33 YEARS OF AIRCRAFT FATIGUE

THE 7TH PLANTEMA MEMORIAL LECTURE

BY

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(1) THE 7TH PLANTEMA MEMORIAL LECTURE

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FOREWARD

It is nearly 27 years since the First International Committee On Aeronautical Fatigue Conference was held at Amsterdam on the initiative of Dr. Frederick Plantema.
Since that date, the use of civil and military aviation has developed extensively throughout the world. The continuing efforts of aeronautical research authorities and aircraft structural engineers has ensured that aircraft fatigue problems have been successfully resolved despite the greatly increased utilisation of aircraft.
The ICAF, by highlighting the problems of aircraft fatigue, monitoring and exchanging fatigue information and offering solutions to current difficulties, has played a great part in ensuring the continuing viability of aircraft operation.
I first met Dr. Plantema at the 1st ICAF symposium at Amsterdam in 1959 and then regularly at the following symposia, and I always remember how his personal approach ensured that ICAF succeeded without a formal organisation.
Therefore it is a great honour for me to present the 7th Plantema Memorial Lecture.

DR. PLANTEMA'S LIFE HISTORY (FIG.1)

Frederik J. Plantema was born on the 21st October 1911 at Leeuwarden in the Netherlands. He graduated from the Technological University Of Delft at the age of 21. For a short period he was assistant to Professor Biezeno at the same university and in 1934 he joined the National Luchtvaart Laboratorium (NLL) in Amsterdam. In 1945 he was appointed head of the Structures Department and in 1950 when the Structures Department and the Materials Department merged into one joint department he became the head of it, which he remained until his death.

Dr. Plantema worked on airworthiness problems, aircraft structures, stress analysis and related problems. He wrote a book in Dutch on The Stress Analysis Of Aircraft Structures and in 1966 published a book in the USA on sandwich construction. As a result of his interest in these subjects and particularly airworthiness he concentrated on the problems of aircraft fatigue.

In 1951 Dr. Plantema took the initiative in founding the International Committee On Aeronautical Fatigue by proposing exchange of reports and periodic meetings between aircraft fatigue institutes. A prelimnary meeting then took place on 14th September 1951 at Cranfield, UK followed by the first ICAF conference at Amsterdam in September 1952. From 1957 until his unexpected death in November 1966 he led the organization of ICAF as its secretary.

(2) INTRODUCTION

My original intention was to call this lecture "33 YEARS OF FATIGUE" but as that was too personal I have changed it to "33 YEARS OF AIRCRAFT FATIGUE" in view of the importance of aircraft fatigue in my professional career.

I had the privilege of lecturing to ICAF in 1961, 63, 65, 67 and 69 as an aircraft structural designer. Now that I am involved in the overall aircraft design sphere this lecture will consider aircraft fatigue from a much wider viewpoint.

This lecture, which is basically a keynote paper for this symposium will:-

2-1 Review the current aircraft fatigue situation.

2-2 Show the value of structural design and development from economic standpoints.

2-3 Demonstrate that the basic fatigue policy for design and development set up many years ago by ICAF is still the correct approach and furthermore show where design management should extend its commitment.

2-4 Discuss current thoughts on safe-life and fail-safe structures particularly with reference to the problems of in-service inspection.

2-5 Present some further data on the relationship between full scale fatigue test failures and in-service failures including scatter.

2-6 Comment briefly on the future including the use of active controls to reduce fatigue damage.

(3) VALUE OF STRUCTURAL DESIGN AND DEVELOPMENTS

Since the end of World War 2 there has been a great deal of research and development on all aspects of aircraft structural design and testing, particularly in the field of aircraft fatigue. It is sometimes considered that in terms of the overall expenditure on aircraft Research and Development (R & D) that structures have had too large a share.

This section of the paper deals with the value of structural R & D as it has been demonstrated by in-service performance from the several aspects shown in fig.2.

3-1 LEADSHIP LIVES, FLIGHTS AND HOURS

3-1-1 MILITARY

In general, military aircraft flight hours are not available for publication but are significantly lower than those of civil aircraft. However the important fact is that, due to financial stringency, air forces have had to continue flying military aircraft for many more years than expected. Fig.3 shows the age of several different types of military aircraft still in-service, expressing the age as years since first flight of 1st prototype.

It must be remembered that some aircraft on fig.3 are expected to be in-service to the 1990s - which means 40 years in-service.

It must also be noted that many aircraft on the figure are undertaking roles whose fatigue damage is much more severe than was called for by the design specification.

This splendid in-service record for military aircraft is due to very extensive in-service inspection coupled with, in many cases, good full scale fatigue testing.

3-1-2 CIVIL

Much more data is available on the leadship performance of civil aircraft. Figure 4 shows the leadship flying hours for eight Boeing and Douglas aircraft types plotted against their actual age since delivery for the aircraft concerned.

The leadship aircraft as regards flying hours is a Douglas DC3 delivered in 1939 and which has flown for 84,875 hours. This is an exceptional record when one remembers that the DC1 first flew in July 1933.

The DC3 design was inherently fail safe in concept and is a striking tribute to its designers.

The next leadship aircraft is one Douglas DC8 delivered in 1960 and which has flown for 70,327 hours.

Figure 4 shows that many civil aircraft types operate for more than 3,000 hours per year for long periods up to 20 years.

One interesting point is that the leadship Boeing 747 has flown for 39,811 hours in nearly 8 years which means that if this rate is continued 100,000 flying hours will be reached in 20 years of operation.

This figure not only shows the great success achieved in civil aircraft operation but it indicates the likely design targets for future aircraft design.

In the early days of ICAF, flying hours were the criteria for establishing fatigue lives but it was soon realized that flights/ landings were probably more significant in view of the fatigue damage during the ground-to-air cycle, climb and take-off. Figure 5, in the inner shaded area, plots the leadship flying hours and landing for 18 types of US, British, Dutch and Canadian civil transport aircraft. The two leadship types with over 60,000 landings are a De Havilland Aircraft Of Canada DHC-6 TWIN OTTER eleven years old and a Douglas DC9 fourteen years old.

Many of the initial discussions at ICAF concerned the relative value of fail-safe and safe-life designs. Figure 5 shows the achievements of safe-life aircraft types and it demonstrates that it is possible to design a safe-life wing structure with a good life. It must be appreciated that wing structural components are replaced at fixed intervals on these aircraft. Figure 5 also shows the expected leadship flying hours and landings for the same 18 different aircraft types if the utilisation rate of each type is continued until all have been in operation 20 years. This means that a target design life of about 110,000 landings seems realistic for a short range civil aircraft and a target design life of about 100,000 hours is realistic for a long range civil aircraft.

3-2 IN-SERVICE PERFORMANCE

Let us consider the in-service performance of aircraft as regards structural accidents.

Since more published information is available on civil aircraft, they will be discussed first. An analysis of the main casual factors for worldwide fatal accidents on public transport aircraft of above 12,500 lb. weight for the period 1962-1971 shows that airworthiness is a relatively small but constant cause at about 12% to 15% of the total. Detailed analysis of the individual accidents attributed to airworthiness shows that only 18% of these are due to the airframe, itself, so this gives a structural causal percentage of only about 2.3% in terms of the total accidents. A more recent analysis shows that the structural causalty percentage is now about 1% of the total fatal accidents. Whilst these figures are encouraging the worldwide use of older aircraft for ever greater utilizations means we must continue to be vigilant in structural design. It is instructive to analyse the type of structural failure which causes fatal accidents.

An analysis of 134 fatal civil accidents for the period 1946-1972 which were caused by airframe factors shows that 20% were static failures under design conditions and only 9% were due to fatigue. As regards military operation the following gives the percentages for structural defects relative to total defects for a recent operational period based on an analysis prepared by the RAF Maintenance Data Centre at Swanton Morley:-

Large operational aircraft	20.3%
Small operational aircraft	16.6%
Transport aircraft	24.9%
Overall mean	18.9%

The apparantly higher figures for transport aircraft is because these figures include maintenance for furnishings and interior equipment and if these are discounted the figure drops from 24.9% to 17.5%. To the civil operator these may seem high but it must be remembered that in many cases military aircraft are operating for more years than expected due to financial stringency. Safe-life military aircraft only continue to fly in-service by means of continual fatigue testing and in-service inspections. Stress corrosion and enviromental corrosion have added to military structural problems. The only evidence I can find (which is over 10 years old) suggests that the structural casulty percentage was only about 3% of total military fatal accidents.

3.3 WEIGHT INCREASES

All aircraft suffer an increase in the design take-off weight during their service life. Figure 6 shows the increase in design take-off weight for six civil and military aircraft.
These increases arise from the need to improve the aircraft's performance very significantly once the airframe is proven. The figure shows that some aircraft have increased their maximum weight by up to 100%. The rapid need to increase the weight is clearly shown for some aircraft as is also the value of being able to increase the weight late in the aircraft's life when its role changes.
These weight increases are only possible with full scale testing to extend the original design values. In many cases the design fatigue stress levels increase significantly and skill is needed to achieve the desired fatigue life.

3-4 ROLE ALTERATIONS

Changes in the military scenario have resulted in considerable increases in the actual fatigue damage experienced compared with the design value. Examples are high altitude bombers which are transferred to low level strike roles. In these cases fatigue damage may be up to four times greater than the original value.

When civil aircraft are put into service there are wide variations in operation - even within a defined role such as short range passenger carrying. Figure 7 shows the fatigue meter returns from one type of civil transport aircraft in-service and their equivalence to the full scale fatigue test conditions. There are over 300 of these aircraft in service and the shaded area embraces all Operators fleet average. This figure shows that some Operators have an average flight time of only 30 minutes compared with the test value of 105 mins. It will be seen from the figure that some Operators experience over 3.5 times the test value both in terms of once per flight and overall fatigue damage. The majority of operations are more severe than the test which indicates the need for care in setting up the test parameters.

3-5 THE COST OF STRUCTURAL DESIGN

It is difficult to obtain genuine costs for aircraft Research and Development due to different accounting methods. The following however gives a typical breakdown for the launching costs of a modern medium sized civil airliner. It has been assumed that about 500 units will be built. "Education in Production" is the total education costs assuming a typical learning curve. "Continuing Support" is the expenditure associated with aircraft development after certification such as continuing expenditure on jigs and tools and product development design. Figure 8 shows the breakdown of launching costs.

		% OF LAUNCHING COSTS		
		TOTAL		STRUCTURAL DESIGN CONTENT
JIGS AND TOOLS		22.9		
EDUCATION IN PRODUCTION		33.2		
CONTINUING SUPPORT		8.9		1.5
RESEARCH AND DEVELOPMENT				
DESIGN	10.7		4.3	
TECH. PUBS.	1.0		0.2	
STRUCT.TEST	3.7		3.7	

SYSTEM TEST	1.6			
FLIGHT TEST	2.9		0.3	
DEV. AIRCRAFT	8.6		0.9	
MIS. DEV.	6.5		1.5	
TOTAL (R & D)		35.0		10.9
TOTAL		100.0		12.4

Thus it will be seen that the structural design content is only 31.1% of the total R & D costs and if the structural design content of the continuing support is included then the total structural design content is only 12.4% of the total launching costs.

But it is the significance of R & D costs in terms of the aircraft first cost, which is important. It is estimated that for a production run of 500 aircraft the launching costs are only about 11.4% of the first cost thus the total R & D costs are only 4% of the first cost and the total structural design content is only 1.4% of the first cost.

3.6 ECONOMICS OF OPERATION

So as to evaluate the economic effects of various aspects of structural design it is necessary to consider Direct Operating Costs (DOC).

The datum project aircraft is shown in fig.9. It has a take-off weight of 52,000 lb., and its design conditions are to carry 52 passengers over three 150 n.mile stage lengths. It is powered by two geared fan engines with a very good SFC. It has a high first cost on account of its current technology.

Economic calculations for this project shows that a 10% increase in first cost gives rise to a 4.7% increase in DOC.

These calculations also show that a 10% increase in total aircraft maintenance and overhaul costs gives rise to a 2.2% increase in DOC.

The effects of an increase in structure weight are not so straight forward. An increase in structure weight leads to:-

(A) An increase in first cost of manufacture

(B) An increase in spares cost

(C) An increase in launching costs

(D) An increase in maintenance and overhaul costs

(E) An increase in fuel costs

(F) A reduction in block speed

It is estimated that with all these effects a 10% increase in structural weight gives rise to a 3.3% increase in DOC.
As regards increased service life it is estimated that a one year extension in-service life would reduce the DOC by 3.3%.
Let us assume that by increasing the structural design effort by 25% that the in-service life was increased by 2 years.
As shown in 3.5 above structural design is only about 1.4% of the first cost so a 25% increase in structural design effort would increase the DOC by only 0.16%.
A 2 year life increase would however reduce the DOC by 6.6% which shows the extra effort is more than justified. This is typical of the great ecomonic advantages which result.

4.0 THE EFFICACY OF STRUCTURAL DESIGN METHODS

This section discusses the relative value of different design and test techniques and the need to extend these activities where necessary.
Fig.10 shows the relative design activities.

4.1 AERODYNAMIC LOAD EVALUATION

Although this is a paper on structural design the interface with aerodynamics must be noted.
The most important function of the aerodynamics department towards the design of the structure is the provision of the aerodynamic loads which form the basis of the design cases. It is vital to get accurate aerodynamic loads early in the design of the aircraft. Too often the wind tunnel programme is running at the same time as the project stressing. Project Managers should always ensure that the wind tunnel tests are well ahead of the basic design. Thus, if any project is being seriously considered, it is well worthwhile spending private venture funding money ahead of the project to ensure that the loads are right.
Whilst aerodynamic analysis has improved significantly it is still important to confirm the loads in a wind tunnel, particularly with respect to three-dimensional shapes.
Load evaluation in the wind tunnel must cover tailplane, elevator, fin and rudders which have been neglected to some extent in the past.
Accurate evaluation of asymmetric loads is also desirable.

The aerodynamicist should also search for areas where aerodynamic buffeting may cause fatigue. I was associated with a twin-boomed aircraft which had very early fatigue damaged buffeting in the fin/tailboom/tailplane intersection - and a simple aerodynamic seal solved the problem for many thousands of flying hours. There has also been a recent case on a major civil aircraft where fin/rudder/ fuselage buffeting occurred with resultant fatigue damage.

As there is some evidence on one or two aircraft types that aircraft configuration changes during the life of the aircraft have altered the fatigue stress levels, the aerodynamicist is an important component in the continuing structural audit of older aircraft.

4.2 FATIGUE LOAD SPECTRA

In our 1965 ICAF paper [1] on full scale fatigue testing we showed th the fact that the actual flight usage was different to design/test assumptions was a significant factor in the discrepancy between service and test defects. This situation has improved somewhat but it is still important to establish the correct load spectra for fatigue design.

Early fatigue testing tended to exaggerate the average times and minimise the total life. It is important to define the expected operational role very carefully. One difficult area is the evaluation of the mean operational weight which depends on the expected passenger load factor or mean weapon load. I do not belie that is is worthwhile employing gimmicks like fuel management at th project stage - there will be plenty of need to use these ploys lat There is evidence that wing and fuselage load spectra can be evalua accurately - apart from dynamic loads - but other aircraft componen load spectra - such as for empennage and undercarriage are still somewhat shortweight. Research in this field is important.

One detail structural component which gives problems is hot air duc and the lives of these are very dependant to the operational techniques used.

4.3 CHOICE OF MATERIAL

It is not intended to discuss the choice of material in depth but must be recorded that the poor selection of material has very significantly increased aircraft fatigue problems over the last few years eg:-

1. The use of Zinc bearing Light Alloys on account of their higher static strength properties - but with their associated poor stress corrosion characteristics has lead to innumerable

problems in-service on military and civil aircraft. The combination of stress corrosion and fatigue has posed very difficult inspection problems because of interpretation and life evaluation. This problem is worldwide - I quote from an European report "Our Air Force has had a lot of fatigue problems with aircraft X in 1500 hours of flight and almost no problems with aircraft Y in about the same life span. We feel that a lot of this is due to the bad choice of material. The design principles of these two types of aircraft are not basically different".

2. The use of steels with poor fracture toughness and thus subsequently poor in-service fatigue performance appears to have led to the growth of damage tolerance criteria. Whilst I welcome the use of this criteria a better in-service experience might have reduced the severity of this design approach. I believe that with today's analysis and testing tools our approach to materials selection is better but we must be very careful when we extensively use composite materials to make sure we do not make the same fundamental mistakes again. In the field of composite materials it's ability to withstand thermal and humidity cycling as well as the effect of production flaws on the structure's integrity are some of the ares to be resolved.

4.4 FATIGUE STRESS LEVELS AND CUMULATIVE DAMAGE THEORY

An aircraft's fatigue characteristics are very dependent on the actual stress levels. We should examine how good the stressman is in evaluating stress levels.

There was some evidence in 1975 (based on a personal analysis of 42 structural tests - including both component and full scale) that 26% of such tests failed to reach the calculated design ultimate strength - but no tests failed to meet 75% of the design ultimate strength. These premature failures could have been due to incorrect stress level estimation or an incorrect evaluation of the allowable ultimate strength. However in view of the sensitivity of fatigue life to stress levels the analysis does show that stress level evaluation is still not completely reliable.

There is however no doubt that the increasing use of large computers using finite element analysis is now giving very accurate structural analysis for both complex reinforced shell structures and detail design features. Comparative data between stresses evaluated by computers and test evidence is difficult to present but all the stress analysts are now confident that they can reliably estimate stress levels. The large volume of work however does mean that errors sometimes occur particularly in interpretation of the design concerned.
To correlate theoretical and test fatigue lives for various operational roles the well known Miner's Law of Cumulative Damage is still the only practical approach. The value of Miner's Law is that it can be applied very easily to any structure or loading conditions and there is no real proof that the answers it gives are not accurate enough for design purposes. This is not to discourage research into alternative approaches - far from it. The key to using Miner's Law is to make sure the full scale fatigue test loading actions are as close as possible to the operational spectra.

4.5 DETAIL DESIGN PHILOSOPHY

One feature which continually arises when reviewing structural tests or in-service failures is the fact that the majority of them are simple design faults, only rarely being threshold of knowledge problems. These simple design faults probably arise from the sheer volume of work associated with aircraft design. The solutions to all these problems are well known and they repeat themselves continually. Typical examples of these simple design failings are absence of adequate corner radii, local offsets, load application in mid panel, rapid changes of cross section, interaction of more than one stress concentration, non-allowance for induced stresses, poor supporting structure on compression panels, tooling holes or identification stamps at critical sections. A typical poor detail design is shown in fig 11. In view of these repeated simple design failings it seems well worthwhile spending more time and money on basic design. Even a 30% increase in effort on structural design would only increase launching costs by 0.42% with an increase in DOC of 0.20%. The resultant improvement in maintenance and in-service life would easily offset this and the weight increase would be very small. This extra structural design effort could be employed as follows:- (Fig.12).

EDUCATION:- Teaching establishments should devote more time to the subject of creative and detail design. Industrial establishments should teach the significance of detail design to designers by reference to historical examples, in-service failures and regular visits to fatigue laboratories.

USE OF MORE SKILLED DESIGNERS:- The importance of detail design should be encouraged by making draughting a well paid career - too many designers switch to semi-management posts where their specialist knowledge is wasted. The real problem today is the shortage of skilled staff and lack of continuity of experience. The most successful aircraft design firms are those with a continuing experience of a family of aircraft. It is also hoped that the wider use of computers which will give fuller stress information throughout the aircraft to stressmen will mean he has the information - and the time - to concentrate on the quality of detail design.

SETTING A CLEAR DETAIL DESIGN POLICY:- There is no evidence that the choice of construction affects the detail design policy. Very successful aircraft have been built with bonded, machined or riveted construction. However, it is essential to decide from the start on such principles as to whether the fatigue design is going to be dependent on manufacturing techniques such as the use of coining or shot peening etc. The type of manufacturing labour used and the type of operator will influence this decision.
Alternative detail designs should also be considered. The approach to repair policy needs defining to improve in-service performance. The decision as to the extent to which fracture mechanics are used in detail design is difficult to take in view of the valuable specialist manpower required for such a task - but a fracture mechanics analysis of typical design features is always worthwhile as a guide to detail design policy.

CONCLUSIONS:- One must remember that the whole purpose of a design organisation is to prepare good quality drawings for manufacturing purposes. The pressure of the Project Manager to get drawings out on time affects quality. One of the problems is that after waiting many years for authority to proceed with a design from a customer or government sufficent time is not allowed for detail design and thus time and cost is wasted on subsequent modifications which are very expensive in paperwork, rectification and clearance.

Making sure that the first issue is the last is the most economic solution. The above are all personal thoughts but in case it is felt that I have over stressed the point I would like to quote from a US paper on a review of the results obtained through analysis, fatigue testing and actual usage of a high speed aircraft subjected to combined peacetime and combat flying, its conclusions were - "The importance of detail design cannot be over emphasised. All the certification programs in the world cannot overcome fatigue - susceptible design. In areas of high cyclic load application, design must not be compromised into creating a fatigue problem. Stress concentrations arising from holes or sharp radii and rapid changes in cross-section are to be avoided in such areas".

4.6 SAFE-LIFE, FAIL SAFE AND DAMAGE TOLERANCE DESIGN PHILOSOPHIES

Ever since the first meetings of ICAF there has always been considerable discussion on the relative merits of safe-life and fail-safe structures. The fail-safe concept, has developed into the damage tolerance philosophy (Fig.13).

Some immediate post-war military and civil aircraft had safe-life structures. A safe-life structure incorporated components which had to be replaced at a stated life to ensure structural integrity. This safe-life was usually established by a full scale fatigue test with a simplified loading. The resultant test failure was then used to establish an equivalent aircraft S/N curve. The safe-life was evaluated by relating the theoretical in-service fatigue damage with this S/N curve and then applying an overall factor of between 3.33 and 5.0. The safe-life structure had the advantage that the operator could plan in advance his structural component replacement policy and have minimal structural inspection requirements - apart from corrosion checks.

The fundamental disadvantage of the true safe-life structure was that the safety and reliability of the structure depended entirely on the ability of the designer to estimate in advance the safe-life. This evaluation was difficult in those days because of the sparse information of load spectra, shortage of data on S/N curves for typical structures and the simplified nature of the loadings on the full scale fatigue test. In addition the safe-life philosophy at that time had no provision to cover manufacturing errors or accidental damage in-service.

There were one or two serious incidents involving safe-life wing structures and about that time the concept of fail-safe design became accepted.

The fail-safe structural philosophy ensures that the aircraft structure after sustaining either accidental or fatigue damage can continue in operation satisfactorily with the ability to withstand all expected in-service loads until the next inspection occurs. Thus with one or more elements cracked or failed completely the rest of the structure must be capable of withstanding a given static load usually assessed on a probability basis as the maximum likely to occur between inspection periods.

Furthermore the rate of growth of fatigue cracks must be slow enough to give a reasonably long inspection interval to detect it with certainity before it reaches a critical length when the structure will not carry the fail-safe design load without a major failure. The problems of fail-safe aircraft structures are the difficulty of providing - and prooving - fail-safe characteristics throughout the aircraft and the vital need for continuing inspection of all structural elements so as to ensure fail-safe. I believe that the design concepts of fail-safe structure are now well known - and proven generally in-service but the question of whether operators can - and do - enough structural inspection to ensure the structural integrity of fail-safe aeroplanes is still unresolved. It is this last unresolved question which challenges the viability of the fail-safe concept.

The design of fail-safe structures has improved significantly over the years with the advent of fracture mechanics, extensive detail testing and the selection of materials for their fail-safe characteristics rather than for static strength. I believe some of the earlier fail-safe designs were incorrect in their approach by the use of multipath structures with poor access to inspection and marginal residual static strength. There is no doubt that the best approach is a low crack propagation rate coupled with the ability to inspect visually.

As shown later in this paper, modern NDT methods using X-ray, magnetic crack detection, eddy current and ultrasonic methods can guarantee to detect very small cracks indeed. But the volume of work involved in continuous NDT for complete aircraft is very high indeed even for sophisticated military and civil operators and I believe well beyond the capability of small operators in undeveloped countries.

Before we seriously consider whether we should return to safe-life structures again we should review the damage tolerance concept. Damage tolerance is really a refinement in fail-safe philosophy in that it makes assumptions on initial damage and subsequent damage coupled with a definition of the safety standards to be met. There is also consideration as to whether a structure is inspectable or not. US military damage tolerance requirements set out the sizes of assumed initial cracks, required growth periods and required residual strength. I do not intend to comment here on the levels set for assumed initial crack length but they appear arbitary and somewhat penalising in the case of slow crack growth structure which has a requirement of both a larger initial crack and the need for a longer required crack growth period. With the US military approach the non-inspectable damage tolerance structure approaches the traditional safe-life structure. Apart from the requirement levels there is no doubt that there is considerable merit in the damage tolerance approach.

A complete assessment of damage tolerance and fail-safe integrity over the complete aircraft is not possible by tests alone as these must be restricted to certain critical areas. The only acceptable theoretical method is by fracture mechanics. There is no doubt that fracture mechanics offers solutions to many crack propagation and static strength problems which were insoluable a few years ago. It is now felt that as the limitations of the technique are quantifiable there is no reason against extending its use as a design tool. However, it is a complicated analysis tool and to apply it throughout the aircraft would be costly and involve a great number of skilled - and thus scarce - structural experts. I believe fracture mechanics should be used to confirm the main detail design features such as wing joints and fuselage frame/skin intersections but the wholesale introduction of fracture mechanics and damage tolerance requirements should be carefully reviewed. It must be noted that even if a complete damage tolerance analysis is carried out on an inspectable structure at a great expenditure of design staff effort it still means that in-service inspection is still vital. Thus I believe we should reconsider the use again of safe-life structures. Modern knowledge of materials, load spectra, S/N curves for details and more representative fatigue tests mean that a safe-life can now be estimated more accurately by the conventional fatigue analysis process.

Alternatively a safe-life could be established by a crack propagation analysis with an assumed initial crack length. I only put this forward in view of my great concern on the standards of inspection possible by operators and would only limit it to critical areas where accidental damage was not possible. The weight penalty would be small but the structural integrity would benefit. These views are put forward as a discussion point.

4.7 DESIGN FOR INSPECTION

At the 1967 ICAF Mr. Lambert and I presented a paper (Lambert & Troughton[2]) on "The Importance Of Service Inspection In Aircraft Fatigue". Since that time my own and other designers experience of in-service aircraft have not only confirmed my views but strengthened them.

A structure can only be regarded as fail-safe if every part of it can be inspected and thus the disadvantage of fail-safe structures is the possible continual heavy in-service inspection time required. One real worry that has been highlighted is the ability of some marginal civil aircraft operators to inspect their aircraft thoroughly enough to guarantee fail-safe. Cracks in-service of several feet in length have been found and a later analysis has shown that they have been present for a very long time. Section 4-16 shows that there is now no shortage of reliable NDT methods available but not all operators are prepared to go to the trouble and cost of using them. This is not true in the case of military operators who are using the modern techniques with great success.

Although there is no doubt that many aircraft flying today would have been grounded without NDT, the major problem in NDT is advance knowledge of the probable location of fatigue cracks in-service. Whilst the full scale aircraft fatigue test can provide much valuable information it may not reveal the location of every naturally occuring fatigue crack possible in-service. The conclusion is therefore that it is vital in a fail-safe aircraft to design a structure that can genuinely be inpsected visually rather than just paying lip-service to the concept. The use of open section stringers and accessible structures in fatigue - sensitive and corrosion - prone areas is most desirable and great care must be taken to design multi-element members so that they can be inspected properly. Fig 14 is taken from the 1967 paper but it is still very valid today.

I still believe a fail-safe aircraft structure should be kept simple relying on low crack propagation rates to achieve a fail-safe structure rather than over complicating the design.

The two real problems regarding actual inpsection even when the design is correct is the operators approach to it and the problems of multi-crack initiation.

There is a great deal of discussion at the moment regarding continual in-service operation of older aircraft. The question of how to inform the operator correctly is not always clear cut. For sometime it was felt that by highlighting only certain critical areas some operators would only examine those areas to the detriment of the safety of the rest of the structure. There is no real doubt that the best approach is a structural audit defining very carefully the most likely type of crack, based on the theoretical and test analysis for given areas (see fig.15). Even with this type of information it is difficult to always pick up small cracks but I believe this is the best approach.

The question of multi-crack intiation is a serious problem and highlights the need for extensive fatigue testing and very good visual access to those areas.

It may be felt that my emphasis on visual inpsection is old fashioned in view of the NDT methods now available but I believe a design which can be inspected visually will be suitable for worldwide operation.

4.8 DETAIL FATIGUE TESTS

When reviewing R & D costs management always criticize the need for detail, component and full scale fatigue tests. In practice they all play their part (fig.16). Let us first deal with detail fatigue tests. Detail structural tests are used to provide design data on new types of design, confirm that the design is correct or provide airworthiness clearance for new material/techniques/configurations. At first it might be thought that aircraft design theory had progressed sufficiently that no detail tests were necessary and that full scale confirmation would be enough but the evidence does not suggest that. As stated previously good detail design is the key to success and this must be associated with detail fatigue tests - and there is some evidence that good detail testing ensures success on the full scale test.

The value of detail fatigue tests is in establishing the following:-

1. Confirming the stress concentrations at critical details.
2. Establishing basic design principles.
3. Establishing project S/N curves.
4. Proving the value of special manufacturing techniques such as interference fits special fasteners, coining etc.
5. Materials and fasteners and surface treatment selection.

6. Confirmation of fail-safe characteristics.

7. Effect of corrosion and thermal heating.

Detail fatigue tests, because of their relative cheapness, can cover alternative design concepts, include multiple specimens to cover scatter and have very genuine random loadings applied. They should be started during the project stage to advise the designer early and they should also be witnessed by the designer as part of his education.

One must not forget the need for continuing detailed fatigue tests to ensure in-service structural integrity as in-service experience reveals changes in configuration, roles and stress levels.

4.9 COMPONENT FATIGUE TESTS

As regards intermediate sized component tests such as those on engine nacelles, fuselage sections etc., in many cases, these can be dispensed with by good full scale testing. These tests may cost up to £400,000 each and need examination on a cost effective basis. Their value lies in their ability to apply more extensive representative random loadings to a given component than a full scale test. Their disadvantage lies mainly in obtaining a satisfactory representation of the balancing loads at their interface with adjoining structures. But there are certain cases where component tests are justified.

The first is in international projects where the design/manufacture split up is clear and the load interface definable.

The second and more important reason is for fail-safe tests. Current fatigue test and in-service evidence shows that fail-safe analysis alone cannot be relied upon and thus I believe that these are vital and it is very difficult to carry out fail-safe tests at several locations on the full scale fatigue test, particularly if that test is carried on for many years to get the maximum value out of it. I consider the use of large component tests such as fuselage sections etc for fail safe testing should be included in any structural audit approach.

4.10 FULL SCALE AIRCRAFT FATIGUE TESTS

I have always believed in full scale aircraft fatigue tests since my first experience on fuselage pressure testing on the Armstrong Whitworth Apollo in 1950. Mr. Harpur and I in our 1965 ICAF lecture justified the use of full scale aircraft fatigue test on both safety and economic grounds and I believe this justification is still good today (fig.17). But not everyone agrees fully with this approach. The US FAA Fatigue Regulating Review Program Amendment Working Draft (Thursday October 5th 1978) part 111 discussion on page 46238 states:-

"Two commenters recommended that full scale fatigue tests of the whole airplane structure be required, so as to insure reliable identification of those locations and detail design points at which a fatigue failure, if not detected in time, could cause catastrophic failure of the airplane. The FAA disagrees. Although full scale testing can be useful in predicting possible locations of fatigue failures, the test results do not always correlate with service experience because of differences in the loading spectrum, varying environmental conditions, scatter in the test data, and unpredictable operational effects".

It has nearly always been the UK approach to carry out full scale aircraft fatigue tests and for the aeroplanes with which I have been associated we have always used two full scale specimens - one static and one fatigue. This section reviews briefly the case for full scale aircraft fatigue tests. There are still critics who feel that with modern computers and advanced structural analysis methods that there is no real need today for full scale aircraft fatigue tests as distinct from immediately postwar when major pressure cabin and wing fatigue failures occurred.

Let us review the case for full scale aircraft fatigue testing.

(1) HIGHLIGHTING DESIGN ERRORS

Whilst one can set up, during the project phase, design philosophies and stress levels for an aircraft it is also possible, in view of the large number of drawings involved, that an individual designer may make an error of judgement in detail design and this can only be found out on a complete aircraft fatigue test. This is particularly important as most service failures are associated with poor detail design.

(2) ABILITY TO DEVELOP AN AIRCRAFT.

Earlier sections of this lecture have shown the real need to develop an aircraft design as regards weight growth and change of operational role. I am certain that full scale fatigue testing is invaluable in this respect as any hidden fatigue margins are demonstrated and can be used - which is not possible if one relies on theoretical analysis alone.

(3) COMPLETE STRUCTURAL COVER.

As fatigue failures occur throughout an aircraft's structure the wide range of component testing required is most economically carried out by testing all at once in a complete aircraft specimen. It has the further merits that loadings and structural interactions are more correct and thus failures which cannot be found by individual large component tests would be highlighted particularly in the vital wing/body region.

(4) INSPECTION SCHEDULES

From personal comparisons of in-service fatigue failures with those occurring on full scale aircraft fatigue tests and from discussions with other designers it appears that full scale fatigue tests can now be set up to give the location of most of the significant in-service fatigue failures but not necessarily the correct number of flights at which they occurred in service. Thus the full scale test gives one the ability to draft a successful inspection schedule with sufficient definition to explain the best NDT for a specific failure.

(5) CONTINUATION OF TESTING.

Since perhaps 75% of the cost of full scale aircraft fatigue testing is in the preparation of the rig and specimen it is false economy to stop testing at an early date. Management tends to stop fatigue testing once the design fatigue life is achieved with a factor of perhaps 2 or 3.1/3. It an aircraft is presumed fail-safe and the tests are discontinued at 2 times the design fatigue life one is faced with problems when the aircraft has exceeded the test life in service since the location of possible failures is unknown. Since modern aircraft are used so extensively it is vital to continue the full scale fatigue test. I believe full scale fatigue tests should be carried out as far as possible to obtain any hidden safe-life failures as well as to highlight the nature of any fail-safe characteristics. Fig.18 shows the fatigue life obtained on an extended complete aircraft fatigue test and the great increase in proven fatigue life over the design fatigue life which has been achieved by continuing the test. One other value in continuing the full scale fatigue test is that modifications arising as a result of the testing can be proven on the full scale fatigue test with very great in-service benefit. It is always important to retain the specimen after completion of the testing in case in-service fatigue defects occur later and then confirmation of specimen condition at that point is vital to analysis.

(6) PROOF OF FAIL-SAFE CHARACTERISTICS.

One of the conflicting demands during full scale fatigue testing is whether to continue the fatigue testing to look for failures or to establish the fail safe characteristics of the structure by initiating artificial cracks. I believe the best approach is to continue the full scale fatigue testing which will highlight some fail-safe failures and to establish the basic fail-safe characteristics on component fatigue tests.

(7) ECONOMIC VALUE OF FULL SCALE FATIGUE TESTS.

It will be recalled that in section 3.5 earlier that structural testing is only about 3.7% of an aircrafts launching cost or only 0.42% of an aircraft's first cost. Thus any improvement in an aircraft's in-service life will result in airline economic savings many times greater than the expenditure - apart from peace of mind.

Whilst there is a further section in this paper on in-service fatigue failures it is worthwhile here discussing the relationship between full scale fatigue tests and service experience. In our 1965 ICAF paper Mr. Harpur and I identified six major reasons for the discrepancies between service and test defects and I would like to update our comments. Let us deal with the reasons for the discrepancies one by one. The data for 1965 is from the ICAF 65 paper - only taking discrepancies outside fatigue scatter and the 1979 data is a new personal analysis from worldwide data (64 examples) again only taking discrepancies outside fatigue scatter.

(A) The first reason in fig.19 as to why service experience does not correlate with test results is the fact that flight usage is different from test assumptions. I am certain that this reason will disappear someday as there is now a much greater appreciation of actual operational usage and its significance in fatigue. But is is still a major reason today.

(B) The second reason in fig.19 covers the case when incorrect loads have been applied on test due to aerodynamic (or in a few cases structural) miscalculations. This is a continuing cause and can be of importance. I believe that the extended use of structural integrity flight testing will remove this reason - but the effect of late configuration changes on aerodynamic loads must be watched carefully.

(C) The third reason in fig.19 is still the biggest factor in the discrepancy between service experience and test results. It covers all the cases where the type of loading which caused the failure was not applied at all on test. If one added all the possible detailed loading cases to the full scale test the testing time would extend unacceptably. A typical example is the imput of undercarriage loads in all respects. I believe that the more extensive detail and component fatigue tests would cover this point. I also consider that today's full scale fatigue tests now cover loading spectra much more comprehensively than in the past as regards basic structural loads. However, the real problem that cannot be represented on the full scale test is the absence of local aerodynamic loads. These are a real problem as any attempt to add local aerodynamic load effects on a full scale test would soon be prohibitive in cost and time. However there have been serious in-service defects due to aerodynamic buffeting or acoustic effects. The only solution to these problems is a careful study of possible problem areas (propeller slipstreams in wing/body/tail junctions or jet efflux regions), first class detail design using proven methods with detail and component fatigue testing.

(D) The fourth reason cover the case when the test specimen was not representative of the aircraft in some respect due to build differences, modification etc., or simplified specimen. This reason should not really occur but test specimens sometimes incorporate non-standard materials or manufacturing techniques as a result of short cuts or early build standard. The specimen is vital to the success of the project and must be manufactured as a typical product. One more important point that has been highlighted recently is the need for updating the test specimen with in service modifications which may affect fatigue life. Specimens appear more representative these days but this reason is still very valid.

(E) The fifth reason covers the case where service environment is more severe that the test environment. This was a rare occurance and covers the presence of local overheating or corrosion. It cannot be represented on test. There is some evidence that this reason is increasing particularly due to corrosion.

(F) The sixth reason has been added as certain defects have been highlighted at an earlier stage by service inspections.

(G) The seventh reason covers the cases where the Aircraft Constructors have not found a conclusive reason.

However, the above discussion is only for when there are discrepancies. A great deal of evidence has been supplied where the full scale fatigue testing agrees well with the in-service experience. Fig.20 shows how by using the correct full scale fatigue test loading it is possible to reproduce in advance the structurally important service crack locations.

4.11 STRUCTURAL INTEGRITY FLIGHT TESTING

For many years delegates to conferences on aircraft structural design have always praised the value of structural integrity flight testing. But, in practice, only lip service is paid to the real principles of structural integrity flight testing. Since actual flight measurement of structural stresses is the only real proof that the aerodynamic and stressing analysis is correct it should always be undertaken.

Section 5 above has shown that the cost of all flight testing is only 2.9% of the total launching costs. If we assume that an extra 30% flight testing was carried out for structural purposes the increase in the aircraft first price would be 0.10% with an increase in DOC of only 0.05%. The confidence obtained by these flight measurements is seen to be obtained very cheaply.

Structural integrity flight testing can be used to obtain the following (fig.21).

(1) Confirming design loads: by combining ground calibration and in-flight stress measurements it is possible to confirm design load assumptions. It particularly enables aerodynamic assumptions to be confirmed and experience has shown this to be necessary. It also enables loads to be evaluated for components apart from the wing and fuselage such as the empennage and undercarriage where load evaluation is difficult.

(2) Establishing loads not considered during design - although Airworthiness Authorities cover most of the design cases which generally occur, flight stress measurements sometimes uncover unexpected loads. A typical case occurred with an outboard aileron hinge failure. Flight measurements showed that the design fatigue case was in fact the inertia loads during taxi-ing on unprepared airfields rather than the in-flight aileron loads.

(3) Establishing stress levels. Apart from confirming the actual loadings by calibration, the local stress levels are established throughout the flight regime. This can be very important in critical structural regions in establishing very local stress distributions.

(4) Dynamic load evaluation. The theoretical evaluation of dynamic loads such as the wing stresses experienced in a gust is not at all reliable and thus continuous recording of in-flight stress measurements together with other parameters is the best way of establishing genuine dynamic loads.

(5) Obtaining load spectra. With continuous recording of stress measurement it is possible to obtain load spectra which were previously unobtainable such as that experienced by undercarriage and empennage. This is important in structural integrity analysis as these components have not normally been extensively tested in fatigue.

(6) Monitoring the effects of configuration changes. During the in-service development of aircraft, configuration changes may occur without the associated wind tunnel tests and in-flight stress measurements are a good check on the implications of these changes.

(7) Establishing fatigue life. I believe one very good solution to establishing an aircraft's fatigue life is to continuously monitor the in-flight stresses over a long period. I have been associated with an exercise where a large military aircraft has been flown for one year with continuous monitoring of wing stresses. Fig.22 shows the estimated fatigue life for this aircraft for typical flights comparing the life based on the stresses recorded with that estimated from the fatigue meter returns for the same flight. Other valuable information gained by this year's flying has been the variation of fatigue life for operation from different runway surfaces and the true value of the ground-to-air cycle. I believe that by comparing the stress spectra obtained during the year's operation with that applied to the full scale fatigue test it will be possible to get a very accurate estimate of the aircraft's fatigue life particularly for a given operational role. Structural integrity flight testing requires good ground calibration under known loads and fig.23 shows a BAe Tornado undergoing ground calibration tests.

4.12 IN-SERVICE FATIGUE FAILURES

Analysis of in-service fatigue failures is always difficult in view of the limited information available which is rarely presented in a consistent form and the need for commercial or military security. There has always been a requirement within the ICAF for consistent information on test and service fatigue failures for comparative purposes.

The only recent rigorous analysis that I am aware of is one carried out by Mr. J. Forsyth of the UK Royal Aircraft Establishment who kindly undertook a statistical analysis on eight aircraft especially for this lecture.

Figures 24 and 25 show the results of the statistical analysis by Mr. J. Forsyth on thirteen fatigue failures which have occurred in-service. This analysis is based on the procedures laid down by Mr. Stagg of the UK Royal Aircraft Establishment (Stagg [3]) using the service failures together with the non-failures at the same location. The principle of maximum likelihood has been used to determine the best fit conditions for the failed and non-failed components in a normal distribution of log life. From the analysis the LOG_{10} SD, mean fatigue life and coefficent of variation (CV) have been assessed. The calculated mean in-service fatigue life for each failure has been taken as 100 to preserve the identity of the aircraft concerned. The safe-life has also been calculated using 3 SD below the mean. It has been the UK military practice to evaluate safe lives by applying a standard factor on life (5 if loads are not monitored or 3.3 if load monitoring is used) and the appropriate safe lives based on this policy are also given. Correlation has been assessed on the basis of either flying hours of fatigue index where most applicable. The time at which the first in-service crack was found is also given in terms of calculated mean in-service fatigue life. In most cases the cracks when found were long and hence the assessments give scatter on life to various crack lengths. It will be noted from the figures that the coefficient of variation covers a range from 1% to just over 11%. Current safe-life assumptions are that LOG SD is constant and with a 3.1/3 life factor its value is 0.176 and it will be seen that this is in line with some results. The lessons to be learnt from this analysis are:-

(1) The UK approach of using a statistical life factor 3.3 or 5.0 on the mean fatigue life as a means of evaluating safe life compares well with the full statistical analysis of actual in-service experience.

(2) The complete statistical approach of establishing a safe life by evaluating the mean fatigue life minus 3 SD gives an accurate guide to the first in-service crack. Thus by analysing both the service failures and non failures it is possible to set up a reliable inspection procedure once cracks have been found in-service. It should be noted that the analysis should include the non-failures as well as the failures since a statistical analysis on the failures discussed previously but only using the failures and not including the non-failures gave a reduction in safe lives to between 32% and 100% of those established with both failures and non-failures. Whilst this would be a safe approach it would not be economic.

Before leaving Mr. Forsyth's analysis it seems worthwhile comparing the in-service safe life estimated from the statistical approach with those promulgated previously from the full scale fatigue tests. Fig.26 shows the safe life (mean - 3 SD) for seven of the cases on the previous two figures with the promulgated safe life based on the full scale fatigue tests. It will be seen that in only one case is there good correlation and in practice the test value over estimates the actual fatigue life in service by up to six times. However, the really interesting point is that the test accurately locates the failure and that the reasons for the discrepancies all lie in the unrepresentative test programme relative to the actual service usage. I still believe that once actual load spectra are established and used on full scale fatigue tests these discrepancies will reduce and full scale fatigue tests give the correct results.

Not all correlation between test and service experience is as poor as shown in the previous figure. I would like to show the relationship between test and service experience for a fatigue failure - designated crack A in fig.27. The value of this case lies in the number of examples used in the statistical analysis - 138 uncracked locations and 64 cracked. The resultant mean in-service fatigue life (LOG_{10} SD = 0.162) is shown together with the $\pm$ 3 SD limits. This in-service life is based on the cracks when first detected. It should be noted that the first crack detected in-service is within the safe life obtained satistically.

The very good relationship between the mean service life and the log mean of the test failures of the full scale fatigue tests is also demonstrated - it is worth noting that structural integrity flight testing confirmed that the full scale fatigue tests stress levels were correct.

However, the situation is not always as clear cut as crack A. The next fig.28 shows the relationship between test and service experience for a fatigue failure - designated crack B. Although the in-service cracks on the port and starboard side are identical in nature the statistical evidence shows that there is a difference between the port and starboard side (as shown by the fact that LOG_{10} SD is 0.0486 on starboard and 0.09381 on port). The comparison between test and service experience is good indeed for the starboard side but is very poor for the port side- the full scale testing spectra is of course the same for both sides. However the figure again shows clearly that the first crack detected in-service is within the safe-life obtained statistically and is of the same type found on test.

4.13 NON DESTRUCTIVE TESTING METHODS

There are many acceptable NDT techniques (visual, ultrasonic, X-ray etc), which are capable of reliably detecting in-service any fatigue cracks which occur in skin or stringers and whose critical crack lengths are measured in inches. But the real problem is detecting very small cracks in bolt holes in machined components such as spar caps with very small critical crack lengths.

The Eddy Current Probe has been shown to be the best method of detecting bolt hole cracks and I would like to comment here on current developments which are a significant step forward in reducing operator fatigue and improved crack detection.

The standard Eddy Current Probe, which is hand held is cylindrical with a small sensing coil situated close to the end. The probe is spiralled down the bore being rotated by hand and incrementally removed after each rotation. This operation is laborious and leads to lack of concentration by the operator. Defect discrimination is poor relying heavily on operator experience to identify cracks or mechanical damage. Defect detection is affected by surface finish and mechanical damage and the defect sensitivity is poor at the material edges particularly with multi layers.

The recent development is the introduction of automated high speed bolt hole flaw detectors. This section will discuss a typical instrument fig.29 - the DEFECTOMAT C2821 produced by Institute Dr. Forster of Germany and developed by Wells - Krauthramer limited of the UK. The DEFECTOMAT is a power operated phase sensitive eddy current flaw detector. The probe incorporates minute twin differentially wound coils near the end. The probe is rotated at 2700 RPM by a low torque motor.

The probe can be passed quickly down the bore giving a complete 100% scan of the hole periphery. Phase changes of the twin coils are fed through the instrument and displayed on an oscillascope. The importance of the power operation is that there is a higher operator reliability due to reduced boredom and fatigue with a greater change of detection when inspecting many holes. The real value of the DEFECTOMAT is its ability to discriminate cracks from mechanical damage. The self comparator coils ensure that defects discrimination is still good at material edges. Figure 30 shows how the DEFECTOMAT can discriminate the difference between defects or hole features on the CRT. The DEFECTOMAT has a poor response to laminar defects but fatigue defects are rarely in this direction - but stress corrosion detection sometimes need this ability.

It is important to know whether NDT is really reliable in crack detection. It is rarely possible to compare NDT indications with actual cracks but fig.31 shows a comparison done where 120 cracks were inspected and then broken open. The NDT was undertaken in the laboratory under ideal inspection conditions so the operating advantages of the DEFECTOMAT were not fully demonstrated as compared to field inspection conditions. It can be seen that the DEFECTOMAT reduced significantly the spurious indications and thus would significantly reduce the repair workload. The very small maximum crack not detected by either means is also shown. My main purpose in this section to show the very high standard of NDT now available and still under further development. But despite these developments NDT is time consuming and costly and should be kept to a minimum. The design should aim to be crack free and NDT only used for the special cases which may arise in services.

5. ACTIVE CONTROL SYSTEMS

Structural design can be assisted in at least two ways by Active Control Systems Viz:-

(1) Use of Active Control Systems to provide artificial stability which permits a reduction in size of the tailplane or fin.

(2) Use of Active Control Systems for load alleviation either in atmospheric gusts or pilot induced manoeuvres.

This section will discuss 2 only.

In 1949, I was involved in the design of the Armstrong Whitworth Apollo aircraft which in its project design had gust alleviating ailerons operating symmetrically to reduce the wing static design bending moments in gusts.

At that date, methods of advance gust detection and quick enough control activation were not good enough to ensure any load alleviation. Today the concept has become viable with modern electronic techniques. I would like to comment on the use of Active Control Systems for improving aircraft fatigue.

As part of a study on load alleviation, a large aircraft was fitted with continuous in-flight recording of its wing stresses. One interesting result of this exercise is the fact that in the case of operation without autopilot the load fluctuations encountered in operational flying which are induced by pilot operation are significant by comparison with turbulence induced loads.

Fig.32 shows a typical flight record of pilot-induced load variations during a flight in preparation for the final approach with little or no turbulence.

Using this flight test evidence to set up a simulation of the loads we can generate a comparison of different flight conditions. Fig.33 shows some of the computor results.

The effect of the presence of the pilot on the fatigue induced loads is quite clear. This figure also shows the effect of possible Active Control Systems in alleviating wing bending moments from a fatigue standpoint. The possible reductions are sufficient to nearly double the fatigue life in the flight regime whether pilot induced manoeuvres are significant or not.

It is also worthy of note that if an Active Control System is being designed to reduce fatigue life rather than just reducing the design proof and ultimate loads it may be essential to take pilots control movements fully into account in designing the Active Control Design characteristics particularly if significant flying without autopilot is a requirement.

6. CONCLUSIONS

This lecture has briefly touched on many aspects of aircraft fatigue as they affect aircraft design and development.

It has shown the great value of structural design and development effort both from economic and airworthiness stand points.

The significant part ICAF has played over the years in focussing designers and research establishments thoughts on methods of solving aircraft fatigue problems has been demonstated.

The correctness of ICAF thoughts on design and testing has been shown by in-service examples. One striking feature is the fact that if basic structural design and development principles are rigidly adhered to - a good fatigue resistant aircraft results. There is really very little new under the sun. The problems of in-service inspection has led to the need to reassess the relative merits of safe-life, fail-safe and damage tolerance structural philosophies.

A brief note was included on the value of active controls in reducing aircraft fatigue.

(7) REFERENCES

(1) "THE VALUE OF FULL-SCALE FATIGUE TESTING" by N.F. Harpur and A.J. Troughton Pages 343-373 of FATIGUE DESIGN PROCEDURES edited by E. Gassner and W. Schutz - Pergamon Press 1969.

(2) "THE IMPORTANCE OF SERVICE INSPECTION IN AIRCRAFT FATIGUE" by J.A.B. Lambert and A.J. Troughton Pages 265-402 of AIRCRAFT FATIGUE, DESIGN, OPERATIONAL AND ECONOMIC ASPECTS edited by J.Y. Mann and I.S. Milligan - Pergamon Press (Australia) Ply Limited 1972.

(3) "PARAMETER ESTIMATION FOR THE LOG-NORMAL PARENT POPULATION OF FATIGUE FAILURES FROM A SAMPLE CONTAINING BOTH FAILED AND NON-FAILED MEMBERS" by A.M. Stagg - RAE T.R. 70145.

(8) ACKNOWLEDGEMENTS

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The figures in this paper were based on a great deal of data from the many sources listed below. I would like to thank all those who supplied data. I am sorry if your particular data does not appear to have been used but I had a considerable amount to analyse and it is possible your data has been used but its identity is concealed. My sources of information included:-

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National Research Council	Mr. J.A. Dunsby
GERMANY	
Deutsche Airbus GmbH	i.A. Frank & i.A. Ganz
ITALY	
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AIFA	Professor L. Lacati
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SWITZERLAND	
SIG	Mr. H. Rhomberg
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Douglas Aircraft Company	Mr. E.F. Dubil Mr. M. Stone
Fairchild Republic Company	Mr. Aaron Merken Mr. Peter Waters

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Lockheed - California	Mr. Warren Stauffer
NASA (Langley)	Mr. H.F. Hardrath

U.K.

British Aerospace	Mr. Cayton
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	Mr. S.A. Grott
	Mr. N.F. Harpur
	Mr. W.G. Heath
	Mr. J. Lambert
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Shorts	Mr. W. Kay

I would also like to thank Messrs Wells-Krauthramer for providing photographs and data on their Defectomat.

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DOCTOR
FREDERIK J PLANTEMA

SECRETARY

ICAF
1951—1966

1

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VALUE OF STRUCTURAL DESIGN AND DEVELOPMENT

2

LEADSHIP LIVES, HOURS AND FLIGHTS

IN—SERVICE PERFORMANCE

WEIGHT INCREASES

ROLE ALTERATIONS

COST OF STRUCTURAL DESIGN

ECONOMICS OF OPERATION

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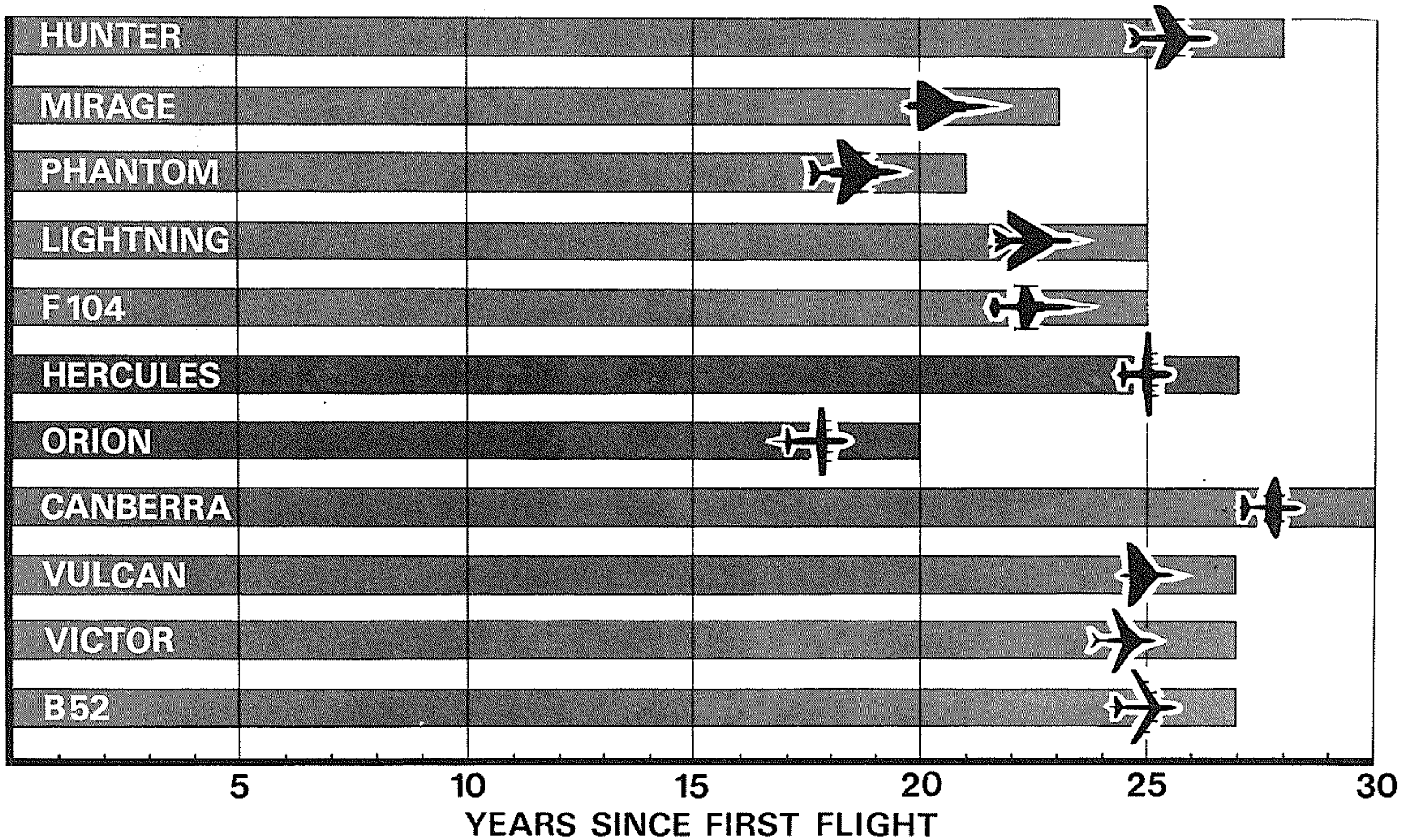
MILITARY AIRCRAFT LIVES

3



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CIVIL AIRCRAFT LEADSHIP LIVES

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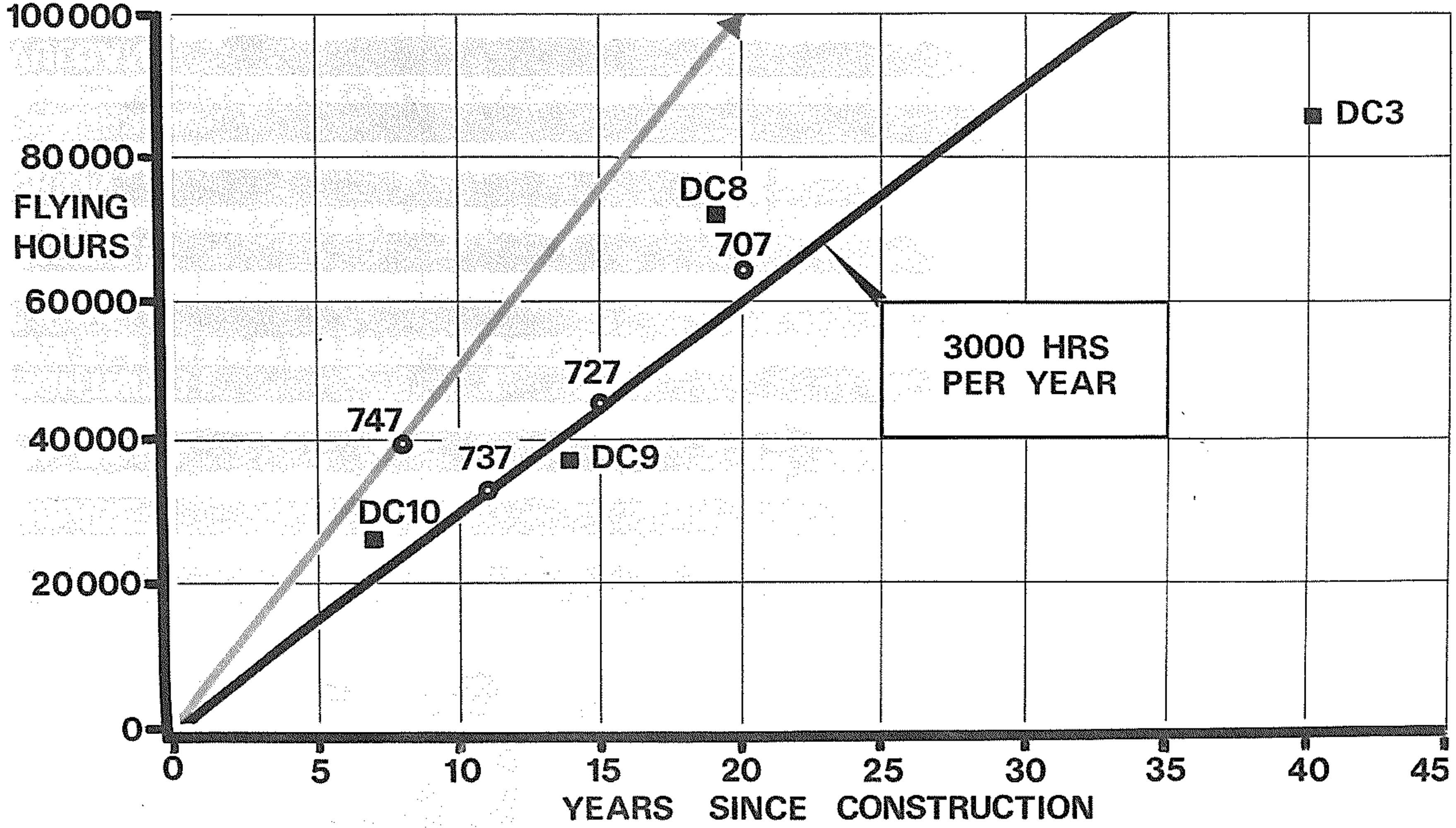

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CIVIL AIRCRAFT LEADSHIP HOURS AND LANDINGS

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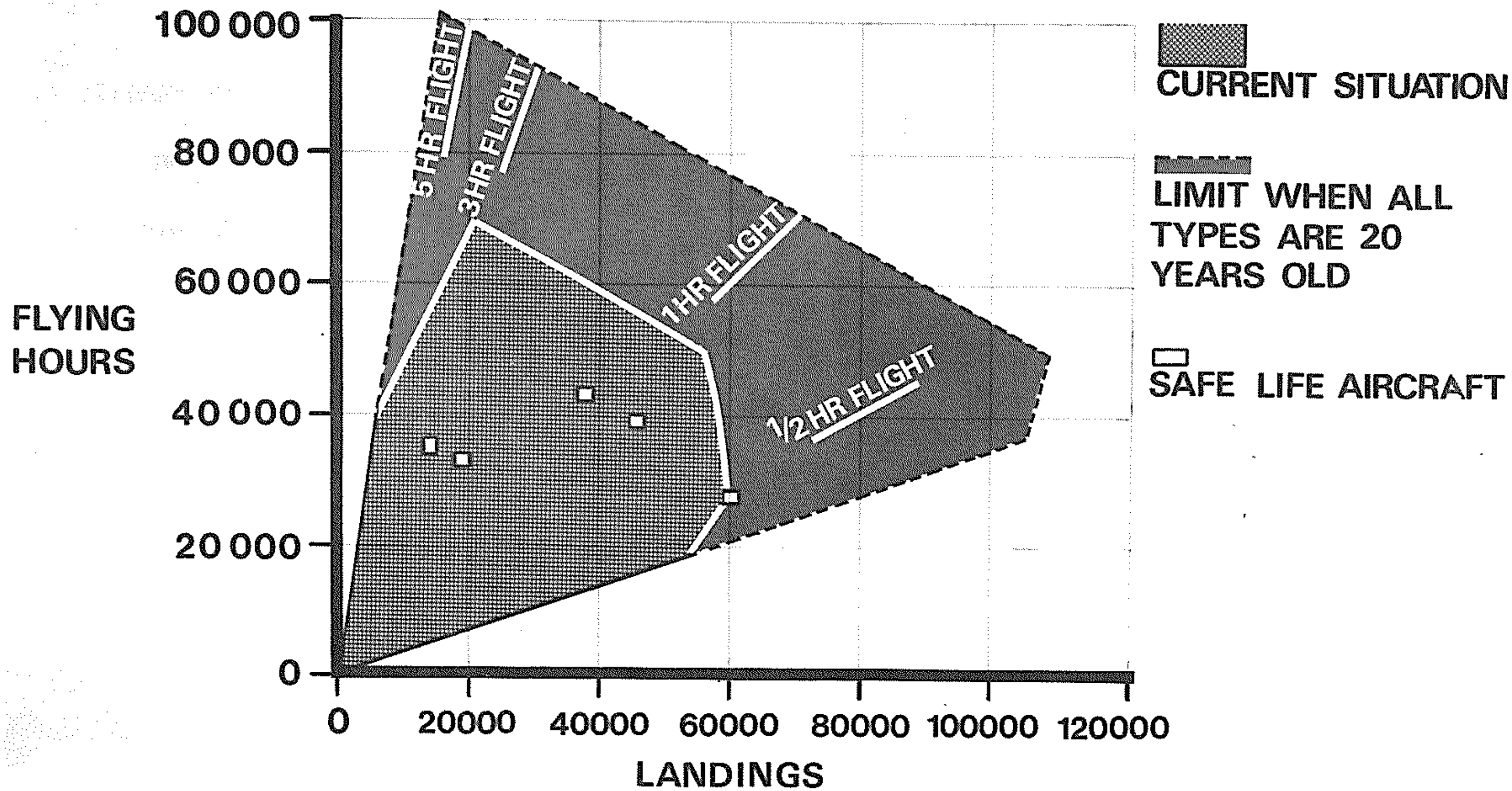

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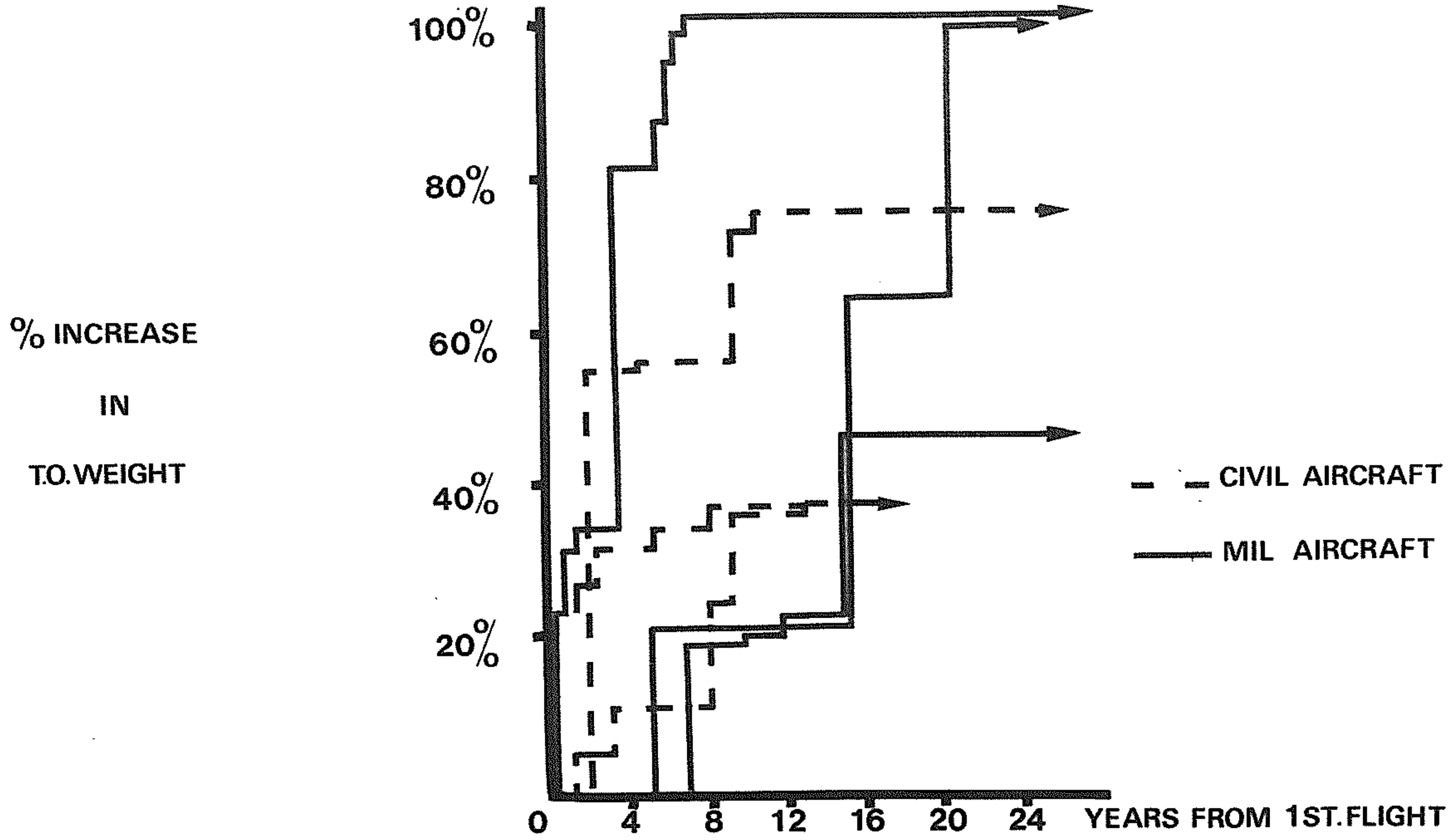

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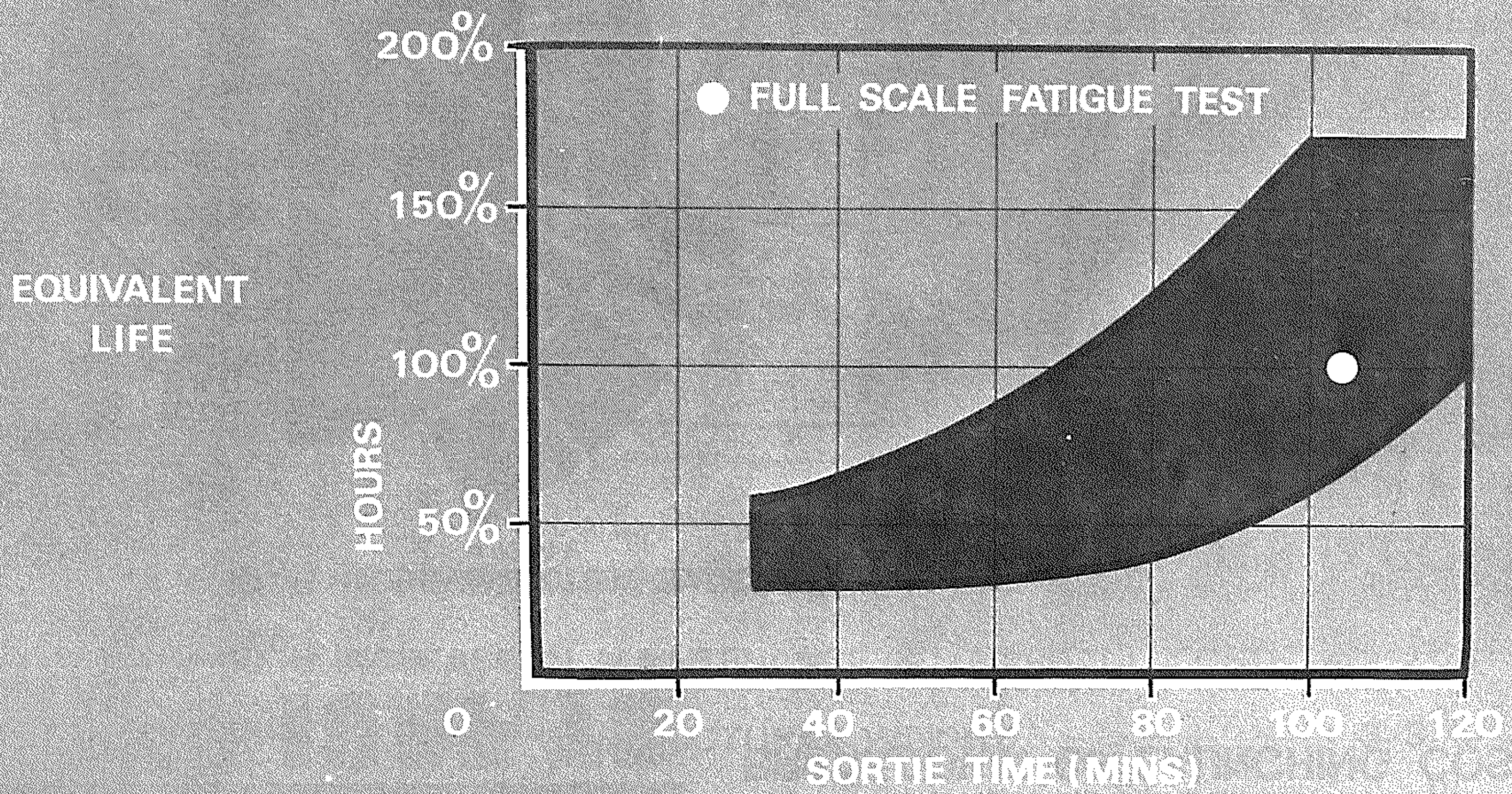
CIVIL IN-SERVICE ROLE VARIABILITY



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BREAKDOWN OF LAUNCHING COSTS

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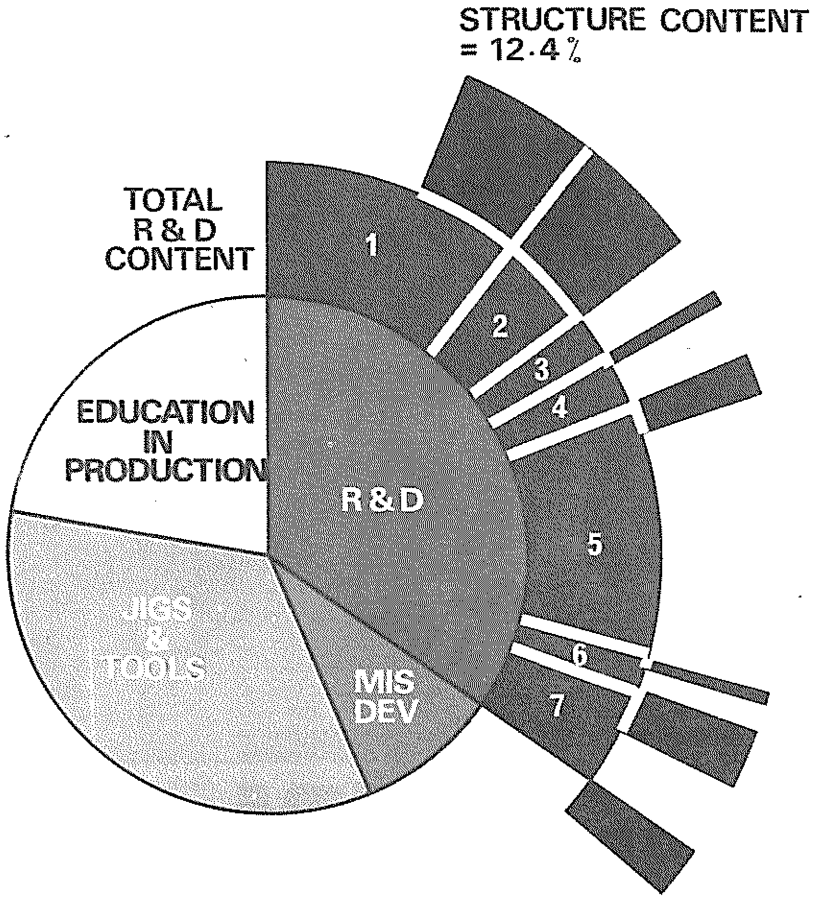

KEY TO R & D CONTENT

1 DESIGN

2 STRUCTURAL TEST

3 SYSTEM TEST

4 FLIGHT TEST

5 DEVELOPMENT AIRCRAFT

6 TECHNICAL PUBLICATIONS

7 MISCELLANEOUS DEVELOPMENT

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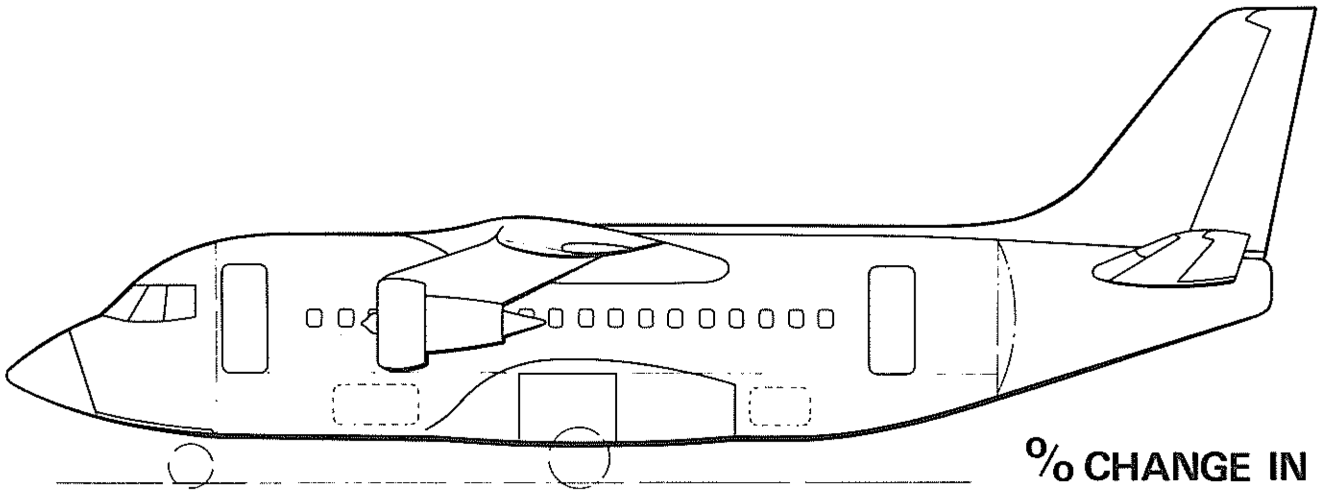
ICAF 79 OPERATIONAL COSTS TRADE-OFFS

9



	% CHANGE IN DIRECT OPERATING COSTS
10% INCREASE IN FIRST COST	+ 4·7%
10% INCREASE IN MAINTENANCE & OVERHAUL COSTS	+ 2·2%
10% INCREASE IN STRUCTURAL WEIGHT	+ 3·3%
2 YEAR INCREASE IN SERVICE LIFE	− 6·6%

AERODYNAMIC LOAD EVALUATION

FATIGUE LOAD SPECTRA

CHOICE OF MATERIAL

FATIGUE STRESS LEVELS

CUMULATIVE DAMAGE THEORY

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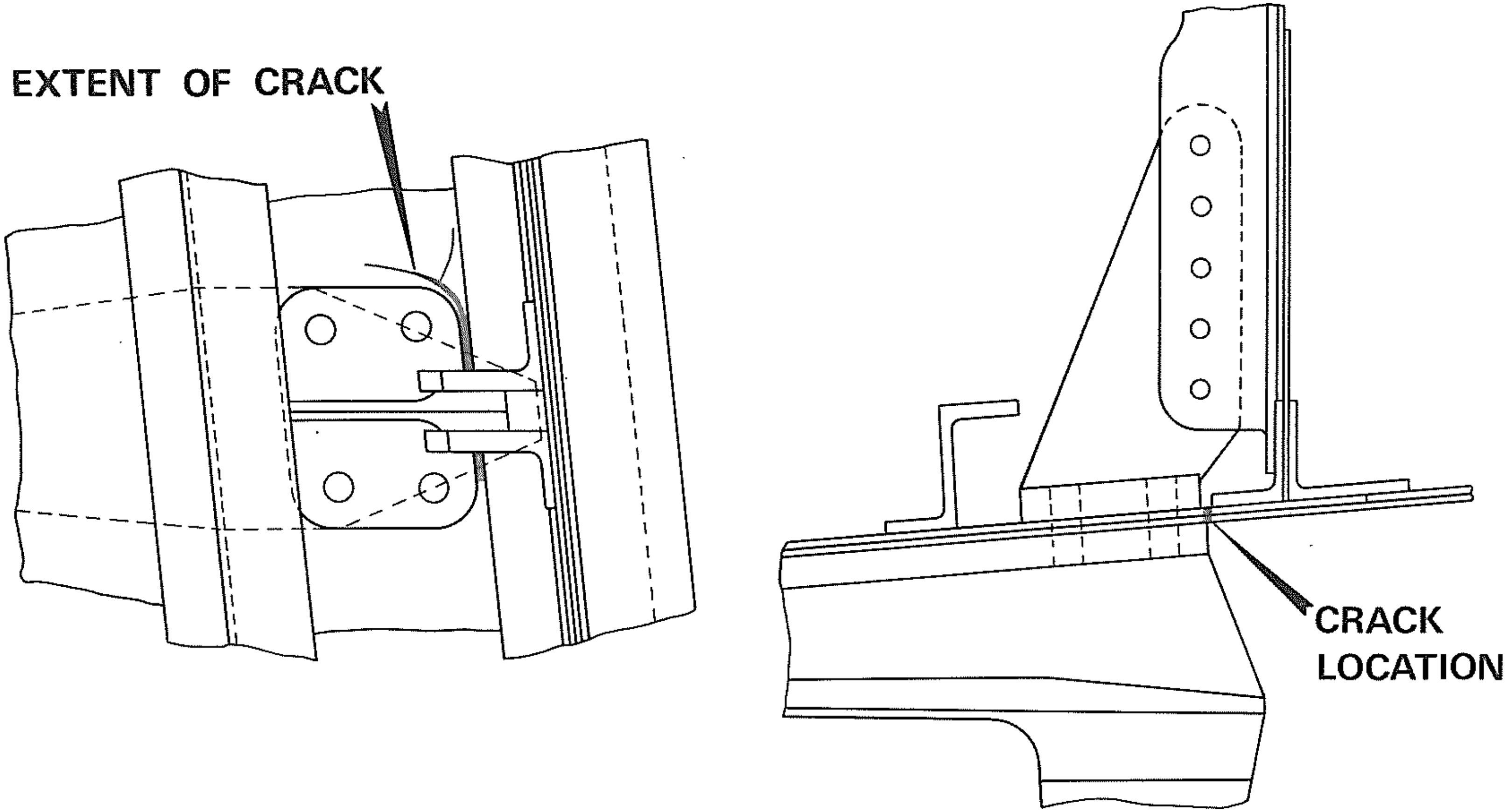
TYPICAL DETAIL DESIGN FAILURE

11



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SOLUTIONS TO THE PROBLEM OF DETAIL DESIGN

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CREATIVE DESIGN IN UNIVERSITIES
IMPROVED INDUSTRIAL EDUCATION
MAKE DRAUGHTING A CAREER
CONTINUING FAMILY OF AIRCRAFT
SETTING DETAIL DESIGN POLICIES
EXTENT OF USING FRACTURE MECHANICS
MAKING THE FIRST ISSUE OF DRAWINGS THE LAST

30% INCREASE IN STRUCTURAL DESIGN ONLY INCREASE LAUNCHING COSTS 0.42%

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SAFE LIFE

FAIL-SAFE OR DAMAGE TOLERANCE ?

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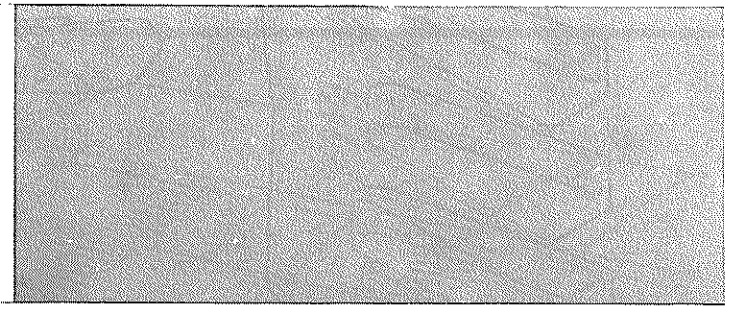
SAFE LIFE:

KNOWN REPLACEMENT POLICY

DIFFICULTY IN EVALUATING SAFE LIFE

NO PROVISION AGAINST MANUFACTURING ERRORS OR IN-SERVICE DAMAGE



FAIL-SAFE:

CAN CONTINUE FLYING ALTHOUGH CRACKED

IN-SERVICE INSPECTION PROBLEMS

PROOF OF FAIL-SAFE CHARACTERISTICS DIFFICULT

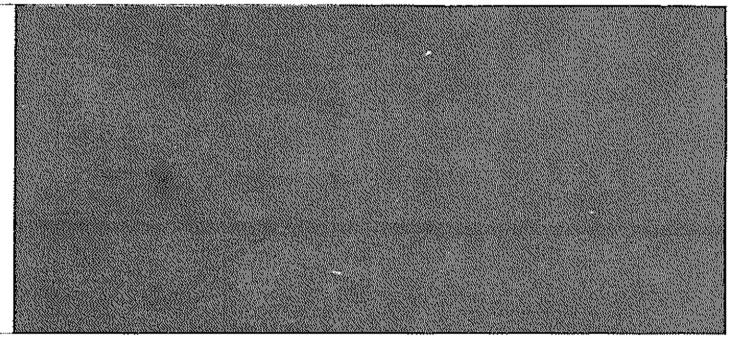
DAMAGE TOLERANCE:

COVERS MANUFACTURING ERRORS

COVERS INSPECTION ABILITY

ASSUMES INITIAL CRACK LENGTH

PROOF OF DAMAGE TOLERANCE CHARACTERISTICS DIFFICULT



Should we reconsider Safe Life Structures ?

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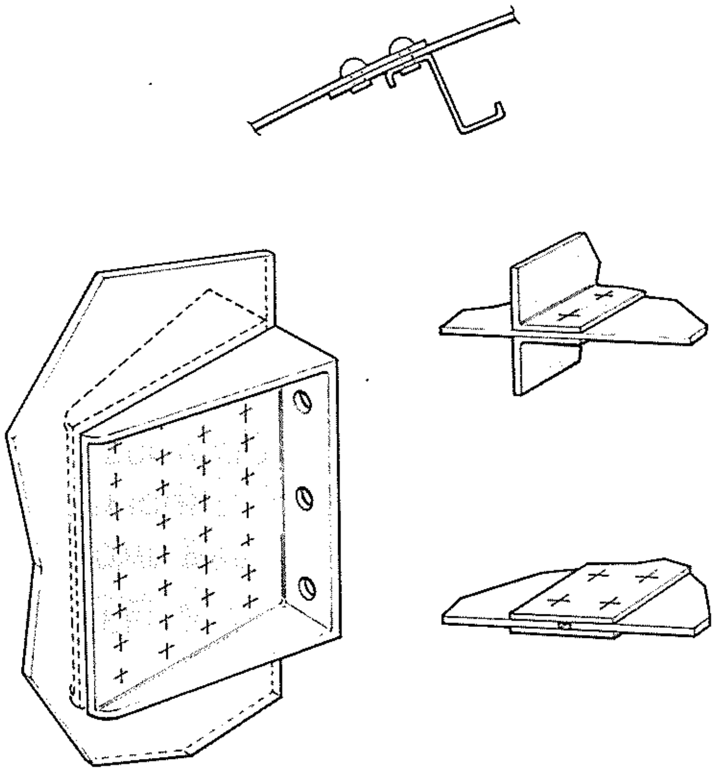
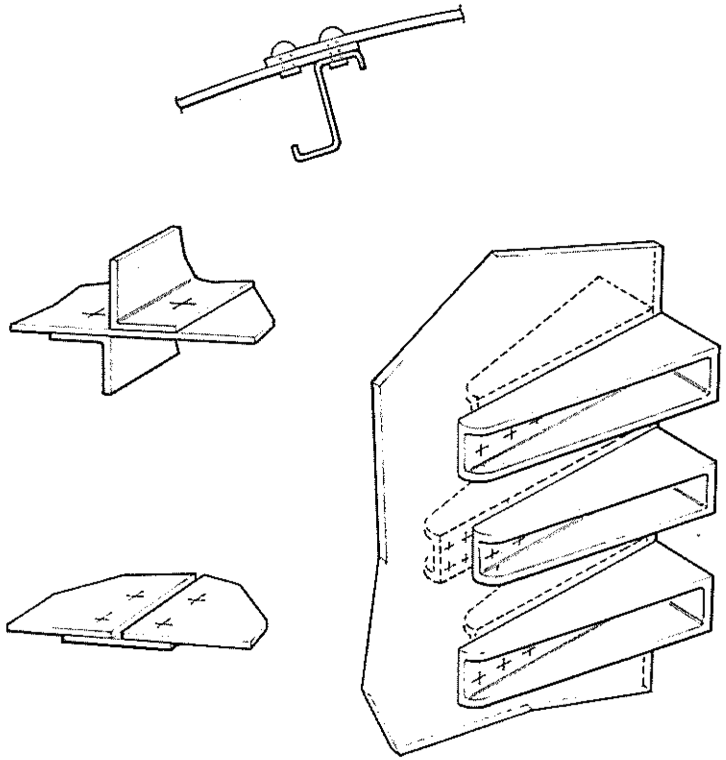
DESIGN FOR ACCESSIBILITY



POOR DESIGN



GOOD DESIGN

TYPICAL SERVICE BULLETIN

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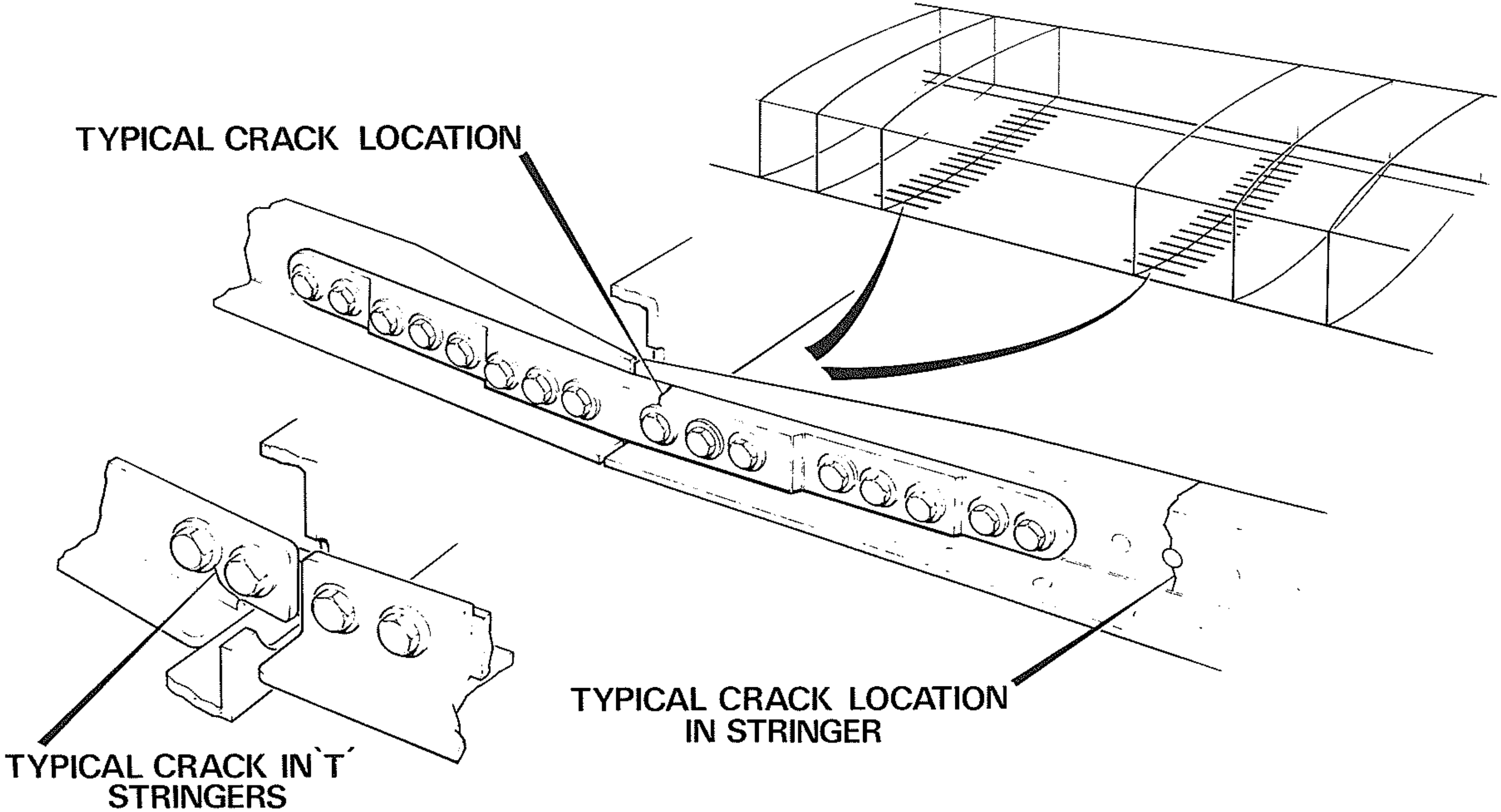

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RELATIVE VALUE OF TYPES OF FATIGUE TESTS

DETAIL FATIGUE TESTS

1. CONFIRM BASIC DESIGN IN ALL RESPECTS.
2. AIDS SELECTION OF MATERIALS, FASTENERS, SURFACE TREATMENT
3. PROVING VALUE OF SPECIAL MANUFACTURING TECHNIQUES
4. CONFIRMATION OF DETAIL FAIL-SAFE CHARACTERISTICS
5. EFFECT OF CORROSION AND THERMAL HEATING
6. CONTINUING STRUCTURAL INTEGRITY

COMPONENT FATIGUE TESTS

1. VALUE IN INTERNATIONAL PROJECTS
2. VITAL FOR ESTABLISHING FAIL-SAFE CHARACTERISTICS

FULL SCALE FATIGUE TESTS

1. ONLY REAL GUIDE TO FAILURE LOCATION
2. BASIS FOR STRUCTURAL INTEGRITY

VALUE OF FULL SCALE AIRCRAFT FATIGUE TESTING

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1 HIGHLIGHTING OF DESIGN ERRORS

2 ABILITY TO DEVELOP AN AIRCRAFT

3 COMPLETE STRUCTURAL COVER

4 INSPECTION SCHEDULES

5 CONTINUATION OF TESTING

6 PROOF OF FAIL-SAFE CHARACTERISTICS

7 ECONOMIC JUSTIFICATION

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FULL SCALE FATIGUE TEST DURATION

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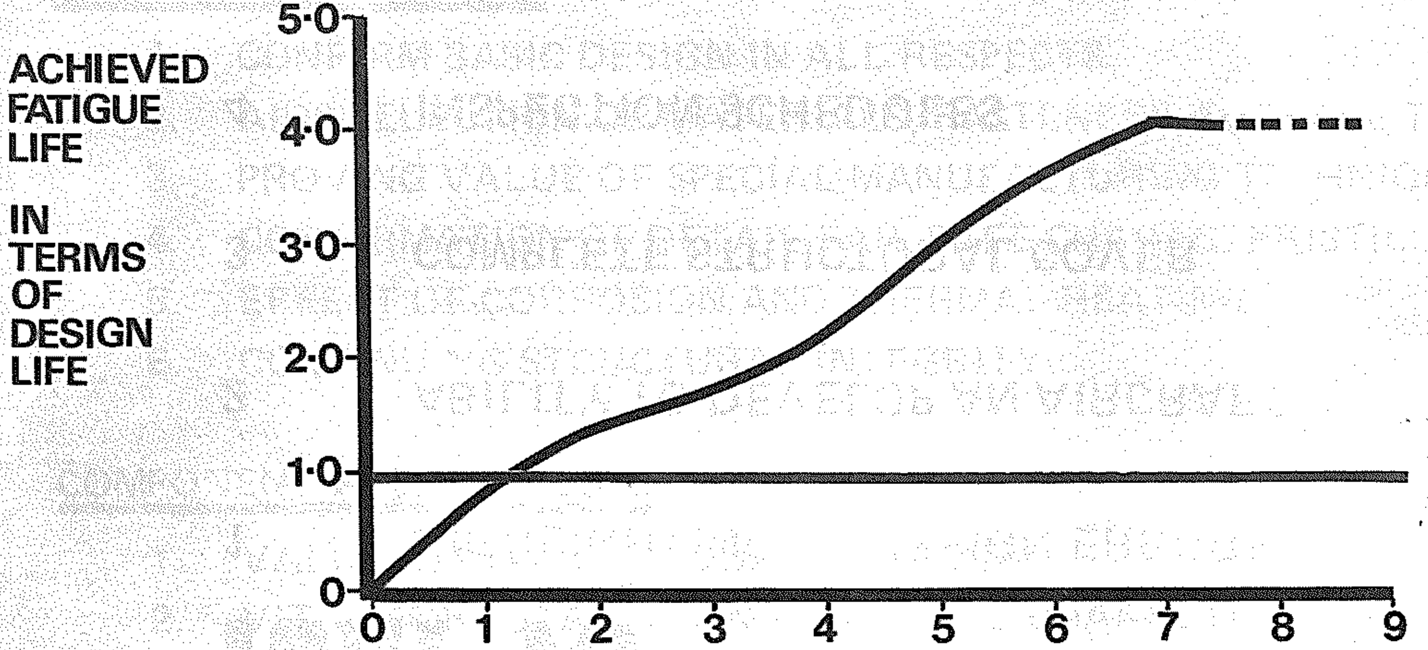

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REASONS FOR DISCREPANCY BETWEEN SERVICE AND TEST DEFECTS

	REASON FOR DISCREPANCY	1965 %	1979 %
1	ACTUAL FLIGHT USAGE DIFFERENT FROM TEST ASSUMPTIONS	24	25
2	INCORRECT LOADS APPLIED ON TEST	11	11
3	TEST LOADINGS NOT REPRESENTATIVE	35	31
4	TEST SPECIMEN NOT REPRESENTATIVE	18	6
5	ENVIRONMENT DIFFERENT : LOCAL HEATING : CORROSION ETC	5	17
6	PROBLEMS HIGHLIGHTED BY SERVICE INSPECTIONS	5	5
7	UNKNOWN	2	5

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SIGNIFICANCE OF CORRECT FULL-SCALE FATIGUE TEST LOADING.

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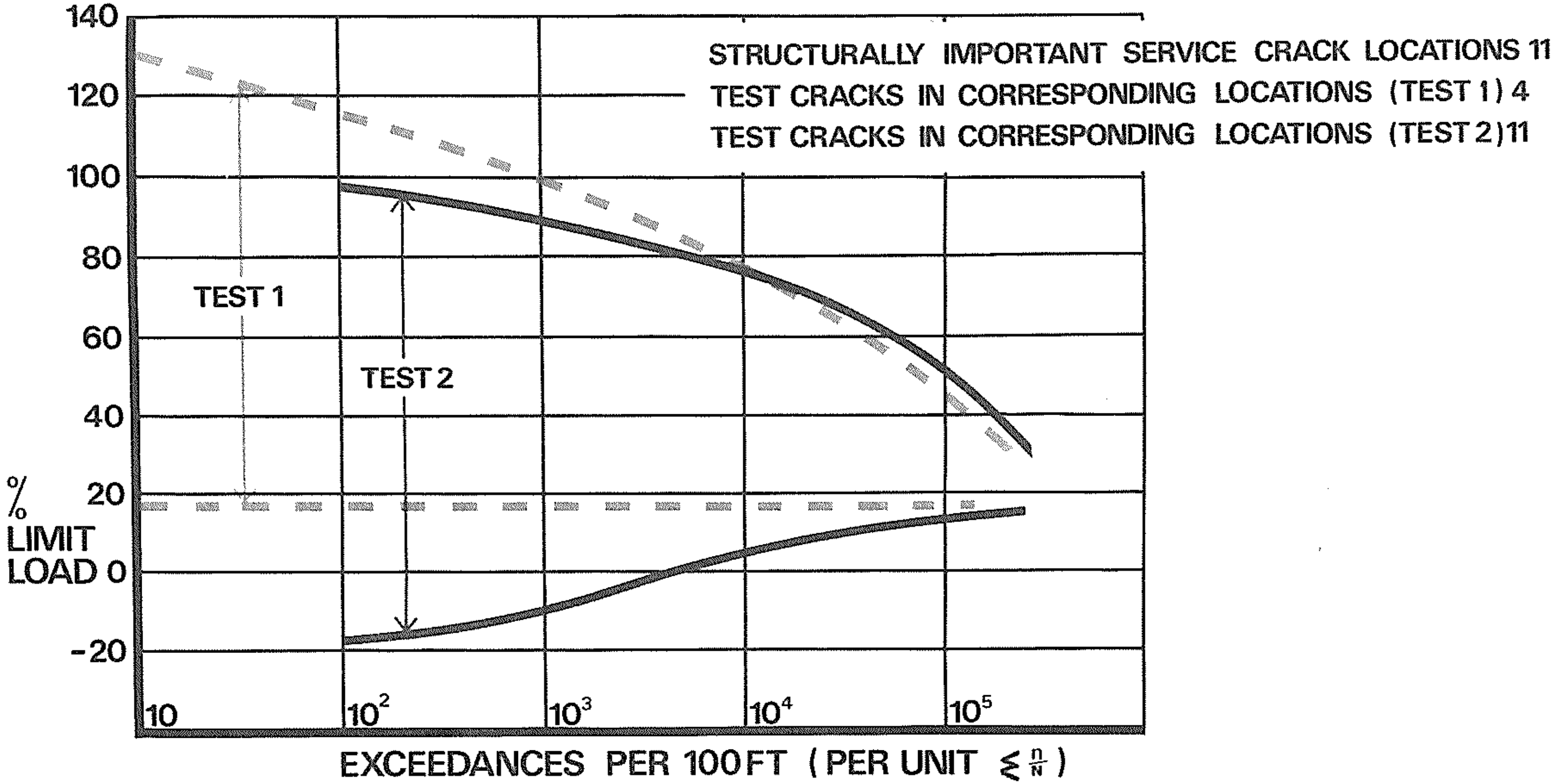

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VALUE OF STRUCTURAL INTEGRITY FLIGHT TESTS

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1. CONFIRMING DESIGN LOADS
2. ESTABLISHING LOADS NOT CONSIDERED DURING DESIGN
3. ESTABLISHING LOAD EVALUATION
4. DYNAMIC LOAD EVALUATION
5. OBTAINING LOAD SPECTRA
6. MONITORING CONFIGURATION CHANGES
7. ESTABLISHING FATIGUE LIFE

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USE OF STRUCTURAL INTEGRITY FLIGHT TESTING AS AID TO EVALUATING FATIGUE DAMAGE

FATIGUE DAMAGE USING FLIGHT MEASURED STRESSES	0.40 (MIN)
FATIGUE DAMAGE BASED ON C.G. COUNTING ACCELEROMETER	1.45 (MEAN)
	9.01 (MAX)

RESULTS OF 19 FLIGHTS WITH CONTINUOUS RECORDING OF WING STRESSES AT CRITICAL SECTION

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GROUND CALIBRATION TESTS ON TORNADO PRIOR TO STRUCTURAL INTEGRITY FLIGHT TESTING

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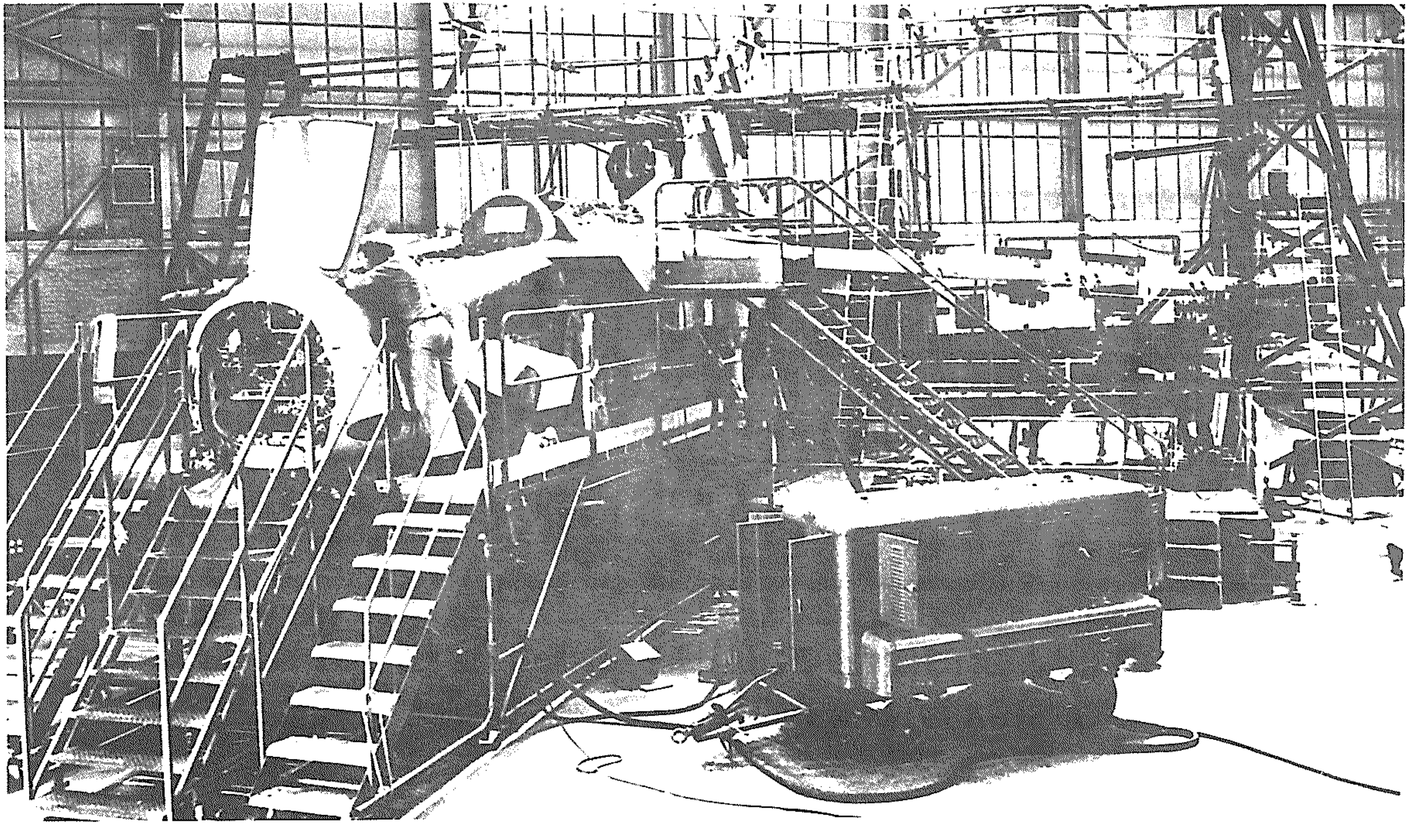

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STATISTICAL ANALYSIS OF IN-SERVICE FATIGUE FAILURES

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AIRCRAFT	PART	$LOG_{10}SD$	C_V	MEAN $\bar{x}$	SAFE LIFE $\bar{x} - 3SD$	SAFE LIFE $\frac{\bar{x}}{3\frac{1}{3} \text{ or } 5}$	1ST IN-SERVICE CRACK
TRANSPORT	FUS	0.1129	0.029	100	45.8	30	22.0
TRANSPORT	FUS	0.0512	0.014	100	69.7	30	75.7
TRANSPORT	WING	0.1781	0.044	100	29. 3	30	42.7
TRANSPORT	WING	0.2302	0.053	100	20.4	30	24.5
TRANSPORT	WING	0.1759	0.1104	100	29.7	30	39.5
TRANSPORT	WING	0.195	0.046	100	26.0	30	41.9

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STATISTICAL ANALYSIS OF IN-SERVICE FATIGUE FAILURES

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AIRCRAFT	PART	$LOG_{10}SD$	C_V	MEAN $\bar{x}$	SAFE LIFE $\bar{x} - 3SD$	SAFE LIFE $\frac{\bar{x}}{3\frac{1}{3} \text{ or } 5}$	1ST IN-SERVICE CRACK
STRIKE A/C	FIN	0.182	0.057	100	28.4	20	19.0
FIGHTER	WING	0.182	0.0927	100	28.4	30	31.5
TRAINER	WING	0.174	0.0979	100	30.0	30	34.5
TRAINER	FIN	0.0738	0.022	100	60.0	20	64.1
LARGE A/C	WING	0.1477	0.076	100	36.1	30	74.4
HELICOPTER	FUS	0.2345	0.081	100	19.7	20	38.6
HELICOPTER		0.4212	0.104	100	5.4	20	5.7

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STATISTICAL SERVICE SAFE LIFE COMPARED WITH PROMULGATED SAFE LIFE FROM FULL SCALE FATIGUE TESTS

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AIRCRAFT	PART	ACTUAL SAFE LIFE ($\bar{x}$ −3 SD)	EXPECTED SAFE LIFE (FTS)	REASONS FOR DISCREPANCY
TRANSPORT	WING	100	298	UNREPRESENTATIVE TEST PROGRAMME
TRANSPORT	WING	100	158	
TRANSPORT	WING	100	570	
STRIKE A/C	FIN	100	215	FIN LOAD SPECTRA UNKNOWN
FIGHTER	WING	100	458	NOT KNOWN
TRAINER	WING	100	102	GOOD CORRELATION
TRAINER	FIN	100	350	DIFFERENT SERVICE USAGE

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RELATIONSHIP BETWEEN TEST AND SERVICE EXPERIENCE - CRACK 'A'

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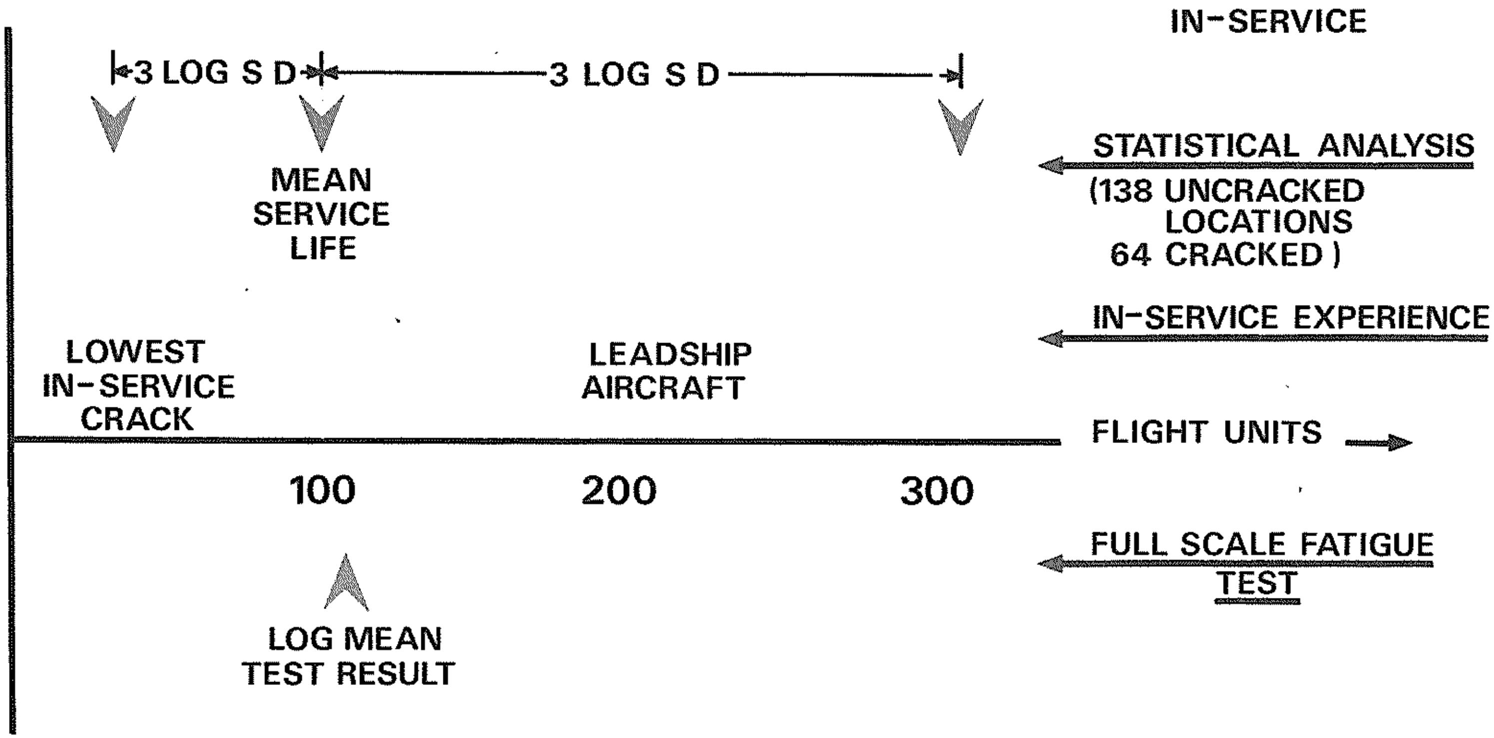

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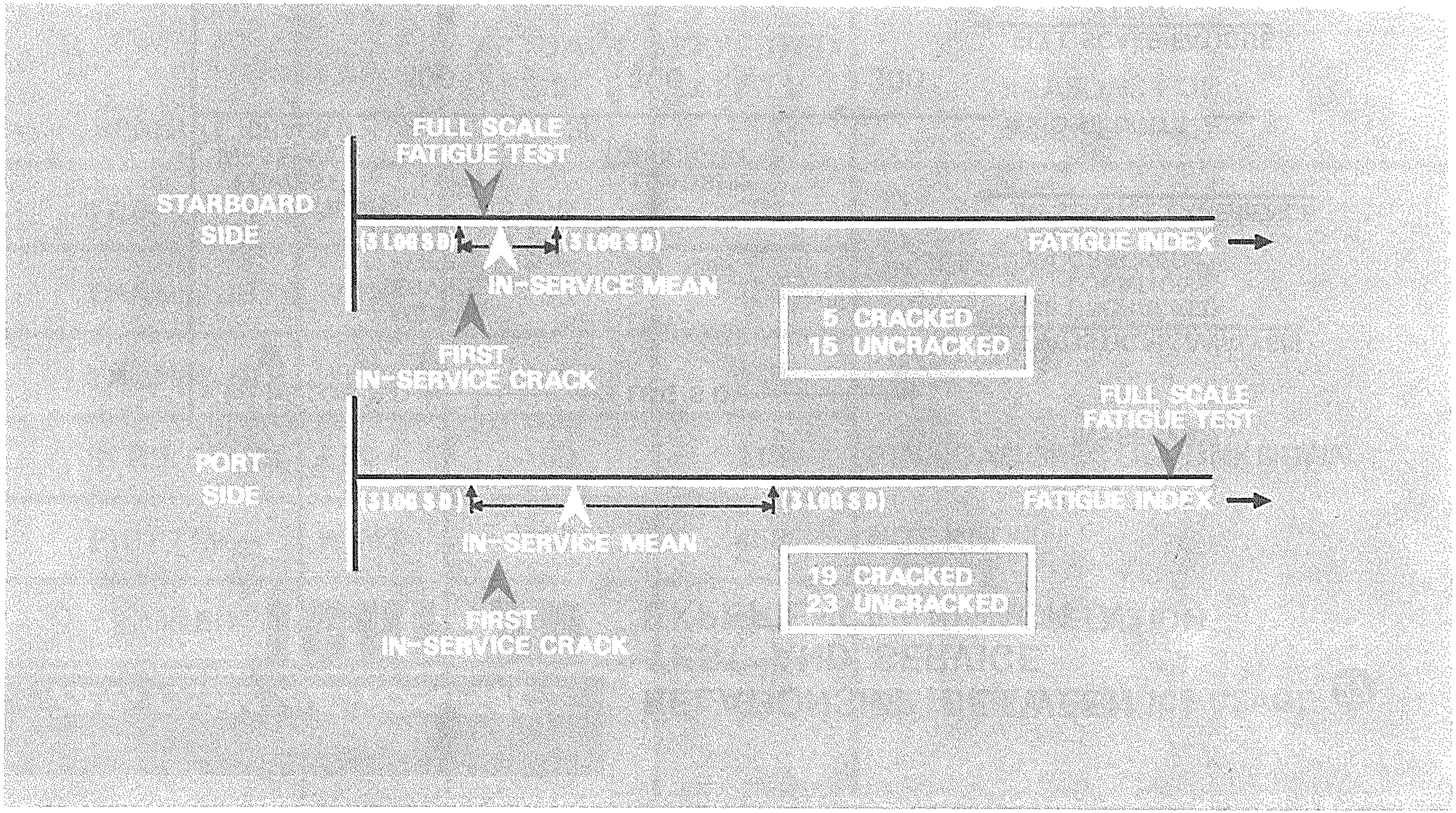
RELATIONSHIP BETWEEN TEST AND SERVICE EXPERIENCE – CRACK 'B'

28



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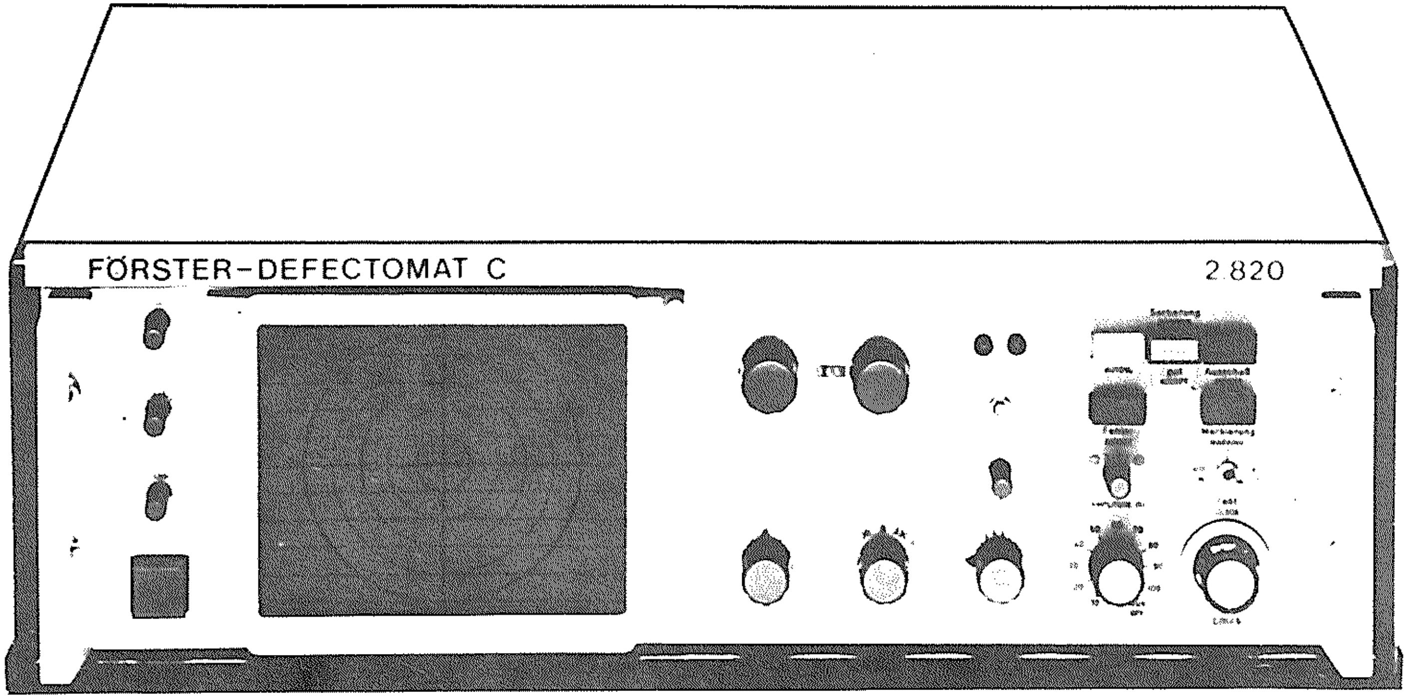
DEFECTOMAT



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DEFECT DISCRIMINATION BY 'DEFECTOMAT'

(SIMULATION OF C.R.T. SHOWING VARYING PHASE ANGLES OF DEFECTS)

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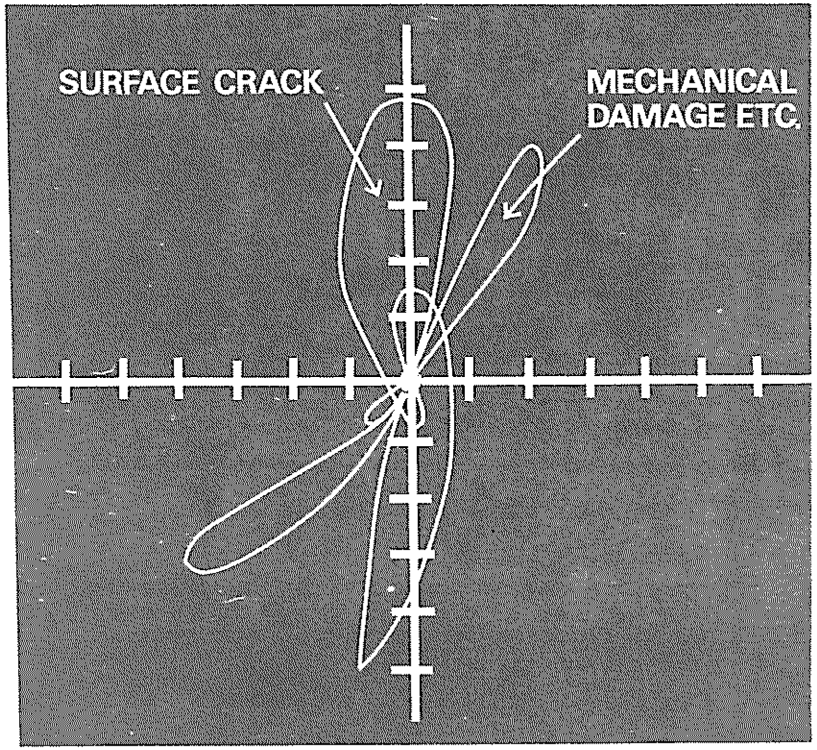
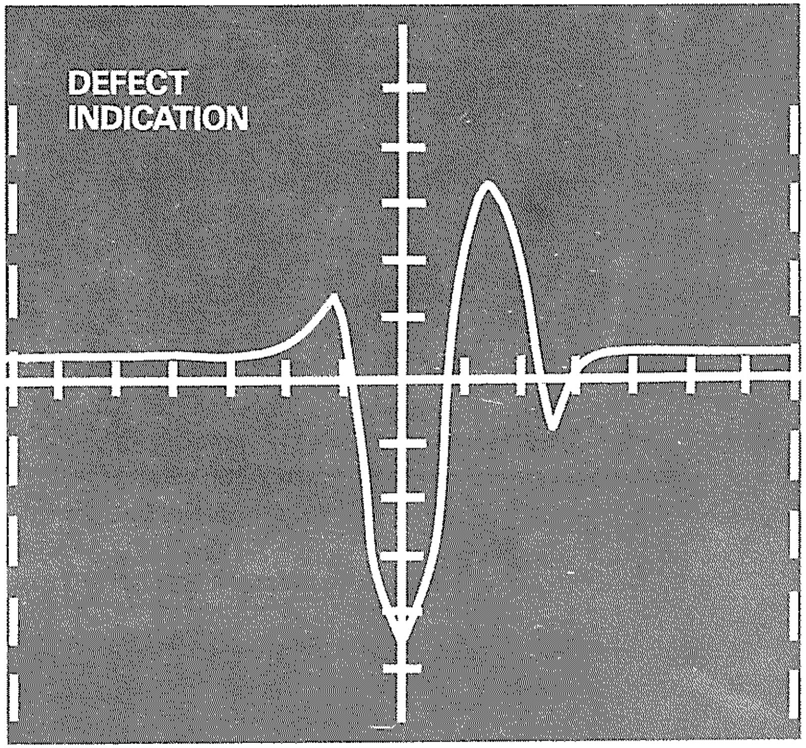
SPOT DISLAY ANALYSIS



SINE WAVE DISPLAY SEARCH



360° OF HOLE PERIPHERY

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ACCURACY OF CURRENT N DT TECHNIQUES

	Standard Eddy current Probe	Defectomat
Number of Holes inspected.	120	120
Holes giving distinct crack indications	35	23
Holes actually cracked within the above number (Microscopic examination)	17	16
Number of cracked holes not detected	6	7
Number of cracks not detected	10	11
Maximum size of crack not detected by either instrument	0.015″ corner crack and	0.015″ x 0.030″ (depth x length)

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MEASURED PILOT INDUCED LOADS IN STILL AIR FLYING

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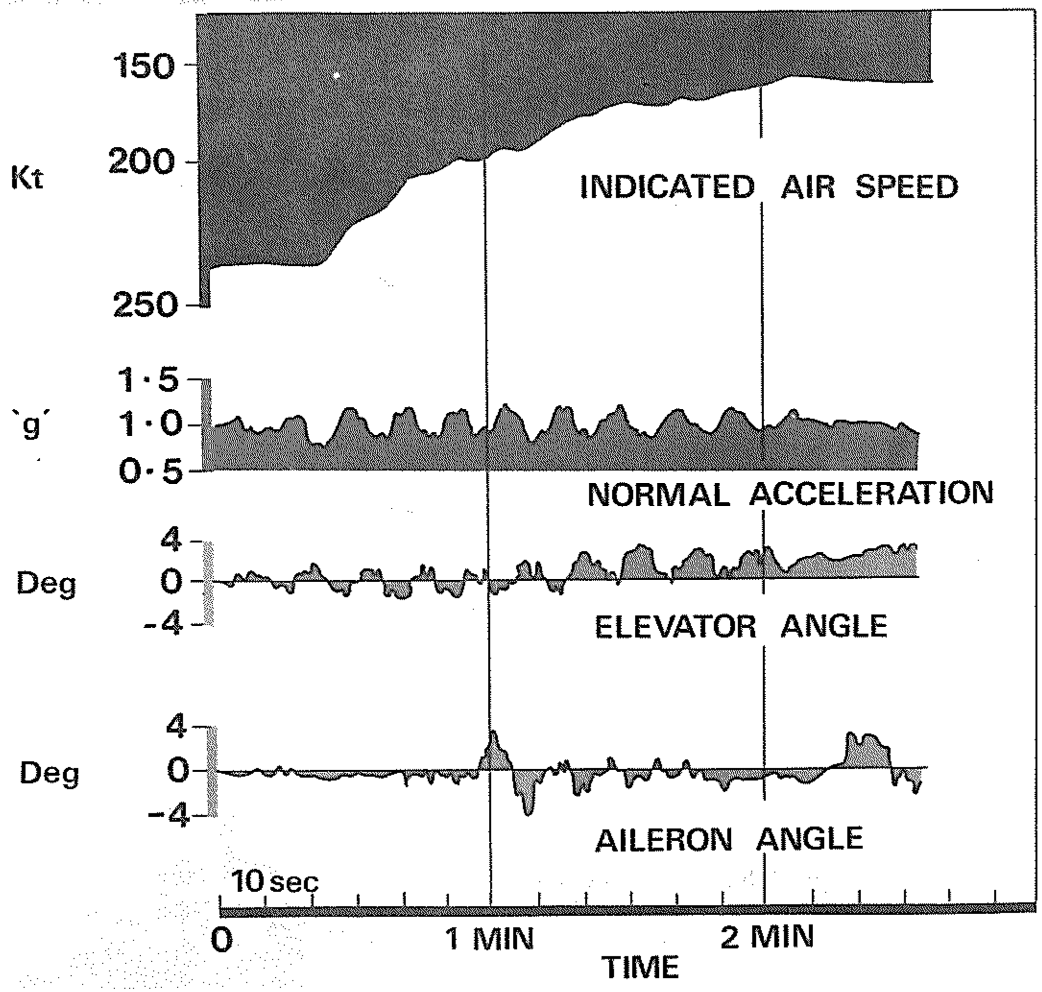

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WING BENDING MOMENT REDUCTIONS WITH ACTIVE CONTROL SYSTEMS (A.C.S)

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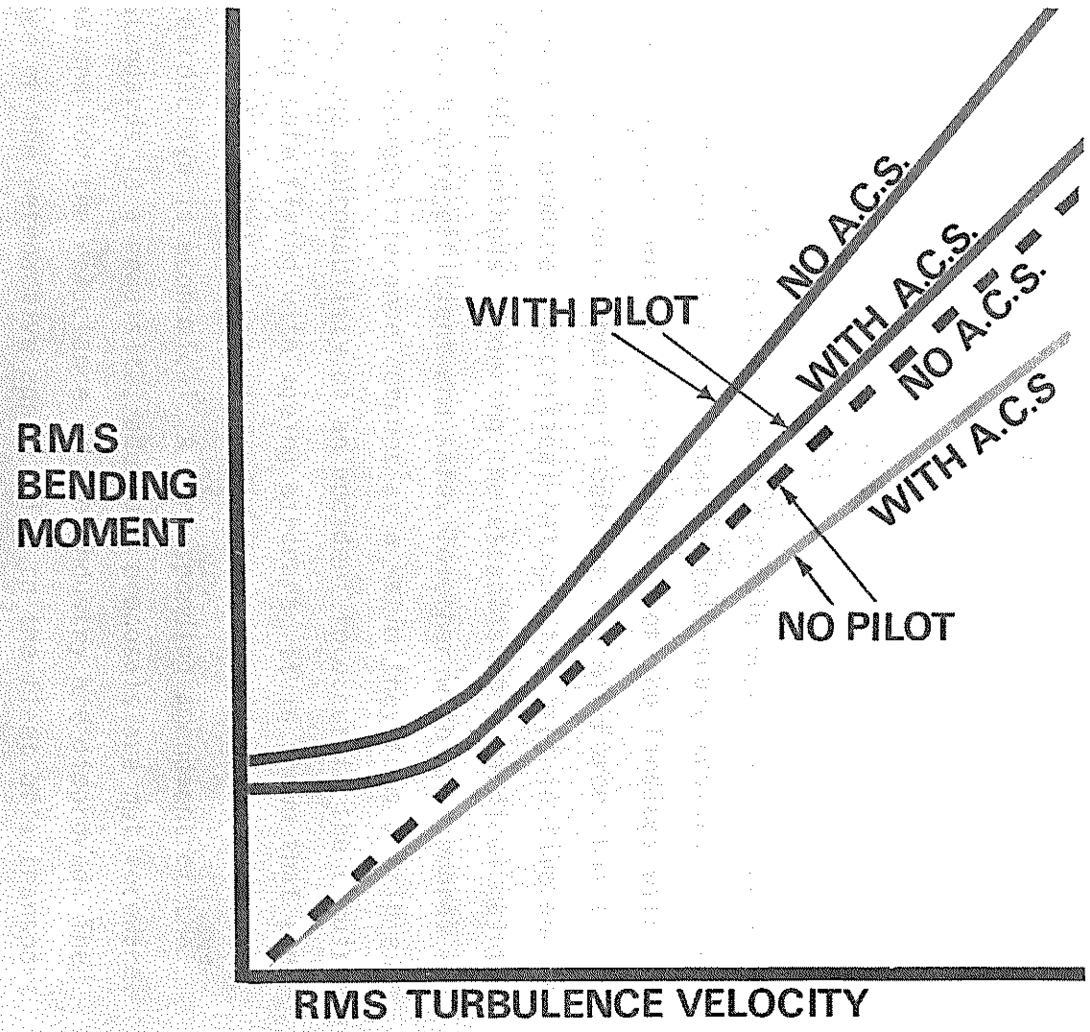

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