Aircraft Structural Integrity Analysis through Fatigue Crack Propagation

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Abstract— *The life of an aircraft depends mainly on the specific usage of the aircraft. The manufacturer defines a lifetime for the general usage of the aircraft. But with the specific usage of the aircraft, the aircraft structure may fail prematurely or sometimes may have a life time exceeding the manufacturer specified limit. Retiring of an aircraft impose a great loss for the operator. In countries with limited economical capacity, this is a vast issue. Therefore in this research a method to analyse the structural integrity of aircraft is discussed which will reflect the condition of aircraft structure based on the usage. For this purpose, factors effecting the structural integrity are discussed in this research and fatigue crack propagation is identified as the major cause for structural failure in aircraft. Several models are identified for the purpose of analysing fatigue crack propagation and NASGRO model is selected as the most suitable model for this purpose. The most critical structural components of the aircraft structure is identified and is analysed using this model. A mathematical relationship is developed to predict the flight-cycles to failure of these structures. Then the structural integrity of aircraft is revalidated based on this calculation.*

Keywords— **Structural Integrity, Fatigue cracking, Revalidation**

I. INTRODUCTION

Aviation is an industry which focuses mainly on safety with a risk free environment, this incur a large capital cost for maintaining the safety. When an aircraft exceeds its lifetime, it should be kept out of service as per the manufacturer's instructions. As Sri Lanka is a country with limited economical capacity and depreciation of an aircraft is comparatively higher to the revenue it makes, retiring an aircraft may cause huge loss for the operator therefore necessary steps should be taken to revalidate and continue the service period. Therefore our approach is to increase the service life of the aircraft to maximize the revenue which it makes for the industry. The goal of this research is to revalidate the aircraft structure to be used for an extended time duration.

Normally manufacturer defined the aircraft life for the general usage, but according to the specific use of the operators, the life of the aircraft will vary. i.e. – An aircraft used for transporting corrosive cargo will age rapidly than a same type of aircraft used for transporting passengers. Thereby according to the usage of an aircraft may still be serviceable after the manufacturer defined life time has exceeded or it may become unserviceable before the manufacturer defined life time is reached.

Aircraft structures ageing mechanisms are fatigue through repetitive cycles, wear, corrosion and deterioration. Fatigue has led to many structural failures in aircraft, therefore this factor is considered in the scope of this research.

Fatigue failures occurred in aircraft structures are mainly due to fatigue cracks, sudden failure, and deformation. In here fatigue cracks has proven to be the major mode of failure. Therefore in this research fatigue crack propagation is used to analyse the structural integrity of aircraft.

In this research sample data is collected from DHC-1(Chipmunk) and Y-12 which are used by the Sri Lanka Air Force. From these collected data and past occurrences, critical points in aircraft structures have been identified. Then fatigue analysis is done through fatigue crack propagation for the sample and remaining life is calculated.

A. FRACTURE MECHANICS

Fracture mechanics is used to predict propagation of a crack in a metal element. In here the driving force on a crack is calculated by using analytical methods.

In fracture mechanics fractures are categorized into 3 modes of crack propagation. (Zehnder, 2012)

- Mode I- Opening Mode
- Mode II- Sliding Mode
- Mode III- Tearing Mode

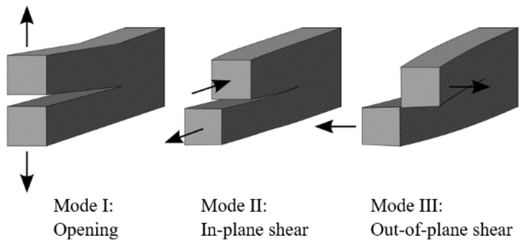

Figure 1: Modes of crack propagation

Here, the stress field inside a linear elastic body with an isolated crack is described by the stress intensity factor (SIF), K. The rate of propagation of the crack is predicted by using this K value. This K factor is an indication of the stress level at the tip of a fatigue crack. Therefore by this a relationship between this K value, fatigue crack growth and failure can be established. The K value is given subscripts KI, KII, KIII depending on the mode of the crack. (Irwin, 1957).

SIF or the K value depends on the geometry of the element and the size and the location of the crack.

Mode I deformation will occur due to tension, mode II deformation will occur due to Shear force and Mode III will occur due to an out-of-plane Shear Force.

Past experimental and theoretical fracture and crack growth studies have indicated that the Mode I crack propagation most common mode in fatigue failures. (Shah, 1974) Therefore in the scope of this research the deformation is assumed to tension mode deformation (I mode) and the crack propagation is assumed to be two dimensional.

Then for mode I cracks, SIF can be expressed as,

$$K = M_e M_p \sigma_\infty \sqrt{\frac{\pi a}{Q}}$$

Where,

M_p = Crack geometry magnification parameter

M_l = Elastic magnification parameter

$\sigma\infty$ = Remote uniaxial tensile stress

a = One-half of the crack length

Q = Plastic correction factor

$$Q = (\emptyset)^2 - 0.212\left(\frac{\sigma\infty}{\sigma y}\right)^2$$

Where,

σ_y = Yield stress

$\emptyset$ = Complete elliptic function

Given by,

$$\emptyset = \int_0^{\pi/2} \sqrt{1 + \left(\frac{c^2 - a^2}{c^2}\right) sin^2\theta}\; d\theta$$

(Kobayashi & Moss, 1969)

Where,

c = half of length of surface crack

a = crack length for an edge crack or half crack length for through thickness crack or depth of a surface crack.

These are discussed in detail in the following section.

The cracks according to their location will have different properties and growth rates. There have been many models developed for this and one used in linear elastic fatigue crack growth prediction is generalizing cracks into categories according to their nature and developing stress intensity factors for them. (Kobayashi & Moss, 1969)

Therefore the cracks that occur in a material can be generalized in to 3 models as

- Surface cracks
- Through-thickness cracks
- Edge cracks (Liu, 1996)

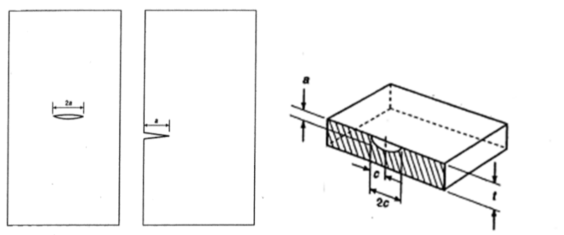

Figure 2: Crack nature in a material

For these crack models, the crack geometry magnification parameter is experimentally taken as (Walker, 1970)

Mp =1 for through-thickness crack

Mp = 1.12 for surface and the edge cracks

And the Elastic magnification parameter as

Ml = 1 for shallow crack

Ml = 1.6 as the depth of the crack reaches the back of the plate (Walker, 1970)

There are several models developed to predict crack growth under fatigue conditions. The most commonly used models in fracture mechanics are,

- Paris-Erdogan Equation
- Walker Equation
- NASGRO Equation

The fatigue crack growth for any material has three stages. It consists of a threshold region where the crack growth is not present below a threshold value and a sudden crack growth is present which will decelerate into a straight line which is called the Paris region where the crack growth is nearly a straight line and it will accelerate again into the third region which is the fracture region where a rapid crack growth will be present until it will reach the fracture toughness which will lead to the fracture of the material.

This graph has been basis for most of the linear elastic fracture mechanics models. (Cui, 2002)

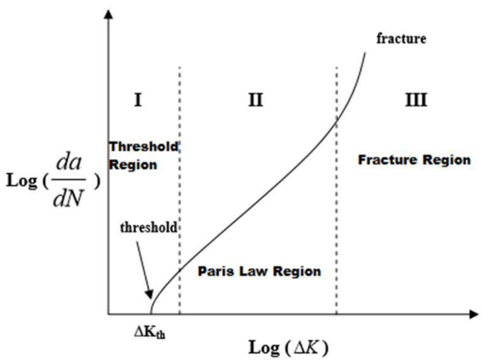

Figure 3: Fatigue crack behaviour graph

1. Paris-Erdogan Equation: This is the earliest approach in linear elastic fracture mechanics which was developed by P.C. Paris. It gives a simple relationship between the fatigue crack propagation and Stress Intensity Factor. This model takes into account only the straight region of the crack growth rate curve which makes this model less accurate. (P.C. Paris, 1961) Though this is the most popular model in facture mechanics many new models have been developed by taking this relationship as the basis or otherwise including factors other than just the Stress intensity factor 'K' which in turn will include other regions of the crack growth rate curve in those models. (Roberts, 2002)

$$\frac{da}{dN} = C\Delta K^{m}$$

2. Walker Equation: Walker equation is one such development taking Paris law as the base and in cooperating factors such as stress ratio. (Walker, 1970) The major problem with this mode is although this model takes in to account correction factors and stress ratio, this model also considers only the Paris region of the crack growth curve. (R.G. Forman, 2005)

Walker equation is given as,

$$\frac{da}{dN} = C(K_{max})^{m}\,(1-R)^{n} = c\,(\Delta K)^{m}\,(1-R)^{n-m}$$

Where,

c, m, n = Material Constants

K_{max} = Maximum stress intensity factor

ΔK = Stress intensity amplitude

R = Stress ratio

3. NASGRO Equation: Another such development is NASGRO equation developed by R. G. Forman and S. C. Newman in 1992. (Forman & Mettu, 1992). A major advantage in this equation is the inclusion of the effect of the non-linear region of the stress intensity range curve, which can lead to more accurate predictions. Further this model has an extensive database developed for the use by NASA and thereby appropriate data are available for wide range of materials abundantly. (Forman, et al., 2002)

Given as,

$$\frac{da}{dN} = c\left(\left(\frac{1-f}{1-R}\right)\Delta K\right)^{n} \frac{\left(1-\frac{\Delta K}{\Delta K_{th}}\right)^{p}}{\left(1-\frac{K_{max}}{K_{c}}\right)^{q}}$$

Where,

N = Number of applied fatigue cycles

R = Stress ratio

ΔK = Stress intensity amplitude

a = Crack length

f = Explanations of the crack opening function

ΔK_{th} = Threshold stress intensity factor

Kc = Critical stress intensity factor

C, n, p, and q are empirically derived constants

This equation is used by NASA to examine the crack propagation in space shuttles and aircrafts. There have been wide range of application of this equation. A well developed and maintained database for the constants is available for NASGRO analysis which is primarily developed maintained by NASA.

For this research the NASGRO equation is selected considering there is a well maintained data base, it includes many parameters other than just the stress intensity factor and it includes non-linear regions of the stress intensity range curve which leads to a higher accuracy of the calculations.

In order to apply this equation, the stress applied throughout the analysis must be constant. But in aircrafts, the loading varies significantly. Therefore in order to apply this variable amplitude loading in to the NASGRO equation, Half-Cycle theory is used.

4. HALF-CYCLE THEORY

Half cycle theory states in a stress cycle applied to an element, the damage caused by the complete stress cycle is equally divided between the increasing phase and the decreasing phase of the load. Therefore each cycle can be divided in to two half cycles of equal constant amplitude loading under the same loading magnitude (ΔK and R). Therefore the damage can be computed separately and the total damage can be assessed at any cycle in the stress cycle history of a variable amplitude loading history.

II. METHODOLOGY

A. DATA COLLECTION

In order to analyse the structural integrity of aircraft, first the critical structural points of the aircraft which are most prone to failures should be identified. In order to identify the critical point data should be collected from structural repair manual, aircraft repair manuals, inspection data sheets, technical news sheets, task cards and also via questionnaire's and expert interviews. Initially data must be collected qualitatively and finally at the end result it must be analysed quantitatively and critical points should be identified.

After having identified the critical points of the aircraft, the crack sizes existing in those points should be identified and measured. Then the size of the biggest crack should be recorded and analysis of the integrity should be done by using this size as a data, since the biggest crack will cause the particular structural component to fail, first. Identifying and measuring the crack size of the identified critical points is given in detail in the following sections.

B. CRACK MEASUREMENT

To identify the cracks present in the structure of the aircraft several type of NDT techniques can be used. Such as Ultrasonic inspection, X-rays, Magnetic particle Inspection, Eddy current inspection, Liquid penetrant inspection and Visual inspection. Out of these, eddy current testing can identify surface cracks only and can be used only for conductive components. Magnetic particle inspection method is only applicable for Ferro-magnetic components. The liquid penetrant test can only be used for detect cracks visible to the surface. But Ultrasonic and X rays NDT methods can detect both surface and subsurface defects with a high degree of sensitivity of any type of material with minimal surface preparation. These two methods can also be used to inspect assembled components. Therefore by considering these factors Ultrasonic and X rays NDT methods are selected as the most efficient methods to be used in this research. By using these two NDT methods, the crack sizes in the critical points of the aircraft is identified. In order to analyse fatigue cracks aircraft structures, size of these cracks must be found and also crack shape must be measured.

Cracks can be measured in different techniques. Following are suitable for measuring of cracks of different sizes. Optical microscopy, Acetate replica method, In-situ scanning electron microscope (SEM) method, Direct current potential drop (DCPD). Out of the 4 methods optical microscopy method with the help of stereomicroscope is the most commonly used method as it's readily available in a country like ours and it has magnification of 600x and above.

C. FAILURE MODES

Once the critical structural points have been identified and the crack sizes of those points is measured, an analysis can be done on these structures and the life time can be defined. The identified critical points of the aircraft may fail under the action of 3 major factors.

1. Corrosion
2. Hydrogen Embrittlement
3. Fatigue

(Findlay & Harrison, 2002)

Out of these factors, Fatigue is the most common failure mode in aircraft structures caused by the process repeating applied load which weakens the material. According to statistics, 60% of failures in aircraft structure have occurred due to fatigue failures. (Findlay & Harrison, 2002) (Bhaumik, et al., 2007)

Mainly fatigue failures which are occurred in aircraft structure can categorized into three types that are,

1. Deformation due to fatigue
2. Brittle failure due to fatigue
3. Fatigue crack propagation

80% of all fatigue failures are due to fatigue cracks, whether in association with cyclic plasticity, sliding or physical contact, environmental damage, or elevated temperatures. (Ritchie, 1998) Most of fatigue cracks can be the cause of catastrophic failures.

The crack growth due to fatigue, it has three separate stages

- Initiation of the fatigue crack
- Propagation of the fatigue crack
- Finally, sudden failures

To reduce the probability of crack generating due to fatigue in an aircraft structure material and its design must be taken in to the consideration when manufacturing. When designing the aircraft structural components normally it is designed with the inspection safe life below the fatigue cracking points. But according to the usage, the life will vary significantly and cracks may develop prematurely. If these are properly analysed, a life can be given to the structural components based on the specific usage of the aircraft and the life can be increased beyond the manufacturer specified life span.
Therefore based on these findings, the scope of this research is limited to analysing of the failures which are occurred due to fatigue crack propagation.

D. CALCULATION DEFINITION

1. NEWMAN'S CRACK OPENING FUNCTION (f)

A crack opening stress equation has been developed by Newman for structures under constant amplitude loading and is used in this research for the purpose calculation of the fatigue crack growth. (Tong, et al., 2007). For this case it is defined as,

$$f = \frac{K_{op}}{K_{max}} = (A_0+A_1R+A_2R^2+A_3R^3)$$

Where,

$$A_0 = (0.825 + 0.34\alpha + 0.05\alpha^2)\left[\cos\left(\frac{\pi}{2}\frac{\sigma_{min}}{\sigma_0}\right)\right]^{\frac{1}{\alpha}}$$

$$A_1 = (0.415 - 0.071\alpha)\frac{\sigma_{min}}{\sigma_0}$$

$$A_2 = 1 - A_0 - A_1 - A_3$$

$$A_3 = 2A_0 + A_1 - 1$$

And,

$$\sigma_0 = \frac{\sigma_u + \sigma_y}{2}$$

Where,

σ_y = Yield stress

σ_u = Ultimate tensile stress

α = Plain stress/ strain constraint factor

The plain stress/strain constraint factor is defined as being between 1 and 3, 1 being for plain stress and 3 being for plain strain. (REMBECK & SJÖBLOM , 2012)

2. STRESS RATIO(R)

Stress Ratio is the ratio between maximum and minimum stresses of a constant amplitude load cycle, given as,

$$R = \frac{\sigma_{min}}{\sigma_{max}}$$

3. THRESHOLD INTENSITY FACTOR (ΔKth)

Threshold intensity factor range is the region of SIF where below this value a crack growth will not occur. The value of Threshold intensity factor is not a constant for a specimen and has to be calculated as the each type of fatigue loading considered. This has been developed by M. Klesnil and P. Lukas and value of it is given by,

$$\Delta K_{th} = \Delta K_o(1 - R) \qquad \text{(Farahmand, 2001)}$$

Where ΔKo is the stress intensity range at R=0

4. CRITICAL STRESS INTENSITY FACTOR (Kc)

The value of Critical stress intensity factor (Kc) is equal to the plain structure fracture toughness (KIC) of a material which is constant for the type of material independent of the load or the configuration of the loading.

$K_c = K_{IC}$ (DAVENPORT & BROOK , 1979)

E. CALCULATION

If the biggest crack on a selected structure is a_0, the maximum permittable crack size for that structure is a_m, and the crack growth per flight is Δa assuming that a load cycle is equal to flight cycle and that crack growth per flight is constant, the number of remaining flights F0 can be given as,

$$F_0 = \frac{a_m - a_0}{\Delta a},$$

By reference to research done on structural integrity analysis (William, 1987)

But in reality, the crack growth per flight is not constant, it will rather increase rapidly with the number of load cycles. Therefore the actual remaining flights will be significantly less than the value calculated from this equation. To correct this the NASGRO equation will be used. In order to use this equation, the load cycle must be of constant amplitude. But in reality, the loading is not constant and is varying. Therefore in order to make the load cycle constant to be applied in to this equation, Half Cycle Theory is used.
According to half cycle theory, the crack length increment at Ith half cycle is summation of all previous half cycles,

$$\Delta a_{\ell} = \sum_{i=1}^{2N_l} \frac{\delta a_i}{2}$$

Where δai/2 is crack growth in the i[th] half cycle under loading ΔK_i and R_i. And it is assumed that it is equal to crack growth by a half cycle of constant amplitude under the same load.

Which gives the number of remaining flights as at the,

$$F_1 = \frac{a_m - a_o}{\sum_{i=1}^{2N_l} \frac{\delta a_i}{2}} = \frac{a_m - a_0}{\Delta a_{\ell}}$$

Therefore according to NASGRO equation, δai/2 can be calculated as

$$\frac{\delta a_i}{2} = \frac{1}{2}\left[\frac{da}{dN}\right]_i = c\left(\left(\frac{1-f}{1-R_i}\right)\Delta K_i\right)^n \frac{\left(1-\frac{\Delta K_i}{K_{th}}\right)^p}{\left(1-\frac{K_{max\,i}}{K_c}\right)^q}$$

By integrating this equation and applying equation for Stress Intensity factor 'K', the crack growth in the lth load cycle can be obtained as,

$$\Delta a_{\ell} = c\left(\left(\frac{1-f}{1-R}\right) M_p M_{\ell}(\sigma_{max}\text{-}\sigma_{min})\sqrt{\frac{\pi a_{\ell\text{-}1}}{Q}}\right)^n \frac{\left(1-\frac{M_p M_{\ell}(\sigma_{max}\text{-}\sigma_{min})\sqrt{\frac{\pi a_{\ell\text{-}1}}{Q}}}{\Delta K_o(1-R)}\right)^p}{\left(1-\frac{M_p M_{\ell}\sigma_{max}\sqrt{\frac{\pi a_{\ell\text{-}1}}{Q}}}{K_{\ell}}\right)^q} N_{\ell}$$

Where $N_{\ell} = \{1, 2, 3, \ldots\ldots, \ell\}$

Assuming that f, R constant and constant amplitude stress cycle for all load cycles, the following equation can be obtained for simplicity of calculations.

$$\Delta a_{\ell} = D(a_{\ell\text{-}1})^{\frac{n}{2}} \frac{\left(1-k_1 a_{\ell\text{-}1}^{\frac{p}{2}}\right)}{\left(1-k_2 a_{\ell\text{-}1}^{\frac{q}{2}}\right)} N_{\ell}$$

Where,

$$D = c\left(\left(\frac{1-f}{1-R}\right) M_p M_{\ell}(\sigma_{max}\text{-}\sigma_{min})\sqrt{\frac{\pi}{Q}}\right)^n$$

$$k_1 = \left(\frac{M_p M_{\ell}(\sigma_{max}\text{-}\sigma_{min})\sqrt{\frac{\pi}{Q}}}{\Delta K_o(1-R)}\right)^p \quad k_2 = \left(\frac{M_p M_{\ell}\sigma_{max}\sqrt{\frac{\pi}{Q}}}{K_c}\right)^q$$

If the component is to fail in the F_x[th] flight cycle, then

$$\frac{a_m - a_0}{\Delta a_1} = \frac{\Delta a_1 + \Delta a_2 + \Delta a_3 + \ldots + \Delta a F_x}{\Delta a_1}$$

If the flight cycles is assumed to be constant which makes σ_{max}-σ_{min} constant.

Therefore, In reference to the method proposed in similar research (William, 1987),

$\Delta a_{\ell}/a$ <<<1, p and q <2, Therefore assuming $(\Delta a_{\ell}/a)^{n/2} = \frac{n}{2}\frac{\Delta a_1}{a_0}$ and $a_{\ell\text{-}1} + \Delta a_{\ell}$ <<1,

$$\Delta a_1 = D(a_0)^{\frac{n}{2}} \frac{(1-k_1 a_0)^{\frac{p}{2}}}{(1-k_2 a_0)^{\frac{q}{2}}} N_1$$

$$\Delta a_2 = D(a_1)^{\frac{n}{2}} \frac{(1-k_1 a_1)^{\frac{p}{2}}}{(1-k_2 a_1)^{\frac{q}{2}}} N_2 = D(a_1)^{\frac{n}{2}} \frac{(1-k_1(a_0+\Delta a_1))^{\frac{p}{2}}}{(1-k_2(a_0+\Delta a_1))^{\frac{q}{2}}}$$

$$\Delta a_3 = D(a_2)^{\frac{n}{2}} \frac{(1-k_1 a_2)^{\frac{p}{2}}}{(1-k_2 a_2)^{\frac{q}{2}}} N_3 = D(a_2)^{\frac{n}{2}} \frac{(1-k_1(a_1+\Delta a_2))^{\frac{p}{2}}}{(1-k_2(a_1+\Delta a_2))^{\frac{q}{2}}}$$

$$\Delta a_{\ell} = D(a_{\ell\text{-}1})^{\frac{n}{2}} \frac{(1-k_1 a_{\ell\text{-}1})^{\frac{p}{2}}}{(1-k_2 a_{\ell\text{-}1})^{\frac{q}{2}}} N_{\ell} = D(a_{\ell\text{-}1})^{\frac{n}{2}} \frac{(1-k_1(a_0+\Delta a_1))^{\frac{p}{2}}}{(1-k_2(a_0+\Delta a_1))^{\frac{q}{2}}}$$

$\Delta a_{\ell}/a$ <<<1, p and q <2, Therefore assuming $(\Delta a_{\ell}/a)^{n/2} = \frac{n}{2}\frac{\Delta a_1}{a_0}$ and $a_0 + \Delta a_{\ell}$ <<<1,

$$\frac{\Delta a_2}{\Delta a_1} = \left(\frac{a_1}{a_0}\right)^{\frac{n}{2}} = \left(\frac{a_0+\Delta a_1}{a_0}\right)^{\frac{n}{2}} = 1 + \left(\frac{n}{2}\frac{\Delta a_1}{a_0}\right)$$

$$\frac{\Delta a_3}{\Delta a_1} = \left(\frac{a_2}{a_0}\right)^{\frac{n}{2}} = \left(\frac{a_0+\Delta a_1+\Delta a_2}{a_0}\right)^{\frac{n}{2}} = 1 + 2\left(\frac{n}{2}\frac{\Delta a_1}{a_0}\right)$$

$$\frac{\Delta a_{\ell}}{\Delta a_1} = \left(\frac{a_{\ell\text{-}1}}{a_o}\right)^{\frac{n}{2}} = \left(\frac{a_o+\Delta a_1+\Delta a_2+\Delta a_3+\ldots+\Delta a_{\ell}}{\Delta a_o}\right)^{\frac{n}{2}} = 1 + (l-1)\left(\frac{n}{2}\frac{\Delta a_1}{a_o}\right)$$

If the component is to fail in the F_x[th] flight cycle,

$$\frac{a_m - a_0}{\Delta a_1} = \frac{\Delta a_1 + \Delta a_2 + \Delta a_3 + \ldots + \Delta a F_x}{\Delta a_1}$$

$$F_1 = (1+1+1+\cdots) + \left(\frac{n}{2}\frac{\Delta a_1}{a_o}\right)(1+2+3+\cdots+(l-1))$$

Therefore by sum of arithmetic progression,

$$F_1 = F_x + \left(\frac{n}{2}\frac{\Delta a_1}{a_o}\right)\left(\frac{F_x - 1}{2}\right) F_x$$

Where F1 is the number of flight remaining assuming $\Delta a_1 = \Delta a_2 = \Delta a_3 = \ldots\ldots\ldots = \Delta a_{Fx}$

Therefore F_x, the number of flights to failure can be obtained as,

$$F_x = \frac{2a_0}{n\Delta a_1}\left(\sqrt{1 + \frac{n\Delta a_1}{a_0} F_1} - 1\right)$$

Where,

n = material constant of NASGRO equation

F_1 = number of flights remaining taking crack growth as constant

$$F_1 = \frac{a_m - a_0}{\Delta a_1}$$

Δa_1 = crack growth in 1st flight

a_0 = Initial crack size present

By using this equation, the life cycles to failure of a particular selected aircraft structure can be analysed according to principles of fatigue crack propagation. Therefore using this method, the life cycles to failure of that particular structure can be calculated. Therefore the structural integrity of that particular structural component can be revalidated based on this calculated life cycles to failure.

F. REVALIDATION OF AIRCRAFT STRUCTURE

Fatigue crack analysis can be used due to extend service of structural component. Evaluation of structural component is directly related to WFD (wide spread fatigue damage) and it requires a complete analysis. For the safe operation of aging aircraft prevention of WFD is an important issue. In this research the structural analysis is done and Number of cycles to failure is predicted. But a structure in an aircraft cannot be used for total of this analysed life-to-failure. Therefore a proper model is defined in this paper to use a structural component for an extended time period without posing a critical safety risk. This includes increasing the maintenance and inspection for the analysed component and if needs to be used even further a structural modification should be done. Two points, ISP and SMP is defined for this purpose.

Inspection start point (ISP) is the point where special WFD inspection must be started it consist of detailed inspection of MED and MSD.

Structural modification point (SMP) which is an operational limit, beyond which a structural item may not be used without modification because of the increase risk of WFD. Beyond the SMP point airplane may not be operate without further evaluation and modification. (Trey & Schmidt, 2001)

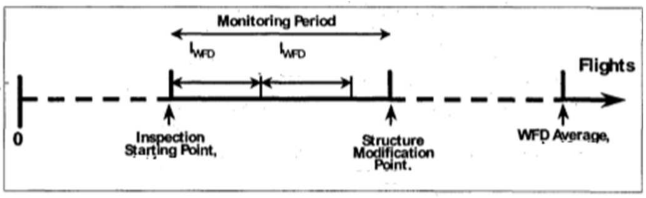

Figure 4: Crack nature in a material

After calculation of the remaining life cycle for the crack,

- First quarter of the flight cycle is done with manufacture recommended inspections for the crack till inspection starting point reaches.
- Then next two quarter followed by the first quarter is done with extra inspections till structure modification point reaches.
- For the last quarter structural modification are done for the structure component.

III. RECOMMENDATIONS

- The life of an aircraft structure will vary according to the specific use of it by the operator.
- Therefore the aircraft may fail before the manufacturer specified time intervals are reached.
- An aircraft structure can be used beyond the manufacturer specified time intervals if the structure is analysed properly and respective safety measures are taken.
- Fatigue is the predominant cause of failure in aircraft structures.
- Fatigue cracking is the major failure mode in fatigue.
- Therefore by analysing the fatigue crack propagation of aircraft structure, a life span can be defined according to the previous usage of the aircraft.
- This can be done using a combination of NASGRO equation and half cycle theory.
- The defined life time to failure should not be used as-is in order to maintain safety of the aircraft.
- A safety limit is defined and inspections and maintenance is increased and a structural modification should be done if the component is nearing the failure life.
- By doing these, a life span can be defined for aircraft structures by a process of fatigue crack analysis of the structure.
- This can be used on ageing aircraft to ensure safety and to determine the actual life remaining of the aircraft according to the usage of the aircraft which may be more than or less than the manufacturer defined general life.

IV. Conclusion

In this research it is shown that the main reason for aircraft structural failure is fatigue failure by fatigue crack propagation. Fatigue is the most common failure occurred in aircraft and it has the ability to cause catastrophic failures and mainly the failure modes of fatigue is briefly explained in this research. In here the remaining life of aircraft structure is predicted by a process of critical point identification and fatigue crack analysis of those structures. Fracture mechanics used to predict the propagation of crack and here the analysis done for the most common failure mode of fracture mechanics. For this analysis NASGRO equation is identified as the most suitable model. In this model constant amplitude load cycles must be used therefore, in order to transform the variable amplitude loading of the aircraft to constant amplitude loading cycles Half cycle theory has been used and application of NASGRO equation for the purpose of life prediction of the aircraft structure is discussed. In this research structural analysis is done through the crack propagation in order to predict the remaining cycle numbers by using a model to revalidate the aircraft structure based on this analysis as a region of life to be used without extra monitoring or maintenance, a region with increased inspection and a region to be used only with structural modification.

Therefore, through this analysis the fatigue life of an aircraft structure depends mainly on the stress level applied to the structure and the stress intensity factor. The maximum and minimum stress level must be measured by using strain gauges and crack sizes also should be measured for this purpose. These cracks and it size should be identified by using NDT inspections, to find the cracks ultrasonic and X ray inspections are used and the size of the cracks are measured by using optical microscopy method, because in NDT methods X ray and ultrasonic methods are more suitable than other methods. In this research paper for the purpose calculating the fatigue crack growth Newman's crack opening function has been used. Case study of this research has been applied for the identified critical structure in the selected samples which include DHC-1 aircraft of SLAF. And the remaining life cycles numbers were calculated with the above mentioned analysis method. Therefore, by doing this analysis the aircraft's inspection starting time and the structural modification can be predicted. This paper includes a method which can be used on ageing of aircraft and its structure to ensure the aircraft safety and also it will help to find out the actual remaining life of the aircraft.

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