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İSTANBUL TECHNICAL UNIVERSITY ★ INSTITUTE OF SCIENCE AND TECHNOLOGY

**FATIGUE FAILURE AND CRACK PROPAGATION IN AIRCRAFT
STRUCTURES**

Ms. Thesis by

Sabriye Sera ATAGÜNDÜZ

511021033

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Supervisor (Chairman) : Assist. Prof. Dr. Vedat Ziya Doğan

Members of the Examining Committee Assoc. Prof. Dr. İbrahim Özkol (İ.T.Ü.)

Assist. Prof. Dr. Ata Muğan (İ.T.Ü.)

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Sabriye Sera ATAGÜNDÜZ
511021033

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Tez Danışmanı: Yar.Doç.Dr. Vedat Ziya Doğan

Diğer Jüri Üyeleri Doç.Dr. İbrahim Özkol (İ.T.Ü.)

Yar.Doç.Dr. Ata Muğan (İ.T.Ü.)

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PREFACE

The purpose of this study is to present a simplified semi-analytical methodology to predict the behaviour of fatigue and crack propagation in aircraft structures. The main goal is to correlate the propagation behaviour of cracks with the numerical models. Analysis of the results shows a highly satisfactory correlation between prediction by calculation and experimental data.

I gratefully thank my supervisor Asst.Prof. Vedat Ziya Doğan for his efforts in helping and supporting me during my studies with his great knowledge and experience in aeronautical engineering. In addition to this, i want to mention my being obliged to my parents Prof. Dr. Gürbüz Atagündüz and F. Aysel Atagündüz for supporting me in all parts of my life.

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S.Sera Atagündüz

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LIST OF SYMBOLS

a	:crack length (half length)
B_{nUSD}	:Beta Non-Uniform Stress Distribution
β_b	:Beta Bulging
B_s	:Beta Stiffening
C	:empirical fatigue material constant
E	:Modulus of Elasticity
$f(g)$	:correction factor that depends on specimen and crack geometry
K	:stress intensity factor
K_I	:opening mode
K_{II}	:sliding mode
K_{III}	:tearing mode
K_{to}	:stress intensity factor for complicated geometry
K_θ	:stress intensity factor in the absence of all boundaries of a form applicable to the loading
K_n	:stress intensity factor for the nth simpler configuration
K_e	:factor that accounts for the effect
K_c	:fracture toughness
ΔK_{eff}	:effective stress intensity factor range
m	:empirical fatigue material constant
N	:cycles
r	:cylindrical coordinate of a point with respect to the crack tip
R	:The stress ratio
S_y	:yield strength
$w(a,x)$	:weight function
T	:Integration period
σ	:stress applied to component
$\sigma(x)$	:the stress distribution in the uncracked body at the location of the crack
$\Delta\sigma$	:the remote stress applied to the component.
θ	:cylindrical coordinate of a point with respect to the crack tip

ÖZET

UÇAK YAPILARINDA YORULMA VE ÇATLAK İLERLEMESİ

Mühendislik tasarımlarında malzemeleri uygulama aşaması, tasarımcıya ve üreticiye çözmesi güç sorunlar getirmiştir. Birçok yüzyıl boyunca metal malzeme kullanımı oldukça yüksek maliyete sebep olmuştur. Metal malzeme kullanımındaki metodların gelişmesi ile birlikte metallerin kullanımı hızla artmıştır. Bir süre sonra, metallerin kullanımının her durumda tatmin edici sonuçlar vermediği, beklenmeyen bazı malzeme yorulmalarıyla karşı karşıya kalındığı gözlenmiştir. Havacılık endüstrisi, uçak yapılarında metallerin kullanımının artmasıyla birlikte yorulma ve çatlak ilerlemesi sorunlarıyla her geçen gün daha fazla karşı karşıya kalmaya başlamıştır. Yorulma ömrü üzerinde durulması gereken esas nokta, malzemenin nasıl ve ne zaman yorulduğunun ve hasarın ne zaman başladığının ortaya çıkarılmasının gerekliliğidir. Bu çalışmanın amacı, uçak yapılarında yorulma ve çatlak ilerleyişi davranışını, analitik bir yöntemle sunmaktır. Bu çalışmada üç farklı geometriye köşe çatlağı modellemesi, örnek bir uçak gövdesi paneline de boylamasına çatlak modellemesi uygulanmıştır. Dairesel bir delik etrafında köşe çatlağı oluşması, uçak yapılarının ömrü üzerindeki etkisinden dolayı kırılma mekaniği otoritelerinin oldukça fazla dikkatini çeken bir konu olmuştur. Bu çalışmadaki esas amaç, boylamasına bir çatlağın uçak gövde panelindeki ilerleyişinin, gerçek veriler ve yapılan analitik çalışma arasındaki yakınsaklığının gözlemlenmesidir. Çatlak ilerleyişi analitik olarak hesaplanmış, gerçek çatlak ilerleyiş verileri ile karşılaştırılmıştır. Analiz sonuçlarında, önerilen yöntemlerle geliştirilen analitik formülasyonun mühendislik açısından yeter yakınsaklığı sağladığı görülmüştür.

ABSTRACT

FATIGUE FAILURE AND CRACK PROPAGATION IN AIRCRAFT STRUCTURES

The aviation industry has been plagued with fatigue failures ever since the vastly increased use of metals during the industrial revolution. The purpose of this study is to present a simplified semi-analytical methodology to predict the behaviour of fatigue and cracks in aircraft structures. In this study the models analyzed are corner cracks applied to three different geometries and a longitudinal crack above a broken frame in a curved stiffened panel. The problem of corner cracks at a circular hole configuration has received considerable attention from the fracture mechanics community because of its effect on the life of aircraft structures. The main goal is to correlate the propagation behaviour of a longitudinal crack for another stiffened curved panel as determined with the numerical curved model. The methodology is based mainly on the application of "bulging" coefficients to stress intensity factors taken from a numerical model of a stiffened cracked panel in order to get the crack driving force of the curved panel. A crack above a broken frame is taken into account in modelling the longitudinal crack on the fuselage panel. The panel has test data of the non-uniform stress distribution. The basic formulation used is the Forman Equation. The bulging phenomenon is included in the stress intensity factor calculations by using the formulation of Swift. The Crack propagation is calculated and compared with the actual test results. Analysis of the results shows a highly satisfactory correlation between prediction by calculation and experimental data.

1. INTRODUCTION

1.1. Introduction To Fatigue Failure And Crack Propagation

Application of materials in engineering design has brought difficult problems to mankind. For many centuries metal working was extremely costly and laborious. With the improving skills of metal working, applications of metals in structures increased progressively. Afterwards it was experienced that structures built of these materials doesn't behave all the time satisfactorily and people has experienced some unexpected failures occuring in those structures.

The transport industry has been plagued with fatigue failures ever since the vastly increased use of metals during the industrial revolution [1]. The vastly increasing use of metals in the nineteenth century caused the number of accidents and casualties to reach unknown levels but the improvements in fracture mechanics provides decreasing the number of possible accidents and damages. The important point which must be focused on fatigue is which means how and when the material fails and the damage starts.

Metal fatigue has been a problem for aircraft since the days of the Wright brothers as indicated by Glen Campbell of NRC Canada [2]. Glen stated that Boggs reported the first flight of the Wright brothers' flyer was delayed because a hollow propeller shaft developed a fatigue crack and had to be replaced by a solid one shipped from Dayton, Ohio, to the testing site Kitty Hawk [2]. According to Campbell, there have been 1519 fatalities in 216 fixed-wing aircraft fatigue related accidents up to December 1980 [2].

Metal fatigue is a process which causes premature failure or damage of a component subjected to repeated loading [3]. It is a complicated, metallurgical process which is difficult to accurately describe and model on the microscopic level.

One of the most important physical observations is that the fatigue process can generally be broken into two distinct phases which are initiation life and propagation life. The initiation life encompasses the development and the early growth of a small crack. The propagation life is the portion of the total life spent growing a crack to failure [1]. However, it is often very difficult, if not impossible, to define the transition from initiation to propagation. This distinction depends upon many variables, including component size, material, and the methods used to detect cracks.

Fatigue of materials has been studied over 150 years . August Wöhler, while not the first, is one of the earliest and most famous fatigue researchers. During the period from about 1850 to 1875 , experiments were conducted to establish a safe alternating stress below which failure would not occur [1]. Full scale axles as well as smaller laboratory specimens were employed to establish the effects of the many variables that influence the long life fatigue strength of metals.

Research in fatigue during 1930's and 1940's was largely devoted to experimentally establishing the effects of the many factors that influence the long-life fatigue strength [1].

Many significant contributions were made during the 1960's. Irwin and others pioneered the development of fracture mechanics as a practical engineering tool. Paris quantified the relations for fatigue crack propagation. Paris, in " Twenty Years of Reflection on Questions Involving Fatigue Crack Growth" comments on his original work : " The paper was quite promptly rejected by three leading journals whose reviewers uniformly 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" [4]. Smooth specimen simulations of notches and cycle counting methods for variable amplitude loading were developed. Much of this work was made possible by the introduction of closed-loop materials testing systems and the introduction of the digital computer for solving engineering problems.

By the 1970's fatigue analysis became an established engineering tool in many industrial applications. Despite all of this knowledge, unintended fatigue failures continue to occur. Many of the failures are a result of fatigue technology being in the hands of the "experts" rather than the people who design and build structures and components [1].

There is no general fatigue and crack growth analysis that is best in all design situations. Each technique has its own advantages and limitations and a selection must be based on material, loading history, service environment, component geometry and consequences of component or structural failure.



2. LINEAR ELASTIC FRACTURE MECHANICS AND FATIGUE CRACK PROPAGATION

2.1. Fatigue And Crack Propagation

The fatigue life of a component is made up of initiation and propagation stages. This is illustrated schematically in Fig.2.1.1 [5]. The size of the crack at the transition from initiation to propagation is usually unknown and often depends on the point of view of the analyst and the size of the component being analyzed. For example, for a researcher equipped with microscopic equipment it may be on the order of a crystal imperfection, dislocation or a 0.1 mm-crack, while to the inspector in the field it may be the smallest crack that is readily detectable with non-destructive inspection equipment. Nevertheless, the distinction between the initiation life and propagation life is important. Fracture mechanics approaches are used to estimate the propagation life.

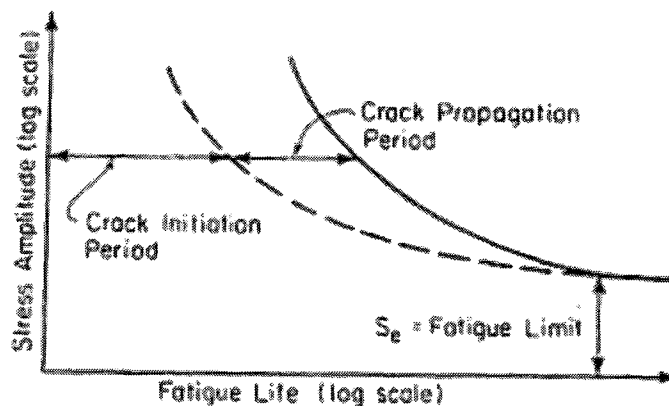


Figure 2.1. Initiation and Propagation Portions of Fatigue Life

Fracture mechanics approaches require that an initial crack size be known or assumed [6]. For components with imperfections or defects, an initial crack size may be known. Alternatively, for an estimate of the total fatigue life of a defect-free material, fracture mechanics approaches can be used to determine propagation.

In the development of stronger materials research has long been devoted to yield phenomena and plastic flow, because an increase of the yield strength usually leads to a stronger material [5]. This work has provided us with a good many new developments, and our knowledge on these subjects is substantial.

Fracture mechanics can be looked upon from a metallurgical viewpoint or a stress analysis viewpoint. The former usually takes place after failure with a fractographic analysis of the fracture surface. The latter is primarily associated with the calculation of crack growth and residual strength in order to establish an inspection program to prevent failure. Since engineers are often involved in conducting or reviewing the damage tolerance evaluation to prevent failure it appears appropriate to concentrate first on the analysis oriented fracture mechanics approach.

In aviation structures fatigue and crack under dynamic loads is more important than fatigue and crack under static loads [7]. Although the values of dynamic loads are not as high as static loads, as they are continuously repeating for long time periods, the fatigue and the crack occurring on structures subjected to dynamic loads causes the structures to be out of service.

2.2. Stress Analysis Of Cracks

Linear elastic fracture mechanics principles are used to relate the stress magnitude and distribution near the crack tip to:

- Remote stresses applied to the cracked component
- The crack size and shape
- The material properties of the cracked component

In the 1920's, Griffith formulated the concept that a crack in a component will propagate if the total energy of the system is lowered with crack propagation. That

is, if the change in elastic strain energy due to crack extension is larger than the energy required to create new crack surfaces, crack propagation will occur [5].

Griffith's theory was developed for brittle materials. In the 1940s, Irwin extended the theory for ductile materials. He postulated that the energy due to plastic deformation must be added to the surface energy associated with the creation of new crack surfaces. He recognized that for ductile materials, the surface energy term is often negligible compared to the energy associated with plastic deformation. Further, he defined a quantity, G , the strain energy release rate or "crack driving force", which is the total energy absorbed during cracking per unit increase in crack length and per unit thickness [1].

In the mid-1950s, Irwin made another significant contribution. He showed that the local stresses near the crack tip are of the general form :

$$\sigma_{ij} = \left[\frac{K}{\sqrt{2\pi r}} \right] f_{ij}(\theta) + \dots \quad (2.1)$$

where r and θ are cylindrical coordinates of a point with respect to the crack tip shown in Figure 2.2. and K is the stress intensity factor [5]. He further showed that the energy approach (the "G" approach above) is equivalent to the stress intensity approach and that crack propagation occurs when a critical strain energy release rate, G_c (or in terms of a critical stress intensity, K_c) is achieved [5].

From Equation 2.1. given above it is apparent that the solution is depending on $1/\sqrt{r}$. When $r \rightarrow 0$ solution has a infinite value while other parameters are reaching a finite value or zero. Due to this, the stress in the crack tip, changes depending upon $1/\sqrt{r}$ while it is independent from the configuration of the cracked structure. So, the crack tip displacement changes depending on $1/\sqrt{r}$ and when $r = 0$ there occurs the stress singularity.

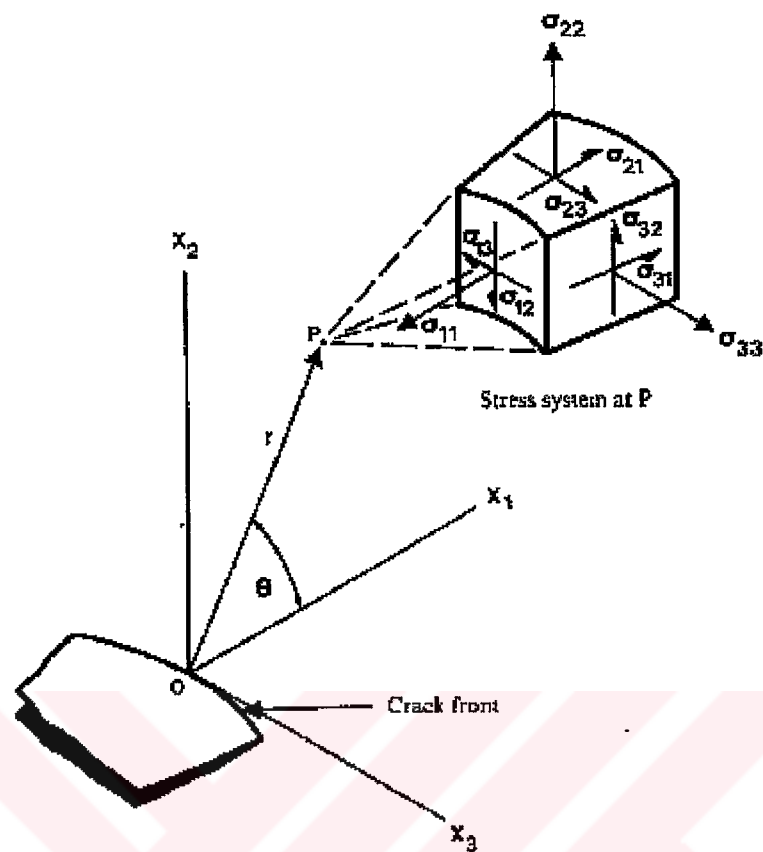


Figure 2.2. Location of Crack Stresses Near a Crack Tip in Cylindrical Coordinates

There are generally three modes of loading, which involve different crack surface displacements shown in Figure 2.3 [5]. The three modes are :

- Mode I: Opening or tensile mode (the crack faces are pulled apart)
- Mode II: Sliding or in-plane shear (the crack surfaces slide over each other)
- Mode III: Tearing or anti-plane shear (the crack surfaces move parallel to the leading edge of the crack and relative to each other)

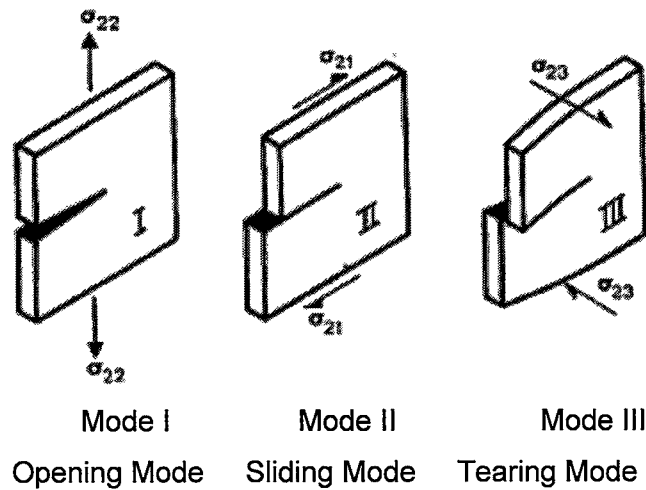


Figure 2.3. Three Loading Modes

2.3. Stress Intensity Factor

The operational life of a structure or a component is often limited by the initiation and subsequent growth of cracks. The presence of cracks may result in a loss of performance or even in the total failure of the structure. Since it may not be possible to prevent cracks occurring in service it is necessary to have a means of assessing their effect in a given structure [7]. This information can then be used in the design of structures in order to minimize the effects of cracks and, also be used during service to assess the likely behaviour of a cracked structure. Fracture mechanics provides a basis for quantifying the behaviour of cracks during both the propagation phase and the final failure. The important parameter in fracture mechanics is the stress intensity factor K which is a measure of the magnitude of the stress occurring in the highly stressed region at the tip of a crack in an elastic solid [8]. There is thus a need to know K for cracked configurations which might be encountered in practice. Many such solutions for K are now available and are presented in a standard form. The solutions are applicable to linear, elastic, isotropic and homogeneous materials containing stationary cracks.

The linear elastic solution for the stress-field around a crack shows that the stress components σ_{ij} are always of the same form; they are, in terms of the polar coordinates r, θ from the tip:

$$\sigma_{ij} = \left[\frac{K}{\sqrt{2\pi r}} \right] f_{ij}(\theta) + \dots \quad (2.1.)$$

If the point (r, θ) is sufficiently close to the tip, i.e. $r \ll$ crack length, the other terms in the above equation are negligible compared to the first term. The elastic solution predicts infinite stress at the crack tip ($r = 0$) which can not occur in practise since there is plastic flow in the highly stressed region near the tip. However, if the region of plastic flow is small compared to the region over which the $r^{-\frac{1}{2}}$ term dominates the stress-field, it may be assumed that the behaviour of the crack is determined by the elastic stress intensity factor. This assumption forms the basis of linear elastic fracture mechanics.

The constant K , the stress intensity factor, is a function of the loading on the cracked configuration and of the size and shape of the crack and other geometrical boundaries; it has dimensions of $stress = \sqrt{length}$ [9]. There are three distinct types of cracking modes which are illustrated in Fig.2.3. Mode I, characterized by K_I and known as the opening mode, is the most common mode found in practical situations. Mode II, characterized by K_{II} , is known as the sliding mode and Mode III, characterized by K_{III} , is known as the tearing mode [5]. The stress intensity factor for each mode can be formally defined, as r tends to zero along $\theta = 0$, in terms of the stress components shown in Fig. 2.2 as follows [5]:

$$\begin{aligned} K_I &= \lim_{r \rightarrow 0} \left[\sqrt{2\pi r} \sigma_{22}(r, \theta) \right] \\ K_{II} &= \lim_{r \rightarrow 0} \left[\sqrt{2\pi r} \sigma_{21}(r, \theta) \right] \\ K_{III} &= \lim_{r \rightarrow 0} \left[\sqrt{2\pi r} \sigma_{23}(r, \theta) \right] \end{aligned} \quad (2.2)$$

For most two-dimensional representations of straight-fronted cracks the coordinate system (x_1, x_2, x_3) in Fig. 2.2 is a cartesian system (x, y, z) with the origin of coordinates translated to the crack centre and the crack lie in the $y = 0$ plane with the front parallel to the z axis [9]. In some three-dimensional configurations with planar cracks which have a curved crack-front the stress intensity factors vary with position on the crack-front. In contrast to two dimensional configurations the crack usually lies in the $z = 0$ plane of a cartesian system but the stress intensity factors

must now be defined in terms of a local coordinate system at the point under consideration; x_1 becomes the outward normal to the crack-front, x_2 the normal to the plane of the crack and x_3 the tangent to the crack-front.

Fracture mechanics is widely used to describe many aspects of crack behaviour, the onset of crack growth due to corrosion or fatigue, the rate of growth due to corrosion or fatigue and the occurrence of final failure. Material constants such as K_{Isc} the value of K at which stress corrosion cracking starts, and K_{Ic} the value of K at which plane strain failure occurs under a continuously increasing load, have been measured for many materials [10]. Crack growth laws have been suggested to explain the strong dependence of the rate of growth of a fatigue crack on the range ΔK over which the stress intensity fluctuates [5]. In order to apply these concepts and laws to a practical situation it is necessary to know the stress intensity factor for the particular configuration.

2.3.1. Relation Of Stress Intensity Factor With Other Parameters

The stress intensity factor, K , as explained before defines the magnitude of the local stresses around the crack tip. This factor depends on loading, crack size, crack shape, and geometric boundaries, with the general form given by [5]:

$$K = f(g)\sigma\sqrt{\pi a} \quad (2.3)$$

where σ = stress applied to component

a = crack length

$f(g)$ = correction factor that depends on specimen and crack geometry

Stress intensity factor solutions have been obtained for a wide variety of problems and published in handbooks. Figure 3.3 gives the stress intensity relationships for a few of the more common loading conditions [10].

Stress intensity factors for a single loading can be added algebraically. Consequently, stress intensity factors for complex loading conditions of the same

mode can be determined from the superposition of simpler results, such as those readily obtainable from handbooks.

One superposition method, the compounding technique, has been used to obtain relatively accurate approximations [10]. The technique consists of reducing a complicated problem into a number of simpler configurations with known solutions. By superposition of these simpler K solutions, a stress intensity factor may be obtained for the complicated geometry. In equation form [10]:

$$K_{tot} = K_0 + \left[\sum_{n=1}^N (K_n - K_0) \right] + K_e \quad (2.4)$$

where

- K_{tot} = stress intensity factor for complicated geometry
- K_0 = stress intensity factor in the absence of all boundaries of a form applicable to the loading
- K_n = stress intensity factor for the n th simpler configuration
- K_e = factor that accounts for the effect of the interaction between boundaries

K_e is the only unknown. Neglecting this term will lead to underestimates of less than 10%.

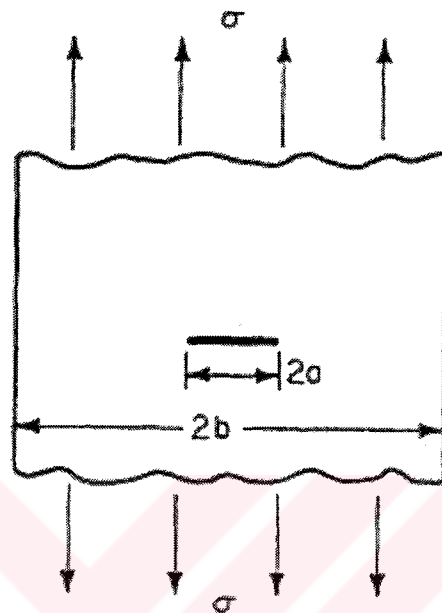
Another approximate method is simply to multiply the individual correction factors for the various geometric effects, such as [10]:

$$K = f_1 \cdot f_2 \cdot f_3 \dots \sigma \sqrt{\pi a} \quad (2.5)$$

Correction factors, f_i , are used to account for [10]:

- Finite width (back wall) effect
- Front wall effect
- Crack shape (i.e., elliptical flaw)

Other superposition methods that are employed include the alternating method and the weight function method.



$$K_I = f(g)\sigma\sqrt{\pi a}$$

$$f(g) = \sqrt{\sec\left(\frac{\pi a}{2b}\right)}$$

Figure 2.4. Center Cracked Plate Loaded in Tension

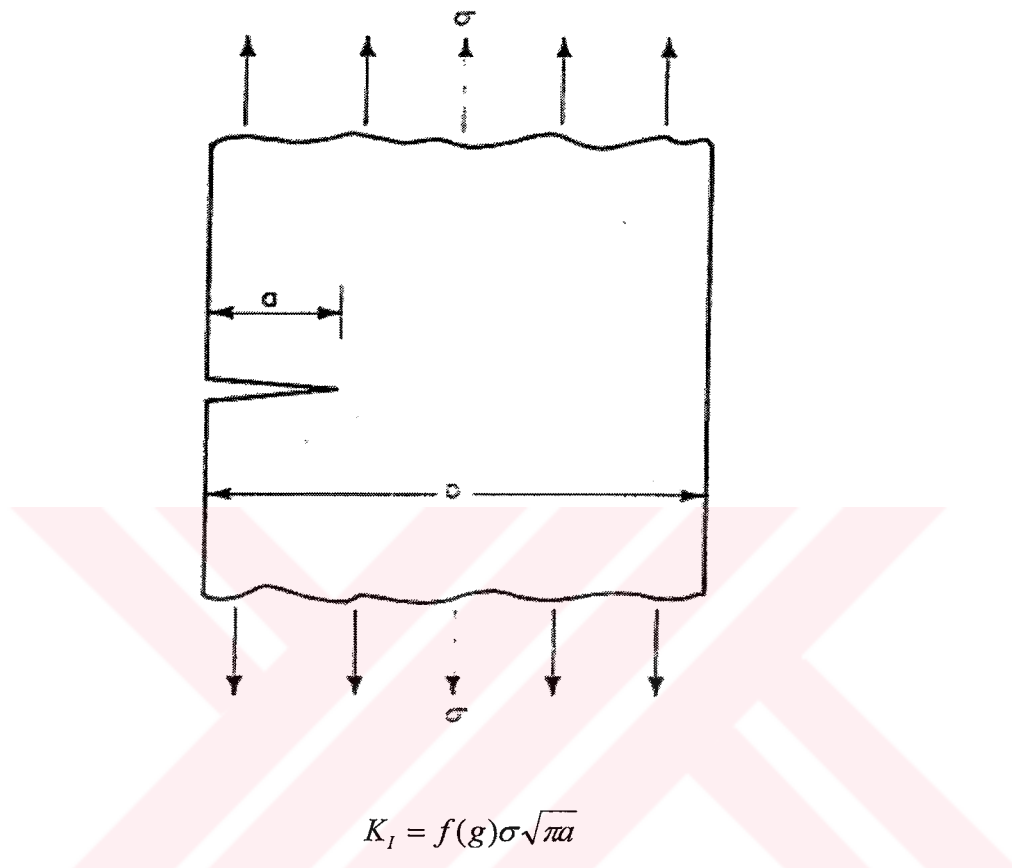
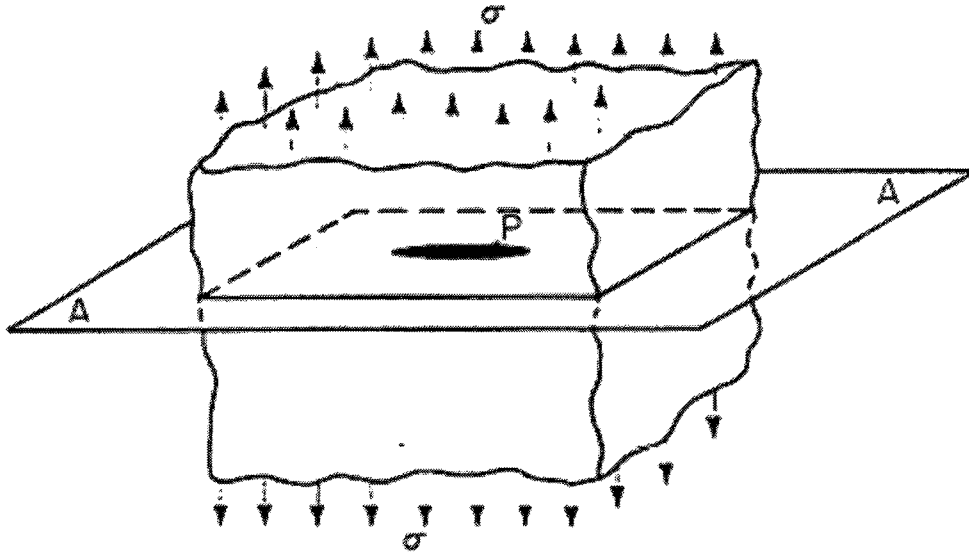


Figure 2.5. Edge Cracked Plate Loaded in Tension



$$K_{I_{atP}} = \frac{\sigma \sqrt{\pi a}}{\phi} \left[\sin^2 \theta + \frac{a^2}{c^2} \cos^2 \theta \right]^{\frac{1}{4}}$$

$$\phi = \int_0^{\frac{\pi}{2}} \sqrt{1 - \left(1 - \frac{a^2}{c^2}\right) \sin^2 \phi} d\phi$$

when $\theta = 90^\circ$, K_I is Maximum

when $\theta = 0^\circ$, K_I is Maximum

Figure 2.6. Elliptical Crack Embedded in Infinite Body Subjected to Tension

Determination methods for K tend to be approximate. In general, values for $f(g)$ tend to be between 1 and 1.4, with the value for many engineering situations being between 1 and 1.2 [10]. Errors in K may be small compared to uncertainties in a fatigue analysis, such as material properties, load levels, load history and service environment.

2.3.2. Weight Function Methods

Analytical solutions for straight, circular, and elliptical cracks are readily available, owing to their simplicity of geometry. On the other hand, irregular cracks and non-uniform loading conditions seldom have simple solutions due to their complex geometry and loading. Often, the finite element method must be applied in order to

determine the distribution of stress intensity factor or stress correction factor, in the case of a pore or cavity. Because of crack irregularity, fatigue crack growth often occurs at points of maximum stress intensity, leading to continuous change in irregularity of the crack front [9]. Under these circumstances, the finite element method calculations must be repeated to trace the change in crack shape and/or to allow for the loading spectrum. However, this is not practical, due to the cost and time involved.

The stress intensity for non-uniform but symmetric loading is obtained from the known stress intensity for another simple case of symmetric loading and the crack face displacements for the latter case is [10]:

$$K_{Isought} = \int_0^a \sigma(x) w(a, x) dx \quad (2.6)$$

where $w(a, x)$ is the weight function and $\sigma(x)$, the stress distribution in the uncracked body at the location of the crack. The weight function is then given by:

$$w(a, x) = \frac{8\mu}{1+k} \frac{1}{K_{known}} \left(\frac{\partial v(x)}{\partial a} \right)_{known} \quad (2.7)$$

where μ is the shear modulus and $k = 3-4\nu$ (ν is the poisson ratio) for plane strain and $k = (3-\nu)/(1+\nu)$ for plane stress. The crack face displacements of the known case are $v(x)$. Provided the reference case is a reference case is a simple one for which $v(x)$ is known, equation 2.6 can be solved numerically in little time.

2.4. Crack Tip Plasticity

Whether a fracture occurs in a ductile or brittle manner, or a fatigue crack grows under cyclic loading, the local plasticity at the crack tip controls both fracture and crack growth [11]. As mentioned previously, materials develop plastic strains as the yield stress is exceeded in the region near crack tip. The amount of plastic deformation is restricted by surrounding material, which remains elastic. The size of this plastic zone is dependent on the stress conditions of the body [5]. It is possible to calculate a plastic zone size at the crack tip as a function of stress intensity

factor, K , and yield strength, S_y , by using the stress equations and the von Mises or maximum shear stress yield criteria [5].

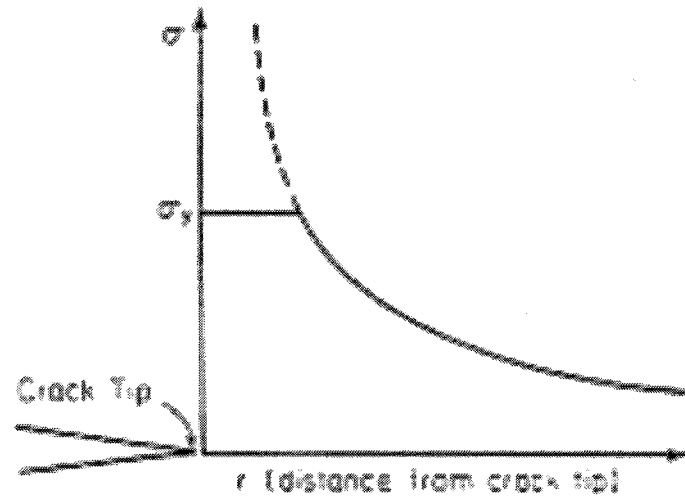


Figure 2.7. Yielding Near Crack Tip

The actual stress-strain distribution within the plastic zone is difficult to obtain; however, this is not significant for the use of linear elastic fracture mechanics in design.

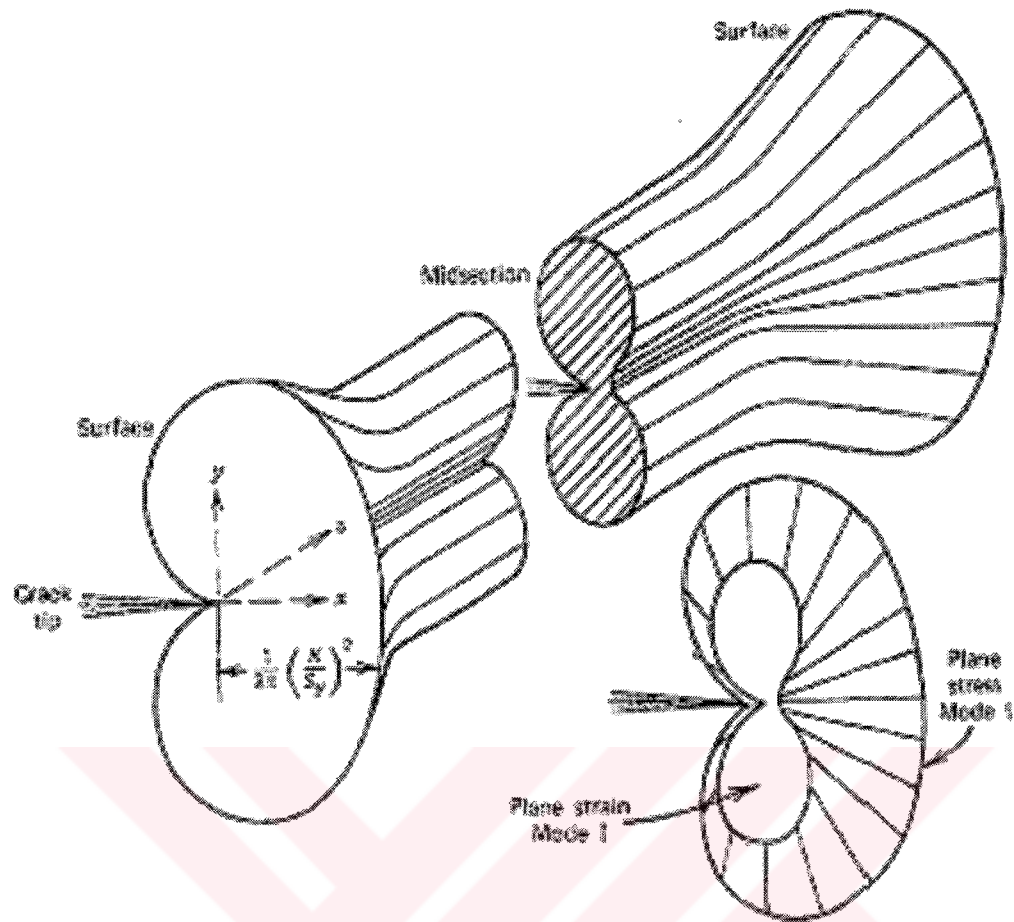


Figure 2.8. Plastic Zone Size at The Tip of a Through Thickness Crack

2.4.1. PLASTIC ZONE SIZE

In practice, materials (especially metals) tend to exhibit a yield stress, above which they deform plastically. This means that there is always a region around the tip of a crack in a metal, where plastic deformation occurs, and hence a stress singularity cannot exist. The plastic region is known as the crack tip plastic zone [6].

In a thin body, the stress through the thickness (σ_z) cannot vary appreciably due to the thin section. Because there can be no stresses normal to a free surface, $\sigma_z = 0$ throughout the section and a biaxial state of stress results. This is termed a plane stress condition [6].

In a thick body, the material is constrained in the z direction due to the thickness of the cross section and $\epsilon_z = 0$, resulting in a plane strain condition. Due to Poisson's

effect, a stress, σ_z , is developed in the z direction. Maximum constraint conditions exist in the plane strain condition, and consequently the plastic zone size is smaller than that developed under plane stress conditions[5].

- **Monotonic Plastic Zone Size:** The plastic zone sizes under monotonic loading have been estimated to be:

$$r_y = \left\{ \frac{1}{2\pi} \left(\frac{K}{\sigma_y} \right)^2 \right\} \quad \text{plane stress} \quad (2.7)$$

$$r_y = \left\{ \frac{1}{6\pi} \left(\frac{K}{\sigma_y} \right)^2 \right\} \quad \text{plane strain} \quad (2.8)$$

where r is defined as shown in Figure 2.9 [5].

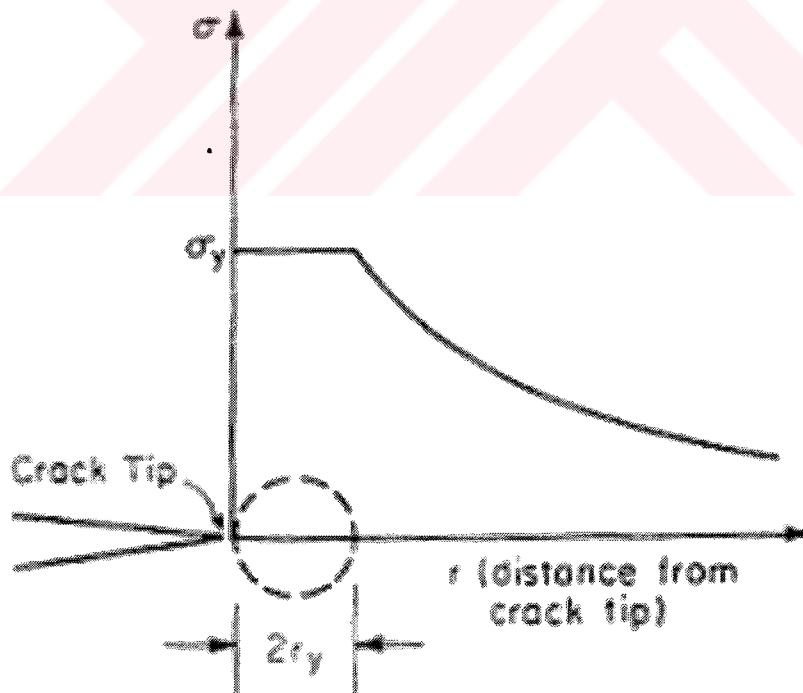


Figure 2.9. Monotonic Plastic Zone Size

- Cyclic Plastic Zone Size:** The reversed or cyclic plastic zone size is four times smaller than the comparable monotonic value. As the nominal tensile load is reduced, the plastic region near crack tip is put into compression by the surrounding elastic body. As shown in Figure 2.10., the change in stress at the crack tip due to the reversed loading is twice the value of the yield stress [5]. The cyclic plastic zone size is smaller than the monotonic and more characteristic of a plane strain state in thin plates. The linear elastic fracture mechanics concepts can often be used in the analysis of fatigue crack growth problems even in materials that exhibit considerable amounts of ductility. The basic assumption that the plastic zone size is small in relationship to the crack and the cracked body usually remains valid.

$$r_y = \left\{ \frac{1}{2\pi} \left(\frac{K}{2\sigma_y} \right)^2 = \frac{1}{8\pi} \left(\frac{K}{\sigma_y} \right)^2 \right\} \quad \text{plane stress} \quad (2.9)$$

$$r_y = \left\{ \frac{1}{6\pi} \left(\frac{K}{2\sigma_y} \right)^2 = \frac{1}{24\pi} \left(\frac{K}{\sigma_y} \right)^2 \right\} \quad \text{plane strain} \quad (2.10)$$

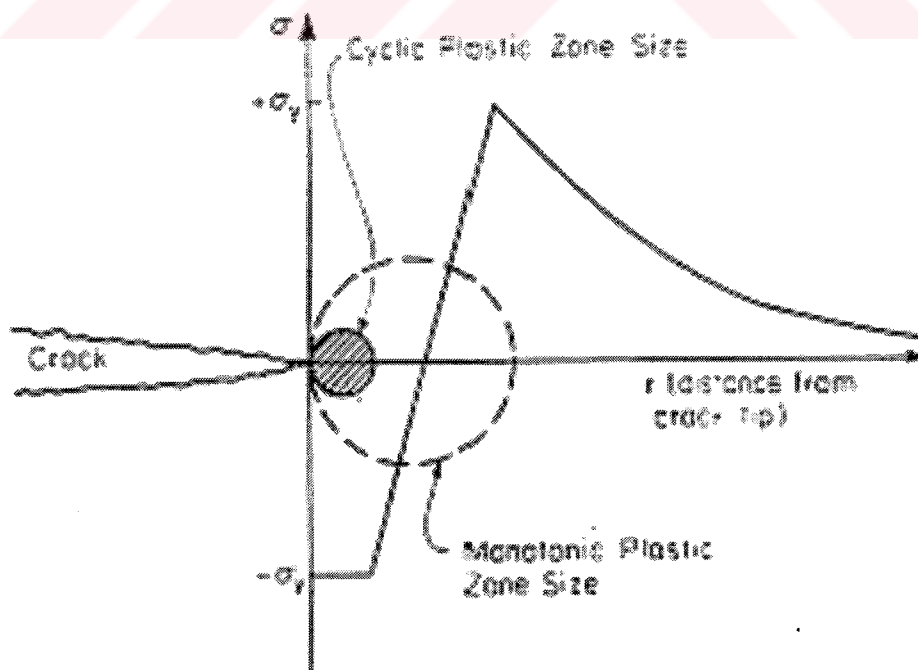


Figure 2.10. Reversed Plastic Zone Size

2.4.2. Critical “K”

As the stress intensity factor reaches a critical value, K_c , unstable fracture occurs [9]. This critical value of the stress intensity factor is known as the fracture toughness of the material. The fracture toughness can be considered the limiting value of stress intensity just as the yield stress might be considered the limiting value of applied stress.

Quantitative monotonic fracture toughness can be obtained for brittle materials from tests using specimens with fatigue cracks with known K expressions [5]. Critical values of K refer to the condition when a crack extends in a rapid (unstable) manner without an increase in load or applied energy. Critical values of K are denoted as K_c as follows [5]:

$$K_c = S_c \sqrt{\pi a_c} f\left(\frac{a_c}{w}\right) \quad (2.11)$$

where S_c is the applied nominal stress at crack instability and a_c is the crack length at instability; w is the width, K_c is called fracture toughness and depends on the material, temperature, strain rate, environment, thickness and to a lesser extent, crack length. If K_c is known for a given material and thickness, and K is known for a given component and loading, a quantitative design criteria to prevent brittle fracture exists involving applied stress and crack size. Since K_c also represents the stress intensity factor at the last cycle of a fatigue fracture, it can be used to obtain critical crack sizes for fracture under cyclic loading. In 1920 Griffith, using elastic energy balance equations and experimentation with brittle glass, was the first to produce a quantitative relationship for brittle fracture of cracked bodies. Effect of specimen thickness on fracture toughness is shown in Figure 2.11 [6].

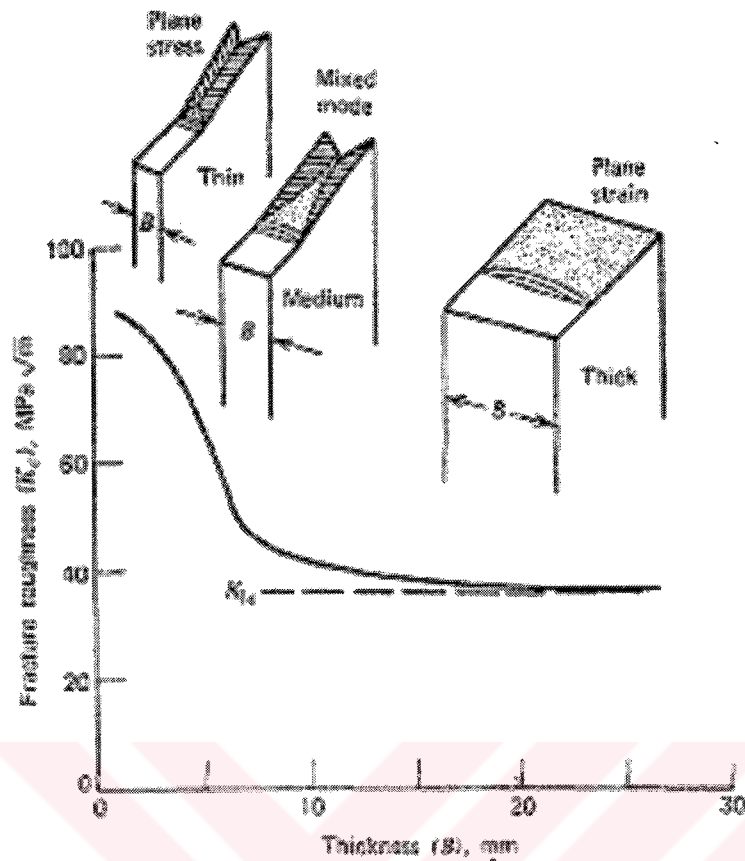


Figure 2.11. Effect of Specimen Thickness on Fracture Toughness

2.5. Fatigue Concept And Fatigue Crack Growth

If an increasing load is applied to a piece of material, a magnitude of load will eventually be reached at which the material fractures. This is the failing or ultimate static load. Now suppose a fraction of this ultimate load is applied to a similar piece of material. The material may bend or stretch but will not break. If the load is removed, the material will return to its original condition (assuming no permanent set). This application of load may be repeated several times. Each time the material will return to its original condition. Close examination may reveal no evidence that it has been subjected to a succession of loads. However, something has happened, because if this cycle of loading is repeated a sufficient number of times the material will break. This phenomenon of fracturing after application of a series of cyclic loads less than the ultimate load is called fatigue [5]. The higher the load applied, the smaller the number of cycles required to cause failure as illustrated by a typical S-N or Wohler curve. The S-N curve gives a curve giving the stress (S) as

the ordinate and the life cycles (N) as the abscissa on a long scale [12]. In some countries this curve is called a Wohler curve after the scientist who developed the foundations of all fatigue work.

The definition of fatigue as currently can be stated as follows [5]:

“ The process of progressive localized permanent structural change occurring in a material subjected to conditions which may produce fluctuating stresses and strains at some point or points and which may culminate in cracks or complete fracture after a sufficient number of fluctuations”.

The word “progressive “ implies the fatigue process occurs over a period of time or usage. A fatigue failure is often very sudden with no external warning; however, the mechanisms involved may have been operating since the beginning of the component or structure usage.

The word “localized” implies that the fatigue process operates at local areas rather than throughout the entire component or structure. These local areas can have high stresses and strains due to external load transfer, abrupt changes in geometry, temperature differentials, self-stresses (residual stresses), and material imperfections.

The word “cracks” is often the most misunderstood and misused part of fatigue. The ultimate cause of all fatigue failures is that a crack has grown to a point at which the remaining material can no longer tolerate the stresses or strains, and sudden fracture occurs. The crack had to grow this size because of the cyclic loading. In fact, because of costly fatigue failures in the aerospace fields current aircraft design criteria for certain safety critical parts require the assumption that small crack like defects exist prior to initial use of the aircraft. The fatigue life of these safetyt critical parts is thus based solely on crack propagation.

The word “fracture” implies the last stage of the fatigue process is separation of a component or a structure into two or more parts.

Fracture mechanics approaches require that an initial crack size be known or assumed. For components with imperfections or defects (such as welding porosities, inclusions and casting defects, etc.) an initial crack size may be known. Alternatively, for an estimate of the total fatigue life of a defect-free material,

fracture mechanics approaches can be used to determine propagation. Strain-life approaches may then be used to determine initiation life, with the total life being the sum of these two estimates.

As discussed, the majority of fatigue life may be taken up in the propagation of a crack. By the use of fracture mechanics principles it is possible to predict the number of cycles spent in growing a crack to some specified length or final failure.

The aircraft industry has been instrumental in the effort to understand and predict fatigue crack growth. They have developed the safe-life or fail-safe design approach. In this method, a component is designed such that if a crack forms, it will not grow to a critical size between specified inspection intervals. Thus, by knowing the material growth rate characteristics and with regular inspections, a cracked component may be kept in service for an extended useful life [13]. This concept can be seen in Figure 2.12 [6].

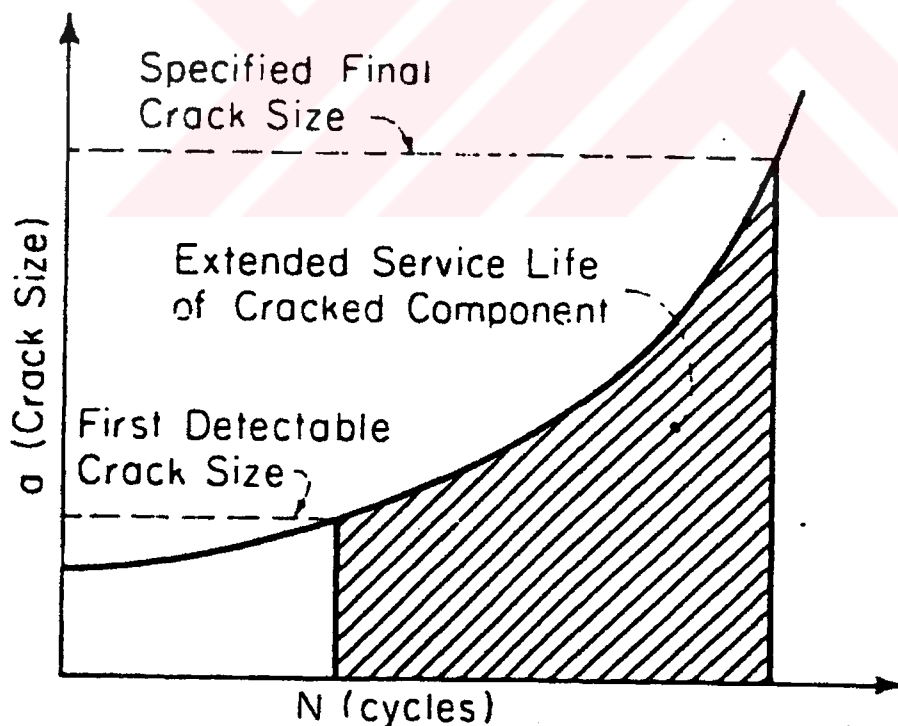


Figure 2.12. Extended Service Life of A Cracked Component

2.5.1. Mean Stress Levels

Most of the life of the component is spent while the crack length is relatively small. In addition, the crack growth rate increases with increased applied stress. The crack growth rate is:

$$\frac{da}{dN} \quad (2.12)$$

Crack growth rate is obtained by taking the derivative of the crack length, a , versus cycles, N , curve. Values of $\log da/dN$ can then be plotted versus $\log \Delta K$, for a given crack length, using the equation [6]:

$$\Delta K = K_{\max} - K_{\min} = f(g) \Delta \sigma \sqrt{\pi a} \quad (2.13)$$

where $\Delta \sigma$ is the remote stress applied to the component.

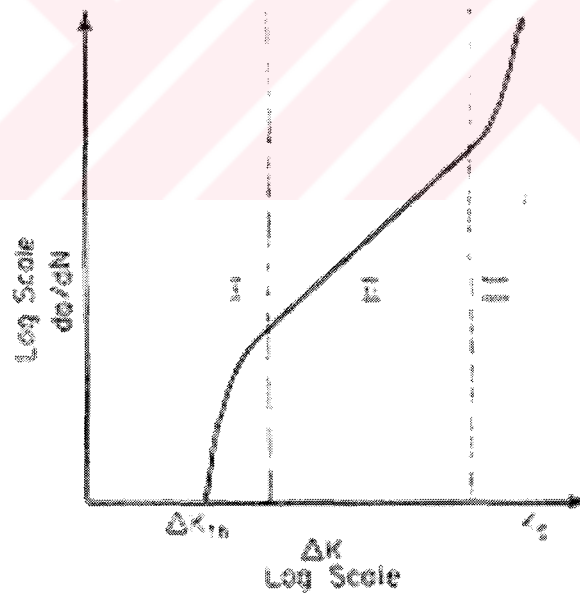


Figure 2.13 Three Regions of Crack Growth Rate Curve

The stress ratio $R = K_{\min} / K_{\max} = S_{\min} / S_{\max}$ is used as the principal parameter. Most mean stress effects on crack growth have been obtained with only tensile stressing, that is, $R \geq 0$ [6]. Increasing R ratio (which means increasing the mean

stress) has a tendency to increase the crack growth rates in all portions of the sigmoidal curve. The increase in region II, however, may be small. In region III, where fracture toughness K_c controls, substantial differences in crack growth rates occur for different R ratios. The most commonly used equation depicting mean stress effects in regions II and III is the Forman Equation [14]:

$$\frac{da}{dN} = \frac{C(\Delta K)^m}{(1-R)K_c - \Delta K} \quad (2.14)$$

where C and m are empirical fatigue material constants and K_c is the applicable fracture toughness for the material and thickness. The Forman equation is a modification of the Paris equation.

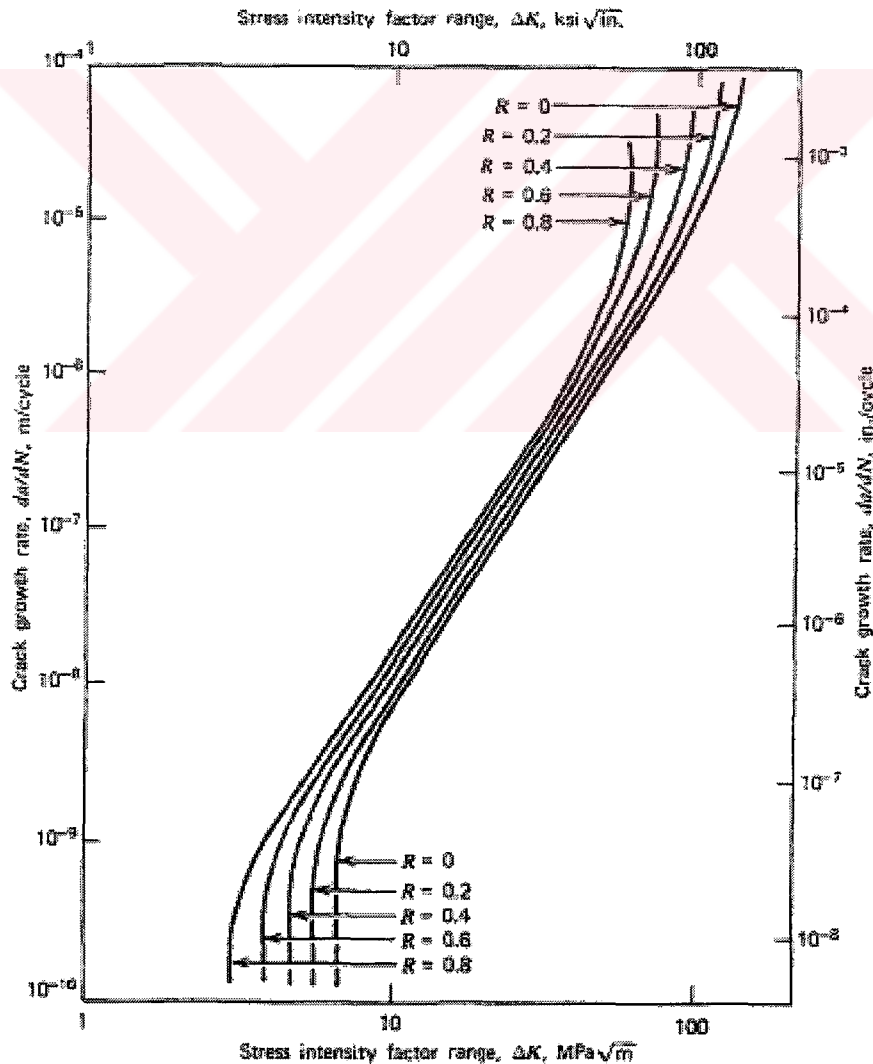


Figure 2.14 Schematic Mean Stress Influence on Fatigue Crack Growth

The effect of negative R ratios, which includes compression in the cycle, has not been sufficiently investigated, particularly at the threshold levels. The results of many negative R ratio tests on wrought and cast steels, cast irons, and aluminum alloys subjected to constant amplitude conditions in regions II and III indicate crack growth rates based on ΔK values (which neglect compressive nominal stresses) are similar to $R = 0$ results, or are increased but not more than a factor of 2 [5].

2.5.2. Loading Environment

Environmental effects on fatigue of metals may be more severe than sharp stress concentrations or almost harmless. Quantitative fatigue life predictions are often not possible because of the many interacting factors that influence environmental fatigue behaviour and lack of significant data [14]. For example, in corrosion fatigue, frequency effects can be quite substantial, while in non-corrosive environments frequency is usually a second order concern [14]. Design against fretting fatigue, particularly in mechanical joints, has not been quantified [7]. At elevated temperatures mean stress effects are extremely complex because of interaction among creep, fatigue, and environment. The linear elastic stress intensity factor K also has more limitations at elevated temperatures because of appreciable plasticity. Substantial reduction in fracture toughness can occur at low temperatures, which reduces critical crack sizes at fracture. Irradiation can reduce both fatigue resistance and fracture toughness. Despite these large difficulties fatigue design with environmental considerations must be accomplished [13]. This places great emphasis on real-life product testing, inspection, service history analysis, experience, and accelerated acquisition of experience [13].

There are many environments that effect fatigue behaviour; however, most engineering components and structures interact with air, water, and salt water. Corrosion fatigue refers to the joint interaction of corrosive environment and repeated stressing [15]. The combination of both acting together is more detrimental than either acting separately. That is, repeated stressing accelerates the corrosive action and the corrosive action accelerates the mechanical fatigue mechanisms. Under static loads, corrosive environments may also be detrimental, particularly in higher strength alloys. Environmental assisted fracture under static loading is called "stress corrosion cracking" [6].

2.5.3. Fatigue Crack Propagation

In a first approximation, the propagation of cracks depends on the alternating stresses that pull the crack faces open; the other principal alternating stresses have a modifying influence [16]. A small alternating tensile stress is required for crack growth.

In the stress plane for biaxial states of stress, the compression-compression quadrant is entirely safe from crack growth. Any state of nominal mean and alternating stresses represented by a line that remains entirely in this quadrant will not propagate small cracks [17].

A small band of tensile stress, corresponding to the critical alternating tensile stresses can be added to this quadrant. If the mean stress is far in the tension-tension quadrant, the zone of stress states that leads to non-propagating cracks is a small square [17].

Nominal net stresses are the criterion for crack propagation. They must, of course, include self-stresses. Local stresses are the criterion for initiation. All these considerations for long life apply only if gross yielding is avoided. The zone of nonyield is an ellipse according to the von Mises-Huber-Henky or octahedral shear stress theory, or a hexagon according to Tresca theory [17].

In Figure 2.15 this approach is applied to the determination of the optimum amount of surface compression for coil springs or torsion bars [6]. For a given compressive mean, line A-A' represents the torsional stress range for crack initiation, while B-B' is that for propagation. The largest allowable alternating stress is achieved for a mean stress where failure due to propagation and yielding are equally likely, as shown by line D-D'. In this, A-A' shifts to C-C' (larger ellipse). Further increases in compressive mean stress would reduce allowable alternating stress because the latter is limited by yielding.

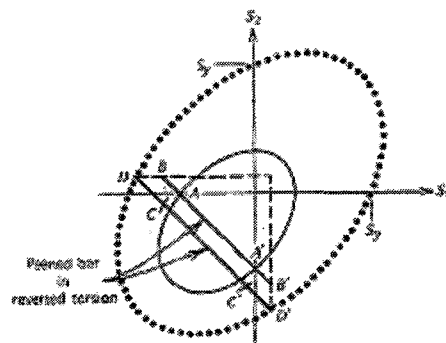


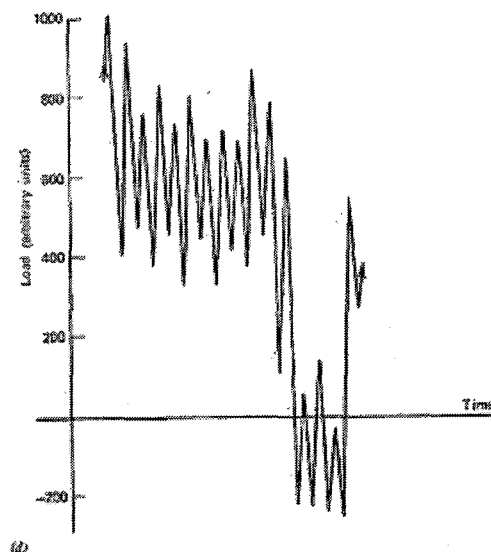
Figure 2.15 Three Criteria used for Optimizing a Biaxial Stress Rate

2.5.4. Cumulative Damage

Analysis of cumulative fatigue damage is a method that addresses the strain-life behaviour of metal fatigue resistance, the effect of mean stress, the effect of geometric notches, the typical strain-time histories of components [18]. To ascertain structural life under other constant-amplitude conditions one must apply cumulative damage criteria to conditions of varying stress or strain amplitudes.

In tests with uniform load cycles the cumulative effect of all the cycles eventually produces fatigue failure (unless the load is below the prevailing fatigue limit) [18]. When the loading and unloading does not occur in uniform cycles but in an irregular manner, the cumulative effect of these events may also produce fatigue failure. The term “cumulative damage” refers to the fatigue effects of loading events other than uniform cycles. The term “ spectrum” as used in fatigue literature often means a series of fatigue loading events other than uniformly repeated cycles [18]. Sometimes spectrum means a listing, ordered by size, of components of irregular sequences. Other parameters, such as maximum load, or pairs of parameters, such as range and mean, are also used to define the classifications or “boxes” in which the counts of cycles are listed.

Figure 2.16 shows three service load records along with details to a larger time scale of spectrum b as examples of irregular real load histories [18]. One approach to variable load histories uses the concept of damage, defined as the fraction of life used up by an event or series of events. These fractions are added together; when their sum reaches 1.0 or 100% we expect and predict failure.



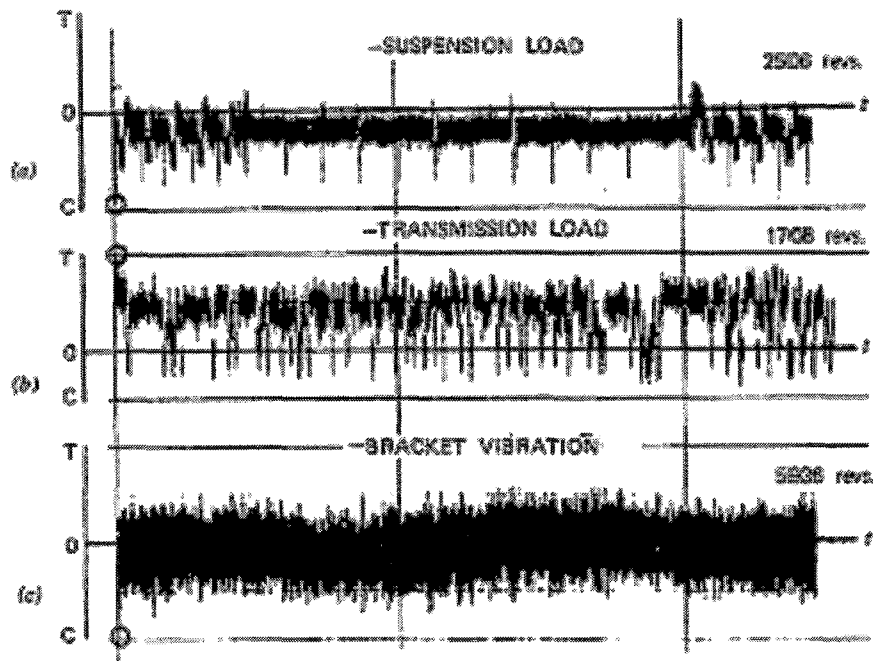


Figure 2.16 Amplitude Time Display of Service Histories

2.5.5. Crack Closure

Crack closure arguments are often used to explain the stress ratio effect of crack growth rates as well as environmental effects on ΔK_{th} . In addition, crack closure theories are very important in variable amplitude fatigue crack growth predictions.

In the early 1970s, Elber observed that the surfaces of fatigue cracks close (contact each other) when remotely applied load is still tensile and do not open again until a sufficiently high tensile load is obtained on the next loading cycle. He developed the theory of crack closure to explain this phenomenon [5].

Elber proposed that crack closure occurs as a result of crack-tip plasticity. A plastic zone develops around the crack tip as the yield stress of the material is exceeded [5]. As shown in Figure 2.17, as the crack grows, a wake of plastically deformed material is developed while the surrounding body remains elastic. Elber proposed that as the component is unloaded, the plastically “stretched” material causes the crack surfaces to contact each other before zero load is reached [5].

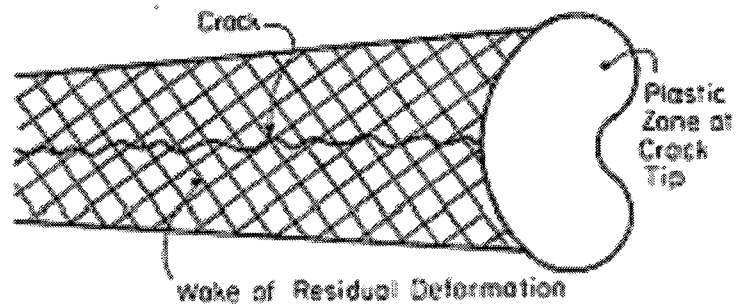


Figure 2.17. Wake of Plastically Deformed Material

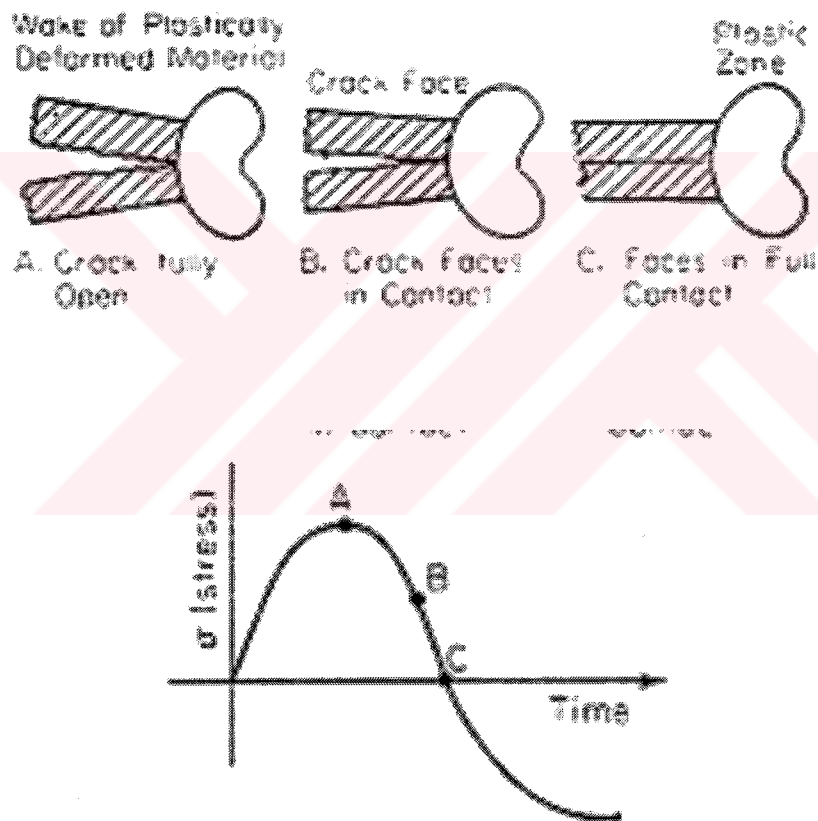


Figure 2.18. Crack Closure Phenomena

Elber further introduced the idea of a crack –opening stress. This is the value of applied stress at which the crack is just fully open, σ_{op} . He suggested that for fatigue crack growth to occur, the crack must be fully open [5]:

$$\Delta K_{eff} = K_{max} - K_{open}$$

$$\Delta K = K_{max} - K_{min}$$

as

(2.15)

$$K_{open} > K_{min}$$

consequently,

$$\Delta K > \Delta K_{eff}$$

Therefore, an effective stress intensity factor range, ΔK_{eff} , which is smaller than ΔK , should be used in fatigue crack growth predictions [5].

$$\frac{da}{dN} = f(\Delta K_{eff}) \quad (2.16)$$

Elber proposed that ΔK_{eff} accounts for the R effect on growth rates. At higher values of R, less crack closure results and ΔK_{eff} becomes closer to ΔK because K_{open} approaches K_{min} . This results in the crack being subjected to a greater range of loading. He obtained the empirical relationship :

$$\Delta K_{eff} = U \Delta K$$

$$U = \frac{\Delta K_{eff}}{\Delta K} = 0.5 + 0.4R \quad (2.17)$$

It is important to note that equation 2.17 is only valid when $R > 0$.

2.5.6. Fatigue Crack Growth Under Variable Amplitude Loading And Overloading

Fatigue is a practical problem for all kinds of structures subjected to a spectrum of many load cycles under normal service utilization conditions. A single load application need not to be harmful, but a repetition of many load cycles can initiate a fatigue crack. The crack will grow until collapse of the structure, unless it is found by inspection.

Fatigue crack growth tests and predictions performed under constant amplitude loading often differ considerably from variable amplitude loading conditions [19]. In contrast to constant amplitude loading where the increment of crack growth, Δa is dependent only on the present crack size and applied load, under variable amplitude loading the increment of fatigue crack growth is also dependent on the preceding cyclic loading history. This is known as load interaction. Load interaction or sequence effects significantly affect the fatigue crack growth rate and consequently, fatigue lives.

2.5.6.1. Load Interaction Effects

In the early 1960s, interaction effects were first recognized [19]. The application of a single overload was observed to cause a decrease in the crack growth rate as shown in Figure 2.19. This phenomenon is termed *crack retardation*. If the overload is large enough, crack arrest can occur and the growth of the fatigue crack stops completely [19].

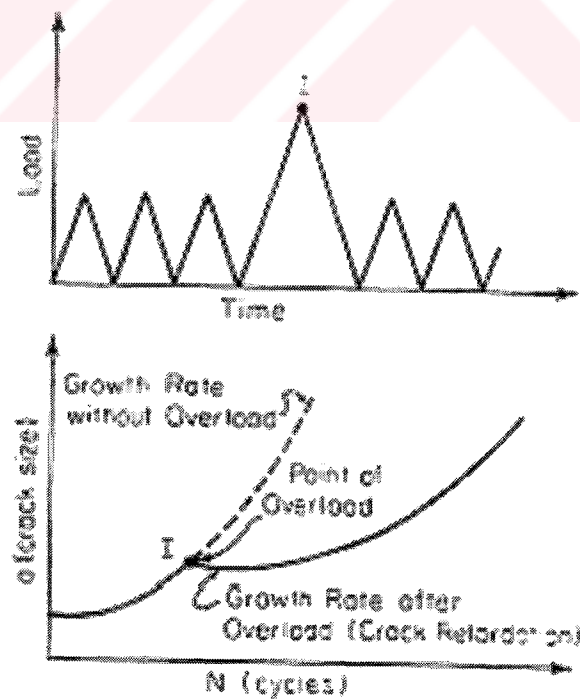


Figure 2.19. Crack Growth Retardation After an Overload

Crack retardation remains in effect for a period of loading after the overload. The number of cycles in this period has been shown to correspond to the plastic zone size developed due to the overload. The larger the overload plastic zone, the longer the crack growth retardation remains in effect [3].

The crack growth rate does not reach a minimum immediately after the overload is applied. Rather, the minimum is reached after the crack has grown a distance approximately one-eighth to one-fourth of the distance into the overload plastic zone. This behaviour is known as *delayed retardation* [19].

A single compressive overload (termed underload), shown in Figure 2.20, generally causes an acceleration in the crack growth rate. In addition, when an underload follows an overload, as shown in Figure 2.21, the amount of crack growth retardation is significantly diminished [3].

In high-to-low loading sequences (Figure 2.21), crack retardation also occurs. Retardation increases for an increasing number of tensile overloads, P , until a limiting "saturation" of overloads is reached. Spacing (M in Figure 2.21) between overloads is critical for this trend. Overloads too closely spaced will eliminate the benefits of crack retardation as the material response tends toward constant amplitude crack growth corresponding to the overloads [3].

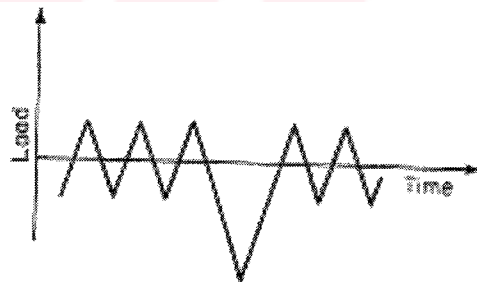


Figure 2.20. Single Compressive Overload

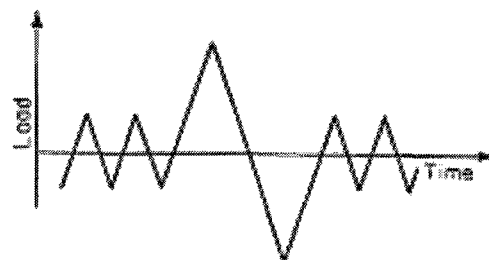


Figure 2.21. Overload Followed by an Underload

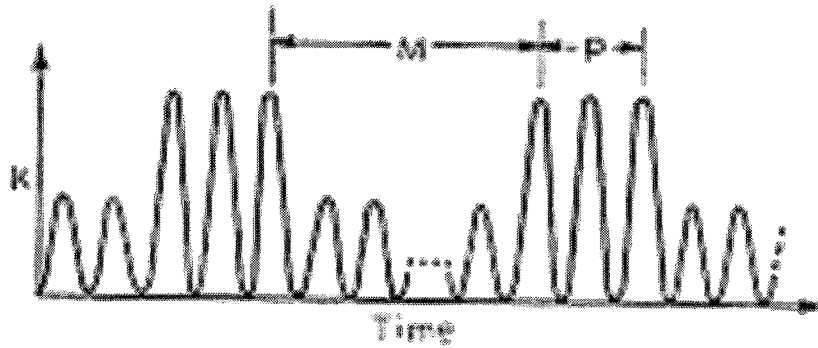


Figure 2.22 Spacing, M , Between multiple Overloads, P , is critical in Producing Maximum Retardation

Finally, low-to-high sequences (Figure 2.23) can cause crack growth acceleration. Fortunately, this acceleration stabilizes quickly when compared to retardation effects [3].

It should be noted that periodic overloads are not always beneficial. In some low cycle fatigue tests, periodic overloads have also been found to cause crack growth acceleration.

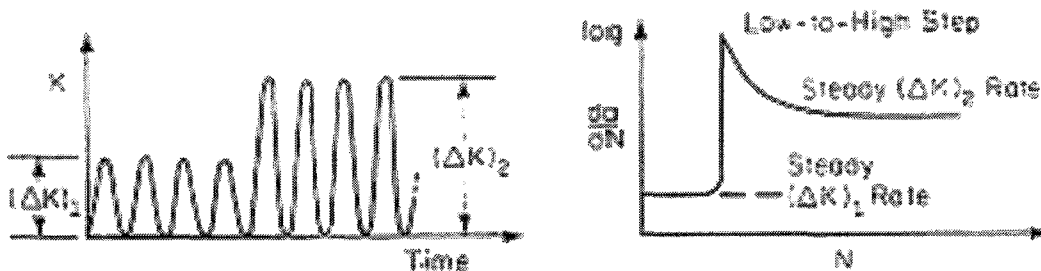


Figure 2.23. Acceleration in Crack Growth Rate Due to Increase in Load

3. NON DESTRUCTIVE TEST METHODS USED FOR DAMAGE INSPECTION

3.1. Purpose Of Non Destructive Test Methods

The aviation industry, because of its very nature, demands a high degree of safety and reliability. Non-destructive testing (NDT) is a science and technology which allows examination of structural members in such a manner that integrity can be determined without disturbing the present condition [20].

Non-destructive inspection has been an integral part of aviation maintenance safety for more than 60 years. NDI encompasses a wide spectrum of inspection disciplines and procedures that range from the conventional to the truly unique. NDI has experienced unparalleled advancement over the last quarter century in both application and innovation. The contribution non-destructive evaluation has made to aviation safety surpasses any existing technology to date [20].

Defects, which may lead to failure, can be present in raw materials prior to part or component manufacturing. Defects can be introduced into materials during manufacturing processes such as forming, bending, drilling, machining, and heat treating. One goal of NDT is to locate these defects in the early stages of production before the part or component goes into service [7].

Once the aircraft, part or component is placed in service, vibrations, structural load stresses, and temperature variations can induce defects, too small to be seen, which will continue to grow in size. The aircraft technician must be equipped with the means to detect these flaws in their early stages so they can be eliminated or repaired before it results in a complete structural failure. This type of testing is referred to as in-service NDT [20].

Non-destructive inspection implies a method that will detect defective or unservicable items while avoiding damage to useful parts and assemblies.

3.2. Important Non Destructive Test Methods And Applications

Non-destructive tests begin with a visual examination. There are many visual inspection aids and techniques which can be employed. Below is a brief overview of the commonly used NDT methods [20].

- **Visual Inspection (VT):** Visual inspection is the most common form of examination and consists of viewing the area with the eye, aided by magnifying glass, borescope, light source, etc. Most defects are detected by visual inspection in aircraft maintenance.
- **Liquid Penetrant Testing (PT):** Liquid penetrant testing is used to detect small cracks or discontinuities open to the surface which may not be evident by normal visual inspection. Penetrant inspection can be used on most airframe parts and assemblies accessible for its application. The inspection is performed by applying a liquid which penetrates into the surface defects. Excessive penetrating liquid is removed and suitable developers applied to draw the penetrant from the surface defects so that visual indications are obtained by color contrast or fluorescence of the penetrant under the influence of "black light".
- **Magnetic Particle Testing (MT):** Magnetic particle testing will indicate surface or near-surface defects in ferro-magnetic parts (parts which can be magnetized) and may be performed on assembled or disassembled parts. The test is accomplished by inducing a magnetic field in the part and applying a dry powder or liquid suspension of iron oxide particles. Local magnetic poles formed by defects in the part attract the oxide particles. Local magnetic poles formed by defects in the part attract the oxide particles so they may be viewed and evaluated by color contrast or fluorescence under "black light".
- **Eddy Current Testing (ET):** Eddy current testing is used to detect surface or near-surface defects in most materials, measure coating thicknesses, and to sort metals by their alloys or heat treat conditions. It can be applied to aircraft parts or assemblies where the defective area is accessible to contact by the eddy current probe. The inspection is performed by inducing eddy currents into a part and electronically observing variations in the induced field. The character of observed field change is interpreted to determine the nature of the defective condition.

- **Ultrasonic Testing (UT):** Ultrasonic testing is a non-destructive test method suitable for the inspection of most metals, plastics, and ceramics for surface or subsurface defects. Ultrasonic inspection requires at least one surface of the part to be accessible in the vicinity of the area in question. The inspection of airframe structure is accomplished by inducing ultrasonic energy (high frequency sound) into the part by a contacting probe and picking up reflections of this sound from within the part. The detected ultrasonic reflections are electronically displayed on a cathode ray tube (CRT) and interpreted for indications of defects.
- **Radiographing Testing (RT):** Radiographing testing will show internal and external structural details of all types of parts and materials. It is a non-destructive test method and is used for the inspection of airframe structure inaccessible or unsatisfactory for the application of other non-destructive test methods. It is accomplished by passing X-ray or Gamma radiation through the part or assembly being tested to expose radiographic film. The processed film shows the structural details of the part assembly by variations in film density. Interpretation of the radiograph will indicate defects or damage.
- **Ultrasonic Excited Thermography:** Ultrasonic Excited Thermography is an efficient, rapid and reliable non-destructive technique developed a few years ago. This technique provides defect selective imaging using thermal waves generated by elastic waves. The mechanism involved is local friction or hysteresis which turns a variably loaded defect into a heat source which is identified thermography even in the presence of complicated intact features. It allows fast measurements with a good reproducibility .

Elastic waves sent into the component propagate inside the sample until they are converted into heat. A defect causes locally enhanced losses and consequently selective heating up. Therefore modulation of the elastic wave amplitude results in periodical heat generation so that the defect is turned into a local thermal wave transmitter. Its emission is detected via the temperature modulation at the surface which is analyzed by lockin thermography tuned to the frequency of amplitude modulation. The amplitude image displays the efficiency of mechanical losses, so it shows the imaginary part of local elastic properties.

Reliable inspection techniques are required for the maintenance of safety relevant structures (e.g. aerospace equipment and vehicles) where one needs to detect defect areas early enough to prevent catastrophic failure. As many structures consist of carbon fibre reinforced plastics (CFRP), the rapid and remote identification of delaminations, impact damages, ruptures, and cracks is a topic of major concern. Therefore a method is required that is applicable during inspection procedures to monitor the integrity of such structures.

The Infrared Thermography for Non-Destructive Evaluation implies to introducesome energy into an object. An infrared camera detects the relevant temperature difference on the surface, thus focusing mainly on the investigation of the subsurface features of the material. The major advantage of this method is its ability of fast inspection.

In ultrasonic excited thermography, sound wave of 20KHz is injected into the sample through the ultrasonic transducer system and part of this sonic excitation causes the interface of the crack, in the specimen, to 'slap' and 'rub' against each other and hence establishing a local source of heat with respect to the surroundings.

After applying the excitation we see the heat dissipation which is produced in the sample through an Infrared Camera System, which is recording the experiment. This Camera registers small temperature changes. This means that where the temperature has increase, there is some damage in the plate.

It must be understood that there is no test which best suits all applications. Having a good working knowledge of each method available and be thoroughly familiar with aircraft parts, components, materials, and their functions to determine the best method or methods for the type or types of defects to be located. All NDT methods are compared in Table 3.1 [20].

A discontinuity is an interruption in the normal physical structural configuration of a part such as cracks, forging laps, folds, seams, porosity, etc. A discontinuity may or may not affect the usefulness of the part. When non-destructive tests are performed, a discontinuity will produce an indication. It is not known whether the indication is a defect until the indication has been evaluated by comparison to some criteria [20].

Fatigue cracks and corrosion are the most common failures or encountered during non-destructive inspection. Fatigue cracks only occur in parts that have been in service under repeated stress reversals or stress variations. The crack starts at a highly stressed area and propagates through the section until failure occurs. A fatigue crack will start more readily where the design or surface condition provides a point of stress concentration, such as fillets, poor surface finish, seams, grinding cracks, and from fastener holes that have a poor hole wall finish or sharp burrs.

Table 3.1. Comparison Of All NDT Methods

METHOD	ADVANTAGE	DISADVANTAGE
VISUAL	1. Low cost 2. Portable 3. Immediate results 4. Minimum Part Preparation	1. Suitable only for surfaces which can be viewed. 2. Generally detects only larger defects. 3. Misinterpretation of cracks and scratches.
LIQUID PENETRANT	1. Low cost 2. Portable 3. High sensitivity 4. Immediate results 5. Fast on multiple parts 6. Minimum skill required to perform	1. Can only inspect surface of parts accessible to penetrant application. 2. Defects must be open to surface. 3. Part preparation, such as removal of finishes and sealant required. 4. Permanent test results not always possible. 5. Direct visual detection of results required. 6. Requires a high degree of cleanliness for satisfactory inspection. 7. Messy.
MAGNETIC PARTICLE	1. Semiportable 2. Sensitive to small indications 3. Detects surface and near-surface defects 4. Sensitive to inclusions as well as cracks. 5. Fast on individual parts. 6. Moderate skill required to perform.	1. Only suitable for ferro-magnetic material. 2. Part must be physically and visually accessible to perform test. 3. Removal of some surface coatings and sealants required. 4. Inspection is directional requiring a general orientation of field to defect. 5. Permanent test results not always available. 6. Not usable in areas where a strong magnetic field may damage instruments. 7. Part must be demagnetized after inspection.
EDDY CURRENT	1. Portable 2. Moderate cost 3. Immediate test results. 4. Sensitive to small indication. 5. Little part preparation. 6. Permanent record easily recorded. 7. Automation possible. 8. No couplant required.	1. Defects surface and near surface defects. 2. Surface to be inspected must be accessible. 3. Rough surfaces interfere with test sensitivity. 4. Suitable for inspection of conductive material only. 5. Considerable skill and experience required. 6. Time Consuming to scan large areas.

ULTRASONIC	1. Portable 2. Suitable for surface and subsurface defects. 3. Sensitive to small defects. 4. Immediate test results. 5. Little part preparation. 6. Wide range of material thicknesses can be inspected. 7. Not limited to metals. 8. Permanent records easily recorded.	1. Surface of part to be inspected must be accessible. 2. Rough surface interfere with test results. 3. Couplant required. 4. Test method is directional. 5. High degree of skill and experience required. 6. Thin or complex parts are difficult to inspect.
RADIOGRAPHIC X-RAY	1. Portable 2. Ability to inspect for both internal and surface defects. 3. Ability to inspect parts covered or hidden by other parts or structure. 4. Permanent test record obtained. 5. Minimum part preparation required.	1. High initial cost. 2. High set-up and preparation time. 3. Radiation hazard. 4. Test method is highly directional. 5. High degree of skill and experience required for set-up, and radiographic interpretation. 6. Film processing equipment required. 7. Subject to regulation by Nuclear Regulatory Commission.

4. DESIGN PRINCIPLES IN AIRCRAFT STRUCTURES

4.1. Concept Of Designing Aircraft Structures

In design for fatigue and damage tolerance, one of two initial assumptions is often made about the state of the material. Both of these are related to the need to invoke continuum mechanics to make the stress, strain, fracture mechanics analysis tractable [21]:

- The material is an ideal homogeneous, continuous, isotropic continuum that is free of defects or flaws.
- The material is an ideal homogeneous, isotropic continuum but contains an ideal crack-like discontinuity that may or may not be considered a defect or flaw, depending on the entire design approach.

The former assumption leads to either the stress-life fatigue design approach. These approaches are typically used to design for finite life or "infinite life". Under both assumptions, the material is considered to be free of defects, except insofar as the sampling procedure used to select material test specimens may "capture" the probable "defects" when the specimen locations are selected for fatigue tests [22].

This often has proved to be an unreliable approach and has led, at least in part, to the damage-tolerant approach.

Another possible difficulty with these assumptions is that inspectability and detectability are not inherent parts of the original design approach. Rather, past and current experience guide field maintenance and inspection procedures, if and when they are considered [7].

The damage tolerant approach is used to deal with the possibility that a crack-like discontinuity will escape detection in either the initial product release or field inspection practices [13]. Therefore, it couples directly to non-destructive inspection and evaluation. In addition, the potential for initiation of crack propagation must be considered an integral part of the design process, and the subcritical crack growth characteristics under monotonic, sustained, and cyclic loads must be incorporated in the design. The final instability parameter, such as plane strain fracture toughness (K_{Ic}), also must be incorporated in design [15]. The damage tolerant approach is based on the ability to track the damage throughout the entire life cycle of the component or system. It therefore requires extensive knowledge of the above issues, and it also requires that fracture or damage mechanics models be available to assist in the evaluation of potential behaviour. As well, material characterization procedures are needed to ensure that valid evaluation of the required material “property” or response characteristic is made. Non-destructive Inspection must be performed to ensure that probability of detection determinations are made for the Non-destructive Inspection procedures to be used. This approach has proved to be reliable, especially for safety-critical components.

4.2. Static Strength – Safe Life

The safe-life, infinite-life philosophy is the oldest of the approaches to fatigue. Examples of attempts to understanding fatigue by means of properties, determinations, and representations that relate to this method include August Wöhler’s work on railroad axles in Germany in the mid-1800s [21]. The design method is stress-life, and a general property representation would be S-N (stress versus log number of cycles to failure) [22]. Failure in S-N testing is typically defined by total separation of sample. General applicability of the stress-life method is restricted to circumstances where continuum, “no cracks” assumptions can be

applied [23]. The stress-life approach seems best applied to components that look like the test samples and are approximately the same size. Much of the technology in application of this approach is based on ferrous metals, especially steels. Other materials may not respond in a similar manner.

The failure criterion for samples are used is usually the detection of a small “crack” in the sample or some equivalent measure related to a substantive change in load-deflection response, although failure may also be defined by separation.

The discovery of fatigue cracks, and failure of major structural elements on airplanes in the late 1930's and early 1940's forced the development of the safe-life design principle [15]. The safe-life design principle requires that major structural elements be replaced after a fixed number of flight cycles. These parts can not be repaired or refurbished to extend the components life. The basis for safe-life design is fatigue analysis.

The safe-life design principle is very conservative. No cracking is allowed during a component's design life [7]. Safe-life structures require extensive fatigue testing. This is expensive and time consuming.

These requirements result in structures which are heavier than parts designed to fail safe design principles. Therefore, safe-life design is usually limited to structures which can not be fail-safe, or where adequate inspections are impractical. One example is landing gear components.

4.3. Fail Safe

The fail-safe design principle uses multiple load paths to ensure structural integrity [13]. If one load path cracks completely through, or sustains accidental damage, the remaining load paths carry the additional load. This type of design is common on modern jet airplanes. Examples include:

- Multiple stringers and ribs in wings
- Multiple wing panels
- Multiple stringers and frames in fuselage construction. This construction also breaks the fuselage skin into redundant panels.

- Bonded and bolted fittings (often called 'back to back' fittings), and bonded and bolted landing gear beams.

Static strength analysis of these structures, with one element severed, satisfies the certification requirement for fail-safety [23]. The loads for fail-safe design conditions are called 'fail-safe loads'. These usually correspond to limit load conditions.

The fail-safe principle also requires that any damage will be detected during an inspection, and then repaired. Some types of damage produce effects that are obvious, such as flapping fuselage skin panels, or wing tank leaks. This obvious damage is considered part of the fail-safe inspections.

Fail-safe design is a good philosophy, and worked well for many decades. In fact, fail-safe design still provides the basis for most new airplane designs. However, operational experience shows that some of the assumptions of fail-safety do not hold true.

Cracks usually develop in several elements at the same time, making the alternate load paths weaker. This is called multiple site cracking. We cannot always rely on obvious damage to alert operators to structural problems before they become catastrophic. Corrosion weakens alternate load paths, and accelerates crack growth.

To compensate for these deficiencies in fail-safe design, the damage tolerance philosophy was developed.

4.4. Damage Tolerance

The damage tolerance principle requires that any airplane damage be detected, and repaired before the strength is below a minimum level [22]. Achieving this goal requires a combination of design, analysis, and inspection methods.

Fail-safe structure forms the basis for damage tolerant design. Damage tolerance improves on fail-safety by considering multiple site cracking, and the residual

strength of partially failed structural elements. Damage tolerance also considers the effects of environmental (corrosion) damage, and discrete (accidental) damage.

Damage tolerant analysis begins with the same static strength calculations as non-damage tolerant structure. This includes analysis of ultimate, limit, and fail-safe design conditions [13].

Crack growth prediction assumes initial cracks which are too small to detect. We then predict how much the cracks grow during normal airplane operations. When the cracks grow to a certain size, you can detect them.

Residual strength predictions tell how the presence of the fatigue cracks affect the strength of fail-safe elements. They also tell how the presence of undetected accidental damage affects the strength of remaining fail-safe elements.

We use the crack growth and residual strength predictions to establish maximum crack sizes. We also establish how many inspection opportunities exist to find a given crack. This ensures that airplane damage is detected, and repaired before the strength is below a minimum level.

Inspection intervals for damage tolerant structure are based on the crack growth and residual strength predictions [23]. The inspections find fatigue cracks, corrosion, and accidental damage before the strength is below a minimum level.

The inspection intervals depend upon the inspection method you choose. A method which can detect very small fatigue cracks (high frequency eddy current, for example), allow a longer inspection interval. Methods which can only detect larger cracks require shorter inspection intervals.

4.4.1. Residual Strength

The residual strength of a member is the minimum load an aircraft must sustain with damage present and without endangering safety of flight or degrading performance of the aircraft for the specified minimum period of unrepaired service usage. Damage present is the length of the crack in the critical structural component. A critical structural component is an aircraft structure whose failure will

either cause the plane to crash or not successfully complete its mission. Unrepaired service usage is a time period dependent on the categorization of the inspectability of the particular structure.

4.4.1.1. Material Property

One way to measure the residual strength of the material is its' fracture toughness [14]. Different materials have different fracture toughness. Usually, the yield strength of a material is inversely related to the fracture toughness. For example 7000 series aluminum alloys have higher strengths than 2000 series aluminum alloys, but the 2000 series alloys have higher fracture toughness [7].

Aluminum alloys within a particular series can also have large differences in fracture toughness. Since damage tolerance was adopted, new aluminum alloys with improved fracture toughness have been developed.

Similarly, steel or titanium alloys with higher yield strength usually have lower fracture toughness. These are general guidelines, and we should consult actual material data before coming to conclusions about the relative toughness of any materials. Large differences in fracture toughness also occur due to part geometry.

4.4.1.2. Thickness Factor

Fracture toughness depends on the part thickness [22]. A thin part can have twice the fracture toughness of a thick part made from the same material. What defines "thin" and "thick" depends on the material.

The reason for this geometric dependence is a difference in the failure mode. Thin parts fail in a condition of plane stress. This means that the induced, out of plane stress is very small, and does not contribute to material fracture. Thick parts fail in a condition of plane strain. In plane strain, the induced, out of plane stress is larger, and does contribute to material fracture.

There is also a failure mode which is a combination of plane strain. This is called transitional failure [23]. Each material has its' own curve defining the three different failure modes.

The effect of material thickness can be accounted for rather well, because the thickness of the component under consideration will be readily known. In sheets there is a small, but a systematic, effect of thickness on crack propagation. The effect appears to exist primarily before the fracture mode transition. Fatigue cracks in sheets always start as a tensile-mode crack perpendicular to the sheet surface. When the crack grows the size of the plastic zone increases and plane stress can finally develop. This causes the fatigue crack to change to single or double shear. Plane stress develops when the size of the plastic zone is in the order of the sheet thickness. Therefore it is conceivable that the thickness effect is related to the fracture mode transition.

Crack propagation in very thick sections encompasses more problems. The cracks may develop as quarter-elliptical corner cracks or semi-elliptical surface flaws. The stress intensity varies along the front of the flaw and its maximum value depends upon flaw shape.

4.4.2. Damage Detection Period

The purpose of damage tolerance inspections is to discover damage, before the structures residual strength capability falls below the residual strength requirement. Residual strength prediction tells what size the cracks correspond to minimum strength (critical crack length). Crack growth prediction tells how quickly cracks will grow to the critical length.

We choose the inspection interval so that a crack smaller than the detection threshold cannot grow to critical length before the next inspection period. This is called the damage detection period.

The detectable crack size depends on the inspection method. Some methods can find very small cracks (high frequency eddy current, for example). Other methods can only find larger cracks (general visual inspection, for example). The detectable

crack size (and therefore the inspection method) affects how often the inspection is required.

4.5. Advanced Materials use

A few comments on what ends up the material for a structure should also be made. First is a composition, essentially the basic chemistry of an alloy or the specific components of a composite. Producing the structure may require a few or many steps beyond this chemistry/components combination. Primary processing plays an important role. As examples, an investment cast super-alloy blade will have different characteristics depending whether it is made using an equiaxed, directionally solidified, or single-crystal process; and fiber reinforced composites clearly have numerous wrap/lay configurations that can influence their response [7]. Subsequent thermal treatment for an alloy, or curing conditions for a composite, also contribute to the end product. Many metallic alloys, far from being the uniform homogenous materials often envisioned, are carefully orchestrated arrangements of microconstituents designed to provide specific property balances from these in situ composites [7].

4.5.1. Aluminum Alloy Usage

- 2024 is copper-magnesium alloy that is generally used in the naturally aged (room temperature aged) condition of the T3 and T4 tempers [13]. This alloy has the best fatigue strength and fracture toughness of the structural alloys and is consequently used for tension applications such as lower wing surface and upper surface of the horizontal stabilizer. The body skins are also 2024 to provide maximum toughness [13]. Body skins are clad to provide corrosion resistance in areas left unpainted.
- 7075 is copper, magnesium, zinc aluminum alloy that is often used in the – T6 or peak aged condition for structure requiring high compression and shear strengths such as upper wing surface, lower surface horizontal stabilizer, vertical stabilizer, and body frames and stringers [7]. The same alloy is used in the overaged condition for fittings machined from forgings, plate and extrusions.
- 7178 is a high strength copper, magnesium, zinc aluminum alloy that is used in the peak aged condition for its high compressive strength [7].

Application has been primarily for upper wing surface and keel beam chord. It is no longer used for design because of its susceptibility to exfoliation corrosion.

4.5.2. Titanium Alloy Usage

There are a variety of titanium alloys chosen for specific applications based upon needed material properties and a balance between material/fabrication costs and component weight. The following are the primary alloys, the types of applications, and the reasons for selection.

Commercially pure (Ti) is the lowest strength titanium grade used. CP Ti is used for applications such as scuff plates and pneumatic ducts [13].

- Ti-3Al-2.5V is used for high pressure hydraulic tubing to reduce weight over previously used stainless steel tubing.
- Ti-6Al-4V is the most extensively used alloy. There are three heat treatments used depending upon the specific requirement for component. Applications are floor support structure, landing gear beam, APU firewall, fasteners and hydraulic fittings.
- Ti-6Al-6V-2Sn is higher strength than Ti-6Al-4V which allows more weight savings. It is used in fittings on landing gear and wing area, flap support structure.
- Ti-10V-2Fe-3Al is a newer titanium alloy with higher strength than either Ti-6Al-6V-2Sn or Ti-6Al-4V. This newer alloy is used in forgings, auxiliary slat tracks, and door hinges. Because of its higher strength, it offers more weight savings.

5. APPLICATION OF CRACK ANALYSIS TO AIRCRAFT STRUCTURES

5.1. Models Used In Application Of Crack Analysis

In this chapter, two different kinds of cracks are examined. Analysis of corner crack in three different geometries are done by the use of the computer program NASGRO. Another broad analysis are done for a longitudinal crack in a fuselage panel analytically and compared with the actual test results of the panel.

5.2. Analysis Of Corner Cracks In Three Different Geometries

The problem of corner cracks at a circular hole configuration has received considerable attention from the fracture mechanics community because of its effect on the life of aircraft structures. The first set of test data was presented at the Air Force Conference in 1969. Since then, considerable efforts have been directed at the development of analytical and semiempirical solutions. In the following sub-chapters corner cracks applied in three different geometries as:

- At Hole in Lug
- At Offset Hole in Plate
- At Edge of Plate

are shown.

5.2.1. Corner Crack At Hole in Lug

The pin load is applied at the direction normal to the base of the lug. Either the bearing stress, σ_{br} , or the gross area stress, σ_0 can be used to compute stress intensity. The solution method that involving analysis at two crack tips is rather complicated. In the one-parameter method, the crack shape and is assumed to be constant. Corner crack at a hole in lug shown in Figure 5.1., and all corresponding data about the model and the analysis are shown in Chapter 6- Numerical Results.

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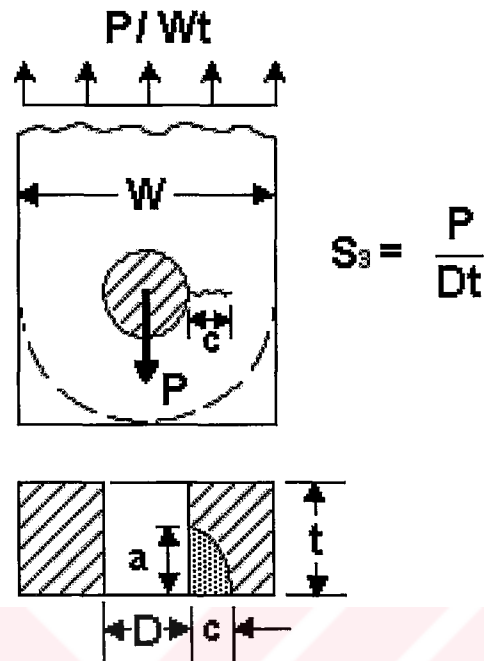


Figure 5.1. Corner Crack – At Hole in Lug

5.2.2. Corner Crack - At Offset Hole in Plate

A part through crack originated on the surface of a plate is usually modeled as one-half of an ellipse. Either the major or the minor axis of the ellipse may be placed on the front surface of the plate, depending on the configuration. The length and depth dimensions of the crack are designated as $2c$ and a , respectively. The crack shape is described by an aspect ratio $a/2c$ or a/c . For a plate subjected to uniform farfield tension, the expression for K , for some point on the periphery of a semielliptical crack in a semi-infinite solid where the applied stress is equated to the applied load (force) over the full cross section of the plate (ignoring the crack).

For a crack in a rectangular plate of finite thickness and width, the solution for K (with a given crack size and shape) should account for the influence of the width and the front and back faces of the plate.

Corner crack at offset hole in plate shown in Figure 5.2., and all corresponding data about the model and the analysis are shown in Chapter 6- Numerical Results.

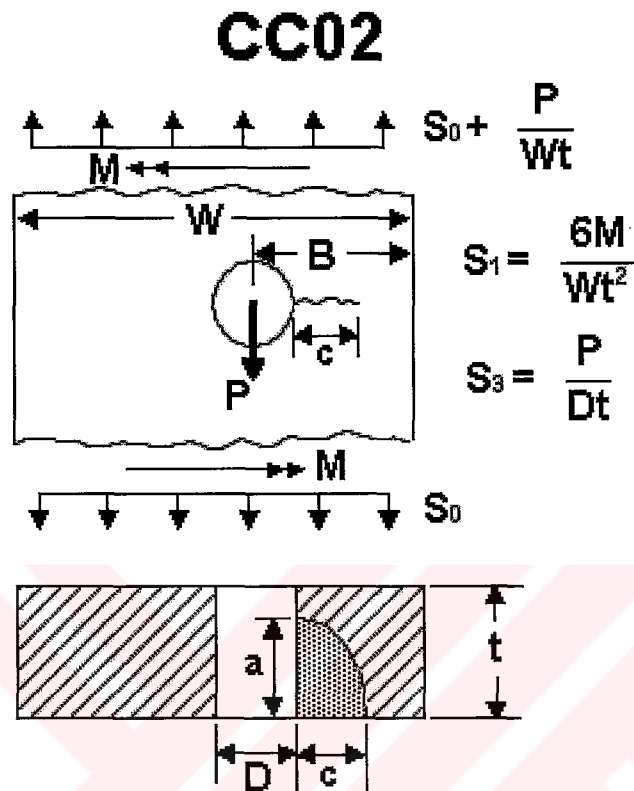


Figure 5.2. Corner Crack – At Offset Hole in Plate

5.2.3. Corner Crack - At Edge of Plate

Corner crack at edge of plate shown in Figure 5.3., and all corresponding data about the model and the analysis are shown in Chapter 6- Numerical Results.

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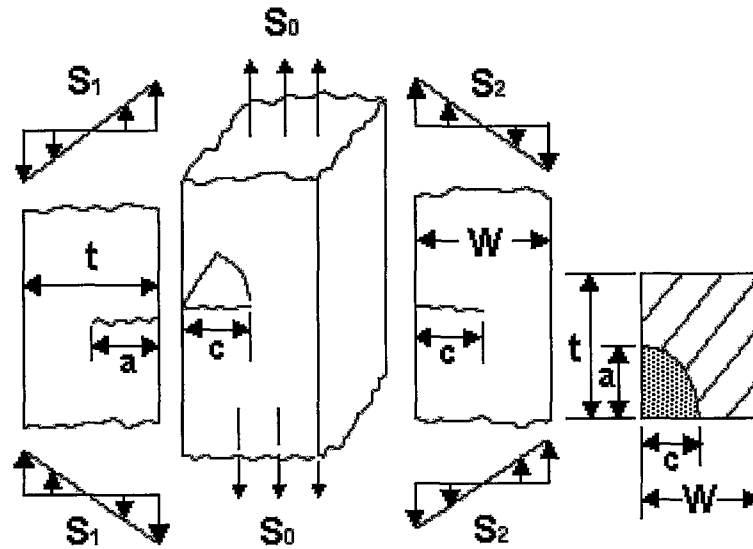


Figure 5.3. Corner Crack – At Edge of Plate

5.3. Analysis Of A Longitudinal Crack In A Curved Stiffened Fuselage Panel

The purpose of this study is to present a simplified semi-analytical methodology to predict the behaviour of longitudinal cracks in cracked stiffened curved panels with frames. Pressure loading acting on the longitudinal cracks induces geometrical non-linearity due to large deflections of crack edges [24]. This methodology is based mainly on the application of “bulging” coefficients to stress intensity factors taken from a numerical model of a stiffened cracked panel in order to get the crack driving force of the curved panel [24]. Analysis of the results shows a highly satisfactory correlation between prediction by calculation and experimental data.

The most well-known people who worked on this type of crack modelling are Chen and Swift. In this type of crack modelling, there are facts that effect the models and the cracks due to the geometry, stress distribution which is non-uniform, and pressure differences to which the panel is subjected due to the inside pressure and the outside pressure [25]. The skin elements are subjected to tension and pressure

loading to simulate fuselage longitudinal loading and cabin pressurization. The boundary loading is constrained to move uniformly in the tension direction.

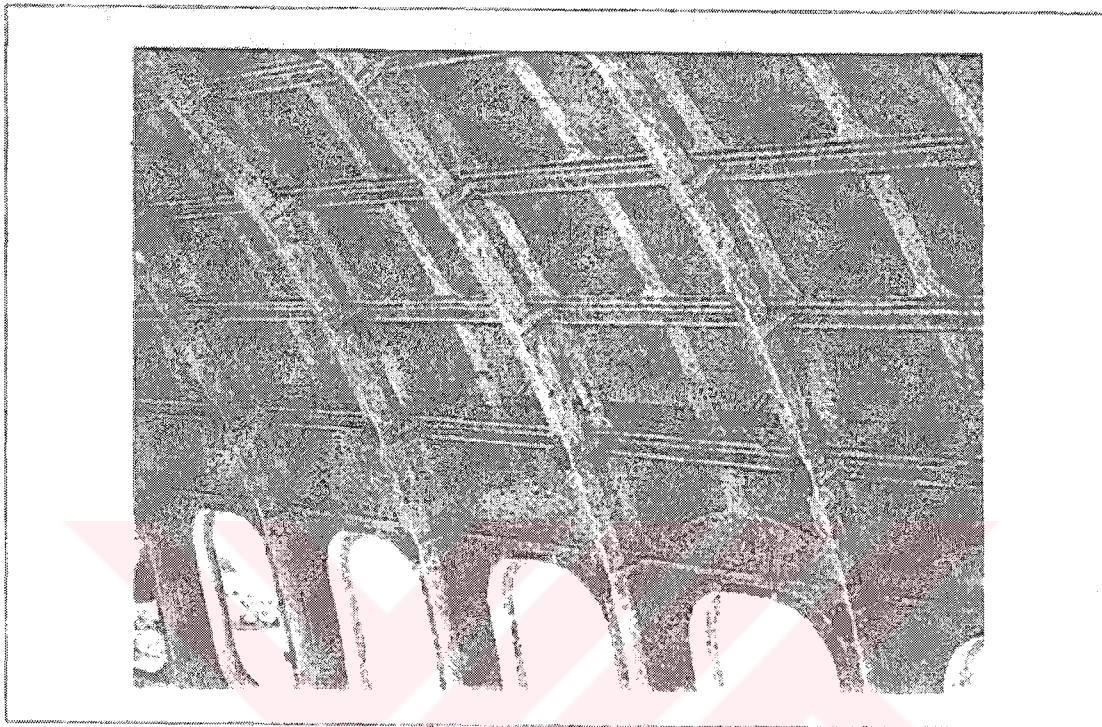


Figure 5.4. Panel Geometry

5.3.1. Bulging Phenomenon

In curved structures subjected to internal pressure, longitudinal crack tips are affected by a phenomenon called the “bulging” effect [26]. This effect, due to pressure loading, creates local bending at the crack tips and increases the stress intensity factor compared with a flat structure subjected to tension loading. That is:

$$K_I(\text{curve}) = K_I(\text{circumferential}) + K_I(\text{local bending}).$$

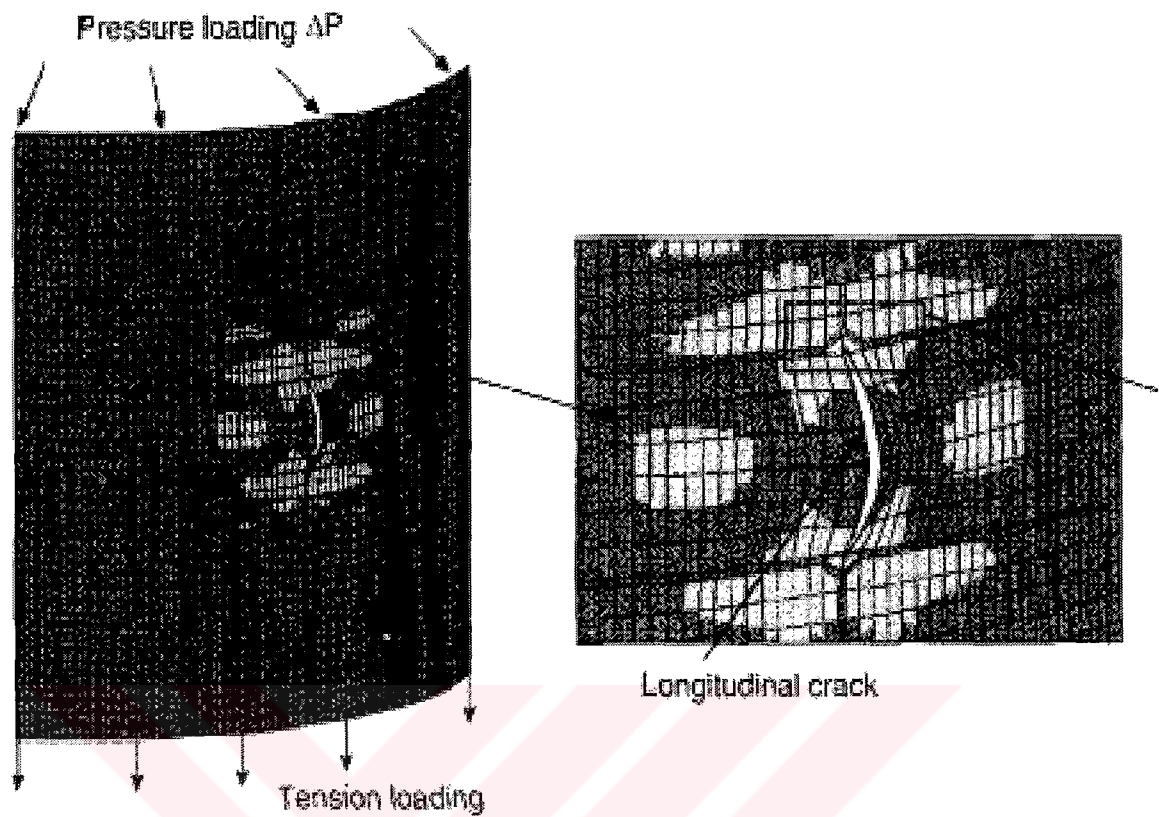


Figure 5.5. Effect of Bulging

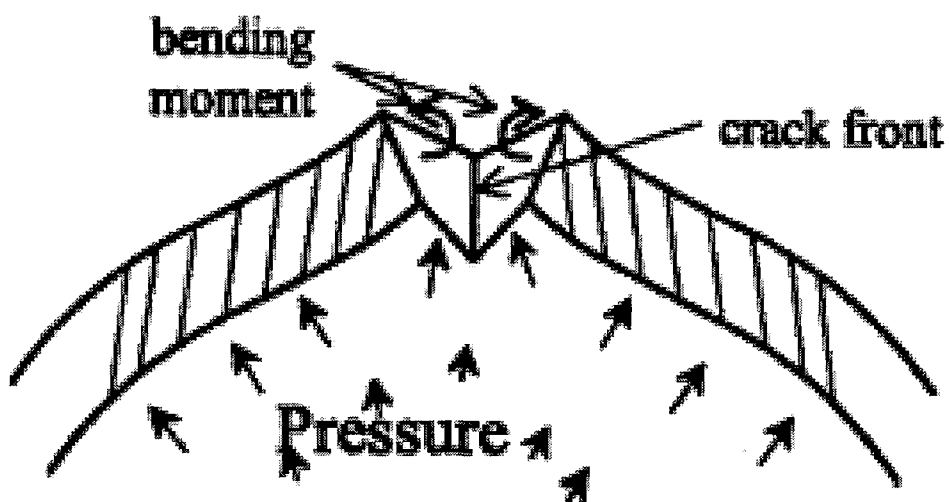


Figure 5.6. Circumferential Section of a Longitudinal Crack

For a longitudinal crack, the frames that stiffen the skin have an influence on the out-of-plane displacements of the crack edges [24]. Swift considers that “bulging” is a maximum when the crack tip is midway between the frames, and tends towards unity when the crack tip reaches the adjacent stiffeners [27]. Based on observations, he considers that the variation of “bulging” coefficients can be approximated by a sinusoidal function of crack length and frame pitch . Swift distinguishes two different configurations:

- a crack midway between two intact frames
- a crack above a broken frame

In figure 5.7., “A” is the crack midway between two intact frames and “B” is the crack above a broken frame [24]. And Figure 5.8. shows a real example of a damaged panel [13].

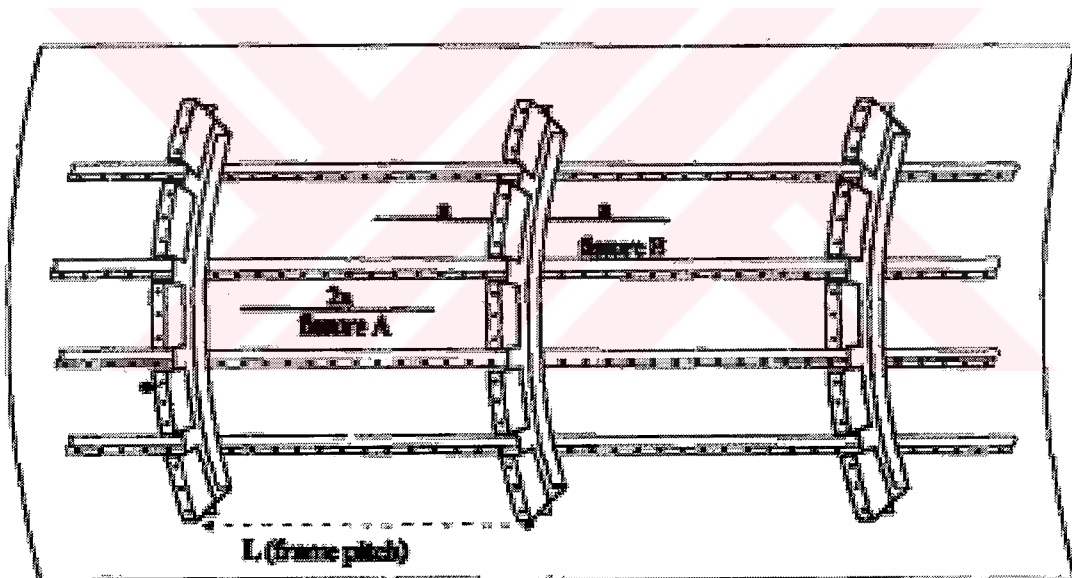


Figure 5.7. A Crack Midway Between Two Intact Frames and a Crack Above a Broken Frame

Figure 5.9. shows the fuselage section with the panel geometry [13].

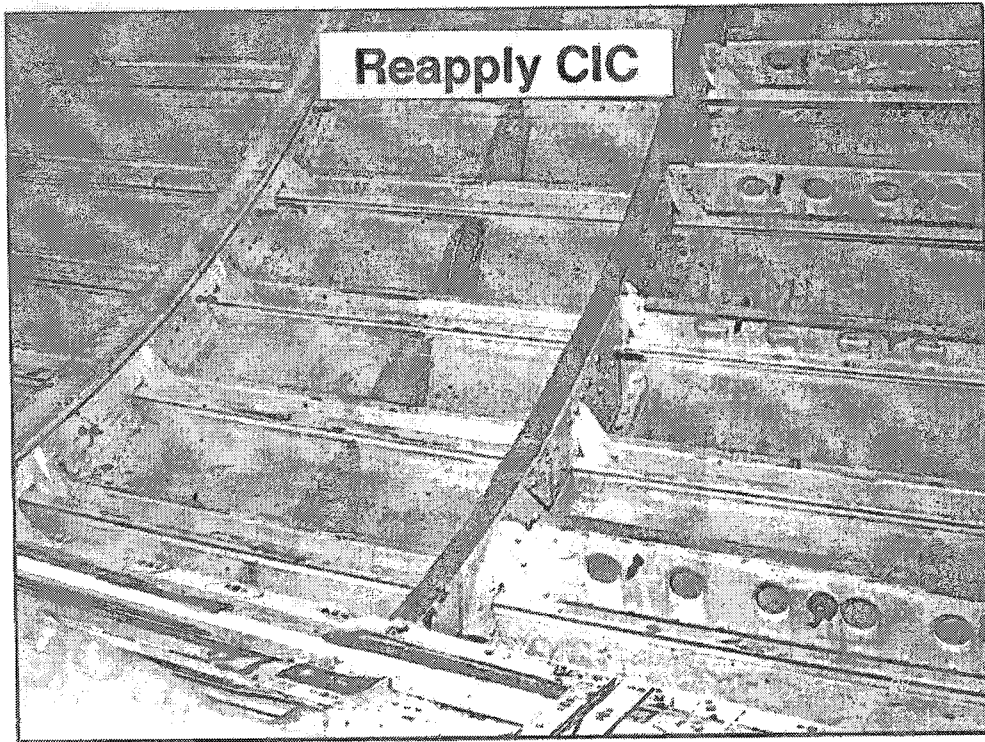


Figure 5.8. Crack Between Two Frames

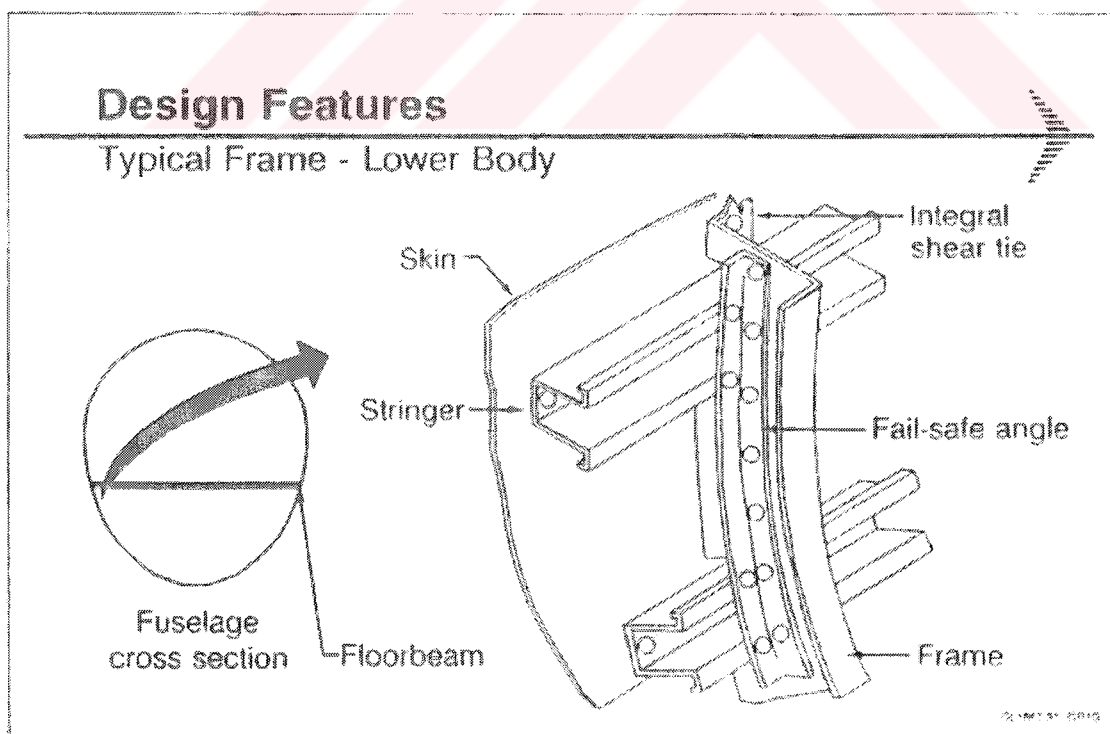


Figure 5.9. Fuselage Cross Section

A crack above a broken frame is taken into account in modelling the longitudinal crack on the fuselage panel. The panel used is IMA-XB5-LC and the material used is Al-2024. The panel has test data of the non-uniform stress distribution. The basic formulation used is the Forman Equation which was explained in Chapter 3. The bulging phenomenon is included in the stress intensity factor calculations by using the formulation of Swift. Bulging coefficient is obtained by both Chen's Formulation and Swift's Formulation and compared to each-other. As a result of comparison Swift's formulation is taken into account for analysis. The crack has an initial length of 37.5 mm in the model. The Crack propagation is calculated and compared with the actual test results. The nomenclature used in this crack analysis is shown in Table 5.1. and the formulations used are shown below. The results obtained are given in Chapter 6-Numerical Results.

Table 5.1. Nomenclature

a (initial)	Half crack length [mm]	37.5
R	Stress ratio	0
T	Integration period	100
R (ρ)	Fuselage radius [mm]	3340
L	Frame spacing [mm]	635
Delta p	Pressure Differential [MPa]	605
E	Modulus of Elasticity [MPa]	71000
e (t)	Thickness [mm]	2
C	Forman Coefficient at R	0.201E-07
S long./S circum. (X)	Biaxiality Ratio	0.33
B_{nugd}	Beta Non-Uniform Stress Distrubution (normalized initial)	0.8406
β_b	Beta Bulging(normalized initial)	1.111793
B_s	Beta Stiffening(normalized initial)	1.017

Forman Equation:

$$\frac{da}{dN} = \frac{C(\Delta K)^m}{(1-R)K_c - \Delta K} \quad (5.1.)$$

$$a_i = a_{i-1} + \Delta a \quad (5.2.)$$

Crack propagation per integration period:

$$\Delta a = \frac{da}{dN} \times T \quad (5.3.)$$

Stress Intensity Factor calculation:

$$K = \beta_{msd} \beta_b \beta_s \sigma_{\max} \sqrt{\pi a} \quad (5.4.)$$

Beta Bulging – Swift Formulation:

$$\beta_b = 1 + \left[10 \frac{a}{R} \cos\left(\frac{\pi a}{2L}\right) \right] \quad (5.5.)$$

Beta Bulging – Chen Formulation:

$$\beta_b = \sqrt{1 + \frac{5ea}{3\pi\rho^2\Delta p} \frac{dw_0}{da}} \quad (5.6.)$$

$$\beta_b = \sqrt{1 + \frac{5ea}{3\pi\rho^2\Delta P} \frac{0.316}{\sqrt{1+18\chi}} \tanh\left(0.06 \frac{\rho}{e} \sqrt{\frac{\Delta Pa}{Ee}}\right)} \quad (5.7.)$$

Beta Stiffening:

$$\beta_s = \frac{1}{2} \left[1 + \cos \pi \left(1 - \frac{2a}{L} \right) \right] \quad (5.8.)$$

6. NUMERICAL RESULTS OF CRACK ANALYSIS TO AIRCRAFT STRUCTURES

6.1. Corner Crack At Hole in Lug

The analysis results of program NASGRO is given below:

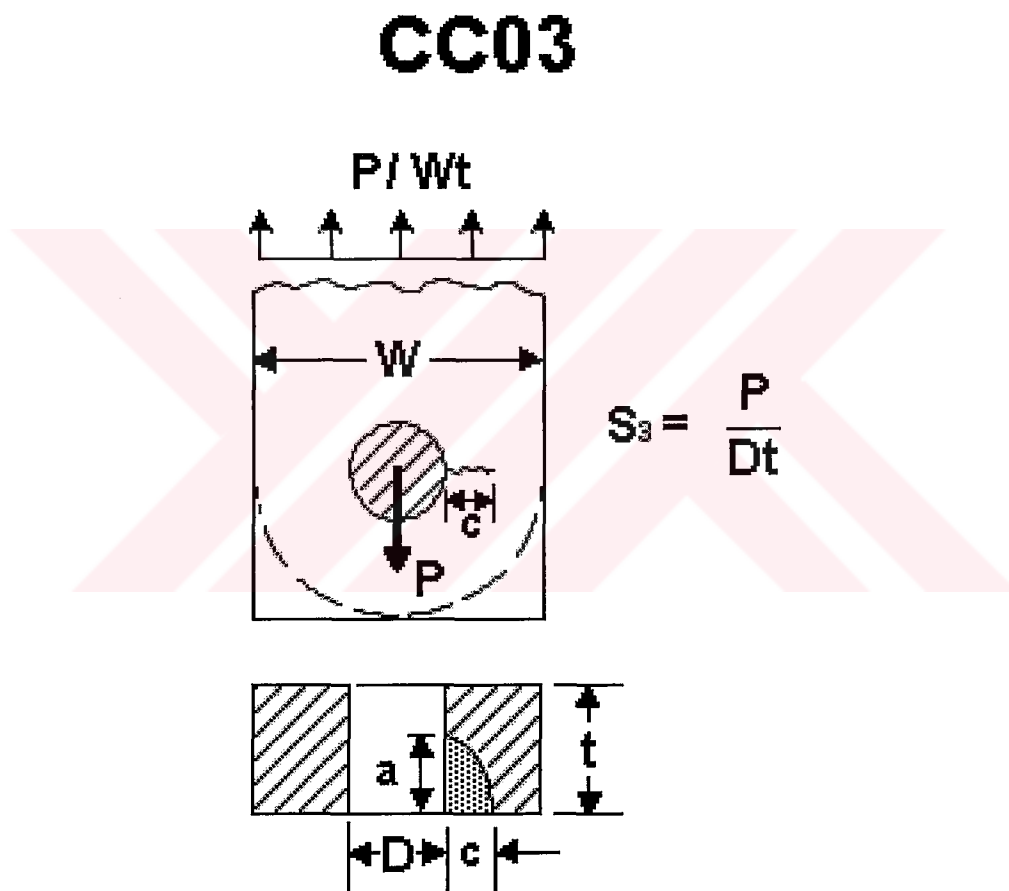


Figure 6.1. Corner Crack – At Hole in Lug

FATIGUE CRACK GROWTH ANALYSIS

DATE: 04-Mar-04 TIME: 12:02:35
(NASGRO Version 4.02, September 2002.)
SI units [mm, MPa, MPa sqrt(mm)]

PROBLEM TITLE

[no title given]

Crack Growth Model: Non Interaction
Ratio of growth increment to current crk size= 0.0050
Equation/Table : NASGRO Equation

GEOMETRY

MODEL: CC03-Corner crack from hole in lug (2D).

Plate Thickness, t = 23.0000
Plate Width, W = 112.0000
Hole Diameter, D = 42.0000

FLAW SIZE: (User specified)

a (init.) = 1.270
 c (init.) = 1.270
 a/c (init.) = 1.000

ANALYSIS RESULTS:

FINAL RESULTS:

Net-section stress exceeds the Flow stress.
(Flow stress = average of yield and ultimate)

at Cycle No. 4414.88
of Load Step No. 3 Description: None
of Block No. 10
of Schedule No. 1
Crack Size c = 33.7167
Total Cycles = 797414.88

FATIGUE CRACK GROWTH ANALYSIS

DATE: 04-Mar-03 TIME: 12:02:35
(NASGRO Version 4.02, September 2002.)
SI units [mm, MPa, MPa sqrt(mm)]

PROBLEM TITLE

[no title given]

Crack Growth Model: Non Interaction
Ratio of growth increment to current crk size= 0.0050
Equation/Table : NASGRO Equation

7050-T7451 Plt & Sht; L-T; DA

• • • • •

1	525.0	100.0	1450.0	1410.0	1.00	0.00	00.0000	1005.5
---	-------	-------	--------	--------	------	------	---------	--------

N O PK1-2H + Alk-2-Quadrant

1000

1 0 1015 11 0 000 0 50 0 50 00 00 0

0 00010

[illegible]

(110070) 1 00 0 1 1 000)

[illegible]

1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57 58 59 60 61 62 63 64 65 66 67 68 69 70 71 72 73 74 75 76 77 78 79 80 81 82 83 84 85 86 87 88 89 90 91 92 93 94 95 96 97 98 99 100

1

1000

II. THE HILBERT-NAGATA FUNCTION

1. *Journal of the American Medical Association*, 1997; 277: 1039-1043.

Book 1, Act 1, Scene 1, Lines 1-10

POLYMER LETTERS

Block Number	Block Case No.
From - To	
1 - 12	1

Stresses: Tension, bending or pin

FATIGUE SCHEDULE BLOCK INPUT TABLE

SINGLE DISTINCT BLOCK

S : M: NUMBER	:	S3	:	S	:
T : A: OF	:		:		:
E : T: FATIGUE	:		:		:
P : L: CYCLES	:	(t1) : (t2)	:	(t1) : (t2)	:
1: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
2: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
3: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
4: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
5: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
6: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
7: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
8: 1:	5000.00 :	0.00:	59.20:	0.00:	0.00:
9: 1:	2000.00 :	0.00:	59.20:	0.00:	0.00:
10: 1:	2000.00 :	0.00:	59.20:	0.00:	0.00:
11: 1:	2000.00 :	0.00:	59.20:	0.00:	0.00:
12: 1:	2000.00 :	0.00:	59.20:	0.00:	0.00:
13: 1:	2000.00 :	0.00:	59.20:	0.00:	0.00:
14: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
15: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
16: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
17: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
18: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
19: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
20: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
21: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
22: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
23: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
24: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
25: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
26: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
27: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
28: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
29: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
30: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
31: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
32: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
33: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
34: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
35: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
36: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
37: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
38: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:

39: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
40: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
41: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
42: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
43: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
44: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
45: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
46: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
47: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
48: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
49: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
50: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:

Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

FATIGUE SCHEDULE BLOCK STRESS TABLE

S : M:	NUMBER :	S3 :	S :
T : A:	OF :	:	:
E : T:	FATIGUE :	(MPa) :	(MPa) :
P : L:	CYCLES :	(t1) : (t2) :	(t1) : (t2) :

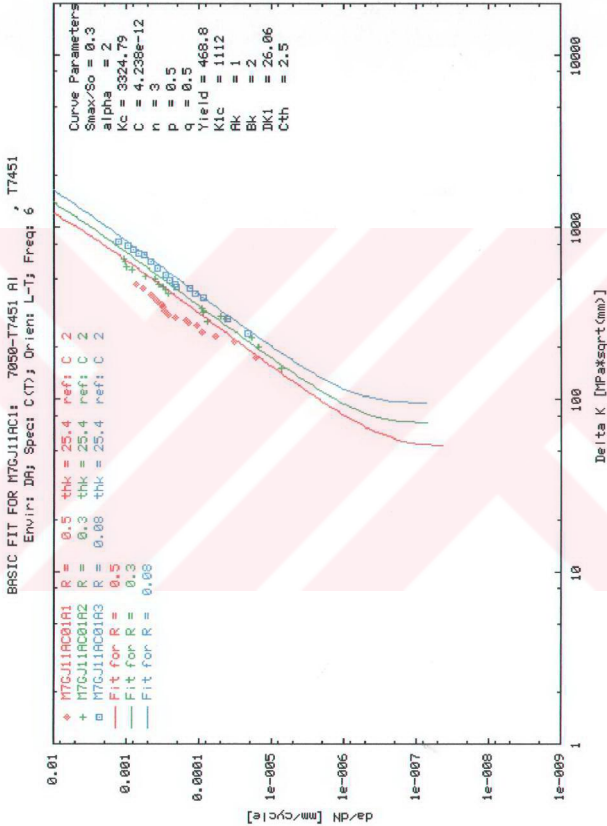
1: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
2: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
3: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
4: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
5: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
6: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
7: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
8: 1:	5000.00 :	0.00:	59.20: 0.00: 0.00:
9: 1:	2000.00 :	0.00:	59.20: 0.00: 0.00:
10: 1:	2000.00 :	0.00:	59.20: 0.00: 0.00:
11: 1:	2000.00 :	0.00:	59.20: 0.00: 0.00:
12: 1:	2000.00 :	0.00:	59.20: 0.00: 0.00:
13: 1:	2000.00 :	0.00:	59.20: 0.00: 0.00:
14: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
15: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
16: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
17: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
18: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
19: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
20: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
21: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
22: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
23: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
24: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
25: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
26: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
27: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
28: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
29: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
30: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:
31: 1:	1000.00 :	0.00:	59.20: 0.00: 0.00:

32: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
33: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
34: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
35: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
36: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
37: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
38: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
39: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
40: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
41: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
42: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
43: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
44: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
45: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
46: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
47: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
48: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
49: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:
50: 1:	1000.00 :	0.00:	59.20:	0.00:	0.00:

Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

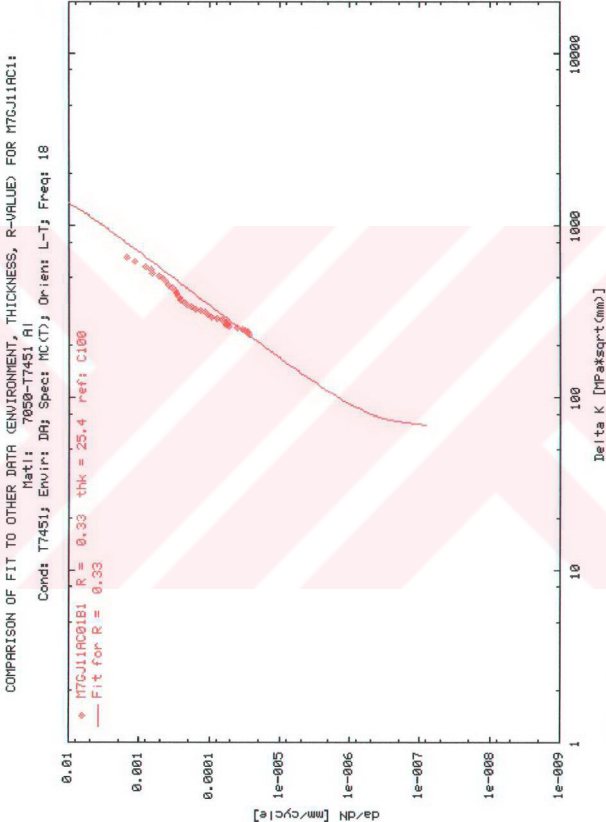


Chart 6.1. da/dN-Delta K



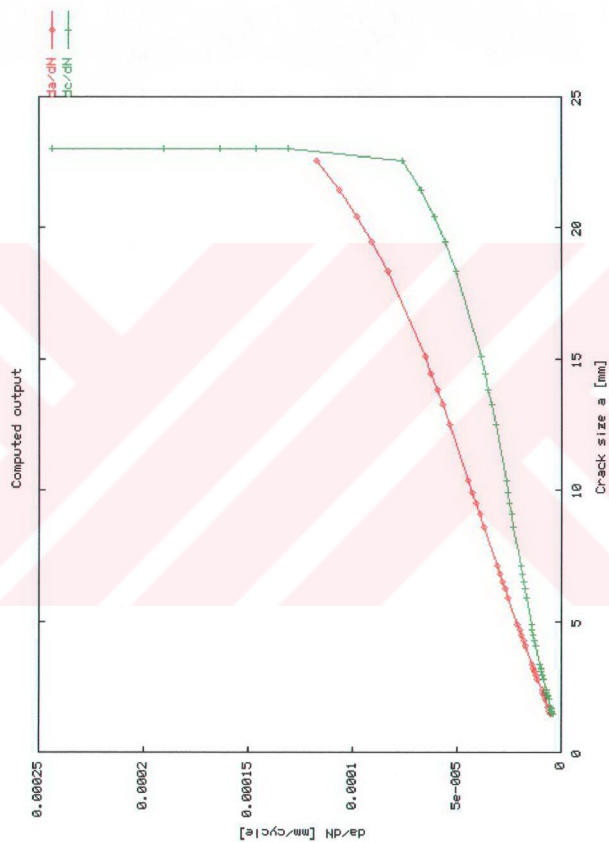
Tue Apr 06 23:02 2004

Chart 6.2. da/dN-Delta K



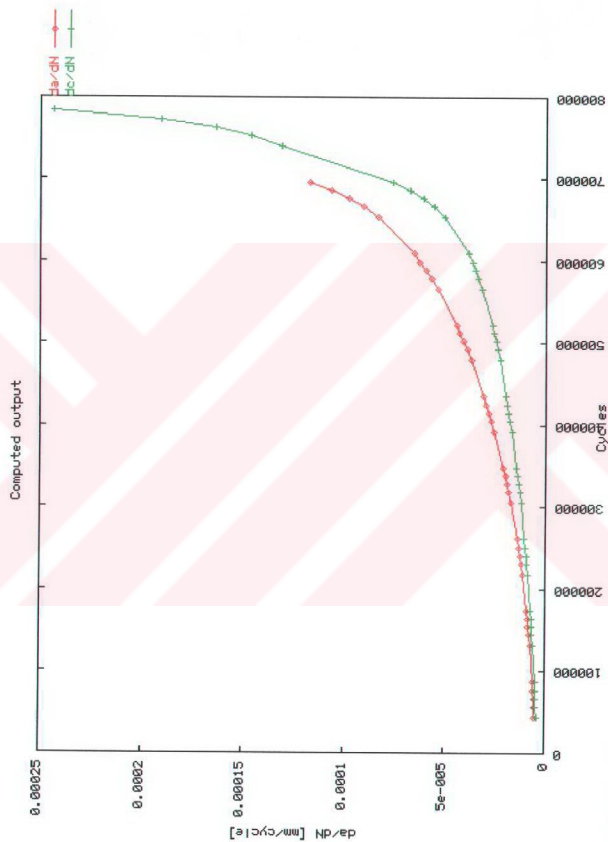
Tue Apr 06 23:04 2004

Chart 6.3. da/dN - a



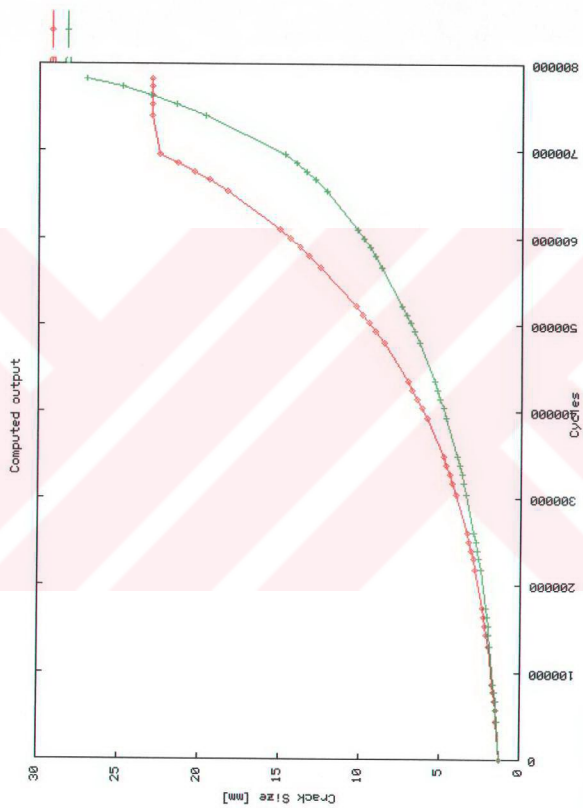
23:26 06Apr2004.

Chart 6.4. da/dN - Cycles



23:12 06Apr2004.

Chart 6.5. Crack Size - Cycles



23:11 06Apr2004.

6.2. Corner Crack At Offset Hole In Plate

The analysis results of program NASGRO is given below:

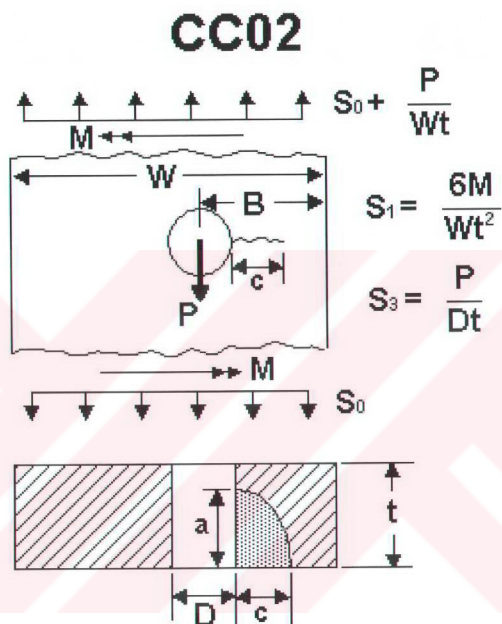


Figure 6.2. Corner Crack – At Offset Hole in Plate

FATIGUE CRACK GROWTH ANALYSIS

DATE: 01-Mar-04 TIME: 10:31:52
 (NASGRO Version 4.02, September 2002.)
 SI units [mm, MPa, MPa sqrt(mm)]

PROBLEM TITLE

[no title given]

Crack Growth Model: Non Interaction
 Ratio of growth increment to current crk size= 0.0050
 Equation/Table : NASGRO Equation

GEOMETRY

MODEL: CC02-Corner crack from hole in plate (2D).

Plate Thickness, t = 10.0000
Plate Width, W = 48.5000
Hole Diameter, D = 7.9000
Edge to Hole Ctr, B = 16.0000
Poisson's ratio = 0.30

Compounding is NOT being used.

FLAW SIZE: (User specified)

a (init.) = 1.270
 c (init.) = 1.270
 a/c (init.) = 1.000

ANALYSIS RESULTS:

FINAL RESULTS:

Net-section stress exceeds the Flow stress.
(Flow stress = average of yield and ultimate)

at Cycle No. 538.66
of Load Step No. 74 Description: None
of Block No. 1
of Schedule No. 1
Crack Size c = 10.9063
Total Cycles = 73538.659

FATIGUE CRACK GROWTH ANALYSIS

DATE: 01-Mar-04 TIME: 10:31:52
(NASGRO Version 4.02, September 2002.)
SI units [mm, MPa, MPa sqrt(mm)]

PROBLEM TITLE

[no title given]

Crack Growth Model: Non Interaction
Ratio of growth increment to current crk size= 0.0050
Equation/Table : NASGRO Equation

MATERIAL

MATL 1:

2024-T351 Plt & Sht; T-L; LA & HHA

Material Properties:

:Matl: UTS : YS : K1e : K1c : Ak : Bk : Thk : Kc : Keac :

```

: No.: : : : : : : : :
:-----:
: 1: 468.8: 358.5: 1355.0: 973.0: 1.00: 0.75: 10.000: 1516.4: :
:
: Matl:----- Crack Growth Eqn Constants -----:
: No.: C : n : p : q : DK1 : Cth+ : Alpha: Smax/:
: : : : : : : : : : : : : : : : : : : : : : : : :
: : : : : : : : : : : : : : : : : : : : : : : : :
:-----:
: 1: 0.109D-10: 3.000: 0.50: 1.00: 25.71: 2.20: 2.00: 0.30:

```

Small crack threshold parameters:
a0 = 0.03810
Limit of ratio DKth(S)/DKth(L) = 0.20

Cth+ = 0, Cth- = 0 are used at start only

FATIGUE CRACK GROWTH ANALYSIS

DATE: 01-Mar-04 TIME: 10:31:52
(NASGRO Version 4.02, September 2002.)
SI units [mm, MPa, MPa sqrt(mm)]

PROBLEM TITLE

[no title given]

Crack Growth Model: Non Interaction
Ratio of growth increment to current crk size= 0.0050
Equation/Table : NASGRO Equation

FATIGUE SPECTRUM

[Note: Stress = Input Value * Scale Factor]

Stress Scaling Factors for Block Case: 1

Scale Factor for Stress S0: 1.0000
Scale Factor for Stress S1: 1.0000
Scale Factor for Stress S3: 1.0000

Schedule info. was input manually

Total No. of Blocks in Schedule = 1

Block Number and Case Correspondences

Block Number	Block Case No.
From - To	
1 - 1	1

Stresses: Tension, bending or pin

FATIGUE SCHEDULE BLOCK INPUT TABLE

SINGLE DISTINCT BLOCK

S : M:	NUMBER	:	S0	:	S1	:
T : A:	OF	:		:		:
E : T:	FATIGUE	:		:		:
P : L:	CYCLES	:	(t1) :	(t2) :	(t1) :	(t2) :

1:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
2:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
3:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
4:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
5:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
6:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
7:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
8:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
9:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
10:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
11:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
12:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
13:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
14:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
15:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
16:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
17:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
18:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
19:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
20:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
21:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
22:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
23:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
24:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
25:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
26:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
27:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
28:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
29:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
30:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
31:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
32:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
33:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
34:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
35:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
36:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
37:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
38:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
39:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
40:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
41:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
42:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
43:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:
44:	1:	:	1000.00 :	51.90:	0.00:	0.00: 0.00:

64: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
65: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
66: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
67: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
68: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
69: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
70: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
71: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
72: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
73: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
74: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
75: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
76: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
77: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
78: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
79: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
80: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:

[no title given]
MODEL: CC02

FATIGUE SCHEDULE BLOCK INPUT TABLE

S : M: NUMBER : S3 :
T : A: OF : :
E : T: FATIGUE : :
P : L: CYCLES : (t1): (t2) :

1:	1:	1000.00	106.00	106.00
2:	1:	1000.00	106.00	106.00
3:	1:	1000.00	106.00	106.00
4:	1:	1000.00	106.00	106.00
5:	1:	1000.00	106.00	106.00
6:	1:	1000.00	106.00	106.00
7:	1:	1000.00	106.00	106.00
8:	1:	1000.00	106.00	106.00
9:	1:	1000.00	106.00	106.00
10:	1:	1000.00	106.00	106.00
11:	1:	1000.00	106.00	106.00
12:	1:	1000.00	106.00	106.00
13:	1:	1000.00	106.00	106.00
14:	1:	1000.00	106.00	106.00
15:	1:	1000.00	106.00	106.00
16:	1:	1000.00	106.00	106.00
17:	1:	1000.00	106.00	106.00
18:	1:	1000.00	106.00	106.00
19:	1:	1000.00	106.00	106.00
20:	1:	1000.00	106.00	106.00
21:	1:	1000.00	106.00	106.00
22:	1:	1000.00	106.00	106.00
23:	1:	1000.00	106.00	106.00
24:	1:	1000.00	106.00	106.00
25:	1:	1000.00	106.00	106.00
26:	1:	1000.00	106.00	106.00
27:	1:	1000.00	106.00	106.00
28:	1:	1000.00	106.00	106.00
29:	1:	1000.00	106.00	106.00
30:	1:	1000.00	106.00	106.00
31:	1:	1000.00	106.00	106.00
32:	1:	1000.00	106.00	106.00
33:	1:	1000.00	106.00	106.00
34:	1:	1000.00	106.00	106.00
35:	1:	1000.00	106.00	106.00
36:	1:	1000.00	106.00	106.00
37:	1:	1000.00	106.00	106.00
38:	1:	1000.00	106.00	106.00
39:	1:	1000.00	106.00	106.00
40:	1:	1000.00	106.00	106.00
41:	1:	1000.00	106.00	106.00
42:	1:	1000.00	106.00	106.00
43:	1:	1000.00	106.00	106.00
44:	1:	1000.00	106.00	106.00

45:	1:	1000.00 :	106.00:	106.00:
46:	1:	1000.00 :	106.00:	106.00:
47:	1:	1000.00 :	106.00:	106.00:
48:	1:	1000.00 :	106.00:	106.00:
49:	1:	1000.00 :	106.00:	106.00:
50:	1:	1000.00 :	106.00:	106.00:
51:	1:	1000.00 :	106.00:	106.00:
52:	1:	1000.00 :	106.00:	106.00:
53:	1:	1000.00 :	106.00:	106.00:
54:	1:	1000.00 :	106.00:	106.00:
55:	1:	1000.00 :	106.00:	106.00:
56:	1:	1000.00 :	106.00:	106.00:
57:	1:	1000.00 :	106.00:	106.00:
58:	1:	1000.00 :	106.00:	106.00:
59:	1:	1000.00 :	106.00:	106.00:
60:	1:	1000.00 :	106.00:	106.00:
61:	1:	1000.00 :	106.00:	106.00:
62:	1:	1000.00 :	106.00:	106.00:
63:	1:	1000.00 :	106.00:	106.00:
64:	1:	1000.00 :	106.00:	106.00:
65:	1:	1000.00 :	106.00:	106.00:
66:	1:	1000.00 :	106.00:	106.00:
67:	1:	1000.00 :	106.00:	106.00:
68:	1:	1000.00 :	106.00:	106.00:
69:	1:	1000.00 :	106.00:	106.00:
70:	1:	1000.00 :	106.00:	106.00:
71:	1:	1000.00 :	106.00:	106.00:
72:	1:	1000.00 :	106.00:	106.00:
73:	1:	1000.00 :	106.00:	106.00:
74:	1:	1000.00 :	106.00:	106.00:
75:	1:	1000.00 :	106.00:	106.00:
76:	1:	1000.00 :	106.00:	106.00:
77:	1:	1000.00 :	106.00:	106.00:
78:	1:	1000.00 :	106.00:	106.00:
79:	1:	1000.00 :	106.00:	106.00:
80:	1:	1000.00 :	106.00:	106.00:
81:	1:	1000.00 :	106.00:	106.00:
82:	1:	1000.00 :	106.00:	106.00:
83:	1:	1000.00 :	106.00:	106.00:
84:	1:	1000.00 :	106.00:	106.00:
85:	1:	1000.00 :	106.00:	106.00:
86:	1:	1000.00 :	106.00:	106.00:
87:	1:	1000.00 :	106.00:	106.00:
88:	1:	1000.00 :	106.00:	106.00:
89:	1:	1000.00 :	106.00:	106.00:
90:	1:	1000.00 :	106.00:	106.00:
91:	1:	1000.00 :	106.00:	106.00:
92:	1:	1000.00 :	106.00:	106.00:
93:	1:	1000.00 :	106.00:	106.00:
94:	1:	1000.00 :	106.00:	106.00:
95:	1:	1000.00 :	106.00:	106.00:
96:	1:	1000.00 :	106.00:	106.00:
97:	1:	1000.00 :	106.00:	106.00:
98:	1:	1000.00 :	106.00:	106.00:
99:	1:	1000.00 :	106.00:	106.00:

Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

[no title given]
MODEL: CC02

FATIGUE SCHEDULE BLOCK STRESS TABLE

S	M	NUMBER	S0	S1	
T	A	OF			
E	T	FATIGUE	(MPa)	(MPa)	
P	L	CYCLES	(t1) (t2)	(t1) (t2)	
1	1	1000.00	51.90	0.00	0.00
2	1	1000.00	51.90	0.00	0.00
3	1	1000.00	51.90	0.00	0.00
4	1	1000.00	51.90	0.00	0.00
5	1	1000.00	51.90	0.00	0.00
6	1	1000.00	51.90	0.00	0.00
7	1	1000.00	51.90	0.00	0.00
8	1	1000.00	51.90	0.00	0.00
9	1	1000.00	51.90	0.00	0.00
10	1	1000.00	51.90	0.00	0.00
11	1	1000.00	51.90	0.00	0.00
12	1	1000.00	51.90	0.00	0.00
13	1	1000.00	51.90	0.00	0.00
14	1	1000.00	51.90	0.00	0.00
15	1	1000.00	51.90	0.00	0.00
16	1	1000.00	51.90	0.00	0.00
17	1	1000.00	51.90	0.00	0.00
18	1	1000.00	51.90	0.00	0.00
19	1	1000.00	51.90	0.00	0.00
20	1	1000.00	51.90	0.00	0.00
21	1	1000.00	51.90	0.00	0.00
22	1	1000.00	51.90	0.00	0.00
23	1	1000.00	51.90	0.00	0.00
24	1	1000.00	51.90	0.00	0.00
25	1	1000.00	51.90	0.00	0.00
26	1	1000.00	51.90	0.00	0.00
27	1	1000.00	51.90	0.00	0.00
28	1	1000.00	51.90	0.00	0.00
29	1	1000.00	51.90	0.00	0.00
30	1	1000.00	51.90	0.00	0.00
31	1	1000.00	51.90	0.00	0.00
32	1	1000.00	51.90	0.00	0.00
33	1	1000.00	51.90	0.00	0.00
34	1	1000.00	51.90	0.00	0.00
35	1	1000.00	51.90	0.00	0.00
36	1	1000.00	51.90	0.00	0.00
37	1	1000.00	51.90	0.00	0.00
38	1	1000.00	51.90	0.00	0.00
39	1	1000.00	51.90	0.00	0.00

40: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
41: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
42: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
43: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
44: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
45: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
46: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
47: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
48: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
49: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
50: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:

Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

[no title given]

MODEL: CC02

FATIGUE SCHEDULE BLOCK STRESS TABLE

S : M:	NUMBER	:	S0	:	S1	:
T : A:	OF	:	:	:	:	:
E : T:	FATIGUE	:	(MPa)	:	(MPa)	:
P : L:	CYCLES	:	(t1) : (t2)	:	(t1) : (t2)	:

51: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
52: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
53: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
54: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
55: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
56: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
57: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
58: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
59: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
60: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
61: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
62: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
63: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
64: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
65: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
66: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
67: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
68: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
69: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
70: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
71: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
72: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
73: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
74: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
75: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
76: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
77: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
78: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:

79: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
80: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
81: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
82: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
83: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
84: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
85: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
86: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
87: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
88: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
89: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
90: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
91: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
92: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
93: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
94: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
95: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
96: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
97: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
98: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
99: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:

[no title given]

MODEL: CC02

FATIGUE SCHEDULE BLOCK STRESS TABLE

S : M:	NUMBER :	S3 :	
T : A:	OF :		
E : T:	FATIGUE :	(MPa)	
P : L:	CYCLES :	(t1) :	(t2) :
1: 1:	1000.00 :	106.00:	106.00:
2: 1:	1000.00 :	106.00:	106.00:
3: 1:	1000.00 :	106.00:	106.00:
4: 1:	1000.00 :	106.00:	106.00:
5: 1:	1000.00 :	106.00:	106.00:
6: 1:	1000.00 :	106.00:	106.00:
7: 1:	1000.00 :	106.00:	106.00:
8: 1:	1000.00 :	106.00:	106.00:
9: 1:	1000.00 :	106.00:	106.00:
10: 1:	1000.00 :	106.00:	106.00:
11: 1:	1000.00 :	106.00:	106.00:
12: 1:	1000.00 :	106.00:	106.00:
13: 1:	1000.00 :	106.00:	106.00:
14: 1:	1000.00 :	106.00:	106.00:
15: 1:	1000.00 :	106.00:	106.00:
16: 1:	1000.00 :	106.00:	106.00:
17: 1:	1000.00 :	106.00:	106.00:
18: 1:	1000.00 :	106.00:	106.00:
19: 1:	1000.00 :	106.00:	106.00:
20: 1:	1000.00 :	106.00:	106.00:
21: 1:	1000.00 :	106.00:	106.00:
22: 1:	1000.00 :	106.00:	106.00:

40: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
41: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
42: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
43: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
44: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
45: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
46: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
47: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
48: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
49: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
50: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:

Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

[no title given]
MODEL: CC02

FATIGUE SCHEDULE BLOCK STRESS TABLE

S : M:	NUMBER :	S0 :	S1 :	
T : A:	OF :			
E : T:	FATIGUE :	(MPa) :	(MPa) :	
P : L:	CYCLES :	(t1) : (t2) :	(t1) : (t2) :	

51: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
52: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
53: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
54: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
55: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
56: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
57: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
58: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
59: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
60: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
61: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
62: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
63: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
64: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
65: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
66: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
67: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
68: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
69: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
70: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
71: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
72: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
73: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
74: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
75: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
76: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
77: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
78: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:

79: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
80: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
81: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
82: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
83: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
84: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
85: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
86: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
87: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
88: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
89: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
90: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
91: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
92: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
93: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
94: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
95: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
96: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
97: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
98: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:
99: 1:	1000.00 :	51.90:	0.00:	0.00:	0.00:

[no title given]

MODEL: CC02

FATIGUE SCHEDULE BLOCK STRESS TABLE

S : M:	NUMBER :	S3 :
T : A:	OF :	:
E : T:	FATIGUE :	(MPa) :
P : L:	CYCLES :	(t1) : (t2) :

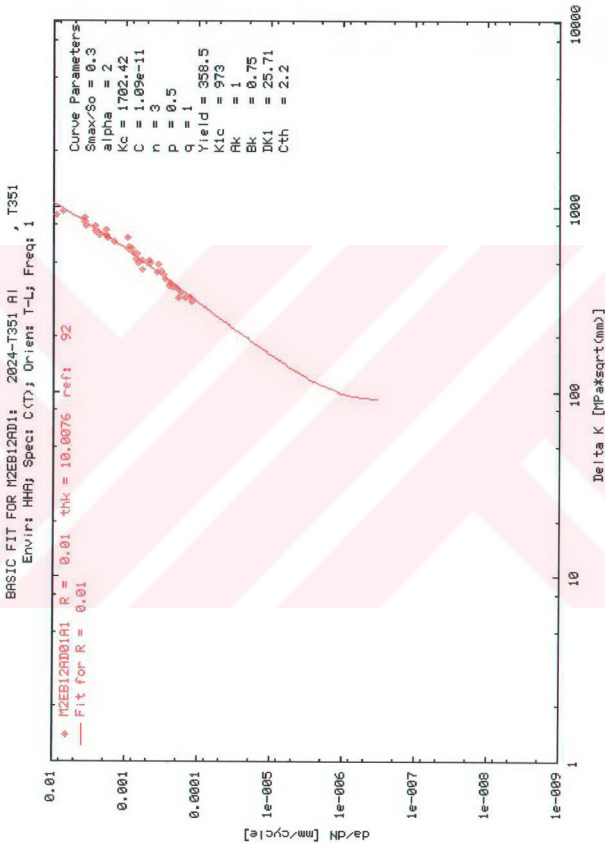
1: 1:	1000.00 :	106.00:	106.00:
2: 1:	1000.00 :	106.00:	106.00:
3: 1:	1000.00 :	106.00:	106.00:
4: 1:	1000.00 :	106.00:	106.00:
5: 1:	1000.00 :	106.00:	106.00:
6: 1:	1000.00 :	106.00:	106.00:
7: 1:	1000.00 :	106.00:	106.00:
8: 1:	1000.00 :	106.00:	106.00:
9: 1:	1000.00 :	106.00:	106.00:
10: 1:	1000.00 :	106.00:	106.00:
11: 1:	1000.00 :	106.00:	106.00:
12: 1:	1000.00 :	106.00:	106.00:
13: 1:	1000.00 :	106.00:	106.00:
14: 1:	1000.00 :	106.00:	106.00:
15: 1:	1000.00 :	106.00:	106.00:
16: 1:	1000.00 :	106.00:	106.00:
17: 1:	1000.00 :	106.00:	106.00:
18: 1:	1000.00 :	106.00:	106.00:
19: 1:	1000.00 :	106.00:	106.00:
20: 1:	1000.00 :	106.00:	106.00:
21: 1:	1000.00 :	106.00:	106.00:
22: 1:	1000.00 :	106.00:	106.00:

23:	1:	1000.00 :	106.00:	106.00:
24:	1:	1000.00 :	106.00:	106.00:
25:	1:	1000.00 :	106.00:	106.00:
26:	1:	1000.00 :	106.00:	106.00:
27:	1:	1000.00 :	106.00:	106.00:
28:	1:	1000.00 :	106.00:	106.00:
29:	1:	1000.00 :	106.00:	106.00:
30:	1:	1000.00 :	106.00:	106.00:
31:	1:	1000.00 :	106.00:	106.00:
32:	1:	1000.00 :	106.00:	106.00:
33:	1:	1000.00 :	106.00:	106.00:
34:	1:	1000.00 :	106.00:	106.00:
35:	1:	1000.00 :	106.00:	106.00:
36:	1:	1000.00 :	106.00:	106.00:
37:	1:	1000.00 :	106.00:	106.00:
38:	1:	1000.00 :	106.00:	106.00:
39:	1:	1000.00 :	106.00:	106.00:
40:	1:	1000.00 :	106.00:	106.00:
41:	1:	1000.00 :	106.00:	106.00:
42:	1:	1000.00 :	106.00:	106.00:
43:	1:	1000.00 :	106.00:	106.00:
44:	1:	1000.00 :	106.00:	106.00:
45:	1:	1000.00 :	106.00:	106.00:
46:	1:	1000.00 :	106.00:	106.00:
47:	1:	1000.00 :	106.00:	106.00:
48:	1:	1000.00 :	106.00:	106.00:
49:	1:	1000.00 :	106.00:	106.00:
50:	1:	1000.00 :	106.00:	106.00:
51:	1:	1000.00 :	106.00:	106.00:
52:	1:	1000.00 :	106.00:	106.00:
53:	1:	1000.00 :	106.00:	106.00:
54:	1:	1000.00 :	106.00:	106.00:
55:	1:	1000.00 :	106.00:	106.00:
56:	1:	1000.00 :	106.00:	106.00:
57:	1:	1000.00 :	106.00:	106.00:
58:	1:	1000.00 :	106.00:	106.00:
59:	1:	1000.00 :	106.00:	106.00:
60:	1:	1000.00 :	106.00:	106.00:
61:	1:	1000.00 :	106.00:	106.00:
62:	1:	1000.00 :	106.00:	106.00:
63:	1:	1000.00 :	106.00:	106.00:
64:	1:	1000.00 :	106.00:	106.00:
65:	1:	1000.00 :	106.00:	106.00:
66:	1:	1000.00 :	106.00:	106.00:
67:	1:	1000.00 :	106.00:	106.00:
68:	1:	1000.00 :	106.00:	106.00:
69:	1:	1000.00 :	106.00:	106.00:
70:	1:	1000.00 :	106.00:	106.00:
71:	1:	1000.00 :	106.00:	106.00:
72:	1:	1000.00 :	106.00:	106.00:
73:	1:	1000.00 :	106.00:	106.00:
74:	1:	1000.00 :	106.00:	106.00:
75:	1:	1000.00 :	106.00:	106.00:
76:	1:	1000.00 :	106.00:	106.00:
77:	1:	1000.00 :	106.00:	106.00:

78: 1:	1000.00 :	106.00:	106.00:
79: 1:	1000.00 :	106.00:	106.00:
80: 1:	1000.00 :	106.00:	106.00:
81: 1:	1000.00 :	106.00:	106.00:
82: 1:	1000.00 :	106.00:	106.00:
83: 1:	1000.00 :	106.00:	106.00:
84: 1:	1000.00 :	106.00:	106.00:
85: 1:	1000.00 :	106.00:	106.00:
86: 1:	1000.00 :	106.00:	106.00:
87: 1:	1000.00 :	106.00:	106.00:
88: 1:	1000.00 :	106.00:	106.00:
89: 1:	1000.00 :	106.00:	106.00:
90: 1:	1000.00 :	106.00:	106.00:
91: 1:	1000.00 :	106.00:	106.00:
92: 1:	1000.00 :	106.00:	106.00:
93: 1:	1000.00 :	106.00:	106.00:
94: 1:	1000.00 :	106.00:	106.00:
95: 1:	1000.00 :	106.00:	106.00:
96: 1:	1000.00 :	106.00:	106.00:
97: 1:	1000.00 :	106.00:	106.00:
98: 1:	1000.00 :	106.00:	106.00:
99: 1:	1000.00 :	106.00:	106.00:

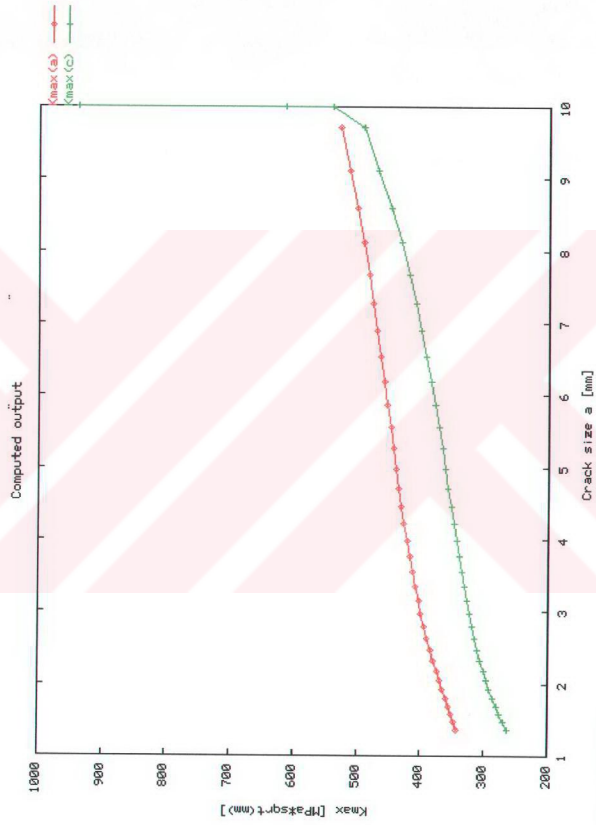
Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

Chart 6.6. da/dN – Delta K



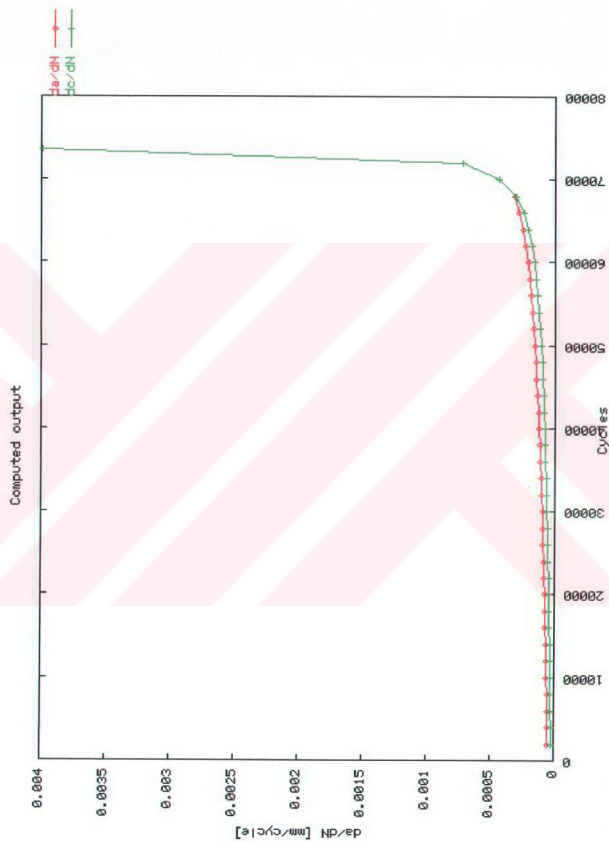
Tue Apr 06 23:19 2004

Chart 6.7. Kmax – a



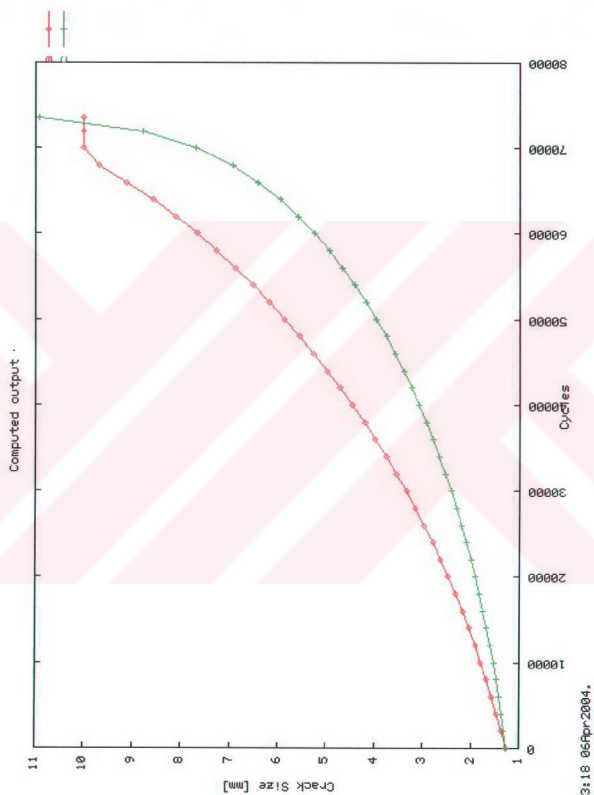
23.18 665pr2004.

Chart 6.8. da/dN – Cycles



23:19 06Apr-2004.

Chart 6.9. Crack Size – Cycles



6.3. Corner Crack At Edge Of Plate

The analysis results of program NASGRO is given below:

CC01

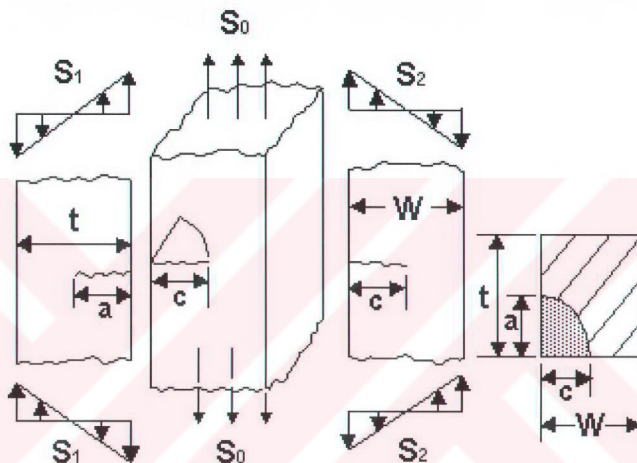


Figure 6.3. Corner Crack – At Edge Of Plate

FATIGUE CRACK GROWTH ANALYSIS

DATE: 01-Apr-04 TIME: 15:06:32
(NASGRO Version 4.02, September 2002.)
SI units [mm, MPa, MPa sqrt(mm)]

PROBLEM TITLE

Sera-edge of plate

Crack Growth Model: Non Interaction
Ratio of growth increment to current crk size= 0.0050
Equation/Table : NASGRO Equation

GEOMETRY

MODEL: CC01-Corner crack in plate or bar (2D).

Plate Thickness, $t = 4.0000$

Plate Width, $W = 500.0000$

FLAW SIZE: (User specified)

a (init.) = 2.540

c (init.) = 1.000

a/c (init.) = 2.540

ANALYSIS RESULTS:

FINAL RESULTS:

Unstable crack growth, max stress intensity exceeds critical value:

$K_{max} = 1842$ $K_{ref} = 0.000$ $K_{cr} = 1840$.

at Cycle No. 771.13

of Load Step No. 1 Description: None

of Block No. 21

of Schedule No. 1

Crack Size $c = 26.8158$

Total Cycles = 20771.127

FATIGUE CRACK GROWTH ANALYSIS

DATE: 01-Apr-04 TIME: 15:06:32

(NASGRO Version 4.02, September 2002.)

SI units [mm, MPa, MPa sqrt(mm)]

PROBLEM TITLE

Sera-edge of plate

Crack Growth Model: Non Interaction

Ratio of growth increment to current crk size= 0.0050

Equation/Table : NASGRO Equation

MATERIAL

MATL 1:

2024-T3 Clad Plt & Sht; T-L; LA & HHA

Material Properties:

:Matl: UTS : YS : K1e : K1c : Ak : Bk : Thk : Kc : Keac :

: No.: : : : : : : : : :

:-----:-----:-----:-----:-----:-----:

: 1 : 448.2 : 330.9 : 1286.0 : 938.2 : 1.00 : 1.00 : 4.000 : 1840.0 : :

FATIGUE SCHEDULE BLOCK INPUT TABLE

SINGLE DISTINCT BLOCK

S : M: NUMBER : S0 : S1 :
T : A: OF : : :
E : T: FATIGUE : : :
P : L: CYCLES : (t1) : (t2) : (t1) : (t2) :

1: 1: 1000.00 : 0.00: 163.00: 0.00: 23.00:

S : M: NUMBER : S2 :
T : A: OF : : :
E : T: FATIGUE : : :
P : L: CYCLES : (t1) : (t2) :

1: 1: 1000.00 : 0.00: 0.00:

Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

FATIGUE SCHEDULE BLOCK STRESS TABLE

S : M: NUMBER : S0 : S1 :
T : A: OF : : :
E : T: FATIGUE : (MPa) : (MPa) :
P : L: CYCLES : (t1) : (t2) : (t1) : (t2) :

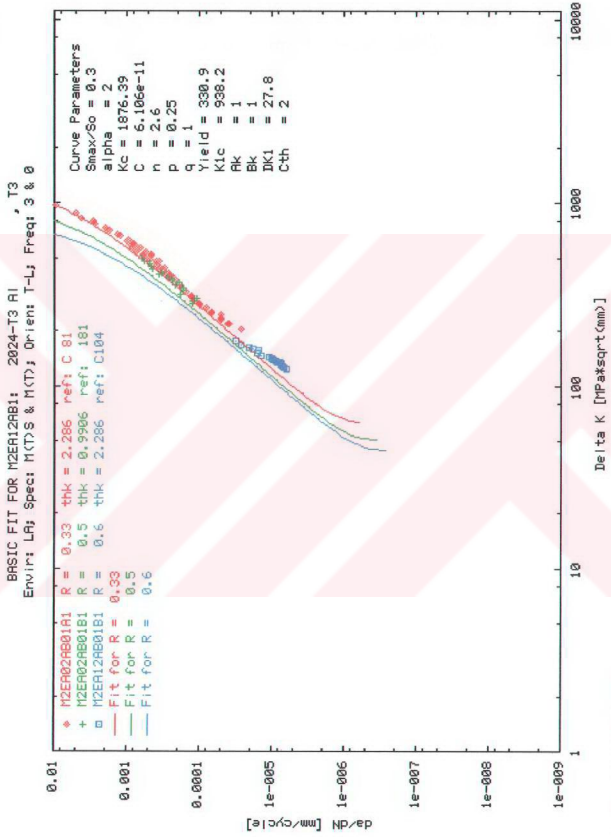
1: 1: 1000.00 : 0.00: 163.00: 0.00: 23.00:

S : M: NUMBER : S2 :
T : A: OF : : :
E : T: FATIGUE : (MPa) :
P : L: CYCLES : (t1) : (t2) :

1: 1: 1000.00 : 0.00: 0.00:

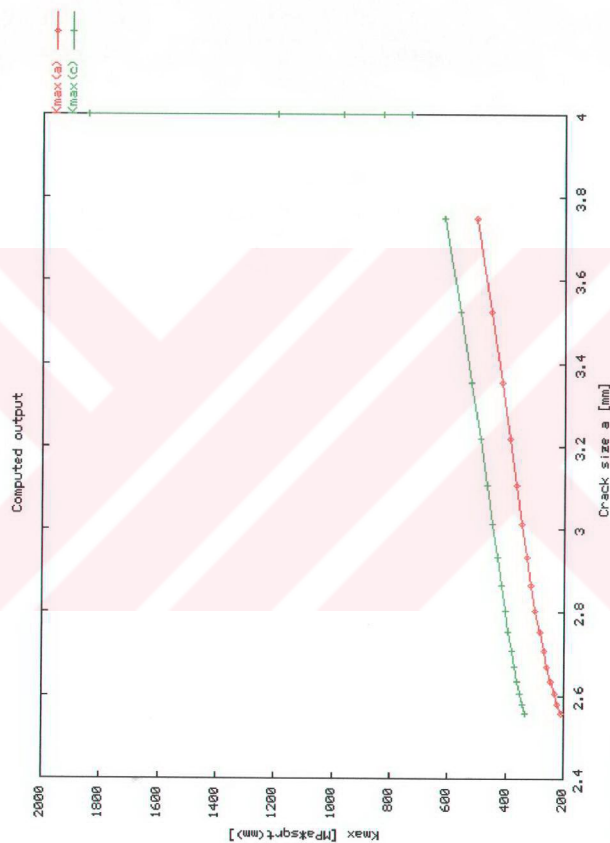
Environmental Crack Growth Check for Sustained Stresses
(Kmax less than Keac): NOT SET

Chart 6.10. da/dN – Delta K



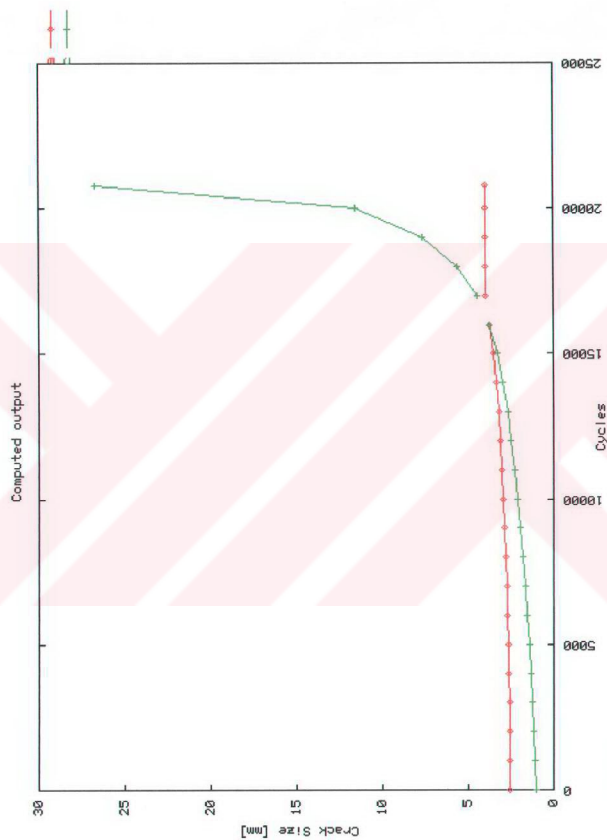
Tue Apr 06 23:29 2004

Chart 6.11. Kmax- a



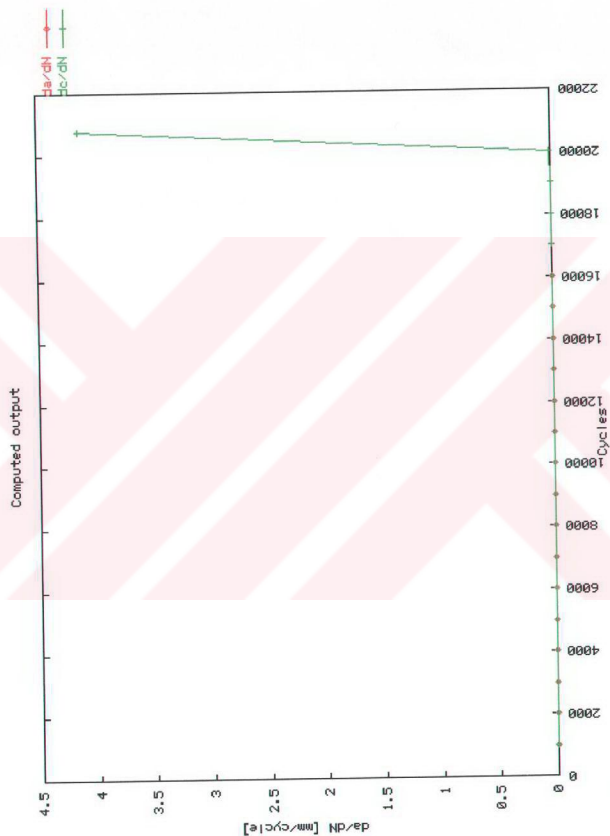
23:34 06Apr2004.

Chart 6.12. Crack Size – Cycles



23:34 06Apr2004.

Chart 6.13. da/dN – Cycles



23:47 06Apr2004.

6.4. Longitudinal Crack In A Curved Stiffened Fuselage Panel

Table 6.1. Nomenclature

a (initial)	Half crack length [mm]	37.5
R	Stress ratio	0
T	Integration period	100
R (ρ)	Fuselage radius [mm]	3340
L	Frame spacing [mm]	635
Delta p	Pressure Differential [MPa]	605
E	Modulus of Elasticity [MPa]	71000
e (t)	Thickness [mm]	2
C	Forman Coefficient at R	0.201E-07
S long./S circum. (X)	Biaxiality Ratio	0.33
$B_{n\text{usd}}$	Beta Non-Uniform Stress Distrubution(normalized initial)	0.8406
β_b	Beta Bulging(normalized initial)	1.111793
B_s	Beta Stiffening(normalized initial)	1.017

Table 6.2. Non-Uniform Stress Distribution

x=0 at frame midbay a	x=0 at frame midbay a/L normalized	NON UNIFORM STRESS DISTRIBUTION		x=0 at frame a	TEST	TEST		ANALYSIS
		x=0 at frame	x=0 at frame			NORMALIZED	ACTUAL	
0	0	0.5		317.5	60	1	1	1
16.51	0.026	0.474		300.99	60	1		0.998
33.655	0.053	0.447		283.845	60	1		0.997
48.006	0.0756	0.4244		269.494	60	1		0.997
50.165	0.079	0.421		267.335	60	1		0.995
67.31	0.106	0.394		250.19	59.5	0.991666667		0.994
83.82	0.132	0.368		233.68	59	0.983333333		0.992
96.012	0.1512	0.3488		221.488	59	0.983333333		0.9913
100.965	0.159	0.341		216.535	59	0.983333333		0.991
117.475	0.185	0.315		200.025	59	0.983333333		0.985
127.635	0.201	0.299		189.865	59	0.983333333		0.981
142.24	0.224	0.276		175.26	59	0.983333333		0.976
144.018	0.2268	0.2732		173.482	59	0.983333333		0.975
159.385	0.251	0.249		158.115	58.36	0.972666667		0.965
175.895	0.277	0.223		141.605	57.672	0.9612		0.949

192.024	0.3024	0.1976	125.476	57	0.95	0.933
193.04	0.304	0.196	124.46	56.915	0.948583333	0.932
207.645	0.327	0.173	109.855	55.698	0.9283	0.902
211.455	0.333	0.167	106.045	55.381	0.923016867	0.894
223.52	0.352	0.148	93.98	54.376	0.906266667	0.856
229.235	0.361	0.139	88.265	53.899	0.898316667	0.838
240.03	0.378	0.122	77.47	53	0.883333333	0.805
247.015	0.389	0.111	70.485	52.553	0.875883333	0.777
255.905	0.403	0.097	61.595	51.984	0.8664	0.743
264.795	0.417	0.083	52.705	51.414	0.8569	0.708
271.78	0.428	0.072	45.72	50.967	0.84945	0.68
281.94	0.444	0.056	35.56	50.316	0.8386	0.638
287.655	0.453	0.047	29.845	49.951	0.832516667	0.615
299.72	0.472	0.028	17.78	49.178	0.819633333	0.579
302.50003	0.476378	0.023622	14.99997	49	0.816666667	0.571
317.5	0.5	0	0	49	0.816666667	0.528

NORMALIZED TEST ACTUAL/ANALYSIS		TEST DATA	
(T/A)		a	
		N	a/L
1	38	0	0.0598
1.002004008	38.5	200	0.0606
1.003009027	39.25	1000	0.0618
1.003009027	40.5	2500	0.0638
1.005025126	43.25	5000	0.0681
0.997652582	49.25	7500	0.0776
0.991263441	53.25	9000	0.0839
0.991963415	56.75	10500	0.0894
0.992263707	60.25	11900	0.0949
0.998307953	64.75	13500	0.102
1.002378525	69.5	15000	0.1094
1.007513661	74.75	16500	0.1177
1.008547009	79.5	18000	0.1252
1.007944732	84.75	19500	0.1335
1.012855638	88.5	20500	0.1394
1.018220793	91.75	21500	0.1445
1.017793276	100	23000	0.1575
1.029157428	105	24000	0.1654
1.032457122	111	25000	0.1748
1.058722741	118.25	26000	0.1862
1.071976929	127.25	27000	0.2004
1.097308489	135.5	27700	0.2134

1.127262977	143	28200	0.2252
1.166083445	151.5	28601	0.2386
1.210310734	160	28901	0.252
1.249191176	164.75	29051	0.2594
1.314420063	170	29201	0.2677
1.353685637	176.25	29351	0.2776
1.415601612	182.5	29501	0.2874
1.430239346	191	29651	0.3008
1.546717172	198	29751	0.3118
	205.5	29851	0.3236
	210	29951	0.3307
	216.5	30051	0.3409
	224	30151	0.3528
	229	30201	0.3606
	234	30301	0.3685
	246.75	30451	0.3886
	267.5	30601	0.4213
	289	30701	0.4551
	319.5	30771	0.5031
	342	30801	0.5386
	360.5	30821	0.5677
	385.5	30841	0.6071
	413	30856	0.6504
	446.5	30872	0.7031
	487	30887	0.7669
	528.5	30903	0.8323

Table 6.3. Beta Bulging (Swift-Chen), Beta Stiffening

Crack Size	Bulging (Swift)	Bulging (Chen)	Stiffening
a	Beta Bulg	Beta Bulg	Beta Stiff
37.5	1.111792726	1	1.017
50	1.148557004	1.00011388	1.013
62.5	1.184893767	1	1.011
75	1.220697414	0.99979728	1.009
87.5	1.255863197	1	1.008
100	1.290287385	1.00030272	1.007
112.5	1.323867429	1	1.007
125	1.356502133	0.99958888	1.006
137.5	1.388091808	1	1.006
150	1.418538441	1.00052536	1.005
162.5	1.447745847	1	1.005
175	1.475619831	0.99935505	1.005
187.5	1.502068335	1	1.004
200	1.527001595	1.00076744	1.004
212.5	1.550332283	1	1.004
225	1.571975654	0.9991059	1.004
237.5	1.591849689	1	1.003
250	1.609875226	1.00102151	1.003
262.5	1.625976097	1	1.003
275	1.640079258	0.99884706	1.003
287.5	1.652114912	1	1.003
300	1.66201663	1.00128322	1.003
312.5	1.669721468	1	1.003
325	1.675170077	0.99858194	1.003
337.5	1.678306809	1	1.002
350	1.679079821	1.00154985	1.002

Table 6.4. Two Different Beta Interpolations

Crack Size	TEST DATA	Beta Interpolation 2	Beta Interpolation 1	Life
a	N			
38	0	37.5	37.5	0
38.5	200	37.57592	37.57592	100
39.25	1000	37.65218	37.64102	200
40.5	2500	37.72875	37.70636	300
43.25	5000	37.80563	37.77194	400
49.25	7500	37.88284	37.83776	500
53.25	9000	37.96037	37.90382	600
56.75	10500	38.03822	37.97013	700
60.25	11900	38.11639	38.03668	800
64.75	13500	38.1949	38.10348	900
69.5	15000	38.27373	38.17054	1000
74.75	16500	38.3529	38.23784	1100
79.5	18000	38.4324	38.30539	1200
84.75	19500	38.51224	38.3732	1300
88.5	20500	38.59242	38.44127	1400
91.75	21500	38.67294	38.50959	1500
100	23000	38.7538	38.57817	1600
105	24000	38.83501	38.64701	1700
111	25000	38.91658	38.71612	1800
118.25	26000	38.99849	38.78548	1900
127.25	27000	39.08076	38.85512	2000
135.5	27700	39.16339	38.92502	2100
143	28200	39.24637	38.99519	2200
151.5	28601	39.32972	39.06564	2300
160	28901	39.41343	39.13635	2400

164.75	29051	39.49752	39.20734	2500
170	29201	39.58197	39.27861	2600
176.25	29351	39.6668	39.35016	2700
182.5	29501	39.752	39.42199	2800
191	29651	39.83758	39.4941	2900
198	29751	39.92354	39.56649	3000
205.5	29851	40.00989	39.63917	3100
210	29951	40.09663	39.71214	3200
216.5	30051	40.18376	39.78539	3300
224	30151	40.27128	39.85894	3400
229	30201	40.35919	39.93279	3500
234	30301	40.44751	40.00693	3600
246.75	30451	40.53623	40.08137	3700
257.5	30601	40.62536	40.15611	3800
269	30701	40.71489	40.23115	3900
319.5	30771	40.80484	40.30649	4000
342	30801	40.89521	40.38214	4100
360.5	30821	40.98599	40.4581	4200
385.5	30841	41.07719	40.53438	4300
413	30856	41.16882	40.61096	4400
446.5	30872	41.26088	40.68786	4500
487	30887	41.35337	40.76508	4600
528.5	30903	41.4463	40.84261	4700

Table 6.5. Crack Propagation in Curved Stiffened Panel

a	BETA NUSD	BETA BULG	BETA STIFF	SIGMA MAX	R	K	da/dN	T	Delta a	Life	a/L
[1]	[2]	[3]	[4]	[5]	[6]	[7]	[8]	[9]	[10]	[11]	
									[8] * [9]		
37.5	0.8406	1.111793	1.017	72.3	0	745.87	0.0008	100	0.0759	0	0.0591
37.576	0.840752	1.1120163	1.01698	72.3	0	746.89	0.0008	100	0.0763	100	0.0592
37.652	0.840833	1.1122406	1.01695	72.3	0	747.85	0.0008	100	0.0766	200	0.0593
37.729	0.840915	1.1124658	1.01693	72.3	0	748.82	0.0008	100	0.0769	300	0.0594
37.806	0.840997	1.1126919	1.0169	72.3	0	749.79	0.0008	100	0.0772	400	0.0595
37.883	0.841079	1.112919	1.01688	72.3	0	750.76	0.0008	100	0.0775	500	0.0597
37.96	0.841162	1.113147	1.01685	72.3	0	751.74	0.0008	100	0.0779	600	0.0598
38.038	0.841245	1.113376	1.01683	72.3	0	752.72	0.0008	100	0.0782	700	0.0599
38.116	0.841329	1.1136059	1.0168	72.3	0	753.71	0.0008	100	0.0785	800	0.06
38.195	0.841412	1.1138368	1.01678	72.3	0	754.7	0.0008	100	0.0788	900	0.0601
38.274	0.841497	1.1140686	1.01675	72.3	0	755.69	0.0008	100	0.0792	1000	0.0603
38.353	0.841581	1.1143015	1.01673	72.3	0	756.69	0.0008	100	0.0795	1100	0.0604
38.432	0.841666	1.1145353	1.0167	72.3	0	757.69	0.0008	100	0.0798	1200	0.0605
38.512	0.841751	1.1147701	1.01668	72.3	0	758.69	0.0008	100	0.0802	1300	0.0606
38.592	0.841837	1.1150059	1.01665	72.3	0	759.7	0.0008	100	0.0805	1400	0.0608
38.673	0.841923	1.1152427	1.01662	72.3	0	760.71	0.0008	100	0.0809	1500	0.0609
38.754	0.842009	1.1154806	1.0166	72.3	0	761.73	0.0008	100	0.0812	1600	0.061
38.835	0.842096	1.1157194	1.01657	72.3	0	762.75	0.0008	100	0.0816	1700	0.0612
38.917	0.842183	1.1159593	1.01655	72.3	0	763.77	0.0008	100	0.0819	1800	0.0613
38.998	0.84227	1.1162002	1.01652	72.3	0	764.8	0.0008	100	0.0823	1900	0.0614
39.081	0.842358	1.1164422	1.01649	72.3	0	765.83	0.0008	100	0.0826	2000	0.0615

39.163	0.842446	1.1166852	1.01647	72.3	0	766.87	0.0008	100	0.083	2100	0.0617
39.246	0.842535	1.1169293	1.01644	72.3	0	767.91	0.0008	100	0.0833	2200	0.0618
39.33	0.842624	1.1171744	1.01641	72.3	0	768.95	0.0008	100	0.0837	2300	0.0619
39.413	0.842713	1.1174206	1.01639	72.3	0	770	0.0008	100	0.0841	2400	0.0621
39.498	0.842803	1.1176679	1.01636	72.3	0	771.05	0.0008	100	0.0845	2500	0.0622
39.582	0.842893	1.1179163	1.01633	72.3	0	772.11	0.0008	100	0.0848	2600	0.0623
39.667	0.842983	1.1181658	1.01631	72.3	0	773.17	0.0009	100	0.0852	2700	0.0625
39.752	0.843074	1.1184164	1.01628	72.3	0	774.24	0.0009	100	0.0856	2800	0.0626
39.838	0.843166	1.1186681	1.01625	72.3	0	775.31	0.0009	100	0.086	2900	0.0627
39.924	0.843258	1.1189209	1.01622	72.3	0	776.38	0.0009	100	0.0863	3000	0.0629
40.01	0.84335	1.1191749	1.0162	72.3	0	777.46	0.0009	100	0.0867	3100	0.063
40.097	0.843442	1.11943	1.01617	72.3	0	778.55	0.0009	100	0.0871	3200	0.0631
40.184	0.843535	1.1196862	1.01614	72.3	0	779.64	0.0009	100	0.0875	3300	0.0633
40.271	0.843629	1.1199437	1.01611	72.3	0	780.73	0.0009	100	0.0879	3400	0.0634
40.359	0.843722	1.1202022	1.01609	72.3	0	781.83	0.0009	100	0.0883	3500	0.0636
40.448	0.843817	1.120462	1.01606	72.3	0	782.93	0.0009	100	0.0887	3600	0.0637
40.536	0.843911	1.1207229	1.01603	72.3	0	784.04	0.0009	100	0.0891	3700	0.0638
40.625	0.844007	1.1209851	1.016	72.3	0	785.15	0.0009	100	0.0895	3800	0.064
40.715	0.844102	1.1212484	1.01597	72.3	0	786.26	0.0009	100	0.0899	3900	0.0641
40.805	0.844198	1.1215129	1.01594	72.3	0	787.38	0.0009	100	0.0904	4000	0.0643
40.895	0.844295	1.1217787	1.01591	72.3	0	788.51	0.0009	100	0.0908	4100	0.0644
40.986	0.844392	1.1220457	1.01588	72.3	0	789.64	0.0009	100	0.0912	4200	0.0645
41.077	0.844489	1.122314	1.01586	72.3	0	790.78	0.0009	100	0.0916	4300	0.0647
41.169	0.844587	1.1225834	1.01583	72.3	0	791.92	0.0009	100	0.0921	4400	0.0648
41.261	0.844685	1.1228542	1.0158	72.3	0	793.06	0.0009	100	0.0925	4500	0.065
41.353	0.844784	1.1231262	1.01577	72.3	0	794.21	0.0009	100	0.0929	4600	0.0651
41.446	0.844883	1.1233995	1.01574	72.3	0	795.37	0.0009	100	0.0934	4700	0.0653

41.54	0.844982	1.1236741	1.01571	72.3	0	796.53	0.0009	100	0.0938	4800	0.0654
41.633	0.845083	1.12395	1.01568	72.3	0	797.69	0.0009	100	0.0943	4900	0.0656
41.728	0.845183	1.1242273	1.01565	72.3	0	798.87	0.0009	100	0.0947	5000	0.0657
41.822	0.845284	1.1245058	1.01562	72.3	0	800.04	0.001	100	0.0952	5100	0.0659
41.918	0.845386	1.1247857	1.01559	72.3	0	801.22	0.001	100	0.0956	5200	0.066
42.013	0.845488	1.1250669	1.01556	72.3	0	802.41	0.001	100	0.0961	5300	0.0662
42.109	0.84559	1.1253495	1.01553	72.3	0	803.6	0.001	100	0.0965	5400	0.0663
42.206	0.845694	1.1256334	1.01549	72.3	0	804.8	0.001	100	0.097	5500	0.0665
42.303	0.845797	1.1259187	1.01546	72.3	0	806	0.001	100	0.0975	5600	0.0666
42.4	0.845901	1.1262054	1.01543	72.3	0	807.21	0.001	100	0.098	5700	0.0668
42.498	0.846006	1.1264936	1.0154	72.3	0	808.42	0.001	100	0.0984	5800	0.0669
42.597	0.846111	1.1267831	1.01537	72.3	0	809.64	0.001	100	0.0989	5900	0.0671
42.696	0.846216	1.127074	1.01534	72.3	0	810.87	0.001	100	0.0994	6000	0.0672
42.795	0.846322	1.1273664	1.01531	72.3	0	812.1	0.001	100	0.0999	6100	0.0674
42.895	0.846429	1.1276603	1.01527	72.3	0	813.33	0.001	100	0.1004	6200	0.0676
42.995	0.846536	1.1279556	1.01524	72.3	0	814.57	0.001	100	0.1009	6300	0.0677
43.096	0.846644	1.1282524	1.01521	72.3	0	815.82	0.001	100	0.1014	6400	0.0679
43.198	0.846752	1.1285506	1.01518	72.3	0	817.08	0.001	100	0.1019	6500	0.068
43.3	0.846861	1.1288504	1.01514	72.3	0	818.34	0.001	100	0.1024	6600	0.0682
43.402	0.84697	1.1291517	1.01511	72.3	0	819.6	0.001	100	0.103	6700	0.0683
43.505	0.84708	1.1294545	1.01508	72.3	0	820.87	0.001	100	0.1035	6800	0.0685
43.608	0.847191	1.1297588	1.01505	72.3	0	822.15	0.001	100	0.104	6900	0.0687
43.712	0.847302	1.1300647	1.01501	72.3	0	823.43	0.001	100	0.1045	7000	0.0688
43.817	0.847413	1.1303722	1.01498	72.3	0	824.72	0.0011	100	0.1051	7100	0.069
43.922	0.847525	1.1306812	1.01494	72.3	0	826.02	0.0011	100	0.1056	7200	0.0692
44.028	0.847638	1.1309918	1.01491	72.3	0	827.32	0.0011	100	0.1062	7300	0.0693
44.134	0.847751	1.1313041	1.01488	72.3	0	828.63	0.0011	100	0.1067	7400	0.0695

44.241	0.847865	1.131618	1.01484	72.3	0	829.95	0.0011	100	0.1073	7500	0.0697
44.348	0.84798	1.1319335	1.01481	72.3	0	831.27	0.0011	100	0.1078	7600	0.0698
44.456	0.848095	1.1322506	1.01477	72.3	0	832.59	0.0011	100	0.1084	7700	0.07
44.564	0.848211	1.1325694	1.01474	72.3	0	833.93	0.0011	100	0.109	7800	0.0702
44.673	0.848327	1.1328899	1.0147	72.3	0	835.27	0.0011	100	0.1096	7900	0.0704
44.783	0.848444	1.1332121	1.01467	72.3	0	836.62	0.0011	100	0.1101	8000	0.0705
44.893	0.848562	1.1335361	1.01463	72.3	0	837.97	0.0011	100	0.1107	8100	0.0707
45.004	0.84868	1.1338617	1.0146	72.3	0	839.33	0.0011	100	0.1113	8200	0.0709
45.115	0.848798	1.1341891	1.01456	72.3	0	840.7	0.0011	100	0.1119	8300	0.071
45.227	0.848918	1.1345182	1.01453	72.3	0	842.08	0.0011	100	0.1125	8400	0.0712
45.339	0.849038	1.1348492	1.01449	72.3	0	843.46	0.0011	100	0.1131	8500	0.0714
45.452	0.849159	1.1351819	1.01446	72.3	0	844.85	0.0011	100	0.1137	8600	0.0716
45.566	0.84928	1.1355164	1.01442	72.3	0	846.24	0.0011	100	0.1144	8700	0.0718
45.68	0.849402	1.1358528	1.01438	72.3	0	847.65	0.0011	100	0.115	8800	0.0719
45.795	0.849525	1.136191	1.01435	72.3	0	849.06	0.0012	100	0.1156	8900	0.0721
45.911	0.849648	1.1365311	1.01431	72.3	0	850.48	0.0012	100	0.1163	9000	0.0723
46.027	0.849773	1.136873	1.01427	72.3	0	851.9	0.0012	100	0.1169	9100	0.0725
46.144	0.849897	1.1372169	1.01423	72.3	0	853.34	0.0012	100	0.1176	9200	0.0727
46.262	0.850023	1.1375626	1.0142	72.3	0	854.78	0.0012	100	0.1182	9300	0.0729
46.38	0.850149	1.1379103	1.01416	72.3	0	856.22	0.0012	100	0.1189	9400	0.073
46.499	0.850276	1.13826	1.01412	72.3	0	857.68	0.0012	100	0.1196	9500	0.0732
46.618	0.850403	1.1386116	1.01408	72.3	0	859.14	0.0012	100	0.1202	9600	0.0734
46.739	0.850532	1.1389652	1.01404	72.3	0	860.62	0.0012	100	0.1209	9700	0.0736
46.86	0.850661	1.1393208	1.014	72.3	0	862.1	0.0012	100	0.1216	9800	0.0738
46.981	0.850791	1.1396785	1.01397	72.3	0	863.58	0.0012	100	0.1223	9900	0.074
47.104	0.850921	1.1400382	1.01393	72.3	0	865.08	0.0012	100	0.123	10000	0.0742
47.227	0.851053	1.1404	1.01389	72.3	0	866.58	0.0012	100	0.1237	10100	0.0744

47.35	0.851185	1.1407639	1.01385	72.3	0	868.09	0.0012	100	0.1244	10200	0.0746
47.475	0.851317	1.1411299	1.01381	72.3	0	869.61	0.0013	100	0.1252	10300	0.0748
47.6	0.851451	1.141498	1.01377	72.3	0	871.14	0.0013	100	0.1259	10400	0.075
47.726	0.851585	1.1418683	1.01373	72.3	0	872.68	0.0013	100	0.1266	10500	0.0752
47.852	0.851721	1.1422408	1.01369	72.3	0	874.23	0.0013	100	0.1274	10600	0.0754
47.98	0.851857	1.1426155	1.01365	72.3	0	875.78	0.0013	100	0.1281	10700	0.0756
48.108	0.851993	1.1429924	1.01361	72.3	0	877.34	0.0013	100	0.1289	10800	0.0758
48.237	0.852131	1.1433715	1.01356	72.3	0	878.92	0.0013	100	0.1297	10900	0.076
48.367	0.852269	1.143753	1.01352	72.3	0	880.5	0.0013	100	0.1305	11000	0.0762
48.497	0.852409	1.1441367	1.01348	72.3	0	882.09	0.0013	100	0.1313	11100	0.0764
48.628	0.852549	1.1445227	1.01344	72.3	0	883.69	0.0013	100	0.1321	11200	0.0766
48.76	0.85269	1.1449111	1.0134	72.3	0	885.3	0.0013	100	0.1329	11300	0.0768
48.893	0.852831	1.1453019	1.01335	72.3	0	886.92	0.0013	100	0.1337	11400	0.077
49.027	0.852974	1.145695	1.01331	72.3	0	888.54	0.0013	100	0.1345	11500	0.0772
49.161	0.853118	1.1460906	1.01327	72.3	0	890.18	0.0014	100	0.1353	11600	0.0774
49.297	0.853262	1.1464886	1.01323	72.3	0	891.83	0.0014	100	0.1362	11700	0.0776
49.433	0.853407	1.1468891	1.01318	72.3	0	893.48	0.0014	100	0.137	11800	0.0778
49.57	0.853554	1.1472921	1.01314	72.3	0	895.15	0.0014	100	0.1379	11900	0.0781
49.708	0.853701	1.1476976	1.01309	72.3	0	896.83	0.0014	100	0.1388	12000	0.0783
49.847	0.853849	1.1481057	1.01305	72.3	0	898.51	0.0014	100	0.1396	12100	0.0785
49.986	0.853998	1.1485164	1.013	72.3	0	900.21	0.0014	100	0.1405	12200	0.0787
50.127	0.854148	1.1489253	1.01298	72.3	0	901.93	0.0014	100	0.1414	12300	0.0789
50.268	0.854299	1.1493364	1.01296	72.3	0	903.67	0.0014	100	0.1423	12400	0.0792
50.41	0.854451	1.1497502	1.01293	72.3	0	905.41	0.0014	100	0.1433	12500	0.0794
50.554	0.854604	1.1501667	1.01291	72.3	0	907.17	0.0014	100	0.1442	12600	0.0796
50.698	0.854758	1.1505859	1.01289	72.3	0	908.93	0.0015	100	0.1452	12700	0.0798
50.843	0.854913	1.1510079	1.01287	72.3	0	910.71	0.0015	100	0.1461	12800	0.0801

50.989	0.855069	1.1514326	1.01284	72.3	0	912.5	0.0015	100	0.1471	12900	0.0803
51.136	0.855226	1.1518602	1.01282	72.3	0	914.3	0.0015	100	0.1481	13000	0.0805
51.284	0.855384	1.1522907	1.01279	72.3	0	916.12	0.0015	100	0.1491	13100	0.0808
51.433	0.855543	1.152724	1.01277	72.3	0	917.94	0.0015	100	0.1501	13200	0.081
51.584	0.855703	1.1531603	1.01275	72.3	0	919.78	0.0015	100	0.1511	13300	0.0812
51.735	0.855864	1.1535995	1.01272	72.3	0	921.63	0.0015	100	0.1521	13400	0.0815
51.887	0.856027	1.1540418	1.0127	72.3	0	923.49	0.0015	100	0.1532	13500	0.0817
52.04	0.85619	1.154487	1.01267	72.3	0	925.36	0.0015	100	0.1542	13600	0.082
52.194	0.856355	1.1549354	1.01265	72.3	0	927.25	0.0016	100	0.1553	13700	0.0822
52.349	0.856521	1.1553868	1.01262	72.3	0	929.14	0.0016	100	0.1564	13800	0.0824
52.506	0.856687	1.1558415	1.0126	72.3	0	931.06	0.0016	100	0.1575	13900	0.0827
52.663	0.856856	1.1562992	1.01257	72.3	0	932.98	0.0016	100	0.1586	14000	0.0829
52.822	0.857025	1.1567603	1.01255	72.3	0	934.92	0.0016	100	0.1597	14100	0.0832
52.982	0.857196	1.1572246	1.01252	72.3	0	936.87	0.0016	100	0.1609	14200	0.0834
53.143	0.857368	1.1576922	1.0125	72.3	0	938.83	0.0016	100	0.162	14300	0.0837
53.305	0.857541	1.1581631	1.01247	72.3	0	940.81	0.0016	100	0.1632	14400	0.0839
53.468	0.857715	1.1586375	1.01245	72.3	0	942.8	0.0016	100	0.1644	14500	0.0842
53.632	0.857891	1.1591153	1.01242	72.3	0	944.81	0.0017	100	0.1656	14600	0.0845
53.798	0.858068	1.1595966	1.01239	72.3	0	946.83	0.0017	100	0.1668	14700	0.0847
53.964	0.858246	1.1600814	1.01237	72.3	0	948.87	0.0017	100	0.168	14800	0.085
54.132	0.858425	1.1605698	1.01234	72.3	0	950.91	0.0017	100	0.1693	14900	0.0852
54.302	0.858606	1.1610618	1.01231	72.3	0	952.98	0.0017	100	0.1705	15000	0.0855
54.472	0.858788	1.1615575	1.01228	72.3	0	955.06	0.0017	100	0.1718	15100	0.0858
54.644	0.858972	1.1620569	1.01226	72.3	0	957.15	0.0017	100	0.1731	15200	0.0861
54.817	0.859157	1.1625601	1.01223	72.3	0	959.26	0.0017	100	0.1744	15300	0.0863
54.991	0.859343	1.1630671	1.0122	72.3	0	961.39	0.0018	100	0.1757	15400	0.0866
55.167	0.859531	1.163578	1.01217	72.3	0	963.53	0.0018	100	0.1771	15500	0.0869

55.344	0.85972	1.1640928	1.01214	72.3	0	965.69	0.0018	100	0.1785	15600	0.0872
55.523	0.859911	1.1646116	1.01212	72.3	0	967.86	0.0018	100	0.1799	15700	0.0874
55.703	0.860103	1.1651344	1.01209	72.3	0	970.05	0.0018	100	0.1813	15800	0.0877
55.884	0.860297	1.1656614	1.01206	72.3	0	972.26	0.0018	100	0.1827	15900	0.088
56.067	0.860492	1.1661925	1.01203	72.3	0	974.49	0.0018	100	0.1841	16000	0.0883
56.251	0.860689	1.1667278	1.012	72.3	0	976.73	0.0019	100	0.1856	16100	0.0886
56.436	0.860887	1.1672674	1.01197	72.3	0	978.99	0.0019	100	0.1871	16200	0.0889
56.624	0.861087	1.1678113	1.01194	72.3	0	981.27	0.0019	100	0.1886	16300	0.0892
56.812	0.861289	1.1683596	1.01191	72.3	0	983.56	0.0019	100	0.1902	16400	0.0895
57.002	0.861492	1.1689124	1.01188	72.3	0	985.87	0.0019	100	0.1917	16500	0.0898
57.194	0.861697	1.1694697	1.01185	72.3	0	988.21	0.0019	100	0.1933	16600	0.0901
57.387	0.861904	1.1700316	1.01182	72.3	0	990.56	0.0019	100	0.1949	16700	0.0904
57.582	0.862112	1.1705982	1.01179	72.3	0	992.93	0.002	100	0.1965	16800	0.0907
57.779	0.862322	1.1711696	1.01176	72.3	0	995.32	0.002	100	0.1982	16900	0.091
57.977	0.862534	1.1717457	1.01172	72.3	0	997.73	0.002	100	0.1999	17000	0.0913
58.177	0.862747	1.1723267	1.01169	72.3	0	1000.2	0.002	100	0.2016	17100	0.0916
58.378	0.862963	1.1729127	1.01166	72.3	0	1002.6	0.002	100	0.2033	17200	0.0919
58.582	0.86318	1.1735037	1.01163	72.3	0	1005.1	0.0021	100	0.2051	17300	0.0923
58.787	0.863399	1.1740999	1.01159	72.3	0	1007.6	0.0021	100	0.2069	17400	0.0926
58.994	0.86362	1.1747012	1.01156	72.3	0	1010.1	0.0021	100	0.2087	17500	0.0929
59.202	0.863843	1.1753078	1.01153	72.3	0	1012.6	0.0021	100	0.2105	17600	0.0932
59.413	0.864068	1.1759198	1.01149	72.3	0	1015.2	0.0021	100	0.2124	17700	0.0936
59.625	0.864295	1.1765372	1.01146	72.3	0	1017.8	0.0021	100	0.2143	17800	0.0939
59.84	0.864524	1.1771602	1.01143	72.3	0	1020.4	0.0022	100	0.2162	17900	0.0942
60.056	0.864755	1.1777888	1.01139	72.3	0	1023	0.0022	100	0.2182	18000	0.0946
60.274	0.864988	1.1784231	1.01136	72.3	0	1025.6	0.0022	100	0.2202	18100	0.0949
60.494	0.865224	1.1790633	1.01132	72.3	0	1028.3	0.0022	100	0.2223	18200	0.0953

60.716	0.865461	1.1797094	1.01129	72.3	0	1031	0.0022	100	0.2243	18300	0.0956
60.941	0.865701	1.1803615	1.01125	72.3	0	1033.7	0.0023	100	0.2264	18400	0.096
61.167	0.865943	1.1810197	1.01121	72.3	0	1036.5	0.0023	100	0.2286	18500	0.0963
61.396	0.866187	1.1816842	1.01118	72.3	0	1039.3	0.0023	100	0.2308	18600	0.0967
61.627	0.866434	1.1823549	1.01114	72.3	0	1042.1	0.0023	100	0.233	18700	0.097
61.86	0.866682	1.1830322	1.0111	72.3	0	1044.9	0.0024	100	0.2352	18800	0.0974
62.095	0.866933	1.183716	1.01106	72.3	0	1047.7	0.0024	100	0.2375	18900	0.0978
62.332	0.867186	1.1844065	1.01103	72.3	0	1050.6	0.0024	100	0.2399	19000	0.0982
62.572	0.867442	1.1851007	1.01099	72.3	0	1053.5	0.0024	100	0.2423	19100	0.0985
62.814	0.867701	1.1857946	1.01095	72.3	0	1056.5	0.0024	100	0.2447	19200	0.0989
63.059	0.867962	1.1864953	1.01091	72.3	0	1059.4	0.0025	100	0.2471	19300	0.0993
63.306	0.868225	1.1872032	1.01087	72.3	0	1062.4	0.0025	100	0.2496	19400	0.0997
63.556	0.868492	1.1879182	1.01083	72.3	0	1065.4	0.0025	100	0.2522	19500	0.1001
63.808	0.868761	1.1886406	1.01079	72.3	0	1068.5	0.0025	100	0.2548	19600	0.1005
64.063	0.869032	1.1893704	1.01075	72.3	0	1071.6	0.0026	100	0.2575	19700	0.1009
64.32	0.869307	1.1901079	1.01071	72.3	0	1074.7	0.0026	100	0.2602	19800	0.1013
64.581	0.869585	1.1908531	1.01067	72.3	0	1077.8	0.0026	100	0.2629	19900	0.1017
64.843	0.869865	1.1916062	1.01063	72.3	0	1081	0.0027	100	0.2657	20000	0.1021
65.109	0.870149	1.1923673	1.01058	72.3	0	1084.2	0.0027	100	0.2686	20100	0.1025
65.378	0.870435	1.1931367	1.01054	72.3	0	1087.4	0.0027	100	0.2716	20200	0.103
65.649	0.870725	1.1939145	1.0105	72.3	0	1090.7	0.0027	100	0.2745	20300	0.1034
65.924	0.871018	1.1947009	1.01045	72.3	0	1094	0.0028	100	0.2776	20400	0.1038
66.201	0.871314	1.195496	1.01041	72.3	0	1097.4	0.0028	100	0.2807	20500	0.1043
66.482	0.871613	1.1963	1.01036	72.3	0	1100.8	0.0028	100	0.2839	20600	0.1047
66.766	0.871916	1.1971131	1.01032	72.3	0	1104.2	0.0029	100	0.2871	20700	0.1051
67.053	0.872222	1.1979355	1.01027	72.3	0	1107.7	0.0029	100	0.2905	20800	0.1056
67.344	0.872532	1.1987674	1.01023	72.3	0	1111.2	0.0029	100	0.2938	20900	0.1061

67.638	0.872846	1.1996091	1.01018	72.3	0	1114.8	0.003	100	0.2973	21000	0.1065
67.935	0.873163	1.2004607	1.01013	72.3	0	1118.4	0.003	100	0.3009	21100	0.107
68.236	0.873484	1.2013224	1.01008	72.3	0	1122	0.003	100	0.3045	21200	0.1075
68.54	0.873808	1.2021945	1.01003	72.3	0	1125.7	0.0031	100	0.3082	21300	0.1079
68.848	0.874137	1.2030772	1.00998	72.3	0	1129.4	0.0031	100	0.312	21400	0.1084
69.16	0.87447	1.2039707	1.00993	72.3	0	1133.2	0.0032	100	0.3159	21500	0.1089
69.476	0.874807	1.2048754	1.00988	72.3	0	1137	0.0032	100	0.3198	21600	0.1094
69.796	0.875148	1.2057915	1.00983	72.3	0	1140.9	0.0032	100	0.3239	21700	0.1099
70.12	0.875494	1.2067191	1.00978	72.3	0	1144.8	0.0033	100	0.328	21800	0.1104
70.448	0.875843	1.2076588	1.00973	72.3	0	1148.7	0.0033	100	0.3323	21900	0.1109
70.78	0.876198	1.2086106	1.00968	72.3	0	1152.8	0.0034	100	0.3367	22000	0.1115
71.117	0.876557	1.2095749	1.00962	72.3	0	1156.8	0.0034	100	0.3412	22100	0.112
71.458	0.876921	1.2105521	1.00957	72.3	0	1161	0.0035	100	0.3458	22200	0.1125
71.804	0.87729	1.2115424	1.00951	72.3	0	1165.1	0.0035	100	0.3505	22300	0.1131
72.154	0.877663	1.2125463	1.00946	72.3	0	1169.4	0.0036	100	0.3553	22400	0.1136
72.51	0.878042	1.213564	1.0094	72.3	0	1173.7	0.0036	100	0.3603	22500	0.1142
72.87	0.878427	1.2145959	1.00934	72.3	0	1178	0.0037	100	0.3654	22600	0.1148
73.235	0.878816	1.2156424	1.00928	72.3	0	1182.5	0.0037	100	0.3706	22700	0.1153
73.606	0.879212	1.2167039	1.00922	72.3	0	1187	0.0038	100	0.376	22800	0.1159
73.982	0.879613	1.2177808	1.00916	72.3	0	1191.5	0.0038	100	0.3815	22900	0.1165
74.363	0.88002	1.2188736	1.0091	72.3	0	1196.1	0.0039	100	0.3872	23000	0.1171
74.751	0.880433	1.2199827	1.00904	72.3	0	1200.8	0.0039	100	0.3931	23100	0.1177
75.144	0.880852	1.2211012	1.00899	72.3	0	1205.6	0.004	100	0.3991	23200	0.1183
75.543	0.881277	1.2222239	1.00896	72.3	0	1210.4	0.0041	100	0.4053	23300	0.119
75.948	0.88171	1.2233642	1.00892	72.3	0	1215.4	0.0041	100	0.4117	23400	0.1196
76.36	0.882149	1.2245226	1.00889	72.3	0	1220.4	0.0042	100	0.4184	23500	0.1203
76.778	0.882595	1.2256995	1.00886	72.3	0	1225.5	0.0043	100	0.4252	23600	0.1209

77.203	0.883049	1.2268957	1.00882	72.3	0	1230.7	0.0043	100	0.4322	23700	0.1216
77.636	0.883563	1.2281116	1.00879	72.3	0	1236	0.0044	100	0.4396	23800	0.1223
78.075	0.884173	1.2293483	1.00875	72.3	0	1241.6	0.0045	100	0.4474	23900	0.123
78.523	0.884794	1.2306069	1.00872	72.3	0	1247.2	0.0046	100	0.4555	24000	0.1237
78.978	0.885426	1.2318882	1.00868	72.3	0	1253	0.0046	100	0.4638	24100	0.1244
79.442	0.88607	1.2331931	1.00864	72.3	0	1258.9	0.0047	100	0.4725	24200	0.1251
79.914	0.886726	1.2345222	1.00861	72.3	0	1264.8	0.0048	100	0.4814	24300	0.1258
80.396	0.887394	1.2358767	1.00857	72.3	0	1270.9	0.0049	100	0.4908	24400	0.1266
80.886	0.888075	1.2372574	1.00853	72.3	0	1277.2	0.005	100	0.5005	24500	0.1274
81.387	0.88877	1.2386653	1.00849	72.3	0	1283.5	0.0051	100	0.5105	24600	0.1282
81.897	0.889479	1.2401015	1.00845	72.3	0	1290	0.0052	100	0.521	24700	0.129
82.418	0.890202	1.2415672	1.00841	72.3	0	1296.6	0.0053	100	0.5319	24800	0.1298
82.95	0.89094	1.2430636	1.00836	72.3	0	1303.4	0.0054	100	0.5433	24900	0.1306
83.494	0.891694	1.2445919	1.00832	72.3	0	1310.3	0.0056	100	0.5551	25000	0.1315
84.049	0.892465	1.2461536	1.00828	72.3	0	1317.4	0.0057	100	0.5675	25100	0.1324
84.616	0.893252	1.2477502	1.00823	72.3	0	1324.7	0.0058	100	0.5804	25200	0.1333
85.197	0.894058	1.2493831	1.00818	72.3	0	1332.1	0.0059	100	0.594	25300	0.1342
85.791	0.894883	1.2510542	1.00814	72.3	0	1339.7	0.0061	100	0.6082	25400	0.1351
86.399	0.895727	1.2527651	1.00809	72.3	0	1347.4	0.0062	100	0.6231	25500	0.1361
87.022	0.896592	1.2545179	1.00804	72.3	0	1355.4	0.0064	100	0.6387	25600	0.137
87.661	0.897478	1.2563052	1.00799	72.3	0	1363.6	0.0066	100	0.6551	25700	0.138
88.316	0.898387	1.2581093	1.00793	72.3	0	1372	0.0067	100	0.6723	25800	0.1391
88.988	0.899323	1.2599607	1.00788	72.3	0	1380.6	0.0069	100	0.6905	25900	0.1401
89.678	0.900283	1.2618622	1.00783	72.3	0	1389.4	0.0071	100	0.7096	26000	0.1412
90.388	0.90127	1.2638165	1.00777	72.3	0	1398.5	0.0073	100	0.7299	26100	0.1423
91.118	0.902286	1.2658267	1.00771	72.3	0	1407.9	0.0075	100	0.7514	26200	0.1435
91.869	0.903331	1.267896	1.00765	72.3	0	1417.6	0.0077	100	0.7742	26300	0.1447

92.644	0.904408	1.2700282	1.00759	72.3	0	1427.5	0.008	100	0.7985	26400	0.1459
93.442	0.905519	1.2722273	1.00752	72.3	0	1437.8	0.0082	100	0.8244	26500	0.1472
94.267	0.906665	1.2744976	1.00746	72.3	0	1448.5	0.0085	100	0.8521	26600	0.1485
95.119	0.907848	1.2768442	1.00739	72.3	0	1459.5	0.0088	100	0.8817	26700	0.1498
96	0.909072	1.2792723	1.00732	72.3	0	1470.9	0.0091	100	0.9135	26800	0.1512
96.914	0.91034	1.2817881	1.00725	72.3	0	1482.7	0.0095	100	0.9478	26900	0.1526
97.862	0.911656	1.2843983	1.00717	72.3	0	1495	0.0098	100	0.9849	27000	0.1541
98.847	0.913023	1.2871107	1.00709	72.3	0	1507.9	0.0103	100	1.0252	27100	0.1557
99.872	0.914447	1.2899339	1.00701	72.3	0	1521.2	0.0107	100	1.069	27200	0.1573
100.94	0.915931	1.2928143	1.007	72.3	0	1535.2	0.0112	100	1.1171	27300	0.159
102.06	0.917482	1.2958151	1.007	72.3	0	1549.9	0.0117	100	1.1699	27400	0.1607
103.23	0.919106	1.298958	1.007	72.3	0	1565.3	0.0123	100	1.2284	27500	0.1626
104.46	0.920811	1.302258	1.007	72.3	0	1581.5	0.0129	100	1.2934	27600	0.1645
105.75	0.922607	1.3057327	1.007	72.3	0	1598.6	0.0137	100	1.3663	27700	0.1665
107.12	0.924504	1.309403	1.007	72.3	0	1616.8	0.0145	100	1.4485	27800	0.1687
108.56	0.926515	1.3132941	1.007	72.3	0	1636.1	0.0154	100	1.5421	27900	0.171
110.11	0.928655	1.3174368	1.007	72.3	0	1656.7	0.0165	100	1.6499	28000	0.1734
111.76	0.930946	1.3218691	1.007	72.3	0	1678.8	0.0178	100	1.7755	28100	0.176
113.53	0.933411	1.3265608	1.00692	72.3	0	1702.4	0.0192	100	1.9226	28200	0.1788
115.45	0.93608	1.3315804	1.00676	72.3	0	1727.9	0.021	100	2.0981	28300	0.1818
117.55	0.938993	1.3370581	1.0066	72.3	0	1755.9	0.0231	100	2.3135	28400	0.1851
119.87	0.942205	1.3430982	1.00641	72.3	0	1786.9	0.0259	100	2.5854	28500	0.1888
122.45	0.945794	1.3498482	1.0062	72.3	0	1821.6	0.0294	100	2.9417	28600	0.1928
125.39	0.94927	1.3574956	1.006	72.3	0	1860.3	0.0342	100	3.4171	28700	0.1975
128.81	0.951784	1.3661312	1.006	72.3	0	1902.5	0.0406	100	4.0637	28800	0.2029
132.87	0.954775	1.376401	1.006	72.3	0	1952.9	0.0509	100	5.0865	28900	0.2093
137.96	0.958518	1.3892556	1.006	72.3	0	2016.4	0.0701	100	7.0146	29000	0.2173

144.98	0.963541	1.4062987	1.0054	72.3	0	2102.1	0.1222	100	12.219	29100	0.2283
157.19	0.972027	1.4360593	1.00442	72.3	0	2252.7	6.8198	100	681.98	29200	0.2475



Chart 6.14. Beta Bulging (Swift) – a

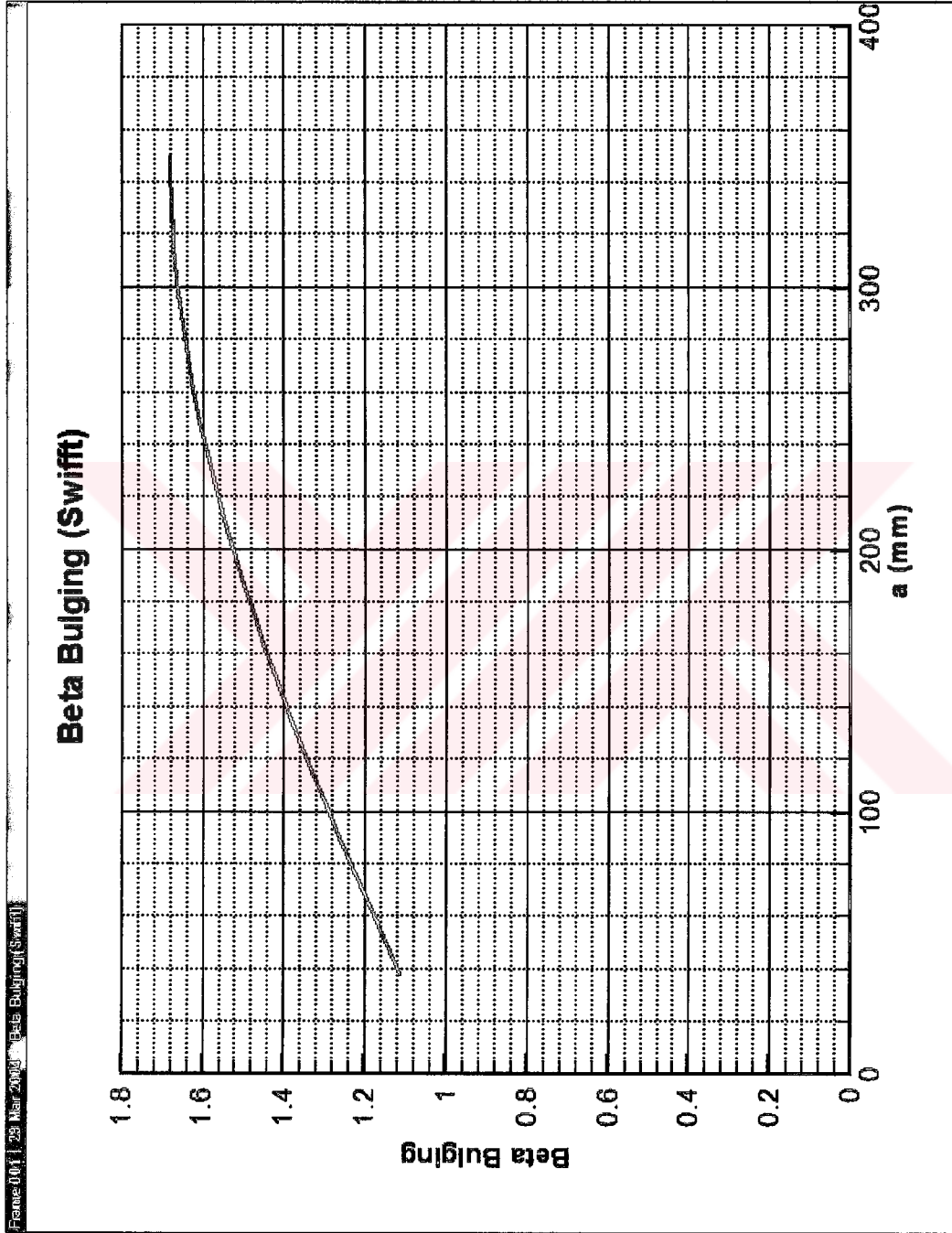


Chart 6.15. Beta Bulging (Chen) – a

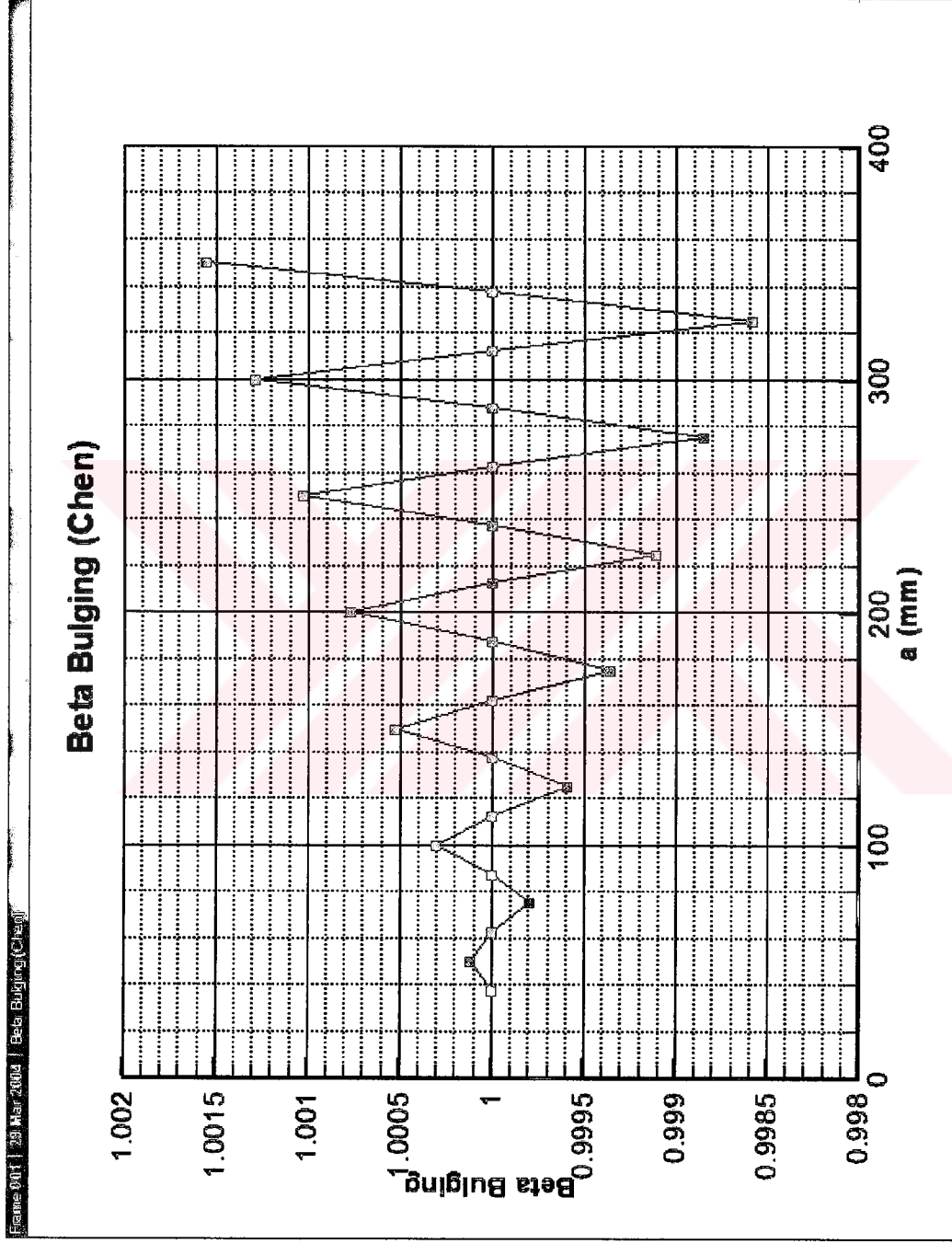


Chart 6.16. Beta Bulging (Chen - Swift)

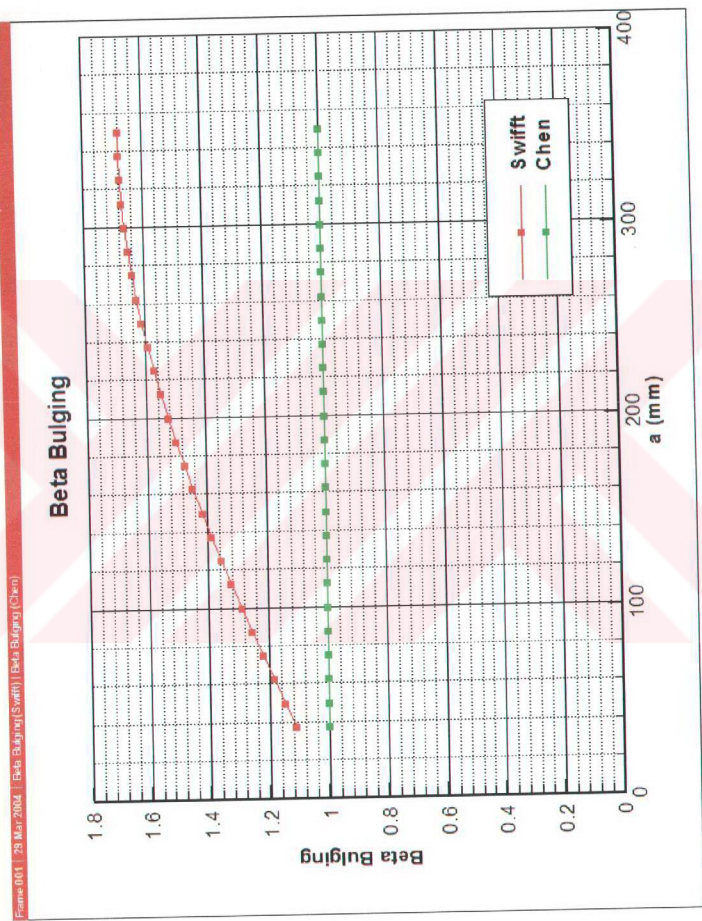


Chart 6.17. Beta Stiffening - a

Frame 101 | 25 Mar 2024 | Beta Stiffening

Beta Stiffening

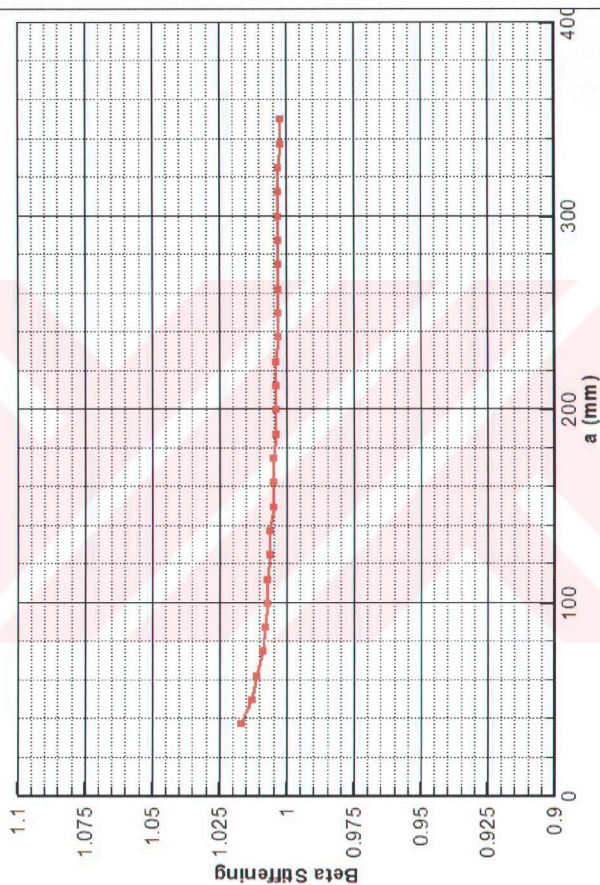


Chart 6.18. Stress Intensity Factor – a

Form 0101 / 29 Mar 2004 / Stress Intensity Factor

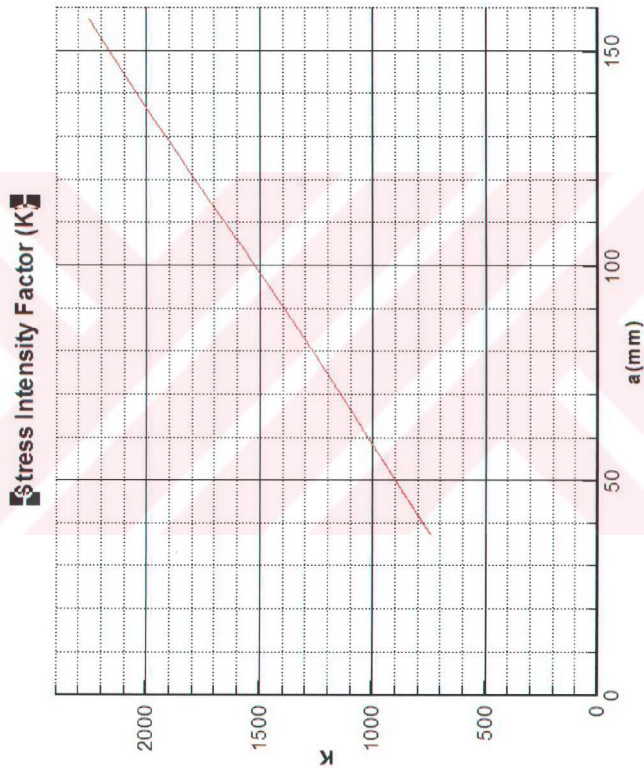
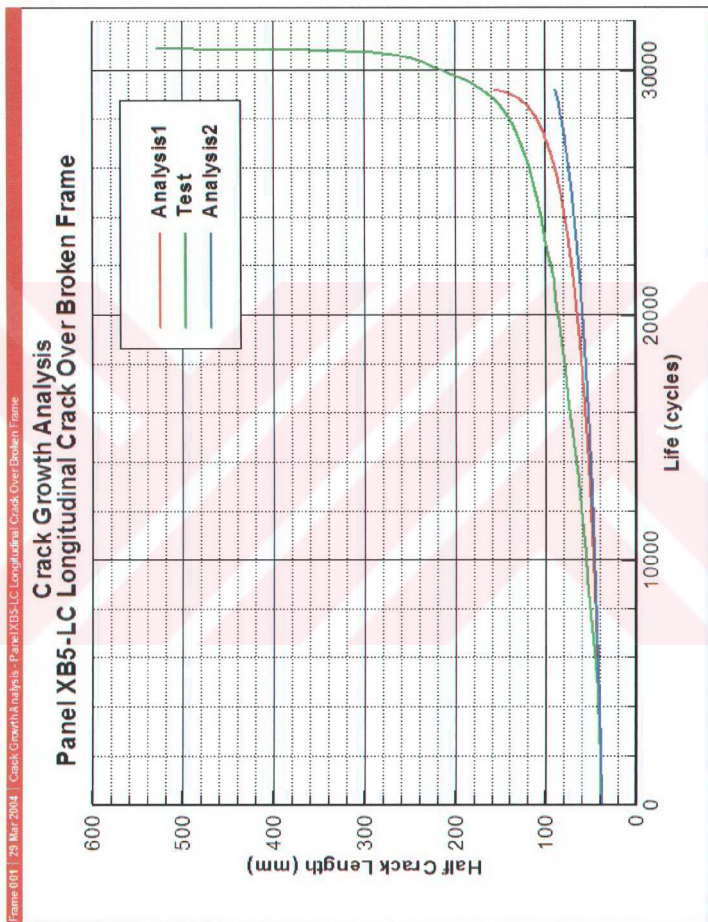


Chart 6.20. Crack Growth Analysis



7. CONCLUSIONS

7.1. Compared Conclusions Of Corner Cracks

In this chapter, the results of four different analysis are discussed in four sub-chapters. The results are discussed and compared with each-other according to the charts and tables given in Chapter 6 taking the literature and the teoritical aspects of fracture mechanics into account.

In the first and second examples of corner cracks analyzized, the stress distrubutions have similar values, only the corner crak – at offset hole in plate has an additional moment value which is written in the results in Chapter 6. The geometries and the materials are chosen different to find out the goemetry and the material effects. The materials used are the most common materials used in similar structures in aviation. The results are summarized below in Table 7.1. showing the most necessary comparison outputs.

In the first example, at hole in lug, the rate of da/dN is higher than dc/dN till the “a” value reaches “t”. After “a” propagates through the whole thickness, “c”s rate of growth is accelerated and “c” grows till it reaches the length of 34 mm. The net section stress exceeds the Flow Stress (the average of yield and ultimate stress) when the size of “c” is 33.7167mm.

When we look at the second application, at offset hole in plate, the width and the hole diameter are smaller than the first example. And the initial crack sizes are same. In this example the net section stress exceeds the Flow Stress when “c” is equal to 10.9063. It is clearly obvious from this example that although the thickness and the width of the material is smaller when compared with the first example and there is even a moment effect, the net section stress exceeds the Flow stress earlier than the crack in lug but nearly in the same cycles. When we take the yielding of Al 2024-T351 into account, that it is smaller than Al 7050-T7451, this can be explained as a geometry effect.

When we examine the charts of third example we see that there is an “Unstable Crack Growth” situation. The maximum stress value exceeds the critical value in the 16193.th cycle when “c” has a size of 26.8158mm. First the crack propagation is very very small for both “a” and “c” before they reach the same size. After this critical point, approximately for 2000 cycles it is impossible to make a prediction for the crack sizes as it becomes an unstable condition. After this time period, “a” is propagating very slowly reaching the thickness value when “c” is propagating with a high rate of propagation.

Table 7.1. Comparison of Three Different Applications of Corner Cracks

	CORNER CRACK AT HOLE IN LUG	CORNER CRACK AT OFFSET HOLE IN PLATE	CORNER CRACK AT EDGE OF PLATE
Material	Al 7050-T7451	Al 2024-T351	Al 2024-T3
t [mm]	23.0000	10.0000	4.0000
W [mm]	112.0000	48.5000	500.0000
D [mm]	42.0000	7.9000	
B [mm]		16.0000	
S0 [Mpa]	0.0000	51.9000	103.0000
S1 [Mpa]	0.0000	0.0000	23.0000
S3 [Mpa]	59.2000	106.0000	0.0000
Number of cycles applied	797414.8800	73538.6590	20771.1270
Yield [Mpa]	460.8000	350.5000	330.9000
a initial [mm]	1.2700	1.2700	2.5400
c initial [mm]	1.2700	1.2700	1.0000
a final [mm]	23.0000	10.0000	4.0000
c final [mm]	34.0000	11.0000	28.0000
			Maximum Stress Intensity
	Net Section Stress Exceeds the Flow Stress	Net Section Stress Exceeds the Flow Stress	Exceeds Critical Value
	At Cycle: 44148.800	At Cycle: 39860.84	At Cycle: 16193.73
	c at cycle 33.7167	c at cycle 10.9063	c at cycle 26.8158

7.2. Conclusions on Longitudinal Crack In A Curved Stiffened Fuselage Panel

The purpose of this study was to present a simplified semi-analytical methodology to predict the behaviour of longitudinal cracks in cracked stiffened curved panels with frames. The most well-known people who worked on this type of crack modelling were Chen and Swift. In the literature, several formulas can be found which allow the stress intensity factor of an unstiffened curved structure to be calculated. These values are often valid over a limited range of crack length and pressure loading. But in this study the model analyzed is a curved stiffened panel. Tests performed by Chen on unstiffened curved panels have indicated that a factor can be applied to the stress intensity factor of the flat structure in order to obtain the stress intensity factor of the curved structure.

In this study. In this type of crack modelling, there are factors that affect the models and the cracks due to the geometry, stress distribution which is non-uniform, and pressure differences to which the panel is subjected due to the inside pressure and the outside pressure [25]. Pressure loading acting on the longitudinal cracks induces geometrical non-linearity due to large deflections of crack edges. This methodology is based mainly on the application of "bulging" coefficients to stress intensity factors taken from a numerical model of a stiffened cracked panel in order to get the crack driving force of the curved panel. Bulging coefficients are taken into account from Swift's work on bulging coefficients.

The goal is to correlate the propagation behaviour of a longitudinal crack for another stiffened curved panel with the same "bulging" coefficient as determined with the numerical curved model. This is a riveted panel with skin, frames, stringers, tear straps and stringer cleats. The skin and frames are made of Al 2024 and the stringers made of Al 7349 alloy.

The crack was initiated with a saw cut located above a broken frame. This long artificial crack length measures 75mm. From this cut, the longitudinal crack propagated approximately perpendicular to the frame axis. The panel was subjected to cyclic pressure loading with a minimum-to-maximum load ratio $R=0$. It was also subjected to longitudinal tension-tension cyclic loading with a load ratio $R=0.1$. The biaxiality ratio was 0.3.

The charts plotted and the crack propagation results of the model shows that the correlation is good. It appears logical because the bulging coefficient has been calibrated with the numerical curved model to which the geometry is similar to that of panel. The conclusions reached can be summarized as follows:

- The experimental crack driving force curve of the curved panel obtained with the Forman relationship merges with the numerical curve obtained with analysis.
- The crack propagation correlation between numerical and test results is good.
- In terms of crack growth life, the analysis yields slightly conservative results as compared with test results.
- This study demonstrates that the growth of longitudinal cracks in a pressurized fuselage structure can be predicted in a conservative way by use of the stiffened flat numerical model corrected by "bulging" coefficients.
- Because of the pressure loading acting on the crack edges, the stress intensity factor is greater for a curved panel than for a flat panel. This overestimation of stress intensity factor can be modified by means of the "bulging" coefficients, which correct the stress intensity factor of a flat panel subjected to tensionloading.
- Geometrical non-linearity effects, which correspond to large deflections of the crack edges, are included in the "bulging" coefficients.
- In the case of a barrel fuselage section where bending is not present, the associated flat numerical model can be computed with the linear model; however, for a stiffened curved panel (part of a barrel fuselage section), the calculation must be computed with the non-linear model because of secondary bending.

A geometrical non-linear analysis of the model provides a more accurate determination of the effect of large crack-edge deflection. Analysis of the results shows a highly satisfactory correlation between prediction by calculation and experimental data.

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ÖZGEÇMİŞ

Sabriye Sera Atagündüz 05.09.1980 yılında İzmir’de doğdu. 1998 yılında İzmir Bornova Anadolu Lisesi İngilizce bölümünden mezun oldu. Aynı yıl İ.T.Ü. Uçak ve Uzay Bilimleri Fakültesi Uçak Mühendisliği bölümünde lisans öğrenimine başladı ve Haziran 2002’de Uçak Mühendisi olarak mezun oldu. Aynı yıl İstanbul Teknik Üniversitesi, Fen Bilimleri Enstitüsü, Uçak ve Uzay Mühendisliği Anabilimdalı, Uçak Mühendisliği Yüksek Lisans Programı’nda yüksek lisans eğitimine başladı.