FATIGUE DAMAGE IN AIRCRAFT STRUCTURES, NOT WANTED, BUT TOLERATED?

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Abstract: *It has been a long road to arrive at the present culture of dealing with fatigue and fatigue damage tolerance of aircraft structures. Some accidents were milestones along the road. New concepts were proposed related to structural design, materials, production techniques, inspection procedures and load spectra. The present state of the art has been affected by various conditions associated with interests of the aircraft industry, the aircraft operator and the airworthiness authorities. At the same time, extensive research efforts have been spent. Our understanding of fatigue damage problems increased significantly in the last 50 years. Simultaneously our tools to tackle these problems have been developed to a high potential efficiency. And still, there are problems. The present paper is a personal impression of evaluating experience, design aspects, predictions and experiments.*

1 INTRODUCTION

When I started to write the present paper, my first problem was to select a relevant title. You might expect something like Designing Damage Tolerant Aircraft Structures because this is the main theme of the present conference. However, I felt some reluctance to talk about designing a structure for something that you do not want to occur, but which you must tolerate. “Designing against fatigue” would be a better opening, something to be achieved. As an aircraft passenger I prefer to think that the structure does not have any significant damage. Moreover, knowing figures about the high safety level of civil aircraft operation and the steadily increasing air traffic, what is our problem ? Apparently significant improvements of designing aircraft against fatigue have been made. But in spite of so much progress, I am still buried by many announcements of conferences on fatigue problems of aircraft structures. Many flyers and repeating messages on the internet. This year there are at least four well-known international conferences on damage tolerance and aging aircraft: in April in the USA, in May in Italy, in September now in Delft, and another one in December again in the USA. Altogether large numbers of papers and posters, and certainly many conference still to come in future years. In other words, we are still facing problems.

In principle, three parties are involved in discussions on fatigue and damage tolerance of aircraft structures: the aircraft industry, the aircraft operator and the airworthiness authorities. The industry and the operator are pushed by different arguments but at the end they must agree with the official airworthiness regulations. Controversial arguments of the industry and the operator are important for technical solutions. The arguments are briefly discussed in Chapter 2. In Chapter 3 several historical aircraft accidents are summarized to illustrate how we learned bit by bit to prevent similar accidents later, learning by practical experience.

Chapter 4 is dealing with the fatigue phenomenon and problems associated with the prediction of fatigue properties of structural elements. Comments on designing against fatigue, also with respect to damage tolerance, are presented in Chapter 5. Summarizing conclusions are given in Chapter 6. The incentive is not to present a thorough treatment of the above topics. This has been done in several excellent surveys in the literature, e.g. in papers of Goranson [1] and Eastin [2]. The present paper is more a personal reflection on some essential concepts and observations which illustrate the present state of the art.

2 THE PRESENT SITUATION

The possibility of fatigue crack initiation must be considered in both civil and military aircraft although circumstances of fatigue issues can be quite different. Aircraft of the airlines should be in the air as much as possible in view of economic profits, i.e. earning money. As an indication some 3000 flying hours in one year may be compiled (there are 8760 hrs in a year). However, military aircraft do not make that many flight hours. Again as an indication, some 300 hrs is possible. In some countries the number has been reduced to save money instead of earning money. Another difference: Civil airlines can not afford serious fatigue problems which would require grounding the fleet. Grounding implies a loss of flying hours and it is also poor publicity. Continuation of flying under special conditions offers safety problems which must be solved in agreement with the airworthiness authorities. If a serious fatigue problem occurs in a military aircraft it is accepted that grounding of all similar aircraft can be necessary to prevent a loss of aircraft and crew. The main duty of a military aircraft is to be available and serviceable depending on international circumstances. Differences between civil and military demands have also led to a different way of organizing safety regulations. In the civil sector three groups can easily be indicated, which are the aircraft industry, the aircraft operator and the airworthiness authorities, see Figure 1. The civil airworthiness authorities are an independent organization. However, airworthiness rules for military aircraft are specified by military authorities. Nevertheless, civil and military aircraft have many fatigue problems in common with respect to designing against fatigue, maintenance, inspections, and damage tolerance.

Safety regulations for civil aircraft operators have been established by airworthiness authorities almost from the beginning of aviation. Arrows in Figure 1 represent all sorts of contacts between the three parties with conflicting economic interests between the operator and the aircraft industry, and on the other hand questions about meeting safety requirements. In essence, conflicting situations are associated with aspects of *economy and safety*, which sometimes implies *economy versus safety*. Airlines obviously want a low-weight structure and low direct operation costs (DOC). The latter argument includes minimal maintenance, simple inspections, long periods between inspections, simple repairs, etc. At the same time, the structure should have sufficient strength and good durability properties, e.g. no fatigue and corrosion problems. The aircraft industry may be capable to design an aircraft structure which meets all these demands. However, it may imply an increased weight of the structure and more expensive design and production concepts with an increased selling price as a consequence. These conflicting conditions are aggravated by a tough competition between aircraft industries on one hand and between airlines on the other hand. After all, these problems must be resolved within the constraints set by the airworthiness authorities in view of safety of aircraft operation in service. At the same time it offers a challenging field for innovative ideas to achieve new solutions and concepts for old problems.

Fatigue problems to be considered in the development of a new aircraft type can be surveyed along the lines depicted in Figure 2. New structural design concepts, alternative production technologies, and new materials will be of interest. They are part of the design analysis. At the same time, problems about predictions and exploring test programs will arise. Relevant fatigue load spectra of the aircraft are also essential. An overwhelming problem setting has to be faced as schematically shown in Figure 2. A discussion of all items is not intended here, but the diagram is shown to illustrate the large variety of problems to be analyzed. It is a highly multi-disciplinary problem setting. Chief-designers must have a good qualitative understanding of the significance of these problems, while quantitative evaluations are covered by specialists. The investments are comprehensive, keeping in mind the complementary experimental efforts to substantiate the fatigue performance of the aircraft structure. Since a few decades satisfactory damage tolerance properties must also be shown. Again as an illustration of the multi-disciplinary problem setting, various steps involved are summarized in Figure 3. A most relevant question now is, how well are we equipped with knowledge and tools to deal with all steps in Figures 2 and 3? Beyond any doubt, if problems can be solved by reliable predictions, a most significant saving of time and money can be obtained. As said long ago by the famous physicist J.C. Maxwell "there is nothing more practical than a good theory". Let us rephrase it here by "nothing is more practical than a good understanding of the mechanical and physical problems involved". Another problem can arise: how can we convince directors and managers about our solutions. But let us first convince ourselves.

3 SOME MILESTONES ALONG THE ROAD

It is generally recognized that fatigue failure accidents had a significant influence on the development of practical knowledge about preventing similar accidents afterwards. A classical example turned up in the 19^{th} century when fatigue failures occurred in railway axles. The problem was studied by Wöhler who carried out systematic fatigue tests in the 1850's. Railway wheels were shrink fitted on the axles by a differential heating technique which produced a very strong joint shown in Figure 4a. Unfortunately it produced an extremely high peak stress at the contact point P. Moreover, fretting corrosion at the same point due to the cyclic bending moment can not be avoided. Fatigue failures were initiated and fretting corrosion particles emerged from the joint. The solution for this problem is shown in Figure 4b. The extreme stress concentration was removed by a generous radius between the shoulder and the axle. Secondly, some fretting might still occur on top of the shoulder, but at this location it will be practically harmless. It is a clever solution at a time that stress concentrations and fretting corrosion were still unknown problems. It was *designing against fatigue*, but after the accidents, just by *learning from service experience combined with engineering judgement*.

Several catastrophic failures of aircraft structures due to fatigue occurred in the 20^{th} century. Some typical case histories will be summarized. A Martin-202 aircraft (40 passengers) crashed in 1948 due to a wing failure caused by a fatigue crack in a joint of the wing spar made of 7075-T6. The fatigue failure started from a corner in a thickness step of the spar cap. Similar cracks were observed in some other aircraft. Poor geometrical design and a fatigue sensitive material. After this accident NASA has started a series of fatigue tests on various modified joints.

A major impact on considering fatigue of aircraft structures emerged from fatal accidents of two Comet aircraft in 1954. Fatigue cracks in the pressurized fuselage structure initiated a fuselage decompression failure at a high altitude. These failures were discussed in a number of previous publications, noteworthy by Tom Swift [3]. However, one interesting observation is not very well known. The two Comet aircraft crashed after 1286 and 903 flights respectively. Unstable crack extension was started by a fatigue crack at the edge of a window in the cockpit section. Before the Comet entered into service, a full-scale fatigue test was performed on a large part of the fuselage including the cockpit section. Small cracks were found after 16000 flights in the test, a life time approximately 15 times longer than in service. Unfortunately, the test article was previously subjected to quasi static loading until a high design load to show sufficient static strength. In order to save money, the same test article was subsequently used for the fatigue test with pressurization cycles. As a consequence, the load applied in the static test was a high pre-load for the fatigue test. It caused small-scale plastic yielding at fatigue critical notches which introduces that favourable residual stresses. As a result, a significant but unrealistic life extension in the full-scale test was obtained. This effect was overlooked.

The accidents of the two Comet aircraft were followed by a full-scale fatigue test on a complete aircraft. A flight-by-flight load history was applied. It included ground-to-air cycles, but the gust load spectrum for the wing was reduced to a single load level which according to a linear damage calculation would represent the same fatigue damaging effect as the full spectrum. The same load history was applied in each flight, see Fig.7c. The idea that full-scale tests are necessary was a step forward. However, the concept of a simplified flight-simulation full-scale test with all flights being equal was not correct.

Another catastrophic fuselage decompression occurred in 1971 in a Vickers Vanguard. The aft pressure bulkhead was blown up due to fatigue cracking along the edge of a reinforcing strip, see Fig.5. The empennage control system was destroyed, a type of failure which is no longer accepted in later airworthiness requirements. Fatigue cracks were initiated by corrosion damage because condensation occurred at the inner side of the bulkhead due to the cold outside temperature. Water was trapped at the edge of the doubler at point A. Sealing was applied but unfortunately not up to the critical point A. It should also be noted that trapping condensation would not occur with a doubler at the other side of the bulkhead. Furthermore, the sheet material of the bulkhead was a fatigue sensitive Al-alloy. Similar corrosion damage was found in other aircraft of the same type.

In 1969 an F-111 fighter crashed as a result of a wing failure. It turned out that the failure was initiated by a large flaw in the material, see Fig.6. The flaw had grown to a small extent by fatigue in 120 flying hours. The material was a D6AC steel heat treated to S_U = 1600 MPa. Surprisingly, the flaw was introduced during the production of the steel plate, and it was not detected. Moreover, the fracture toughness of high strength steels can be relative low. The impact of this fatal accident on the airworthiness regulations of the USAF was far-reaching. The damage tolerance concept was introduced, and more specifically the assumption about small initial flaws being present in new aircraft. A problem was associated with the size of the initial damage and the locations in the aircraft structure where initial flaws must be assumed to be present.

A catastrophic wing failure occurred in 1976 due to a fatigue crack in the lower wing skin of an Hawker Siddeley 748. The crack had a length of 90 cm (35"). The same crack was found in 19 other aircraft, most of them still small, except for one crack of 70 cm (28"). The cracks occurred along a rivet line. A full-scale fatigue test had been performed, but the applied load history was not realistic. Flight loads were arranged in a block programmed sequence (see Figure 7b), and ground-to-air cycles were not included. Un-conservative results can then be obtained.

Another remarkable disaster is known as the Lusaka-accident, which occurred in 1977. A

Boeing 707 lost a stabilizer during landing at the airport of Lusaka. It was caused by a large fatigue crack in the upper spar of the stabilizer, see Fig.9. The total life of the aircraft was 16723 flights (47621 hrs). However, a number of years ago the take-off weight had been increased implying higher design loads, and redesigning of the stabilizer was necessary. The upper skin (which is the tension skin for a stabilizer) of 7075-T6 was replaced by a similar skin of steel. Because of the higher strength of the steel plate the thickness could remain the same. Fail-safe concepts were also introduced by multiple lug attachments of the stabilizer to the tail section and splitting the spar web in an upper and lower part connected by a kind of an extra spar. The significantly increased stiffness of the upper skin had a large effect on the load transmission from the skin to the spar, especially at the root of the stabilizer near the lug attachments. The load transmission of bolt A (see Fig.9) should be very high, but fatigue cracking of the flange of the spar occurred at the more remote bolt B. Because of the high load transmission of all bolts between A and B plastic deformation occurred around the bolt holes which makes the bolts more or less ineffective in the load transmission. As a result of the loose bolts crack initiation can occur at the more remote bolt B. Actually, an operator had noticed that loose bolts were present between A and B, but he did not understand why, and he replaced the bolts by other ones. After the accident had occurred some 20 smaller cracks at the same location were found in 11 other aircraft with more than 26000 flying hours. The term “geriatric aircraft” has been used for some time, which is now replaced by “ageing aircraft”. Anyway, the introduced fail-safe design was not effective in this accident. The very high load transfer by bolt A and nearby bolts could have been expected and confirmed by FEM analysis. Furthermore, the quality of the redesigned structure was not supported by full-scale testing. Such a test was carried out after the accident and it indicated a fracture mode very much similar to the Lusaka case. Generalizing this experience: First recognize the problem, second calculate loads and stress levels, and third incases of doubt, carry out relevant experiments or improve the design.

The accident of a very old Boeing 737 of Aloha Airlines has also drawn much attention. At an altitude of 7300 meters the aircraft lost a large part of the fuselage skin. It is a wonder that the aircraft could still continue flying to an airport. The failure was caused by a large number of cracks started at many rivet holes in the same lap joint, a phenomenon which is now generally labelled as multiple-site damage (MSD). The aircraft was old, 89680 flights, 35496 hrs, 19 years, and unfortunately corrosion occurred in the lap joints promoted by disbonding of doublers. The corrosion problem was known and Boeing had provided inspection instructions, but they were not yet carried out. It is a typical example of ageing aircraft. Remedial actions are primarily related to design and quality of riveted lap joints.

Fuselage skin cracks in the Boeing 747 have been detected during walk around inspections. At cruising altitude humid air in the cabin is escaping through these cracks. Because of the low outside temperature condensation occurs immediately causing a dark staining on the crack edges which facilitates detection. Nicotine in the cabin air may well improve the visibility of the cracks. The escaping air is easily compensated by the cabin blowers and it does not warn for possible cracks of a limited size. However, once a crack occurred under the fairing between the fuselage and the wing. This invisible crack became unstable and a loud bang was heard. Oxygen masks had to be used, and the pilot immediately descended to a low altitude and made a safe landing. Inspection revealed that the crack was stopped at a length of 180 cm (71"), i.e. a 4-bay crack. It would have been instructive if full details of the crack and the structure were published.

A last case history to be summarized here is the catastrophic crash of a Boeing 747 in 1992. Shortly after take-off the aircraft lost an engine which then hit the second engine of the same wing which was also lost. In addition the structure of leading edge slat of the wing was damaged.
The pilot tried to return to the airport but the aircraft crashed and came down on an apartment building in Amsterdam. The first engine was lost because of a fatigue failure in the fuse-pin connecting the engine pylon to the front spar. Analysis revealed that the design of the fuse-pin was not optimal, but also the load spectrum of the pylon attachment was more severe than originally expected.

It may well be concluded that various accidents occurred because of structural design deficiencies including a selection of a material with relatively poor fatigue properties. On the other hand, several accidents could have been avoided if flight-simulation tests had been carried out with realistic load histories. Beyond any doubt, much has been learned from accident investigations, but it is the hard way of learning.

4 PREDICTIONS ON FATIGUE LIFE AND CRACK GROWTH

The case histories in the previous Chapter 3 illustrate how fatigue failures in service can be avoided if the reason for the type of failure is understood. It should be supported by research on design concepts and prediction models. It is then essential to consider two different successive periods of the fatigue life completed by a final failure. The sequence is:

(1) Fatigue crack initiation ! (2) Crack growth ! (3) Final failure

The first period is generally specified as being the crack initiation life. A precise definition of this period is difficult. Theoretically, an initial microcrack starts in the very first load cycle if the fatigue life is considered to be finite. It implies that a crack initiation life should not exist. However, a more pragmatic and still physical definition may be preferred based on the following arguments. As long as the material surface is affecting the growth of the initial microcrack, the material is still in the crack initiation period. The crack initiation period has been terminated if further crack growth is no longer depending on the surface condition and primarily depending on the crack growth resistance of the material as a bulk property. As an example this approach can be illustrated by considering fretting corrosion. The crack initiation period covers the period of increasing fretting damage. As soon as a fatigue crack is growing away from the damaged surface the crack growth period has started. A closely related definition of the transition from the initiation period to the crack growth period can be formulated as follows. In the crack initiation period the fatigue damaging process is depending on the material surface condition and the local stress cycles. However, after the transition to the crack growth period the material surface condition becomes irrelevant and fatigue crack growth is primarily depends on the crack tip driving stress history and the crack growth resistance of the material. In this second period "Fracture Mechanics" is meaningful with the crack tip stress intensity factor K as an important variable. In the crack initiation life the)K-concept is unrealistic.

Unfortunately, fatigue prediction problems are different for constant-amplitude (CA) loading and variable-amplitude (VA) loading. Comments on the CA problem will be made first, followed by the more complex problems offered by VA loading.

4.1 Prediction of the crack initiation life under CA loading

As said before, it is problematic to define the end of the crack initiation life. In the literature, it is generally assumed that fatigue predictions are covering the life to failure. It is tacitly assumed that the crack initiation period covers the major part of the fatigue life and the crack growth life is assumed to be relatively short. However, in aircraft structures built up from sheet material the crack growth life may be long and should be evaluated separately for a damage tolerance evaluation. The crack initiation life should then be considered as the fatigue life until some small cracks are present. The most simple case of fatigue critical locations in an aircraft structure is related to open holes or edge fillets for which K_t-values are available or can be calculated by a FE program. Predictions are then based on an empirical K_f - K_t relation which implies an extrapolation from data for unnotched specimens ($K_t \approx 1$) to notched elements [4]. It should be realized that this is not a small extrapolation step. The problem becomes more complex for joints where fretting corrosion can have a large influence on crack initiation. For riveted lap joints and for pin-loaded hole connections (lugs) this has led to empirical models [5,6] which implies that S-N curves are obtained by an extrapolation of standardized test data for these joints. Empirical equations are used to account for the geometry of the joint for which an S-N curve must be estimated. Extrapolation is still involved but now it means comparing similar types of structural elements. As always, extrapolations require a good deal of understanding and engineering judgment to assess the reliability of the predictions.

4.2 Prediction of fatigue crack growth under CA loading

The situation with respect to prediction of crack growth is entirely different. The growth of macrocracks depends on the crack growth resistance of a material, while the crack driving force is characterized by a ΔK-value. Considering that fatigue crack growth is a cycle-by-cycle process, a crack extension Δa in principle occurs in every cycle. This is illustrated by the striations in an old classic picture obtained at NLR, see Fig.10. Because of striation observations a prediction of fatigue crack growth might appear to be a simple procedure. For each cycle the crack extension Δa is equal to the crack rate in that cycle which can be obtained from a calibration curve, $da/dN = f(\Delta K)$*, as obtained in a crack growth test on a simple specimen. However, there are some pitfalls. Crack growth is depending on the stress ratio R which implies that $da/dN = f(\Delta K,R)$. Secondly, crack growth is also depending on the material thickness which is associated with plane strain / plane stress conditions. Moreover, a through crack might still have one simple dimension, which is the crack length a.

But for a part through crack, the shape of the crack front is no longer a simple straight line. The shape may become semi- or quarter elliptical. As a consequence the ΔK-value varies along the crack front. However, significant progress for such cracks is possible by FE analysis. Fawaz and Andersson have provided numerous K-solutions for such cracks starting from open holes and loaded holes [7].

An important aspect of fatigue crack growth is the occurrence of plasticity induced crack closure. This phenomenon was discovered by Elber in 1968 [8,9]. He wanted to open a central crack. After a saw cut through the first edge ligament, the specimen exhibited an unexpected but visible in-plane deformation of the sheet. Elber realized that this must be caused by some plastic deformation left in the wake of the crack during the preceding crack growth. It implies that during unloading the upper and lower fatigue crack surfaces must touch while the

* Results of experimental research to obtain crack growth data are often presented in graphs with da/dN as a function of ΔK. Because ΔK accounts for both stress level and crack length, the range of crack sizes involved is then unknown. Graphs of $a = f(N)$ and $da/dN = f(a)$ should also be given as unbiased and direct information of the test results. Futhermore, visual observations of the fatigue fracture surface are not always presented. However, they are part of the result of a fatigue test. Information should be provided on any characteristic feature of the fracture surface. For notched specimens and joints the location of crack nuclei can provide significant information to understand the test results`

specimen is still under tension. Crack closure starts at the crack tip which removes the singularity of the stress distribution at the crack tip. During reloading the crack tip will open at a positive stress level, S_{op}. Only the stress range with a fully open crack tip, $\Delta S_{eff} = S_{max}$ - S_{op}, is assumed to determine the crack length increment, Δa. The concept of a ΔK_{eff} is now generally accepted and used in prediction models, but it should be realized that the concept includes the assumption that the fatigue crack extension Δa is fully controlled by ΔS_{eff}. The Elber-concept is also referred to as plasticity induced crack closure.

4.3 Variable Amplitude (VA) loading

Fatigue loads in service are rarely a pure CA loading with a constant S_{max} and S_{min}. This offers the well-known problem of fatigue damage accumulation under variable-amplitude (VA) loading. The variable character is most obvious for gust load spectra on a wing, while the pressurization cycle of a fuselage is more related to CA loading. The problem of predictions on fatigue under VA load was recognized quite long ago. A classical paper was published by Pålmgren in 1924. He proposed the summation of percentages of fatigue damage (Σn/N = 1) which now is usually referred to as the Miner rule. In the early days this rule was frequently checked in laboratory investigations by fatigue tests with simple block load histories as shown in Fig.7a. Sequence effects were observed indicated by En/N values significantly lower and also higher than 1. Also in crack growth tests under similar VA block loading sequences, it was observed that a simple summation of crack growth increments did not occur. A most noticeable effect was crack growth retardation after an intermittent high load as observed already in the sixties 1962 [10].

Block program loading (Fig.7b) was introduced by Gassner [11] in 1939. The purpose was to represent load spectra occurring on structures in service. Actually, the fatigue machines in those days could not apply a more realistic load history. However, in the 1950's it was realized that ground-air-ground load cycles could significantly contribute to fatigue damage of aircraft structures. Simple flight-simulation load histories were then adopted in full-scale fatigue tests with the same load sequence applied in all flights, see Fig.7c. This type of flight-simulation test was applied in the full-scale test on the Comet aircraft as discussed before.

Laboratory investigations in later years have shown that significant sequence and interaction effects are not taken into account in the simplified flight-simulation tests. It can lead to misleading information in a full-scale fatigue test which can be understood with the present knowledge about these effects. A real breakthrough occurred after the introduction of hydraulic fatigue machines with closed loop load control. Any load sequence can now be applied by generating computerized command signals. The technology was first introduced in laboratory fatigue machines, but it is now generally used in full-scale testing of structures. A sample of a load history in a flight-simulation fatigue test is shown in Figure 8.

With respect to fatigue lives to be obtained under VA-loading, the Miner rule was extensively verified. A survey paper was published by Schütz [12]. It illustrates the wide range of Γ(n/N) values obtained varying from much smaller than 1 to much larger than 1. The basic reason is that fatigue damage cannot be characterized by a single damage parameter [4]. Load sequence effects cannot be accounted for by the Miner rule. As a consequence, the rule is unreliable for comparison between the severity of different load spectra. Verification by experiments is then obligatory.

A prediction on fatigue crack growth under VA-loading is significant for the analysis of damage tolerance. Crack growth under VA loading has been studied in numerous research projects which have been surveyed in a number of papers [13-15]. Because of the well known interaction effects, prediction models have been published to account for these effects. The oldest models (plastic zone models: Wheeler, Willenborg and derivatives) were followed by crack closure models where plasticity induced crack closure was included (Onera model, Preffas model, Corpus model). In these models the crack opening stress level was still calculated with empirical relations [16,17]. At the present time, the so-called strip yield models are the most advanced prediction models [18-22]. The basic idea for these models is coming from the original Dugdale model for yielding of the plastic zone ahead of the crack tip [23]. In the strip yield models crack closure is now calculated by considering plastic deformations in plastic elements at the crack tip. However, some empirical tuning of the model for the material to which it will be applied appears still to be necessary. An advantage of the strip yield models is that delayed crack growth retardation (Fig.11) can be predicted, where this is not possible with the older crack closure models.

An extensive comparison has been made between predictions with the Corpus model and experimental crack growth results obtained under a large variety of flight-simulation testing of 2024-T3 and 7075-T6 sheet specimens [16,17]. The test programs included the effects of the design stress level, the shape of the load spectrum, truncation of the most severe loads of the load spectrum, and the severity of the ground-to-air cycle. In general the predictions of the crack growth curves were satisfactory. However, an elaborate fractographic analysis indicated that the crack extension in the most severe flights was underestimated by a factor of 2 to 3 [24]. Because the number of the most severe flights was rather small the incorrectly predicted crack extension of these severe flights had an insignificant influence on the crack growth curve. This raises another question about how a comparison between test result and prediction should be made. It can be done in different ways which are: (i) Comparison of crack growth lives, which is a crude criterion. (ii) A comparison of crack growth curves, which makes more sense for damage tolerance evaluation.(iii) A comparison of crack rate *da/dN* as a function of the crack length *a*. Some averaging is still involved. Local overestimates and underestimates may cancel. (iv) Comparison between striation patterns and cycle-by-cycle predictions. The last option is the most precise one to judge the physical validity of a prediction model. The criterion to be chosen should obviously depend on the purpose of the prediction. The more complex strip yield models are still insufficiently verified by comparisons between results of a large variety of VA test programs and the corresponding predictions. There is still a challenging research issue.

5 DESIGNING AGAINST FATIGUE AND DESIGNING FOR DAMAGE TOLERANCE

The discussion in the previous chapter was touching on available prediction models for fatigue lives and crack growth. It should be admitted that highly accurate predictions can not be guaranteed. As a result, experimental guidance and verifications may be necessary. Although available tools for calculations and experiments are powerful by now, the initial efforts in the design office should be associated with relevant design options including new design concepts. In the present chapter attention will be focused on designing against fatigue and designing for damage tolerance. It will be done by considering some topics of current interest associated with structural concepts, material selection and production, the items drawn from the first column in Figure 2. Secondly, the accuracy of fatigue predictions will be addressed in view of supporting experiments being necessary. Finally some comments are made on realistic component and full-scale testing. Solutions for problems have economic consequences, but this aspect will not be addressed explicitly because arguments depend very much on strategies of the industry and the airlines.

5.1 Design aspects

1. <u>Fuselage skin cracks and crack stopping elements</u>

Since the Aloha accident in 1988 fatigue cracks in riveted lap joints of an aircraft fuselage are supposed to be fatigue critical elements of the structure. The occurrence of MSD has also been observed in other aircraft as shown by the data in Figure 12 [25]. Small cracks wer initiated at rivet holes of a longitudinal lap joint more or less midway between the frames. This should be expected because the hoop stress is usually lower at the frames due to the constraint on radial expansion. This pillowing effect depends on the stiffness of the frame to skin connection. Significant differences between the hoop stress at the frames and midway between the frames were reported by Miller et al. [26] for barrel tests with crack stopper bands (tear straps) at the frame section. Differences were in the order of some 30%. A good agreement was obtained between calculations and strain gage measurements. In other words, the problem is open to stress analysis. It may be noted here that an inhomogeneous hoop stress distribution is favourable for detection of MSD before it becomes widespread. However, an inhomogeneous stress distribution is not optimal for minimizing the weight of an aircraft structure.

The Aloha accident has stimulated the application of crack stopper elements. Two different options are sketched in Figure 13b. Local patches of a Ti-alloy are applied in Airbus aircraft in order to stop fatigue cracks in the longitudinal lap joints. However, crack stopper bands fully around the fuselage (Fig.13a) can also stop crack growth from unintentional damage outside the lap joints. Obviously, the choice has consequences for the damage tolerance evaluation, and also for the construction of the fuselage and the production in the shop. It may be repeated that FE calculations can be highly instructive for the comparison of different design concepts. The frame distance could also be an interesting design variable. Obviously, such an exploratory design analysis requires skill, imagination and engineering judgment, quite a challenge.

2. Riveted lap joints and single strap joints

Riveted joints in fuselage structures have caused fatigue problems in several aircraft types. Actually, this is not surprising if it is realized that a connection between two skin sheets is made by a number of local connections, i.e. by fasteners. A continuous connection obtained by an adhesive bonded lap joint is a much superior joint with significantly better fatigue properties. True enough, it requires a different production technology and quality control. Moreover it may be recalled here that the number of pressurization cycles of a fuselage in the service life of an aircraft is not extremely high from a fatigue point of view. The old Aloha Airlines aircraft compiled 89680 flights which is a relatively large number for a transport aircraft, and still not exceeding $N = 10^5$ cycles.

A major aspect for fatigue of a lap joint is the eccentricity occurring in the overlap, see Figure 14. A tension load introduces so-called secondary bending. This also applies to single-strap joints used for circumferential fuselage joints. Depending on the joint design, the bending stress can easily be as large as the applied tensile stress. A first estimate of secondary bending can be made by the simple neutral line model [27]. It shows that a larger overlap significantly reduces secondary bending; see the bending factor in Fig.15 [28]. In general, the load transmission in a multi-row lap joint occurs for a larger part by the first and the last rivet row (rows 1 and 3 in Figure 14). Fatigue cracks are initiated in these rows. Calculations on the load transmission by all rivets rows are more realistic if fastener flexibility is taken into account, see [29, 30]. However, the outer rivet rows are still the most critical ones. Local reinforcements of the overlap have been applied in fuselage structures to improve the fatigue properties. It reduces the nominal stress level but it increases the eccentricities, and thus secondary bending. Secondary bending calculations can then be instructive. Instead of local reinforcements, secondary bending can be reduced by a local reduction of the sheet thickness, see the right hand part of the table in Figure 15. Moreover, it will reduce the load transmission in the critical outer rows, but the effect must still be verified by fatigue tests. Anyway, it may be stated that stress analysis can contribute to an improved design of riveted joints.

3. Lug connections

In an instructive movie produced by the FAA [31] on fatigue damage tolerance of aircraft structures, a sketch of a triple lug is presented as a kind of a fail-safe design, see Fig.16. Lugs are known to have relatively low fatigue strength. The movie suggests that serious fatigue cracking in one lug, implies that the other two lugs will still have a substantial load carrying capacity. However, a fatigue crack in one of the lugs will substantially reduce the load transmitted by this lug because of the reduced stiffness. As a consequence, the other two lugs must carry an increased load. It may not be expected that crack initiation in the other two lugs will wait until the first one has a large crack. Cracks will probably grow more or less simultaneously in all three lugs. The Leonardo da Vinci argument for the extra chord is not applicable to the triple lug. Furthermore, inspection of a triple lug for cracks starting inside the hole requires dismounting of the joint and a special inspection technique. This example illustrates that a realistic scenario of possible failure modes and consequences for maintenance and inspections should always be made. Again, imagination is essential.

4. Back-to-back structure
The so-called back-to-back structures have been adopted by some aircraft industries. Components, usually small ones, were cut in two parallel parts which then were again joined by adhesive bonding. An example is shown in Figure 17. The idea is similar as for the triple lug. If a fatigue crack occurs in one part, the other part can still carry some load. Also in this case the benefit is questionable, and it requires an extra production step.

Material selection

Relevant material properties for fatigue and damage tolerance are fairly well documented in the literature, and also in industry handbooks. In other words, the designer knows whether a material has good or poor fatigue properties. Also data about residual strength properties and the fracture toughness of a material are usually available. If a high static tensile strength is required for a component, high strength alloys may be a good choice. For aluminium alloys the relatively high $S_{0.2}$ and S_U of 7075-T6 can be advantageous if compared to the lower values for 2024-T3. However, fatigue lives of notched elements and crack growth properties are usually better for the 2024-T3 alloy which may well be related to a better ductility of this alloy. It should be remarked here that good fatigue properties of a structural component can also be achieved by a locally reduced nominal stress level. The weight penalty may be limited. Moreover, a special production treatment can be adopted to introduce favourable compressive residual stresses in the fatigue critical notch area. A well-known example is plastic hole expansion for which apparatus is commercially available. It then is noteworthy that larger compressive residual stresses can be introduced in 7075-T6 if compared to 2024-T3. It is a matter of judgment whether a designer will choose for such a "trick". Some reluctance stems from a more elaborate quality control. Another solution for lugs is to insert a bush with an interference fit which causes favourable pretension around the hole.

New Al-alloys are almost continuously under development. Al-Li-alloys are not yet generally considered to be a good choice, but it may change in the future. The purified 2524 alloy is an alternative for the 2024-T3 material because of a high K_{Ic} which is favourable for residual strength of a cracked structure.

An entirely different approach to fatigue problems is to use fatigue insensitive materials. Two candidates are fiber-metal laminates, e.g. Glare, and carbon fiber reinforced plastics (CFRP's). The application of CFRP will not be discussed here because these composites have an essentially different material behaviour as compared to alloys of Al, Ti and steel. Advantages and disadvantages of CFRP's do not allow a simple comparison if fatigue and damage tolerance properties must be considered. This aspect does not apply to fiber-metal laminates which behave more or less as a metallic material, except for the extremely high crack growth resistance [32]. The fiber-metal laminates should be considered instead of the pure aluminium alloys if fatigue is a prominent design criterion. A well-known case is the application of Glare sheet material in the fuselage of the Airbus A380. Another research example is associated with the connection between the wing and fuselage of the CN-235. The connection is made by four aluminium alloy lugs which for obvious reasons should have a superior fatigue strength and a low design stress level. In a master study such a lug was made from Glare, thickness 19 mm, 25 thin metal layers, see Figure 18. In a flight-simulation fatigue test (10 different types of flight severity) the design stress level was increased to about twice the original level. Failure did not occur in the lug, but outside the lug component in the clamping after 92000 flights. Small cracks were detected in three layers which is insignificant for the static strength.

Production

The surface quality of a component can be significant for the fatigue performance if the joints are not the most critical part. Usually the surface quality is rarely included in a damage tolerance evaluation. It is expected that the surface quality will be good anyway. Of course the material surface is important in view of corrosion and durability issues. In the past fatigue crack initiation in service was occasionally accelerated by poor machining, for instance due to a poor hole quality, but actually such mistakes should not be made. On the other hand, certain production variables can be chosen in order to improve fatigue properties. A most remarkable improvement of the fatigue properties of riveted joints was achieved by increasing the squeeze force during the riveting operation [29]. The application of this procedure requires load controlled rivet squeezing instead of the displacement controlled squeezing.

5.2 Accuracy aspects of fatigue predictions

Predictions on fatigue properties are part of design studies. It then is important to have some idea about the accuracy of the results. Predictions on fatigue life and crack growth are two different issues as already discussed in Chapter 4. Some comments on accuracy aspects of both topics are separately presented below. It includes remarks about supplementary experiments which may be necessary in view of unreliable prediction models.

Prediction of fatigue lives

It was emphasized in Section 4.1 that predictions on the crack initiation life imply extrapolations from available experimental data. It must be realized that the extrapolation step can be fairly large which means that the accuracy may be limited. The best results to be obtained should start from data for specimens which have a good deal of similarity with the structure for which estimates have to be made. Understanding, experience and judgment are very much necessary to obtain some idea about the significance of the results obtained. It may then be concluded that fatigue tests are necessary. In order to avoid new uncertainties, the tests should be as realistic as possible, both with respect to the test article and the load history.

If S-N data are available the Miner rule may be adopted to calculate the fatigue life under spectrum loading. However, the Miner rule is unreliable for this purpose. Actually, the Miner rule at best gives some weighted average of the shape of a load spectrum, but it may not be expected to provide a reliable indication of the severity of a load spectrum. Simplified flight-simulation tests have also been considered to be attractive because of the simple load sequence, overlooking that the most simple approach is to simulate the service load spectrum with a randomization as it occurs in service. In this respect, the Lo-Hi-Lo sequence between ground-to-air cycles as shown in Fig.7d is artificial.

Predictions of fatigue crack growth

Several prediction models for fatigue crack growth under VA loading were listed in Section 4.3 based on ΔK_{eff} to account for sequence effects. The strip-yield models are the most advanced ones. It was mentioned that problems were still left in view of plane-strain / plane-stress situations, and thus to material thickness effects. Interaction effects on fatigue crack growth are larger for the 2024-T3 alloy than for the 7075-T6. In general, interaction effects are larger for more ductile materials whereas these alloys usually have the better fatigue properties. The 2024-T3 alloy is a difficult material for crack growth predictions.

An interesting question may be raised here. A realistic simulation fatigue test for crack growth information is much simpler than for the fatigue life of a component. For tests on a component, the test article must be an exact copy of the component in the aircraft structure. Also the load introduction in the structure must be correctly simulated because crack initiation can be sensitive to these conditions. However, for the crack growth period the crack initiation conditions are no longer relevant. This implies that boundary conditions can more easily be relaxed. The question may then be raised, why not prefer a flight-simulation test rather than adopting a complex crack growth model which still must be tuned and which still leaves us with some unknown uncertainties. If the industry is using a fatigue prediction model some experimental verifications are thought to be obligatory anyway.

5.3 Results from full-scale fatigue tests

Full-scale fatigue tests on large parts of the aircraft are generally supposed to be a final key to confirm the fatigue performance of the structure. In the past it was though that such a test was necessary in order to reveal some unexpected weaknesses associated with inadequate design. In chapter 4 some cases were discussed where a realistic flight-simulation test would have indicated the failures which led to a fatal accident. Moreover, those accidents would also be prevented by a good fatigue analysis with a more refined FE stress analysis and a more careful material selection. Anyway, the full-scale test is still considered to be necessary in view of possible deficiencies of the aircraft structure. Further more, artificial damage can be introduced later in the test in order to see how it will spread by fatigue crack growth. Also residual strength tests can then be carried out.

It has been a matter of debate about how many times the planned service life should be simulated in a full-scale test. It now is generally focused on twice the service life. It is not clear why safety factors on life and inspection periods should be whole numbers, and not fractional numbers, although a safety factor of 2.5 should be more safe than 2.0. Full-scale tests also offer well-known experimental problems with respect how to apply aerodynamic loads on the structure, and how to run the test as fast as possible. Full-scale tests require large investments of time and money. In view of obtaining a maximum profit of such a test a most careful planning is necessary.

6. CONCLUSIONS

1. Designing against fatigue is much appreciated by the aircraft operator. It can be done along lines associated with clever structural design concepts and improved joints. Evaluations along these lines can benefit from detailed stress analysis on the influence of dimensions of the structural concepts, but potential improvements must be verified by realistic flight-simulation tests.
2. Designing for damage tolerance is necessary in view of safety arguments. It encompasses two different problem settings. Much attention is paid to scenarios how to prove that a structure can tolerate growing damage in agreement with airworthiness regulations. In this way it is a safety argument. The other problem is a design problem. How can we arrive at structural concepts which will assure slow growth of damage, and thus less frequent inspections and limited repairs. Although these aspects are still safety arguments, improvements will be welcomed for economic reasons.
3. Predictions of fatigue properties and experimental verifications are most important tools. The present qualitative understanding of problems involved is reasonably well developed. Accidents occurring in the past can now be prevented. However, it must be admitted that fatigue predictions on the crack initiation life are still problematic. They can not be satisfactory solved without relevant fatigue tests. With respect to fatigue crack growth in structure predictions appear to be possible, but also for this topic some more competence should be developed. The same is certainly true for final failure modes.
4. Various aspects of fatigue and damage tolerance of aircraft structures are qualitatively well understood, certainly good enough to define the problems for which better expert knowledge should be achieved. Analysis in depth of failure modes and improved prediction models must be recommended keeping in mind the relation to practical problems. Engineering judgment is essential.
5. Because the full problem setting is characterized by multi-disciplinary aspects, it is very much necessary to present relevant courses, both in the industry and at universities.

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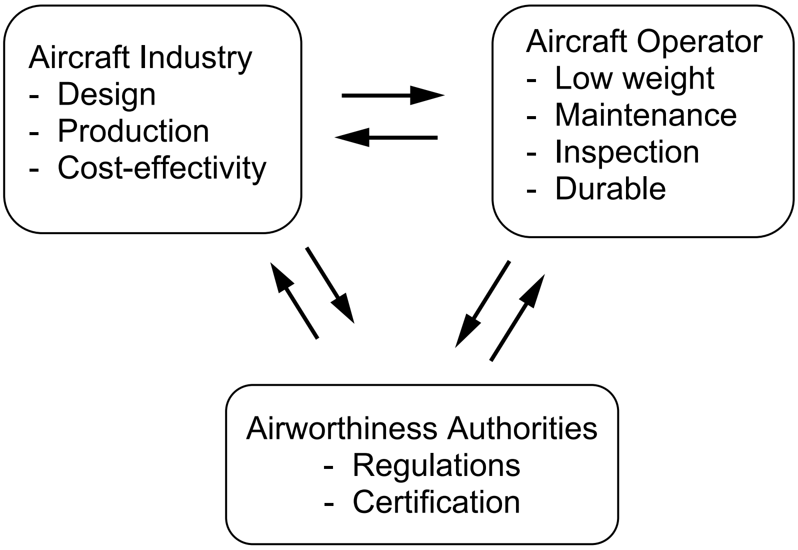


Fig.1: Three parties with mutual contacts

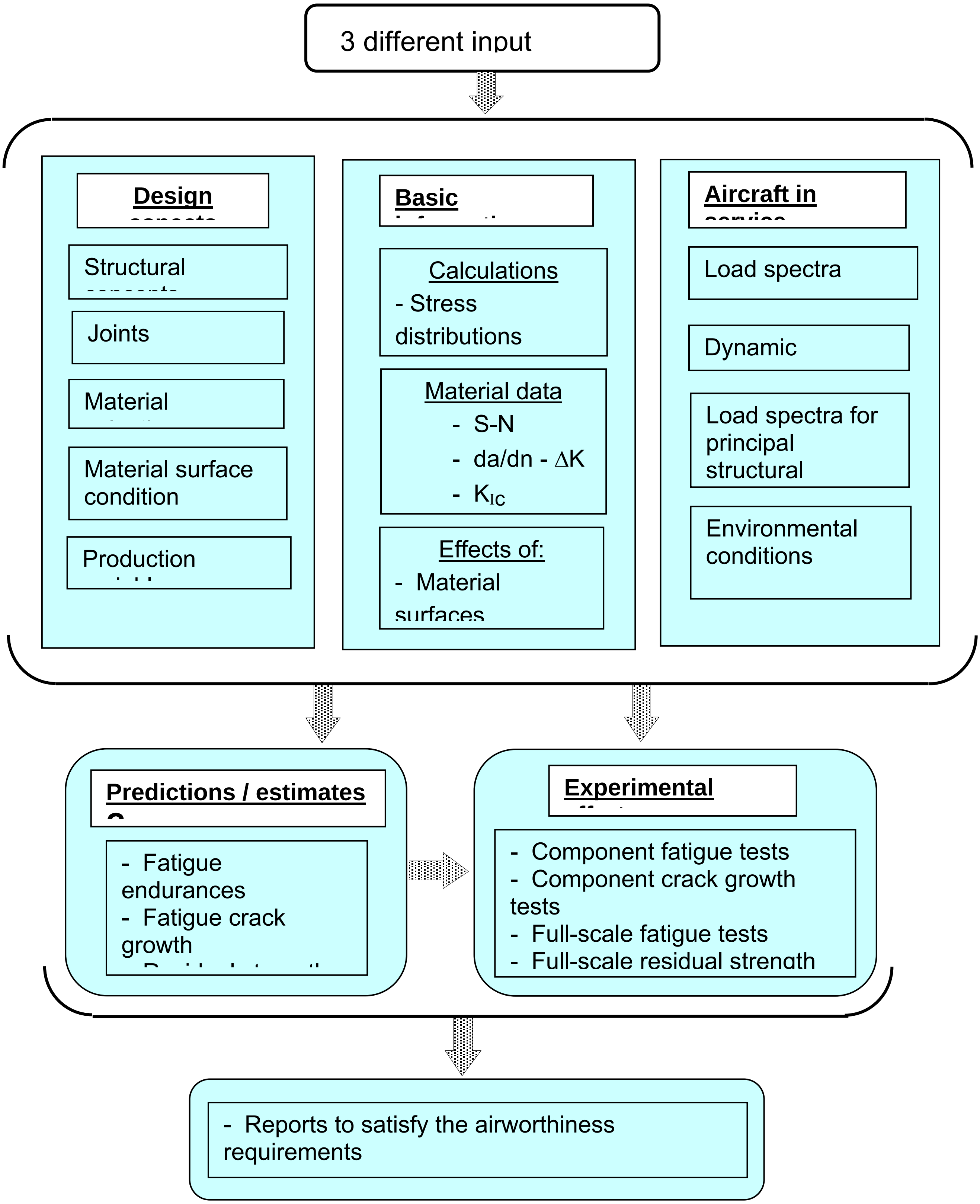


Fig.2: Highly different disciplines are involved in fatigue and damage tolerance of aircraft structures.

Elemtary steps in the Damage Tolerance Evaluation

1. Define the aircraft usage

2. Develop global load spectra

Aircraft Missions
Different types of loads

3. Select critical locations for analyzing Principal Structural Elements (PSE)

4. Consider possible sources of damage
   Fatigue cracks, corrosion, impact damage

5. Develop load spectra for each PSE

How is the PSE loaded?
Single load path or multiple load path ?
Consider failure scenario.

6. Calculate nominal stress levels for PSE's and local stress levels at critical locations

FEM calculations
K_t - values

7. Collect crack growth data and K_{Ic}

8. Predict crack propagation starting from initial damage

9. Calculate residual strength as a function of the crack size

Initial damage = damage present in a new aircraft, or damage caused early in service (conservative assumption)

Difficult, but simplified and conservative calculation can be acceptable

10. Decide on the inspection method

11. Determine detectable crack length

12. Decide on inspection intervals

Operators prefer visual inspections

Fig.3: Elementary steps in the Damage Tolerance evaluation.

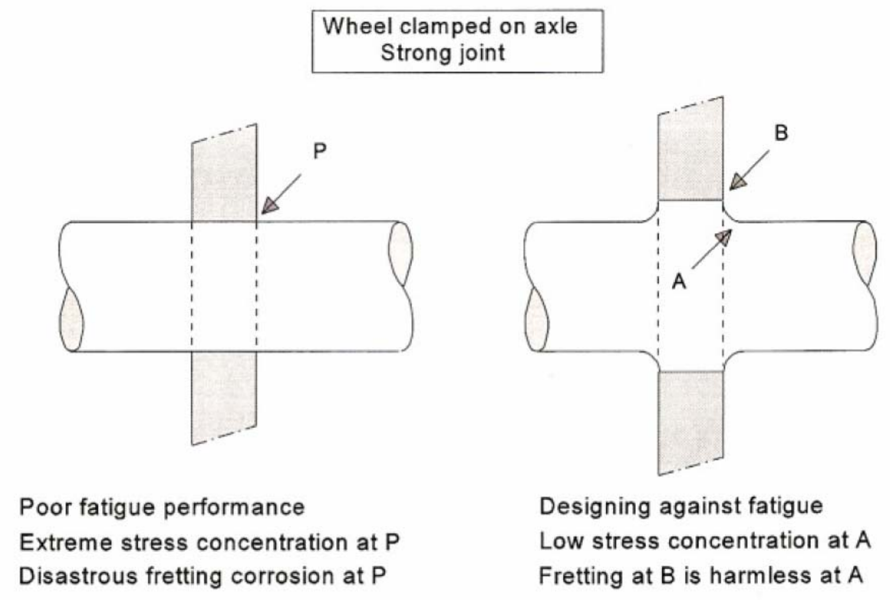


Fig. 4: Designing against fatigue of a railway axles in the 19th century by Wöhler.

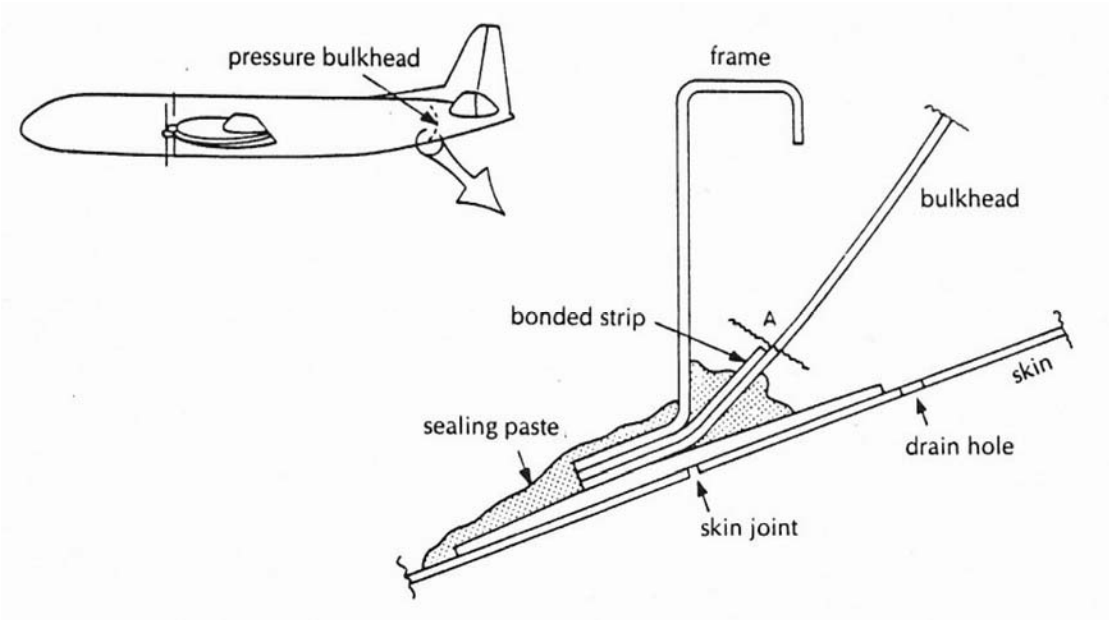


Fig.5: Circumferentially growing fatigue crack in a pressure bulkhead initiated by corrosion damage at location A. Complete failure of the bulkhead caused a crash of a Vanguard aircraft.

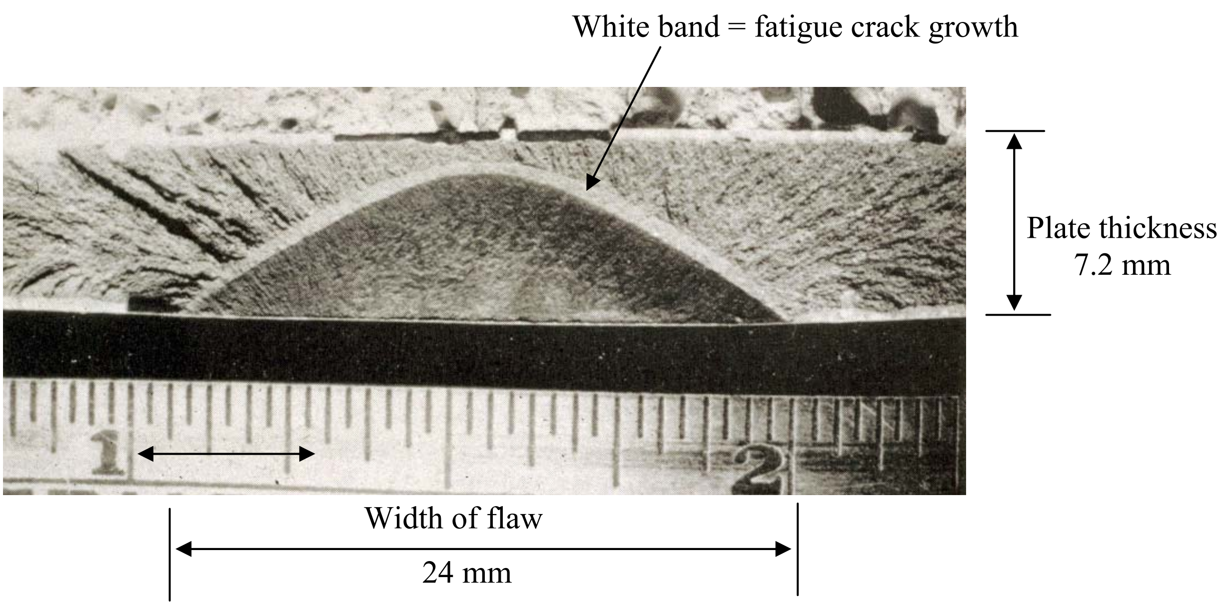


Fig.6: Wing failure of the F-111 due to a material flaw in the lower wing skin with a small crack extension by fatigue in 120 flying hours.
Material: D6AC steel, S_U = 1600 MPa, K_{Ic} between 40 and 100 MPa√m for different lots.

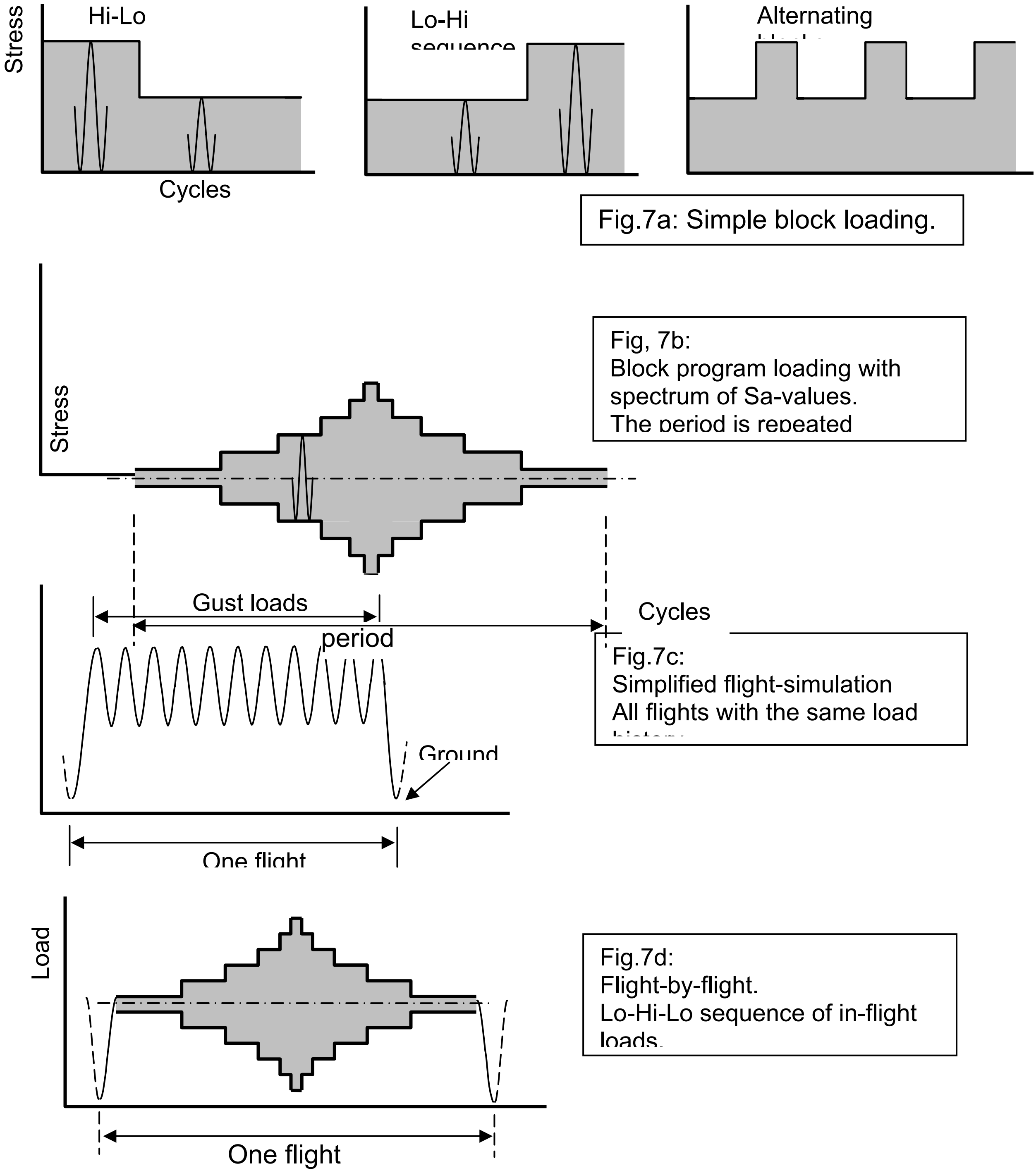


Fig.7: Different types of Variable-Amplitude loading.

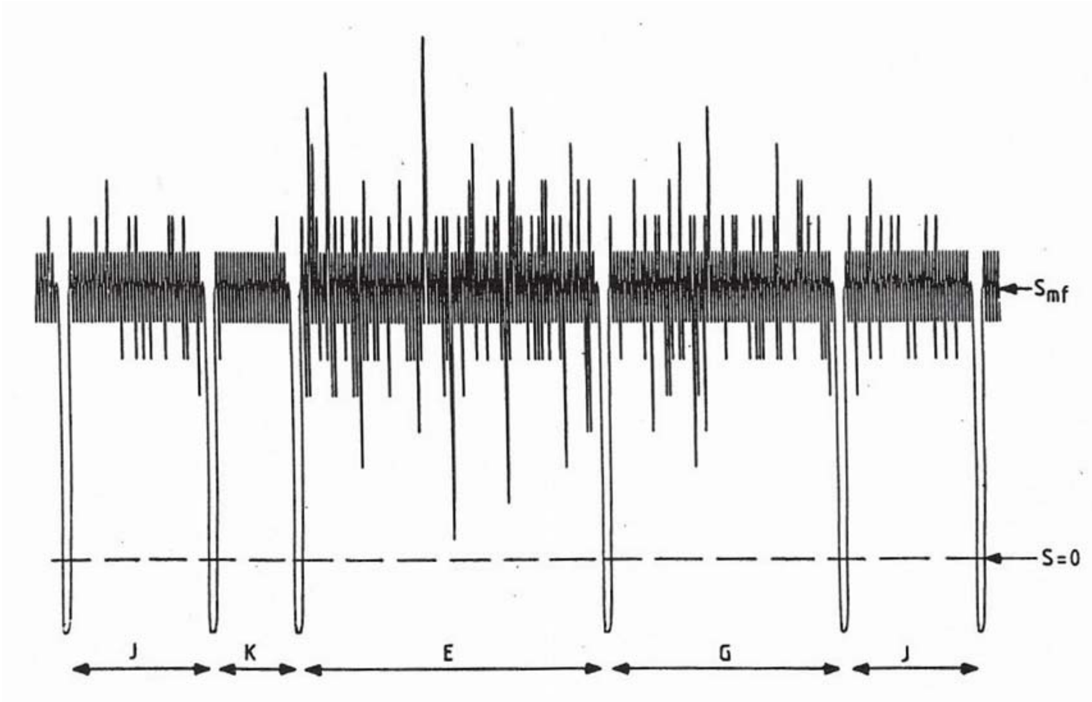


Fig.8: Sample of a load history applied in a flight-simulation fatigue test employing the load spectrum of the Fokker F-28 wing structure. Five flights are shown with gust loads corresponding to different weather conditions. Ten different weather conditions are simulated. For each flight the the sequence of gust loads in randomized. S_{mf} is the mean stress in flight.

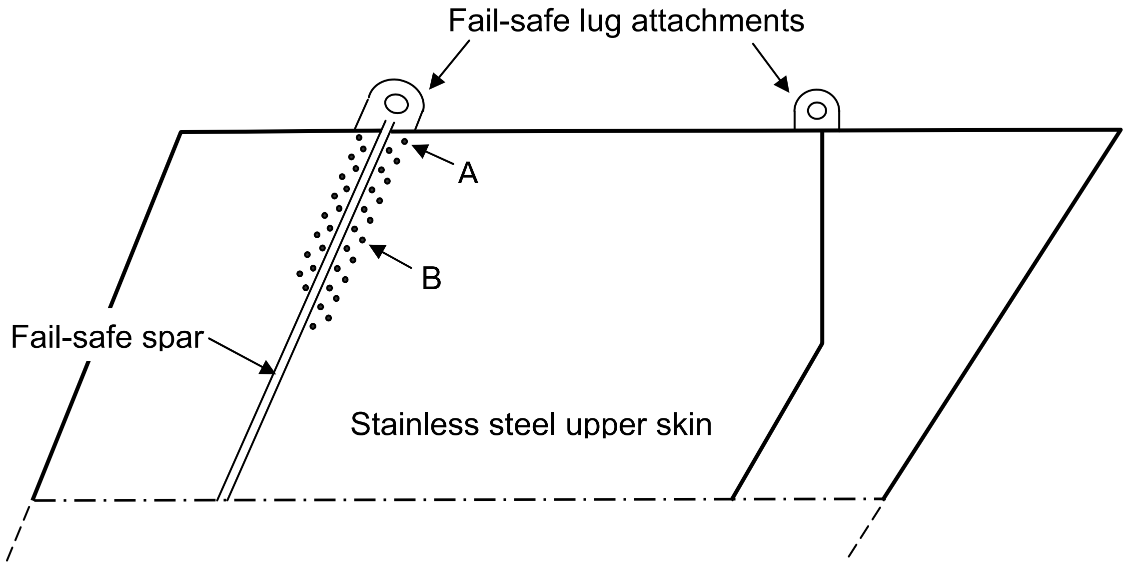


Fig.9: Lusaka accident. Horizontal stabilizer failed by fatigue.
Fatigue crack in the spar expected at bolt hole A, but it occurred at bolt hole B.

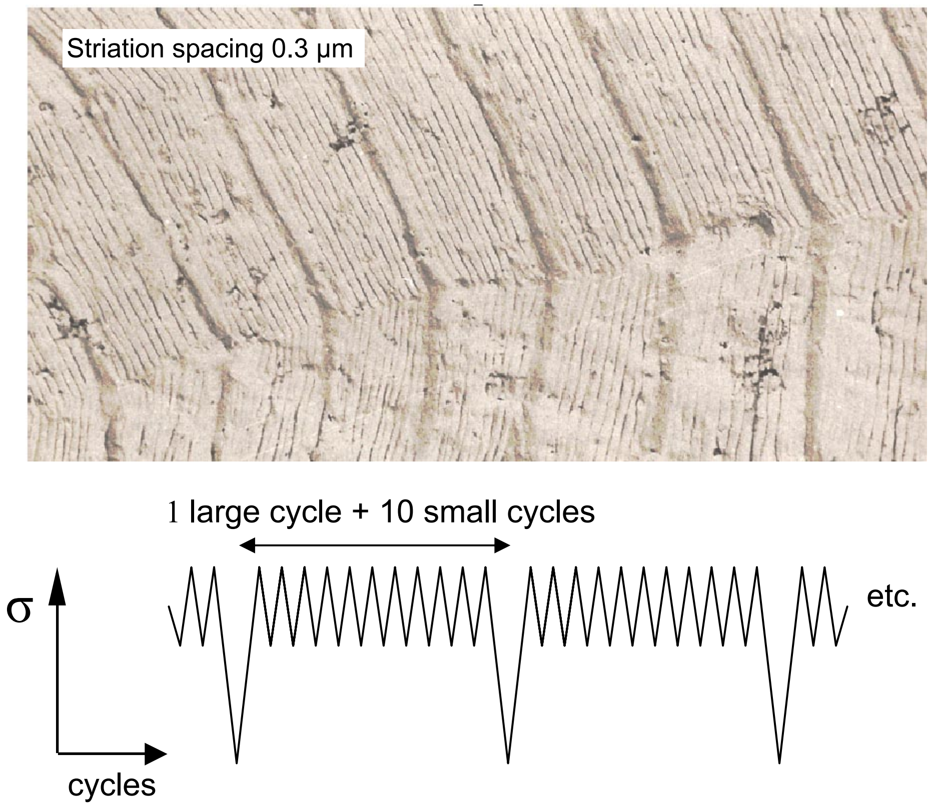


Fig.10: Correspondence between striations and simplified flight simulation. Electron microscope picture of fatigue crack in 2024-T3 specimen. Picture of the National Aerospace Laboratory NLR, Amsterdam.

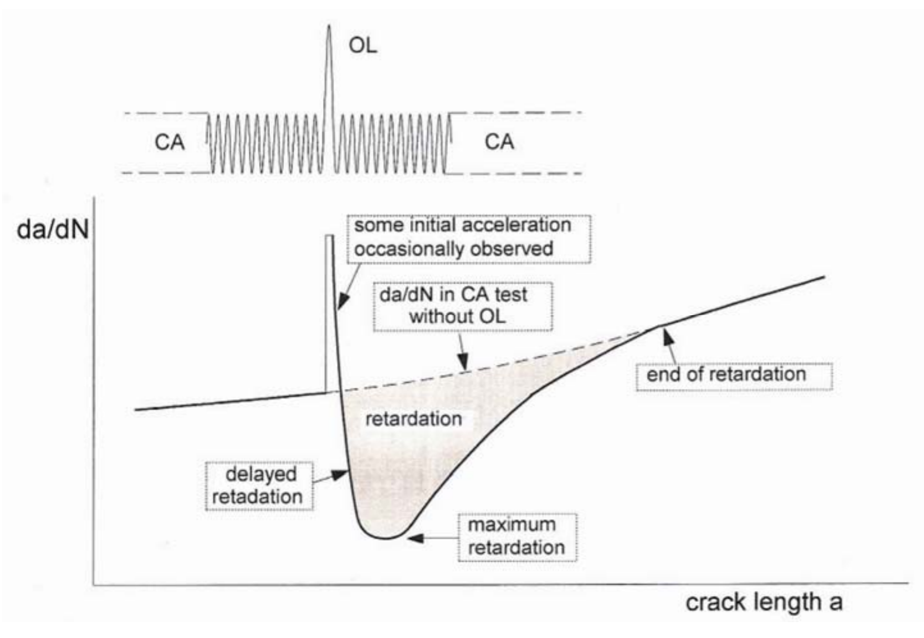
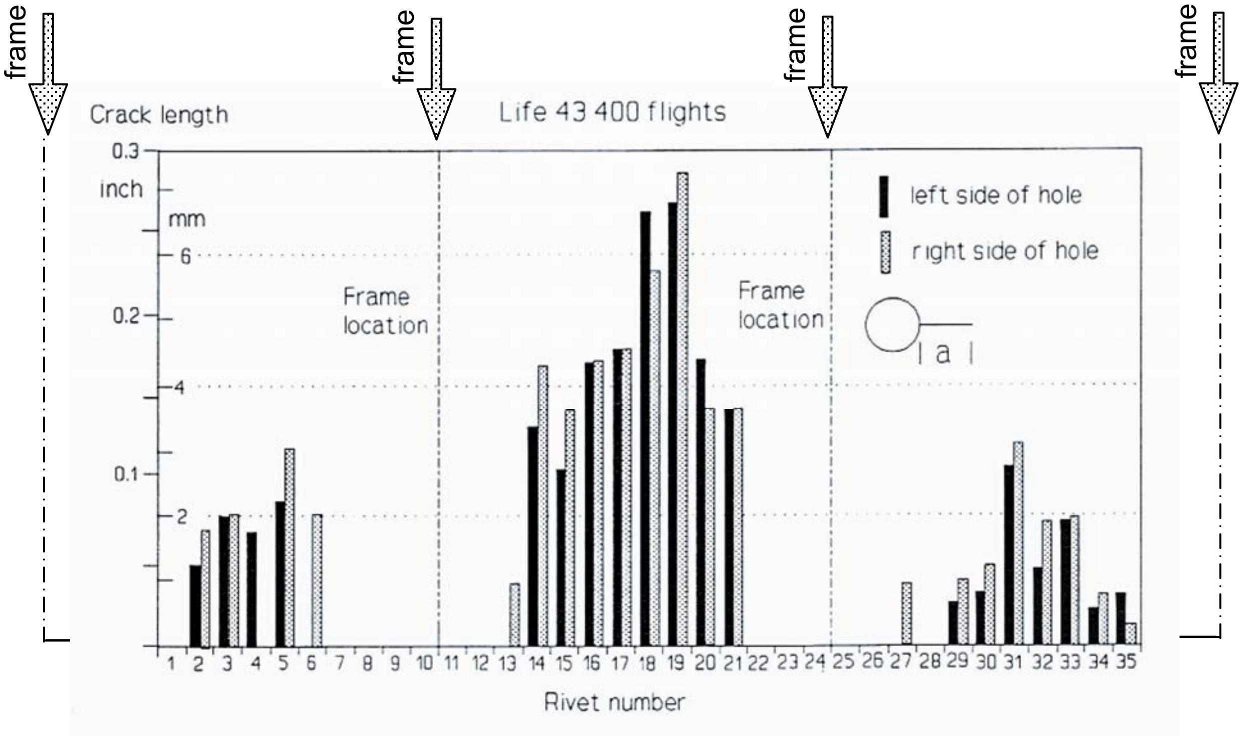
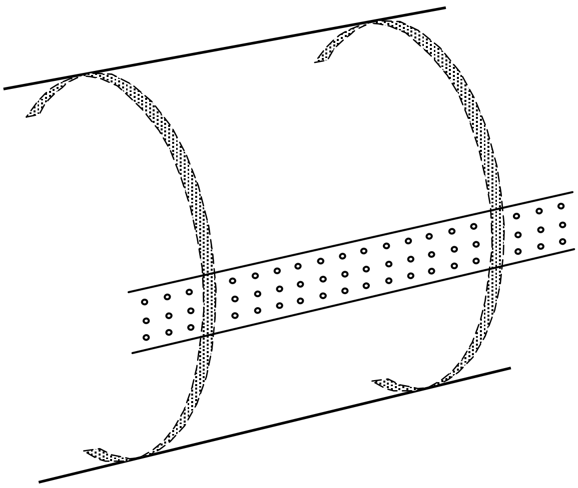


Fig.11: Delayed retardation after an overload (OL) predicted by strip-yield models only.



Mutilple-site damage (MSD) in a Boeing 727 fuselage

Fig.12: Many small cracks at rivet holes (MSD) in three bays between fuselage frames. No cracks near the frames because of a lower hoop stress.



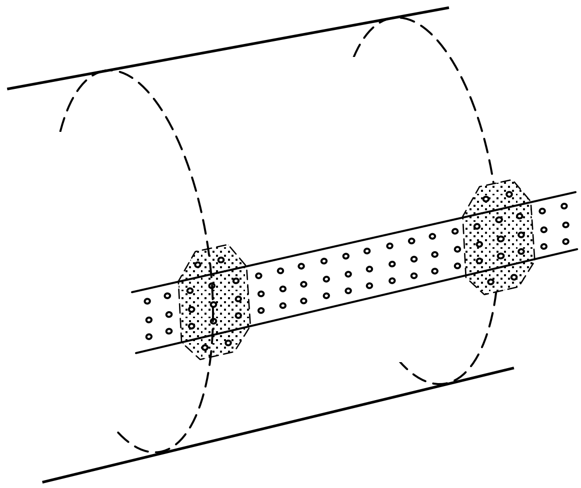


Fig.13a: Continuous crack stopper bands around the fuselage, stopping fatigue cracks in the lap joints and other cracks outside the lap joint.

Fig.13b: Local crack stoppers at the lap joints only.

Fig.13: Schematic pictures of two different designs of crack stoppers in a fuselage.

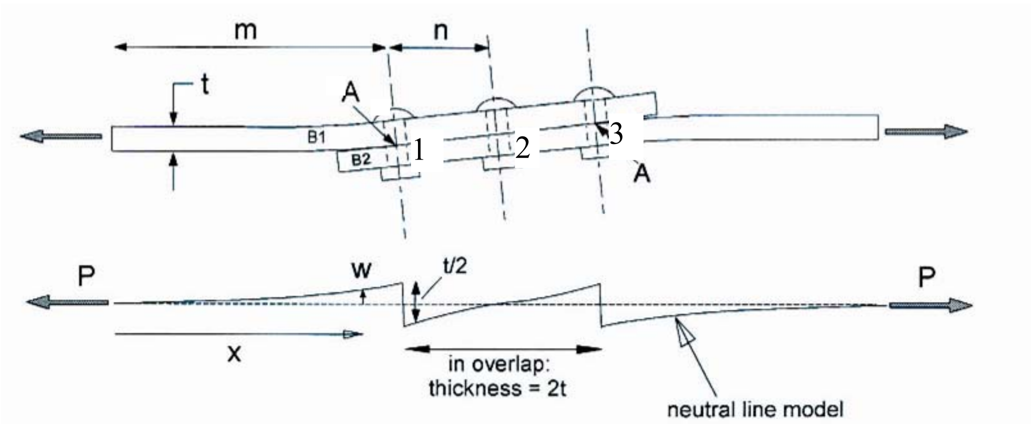


Fig.14: A tensile load on a riveted lap joint induces secondary bending.
The bending stress can be estimated with the neutral line model.

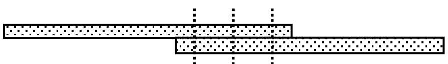
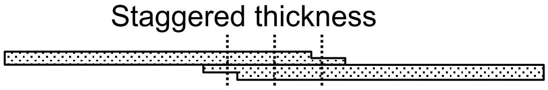
		Staggered thickness 	
Row spacing (mm)	K_b	Row spacing (mm)	K_b
20 30 40	1.46 1.21 1.06	30 45 60	1.14 0.97 0.78

Fig.15: The effect of the distance between the rivet rows on the bending factor K_b [28]

$K_b = \sigma_{bending} / \sigma_{tension}$

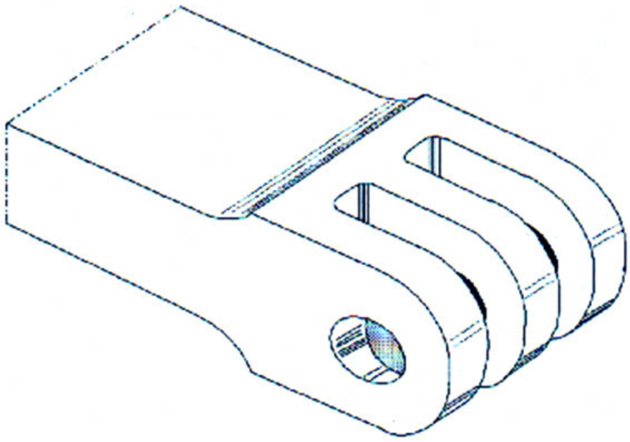


Fig.16: Lug connection with three lug heads.
Should it be considered to be damage tolerant ?

Fig.17: Back-to-back component.
Damage tolerant ? Fail-safe ?

Fig.18: A lug of Glare-1 designed for a wing to fuselage connection.
In the critical section the lug thickness was 19 mm, Glare was built up with 25 thin layers. Hole diameter 44 mm. In flight-simulation tests at a high stress level failure did not occur.