FATIGUE AND DAMAGE TOLERANCE WORK DURING THE AIRCRAFT DESIGN PROCESS
10th Plantema Memorial Lecture

L. Jarfall*

Stress office routine procedures for the handling of fatigue life and damage tolerance work during the design process are reviewed and discussed. Four main tasks are identified: Handling of spectra, Planning of tests, Fatigue life and Crack growth predictions. Examples of procedures and experiences are taken from an actual fighter aircraft project.

INTRODUCTION

During the design rush period of a new aircraft project, the stress office needs established and rational routine procedures to handle the daily fatigue work. This is an attempt to review and discuss methods which already are, or may be considered as routine procedures.

The four dominating activities within the fatigue and fracture mechanics group during the design of a new aircraft are (Fig 1):

- Handling of loads spectra
- Planning of fatigue tests
- Prediction of fatigue lives
- Prediction of crack growth rates

In a lecture like this one it comes very natural also to ask, what progress has been done in these four fields during the 34 years that ICAF has existed. Making the historical description very brief:

* SAAB-SCANIA Aircraft Division, S-58188 LINKÖPING SWEDEN

- The greatest theoretically supported methodology development is the birth of fracture mechanics.

- The greatest change in the way of working undoubtly comes from the employment of computers.

These two changes had taken place already 20 years ago. In 1965, linear damage rule calculations as well as its equivalent correspondence in the way of crack growth calculations were done essentially the same way, as it is being done today. The linear damage rule calculations were already computerized - The crack growth calculations in most cases not yet.

Focusing on rational routine procedures automatically puts the computerization in the foreground. Although it has in all cases speeded up the work, it has not always improved the quality. Looking at the four main tasks listed above, the employment of the computer has allowed a great progress in the handling of loads spectra and thereby also in the planning of spectrum fatigue tests. When it comes to the prediction of fatigue life and fatigue crack growth, however, the shortcomings of our models for material behavior has hampered the progress. The improvements in prediction accuracy is more due to the increase in volume of realistic spectrum fatigue data furnishing empirical correction factors to our prediction models. And to be fair - also here the employment of computers, now for the control of testing machines, has considerably facilitated the progress. Most of our experience from variable amplitude testing has been gained during the last 20 years.

Although outside the subject of this paper, one must not forget the tremendous progress in the methods for calculation of internal loads and local stresses provided by means of modern finite element techniques. The improved quality of the stress analysis, here again provided by means of the computerization, has of course had a strong impact on the reliability of the fatigue and crack growth predictions.

HANDLING OF LOADS SPECTRA

A necessary prerequisite for all fatigue and crack growth work is access to local load spectra for any part of the structure. In the design work on the two latest projects at SAAB-SCANIA a methodology, that we refer to as the "Global Spectrum Approach", has been employed with a great success.

The global spectrum approach implies, that the aircraft design spectrum is defined as a sequence of instantaneous load cases, defined for the aircraft as a whole and expressed in terms of configuration, fuel distribution, thrust, speed, altitude, accelerations and several other flight mechanics parameters. The sequence is defined by the operational analysis group. For every load case, aerodynamic loads are calculated by the loads group and applied to a finite element model of the complete aircraft. Fig 2. Inertia loads are calculated from a mass distribution tied to the finite element model. When the loads specialists have fed all the required input data into the analysis system, the stressman can obtain local load sequences for any member of the finite element model. He can also define a local stress by means of factoring and superposition of several element loads and thus obtain a local stress history.

Programmed into the same analysis system is also a facility to perform a rain-flow-count (RFC) analysis of the local load or stress history and produce a matrix (RFC-matrix) of associated peaks and troughs. Fig 3. The RFC-matrix is the input to plotting programs for peak and trough distributions and range distributions. The RFC-matrix also is the input to programs for fatigue analysis and crack growth analysis, which are programs designed to communicate with the global spectrum program. For cases like this one, with very long irregular sequences stored in a computer, it is very important to use a counting algorithm that in a relevant way combines load cases into cycles as does the rain-flow-count, the hysteresis loop count or the range pair range count, which all produce the same result. Manipulation based on engineering judgement is out of the question when handling such a big quantity of data, that has to be treated in this case.

This global spectrum approach has proven itself to be an indispensable tool for the fatigue and fracture mechanics work, as well as for the planning of component tests and the full scale fatigue test. Any engineer of the stress office can without any waiting time obtain a design spectrum for any part of the structure, that may be described in terms of internal forces of the finite element model. It will also allow convenient modifications to any local spectrum, whenever feed back from flight loads surveys or changes in projected usage may so require. The drawback with this approach is, that it imposes a very heavy work load on the loads department early during a project. While doing this big job, nothing in the way of local spectra can be produced with this software package. Urgent needs for local spectra during the very early design work have to be satisfied by means of traditional working methods.

Specifics of the SF-340 global spectrum

After having defined 1540 unique load cases and their frequencies of occurrence, a sequence of 3600 different flights was generated by means of a "drawing without replacement procedure". The deterministic details of a flight by flight sequence were carefully realized. The sequencing of flights with different gross weight, different altitude, different gust environment, speed for flap extension and retraction as well as intensity and mode of touch down during landing were drawn at random. The sequence of really random events within a segment, as for instance the mixture of brakings, bumps and turns during taxi, was of course randomized. Events of low freguency, such as towing between taxi in and taxi out and such as crew training involving engine shut down, were distributed at regular intervals among the 3600 flights. The actual service load history was assumed to consist of successive repetitions of the 3600 flights sequence. The sequence contained a total of 986212 load cases.

Specifics of the JAS39 global spectrum

Based on pilot training programs following 57 deterministic subsequences and their expected frequencies of occurrence were defined

12	different taxi load sequences	with	57	unique cases
32	different flight load sequences	with	563	unique cases
13	different landing load sequences	with	532	unique cases
57			1152	

The 57 subsequences were combined to a sequence covering 313 complete missions (taxi out, flying, landing, taxi in) containing a total of 406874 load cases. The actual service load history was assumed to consist of successive repetitions of the 313 flights sequence.

PLANNING OF FATIGUE TESTS

The planning of a fatigue test with a component or with the complete aircraft structure involves

- Identification of the most significant load cases

- Design of a testing arrangement that produces a good simulation of the most significant load cases

- Elimination of insignificant load cases to shorten the testing time

By adding a few facilities to the global spectrum software, a very powerful program for test planning has been obtained.

The first requirement, when trying to design a test set up for a component test, is that one knows, which load cases are the most significant from fatigue viewpoint. Again, by means of a rain flow count procedure, variations and frequencies of occurrence for any internal load of the structure can be listed, including code numbers identifying the two load cases forming each variation (range). This has been obtained by retaining the load case identification codes all the way through the RFC-analysis. Knowing which load cases, that should be simulated with maximum accuracy, a test set up of acceptable complexity can be designed. Figs 5 and 6.

Insignificant load cases, that are either intermediate (neither peak nor trough) or contribute to insignificant RFC-defined load ranges are eliminated by means of a multi-conditional rule: The test planner defines a number of external/internal loads, for which conditions of intermediateness as well conditions of range limitations shall be applied. The result of these operations is a testing sequence with all the deterministic characteristics maintained, but considerably shorter than the original (design) sequence. Figs 7 and 8.

As a decisive check of a proposed testing arrangement, a finite element model is made of the test set up with its specific boundary conditions. Loads according to the intended testing sequence (after elimination of insignificant load cases) are then applied at the loading actuator points, and distributions of ranges, peaks and troughs for the relevant internal forces are calculated and plotted. A direct comparision with the corresponding distributions for the complete aircraft, when subjected to the original sequence, then furnishes the base for a decision to accept or reject the proposed test set up. A good design of the testing arrangement as well as the testing sequence is characterized by a good agreement between the range distributions. Fig 4. The distributions of peaks and troughs may turn out rather different because of the elimination of small ranges.

Example: Planning of a test with an engine mount bracket ant its attachment to the fuselage structure. Fig 5.

The four corners of the primary test region is defined by the points 3 through 6. A first axiomatic requirement is that the engine reaction forces P_1 and P_2 must be applied with a high degree of accuracy. An exact simulation of the by-pass forces in the fuselage structure, however, will require a rather complicated testing arrangement. In order to be able to make a rational simplification by replacing controlled boundary conditions by (uncontrolled) reactions, it is necessary to identify which types of load cases that form the most significant load ranges (magnitudes and frequencies). This can be done by listing the upper portions of the range distributions of the fuselage frame bending moments M_3 .... M_6. The listing shows the ranges ΔM in order of magnitude as well as their number of occurrences and most important also the identification numbers of the associated peak and trough values. Knowing which types of load cases that are most significant for the considered region, also the most important internal distributions of forces are known. This knowledge then forms the basis for the design of the testing arrangement. Fig 6.

As a next step load cases, which are intermediate for the external loads P_1 and P_2 as well as for the most important internal loads M_3 .... M_6 are eliminated from the sequence. Fig 7. At the same time also such load cases which contribute only to insignificant local load variations for the same external/internal loads are eliminated, if the program user specifies range limits ΔP_{01}, ΔP_{02}, ΔM_{03} ... ΔM_{06}. This implies that load cases are eliminated only if the elimination condition is satisfied for all the specified external/internal loads. The result of this work is a testing sequence of acceptable length. Fig 8.

CRACK GROWTH CRITERIA vs TOTAL LIFE CRITERIA

During the last 10 years there has been a marked change over from fatigue life requirements to fatigue crack growth requirements even for what used to be referred to as safe life structures. Largely responsible for this change are the initiators to the U.S. Air Force damage tolerance specification Mil-A-83444.

This changed approach, no doubt, has several advantages from safety viewpoint

- A push for sound material selection criteria striving for slow fatigue crack growth and high fracture toughness

- A stop to fatigue improvement methods only influencing the crack initiation process and thereby resulting in structures, which are very intolerant to manufacturing defects and damages during service.

- A push for improved quality control because of the quantifyable relationship between quality control and life requirement.

- A quantifyable relationship between inspectability and life requirement

- A (theoretical) possibility to extend the service life by means of a quantified inspection only.

The quantitative consequences of the approach, however, contains several aspects, which should be subjected to questioning and further consideration:

- Our capability to predict rate of crack growth (particularly for short cracks, which is most important), path of cracking (continuing damage) and residual strength, leaves much to be desired in terms of accuracy.

- For the normally specified initial flaw sizes, a crack growth criterion with a factor of 2 on life is a much stronger requirement than a traditional fatigue life requirement with a factor of 4 on the total life. This of course imposes a weight penalty compared with earlier design requirements. Keeping in mind, that the standard design procedures, which should be neutral or conservative, very often contain a considerable amount of conservatism, the weight penalty caused by the stronger safety requirement gets even more enhanced.

- Complience with a damage tolerance requirement like Mil-A-83444 in its very detail may lead to costly and questionable inspection procedures. Example: Requirement to inspect for 1 mm (tightly closed) corner cracks in every fastener hole of a critical part. Is that really meaningful?

- There is no simple and straightforward way to verify complience with crack growth criteria for a complete structure, the way its total fatigue life may be verified by the full scale fatigue test. The verification can only be done for selected points, where artifical flaws are made, or points which are simulated by means of specific specimens. The weakest link in the verification procedure is the selection of points to be verified. The most critical point may easily be missed.

- Looking specifically at the Mil-A-83444 specification, the potential weight savings from a fail-safe design are normally refrained from by designers of fighter aircraft, because of the big effort needed in the way of analysis and testing to prove complience with the fail safe design requirement. Thus, the present specification favours the slow crack growth requirement.

Considering the simple routine procedures, there is a very important difference between crack growth prediction and fatigue life prediction. While the traditional linear damage rule acc to Palmgren-Miner normally produces results on the unsafe side, its exact equivalence applied to crack growth normally yields results on the safe side. Ref (1). Assume the stressman has access only to a minimum of data, which always means constant amplitude data only, he can be rather confident to have made a safe design, if his simple linear summation calculation tells, that a detectable flaw will not cause failure within the double service life. In the early design process, overdesign is preferred to underdesign.

THE APPLICATION OF DAMAGE TOLERANCE REQUIREMENTS AT THE SAAB-SCANIA MILITARY DIVISION

The Swedish Defence Material Administration has required, that critical parts of the new fighter JAS39 "Gripen" shall comply with damage tolerance requirements, which in most aspects are identical with Mil-A-83444. The same requirement had been discussed for an earlier project, that never was realized. (The B3LA-project, which was very similar to the Aeritalia-Aermacchi-Embraer aircraft AMX). Between the B3LA-project and the JAS39-project an exercise evaluation of the damage tolerance qualities of the JA37 "Viggen" fin was done in cooperation with the Aeronautical Research Institute of Sweden (FFA).

The exercise with the Viggen fin structure put our attention to following problems

- The selection of critical points for damage tolerance analysis did not agree well with the critical points acc to a full scale fin fatigue test. (The most fatigue critical point was missed!)
  Conclusion: The full scale fatigue test is indispensable. It cannot be replaced by a damage tolerance verification test.

- Crack growth predictions using non retardation models in all cases turned out to be far on the safe side.

- Stress intensity factor solutions often required a great and very costly effort.

Damage Tolerance Application in the Design Process

The damage tolerance application to the Gripen structure is limited to critical parts. A part, that if it fails, alone may cause the loss of an aircraft is classified as a critical part. This definition means, that e.g. the complete control system from the control stick and the pedals to the servo actuators and control surface attachments must comply with the damage tolerance requirements. The main goal is a safe life design, i.e. a slow crack growth structure not requiring any inspection during its full life. (A small compact aircraft will require very expensive disassembly in order to allow a meaningful inspection). Up to now, i.e. before the weight savings campaign, all parts except the wing and fin attachment tension bolts fulfill the requirement for a crack growth period exceeding the double design life. For parts complying with the "full life without inspection" requirement, the total fatigue life requirement automatically is fulfilled and no checking is needed.

Very important and also very intriguing is the question of how much weight penalty that is imposed by the damage tolerance requirement. Unfortunately no figure has jet been estimated, but it is much less than originally expected. The reason is very simple. The critical parts, which are the only ones of our aircraft that must comply with the damage tolerance requirement, in most cases are designed with extra safety factors against ultimate failure. This is because they very often are either fittings joining components together (wing to fuselage, control surface to wing) or parts of the control system or both. The ultimate load requirement for a fitting is 1.15 x 1.50 x limit load = 1.725 x limit load. For the attachment of a hydraulically operated control surface the ultimate load requirement is 1.20 x 1.15 x 1.50 = 2.07 x limit load. Particularly when the ratio between ultimate load and limit load is as high as 2.07 the damage tolerance requirement may not cause any weight penalty at all, if the material selection is right. Example: After changing alloy and heat treatment of the nose wing pivot from Ti6Al4V in the STOA condition to Ti6Al4V ELI in the RA condition, the weight penalty due to the damage tolerance requirement was eliminated.

For certain parts of the control system there is not only a high ultimate design safety factor, but there may also be a significant difference between the limit load and the largest load of the service spectrum, resulting in negligible fatigue crack growth. As a consequence, we had to introduce a new design requirement, saying that a component containing the largest flaw, that may be overlooked by quality control during the manufacturing, must be able to sustain 150 % limit load. This is a stronger requirement than the ordinary residual strength requirement of 120 % limit load after two life times.

Damage Tolerance Verification Testing

If the damage tolerance verification rules require a demonstration, that growing cracks exhibit a lower than specified growth rate (The Slow Crack Growth category) following procedure appears to be the only rational approach:

a. Verify calculated local loads and stresses by means of strain gauge measurements on a full scale structure.

b. Design specific specimens, carefully simulating a very limited structural region including the single point to be verified.

c. Demonstrate fatigue crack growth due to realistic spectrum loading.

Not having access to a complete structure for strain measurements (a) as early as desired, much of the unbias, being so important in a verification procedure, may easily be lost during the specimen design, (b). Therefore we have agreed with our customer, the Swedish Defence Material Administration, to use following approach:

a. Critical parts shall be manufactured and installed exactly in accordance with the A/C drawings on rather comprehensive component test specimens.

b. A first test will be run as a pure fatique test for at least four life times subjecting the specimen to a realistic flight by flight spectrum loading.

c. A second identical specimen will then be damaged by artifical flaws at several locations before assembly. Some of those flaws will be impossible to inspect without dissassembly.

d. The second specimen will be subjected to a minimum of two life times of realistic flight by flight spectrum loading.

e. The testing will be concluded by a residual strength test to 120 % limit load.

f. Those initially flawed locations, which do not cause a residual strength failure or do not exhibit any excessive crack growth from the artificial flaws, will be accepted as damage tolerant.

This procedure is supported by an experimental study on the behavior of artificial flaws (comparing various manufacturing methods) when subjected to realistic spectrum loading.

Examples of the largest component tests (one for fatigue plus one for damage tolerance):

- Wing to Fuselage Joint, consisting of half a fuselage frame incl. some outer skin, a wing box with upper and lower attachment brackets.

- Fin to Fuselage Joint, consisting of parts of three fuselage frames incl outer skin, a fin box with left and right attachment brackets, the fitting for the aft engine mount.

- Left side Engine Support Structure consisting of portions of four fuselage frames, outer skin, longitudinal stiffening, engine room wall and engine support bracket.

- Right side Engine Support Structure (similar to above).

- Canopy with hinges and locks

The smaller component tests involve several control surface attachment brackets incl their nearest surroundings as well as several actuator attachment brackets with their nearest surroundings.

The final verification, that no fatigue critical point has been overlooked in the damage tolerance verification process, will be obtained from the full scale fatigue test with a complete early series production version of the structure.

ON THE ACCURACY OF CRACK GROWTH PREDICTION

An analysis to show complience with a crack growth criterion, as required by a damage tolerance specification, involves following steps

a. Calculation of the stress intensity factor
b. Calculation of the critical crack size
c. Compilation of crack growth data
d. Adoption of a model for prediction of fatigue crack growth.

Below these four tasks will be discussed in separate.

The Stress Intensity Factor

In the design analysis of a slow crack growth structure it is most important to make correct estimates for the early portion of the crack growth process, because it is there the life is. In most cases this implies that maximum accuracy is needed for small corner cracks. In a recent review, ref (2), a large number of stress intensity factor solutions for corner cracks were compared. Looking at fig 9 one may wonder: Which function is used in our computer program?

The problem, that we just considered, was the really easy one: The open (unloaded) hole. Most holes in real structure are filled by a fastener or by a bushing and often subjected to a bearing load. Deformation restrictions due to a member inside the hole as well as the contact pressure distribution in the bore of the hole will add more sources of error to the problem of stress intensity factor calculation. Also the pressure from a mating member on the flat surface around the hole has a considerable influence on the growth of a small corner crack. Acc to the results of fig 11 and fig 12, the fastener clamping force increased the crack growth life by factors of 5 and 8 respectively. Ref (3). Expressed in terms of stress intensity factor corrections, the Falstaff test results yield a correction factor of 0.6 while the Gauss results yield a correction factor of 0.55 to 0.65.

Recent test results with single shear double row four column joints, having single a = 1 mm artificial corner flaws at the faying surface under the countersink for Hi-Lok shear heads, indicate that small corner flaws do not have any strong influence on the fatigue life. In this case the friction between the faying surfaces was low due to paint primer and wet assembly with a sealant.

Critical Crack Length

Safe estimates of the residual strength and thereby also of the critical crack length may be obtained from the following two failure criteria:

a. Safe domain for nominal net stress at the crack tip region, $\sigma_{net} < R_p$ = Material yield strength.

b. Safe domain for crack tip stress intensity factor, $K < K_{Ic}$ = Plain strain fracture toughness.

For small size and particularly for thin gauge members, the approaches a and b above yield too much of conservatism, so other approaches needing other materials data must be used. In fig 13 and fig 14, a number of different approaches have been applied to two examples. Ref (4). In both examples the structural members are 4 mm thick and 50 mm wide straps assumed to be cut from 150 mm thick AA 7010-T73651 plate. For that material, K_{Ic}-data had been determined using 25 mm thick CT-specimens. Also R-curve measurements had been made using 4 mm thick CT-specimens (w=125).

Based on the results from the tests mentioned above, residual strength vs crack length was calculated using three more approaches besides a and b, as defined above. Ref (5).

c. $K = K_a$ = Apparent fracture toughness
$= \beta(a_o)\ \sigma_c\sqrt{\pi a_o}$ with a_o = initial crack length
σ_c = fracture stress

d. $K = K_q$ = Stress intensity at onset of crack growth

e. $$\left.\begin{aligned} K &= K_r \\ dK/da &= dK_r/da \end{aligned}\right\} = \text{The R-curve tangency criterion}$$

The two examples of fig 13 and fig 14 clearly show, that there is a risk for big errors in the calculation of the critical crack length, if one does not have access to applicable test results. Looking for published data, one also gets a feeling that we, in the field of aircraft engineering, have spent too much of our resources on K_{Ic}-data, and as a consequence are lacking data, which are more applicable to our designs. May we in the future expect to get something more than K_{Ic}-data from our material suppliers?

Compilation of Crack Growth Data

For every new project, normally also some new alloys are introduced. This means, that the volume of available data during the early project is very limited. A great problem is, that most of the data, due to supply difficulties, often has to be determined from a single batch. According to our experience, the scatter within a batch is very small compared to scatter between batches - particularly so in the case of spectrum loading. Therefore our new policy is not to make more than two identical tests from the same batch. With this approach, more capacity can be spent on studies on the influence from various parameters. Furthermore, we do not find it meaningfull to measure threshold values because of high cost as well as doubtful application, when predicting spectrum fatigue crack growth. We repeatedly have recorded kinks in the plots of crack growth rate vs stress intensity variation - both in constant amplitude tests and in spectrum tests. Although we do not understand them and can not observe any fractographic changes, we acknowledge them, when we use our data.

Recently we have been caught by a couple of surprises. Even if we have not yet had time to fully certify the results, they may be worth some attention:

- Constant amplitude crack growth data for heavy plate and forging of the alloy AA 7010-T736 were found to be identical. Spectrum fatigue crack growth data, however, turned out considerably different.

- Constant amplitude tests with CC-specimens have been duplicated by making CT-specimens from the remains of the CC-specimens. For AA 7010 heavy plate as well as for Ti6Al4V forging, the CC-specimens have exhibited 2 to 4 times higher crack growth rate. This corresponds to a factor of 1.3 on the stress intensity factor.

In both cases referred above, a possible explanation may be the presence of residual stresses, which will be relieved when cutting the notch in a CT-specimen contrary to a CC-specimen.
Do we really have experimental proof, that CC-and CT-specimens do produce identical data?

Models for fatigue crack growth prediction

Several problems when trying to predict fatigue crack growth are the same as those previously encountered when trying to predict total fatigue life:

- Most of the available data are constant amplitude data

- Aircraft structures are subjected to spectrum loading

- The volume of spectrum test data is still rather limited

- The utilization of spectrum test data is often hampered by irrational documentation methods.

- In both cases the influence of peak load truncation presents one of the greatest sources of prediction error.

Both in the case of total life prediction and in the case of fatigue crack growth prediction, we are still missing good models, that are capable to describe material memory. Evaluating various interaction models for spectrum crack growth prediction based on constant amplitude data, one finds that those methods (as well as those for total life prediction) often yield inconservative results. The old equivalent to the Palmgren - Miner law, the linear summation method, on the other side, appears always to yield conservative results. A significant improvement in prediction accuracy presently seems to be achievable only with access to relevant spectrum fatigue data. Based on this experience, following two methods for fatigue crack growth prediction have been adopted for the JAS 39 Gripen project work.

Always conservative method:

The linear summation of constant amplitude data without any interaction. The sequence effect and the material memory function is replaced by the employment of the rain flow count (RFC) method to provide the stress cycle idenfification.

Growth rate when the crack length = a:

$$\left\{\frac{da}{df}\left(\frac{mm}{flight}\right)\right\}_a = \frac{1}{f} \int_1^{n_f} \frac{da}{dn} (\Delta K_i)_a$$

with n_f = the number of ranges, ΔK_i, acc to RFC in a sequence of f flights

$$\text{Crack Growth Life, } F(a_o \rightarrow a) = \int_{a_o}^{a} \left(\frac{da}{df}\right)^{-1} da$$

Method without hidden margins:

Whenever relevant spectrum fatigue crack growth data is available, and particularly when oversizing can not be accepted due to high weight penalty, spectrum test data is used.

$$\left\{\frac{da}{df}\left(\frac{mm}{flight}\right)\right\}_a = D_f \cdot \hat{n}_f \cdot \hat{K}^k; \qquad K = K(a);$$

with D_f = a constant obtained from a test with a spectrum of the same category

$\hat{n}_f$ = spectrum intensity based on RFC. Ref (6).

$\hat{K}$ = spectrum peak stress intensity factor = $\hat{\sigma}\beta\sqrt{\pi a}$ with $\hat{\sigma}$ = spectrum peak stress

$$\text{Crack Growth Life, } F(a_o \rightarrow a) = \int_{a_o}^{a} \left(\frac{da}{df}\right)^{-1} da$$

$$= (D_f \, \hat{n}_f \, \hat{\sigma}^k)^{-1} \cdot I_g$$

For further details: See the Appendix and ref (6)

Our latest comparison of different crack growth prediction methods was applied to a testing program covering 5 different spectra with CC-specimens from the same batch of AA 7010 heavy plate. Ref (7). One of those test series is shown in fig 15 together with predicted crack growth rates acc to ref (8). Predicted growth rates were based on constant amplitude (da/dn) data obtained with CT-specimens from another batch. At a first glance one gets the impression that the top curve, i.e. "Linear summation based on RFC" is too much on the safe side. This is, however, an incorrect conclusion. Considering a realistic example that barely fulfils a crack growth life of 2 life times, one finds, that the first life time is spent between $\hat{K}$ = 400 and 550 and the second life time between $\hat{K}$ = 550 and 750. In the region $\hat{K}$ < 750, all the methods exept the linear summation predict unsafe results. Fig 16.

Among the test series from ref (7), that were compared with analysis methods in ref (8), are two spectra that are different only with regard to the peak load truncation. Comparing the same five methods as in fig 15, the ratio (measured rate)/(predicted rate) at $\hat{K}$ = 600 turns out as follows:

<u>Method</u>	Full peak	Truncated peak
Linear summation	0,6	0,7
RMS acc to Barsom	1,2	1,7
de Koning	1,7	1,7
Generalized Willenborg	2,1	2,3
Original Willenborg	5,9	4,4

Here again, only the linear summation yields results on the safe side. Although on the unsafe side, the crack closure approach acc to de Koning appears promising just because it took care of the peak truncation effect.

A very important deficiency of <u>all</u> the methods based on constant amplitude data are their inability to produce a correct response to changes in stress level, i.e. they produce an incorrect estimate of

"the slope exponent", $k = d(\frac{da}{df}) / d\hat{K}$

In fig 15 all the calculated curves have typical slopes of k = 3.5, while the spectrum test results yield an average k = 2.0. Refer also to the Appendix with the figures A2 and A3.

One should not disconsider differences in computing cost when comparing various calculation methods. The linear summation method, the RMS/Barsom method and the spectrum data method acc to ref (6) have computing costs which are only 0.001 of the computing costs for most interaction models, which require cycle by cycle calculation.

Assume that we only have access to one series of the four fighter wing types of spectra of ref (7). What prediction accuracy would we achieve using the method of ref (6)?

a. Linearization
A straight line fitting the test results of fig 15 in the range $400 < \hat{K} < 800$ and disconsidering all threshold phenomena, will for the same spectrum predict a crack growth coinciding with the curve "TEST" of fig 16. The linearization itself thus is a negligible approximation.

b. Truncation effects
Considering the same example of peak load truncation, as was discussed above, the method of ref (6) would be in error by a factor 1.27. Conservative or unconservative depends on which spectrum that supplies the data and which spectrum is predicted.

c. Maximum prediction error
Picking out the two most extreme results among the four fighter wing types of spectra from ref (7) and using one result to predict the other one, will produce a maximum prediction error of 32 %.

Considering that the number of specimens required for each spectrum is the same as for each R-value in constant amplitude loading, and that the testing time for spectrum crack growth data is much shorter than for constant amplitude data, spectrum test results must be rated as a best buy. And if it is true, that materials rate differently in spectrum loading than in constant amplitude loading, there is really no way to get around the spectrum testing.

FINAL REMARKS

Great progress has been made in the handling of loads spectra and in the technique for spectrum testing. This is mainly due to the employment of computers.

The development of fracture mechanics has provided methods for safer designs, although the prediction accuracy in several aspects (e.g. for very short cracks) is as bad as for total fatigue life. The most important feature of fracture mechanics is the quantifyable relationship between Quality Control/Service Inspections on one side and Life Requirement/Residual Strength on the other side.

As little as materials data and stress concentration factors are sufficient for fatigue life prediction, materials data and handbook stress intensity factor solutions are sufficient for crack growth predictions. Realistic testing with real detail design configurations (e.g. joints) will continue to require a big effort.

More testing with systematic variations of spectra has to be done. Spectrum testing in the past has too often been "single shots". With the global spectrum approach systematic variations in local spectra for different details of the same aircraft are evident. From engineering viewpoint the number of spectrum shapes is not infinite. Having gained more experience from spectrum testing, we will in the future be able to specify a limited number of variations, that will allow interpolations providing predictions of acceptable accuracy.

More attention should also be paid to making data development tests at realistic stress levels. Too often a big portion of the data is produced in the high stress/short life region of little practical interest.

The real problem with peak load truncation, is not to know how to account for differences in truncation, but to know which peak load truncation to design for. Ref (6). For transport aircraft, known statistics of the gust environment forms a basis for a probabilistic design criterion. For a fighter aircraft, however, our knowledge about the statistics of the extreme loads is very insufficient for the formulation of a design criterion.

REFERENCES

(1) D Schütz, H. Oppermann: "Kritische Bewertung von rechnerischen Verfahren zur Lebensdauervohersage" LBF Bericht Nr TB-171 (1984). Vorträge des vierten LBF Kolloquiums am 31. Januar und 1. Februar 1984 in Darmstadt.

(2) B. Palmberg: "A Review of the Stress Intensity Factors for Quarter-Elliptical Corner Cracks at Circular Holes". The Aeronautical Research Institute of Sweden. Informal Report FFAP-H-719.

(3) B. Palmberg: "Spectrum Fatigue Testing of Simple Test Specimens Containing Artificial Flaws" FFA Technical Note HU-2108.

(4) H Ansell: "Verification of methods and data for calculation of residual strength". SAAB-SCANIA Testing Program TKH P-85021 (in Swedish).

(5) H Ansell: "Calculation of Residual Strength for a Plane or Biaxial Stress Condition. Determination of Fracture Toughness and R-curve Data for AA 7010-T73651 Heavy Plate" SAAB-SCANIA TKH R-3432 (in Swedish).

(6) L Jarfall: "Prediction of Fatigue Life and Crack Growth using Spectrum Test Data" SAAB-SCANIA TKH R-3379. See APPENDIX

(7) H Bjermert, B Palmberg: "Spectrum Fatigue Crack Growth of material AA 7010-T736" FFA TN 1984-64 (in Swedish).

(8) S Wikström: "Calculation of Crack Growth Rate using different Models to Account for Sequence Effects" SAAB-SCANIA TKH R-3417 (in Swedish).

(9) Anders Lundqvist: "A Rational Method for Crack Growth Prediction" SAAB-SCANIA TKH R-3401 (in Swedish)

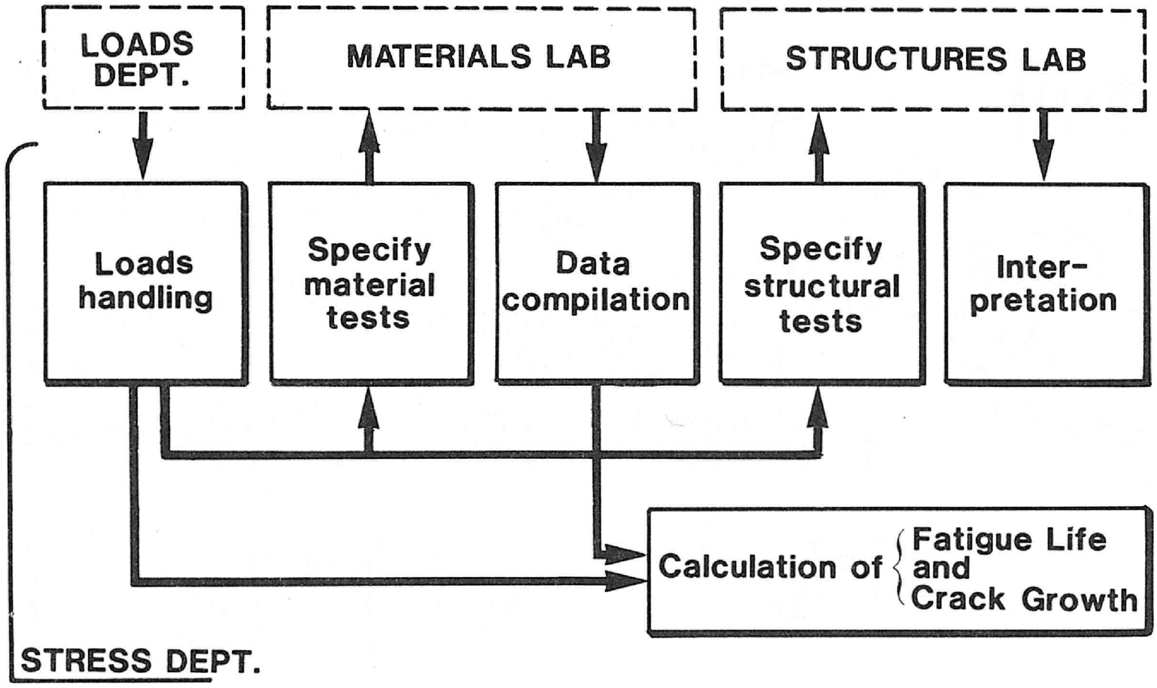


Fig. 1 Fatigue and fracture mechanics activities during the design phase

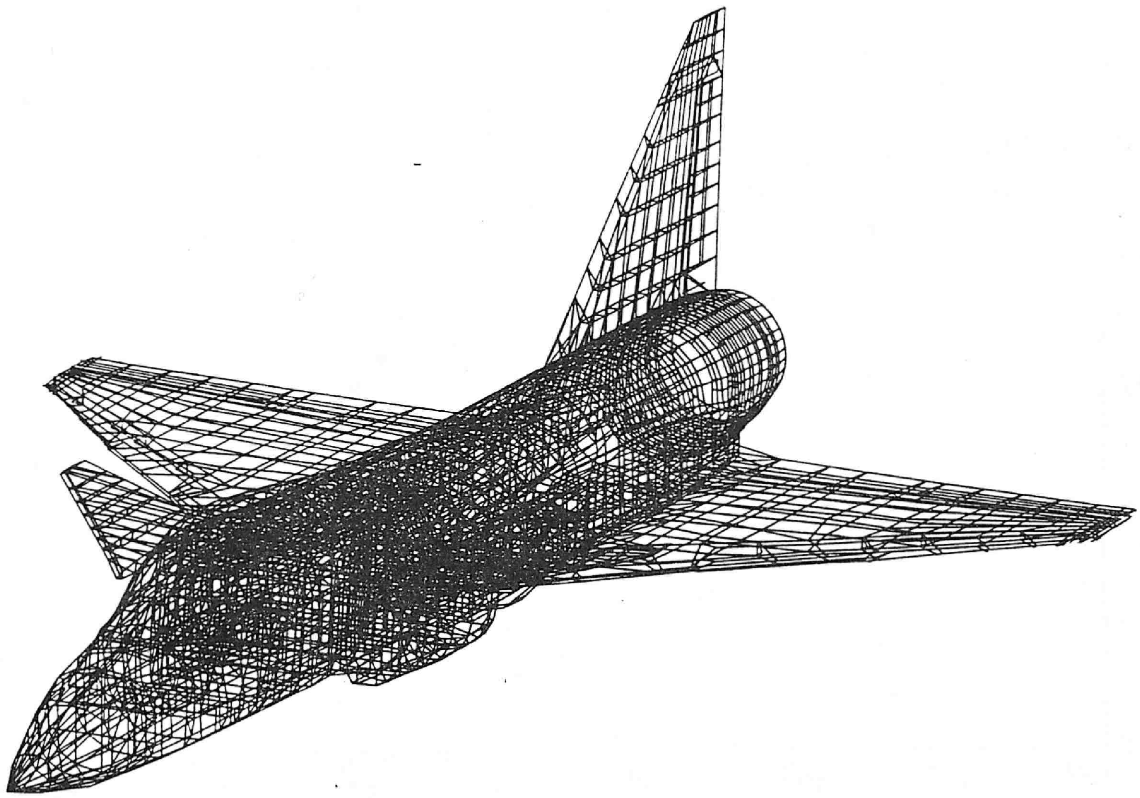


Fig. 2 Finite element model of the JAS 39 "Gripen" (almost 10^5 degrees of freedom)

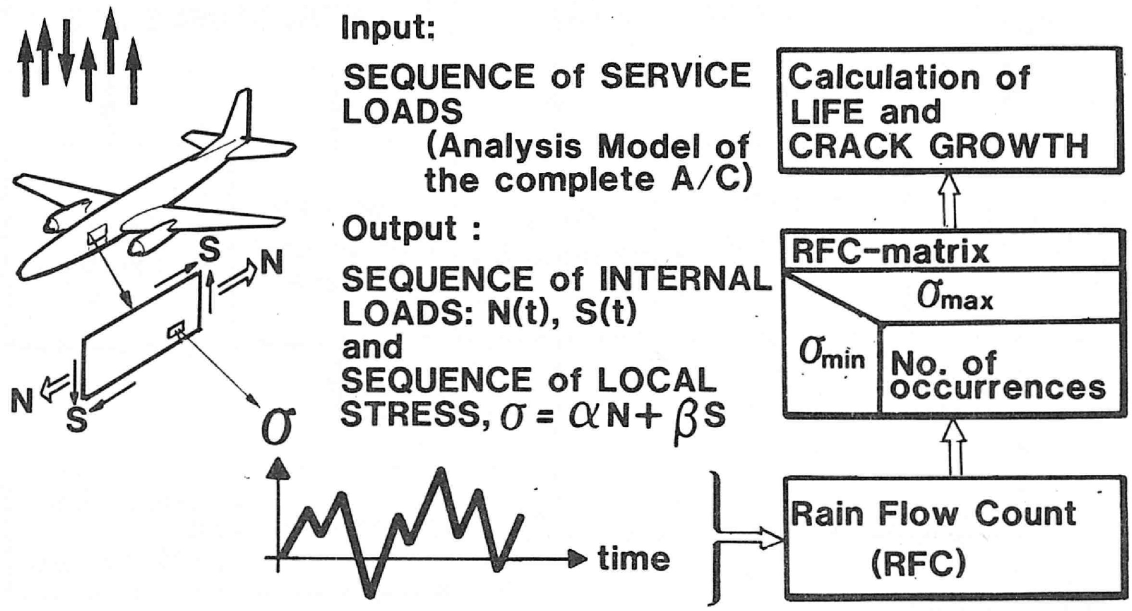


Fig. 3 The global spectrum approach

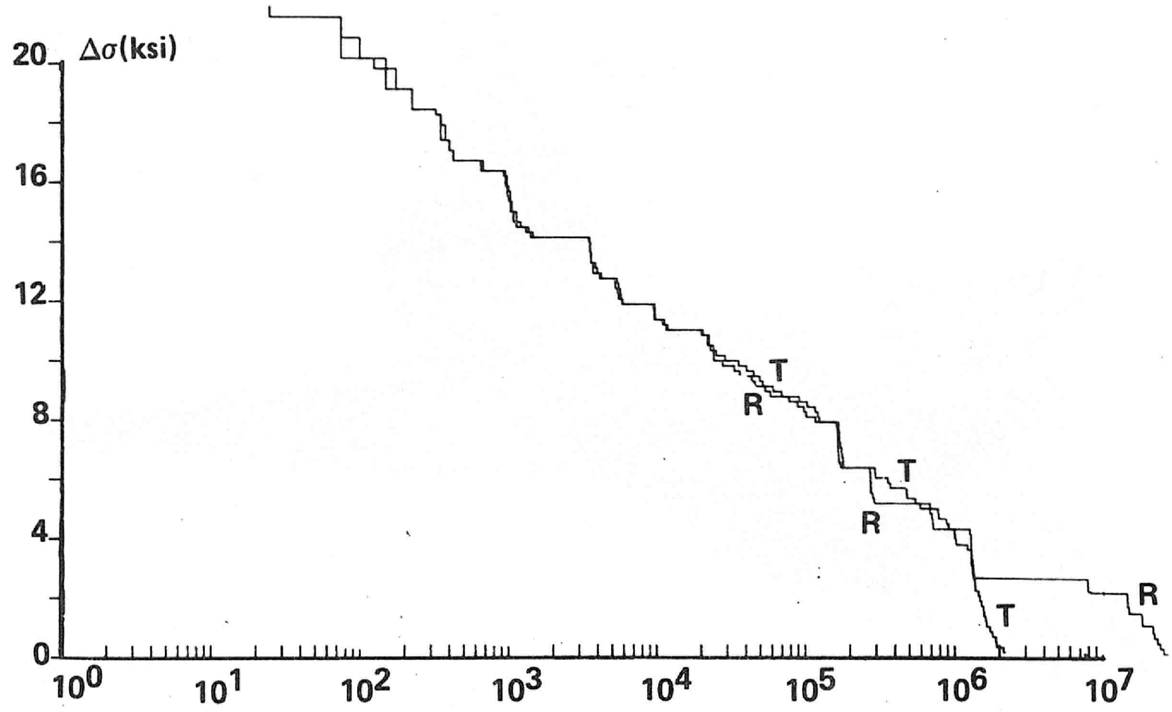


Fig. 4 Comparison of local stress range distributions (from RFC) of the testing sequence (T) and the reference sequence (R)

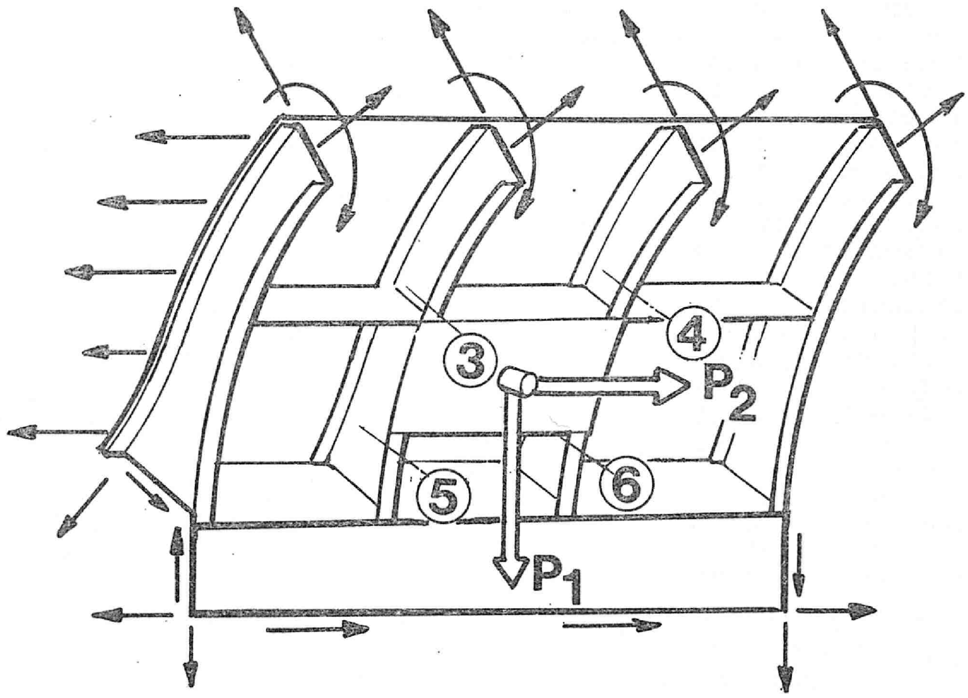


Fig. 5 Part of the real structure with external loads and boundary loads (very schematic)

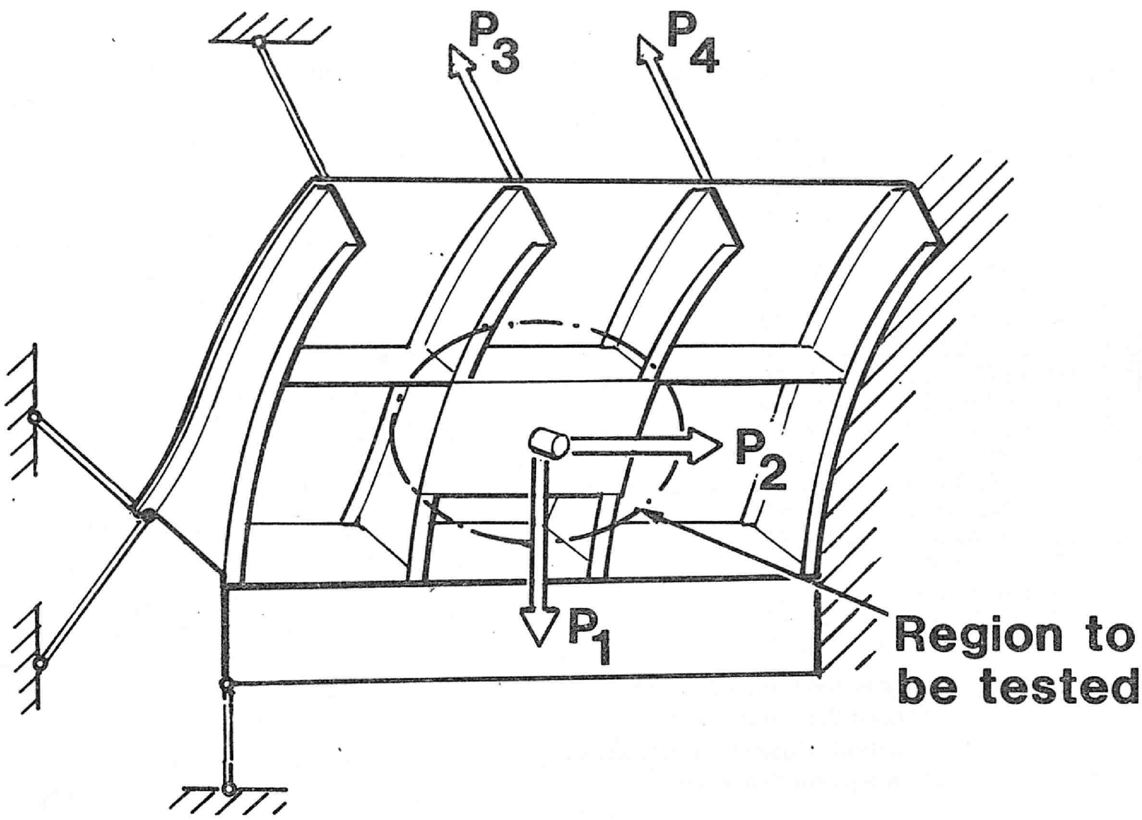
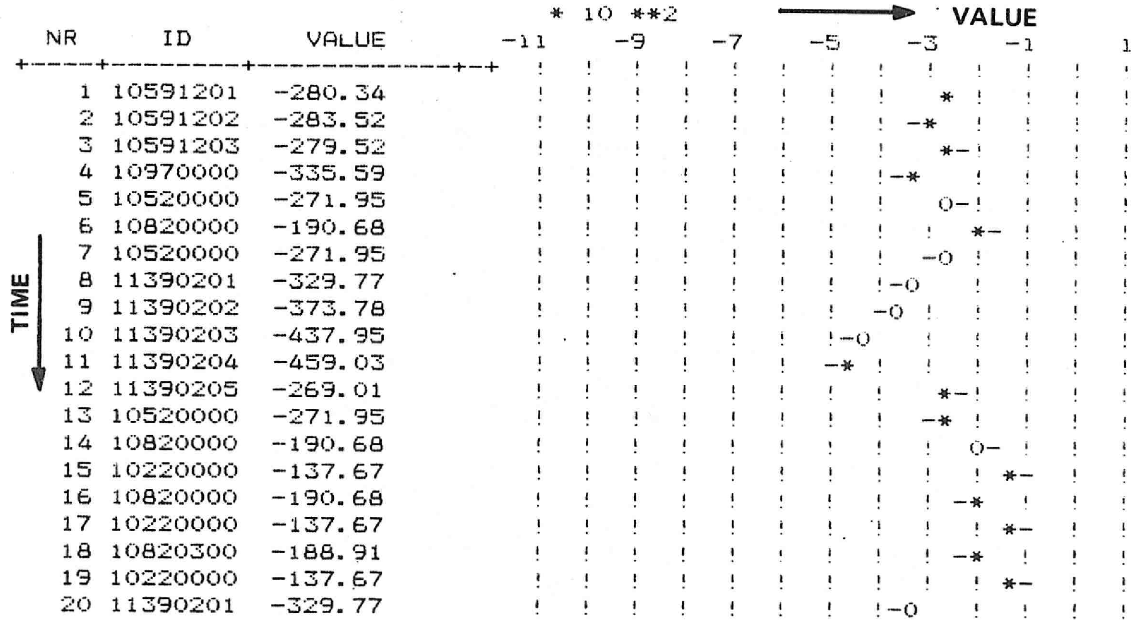


Fig. 6 Proposed test specimen with external loads and reactional supports

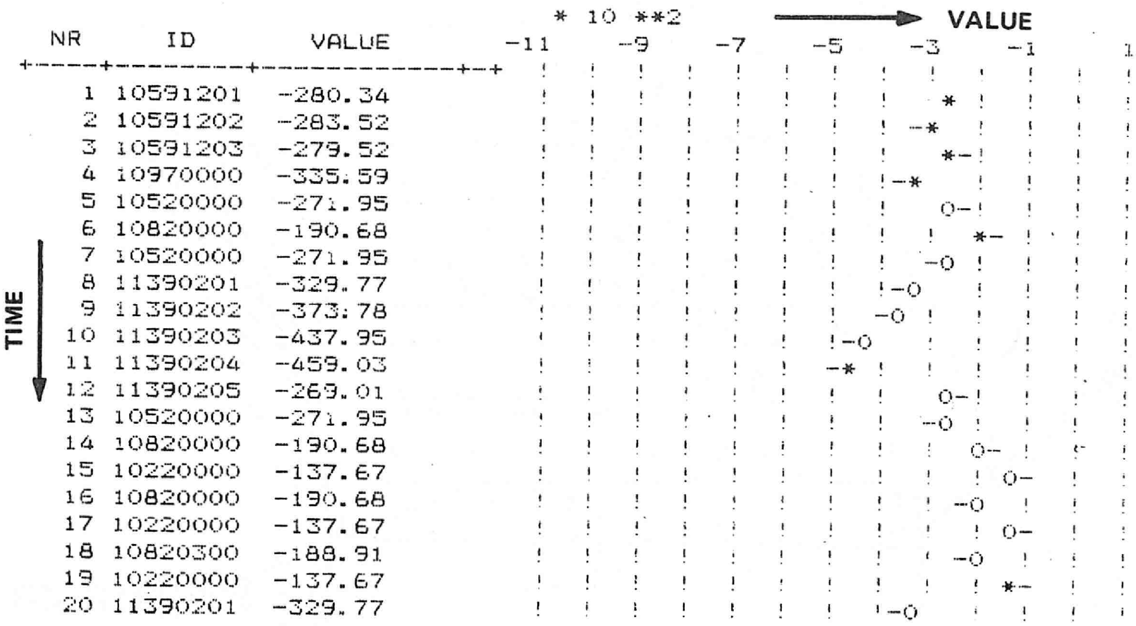
NR	ID	VALUE
1	10591201	-280.34
2	10591202	-283.52
3	10591203	-279.52
4	10970000	-335.59
5	10520000	-271.95
6	10820000	-190.68
7	10520000	-271.95
8	11390201	-329.77
9	11390202	-373.78
10	11390203	-437.95
11	11390204	-459.03
12	11390205	-269.01
13	10520000	-271.95
14	10820000	-190.68
15	10220000	-137.67
16	10820000	-190.68
17	10220000	-137.67
18	10820300	-188.91
19	10220000	-137.67
20	11390201	-329.77



Notations: * =Load Case to be retained
o =Load Case to be deleted
– Indicates location in relation to the preceeding value

Fig. 7 Elimination of intermediate load cases

NR	ID	VALUE
1	10591201	-280.34
2	10591202	-283.52
3	10591203	-279.52
4	10970000	-335.59
5	10520000	-271.95
6	10820000	-190.68
7	10520000	-271.95
8	11390201	-329.77
9	11390202	-373.78
10	11390203	-437.95
11	11390204	-459.03
12	11390205	-269.01
13	10520000	-271.95
14	10820000	-190.68
15	10220000	-137.67
16	10820000	-190.68
17	10220000	-137.67
18	10820300	-188.91
19	10220000	-137.67
20	11390201	-329.77



Notations: * =Load Case to be retained
o =Load Case to be deleted
– Indicates location in relation to the preceeding value

Fig. 8 Elimination of load cases forming load ranges below a specified value

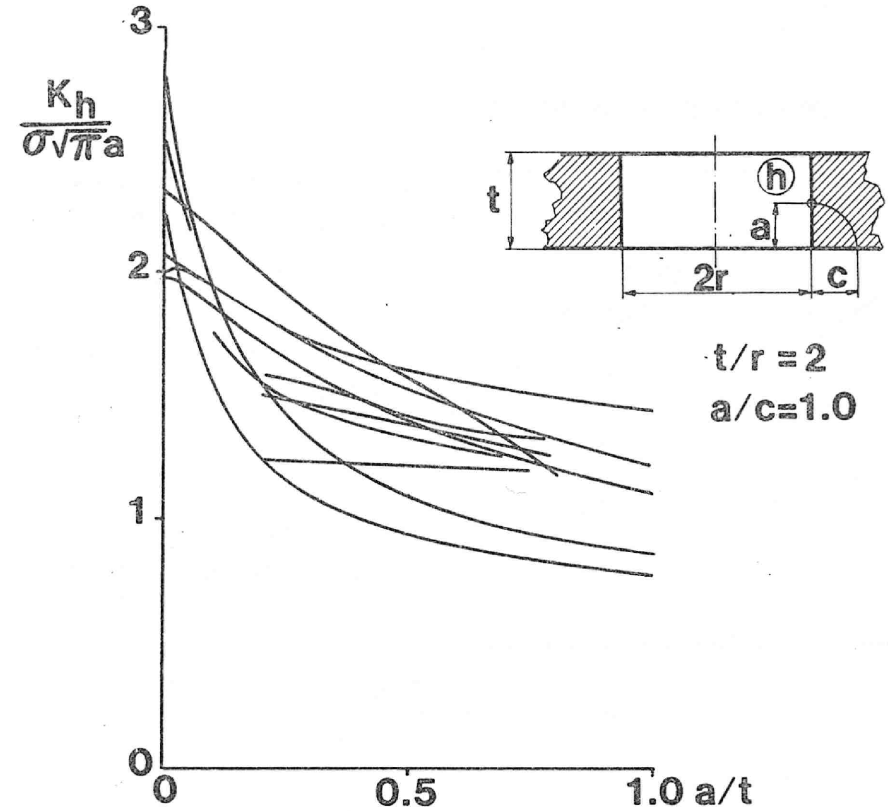


Fig. 9 Comparison of various stress intensity factor solutions for a corner crack. Ref (2)

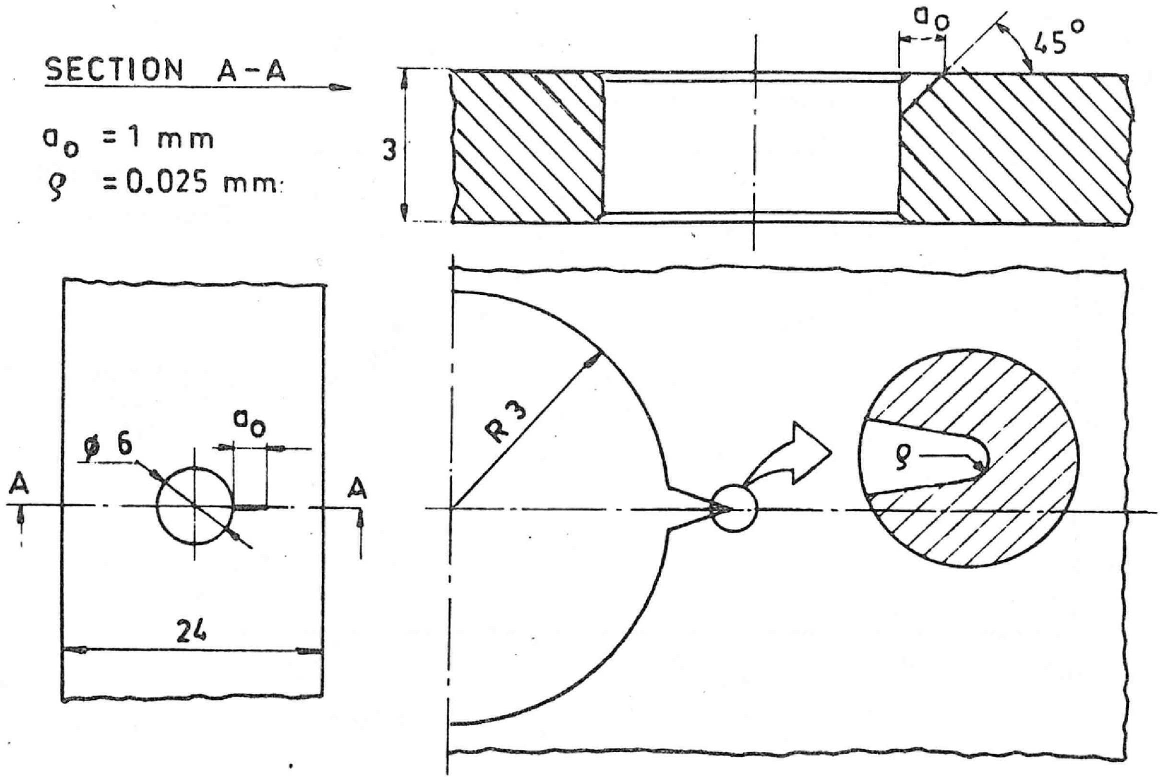


Fig. 10 Artificial corner defect geometry. Ref (3)

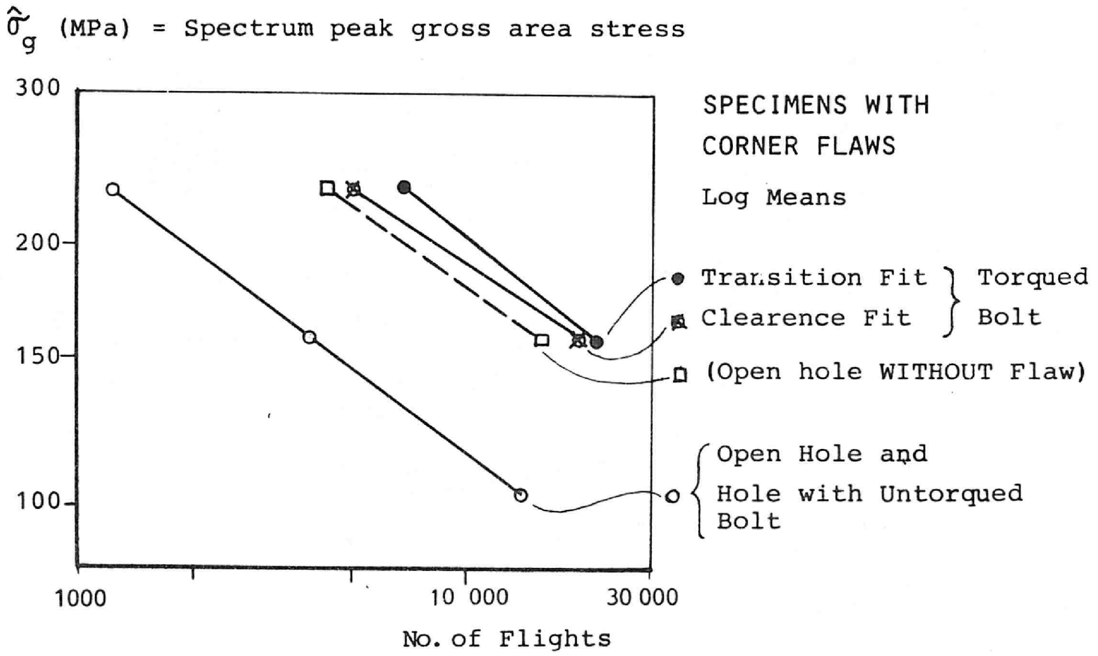


Fig. 11 Experimental fatigue lives with the FALSTAFF sequence. Ref (3)

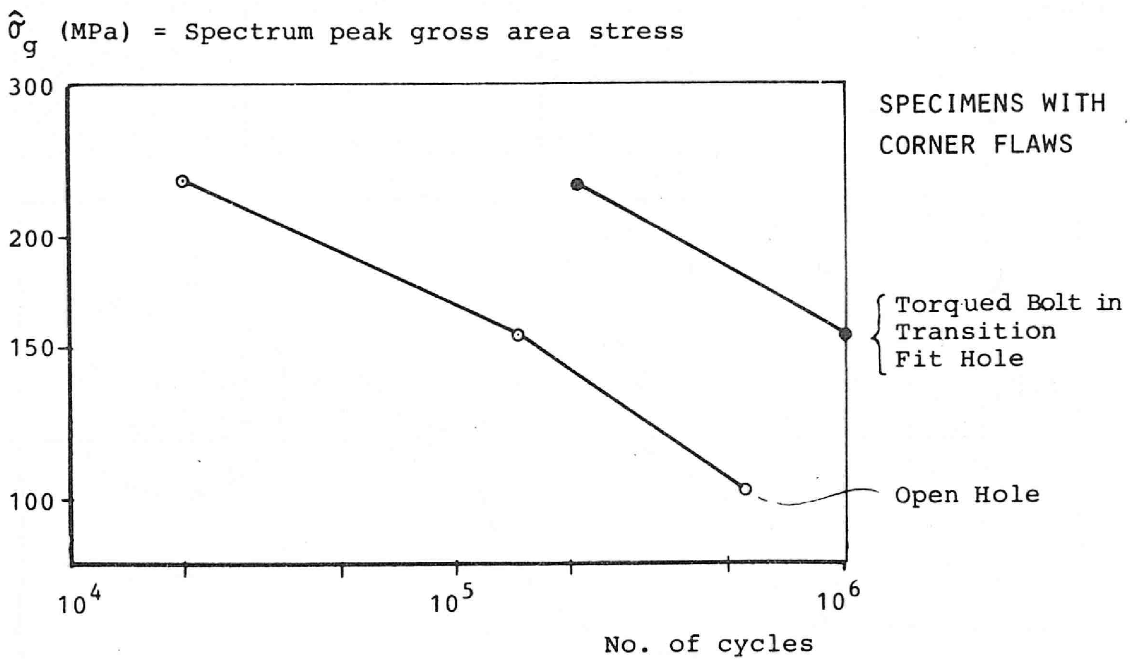


Fig. 12 Experimental fatique lives with a symmetric Gauss distribution. Ref (3)

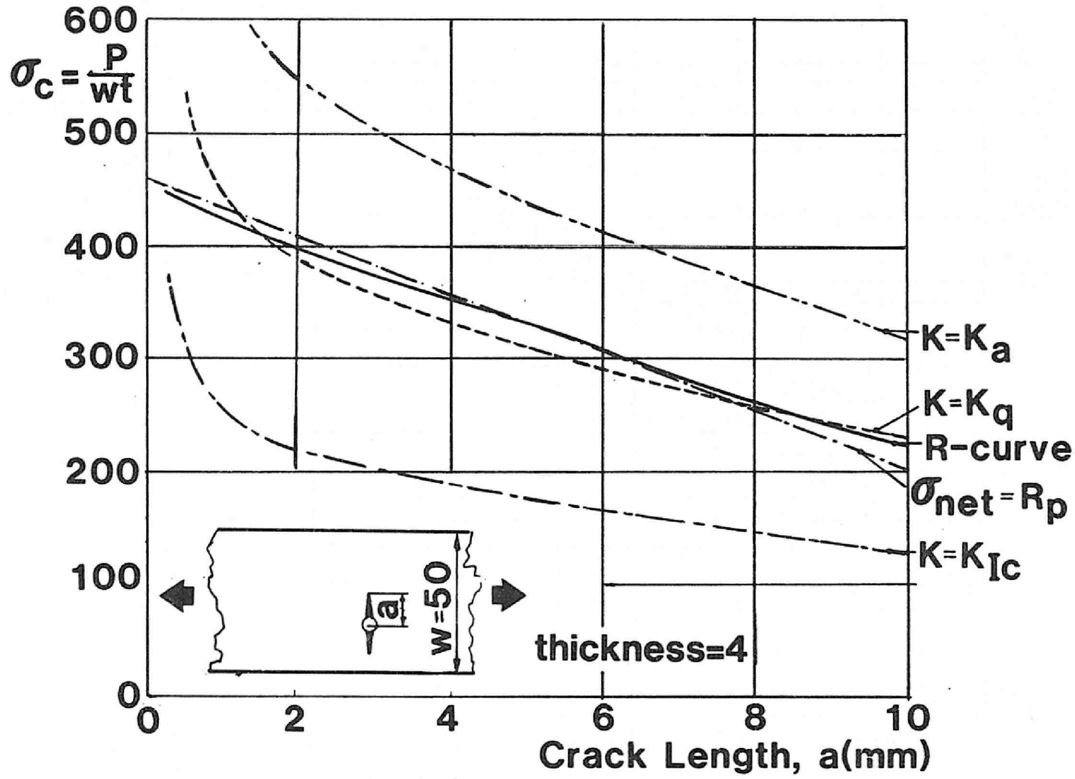


Fig. 13 Calculated residual strength using different approaches. Ref (4)

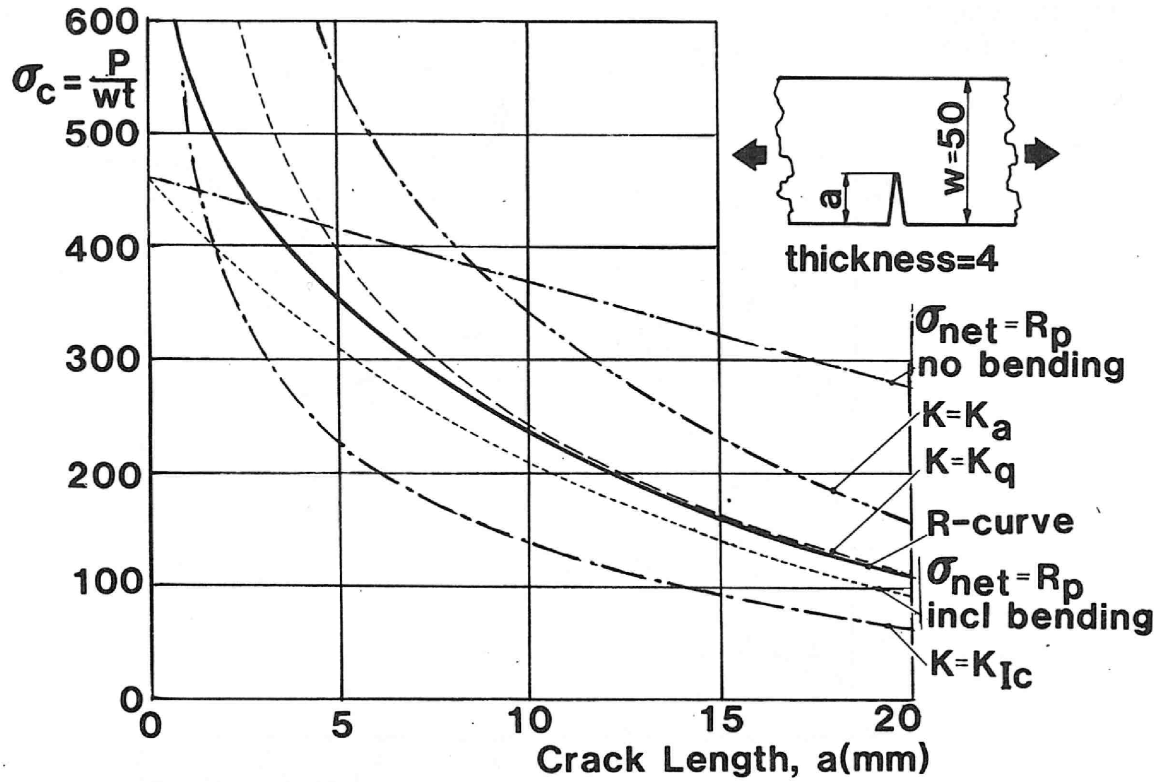


Fig. 14 Calculated residual strength using different approaches. Ref (4)

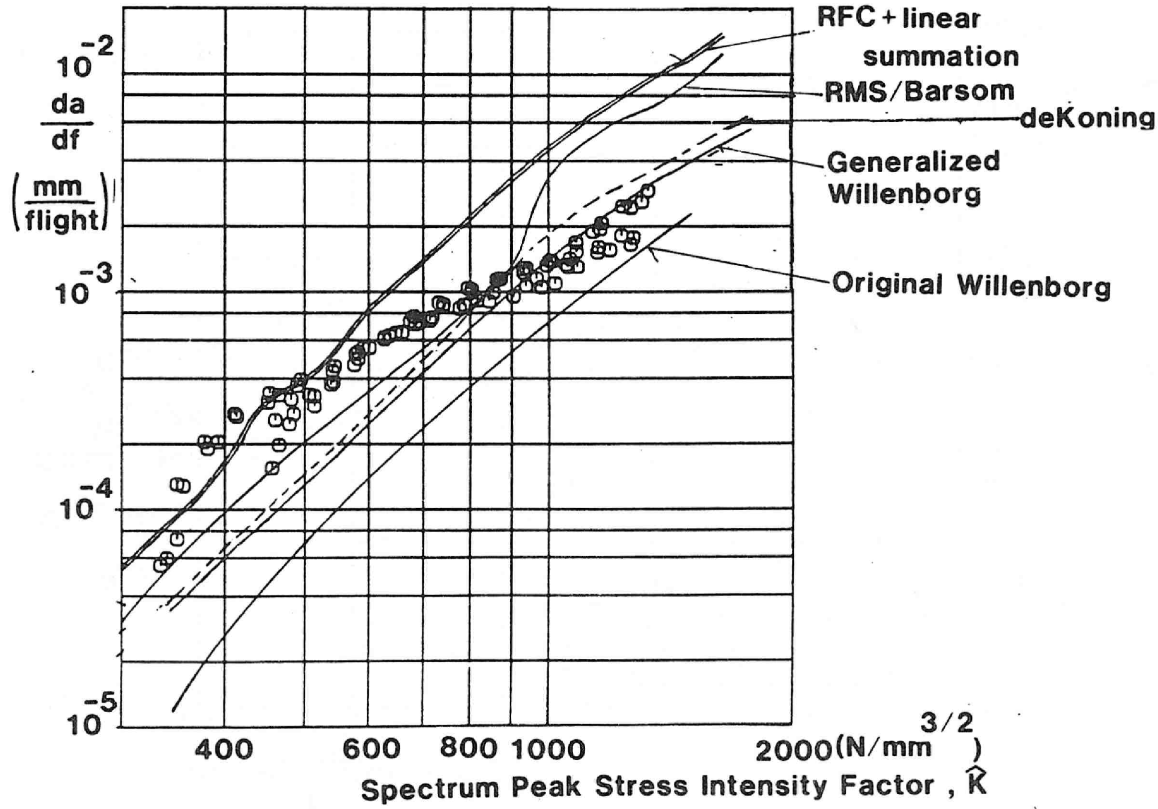


Fig. 15 Predicted and measured spectrum fatigue crack growth rates. Ref (8)

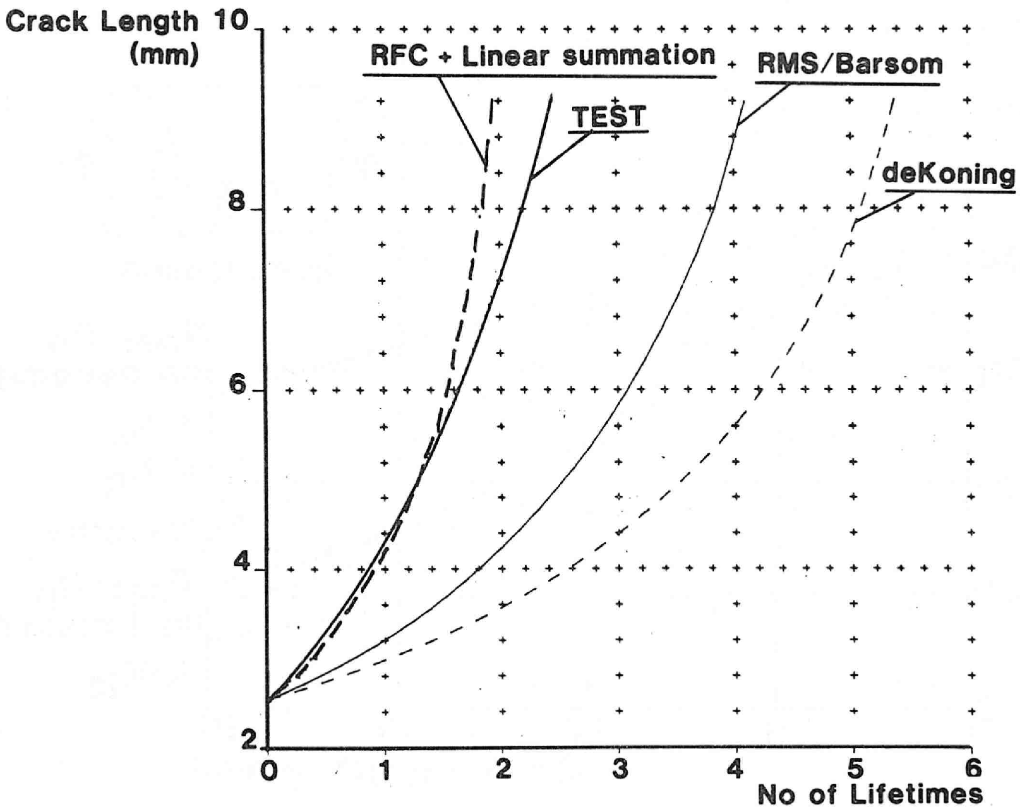


Fig. 16 Calculated crack growth for a component that barely fulfills the crack growth requirement.

APPENDIX

A Rational Approach for Fatigue Crack Growth Prediction

This method makes use of linearized spectrum fatigue crack growth data and in case of hand calculations: integrated stress intensity factor functions, Ref. (6).

Assume that each stress variation $\Delta\sigma_i$ of a spectrum (on the average) contributes to the crack growth rate by

$$\Delta\left(\frac{da}{df}\right) \frac{mm}{flight} = D \cdot (\Delta\sigma_i \cdot \beta\sqrt{\pi a})^k$$

Summing the contributions from all the $\Delta\sigma_i$ of a spectrum having n_f stress variations $\Delta\sigma_i$ during f flights one obtains

$$\frac{da}{df} = \underbrace{\frac{D}{f}}_{= D_f} \cdot \underbrace{(\hat{\sigma}\beta\sqrt{\pi a})^k}_{= \hat{K}} \cdot \underbrace{\sum_{i=1}^{n_f} \left(\frac{\Delta\sigma_i}{\hat{\sigma}}\right)^k}_{= \hat{n}_f} = D_f \cdot \hat{K}^k \cdot \hat{n}_f \quad \ldots \text{Ref (6)}$$

$= D_f$
= a constant

$= \hat{n}_f$ = spectrum intensity
= a function of spectrum shape

$= \hat{K}$ = spectrum peak stress intensity factor

$\hat{\sigma}$ = spectrum peak stress

The number of flights required to propagate a crack from length a_o to the length a is obtained by a simple integration

$$F(a_o \rightarrow a) = \int_{a_o}^{a} \frac{da}{(da/df)} = (D_f \cdot \hat{n}_f \cdot \hat{\sigma}^k)^{-1} \int_{a_o}^{a} \frac{da}{(\beta\sqrt{\pi a})^k}$$

$$= (D_f \cdot \hat{n}_f \cdot \hat{\sigma}^k)^{-1} \cdot I_g$$

For a general case, crack length versus number of flights is calculated, listed and plotted by means of a computer program working with da/df ($\hat{K}$) and $\beta(a)$ as inputs. As an alternative for hand calculations, the geometry integral I_g has been plotted for 5 common structural configurations with initial flaw sizes according to MIL-A-83444 and for standarized values of the exponent $k = k_{std}$ = 1.5 2.0 2.5 3.0 3.5 4.0 5.0. An example is shown in fig A1. Ref (9).

The slope exponent, k, is obtained from plots showing no. of flights required to propagate a crack to various end values as a function of stress level. The value is then rounded off to the nearest k_{std}. Fig A2.

The constant D_f is then obtained from a straight line having the slope k_{std} and fitted to the plotted data of lg(da/df) vs lg $(\hat{K})$.

$$\lg\left(\frac{da}{df}\right) = \lg D_f + k \lg \hat{K} + \lg \hat{n}_f$$

where n_f = the spectrum intensity = $\sum_{i=1}^{n_f} \left(\frac{\Delta\sigma_i}{\sigma}\right)^k$ for the actual testing spectrum with $\Delta\sigma_i$ obtained by means of the rain flow count technique.

For best prediction accuracy the straight line should be fitted to the data points in the transition region towards the apparent threshold value as shown in fig A3. The spectrum peak stress intensity factor, $\hat{K}$, should reflect retardation effects due to peak loads, and is therefore believed to be a more relevant correlation parameter than, e.g. the largest range of the spectrum.

Application of the method requires access to a D_f-value and a k_{std}-value obtained from a crack growth test with a spectrum of the same category as the one being considered in the prediction. The geometry integral I_g is read off from a diagram like fig A1. For a design spectrum of the same category (e.g. fighter wing spectrum) characterized by a spectrum intensity $\hat{n}_f$ and a local spectrum peak stress $\hat{\sigma}$, the crack growth life is then calculated from

$$F(a_o \rightarrow a) = (D_f \cdot \hat{n}_f \cdot \hat{\sigma}^k)^{-1} \cdot I_g$$

For stress levels and life requirements typical for military aircraft, the linearization in itself does not significantly contribute to prediction errors. The biggest source of error is the constant D_f. In ref (7), da/df-data were recorded for four different fighter wing spectra. Normalizing the D_f-value for the FALSTAFF spectrum to 100 %, the following relative D_f values were measured: 100, 87, 81 and 68 %. Using the D_f-value from just one test series, the maximum prediction error would be 32 %.

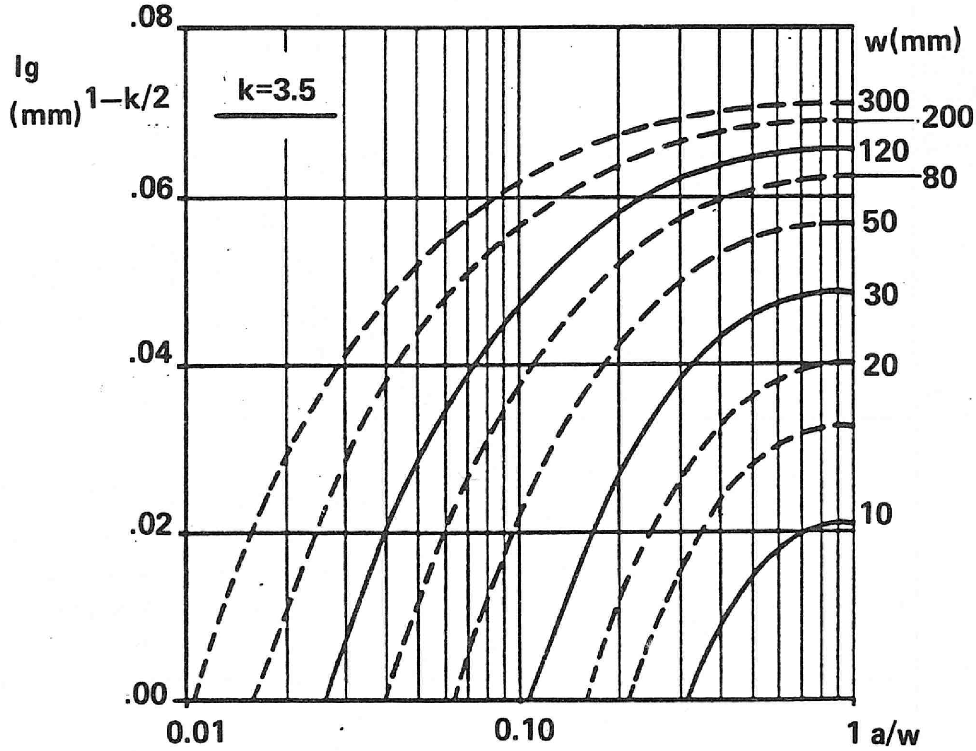


Fig. A1 The geometry integral I_g for a center cracked panel: Width = 2w. Crack length = 2a. Initial crack length, a_0 = 3.175 (1/8") Ref. (9)

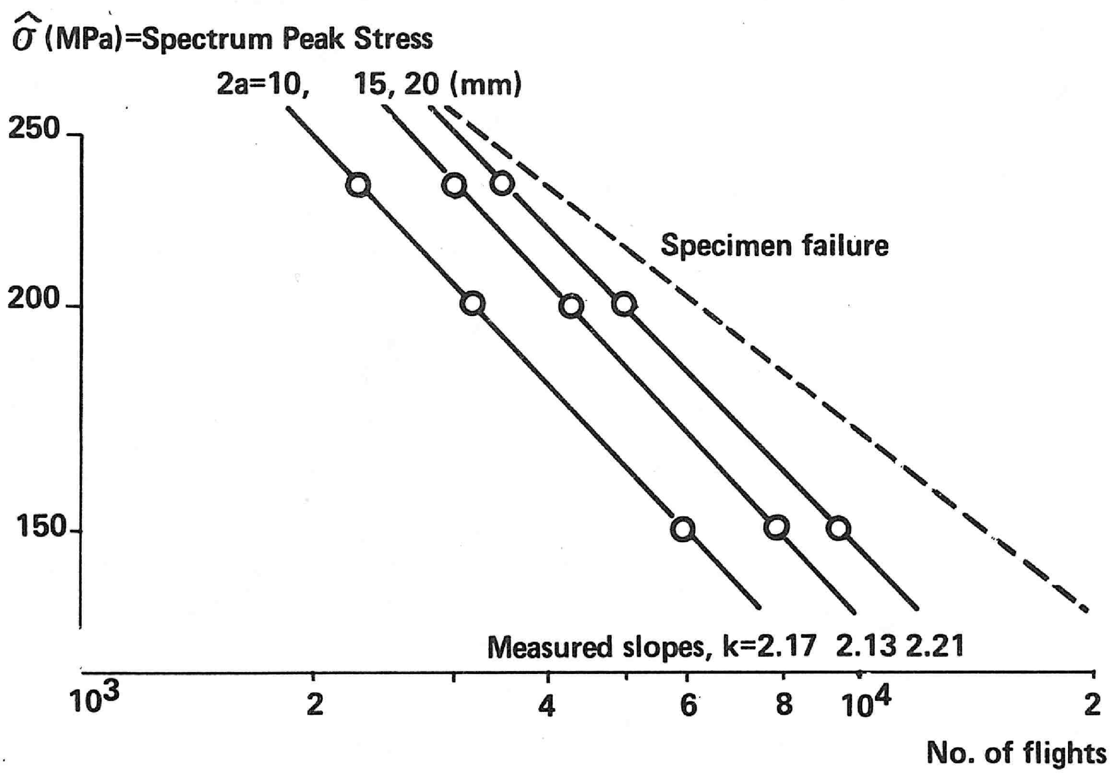


Fig. A2 Fatigue life from an initial flaw size $2a_0$ = 3.5 mm. FALSTAFF sequence. Ref (7)

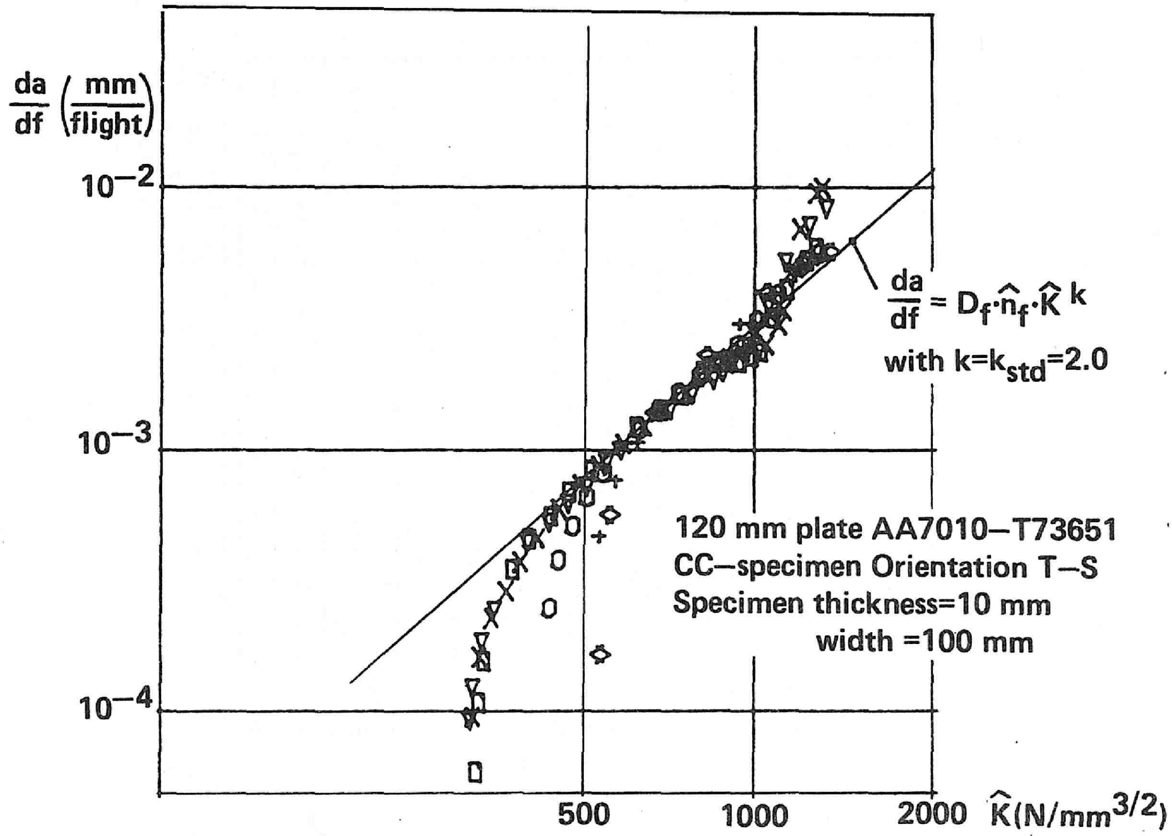


Fig. A3 Straight line fit to the da/df-data measured in the tests of Fig A2. Ref (7)