18TH PLANTEMA MEMORIAL LECTURE

Anders F. Blom

FATIGUE SCIENCE AND ENGINEERING – ACHIEVEMENTS AND CHALLENGES

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Abstract: Fifty years after the foundation of ICAF, fatigue of aircraft remains a fundamental problem although significant achievements have been made over the years. A brief summary of historical incidents and breakthroughs is given together with a discussion of developments in academic fatigue research. The various ingredients in fatigue design, certification and management of aircraft structures are discussed and summarised in a brief state of the art review. Current demands and future trends for military and civil aircraft are discussed and potential fatigue related problem areas are highlighted with a few warnings made. Not only technical and financial aspects will govern the future of aircraft design and operations, but also the ongoing decline in military spending and related political decisions. Together with new societal values in an ever increasing globalisation of the world, future education systems and availability of skilled personnel are factors that need consideration if aviation is to remain as relatively safe as it has now become. A few words along these lines end this paper.

INTRODUCTION

As ICAF celebrates its 50[th] birthday, we are also experiencing the first ICAF-meeting in the new millennium. At this moment many things in aeronautics are undergoing substantial changes, largely due to reduced military developments, resulting from new political balances in the world, but also from mergers into fewer but larger aeronautical companies in the western world. Reduced manufacturing costs and competition based on finances, passenger comfort and environmental issues are taking the lead from competition based on technical superiority, safety and total operating cost.

Whilst structural failures and fatalities due to fatigue related problems have been dramatically reduced since the pioneering days of aeronautics, it would be wrong to state that the problem of structural fatigue has been solved and that such failures will not occur in the future. In the last decade the problem of ageing aircraft has drawn much attention and the world wide interests in the so-called multi-site damage or wide spread fatigue damage problem has been enormous. Especially in the USA the total research funding on this problem must have been one of the largest amounts ever spent to solve an engineering problem.

In this paper we will provide a historical overview of developments in aeronautical fatigue, in academic fatigue research with interests for the aeronautical sector, and finally

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present a subjective view of what remains to be done. This includes areas that need further research, future potential problem areas and also educational and societal changes that need consideration in order to maintain the high safety of flying that is now taken for granted.

Fatigue prior to ICAF

As presented by John Mann in his excellent review of 1958, Ref. (1), there were very many papers presented on fatigue research before the foundation of ICAF in 1951. Frequently, the German engineer Albert is credited with having been the first to carry out tests under repeated loads (already 1829) although his countryman Wöhler published the first systematic investigations on fatigue of materials (in 1858). It is remarkable that Wöhler not only observed a fatigue limit in many materials, but he also realised that the magnitude of the stress range, rather than the peak stress itself, was the driving force behind fatigue failures. He also demonstrated an influence of mean stress on limiting fatigue stress through his experiments. The paper by Mann is recommended to anybody interested in the early history of fatigue research. It may be of interest to note that the word fatigue appears to have been firstly coined by the French engineer Poncelet in 1839.

Fatigue in aircraft has been a problem since the very first days of aviation. A hollow propeller shaft (hollow to reduce weight) developed a fatigue crack and the first flight of the Wright Brothers' aircraft of 1903 was delayed until a solid spring steel shaft could be brought in from Dayton, Ohio, to the testing site in north Carolina, Ref. (2).

According to yet another remarkable review by Mann, presented as Plantema memorial lecture in Toulouse in 1983, Ref. (3), the crash of a Dornier Merkur aircraft in Germany in September 1927 was the first recorded instance of an in-flight structural fatigue failure. The crash, with loss of six lives, of the high-wing monoplane of the German airline Lufthansa near Schleiz resulted from the failure of a wing to strut fitting. Mann further mentions a Handley-Page W. 10, operated by Imperial Airways, crash into the English Channel on June 1929 as an early failure due to fatigue of an engine connecting rod. Seven lives were lost in that accident. Another 11 lives were lost because of fatigue failure of a wing strut in the crash, on 27 July 1934 near Tuttelingen in Germany, of a Swissair Curtiss Condor biplane. In Mann's native Australia, the first aircraft structural fatigue failure occurred in 1945. In the USA, the Air Force firstly became involved in structural fatigue testing in 1944 when it had a problem with the B-24 fleet's nose landing gear, (2).

Frederik J. Plantema and the foundation of ICAF

The foundation of ICAF is summarised in the first Plantema memorial lecture, held by J. Branger at the 5th ICAF symposium in Melbourne, Australia, 1967, Ref. (4). More details both on Dr. Plantema's course of life and his contribution to aeronautical fatigue, as well as on the formation and the evolution of ICAF can be found in Ref. (4). Here, a brief summary is given:

Dr. Frederik J. Plantema was born in Leeuwarden, The Netherlands, in October 1911. He graduated from the Technological University of Delft in 1932 where he remained as assistant to Prof. Biezeno until 1934 when he joined the National Aeronautical Research Institute (Nationaal Luchtvaartlaboratorium "NLL") in Amsterdam. He became leader of the Structures Department in 1945 and in 1950 he was appointed as head of the joint Structures and Materials Department which he continued to direct until his death in November 1966. Dr. Plantema performed research and published reports on a wide range of topics, including: Torsion of aircraft structures, allowable stresses in thin-walled cylinders, loads on tricycle landing gears, buckling of flat and slightly curved plates, loads on wings and tailplanes due to displacements of rudders or flaps, stress distribution in shells, rationalisation of gust load requirements, rolling manoeuvre loads, fatigue of structures and components, flexibility effects of aircraft during landing, fatigue tests on stiffened panels, strength testing of airplanes, buckling of struts, cumulative damage, fatigue tests on sandwich panels, airworthiness requirements for pitching manoeuvres, experimental investigations on runway waviness, and bending of orthotropic plates under transverse loading. His most well known contribution outside of ICAF, the book "Sandwich Constructions", was published by John Wiley and Sons in 1966 shortly before his death.

Dr. Plantema was a member of the Structures and Materials Panel of AGARD (NATO:s Advisory Group for Aeronautical Research and Development) and of the Fatigue Committee of the same Panel. He was further a member of the Netherlands Committee on Structural Strength Requirements for Civil Aircraft, and Associate Fellow of the Institute of the Aeronautical Sciences. In April of 1966 he was royally distinguished as Officer of the Order of "Oranje Nassau".

In 1949 Dr. Plantema published an analytical study entitled "Fatigue of Structures and Structural Components", Ref. (5). In this study he concluded that it would be necessary to consult laboratories in other countries to see whether his recommendations for fatigue research were in agreement with test programmes going on elsewhere, and that it could lead to a useful international exchange of results. Here the idea of ICAF was born.

The formal initiative to found ICAF was taken when Dr. Plantema wrote letters (dated May 11 1951 and signed by the director Koning) to the College of Aeronautics in Cranfield, UK, and the director of the FFA in Stockholm, Sweden. In these letters Plantema proposed a close co-operation between various institutes. The co-operation should consist of an exchange of reports and other information at the earliest possible date and the establishment of common research programmes in order to avoid unnecessary duplication. He further proposed periodic meetings of the people responsible for the fatigue work. These guide lines were agreed on during a preliminary meeting at the College of Aeronautics, Cranfield, September 14, 1951, attended by Dr. Plantema, Mr. E.J. van Beck (Fokker), Prof. W.S. Hemp (College of Aeronautics) and Mr. Bo Lundberg (FFA), Ref. (6). During the Cranfield meeting, which can be considered the birth of ICAF, it was also decided to approach representatives from Switzerland and Belgium about joining the co-operation. This was done before the first conference, held in September 25-26, 1952, in Amsterdam, Ref. (7).

After the foundation of ICAF the research activities on aeronautical fatigue at Dr. Plantema's department steadily increased. It started with fatigue tests on riveted, bolted and adhesively bonded joints and with research on cumulative damage. Later on subjects as crack propagation, notch and size effects and strength of fatigue-cracked panels were added. An extensive programme concerning full-scale tests with programme and random loading was completed one year before Plantema passed away in Nov. 1966. Another subject on which Plantema had been working for a long time was concerned with fatigue loads on aircraft. The integration of aircraft loads and the structural response to these loads, as well as his general interests in airworthiness problems, may explain why Dr. Plantema focused so much effort on fatigue studies.

The increasing fatigue activity at Plantema's department was paralleled by the growth of ICAF. This organisation started with five countries holding conferences from time to time. The number of countries increased to thirteen, but recently Belgium has ceased being an active member, such that the number of participating countries now is twelve. During the first conference in Amsterdam two main guidelines were adopted:

- An effective collaboration could only be obtained by regular personal contacts of the persons responsible for the work.
- An exchange of information on fatigue equipment, programmes and test results should be started as soon as possible.

During the early years of ICAF some serious aircraft accidents due to fatigue occurred and greatly stimulated research on fatigue testing all over the world. Our understanding of fatigue and fracture related problems have expanded tremendously over the 50 years since the foundation of ICAF and the contribution of ICAF to this understanding has been substantial by encouraging relevant research and by sharing the obtained information. This contribution has significantly helped to obtain the high safety levels of aircraft structures. Obvious examples of improvements include the development of design principles from static strength design to safe-life design after World War II. Then fail-safe concepts were developed in the late 1950ies following the Comet accidents, and then the present damage tolerance philosophy evolved in the early 1970ies following a few failures to be discussed below.

The early activities of ICAF from 1952 to 1957 concentrated on Conferences of two to three days, where national delegates from each member country presented summaries of significant research and where also technical lectures were held by delegates. The present format of biennial ICAF meetings started in 1959. The two-day conference, at which the national delegates present summaries of significant research performed in respective country during the last two years, is followed by the three-day symposium consisting of a single session presentation of technical papers. Since 1993 a poster session is also part of the symposium. Plantema memorial lectures have been given since 1967 when Jurg Branger presented the lecture on the foundation of ICAF referred to above. A summary of ICAF Conferences and Symposia is given in Table 1 and details of the Plantema memorial lectures are shown in Table 2.

1951 – 1959

The first few years of ICAF activities dealt partly with finding the correct format of ICAF but also with lots of technical activities. Early reports circulated between the member countries reflect interest in devising new and better fatigue testing machines, and in SN (stress versus number of cycles) fatigue testing of test specimens with different stress concentrations, joints, and stiffened panels. Interestingly enough, these early reports were frequently circulated in the original language version, not only when English, German or French were being used, but even in such esoteric languages as Swedish and Dutch. Today, as in most technical fields, the vast majority of ICAF documents are written in English.

Two sentences from the minutes of the second ICAF Conference in Stockholm, 1953 are of interest, Ref. (8). Firstly, the general director of FFA, Bo Lundberg, quoted during his opening lecture a statement by Dr. R. V. Rhode of N.A.C.A. in his paper "Some Observations on the Problem of Fatigue of Aeroplane Structures" presented at the fourth Anglo American Aeronautical Conference, Sept. 1953. The quotation is:

- In short, present inability to calculate the fatigue life is such that any number from such calculations and purporting to represent the fatigue life on an absolute basis is meaningless.

Secondly, in a paper entitled "Scatter in Fatigue Test" presented by Prof. W. Weibull he states, in order to clarify the notion of "rare events" that follow extreme value distributions, a classical example that is worth revitalising, namely:

- The number of men killed by a kick of a horse in the Prussian army.

The following years show an increased activity in studies of loads on aircraft structures, statistical aspects of fatigue, and cumulative fatigue damage, e.g. Ref. (9). At the fourth Conference in Zurich, Ref. (10), also France and Germany participated for the first time. Italy was represented for the first time at the fifth Conference held in 1957, ref. (11).

The Structures and Materials Panel of AGARD was formed in 1955 and various types of interaction between this body and ICAF were set up. A formal collaboration started on January1, 1958. Several early AGARD reports were distributed as ICAF Documents and during the 1959 Amsterdam meeting where for the first time both an ICAF Conference and a Symposium were organised, the latter was jointly organised by both organisations. Here, the baseline for all subsequent ICAF meetings was laid. The Conference, Ref. (12), which as before (and afterwards) was aimed only at persons invited by the National Delegates attracted some 30 persons, whereas the first Symposium, specifically dealing with full-scale fatigue testing of aircraft structures, Ref. (13), attracted a total of some 110 persons. In Amsterdam, Australia participated with a National Review for the first time and US citizens were attending the Symposium. The format for future ICAF meetings was shaped and the technical activities presented in the national reviews were already covering many of the topics that are dealt with today.

Two interesting papers being circulated to the different ICAF countries at this time period are Refs. (14) and (1). Cumulative damage was studied under block loading conditions in several ICAF countries, but Dr. Gassner's early work is worth special mentioning. In Ref. (14) he presented results from various programme tests where the continuous service loading was replaced by eight-stage distributions repeated until failure of the test specimens. Experiments were compared to the then (and even now) commonly used linear cumulative damage rule proposed by Palmgren in 1924, Ref. (15) and then independently by Miner in 1945, Ref. (16). Gassner concluded that this hypothesis was not valid and its employment was not recommended as its use in the majority of studied cases led to values on the unsafe side. The other selected paper, Ref. (1) by Mann, is the already mentioned early and excellent review of fatigue research from the early work by Albert and Wöhler up to the then recent work performed in the late 50-ies. Already at this time (1958) there were more than 5000 papers published on the various aspects of fatigue and the increase rate was some 300 – 400 papers per year.

On January 10, 1954, a Comet I aircraft, see Fig. 1, known as Yoke Peter, disintegrated in the air, at approximately nine thousand meters altitude, and crashed into the Mediterranean Sea off the island of Elba. Since the flight, BA 781 on the way from Singapore to London, having refuelled in Rome, crashed in daylight witnesses could report three explosions. The remains of the aircraft were 150 meters deep in the sea. Although this accident has been discussed in many publications, an interesting summary is given by Tom Swift, presented in the Plantema memorial lecture in Ottawa in 1987, Ref. (17). Design of the Comet started in 1946 and the first prototype flight was in 1949. BOAC started proving flights in April 1951 and Yoke Peter first flew on January 9, 1951, and was delivered top BOAC after having accumulated 339 flight hours. Yoke Peter was the first high-altitude jet-propelled passenger aircraft in the world to enter scheduled service. The aircraft was advancing the state of the art in a number of areas, e.g. it had a cabin pressure that was almost double that of any other pressurised transport aircraft in the world, (17). At the time of the accident, Yoke Peter had flown 3.680 hours and had experience 1.286 pressurised flights.

After the Elba crash, the Comets were removed from service on January 11, 1954. Modifications to the fleet were made to rectify more than sixty items that were possible reasons for the accident. Protections were added in the case of an engine explosion. New fuel pipes, fire and smoke detectors were added. Service was resumed on March 23, 1954. Only 16 days later, on April 8 1954, another Comet aircraft, known as Yoke Yoke, disintegrated in the air, at some ten thousand meters altitude, and crashed into the Sea in the area of Stromboli. At the time of the crash, this aircraft had flown 2.703 flight hours and had experienced 903 pressurised flights.

Following this accident, BOAC immediately grounded the fleet and on April 12, 1954, the certificate of airworthiness was withdrawn by the Air Registration Board. The ministry of supply instructed the director of the Royal Aircraft Establishment to complete an investigation of the cause of the accidents. It was decided to perform repeated loading of the pressure cabin and to carry out this under water to minimise damage in the event of a failure. A Comet aircraft removed from service, Yoke Uncle, was used for the testing. Failure of the test article occurred during the application of a proof cycle, which was applied at 33% higher loading approximately one time per thousand pressure cycles. Examination of the failure provided evidence of fatigue. The failure origin in the test article was at the aft lower corner of the forward escape hatch. Structural parts from Yoke Peter were recovered from the Sea, Figs 2 - 3, and also confirmed fatigue of the pressure cabin as primary cause of failure. The origin in this case was at the right hand aft corner of the rear automatic direction finding window on top of the aircraft, see Fig. 4.

Swift discusses the failures of the Comet aircraft in detail in Ref. (17). He considers local stress distributions and shows that the failures may be due to high stresses caused by induced secondary bending effects due to shell curvature. Swift also considers the Comet from a modern damage tolerance standpoint, looking at the lack of crack-stopper stringers and the residual strength capability of the aircraft.

At the time of the accidents, the fatigue community realised that both stress level and local geometry caused the early fatigue cracking in the Comet fuselage. Also, it was observed how relatively fast the crack propagation led to failure of the aircraft. The Comet failure led to significant studies of crack propagation. The ICAF community started on such research soon after the failures and efforts grew in the 60-ies.

1960 – 1969

During this decade ICAF activities continued much in the same way as had now become the standardised format under the Amsterdam meeting of 1959. Conferences and Symposia were arranged in Paris, Rome, Munich, Melbourne and Stockholm, although the format at the latter meeting deviated from the others, see Table 1. The USA participated for the first time in ICAF, with a National Review, at the Rome meeting in 1963. A single Canadian observer participated in the Munich meeting in 1965, although no Canadian returned for several more meetings. The Symposia held in 1959, 1961 and 1963 were all co-organised with the Structures and Materials Panel of AGARD. However, changes in the AGARD organisation in 1963, made it impossible to co-operate with a non-NATO body.

As mentioned above, Dr. Plantema died in 1966. In the period from Nov. 1966 until May 1967 Dr. J. Schijve acted as ICAF-secretary ad interim until Mr. J. Branger was elected successor to Dr. Plantema as new secretary of ICAF. As a special honour of the late Dr.

Plantema the organising committee of the Melbourne meeting in 1967 suggested that a special lecture be instituted as part of the bi-annual ICAF-meetings. This idea received support from all delegates and the first Plantema lecture on the foundation and growth of ICAF was presented by Mr. Branger at that meeting. The second Plantema lecture was held by Jaap Schijve in Stockholm, 1969. He spoke on various aspects on estimating fatigue lives and conducting fatigue tests, as well as presented an early summary of fatigue crack growth under flight simulation loading. The following Plantema memorial lectures are summarised in Table 2.

Although fatigue crack growth had now been studied for a long time, it was only in 1961 that Paul Paris and co-authors published the now classical paper that used fracture mechanics to correlate fatigue crack growth rates, Ref. (18). Although the original paper was rejected in three leading journals, whose reviewers felt that "it is not possible that an elastic parameter such as K can account for the self-evident plasticity effects in correlating fatigue crack growth rates", the concept spread reasonably fast. For example, already 1963 David Broek and Jaap Schijve evaluated this concept and other fatigue crack propagation theories in Ref. (19), and concluded that the theory of Paris gives some promise for further development.

The phenomenon of crack closure, to be discussed in more detail subsequently, was discovered by Elber already at the end of the 60-ies as part of his Ph.D. studies. Elber presented an early paper on the subject at the 1969 ICAF-meeting in Stockholm, Ref. (20). It may be of interest to note that Elber had similar problems as Paris and his colleagues mentioned above. His results were firstly considered erroneous and only after considerable problems did he manage to obtain his Ph.D. degree.

Another important contribution to the future of fatigue research was made by H. C. Johnson who developed the closed loop servohydraulic test system in the late1950-ies, Ref. (21). The development of this type of test equipment made it possible for the first time to test completely random, or spectrum, loading in addition to block programming that been used in the past. This type of testing spread across the ICAF nations during the 1960-ies. An early evaluation of different test systems is presented in Ref. (22). Analog computers were being introduced in materials systems in the second half of the 60-ies and at the end of this decade digital minicomputers were used to aid in data collection, Ref. (23).

The rainflow cycle counting method was also invented during this decade. Prof. Tatsuo Endo invented the method and published the three first papers in Japanese together with his co-workers, Refs. (24-26). The co-author of Ref. (26), M. Matsuishi, provided the first English description in his Masters Thesis of 1969. The technique rapidly spread around the world and rainflow cycle counting is today the accepted standard practice around the world. An excellent summary of the development and extension of the original rainflow algorithm is provided in the Conference Proceedings from a meeting dedicated to the memory of Prof. Endo, Ref. (27).

On Dec. 22, 1969, an in-flight structural wing failure occurred in the newly introduced U. S. Air Force F-111 aircraft. This particular accident, together with fatigue problems of the Lockheed C5-A aircraft, started a revolution in the development of fracture technology in the United States and led the Air Force, in July 1974, to mandate a new design philosophy known as Damage Tolerance, Ref. (28).

The F-111 had a design life of 4000 flights in the same number of flight hours. The accident mentioned occurred when aircraft #94 crashed while doing a low level bombing run, Ref. (29). This aircraft was delivered to the US Air Force on Nov. 26, 1968, and had only flown a total of 107 flight hours at the time of the accident. The crash was caused by separation of the left wing pivot fitting, which happened at a load level of only some 3.5 g's, which is less than half the design limit load factor. The direct cause of the failure was the growth of a surface flaw present already when the plane was delivered to the USAF, Fig. 5. The material was forged D6-ac steel, which has a fracture toughness sensitive to heat treatment. Further, it appears that design data were taken from thin sheet material not representative of the fitting.

Eventually, the F-111 passed all of its contractual static strength as well as fatigue tests up to four design life times. However, the sensitivity to small flaws made it necessary to run a proof test of each individual aircraft prior to operation. This test was devised to take place in a chamber at low temperatures, -40 degrees F, where the fracture toughness of the steel is substantially lower than at operating temperatures of the F-111. Together with lowered flight load factors this cold proof test has led to safe operating flight life of the aircraft.

The fatigue problems of the C5-A cargo transport included very early formation of cracks in the spanwise splices of the wing during the full scale fatigue test, scheduled to run up to four design lifetimes (1 design life = 30.000 flight hours, including 5.950 pressure cycles of the fuselage), Ref. (29). Cracking also occurred in adjacent planks and during the full-scale fatigue test multiple site damage developed, although this terminology was not used at that time. The original wing design of the C5-A did not meet with USAF requirements and a new design of the centre, inner and outer wing boxes was developed. The materials were changed and, in particular, much lower stress levels were achieved.

1970 – 1989

During these two decades ICAF grew into its present format. Israel and Japan became member nations and the ICAF activities settled firmly. Much effort went into fracture mechanics research and studies of fatigue crack propagation. The schematic variation of fatigue crack growth rates with stress intensity range, showing primary regimes of growth rate mechanisms, Fig. 6, became understood in more detail. A review of fatigue crack growth mechanisms and modelling issues is presented in Ref. (30). Here, we may only observe that the best correlation of crack growth data in different materials is obtained by normalising the stress intensity range with Young's modulus. Hence, for crack growth cases a difference of roughly three is what will result by changing aluminium for steel. In contrary to fatigue crack initiation (fatigue life of smooth specimens is proportional to material strength (or yield stress)) there is virtually no effect of strength on fatigue crack propagation. It should also be pointed out that there is no physical reason to the use of log-log co-ordinates to represent fatigue crack growth data. It is only a practical way as the growth rates occupy many order of magnitudes. Several persons have also tried to develop mystical models to correlate the constant and the exponent in Paris' law, but this is true nonsense to this writer.

The finite element model, originally developed by aeronautical engineers, was introduced in the teaching at technical universities around the world in the early 70-ies. The impact can only now be fully appreciated. With the explosion in computer development together with a stringent mathematical treatment of finite elements, we have today a situation nobody could envisage as those early times. Besides from stress analysis and determination of stress intensity factors, finite elements were being used to simulate the cyclic elastic-plastic growth of fatigue cracks, Refs. (31 – 38). Early studies of crack closure under plane stress conditions include Refs. (31 – 32). The first analysis of crack closure under plane strain conditions is presented in Ref. (33) and there still exist very few studies of the three-dimensional problem, Refs. (34 – 38).

Typical results of two-dimensional finite element plasticity-induced crack closure calculations are shown in Fig. 7. These results were obtained on CT-specimens (W=60 mm) of Al2024-T3, Refs. (33, 39), and show the normalised crack closure loads P_{op}/P_{max} as function of stress ratio $R=P_{min}/P_{max}$ for both plane stress and plane strain conditions. It is seen that the general trend of the plane strain results is similar to the plane stress results, only with the difference that less crack closure develops, due to more plastic constraint. Although not plotted in the figure, plasticity-induced crack closure also occurs at negative stress ratios. In this case, the influence of stress level becomes of more importance, Ref. (40).

As an example of three-dimensional closure results, the crack surface contact profiles on the crack plane during a certain load cycle are shown in Fig. 8, Ref. (38). The shaded regions show the area in contact at different applied stress levels. The stress ratio is R=0.1. As seen in the figure, the crack was closed over most of the newly created surface (from c_i to c) at minimum load. Increasing the applied stress caused the interior region to open while the surface region still was closed. At $P_{op}/P_{max} = 0.34$ the interior region was fully open and at $P_{op}/P_{max} > 0.50$ the entire crack plane was open. This sequence was reversed upon unloading. The results shown in Fig. 8 were obtained on a 25.4 mm middle-crack tension aluminium specimen. Similar analyses were also performed at other thicknesses down to 1.27 mm, Ref. (35). The variation of crack opening stresses through the specimen thickness is shown in Fig. 9 for a 9.56 mm thick specimen. Also shown in Fig. 9, as a dashed line, is the weighted average opening stress level. This value is obtained as the area under the solid line divided by the thickness, t, used in the analysis. Similar results were also obtained for other specimen thicknesses. For thicknesses 2t = 25.4 mm, 9.56 mm, 2.54 mm and 1.27 mm, the ratios of weighted average opening stresses to maximum stress were found to be 0.29, 0.32, 0.40 and 0.45, respectively. By using these values and also limiting values, representative for plane stress and plane strain conditions, Fig. 10 can be constructed, Ref. (35). This plot shows the variation in normalised weighted average crack opening stress as function of any normalises specimen thickness and should therefore be useful both for experimentalists as well as support for predictions of fatigue crack growth behaviour in three-dimensional structures.

Whilst finite element modelling of fatigue crack growth and closure is important to study basic issues and to validate other types of models, there is also a need for simpler, faster and more easy to use models for the analysis of spectrum fatigue crack growth behaviour. The currently most advanced and versatile models for such analysis are of modified Dugdale type, and of these the so-called strip-yield model originally proposed by Newman, Ref. (41) has won most acclaim. This model accounts for crack closure by leaving plastically material along the crack faces as the crack extends. The cleverness of the Dugdale model is that both crack-surface displacements and the plastic zone size are obtained by superposition of two elastic problems. The primary advantage of the Dugdale model is that such superposition holds even when the non-linear effect of crack closure is included. This is true because the crack closure effects take place only from residual plasticity in the line of the crack. This type of model was further extended to include the concept of weight functions in order to facilitate the analysis of any two-dimensional geometry, Ref. (40). Applications of the model are summarised in detail by Newman in his recent Plantema memorial lecture, Ref. (42).

The Dugdale model is strictly restricted to plane stress conditions. However, by introducing a constraint factor, α, to increase the effective flow stress, $\alpha\sigma_0$, it is possible to simulate plane strain conditions. Verification of the crack closure methodology developed in Ref. (40) is given in Fig. 11, Ref. (30), which shows normalised plasticity-induced crack closure loads for both plane stress and plane strain conditions for a CT-specimen loaded to $P_{max}/\sigma_0 W = 0.05$. In this figure are also included the finite element results from Ref. (39), already shown in Fig. 7, and also for comparison some experimental results obtained by Hudak et al., Ref. (43). Obviously, all results shown in Fig. 11 exhibit the same general trend. However, despite the close agreement between results by the finite element model (FEM) and the modified Dugdale model (MDM), shown in Fig. 11, it could still be argued that such comparison is merely fortuitous. Therefore, a detailed study between the three-dimensional FEM and MDM was performed in Ref. (37). In this study it was found that good predictions of crack opening stress levels in a centre cracked tension specimen were obtained when Irwin's plane strain factor, α = 1.73, is introduced in the MDM model. Furthermore, crack surface displacements, See Figs 12 and 13, and residual stress distributions, see Figs 14 and 15, in the specimen mid-plane, are also in reasonable agreement between FEM and MDM. Clearly, the modified Dugdale model is not used to derive detailed stress distributions at crack tips, but the reasonably good estimates of crack surface displacements and normal stress distributions, on an average means, lend support to the capability of the model for crack closure predictions. It may be of interest to note that the MDM analysis performed on a VAX 780 required about a factor of one thousand less CPU time than the FEM analysis performed on a CYBER-205 super-computer, Ref. (37).

So far, plasticity-induced closure only has been considered. At the end of the 70-ies a seemingly controversial observation was being made. Experimental measurements of crack closure levels indicated that a sharp increase occurred as the near-threshold regime was approached. Yet, at very low crack growth rates plane strain conditions prevail whereas at high crack growth rates a state of plane stress is approached. If we now again consider the results shown in Fig. 7, it is obvious that the computed levels of crack closure are at odds against such experimental data. After a rather long time of controversy it turned out that both results could in fact be correct and that the missing factor was the prevalence of other types of crack closure mechanisms besides from the plasticity-induced one. It is now recognised that that several additional closure mechanisms may be operating at near-threshold crack growth rates. Here, it suffices to mention the two most important ones, namely oxide-induced and roughness-induced crack closure.

As suggested by Ritchie and co-workers, Refs. (44,45) and by Stewart, Ref. (46), corrosion debris formed on freshly exposed surfaces at the crack tip may wedge-close the crack at stress intensities well above K_{min} when the oxide deposits reach a thickness comparable to the crack tip opening displacement. This effect, oxide-induced crack closure, is strongest at low positive stress ratios, where the effects of plasticity-induced closure are largest, and at low stress intensities where oxidation may be enhanced by fretting due to the small crack tip opening displacements involved. As discussed in detail in Refs. (44 – 47) this mechanism has successfully been used to explain many observations on the role of environment in influencing near-threshold fatigue behaviour.

A more general source of closure arises from an irregular or rough surface morphology in conjunction with shear displacements. This results in enhanced crack closure since the crack may be wedge-closed at discrete contact points along the crack surfaces as firstly reported by Walker and Beevers, Ref. (48) and subsequently substantiated by others, Refs.

(49 – 51). This closure mechanism again is promoted at low positive stress ratios, for the same reason as above, and at near-threshold levels, particularly where crack advance is strongly crystallographic, such as in coherent particle hardened (planar slip) systems.

The prevalence of additional closure mechanisms to plasticity-induced closure to a large extent explains the existence of a fatigue threshold stress intensity factor. As is well known the threshold is largest at small positive stress ratios, where crack closure is dominant, and decreases with increased stress ratio. It is still not fully understood exactly how the very load shedding procedure adopted to reach threshold conditions influence the obtained results. However, by applying one single compressive overload after the threshold had been approached, it was found that the crack started to grow a certain distance again, before it became dormant. The amount of crack extension is dependent on microstructure and stress ratio and can be explained in terms of the build up of the closure destroyed by the compressive overload, Ref. (52). The reinitiation of growth was found to be associated with a measured reduction in crack closure. This was attributed principally to a smaller contribution from roughness-induced closure, arising from the compacting and cracking of fracture surface asperities close behind the crack tip.

From the discussions above it becomes clear that the application of any type of continuums mechanics based model for fatigue crack propagation will have certain limitations. At near-threshold conditions the growth rates are so low that the average crack advance per cycle is less than one lattice spacing. Hence, crack growth may be considered an irregular growth process where stochastic effects enter the picture. Obvious modelling problems include the growth of very small defects, where the crack is short compared to crack length, the plastic zone size, or physical dimensions of the test specimen. To a surprisingly large degree, the modified Dugdale type of models mentioned above have been used to successfully model the growth of short cracks, Refs. (42, 53). However, various microstructural effects, such as grain boundary arrest, Ref. (54), and closure build up during short crack extension in different microstructures, Refs. (55) cannot yet be described by fracture mechanics based models.

Combined modelling and experimental measurements of crack closure have been performed to explain the initiation and growth of cracks under cyclic compressive loads, Refs. (56 – 57), to study mechanisms of fatigue crack growth under variable-amplitude loading, Ref. (58), and to study fatigue threshold behaviour as well as short fatigue cracks as already mentioned. Yet, few critical studies of experimental measurements of crack closure have been reported in the pertinent literature. A variety of experimental techniques for determining closure loads levels have bee devised. These include electric potential techniques, ultrasonics and interferometric methods, but by far most common are compliance techniques, e.g. Refs. (20, 39, 43 – 52, 55 – 58). As shown in Fig. 11, experimental data may correlate rather well to numerical predictions of crack closure. Yet, this apparent similarity is almost certainly nothing but coincidence and in reality due to misinterpretation of data. In fact, it seems quite impossible that compliance techniques, using for example back face strain, could monitor the same closure levels as predicted by finite element modelling if the monitored closure mechanism was restricted to plasticity alone. It rather appears as if good correlation can only be obtained over certain crack growth rates where other mechanisms also prevail and the resulting compliance signal is large enough to be monitored accurately. On the other hand, a comparison with numerical modelling is then erroneous as one compares different closure mechanisms.

In order to study this problem, a plane stress elastic –plastic finite element analysis of a CT-specimen loaded at $R = 0$ was performed to critically appraise various common compliance techniques for closure measurements, Ref. (59). Results from this study showed that closure values obtained from compliance curves are always lower than values obtained from the first nodal contact point in FEM-analysis unless other types of closure exist besides plasticity-induced closure. It was also shown that reduced compliance curves are much more sensitive than direct compliance curves and should always be used for determination of closure loads. Further, commonly used back face strain and clip gauge displacement techniques are too insensitive to be used. Instead, compliance should be evaluated very close to the crack tip, but outside the plastic zone. Also, closure levels ought never to be given without presentation of actual graphs used for evaluation. Finally, it was concluded that much of the data presented in the archival literature is likely to be erroneous and the agreement between experiments and numerical modelling may only be due to misinterpretation of the different closure mechanisms involved.

Aircraft failures due to fatigue continued to happen in the 70-ies and 80-ies, and just like the earlier mentioned failures, we have learned important lessons from these failures. In Sweden, early production versions (< #28) of the AJ37 Viggen fighter aircraft (strike version) had some serious wing failures in the mid 70:ies. The design safe service life of those aircraft was 2,000 flight hours.

In July 1974 one aircraft (37.011) crashed after only 152 flight hours due to a main wing failure in the left wing. The investigation carried out in the aftermath looked for several causes, e.g. static overload, flutter etc. but no definite cause of failure was found. Flying continued but after two other wing failures (aircraft 37.005 after 286 flight hours and aircraft 37.014 after 275 flight hours), within one week, in October 1975 the aircraft type was grounded. The ensuing investigation revealed fatigue as the cause of the failures. Fatigue cracks were found in the lower flange of the main wing spar. The critical crack size was quite small, in the order of 1-2 mm. The main reason was found to be the high gross stress level, about 300 MPa at limit load. A bad edge distance of a bolt hole in the lower flange made the situation worse. Fig. 16 shows a summary of the main wing spar failure in AJ37 Viggen. Extensive fatigue testing was carried out and a new wing spar design was developed. The new design resulted in much lower stresses and a new alloy was also introduced. Flying with the aircraft type was resumed in March 1976. The later fighter version, JA37 Viggen, was furnished with another wing spar design. From this moment, all following Saab aircraft, both civil and military, have been designed for damage tolerance.

On May 14, 1977, a Boeing 707 operating as a freighter came in for landing at its final destination, Lusaka airport, Zambia, on its way from London Heathrow – Athens – Nairobi. At 09.28 the co-pilot reported that the airfield was in sight. Lusaka then cleared the aircraft to descend to 6000ft (2221ft above touchdown elevation) and moments later a clearance was given to make a visual approach for Runway 10. At 09.32 flaps were selected to 50deg. Suddenly, at 09.33, the complete right hand horizontal stabiliser and elevator assembly were seen to separate in flight. The aircraft pitched rapidly nose down and dived vertically into the ground from a height of about 800ft and caught fire. The main wreckage was located 3660m from the runway threshold, see Fig. 17. The design of the spar was meant to be fail-safe as the spar had an extra mass of material in the middle of the web that normally does not carry any load. In the case of failure of the upper or lower spar cap, however, this material becomes a smaller load-carrying member. The failure still occurred despite this apparent fail-safe design, due to the inability to detect the crack before it became critical, Ref. (60), see Fig. 18.

Another serious aircraft failure occurred on August 12, 1985, as flight JA8119, a Boeing 747 SR-100 of Japan Air Lines on its way from Tokyo to Osaka, crashed in the mountains near Ueno Village. The aircraft had accumulated 25.030 flight hours and 18.835 landing cycles. Of 524 passengers only four survived. Deterioration of flight characteristics and loss of primary flight controls due to rupture of the aft pressure bulkhead with subsequent ruptures of the tail, vertical fin and hydraulic flight control systems was the direct cause of the accident, see Fig. 19, Ref. (61).

Prior to the accident, this aircraft aft fuselage was seriously damaged during a bad landing in Osaka Airport on June 2, 1978. The aircraft was then repaired at Tokyo International Airport by a Boeing Company repair team from June 17 to July 11, 1978. Both Japan Air Lines and the Boeing Company had agreed to what is still considered an appropriate repair plan. During the repair, in which the lower half of the aft pressure bulkhead was to be replaced, a splice plate had to be inserted between the webs of the upper and the lower halves of the bulkhead. However, the actual repair was not carried out in the correct way, see Fig. 20, Ref. (61).

Due to the incorrect repair, the strength of the L18 splice plate with one-row rivets is estimated to be less than 70% of the intended two-row rivets splice plate design, Ref. (61). Scanning electron microscopic investigations of the failed parts showed striations typical of fatigue loading due to pressure difference to correspond well with the number of flights following the incorrect repair. Hence, the Japanese Aircraft Accident Investigation Commission was led to the following probable failure scenario.

The flight took off from Tokyo-Haneda at 18.12. At 18.24, while climbing up to about 7300 m altitude, all of a sudden, fracture started in bay 2 of the aft pressure bulkhead and subsequently total fracture of the L18 splice occurred as shown in Fig. 21. Within a few seconds, portions of the aft pressure bulkhead were blown off and the passenger cabin pressure increased the inner pressure in the tail section. This resulted in loss or fracture of the APU (auxiliary power unit), most of the vertical fin, rudder and four hydraulic pressure line systems, as indicated in Fig. 19. The reason for the aft pressure bulkhead rupture was that its strength was reduced by the fatigue cracks propagating in the spliced portion of the bulkhead's webs. The initiation and propagation of the fatigue cracks are attributable to the improper repairs of the bulkhead, and since the fatigue cracks were not found in later maintenance inspections, this contributed to the accident.

As final example of fatigue failures of aircraft, one of the most spectacular incidents ever is discussed. On April 28, 1988, Aloha Airlines flight 243, a Boeing 737-200, took off from Hilo for a flight to Honolulu and climbed to cruise altitude. When the aircraft levelled off approximately 18ft from the cabin skin and structure aft of the cabin entrance door separated from the aircraft, see Fig. 22. Of a total of 95 people there was only one fatality, one of the cabin attendants was sucked out in the decompression. Both cockpit crew members immediately initiated an emergency descent of 4100ft/min. A successful emergency landing was made at Maui Airport with a speed of some 40kts above normal landing speed.

In the aircraft accident report, Ref. (62), it is stated that the probable cause of this accident was the failure of the Aloha Airlines maintenance program to detect the presence of significant disbonding and fatigue damage, which ultimately led to failure of the lap joint at S-10L and the separation of the fuselage upper lobe. Contributing to the accident were the failure of Aloha Airlines management to supervise properly its maintenance force as well as the failure of the Federal Aviation Authorities (FAA) to evaluate properly the Aloha Airlines maintenance program and to assess the airline's inspection and quality control deficiencies. Also contributing to the accident were the failure of the FAA to require Airworthiness Directive 87-21-08 inspection of all the lap joints proposed by Boeing Alert Service Bulletin SB 737-53A1039 and the lack of a complete terminating action (neither generated by Boeing nor required by the FAA) after the discovery of early production difficulties in the 737 cold bond lap joint, which resulted in low bond durability, corrosion and premature fatigue cracking.

This particular failure led to awareness of the so-called ageing aircraft problem, which was the major focus for airframe fatigue related research during the last decade. Very many conferences have been devoted singularly to this topic, and sessions on wide spread fatigue damage have been included in virtually all large fatigue conferences. At ICAF meetings many relevant papers have been presented that includes ageing aircraft aspects. In Refs. (63, 64) Douglas Aircraft Company and Boeing Commercial Airplanes, respectively, presented their early ageing aircraft programs. This included overviews of the ageing aircraft fleets, on-site visits with operators, teardown inspections, structural repairs etc. Of particular interest were the findings of the relatively poor conditions of a limited number of inspected aircraft. Ref. (64) mentions a number of such cases and concludes that particularly corrosion damage was found to an extent not expected on well-maintained aircraft. Additional findings included a number of aircraft with improper modifications or repairs, such as the excessive use of blind rivets and improper rivet patterns in primary structure, improper use of screws to attach repairs in primary structure, applying sealant or paint over existing corrosion, and creating knife edges at fastener holes by using countersunk rivets in thin skins Lack in continuity in maintenance was found particularly prevalent when it came to leased aircraft, a situation that might become dangerous with the steady increase of leased aircraft around the world.

In his Plantema memorial lecture, Ref. (65), Ulf Goranson covers general state of the art aspects of damage tolerance including ageing jet transport problems. This paper contains a good summary of the various ageing aircraft tasks that the FAA, aircraft manufacturers, and airline operators have been focussing on ever after the Aloha Airlines accident in 1988.

1990 – present

ICAF activities during the last decade have continued in the format described earlier. The same 13 countries as mentioned above, i.e. The Netherlands, The UK, Sweden, Belgium, Switzerland, Germany, France, Italy, Australia, The USA, Canada, Israel and Japan, have continued with exchanging technical reports and arranging the biannual ICAF Conferences and Symposia. However, Belgium participated with a national review for the last time in 1993, and has since not taken part in ICAF activities. On the other hand, in this meeting, Toulouse 2001, Finland intends to distribute a national review for the first time.

Although the papers presented at ICAF meetings have dealt with virtually all possible aspects of aeronautical fatigue, it is fair to state that focus has been on damage tolerance and ageing aircraft aspects. Yet, it deserves to be mentioned that although damage tolerance now is the mandated design philosophy for all civil aircraft, there appears to be only the USAF and the Swedish Air Force that also adopts this philosophy for military aircraft. In an AGARD meeting devoted to fatigue damage and crack growth prediction techniques, Ref. (66), it became clear that certain countries still rely on manufacturing quality control rather than damage tolerance to ensure structural safety of military aircraft structural components. This is somewhat surprising considering the possibilities for flaws either not to be found in the quality control process, or to appear at a later stage due to mishandling or fatigue loading. The examples of the F-111 and the C5-A mentioned above are still valid despite any attempts at improved quality control procedures. Another example is shown in Fig. 23. A single edge notch specimen of aluminium was found to have an accidental damage at the edge of the notch root. The specimen was tested in spectrum loading and the resulting fatigue life was found to be an order of magnitude less than for undamaged test specimens. This author believes that in the future, all military aircraft should be designed based on the damage tolerance philosophy, similar to the situation for civil aircraft today. The competition to sell fighters on the export market will probably force manufacturers into this step even if technical, financial, or other aspects keep them from doing so now.

Below, the author will present a short overview of the various ingredients contained in aircraft design, analysis and verification of damage tolerant structures. The aim is not to delve into technical details, but to indicate what the present state of the art knowledge is, and what limitations or problems still remain. However, a few chosen areas will be discussed in more detail. For this discussion, the outline of service life management, shown in Fig. 24 is used, Ref. (67).

Regulations and specifications are mandatory for civil aircraft where FAA and JAA (Joint Aviation Authorities) are the certifying agencies of USA and Europe, respectively, e.g. Ref. (68-70). For military aircraft the situation is different as each country may certify their own aircraft rather independently. However, the military specifications of US Air Force, e.g. Ref. (71-75) are well known and these or modifications thereof are used in other countries as well, e.g. Ref. (76).

Loads

The development of load spectra for use in fatigue and damage tolerance analyses are based on the expected future usage of the structure considered. Design parameters in the mission analysis originate from estimated threats and expected usage and are expressed as a sequence of flights and ground conditions, see Fig 25. The conditions are defined by flight mechanics parameters such as load factor, roll rate, speed, control surface deflections, thrust, fuel burn, weapons etc. Each set of flight parameters defines a certain flight condition. Determining the external loads for those conditions (manoeuvres) at different aircraft configurations requires analysis of e.g. structural dynamic response, aerodynamic pressure distributions at different speeds, angles of attack etc, see Fig.26. Ground loads analysis includes such events as landing impact, taxiing, braking, turning etc. The load analysis makes use of techniques like the finite element method to predict dynamic transient response, e.g. landing, computational fluid dynamics for prediction of aerodynamic pressure fields and six degree of freedom flight mechanics model with control system logics to predict in-flight manoeuvres. Numerical predictions are supported by wind-tunnel tests of models.

For fighter aircraft techniques to handle the above are reasonably well developed although problems still exist regarding unstationary aeroelasticity and buffet loads. In order to achieve high manoeuvrability modern fighters are capable to fly under conditions of separated flow. The aircraft structure will, under such conditions, be subjected to random aerodynamic loads arising from pressure fluctuations due to flow separations and/or impact of vortices on the structure. In the case of the F/A-18, vertical tail buffet loads are of comparable order to manoeuvre loads. These buffet loads arise due to turbulent flow resulting from bursting of the lead edge extension vortices. In order to study the effect of the combined effect of this type of loading, on fatigue and damage tolerance properties of the structure, a complex test set up has been devised in Australia, Ref. (77). The test is a full-scale fatigue test of the F/A-18 aft fuselage and empennage using airsprings and hydraulic shakers to apply combined manoeuvre and dynamic loads to the four tail surfaces of the test article.

The probability of fatigue failure (crack initiation) should not exceed 10^{-4} in one service life. To handle this for an unmonitored safe life application, safety factors are used. Assuming that the average fatigue strength data (SN-curve) follow the log-normal distribution, a probability of less than 10^{-4} can be achieved by reducing the fatigue strength with three standard deviations (-3σ leads to p approximately equal to 10^{-3}) and increase the load spectrum with two standard deviations ($+2\sigma$ leads to p approximately equal to $2*10^{-2}$). The values of the safety factors f_n depend on the standard deviation , i.e. $f_n = 10^{n*\sigma}$ where n is the number of standard deviations. A typical value of fn = 1.5 for the allowance of uncertainties in service loading is given in Ref. (76) (the reference gives the same factor for derivation of crack growth curves for damage tolerance analysis). In Ref. (78) load factor data from a group of 145 Viggen aircraft are used to estimate the uncertainties in loading. The data were collected over a period of five years and represent 75.741 flight hours. Using two standard deviations (one $\sigma = 0.072$) the load scatter factor became 1.39. However, once the data for all the 145 aircraft is split up into four versions of the Viggen aircraft, the data no longer fall on one single curve. Fig. 27 shows the normal distribution of the relative crack severity index, CSI, for these four different versions, Ref. (78), i.e. the AJ36 strike-, SH37 sea reconnaissance-, SF37 photo reconnaissance, and SK37 trainer-versions. All of these versions were designed for one fatigue spectrum although they differ in hardware. The JA37 fighter version is not included since it is designed for a different spectrum. As can be seen in Fig. 27, the SH37 version has the mildest usage due to often long approaching distances and moderate manoeuvring. The most severe usage is found for the SF37 version since it manoeuvres rather aggressively at low altitudes. The AJ37 version is found in between these two versions as is the SK37 version. However, the latter differs from the other versions regarding the slope of the curve, which is related to fairly strict training programmes.

The differences shown in Fig. 27 are interesting to observe when one considers the new fourth generation of military aircraft. This generation will certainly exhibit larger variability in loading for individual aircraft than earlier generations did. Reasons for such variability and, hence, deviation from the fatigue conditions it was once designed for include the capability for multi-role missions. This means that the aircraft not is optimised for one single mission but can be assigned to any one of its mission types without any changes to the aircraft software or internal hardware. The aircraft will also be subjected to a swing-role of combined air defence and surface strike operations. Some aircraft may be used exclusively for air patrol whereas others mainly will operate in strike mission training or a mix of these and other mission types. Hence, although all individual aircraft are designed for one single fatigue spectrum, no individual aircraft will follow this exactly and deviations may be large. The new generation of aircraft also has digital flight control systems that may be used to improve aircraft performance even after the aerodynamic and structural designs are complete. This flexibility will be used whenever tactical advantages can be gained. For the new Swedish Gripen aircraft, examples of how the flight control system can be used to reduce loads is given in Ref. (78). For example, load alleviation of the bending moment at the root of the wing can be achieved by an elevon split mode that is operable when load factors are getting closer than 2g from the limit load factor. Load alleviation is then achieved by moving the aerodynamic centre of the wing inwards. Gust load alleviation for aiming purposes and differential control surface deflections for handling the aircraft at higher angles of attack than it was designed for, are other examples of how the flight control system may be used. In all cases, the resulting load spectra will change from the initial design assumptions. Other reasons for uncertainties in future loading conditions include the new safety political situation of the world. In Sweden this means that the old invasion threat is no longer considered and that the defence forces instead may be involved in international peace keeping operations. Both the tactics and the way of using the aircraft, including future armaments, may then change substantially from the design considerations. Hence, the load monitoring systems used in the past are no longer adequate for the new generation of military aircraft. Individual aircraft tracking will be used both to maintain flight safety requirements and in order to aid cost effective maintenance procedures. Fig. 28 shows the load monitoring system for the Gripen aircraft. Although traditional flight parameters are still measured, there are also direct measurements of loads using calibrated strain gauges. Service experience from strain gauge measurements is good until now. No mechanical failures have occurred in the current fleet of some 90 aircraft with a total of over 10.000 flight hours. However, calibration of strain gauge bridges is done every 200 hours of flying. More information on system details as well as a discussion of certain primary items that need to be designed based on limit spectra is provided in Ref. (78).

For civil aircraft, systematic monitoring of service fatigue loading or mission usage has found very limited application until now. However, with ageing fleet of aircraft that are also being leased or resold to new operators it seems unwise not to improve this situation. Already in 1989, Ben de Jonge in his Plantema memorial lecture, Ref. (79), suggested that systematic monitoring of at least a few relatively simple parameters ought to be done in order to extend the service life of ageing aircraft under safer conditions. This author agrees and feels that airworthiness authorities should consider making such practices mandatory.

Stress Analysis and Fracture Mechanics

Models used for assessment of fatigue life and damage tolerance of fighter aircraft vary from rather simple to highly complex depending on factors such as; if the structural component is primary or secondary, local stress levels, inspectability etc. Typically, simple models are used initially and more sophisticated analysis is resorted to in complex geometries and when safety margins are lower. This means that conservative estimates of stress gradients may be used together with weight functions to derive stress intensity factor solutions as a first estimate. Weight function methodology is now available both for plane problems, Ref. (80), and three-dimensional problems, Ref. (81). When necessary the solution complexity is refined until the accuracy is sufficient. It should be realised that stress analysis for large structural details is performed on different levels. A large structural model is developed to handle load distributions for different load cases, see Fig. 29. Such a model is, however, not detailed enough to provide local measures as stresses and strains. Detailed modelling is then performed on areas of interest in order to generate correct stress distributions, stress intensity factors etc, see Fig. 30. Finite element modelling is virtually always done of the un-cracked structure to derive stress distributions. The most recent developments of stress analyses of both cracked and un-cracked structures have been in the area of so-called p-version finite elements, Refs. (82-87). By such techniques, it is now possible to derive numerically exact solutions of stresses and stress intensity factors for any arbitrary elastic three dimensional problem. Even more complicated problems, such as vertex intensity factors, e.g. where a curved crack front or delamination intersects a free surface, can be solved numerically accurately. Stress analysis of problems involving cyclic plasticity or friction is feasible although the physical understanding of the actual phenomena may be more limiting than the numerical solutions of the formulated equations.

Fatigue Crack Growth and Residual Strength

Newman provided a recent review in his Plantema memorial lecture, Ref. (42). What still needs consideration regarding the crack closure modelling presented there include the following aspects. As discussed previously, fatigue crack growth at low growth rates, see Fig. 6, actually takes place as local increments less than the lattice spacing of the material. Hence, crack growth must be occurring as local stochastic processes even if continuum based models may frequently predict average observed behaviour rather well. In order to simulate the three-dimensional aspects of the crack growth process, including the transition from essentially plane strain to gradually more plane stress as the crack extends, a constraint factor is introduced. This factor is sometimes used as a fitting parameter and more studies seem motivated to establish a strict choice of constraint as function of applied load, crack length, sheet thickness, and possibly type of load spectra for various materials. Especially under spectrum loading this is needed in order to avoid having to resort to a variable constraint factor as proposed in Ref. (88). There it was shown that under TWIST spectrum loading, an initial increase in crack growth rate was followed by a decrease and then once again an increase until failure. This effect led to the unusual phenomenon that at certain crack lengths the cracks actually grew faster at lower applied stress levels. The reason for this anomalous behaviour appears to be a plane stress - plane strain - plane stress transition with resulting differences in plastic zone sizes and crack closure levels.

Another basic problem concerns the usage of constant amplitude data, as basis for predictions of crack growth behaviour under spectrum loading, as is common to all frequently used models of fatigue crack growth. It appears that this seemingly straightforward task merits more attention than what is normally the case. If we consider any experimental data set of fatigue crack growth, several things emerge. Firstly, the stress ratio effect has to be accounted for in spectrum modelling by interpolation or by invoking a closure model to represent the data, i.e. as function of the effective stress intensity range. Secondly, there is always a fair amount of scatter in fcg-rates as discussed in more detail elsewhere, e.g. Ref. (89). Suffice it to state here that intra-laboratory scatter in crack growth rate (scatter defined as the ratio between maximum and minimum crack growth rate for constant ΔK-values) frequently may be in the order of three to four and that inter-laboratory scatter may exceed five. There is also a problem to define mean data as no statistical distribution fits typical fcg-data very well. Thirdly, the handling of data in the near-threshold regime may influence predicted results significantly. In a recent AGARD round robin prediction of crack growth in turbine discs subjected to the TURBISTAN sequence, it turned out that performing a crack growth prediction without account for retardation but only handling the fcg-data either as a linear Paris-equation, disregarding near-threshold data, or as a complete curve including such data, yielded approximately a factor five difference in fatigue crack growth life, Ref. (90). This number is of the same order as the differences obtained between all participants through the use of different numerical crack growth prediction models. Clearly, it follows that it may be very difficult to assess crack growth prediction results presented in the literature, as it is sometimes not clear what effects result from the model used and what is due to materials data handling. Although a threshold does appear even under load shedding during spectrum loading, e.g. Ref. (53), it is now clear that fatigue thresholds are partly artefacts due to the actual test procedures being used. The correct physical basis for using constant amplitude data as input for spectrum predictions does not seem to be clear. The issue appears to merit much more fundamental research. The present author is aware of predictions of crack growth in real aircraft structures where the use of threshold data without retardation gave excellent results compared to experiments. On the other hand, so did predictions based on a straight line Paris-relationship together with retardation modelling. Hence, the best procedure is yet to be revealed.

Virtually all fatigue crack growth modelling now relies on the range of the stress intensity factor although this necessarily involves certain limitations. Alternatives have been proposed though. To allow for large plastic deformations the range of the cyclic J-integral has been used. This, however, seems doubtful as the J-integral is derived based on deformation based plasticity and, hence, does not allow for the unloading that necessarily takes place as the crack extends. To model the growth of small flaws various micro mechanical models have been proposed in the literature. These all have in common that they fit the performed experiments, but that they are dependent on planar geometry and therefore not easily adapted to predictions for arbitrary geometries and loadings. The latter point is the essence of fracture mechanics based models and considering the vast amount of money spent on generating experimental data now existing it is hard to see that any new model is likely to replace the current methodology, at least not in the near future. Within the solid mechanics community many models based on damage mechanics have appeared through the years. These, however, seem more of an intellectual exercise than of practical interest, not least because of the vague notion of the concept of damage. Fracture mechanics has a vast superiority in the use of a crack that can actually be physically observed.

Residual strength of aircraft structures is typically handled by using R-curves. An interesting alternative, however, is the crack tip opening angle (CTOA) fracture criterion summarised in Ref. (42). The cleverness of the latter concept, which is considered a local approach, is that a combination of analytical methodology and experimental work is used to predict the failure of any structural component. In essence, the computer code is calibrated for laboratory specimens and then the same model is being applied to the real structure. Finally, it ought to be remembered that the possibility of plastic collapse always must be considered apart from fracture predictions.

Structural Testing

Here, we will discuss the testing carried out to substantiate the analytical fatigue and damage tolerance methodologies used. Such tests include simple coupon tests to study influence of load spectrum truncation etc., component tests to verify numerical predictions and to reveal any unexpected problems, and full-scale fatigue and damage tolerance testing of an entire aircraft. The overall test programme for a new fighter aircraft typically takes more than 10 years to complete. The comments below refer to the testing of the JAS 39 Gripen aircraft. More details on the damage tolerance verification policy for the aircraft and specific details like manufacturing of artificial flaws etc are given in Ref. (67). Component tests for verification of both fatigue and damage tolerance are first subjected to two service lives of fatigue testing, then artificial flaws are introduced, and the test is subjected to two more service lives of fatigue testing. Finally, a residual strength test is carried out with the purpose to verify a load capacity in excess of 120% LL. The pure damage tolerance verification tests are performed with artificial flaws introduced from the very beginning and are subjected to two lifetimes of fatigue testing followed by the residual strength test.

The damage tolerance analysis creates the necessary conditions for structural integrity. This integrity also needs to be demonstrated in structural testing according to the sizing approach shown in Fig. 24. Besides from testing for obtaining data for predictions, three main levels of testing are detail testing, major component testing and final full scale verification testing. Detail testing is mainly performed early in the sizing work. It is used to verify detail design of vital structural members and to qualify the application of prediction methods to typical structural configurations. Major component testing is done for early fatigue and damage tolerance verification. The key point in these tests is that a critical part is tested while properly installed in its nearest boundary structure. This test will spot fatigue critical areas and demonstrate the stable growth of those natural cracks that may initiate and of any artificially made cracks. An example is the testing of attachment of wing to fuselage. Damage tolerance testing of large components involve very many initial flaws, for the rear fuselage tested together with fin and rudder more than 100 flaws were introduced. The final verification of the fatigue and damage tolerance performance is made with a complete airframe tested for several service lives, see Fig. 31. It is also interesting to observe that detailed structural fatigue and damage tolerance testing not only is performed on structural parts, but also on mechanical systems components, see Fig. 32.

The outcome of the completed analytical and experimental damage tolerance work will yield information on what inspection programmes are necessary to perform during the life of the aircraft. For financial reasons, it is attempted to keep down inspections as far as possible and only critical areas will be regularly monitored. After a life extension programme, the inspection programme may need to be changed.

Mechanical Joints

In 1985 Schijve presented a comprehensive state-of-the-art review on flight-simulation fatigue tests where he systematically discusses subjects such as truncation/omission level, sequence effects, design stress level and material, Ref. (91). Schijve concluded that it would be problematic to assign some "effective stress concentration factor" to a notched element subjected to fatigue spectrum loading, which would be useful in the design process. In particular for joints Schijve found this approach to be unrealistic. He mentioned several reasons, the variants of joints are large, and load transmission in a joint is complex and often depends on frictional forces that also often change during the fatigue life due to fretting of interfaces. Further, the secondary bending may be high in asymmetric joints and hence, fatigue cracks will grow as part through cracks for the major part of the fatigue life whereas the growth pattern for other types of notched elements, even for other joints with less secondary bending, may be quite different. The author's laboratory started a fairly comprehensive programme to study some fundamental aspects of fatigue behaviour in mechanical joints in 1993 and this is still on-going. Results have been presented in a number of reports and articles, e.g. Refs. (92-95).

To evaluate and compare the fatigue performance of different fastener systems, within various test programmes run throughout the years, several test specimens have been designed with the scope of simulating the load conditions of a certain structural design. Here, the load transfer (LT) is defined as the percentage of the applied load, which is transferred from one plate to the other by means of the fasteners and friction between the plates. Joints may be divided into four groups according to the amount of LT: no load transfer, low load transfer ($0 < LT < 10\%$), medium load transfer ($10\% < LT < 40\%$), and high load transfer ($40\% < LT < 100\%$). The secondary bending (SB) is defined as the ratio of bending strain to axial strain in the section of interest. This leads to SB being a function also of the applied load level itself. The no-SB joints are of the double shear type, while the high-SB joints have a single shear configuration.

The Double Reverse Dogbone (DRD) joint is a *low-LT, medium-SB joint*, representative of the lower wing skins attached to spars, see Figure 33. The LT is dependent on the fastener type and fit, i.e. when a rigid fastener is used in combination with an interference fit installation, the LT has been measured to approximately 10%. If, however, a clearance fit fastener is used, the LT may disappear. Secondary bending measurements have been in the range of 0.1-0.25. The 1 1/2 Dogbone joint (D) is considered a standard design, originally developed by LBF (Laboratorium für Betriebsfestigkeit) in Germany with the aim of simulating the load conditions of runouts of stiffeners attached to the outer skin, see Figure 33. The design goal for the LT was 40% and the SB was expected to be .5, making it a *medium-LT, high-SB joint*. This joint is sensitive to installation parameters and LT depends on fastener fit, clamping force and fastener flexibility. During the last 20 years, investigations indicated an LT of approximately 20-30% which is substantially lower than anticipated. However, recent measurements and corresponding detailed evaluation show values of LT well in agreement with the original design goal. Lap joints (L) are *high-LT joints* which, in their single shear configuration, are characterised also by *high SB* due to the asymmetric eccentricity of the load carrying members, see Figure 33. Load transfer in the two-row joint should be around 50% for each rivet row, but measured values give a range of ±10% possibly due to asymmetric effects of the formed and manufactured heads of countersunk fasteners. The effective significance of the single shear lap joint specimen with respect to their use in aircraft construction has been questioned because of their high SB level which tends to generate too short fatigue lives, but nonetheless, their use is still widespread to gain data for worst critical conditions. The joint is often fatigue tested with anti-buckling devices that also prevent bending. Secondary bending measurements were found to be in the range of 1.1-1.7. However, the SB in this joint is critically dependent on joint geometry, number of fastener rows and fastener characteristics.

A fastener system includes all the fastener installation parameters, which have to be determined when designing a joint. Thus, the type of fastener, hole geometry, amount of fastener fit, hole and faying surface quality and clamping force all influence and characterise the joint. Five different main configurations of fastener systems are shown in Figure 34, i.e. the Lockbolt and Hi-Lok screws, the solid Aluminium and Monel rivets and blind fasteners. Subgroups are formed based on fastener material, fastener fit and fastener pre-load after installation (i.e. the amount of clamping force (CF)). The fastener fit is dependent on the chosen fastener system. The Hi-Lok and Lockbolt systems are used with approximately 50 μm interference fit. All solid rivet systems are hole filling. However, blind fasteners such as B-bolt (steel) and Ti-Matic (titanium) are not regarded as hole filling fasteners since they guarantee a minimum fastener clamping force, approximately 2 kN and 1 kN respectively. The Cherry SST blind rivet is a hole filling type of fastener, and thus, guarantees no fastener pre-load after installation. The manufacturers of Hi-Lok and Lockbolt fasteners guarantee a minimum fastener CF. The guaranteed magnitude depends on fastener diameter and is approximately 3 kN and 4 kN, respectively, which is, however, not a representative value of fastener CF. Experiments indicate substantially higher values. Thus, the tested Hi-Lok systems are characterised by a CF of approximately 5 kN and the Lockbolt system by a CF of approximately 7 kN. Further, the CF depends on the level of friction/lubrication between nut and plate as well as between fastener threads and nut. A substantially higher clamping force was found in Hi-Lok systems when nut/plate and/or nut/threads were lubricated. The solid rivet systems are characterised by almost zero CF, i.e. CF=0.0-0.5 kN. In design and fatigue life estimations the CF of rivet systems is usually neglected. All fasteners are countersunk (CSK) with a nominal diameter of approximately 5 mm. Low and high interference fit fasteners, as well as clearance fit, are used.

Testing was performed on specimens of aluminium alloys 2024-T3 and 7475-T761. In Fig. 35 test results obtained on 2024-T3 specimens subjected to miniTWIST loading. The gust-dominated miniTWIST load history is representative for the wing root of a transport aircraft. One block consists of 4000 flights with an average flight length of 15 cycles. All specimens were tested at a maximum gross stress level equal to 250 MPa. The results indicate that the fastener systems in Hi-Lok and Lockbolt specimens show a significantly higher fatigue resistance than all the other systems tested. Also, it may be noticed that the fastener systems in the B-bolt specimens particularly, but also in the solid aluminium rivet and the Cherry SST specimens show a lower fatigue resistance than remaining systems.

Depending on the fastener system there are several locations where fatigue cracks may initiate. For example, the crack initiation site may occur in the minimum net section at the edge of the fastener holes in the form of a part-elliptical corner crack, see the left side of Figure 36. This type of crack initiation consistently corresponds to the fastener systems with no or minor clamping force, i.e. solid and blind rivet systems. For a fastener system with a slightly higher clamping force and a stiffer fastener, the crack may also initiate at the intersection of the countersunk and the hole as shown in the middle of Figure 36. For fastener systems with considerable clamping force, i.e. high-performing fastener systems, the fretting mechanism is dominant and cracks always initiate at some distance away from the fastener hole, see the right side of Figure 36 and Fig. 37 to get a different view.

Detailed stress analyses of the medium-LT joint with a systematic, step-by-step improvement in each model were carried out in order to evaluate the degree of modelling complexity needed to study stress distributions accurately. Results show that the model complexity needed for determination of LT and SB is relatively low. However, if more sophisticated and detailed analyses of LT and SB, or stress distribution and contact problem analyses, e.g. fretting, are to be performed, the fastener clamping force (CF), fastener geometry and fastener fit ought to be taken into account. A series of non-linear FE-analyses of the medium-LT joint including contact analysis and fretting studies was performed. It was shown that it was possible to identify the critical area where fretting fatigue cracks initiated. In the fatigue crack growth analyses, the stress amplitude at the crack initiation sites is of interest. At the edge of the hole, the stress amplitude is substantially higher for solid rivets than for fastener systems with CF. For threaded fastener systems, the Ti Lockbolt fasteners have the highest stress amplitude at the crack initiation site (3 mm ahead of minimum net section), see Figure 38.

For threaded fasteners, cracks are initiated 2-3 mm ahead of the minimum net section. Cracks in the Ti Hi-Lok specimens grew around the fastener hole and hence, subsequent analyses are relatively simple. However, cracks in both steel Hi-Lok and Ti Lockbolt specimens grew into the fastener hole relatively early during the fatigue life which implies a sudden step-increase in crack size from some 10th of a millimetre to 3 mm and from approximately 1 mm to 3 mm, respectively. This behaviour affects the K-factor and hence crack growth rates as well as the residual strength of the specimen, see Figure 39 which shows data obtained by fractographic analyses. Due to the described crack growth behaviour of the steel Hi-Lok and Ti Lockbolt specimens only the Ti Hi-Lok specimens were analysed. Numerical results shown in Figure 40 were obtained by using weight function techniques to determine stress intensity factors and then the modified Dugdale model discussed previously was used to predict fatigue crack growth. The analytical results show good agreement for crack growth up to 0.5 mm which is typically within the short crack growth regime. For a crack size larger than 0.5 mm, the analytical results are conservative. This is probably due to the increased boundary effect for a large crack in this complex joint configuration. Detailed finite element analyses seem to be necessary for an estimation of stress intensity factors for large crack sizes in this specimen. Both the numerical result and test results show that the major part of fatigue life for the Ti Hi-Lok specimens is in the small crack growth range which is sensitive to the size of initial flaws. A large initial flaw size, accidentally introduced, at the location of crack initiation may significantly reduce the fatigue life.

Apart from a more extensive study similar to what has been described above, some statistical modelling aspects were also studied, Ref. (93). A Monte-Carlo simulation was repeated approximately 60 times in order to visualise the individual crack growth behaviour and to be in the same order as the actual testing carried out. Results show that the influence of crack growth rate variation is not as significant as the initial flaw size distribution. The obtained total scatter was found to be in good accordance with experimental results. From an experimental point of view, these studies have shown that the fatigue behaviour of 1 1/2 Dogbone type of joints is crucially dependent on the type of fastener system, especially on fastener installation parameters such as fastener material and fastener pre-load after installation. These parameters not only determine total fatigue life but also govern the damage initiation process. Fastener systems with high clamping force exhibit extensive oxide debris 2-3 mm ahead of the fastener hole at the crack initiation site. The fretting mechanism is prominent in such specimens and fatigue life is relatively high. Hence, the fretting process is dominant in specimens with relatively high fatigue resistance. Regarding fastener systems with low or almost no clamping force, all cracks initiated in the minimum net section at the edge of the fastener hole. The corresponding fatigue life is relatively low. The fastener material is clearly affecting fatigue life. Steel fasteners, both in systems with high and low clamping force, tend to decrease fatigue life when compared to less stiff fastener materials such as titanium and aluminium. The fatigue life of solid rivet and blind fastener specimens is relatively insensitive to the initial flaw size at the edge of the fastener hole. Hence, differences in the fatigue crack extension process determine the total fatigue life of these joints. However, for fastener systems with high clamping force, the initiation mechanism and initial flaw size determine the major difference in total fatigue life.

All in all, these mechanical joint studies have, apart from producing the experimental data sets, clarified some interesting points on the degree of model complexity needed in order to reasonably well predict the fatigue crack growth behaviour in the joints studied. However, it appears as if much more work is still needed before analytical damage tolerance analyses can replace the current design approaches using large experimental data sets and applying appropriate safety factors.

Ageing Aircraft

Vast amounts of money have been spent, primarily in the USA but also in Europe, in order to solve the problems of the ageing aircraft fleet around the world. Although the problem of multi-site damage (MSD), or wide spread fatigue damage, certainly is real enough this author wonders if the efforts made really have produced value for money spent. It seems feasible that a portion of the funding was given in order to show the general public that their government officials take responsibility. In any case, the problem has certainly been taken seriously by airworthiness authorities, aircraft manufacturers and airline operators. Many conferences have been devoted exclusively to ageing aircraft problems, e.g. Ref. (96). It is not possible to review detailed technical activities here, but a good overview of damage tolerance including ageing jet transport problems is given in Ref. (65).

The question is really whether the ageing aircraft problem has been solved or if there are still risks for structural failures of individual aircraft becoming of age. This author believes that tremendous progress has been made, but that the problem not yet is solved. Lessons learned for the design of new aircraft include what may be the most important factor in not risking future development of wide spread fatigue damage. This is to decrease the nominal stress levels in new aircraft and to avoid poor joints. Using a probabilistic analysis to handle the MSD problem, it was found in Ref. (97) that by designing areas of the structure prone to MSD with an extra stress margin of about 10 – 15 % the problem could be avoided. Such a solution is quite feasible and should not be too expensive. Recently, more refined models have been developed that can aid in the evaluation of the probability of multiple crack initiation as function of different load levels, e.g. Refs. (98,99). For existing aircraft the nominal stresses cannot be changed and very many models have been proposed to predict the increase in fatigue crack growth rates and the decrease in residual strength from different arrays of multiple cracks. However, the problem remains to find the small flaws that may suddenly coalesce into a large lead crack that may rapidly propagate to failure. The possible degradation of material properties is another factor that has been studied, but that is hard to simulate by accelerated testing. To this author, these two latter points, i.e. improved and reliable non-destructive testing, and better understanding of environmental degradation are the two key factors for better understanding and control of the ageing aircraft problem.

Composites

The discussion has so far been focussed on metallic structures. More and more composite structures, sandwich structures, and possibly hybrid material systems (Glare or Arall) are being introduced in new aircraft. Until recently, fatigue of composites has not been considered a potential problem in aircraft, but designs have been driven by stiffness, static strength, joints, or impact damage. However, fatigue tests on the Airbus 320 full-scale composite vertical tail revealed that sufficient static strength does not automatically guarantee sufficient durability, Ref. (100). Damage was formed at the left main attachment fitting of the front spar. The damage was caused by delamination of the root rib connecting angle from the main fitting. The initial delamination caused further delaminations in the fitting laminate until failure at the load introduction section. After introducing an additional rivet line, as a serial solution and retrofitted in the test programme, the test was successfully completed. Reasons for the relative weak interest in fatigue of composite structures probably derives from several factors, one being the early focus on fatigue testing on unidirectional composites in tension, another being the fairly low strain levels currently used in composite structures. For quasi-isotropic, or other angle-ply combinations, composites tested in compression fatigue may result, and the same holds for composite joints. In the future, it seems likely that composites need to be able to operate at significantly higher strain levels than today, in order to remain competitive with new cheap production methods for metallic structures.

In the literature, quite a few studies of composite fatigue have been performed. These are largely divided into two fields, delamination studies and others, with a clear dominance on the former. For delamination studies, the two frequently studied specimen types, Double Cantilever Beam (DCB) and Edge Notch Flexure (ENF), are shown in Fig. 41. Refs. (101, 102) contain detailed numerical modelling results as well as experimental results for these two specimen types, respectively. The numerical evaluation of strain energy release rates was performed by an hp version of the finite element method with techniques detailed in Ref. (103). Comparisons of numerical FEM results, for the DCB-specimen, with various beam models presented in the literature showed that beam models tended to predict slightly too high values of G_I, especially for short crack lengths where deviations of a few up to some six or seven % are common, depending on the interface properties. Cyclic properties for the DCB-specimen were also studied in Ref. (101). As expected, interfaces with 0°/0° had the lowest resistance to delamination growth, followed by specimens with 45°/45° interface while specimens with 90°/90° interface had the highest resistance. Common to all interface types are steep delamination growth slopes, indicating rapid growth of delaminations once initiated. Taking the ratio of G_{IC}/G_{th} , i.e. the ratio between fracture toughness and fatigue threshold, as a measure of when delaminations will grow during static or cyclic loading it is possible to calculate the ratio between the associated stress levels. This value, i.e. the ratio between the stress needed to cause static delamination initiation and stress at the threshold condition, became roughly 1.7 in mode I, Ref. (101), and roughly 3 in mode II loading, Ref. (102). It is interesting to compare these values to the ratio of ultimate stress to limit static stress, frequently selected to 1.5, to which composite structures are tested during certification. As the limit stress level is also the maximum/minimum fatigue stress level to which the structure is subjected during fatigue loading, it follows that a delamination that does not grow under static load might well do so under fatigue testing

For ENF-specimens the strain energy release rate computations are much more elaborate than for the DCB-specimens. Not only mode II but also mode III prevails at the delamination front, Ref. (102). Also, the energy release rate approaches infinity as the free surfaces are approached due to the vortex singularity located at these positions. Another interesting factor to consider is that, for mode II loading, friction might influence the strain energy release rates. However, the friction is likely to be a factor of both the contacting surfaces (which layers the delamination grows between) and possibly the cyclic loading itself. It was found in Refs. (102, 104) that the effective coefficient of friction depends on the amount of load transfer being carried by the matrix/matrix, fibre/matrix, and fibre/fibre contacts. The evolution of the coefficient of friction, see Fig. 42, depends on whether the initial surfaces are covered with a matrix layer or if the fibres are visible. Any matrix layer will flow to the sides of the contact region, causing fibres to become visible. No wear of fibres themselves was observed. Coefficients of friction were found t stabilise in the range of 0.2 to 0.6 depending on surface details. This effect does indeed influence the value of the strain energy release rate, as indicated in Fig. 43, but it may be of more practical relevance for fatigue behaviour of composite joints where similar effects could take place.

In experimental studies of delamination growth in structural components, delaminations are normally simulated by teflon inserts. Under compression loading, delamination buckling may occur. However, in the author's laboratory delamination growth under cyclic loading has not been found at load levels less than that for local buckling of the tested panels. Whether teflon layers represent delaminations well or not does not seem obvious. A simple experiment to study this question is illustrated in Fig. 44. Here, a small hole is drilled through the side of the test specimen, until the teflon layer is reached. At this point a light pressure is added to the teflon layer, such that any adhesion between the teflon and the composite layers is expected to disappear. As shown in the figure, this small pressure caused a decrease in buckling loads with two thirds. Although this early test needs further substantiation, it indicates that teflon inserts may not be altogether suitable for use as artificial delaminations in certification testing. Cyclic delamination growth can be modelled reasonably well with growth models similar to Paris' law for metals. In Refs. (105, 106) such a model is derived and applied to notched specimens tested under both constant amplitude loading and different load spectra. Fig. 45 shows the load spectrum for the aft fitting of the vertical fin of the JAS39 Gripen aircraft. This is an almost symmetrical spectrum consisting of 5767 load cycles. Also shown in the figure are resulting spectra for elimination of 30 and 50 % of all load cycles in the original spectrum. Fig. 46 shows the number of resulting load cycles, for the spectrum shown in Fig. 45 and also for a load spectrum for the upper fitting of wing of the same aircraft, at different elimination levels. Also shown in Fig. 46 are the predicted normalised fatigue lives due to the different elimination levels. These predictions, and others, were compared to experimental results presented in Ref. (107) for notched specimens made of HTA7/6376C. The fatigue life predictions correlated reasonably well with experimental data and it was concluded that some 50% load cycle elimination may be used without significant influence on overall fatigue life behaviour.

The fatigue behaviour of composite joints is not yet well understood and the number of influencing parameters is large. In an experimental test programme on carbon fibre laminates joined by protruding-head bolts (hexagon bolts) four different configurations were studied, Refs. (108,109). Both the specimen type shown in Fig. 47 with one, two and three rows of two bolts, as well as single overlap joints with three rows of two bolts, were studied. The testing was done at R = -1 and almost all specimens failed due to fatigue failure of the bolts. By plotting the average bearing stress on one bolt versus the number of cycles to failure all data merge into one linear scatter-band. As long as the bolt diameters are the same it then becomes simple to calculate the fatigue life for any other joint configuration. In another test programme, Ref. (110), the fatigue behaviour of the specimen shown in Fig. 47, with three different types of countersunk fasteners, was studied. The specimens with Torque-set bolts were fastened by 9 Nm torque. The Huck-comp fasteners were installed with a preload of 6 kN and the average torque of the composite bolts was 2.7 Nm. Fatigue test results obtained at R = -1 are shown in Fig. 48 together with results from specimens with hexagon bolts discussed previously. As the results indicate there is quite a difference in fatigue life for specimens with different fastener types. Testing was done both on specimens with quasi-isotropic properties and with 0°-dominated lay-ups and in general the former showed better properties both statically and in fatigue when evaluating the behaviour in terms of applied strain levels.

Additional test programmes under way in the author's laboratory include composite joints under spectrum loading and fatigue testing of both notched specimens and joints with impact damage. In order to eventually build up a similar understanding of durability and damage tolerance for composites as that which exists for metals such testing is necessary combined with proper numerical modelling and basic studies of fatigue mechanisms involved.

Future Developments and Problem Areas

New international trends in the manufacturing of aircraft include the focus on production costs, new manufacturing techniques, passenger comfort (like reduced internal noise levels by active control), and environmental issues (both engine emissions and noise). These trends are driven partly by the changes in safety political issues that in turn have led to reduced military spending and fewer but larger aircraft companies in the world. Some obvious problems may arise from these developments. In the past, science and technology have largely been driven by developments in the military sector. It has, however, been hard to produce transport aircraft with a profit. Yet, maybe partly due to national prestige and partly due to a perceived need for maintaining this high technological industry independently, various countries have continued to put money into new aircraft projects. With the gradual build up of the Airbus consortium in Europe, frequent and hard accusations of illegal support to the aeronautical sector has been aimed at Europe by the USA. In return Europe has accused the USA for providing unfair support to its civil aircraft sector through technological developments obtained by governmental grants within the military divisions of the same companies. It seems likely that both accusations are correct. Now then, how to ensure necessary development and maintained safety for the future? It appears that the civil aircraft industry not likely can bear such costs and simultaneously make a profit. There may be a few possible answers to this question. One answer is that governments continue to subsidise the civil aircraft sector for job safety, prestige, or whatever reasons. Another answer is the change from a focus on technical superiority into financial aspects only, i.e. not really developing today's technology but rather developing production techniques that makes it possible to manufacture a similar aircraft as today at a lower cost. Yet another solution would be to sell the aircraft at higher price, which would lead to higher ticket prices for the end customer, the passenger.

Considering airline prices after World War II, such a ticket price increase does not seem unreasonable. However, with the competition hinted at above, perhaps this is not the obvious way to go.

New manufacturing techniques include high speed machining (HSM) for metallic parts and resin transfer moulding for composite parts. HSM, however, suffers from a few problems at present. In the manufacturing of complicated integral curved components, down to mm thickness or less, residual stresses of unknown magnitude arise. Preliminary fatigue testing of notched HSM specimens of aluminium alloy 7010-T74, indicate lower fatigue strength than for conventionally manufactured specimens. In constant amplitude loading, at cutting speeds of1800 m/min, the decrease was some 15 %. In spectrum loading the difference was higher, for a tension dominated wing spectrum, and lower, for a symmetrical fin spectrum. The formation of residual stress fields and their relaxation during flight loading needs further studies. Potentially more dangerous is the fact that HSM and other new techniques make it possible to produce integral, i.e. monolithic load path, structures. This is indeed what is aimed for, in order to save costs not only for fastener systems and assembly, but also for tools. The downside then, is the reduced damage tolerance properties that certainly must result from this development. This author considers this a potential safety hazard for future large transport aircraft. This is particularly so for the new large aircraft, that typically have weight problems and where higher strength materials and higher applied stress levels are likely to be used to rectify this. As mentioned above, the use of lower nominal stresses is highly advocated in order to avoid future ageing aircraft problems. For composites, and sandwich constructions, there is probably a need to increase the applied strain levels if these systems will be competitive against the new metallic manufacturing techniques that are evolving presently. This might lead to a situation where fatigue failures will appear, see the discussion on composites above. Clearly, similar knowledge for the durability and damage tolerant design of composites to that one built up for metals is still needed.

For military aircraft a few new possibilities/problems regarding the fourth generation of fighters were discussed previously. Multi-role capabilities, advanced flight control systems, new payloads, and new usage due to new safety political issues like international peace keeping missions, all lead to different load spectra than that one originally designed against. With onboard load monitoring systems such differences should be possible to handle. However, certain combinations of advanced manoeuvre loads and dynamic loading due to buffets or other separated flow are difficult to predict and in need of better understanding both to guarantee safe flight handling and lasting structural response. A particular new problem for most aeronautical companies is the long time between new projects. In the past, a person spending a career in aeronautics would typically be involved in several new aircraft projects. Today, few people will have this chance, at least in the military sector. Hence, it is now more difficult to build on the expertise developed in earlier projects. This might become a serious problem both for unexpected problems in a current fleet of aircraft, but also in the design stage of a new aircraft generation.

The decline in military spending has led to a situation where military technology no longer is leading the technical development. Since roughly 1990 other technical areas such as biotechnology, information technology, sporting goods, and the electronics game industry are new technology leaders. This development together with new societal values, or the lack of them, will ultimately lead to a situation where the aeronautical sector will have large problems to attract the most skilled workers and the best engineers/scientists. To some extent this is already seen, at least in the author's country where it is harder than ever to attract very good young engineers to a career in the aeronautical field. Some of this development, i.e. the development of new high tech areas, is probably good in the overall sense. However, there appears to also arise a situation where higher education is frown upon by young people and where society makes little effort to change this. In the author's country the present young generation, of some 20 – 30 years of age, is the first since the beginning of last century to have a lower average education than their parents. To some extent this might be blamed on the political leadership that appear to make a point of hailing their own lack of education as proof of being well suited to represent the "people". To some extent the media are to be blamed, with ever more stupid programmes to entertain the masses and with a mostly genuinely incorrect descriptions of industrial development, economical welfare as function of this development, and frequently biased rather than objective treatment of news. However, also the aeronautical industry is to be blamed for not having worked harder to attract the best people, not having made salaries on par with what less skilled personnel typically earn in other areas, and not having tried hard to persuade the very young generation that aeronautical science and engineering actually is fun and challenging instead of dull and introvert. Complaints like above may be wrong, but the general observation of new societal values is hardly so. Hence, a perceived future competence shortage may become a major problem for the aeronautical sector.

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Table 1. ICAF CONFERENCES AND SYMPOSIA

Year	Conference	Symposium	Location	Plantema Lecturer *
1951	Foundation	of ICAF	Cranfield	
1952	1		Amsterdam	
1953	2		Stockholm	
1955	3		Cranfield	
1956	4		Zurich	
1957	5		Brussels	
1959	6	1	Amsterdam	
1961	7	2	Paris	
1963	8	3	Rome	
1965	9	4	Munich	
1967	10	5	Melbourne	J. Branger
1969	11	**	Stockholm	J. Schijve
1971	12	6	Miami	E. L. Ripley
1973	13	7	London	E. Gassner
1975	14	8	Lausanne	S. Eggwertz
1977	15	9	Darmstadt	H. F. Hardrath
1979	16	10	Brussels	A. J. Troughton
1981	17	11	Noordwijkerhout	O. Buxbaum
1983	18	12	Toulouse	J. Y. Mann
1985	19	13	Pisa	L. Jarfall
1987	20	14	Ottawa	T. Swift
1989	21	15	Jerusalem	J. B. De Jonge
1991	22	16	Tokyo	R. M. Bader
1993	23	17	Stockholm	U. G. Goranson
1995	24	18	Melbourne	W. Schütz
1997	25	19	Edinburgh	J. W. Lincoln
1999	26	20	Bellevue (Seattle)	J. C. Newman, Jr.
2001	27	21	Toulouse	A. F. Blom

* Frederik J. Plantema, October 21, 1911 – November 13, 1966

** No Symposium – Two Day Technical Session

Table 2. PLANTEMA MEMORIAL LECTURES

Year	Author	Title
1967	J. Branger	The ICAF, its Foundation, Growing and To-Day's Philosophy
1969	J. Schijve	Cumulative Damage Problems in Aircraft Structures and Materials
1971	E. L. Ripley	The Philosophy of Structural Testing a Supersonic Transport Aircraft with Particular reference to the Influence of the Thermal Cycle
1973	E. Gassner	Fatigue Life of Structural Components under Random Loading
1975	S. Eggwertz	Reliability Analysis of Wing Panel Considering Test Results from Initiation of First and Subsequent Fatigue Cracks
1977	H. F. Hardrath	Advanced Composites – The Structures of the Future
1979	A. J. Troughton	33 Years of Aircraft Fatigue
1981	O. Buxbaum	Landing Gear Loads of Civil Transport Aircraft
1983	J. Y. Mann	Aircraft Fatigue – With Particular Emphasis on Australian Operations and Research
1985	L. Jarfall	Fatigue and Damage Tolerance Analysis in the Aircraft Design Process
1987	T. Swift	Damage Tolerance in Pressurized Fuselages
1989	J. B. De Jonge	Assessment of Service Load Experience
1991	R. M. Bader	Structural Integrity Challenges
1993	U. G. Goranson	Damage Tolerance – Facts and Fiction
1995	W. Schütz	Corrosion Fatigue – The Forgotten Factor in Assessing Durability
1997	J. W. Lincoln	Aging Aircraft – USAF Experience and Actions
1999	J. C. Newman, Jr.	Advances in Fatigue and Fracture Mechanics Analysis for Aircraft Structures
2001	A. F. Blom	Fatigue Science and Engineering – Achievements and Challenges

Figure 1. On 2 May 1952, the world's first commercial jet airline service commenced with the departure from London's Heathrow Airport by deHavilland DH-106 Comet Mk1 G-ALYP (Yoke Peter) operated by British Overseas Airways Corp.

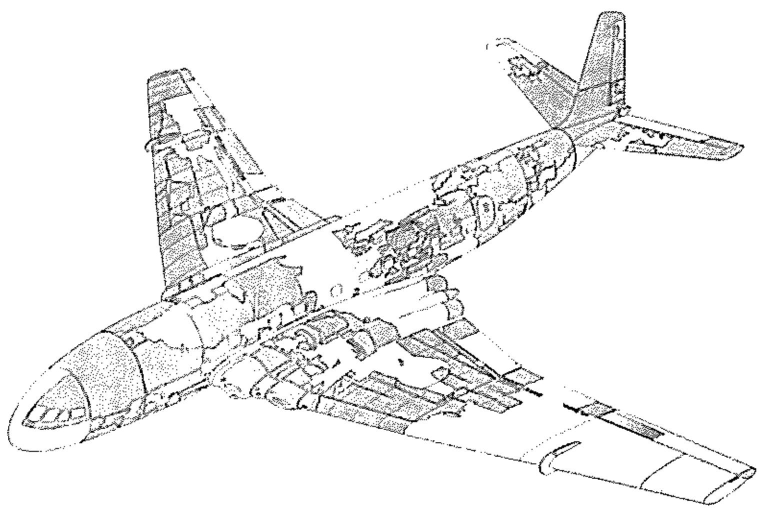


FIG. 2. DIAGRAM SHOWING AMOUNT OF WRECKAGE RECOVERED—G-ALYP.

Figure 2. Diagram showing amount of wreckage recovered of G-ALYP (Yoke Peter). From the official accident report (see http://surf.to/comet)

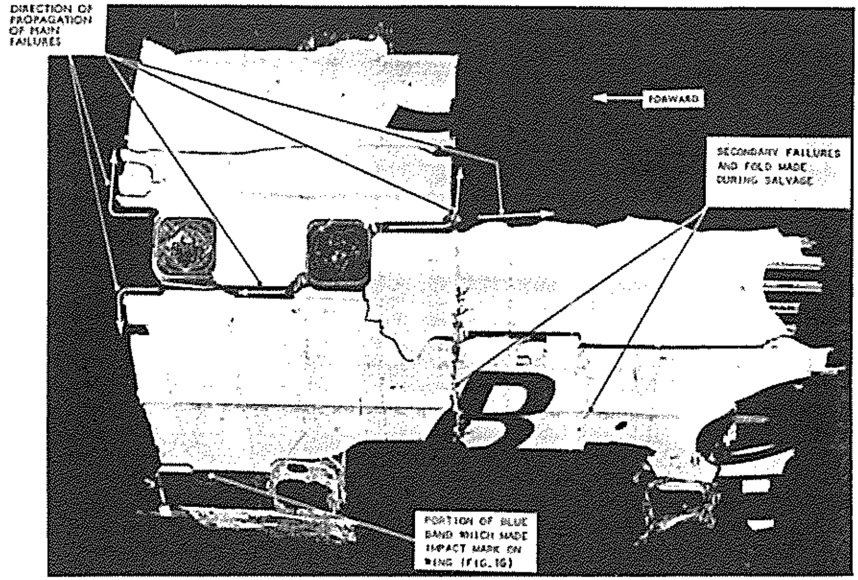


FIG. 12. PHOTOGRAPH OF WRECKAGE AROUND ADF AERIAL WINDOWS—G-ALYP.

Figure 3. Photograph of wreckage around A.D.F. aerial windows - Comet G-ALYP (Yoke Peter). From the official accident report (see http://surf.to/comet)

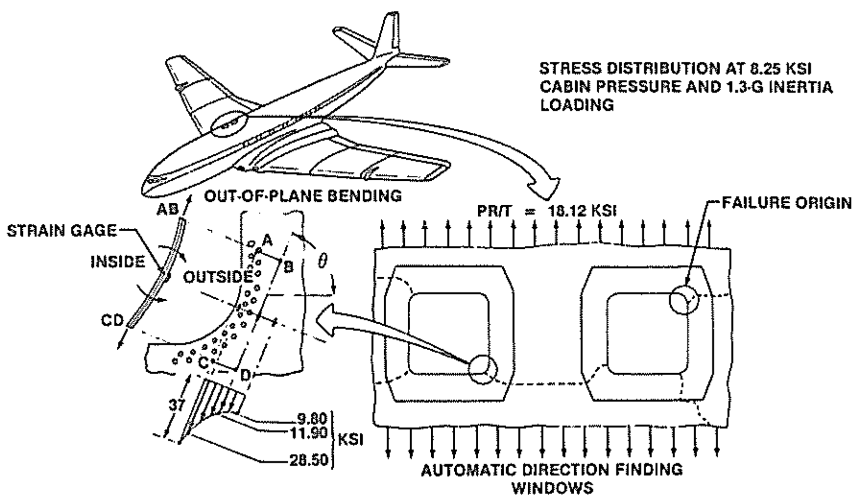


Figure 4. Probable failure origin in Comet I Yoke Peter. From Ref. (17)

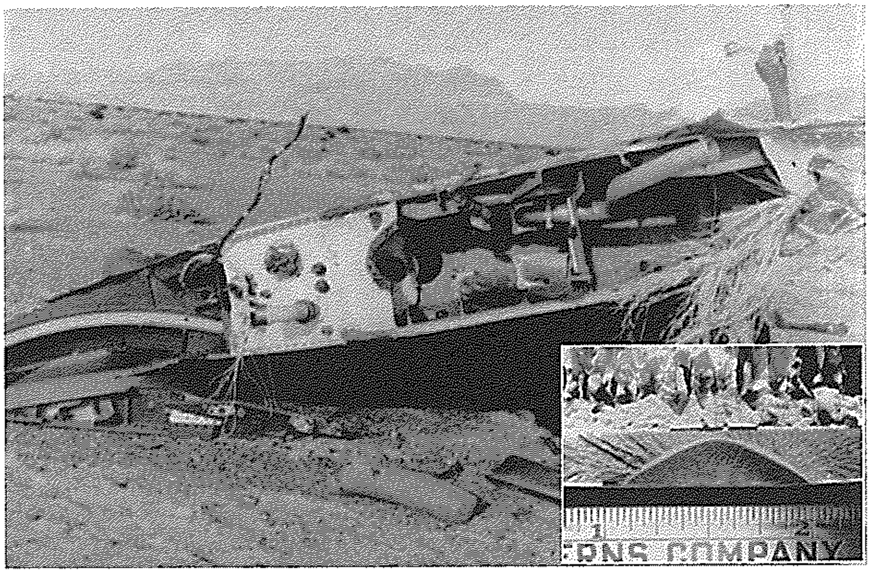


Figure 5. F-111 # 94 wing on desert floor. Fracture face shown in inset

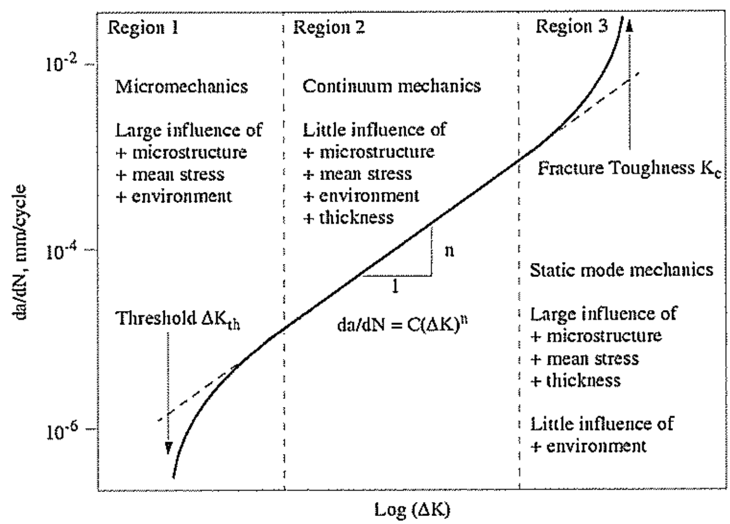


Figure 6. Schematic variation of fatigue crack growth rates (da/dN) with stress intensity range (ΔK), showing primary regimes of growth rate mechanisms

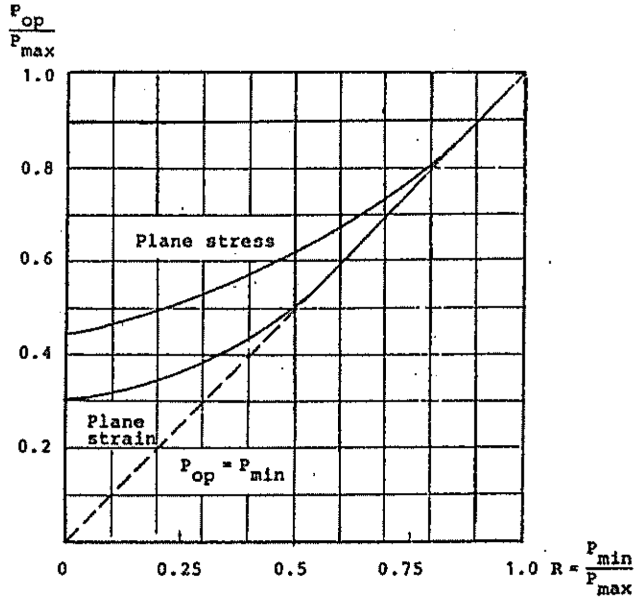


Figure 7. Normalised plasticity-induced crack closure loads versus stress ratio under plane strain and plane strain. Obtained by 2D elastic-plastic FEM-calculations, Ref. (39)

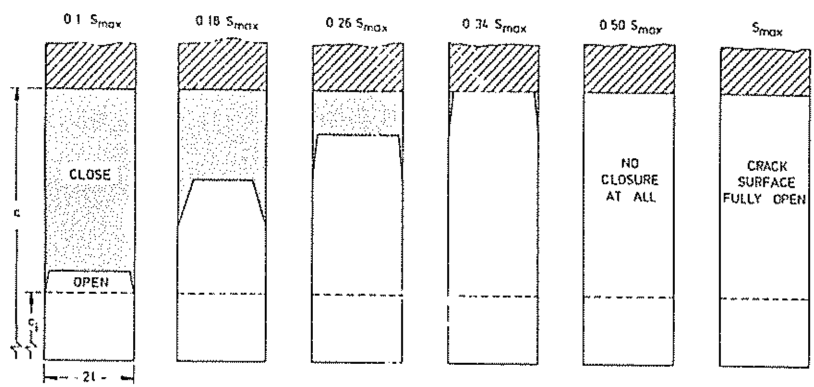


Figure 8. Closure and opening profiles on the crack surface plane obtained by 3D FEM-calculations with $S_{max} = 0.25\ \sigma_0$ and $R = 0.1$. The thickness is 12.7 mm, Ref. (38)

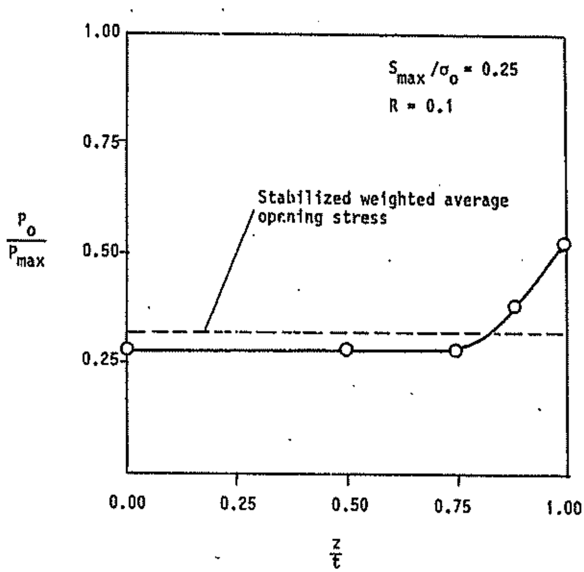


Figure 9. Variation of normalised crack opening stresses as function of normalised specimen thickness (t = 4.78 mm), Ref. (35)

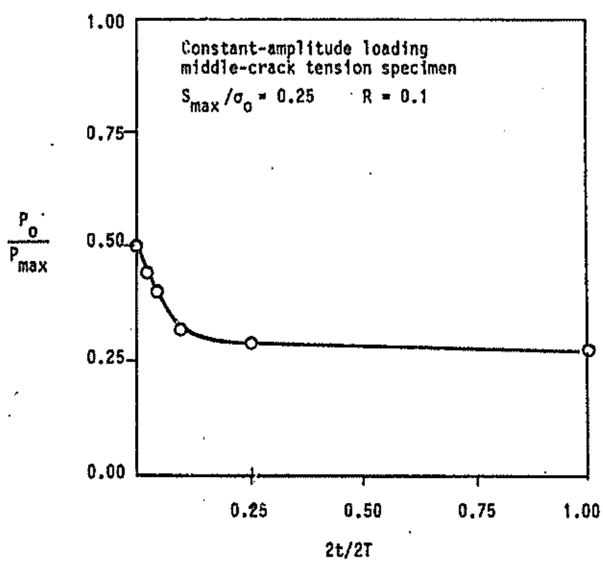


Figure 10. Normalised weighted average crack opening stress as function of normalised specimen thickness for a middle crack tension specimen loaded at R = 0.1. 2T = 101.6 mm is arbitrarily chosen to represent plane strain conditions, Ref. (35)

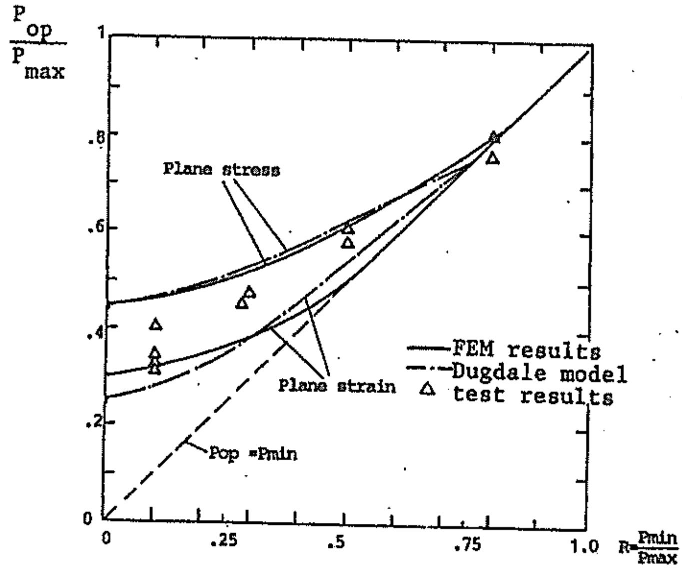


Figure 11. Normalised plasticity-induced crack closure loads versus stress ratio. Comparison of modified Dugdale model with FEM results and experimental observations, Ref. (30)

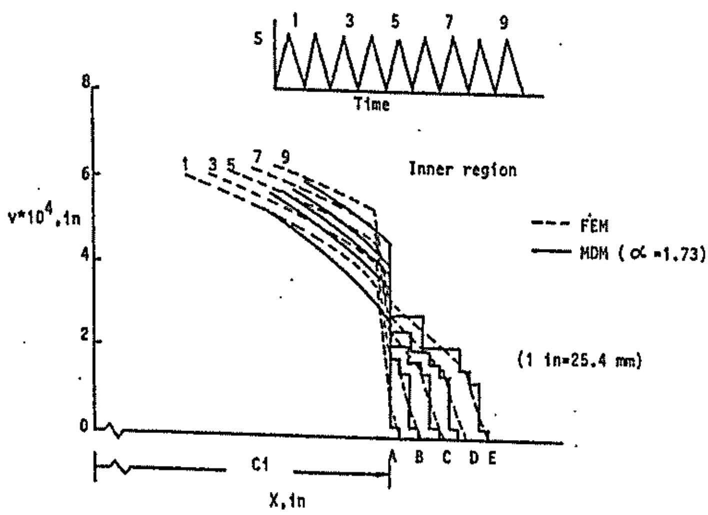


Figure 12. Comparison of crack surface displacements obtained by FEM and modified Dugdale model (MDM) at maximum applied stress with $S_{max} = 0.25\ \sigma_y$ and R = 0.1, Ref. (37)

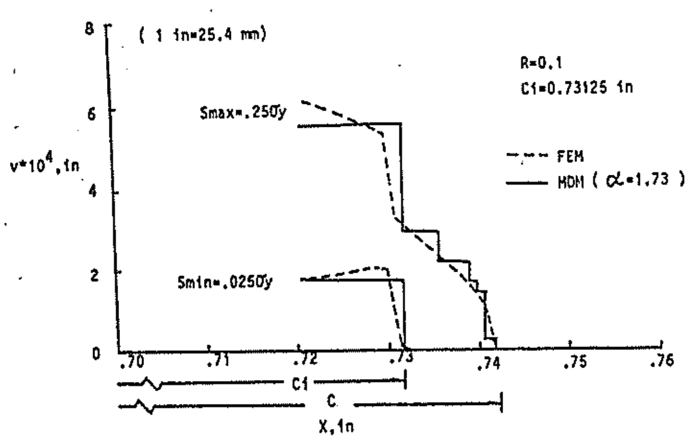


Figure 13. Comparison of crack surface displacements at the inner "plane strain" region at maximum and minimum applied stress, after crack extension, obtained by FEM nad MDM analyses, Ref. (37)

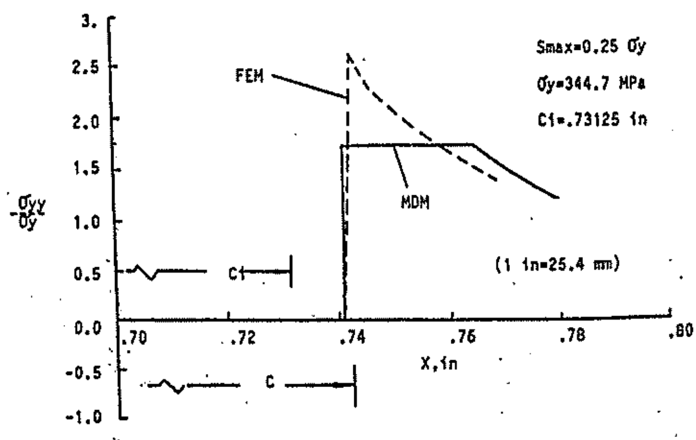


Figure 14. Comparison of the normalised stress along the crack plane at maximum applied stress, after crack extension, obtained by FEM and MDM analyses, Ref. (37)

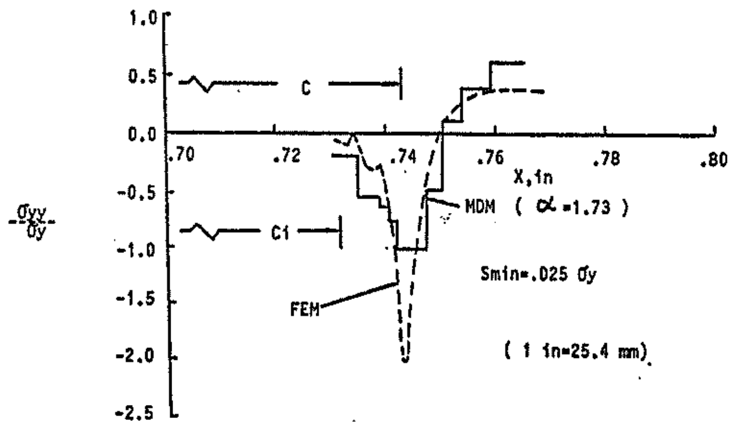


Figure 15. Comparison of the normalised stress along the crack plane at minimum applied stress, after crack extension, obtained by FEM and MDM analyses, Ref. (37)

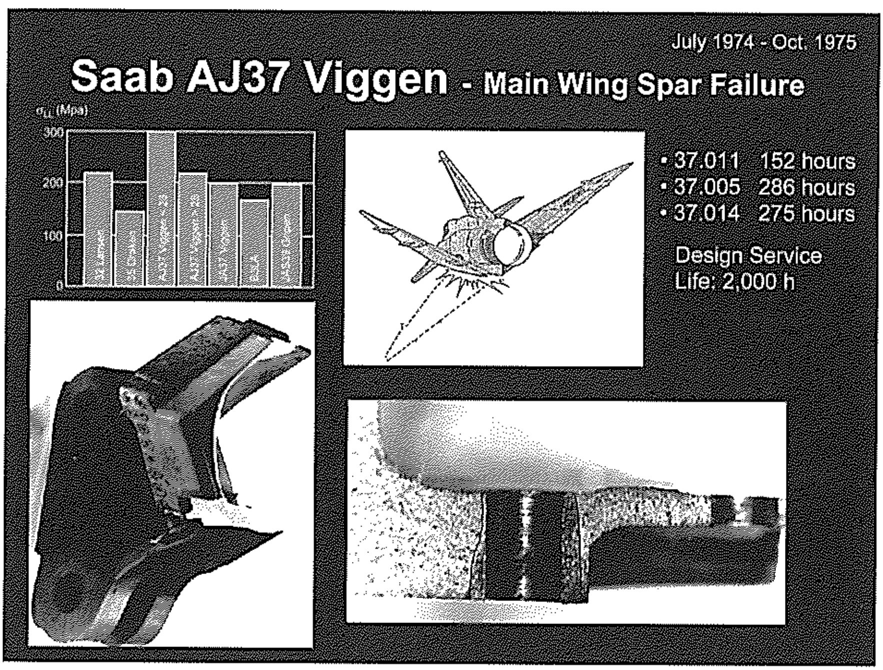


Figure 16. Summary of main wing spar failure in AJ37 Viggen fighter aircraft

Figure 17. Boeing 707 crashed short of Lusaka Airport, Zambia, on May 14, 1977. Inset shows horizontal stabilizer which is a single spar structure

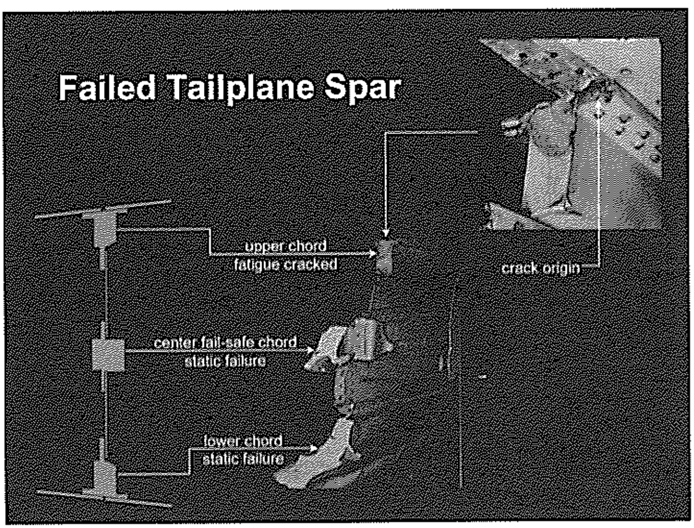


Figure 18. Failure of the Boeing 707 stabilizer rear spar chord due to decreased residual strength as the fatigue crack at the upper chord had grown to critical size

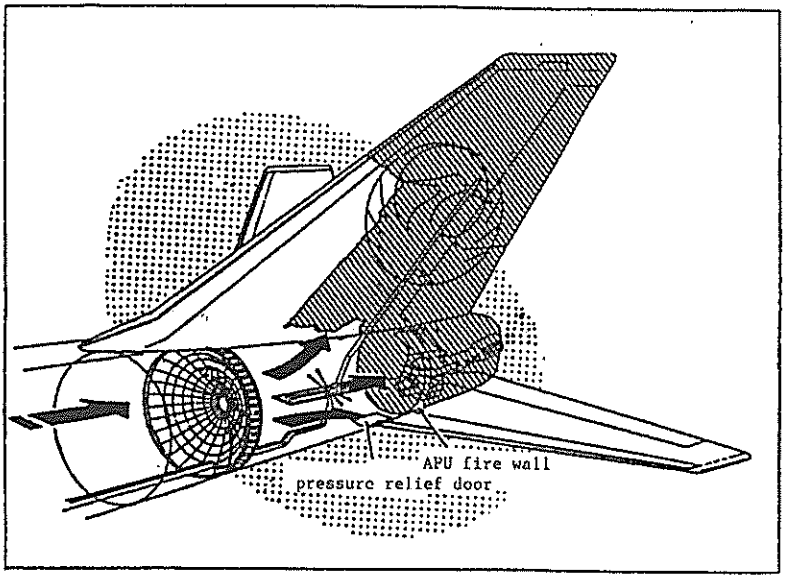


Figure 19. Estimated fracture of rear section of JA 8119 Boeing 747 SR-100 crashed in Japan, 1985, Ref. (61)

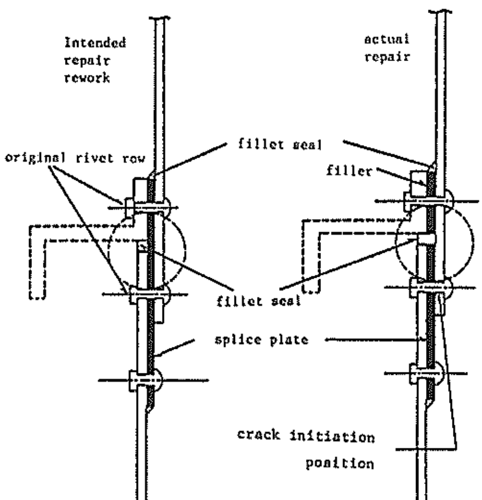


Figure 20. L18 splice section. Intended repair shown left and actual incorrect repair to the right, Ref. (61)

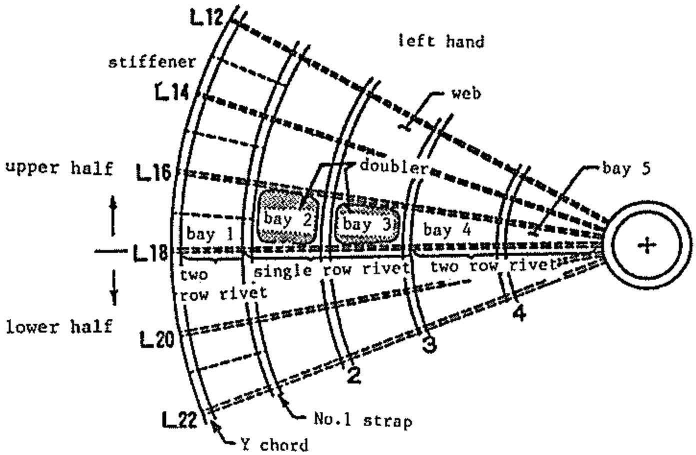


Figure 21. Aft pressure bulkhead of JA 8119 Boeing 747 SR-100 crashed in Japan, 1985, Ref. (61)

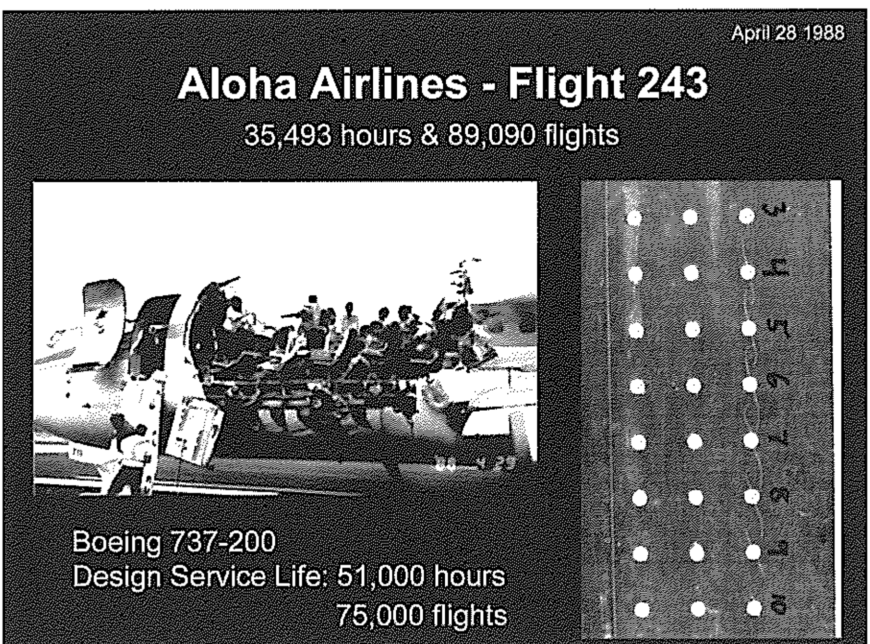


Figure 22. Aloha Airlines Flight 243 Boeing 737-200 after an explosive decompression and structural failure. The inset shows an example, unrelated to the accident, of multiple site fatigue damage

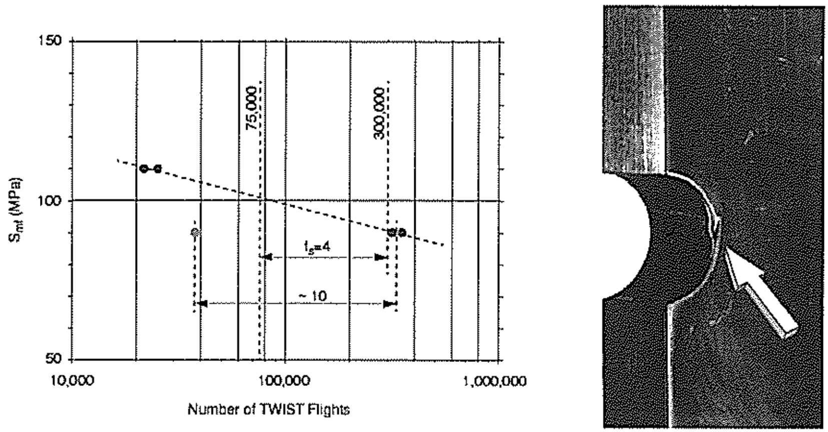


Figure 23. Example of large decrease in fatigue life due to a small flaw, undetected in the manufacturing stage

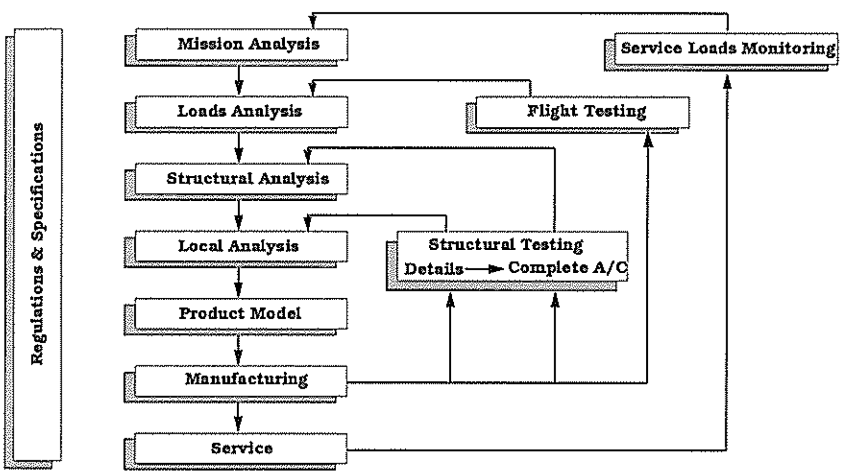


Figure 24. Schematic of Service Life Management

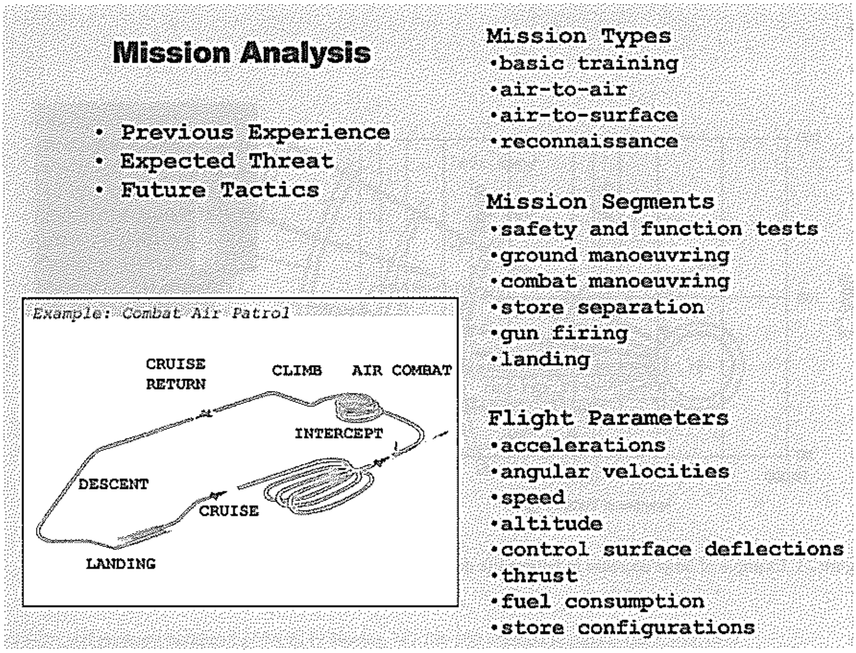


Figure 25. Examples of development of mission analysis used for derivation of load spectra

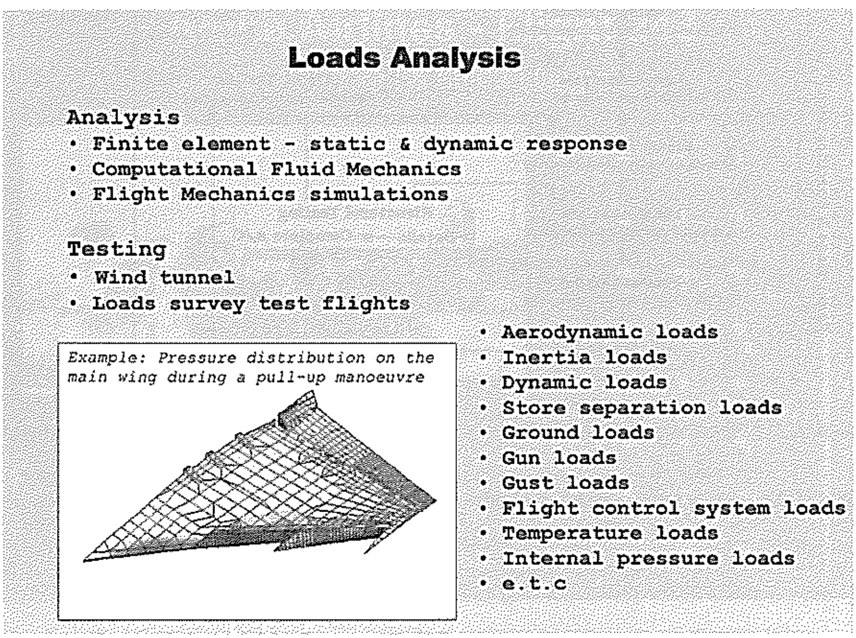


Figure 26. Examples of contributing factors in the derivation of load spectra

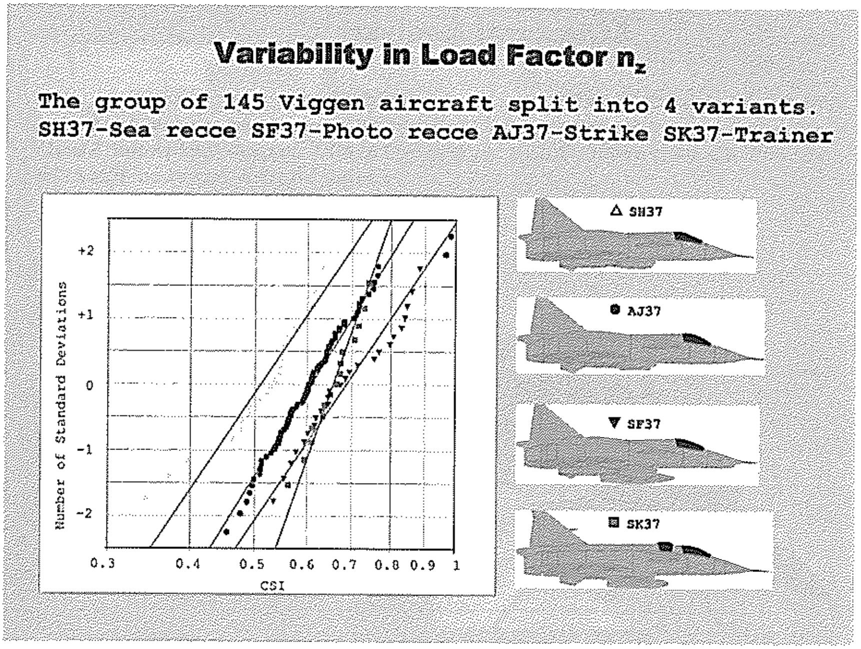


Figure 27. Normal distribution of relative crack severity indices for 4 versions of the Viggen aircraft, Ref. (78)

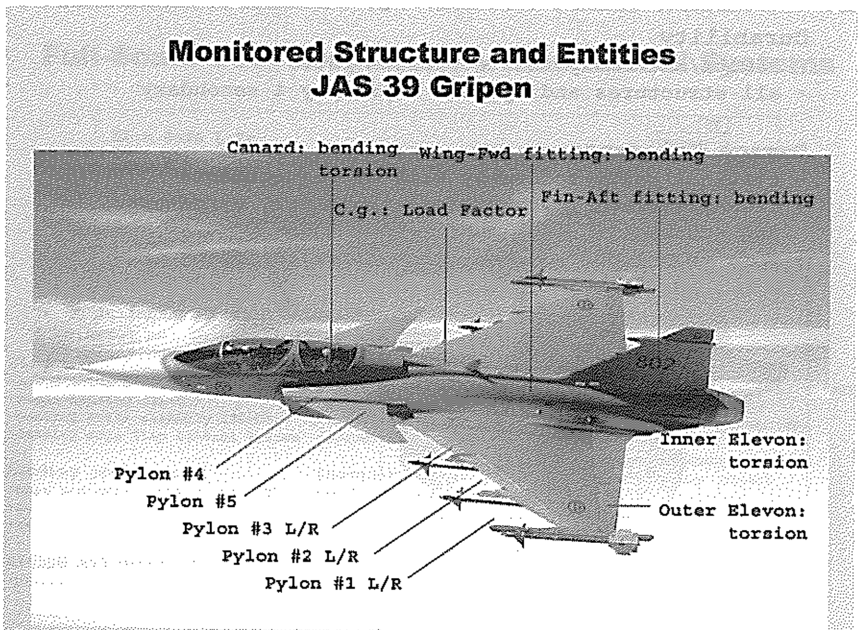
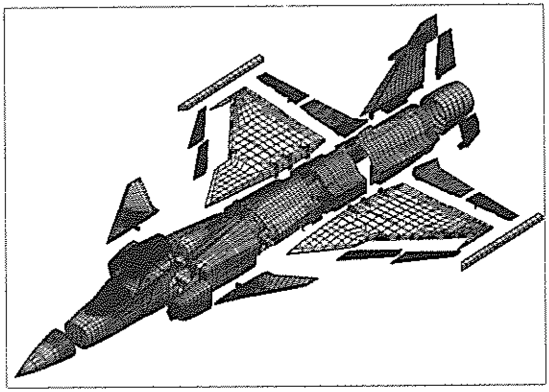


Figure 28. Monitored structures and load entities for JAS 39 Gripen

Structural Analysis

Finite Element Model
- sub-structured models
- 80,000 elements/400,000 d.o.f

Load Cases
- 750 unit load cases solved
- 13,000 unique balanced load cases in the mission profile



Example: Sub-structure of wing attachment unit

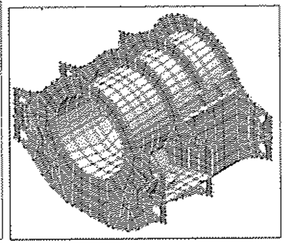
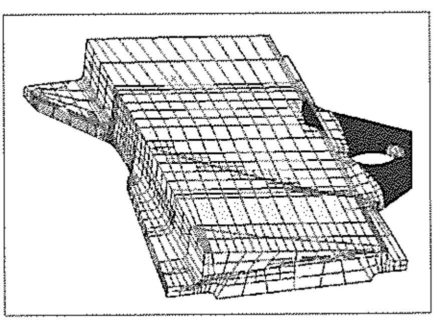


Figure 29. Example of modelling details for global structural analysis

Stress Analysis

Durability
- remote/extremely remote failure rates
- all structures and systems - primary & secondary

Damage Tolerance
- critical parts to comply with DT requirements
- cracks assumed to be present in primary structures and systems
- more than 1,000 assumed crack sites analysed



Example: Stress concentration at a lower flange of the forward wing carry-through bulkhead.

Figure 30. Example of detailed analysis for local stresses, strains, stress intensity factors, or displacements

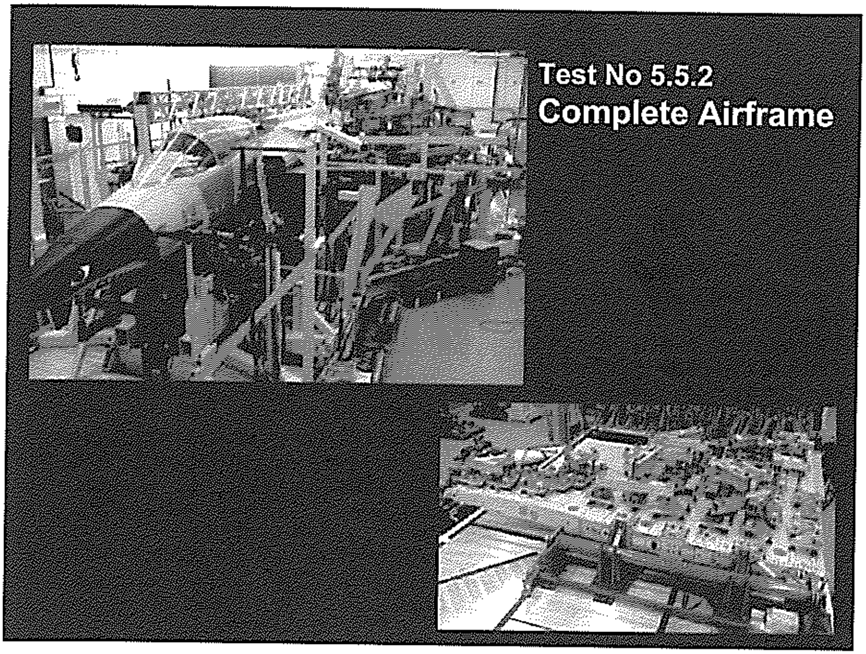
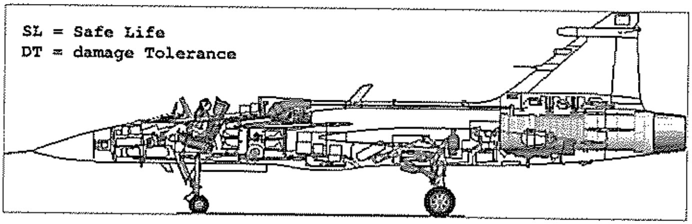


Figure 31. JAS 39 Gripen full scale test rig

Full-Scale Test Programme - Mechanical Systems



Flight control system
- servo actuators (SL+DT)
- pedal housing (SL+DT)
- control stick assembly (SL+DT)
- leading edge flap control system

Landing gear system
- nose and main landing gear (SL)
- actuators (SL)
- wheels and brakes (SL)

Escape and oxygen system
- pressure vessel (SL)

Hydraulic system
- tubes and fittings (SL)
- pumps (SL)
- valve units (SL)
- accumulators (SL)

Secondary power systems
- auxiliary power unit (SL)
- air turbine starter (SL)
- aircraft gear box (SL)
- power transmission shaft (SL)

Fuel system

Environmental system
- reduce and shut off valve (SL)
- heat exchangers (SL)
- engine bleed systems (SL)

Gun and armament install.
- Gun deflector (SL+DT)
- Gun fwd attachment (SL+DT)
- weapon pylons (SL)

Figure 32. JAS 39 Gripen fatigue and damage tolerance test programme of mechanical systems components

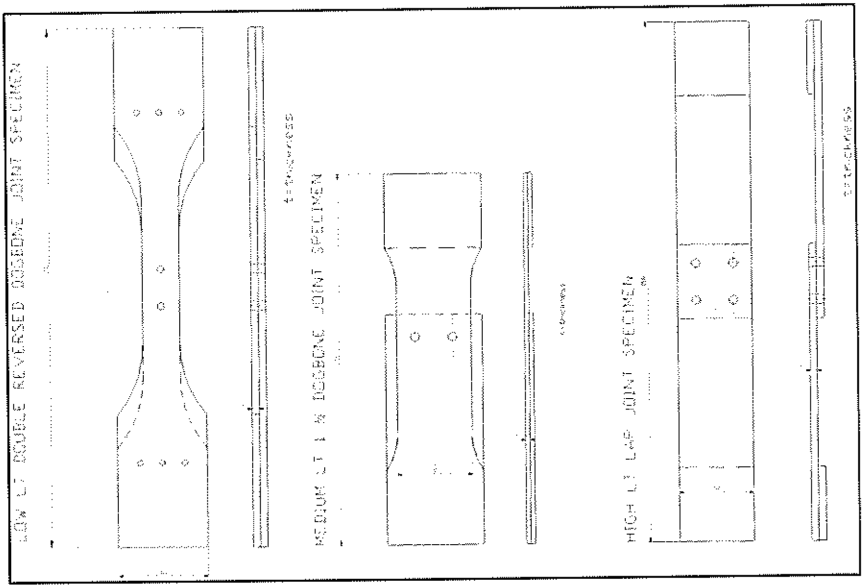


Figure 33. Low, medium and high Load Transfer (LT) test specimens in terms of the double reversed dogbone, 1 1/2 dogbone and lap joints

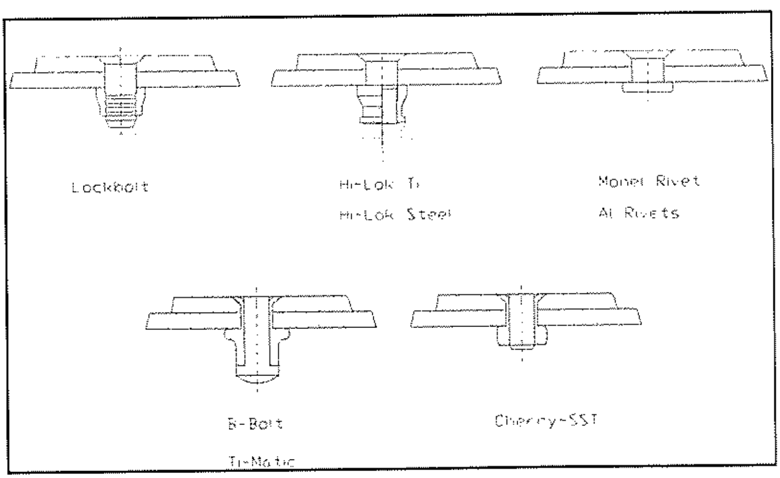


Figure 34. Fastener systems used. Five different main configurations are shown where subgroups are formed based on fastener material, fastener fit and fastener clamping force

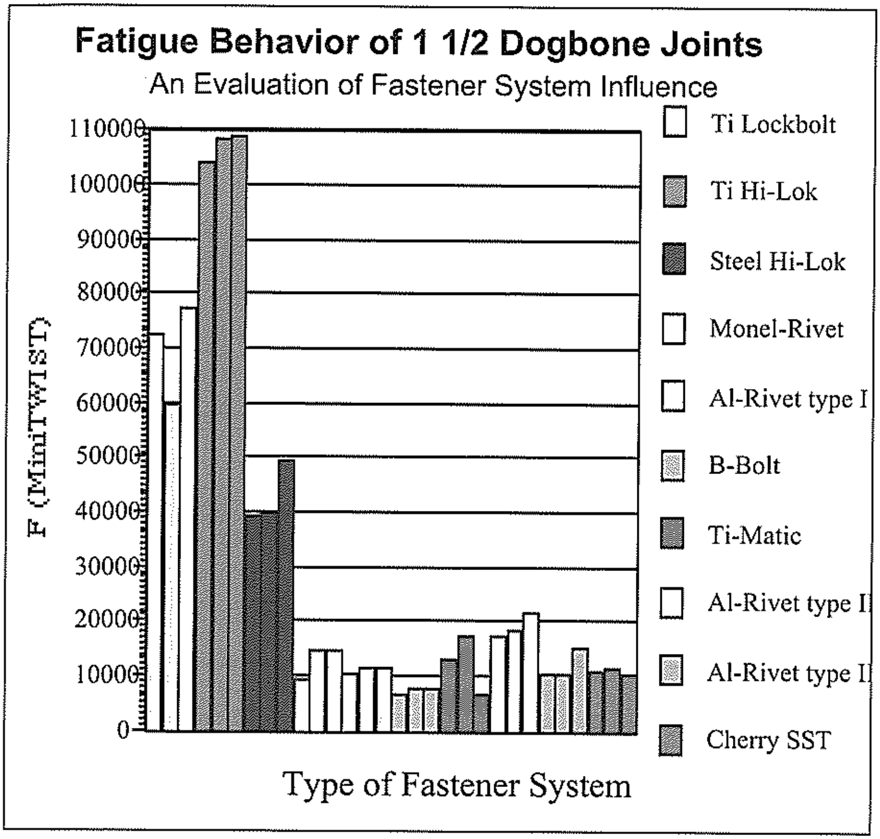


Figure 35. Fatigue test results of 1 1/2 dogbone joints subjected to miniTWIST flight-simulation loading

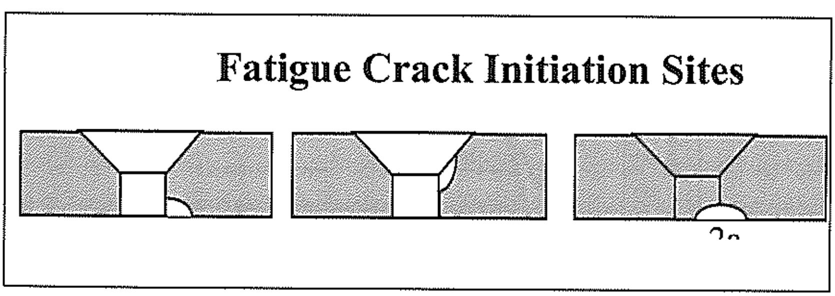


Figure 36. Fractographically observed fatigue crack initiation sites in 1 1/2 dogbone joint specimens

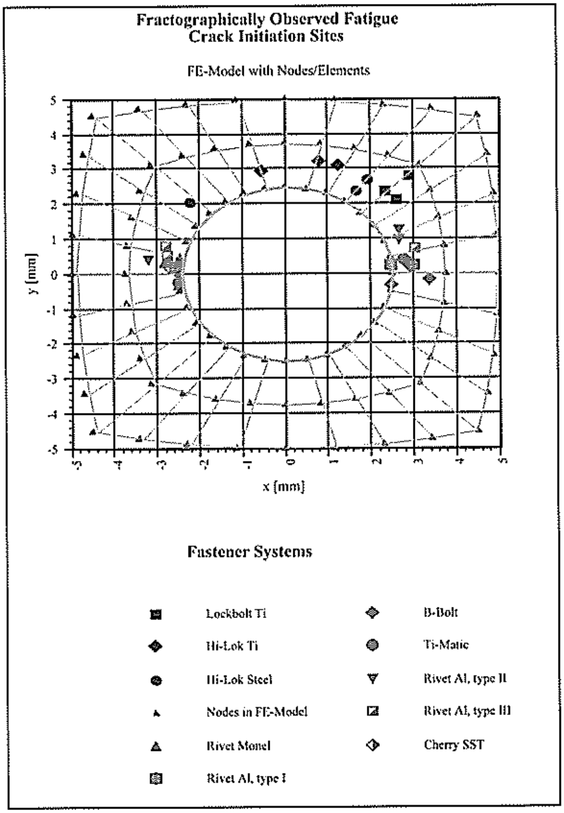


Figure 37. Fractographically observed fatigue crack initiation sites mapped on FE-model

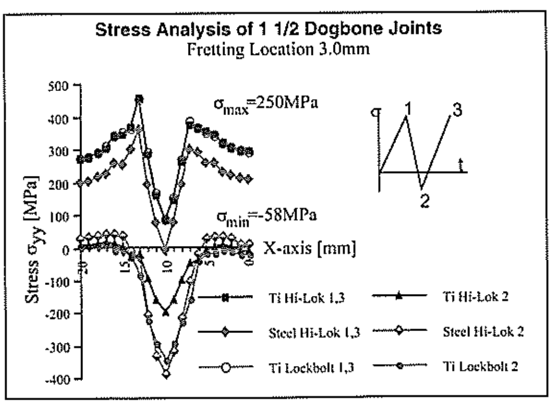


Figure 38. Computed stress distributions ahead of the fastener at the contact interference for maximum and minimum stress in MiniTWIST

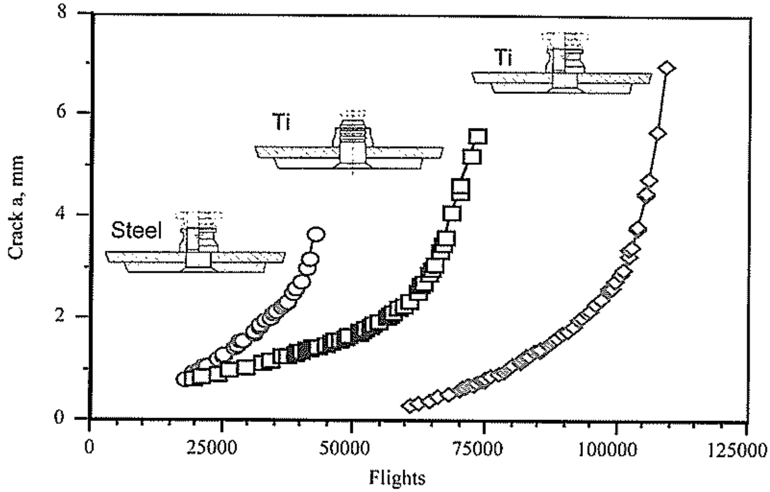


Figure 39. Fatigue mean crack growth curves of Ti Hi-Lok, Steel Hi-Lok and Ti Lockbolt specimens. Results obtained by SEM observations

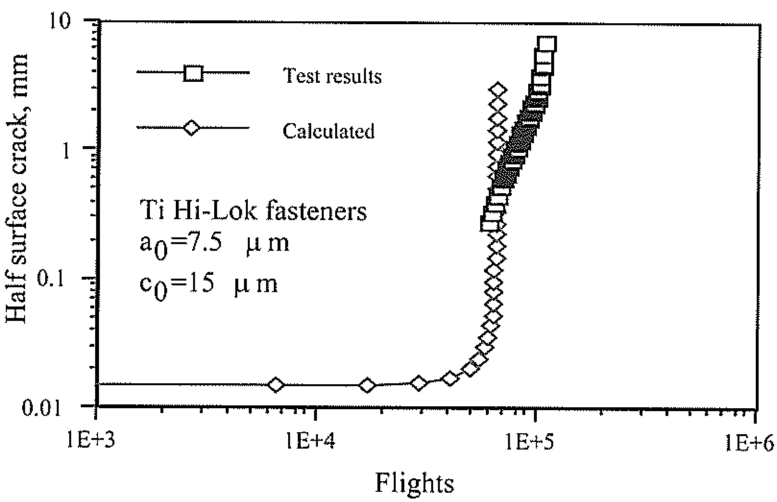


Figure 40. Fatigue crack growth predictions based on the weight function technique for the Ti Hi-Lok fastener systems compared to experimental results

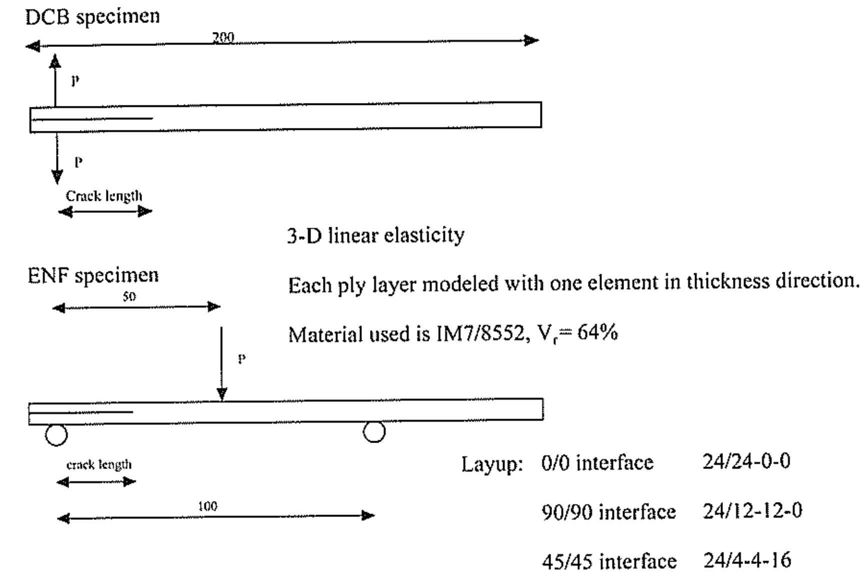


Figure 41. DCB- and ENF-specimens. Geometries, material, and layups

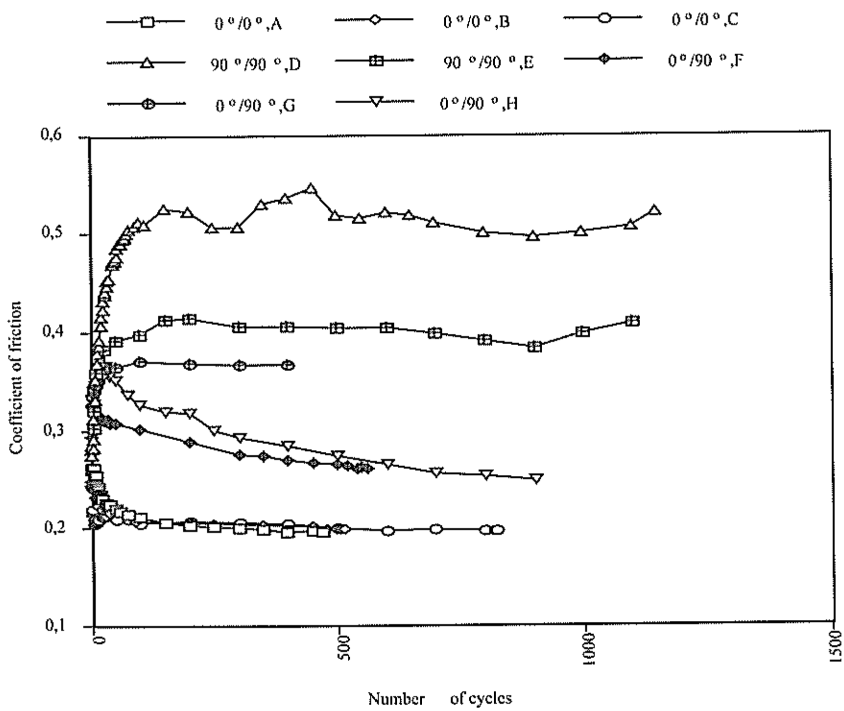


Figure 42. Change in coefficient of friction under cyclic loading for different surface combinations

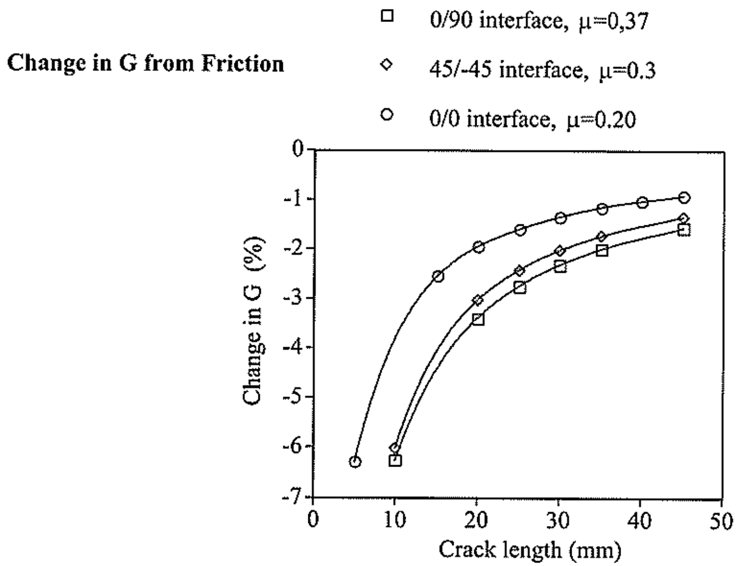


Figure 43. Change of total strain energy release rate in ENF-specimens for various interfaces and crack lengths

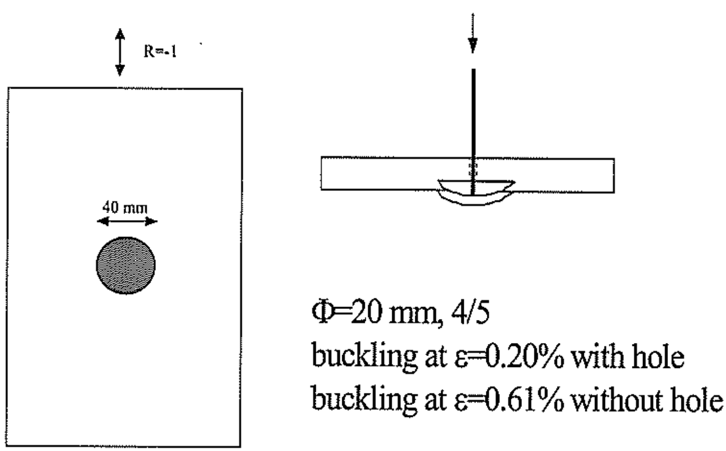


Figure 44. Experimental test to study the effect of teflon layer adhesion on local buckling loads

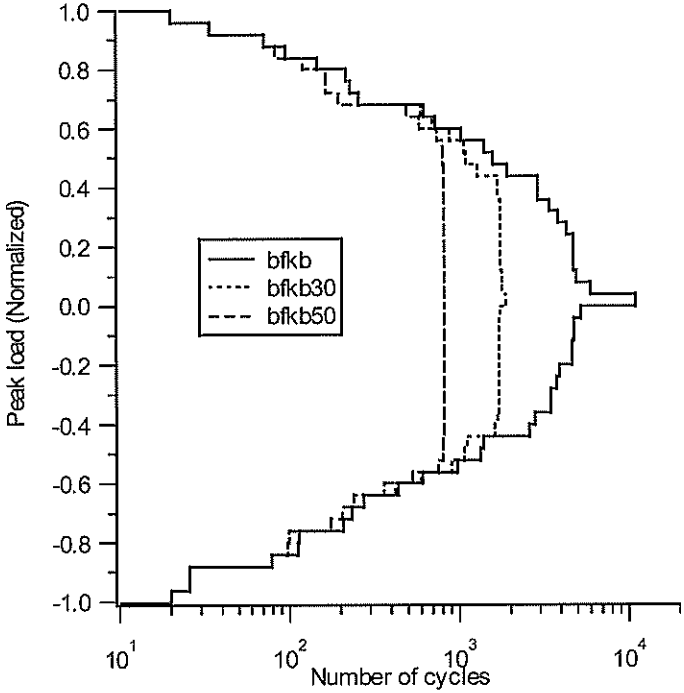


Figure 45. Vertical fin spectrum with different elimination levels shown

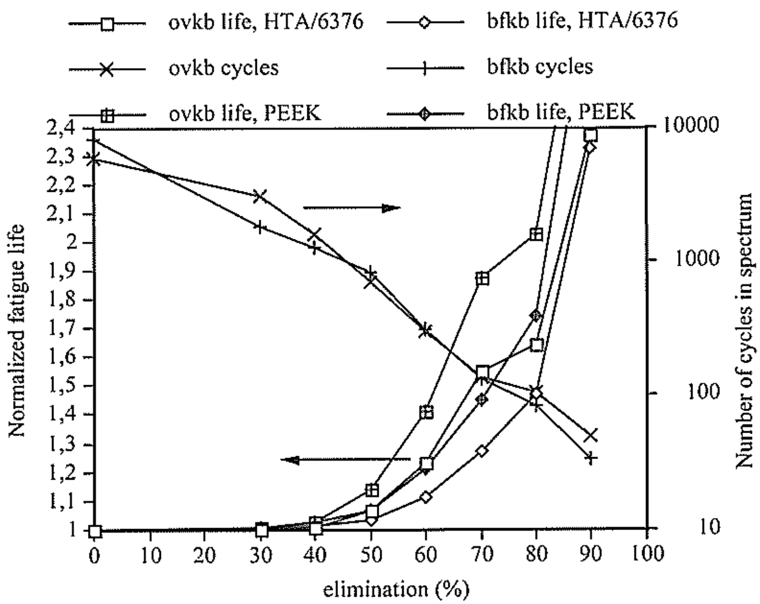


Figure 46. No. of cycles in load spectra after elimination and predicted related fatigue lives

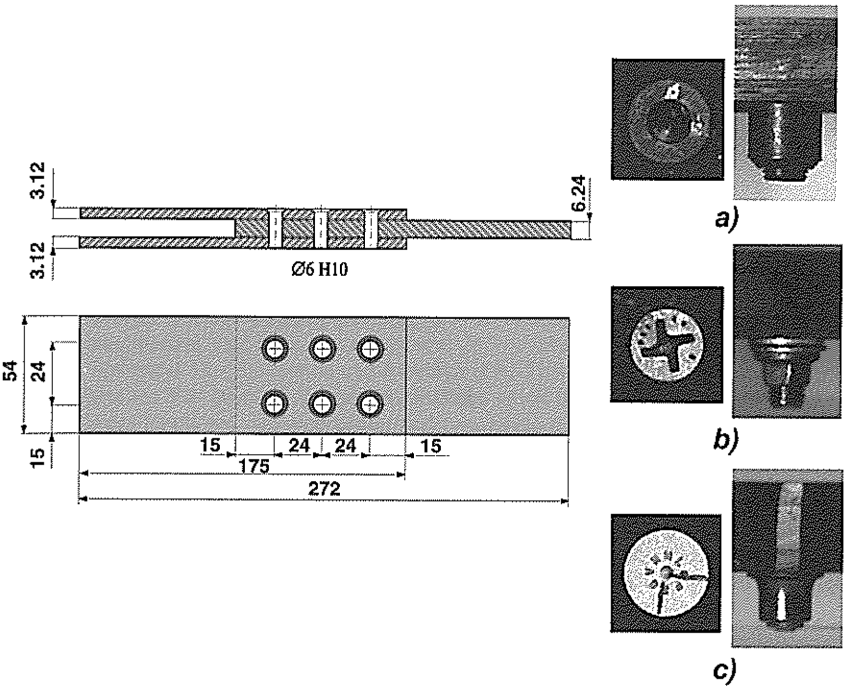


Figure 47. Specimen and fastener configurations: (a) composite bolt; (b) titanium Torque-set bolt; c) titanium Huck-comp fastener

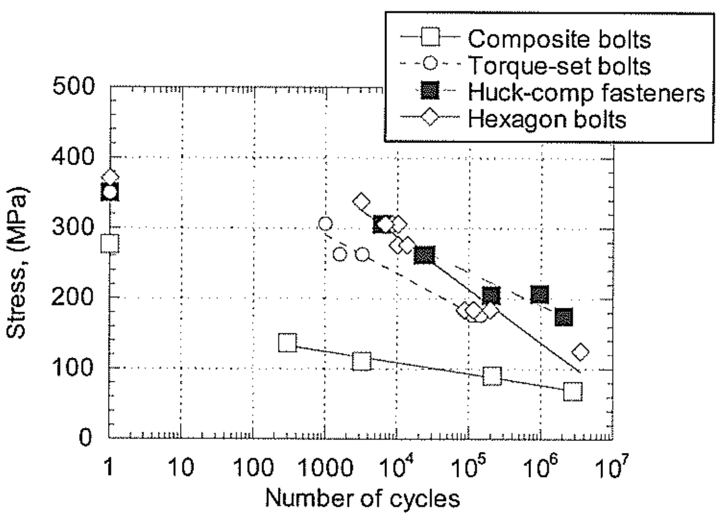


Figure 48. Fatigue life results for specimens shown in Fig. 47