fighters be increased bringing in its train an increased probability of fatigue defects and failures. Apart from the safety aspects, these fatigue defects can cause a reduction in the total state of readiness of the NATO air forces and, with the more sophisticated materials and structural forms now being employed, can result in expensive repair bills.

Recognising this situation, the Structures and Materials Panel of AGARD have explored this area of concern and have proposed, wherever possible, generally accepted procedures for its solution. The AGARDograph on Fatigue Design of Fighters provides such guidelines for obtaining and monitoring adequate fatigue performance of fighter aircraft. It is commended to the structural designers, the procurement agencies, the safety agencies and the air forces in the expectation that the wider adoption of the procedures described will result in overall improvements in the cost-effectiveness of new fighter aircraft structures.

A

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Recognising this situation, the Structures and Materials Panel of AGARD have explored this area of concern and have proposed, wherever possible, generally accepted procedures for its solution. The AGARDograph on Fatigue Design of Fighters provides such guidelines for obtaining and monitoring adequate fatigue performance of fighter aircraft. It is commended to the structural designers, the procurement agencies, the safety agencies and the air forces in the expectation that the wider adoption of the procedures described will result in overall improvements in the cost-effectiveness of new fighter aircraft structures.

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