ANALYSIS OF FATIGUE CRACK INITIATION OF ON-CONDITION AIRCRAFT STRUCTURAL COMPONENTS

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Abstract— *This project aims to analyses the fatigue crack initiation of an on condition component and assigns more reliable and accurate inspection intervals to identify the failure soon after it occurs. The current method of maintaining On-condition parts of an aircraft involves the method of identifying the failed components after some time of its failure by next scheduled inspection of that component. The new method is introduced by this research to overcome the previous method and develop more accurate inspection intervals for those components. From this method the use of failed on condition components and the adverse effects that can occur from that particular components to other components such as excessive stresses that applies on the fail safe designs can be eliminated and increase the usability of the components for some extend. Other than this by identifying a miner failure of an on-condition component by the use of revalidation techniques the usage of the components can be extended. For the development of the new method the approach is taken by implementing Re-Engineering methods for such the components. The material behaviour under fatigue loads and the relationship between the fatigue, stress, material properties and life cycles are given by the Ramberg-Osgood Equation and Morrow equation. For the calculation the details such as mean stress of the component, Monotonic properties and cyclic properties of the material, strain-Life properties are required. After the calculation the life cycles for a fatigue crack initiation of that component can be identified and this data can be used for Maintenance program development of assigning inspection intervals.*

Keywords— **On-condition, Ramberg-Osgood equation, Morrow equation, Fatigue Crack**

I. INTRODUCTION

In aircraft's there are millions of parts, they are mainly categorized into different systems and for sections in aircraft. They are also categorized based on its maintence as well. Among them On-condition parts are fall in to one category. On condition is a preventive primary maintenance process that involves in inspection periodically or check against some appropriate physical standards to determine if it can continue in service. These standards can be adjusted based on operating condition, tests as appropriate to maintenance or reliability programs. In this research it's only focuses on on-condition components. Most of the maintenances that we do in the aircraft are scheduled maintenance. This type of maintenance will be conducted as per the manufacture given in the maintenance manual.

The concept of On-Condition Maintenance came into view from the aviation industry as for the purpose to reduce cost of operation without compromising reliability and performance. The maintenance program published by CAA (Civil Aviation Authority) of India state that the On Condition is a failure preventive process which the component is monitor or inspected at specified intervals to determine the condition of the component and to determine whether it can continue in service. The primary purpose of On Condition is to remove an item before its failure in service. It is not a philosophy of 'fit' until failure or 'fit and forget it'.

This research is done by only considering mechanical components. The reason for selecting mechanical components is because majority of aircraft mechanical components do not fail unexpectedly, it gives some warning or sign of the fact that they are about to fail. The sign will be a crack or a deformation. These warnings or signs are called Potential Failures the word “Potential failures” can be defined as identifiable physical conditions which indicate that a functional failure is about to occur or is in the process of occurring. The amount of warning given by different potential failures varies from microseconds to decades. Depending on the material and the operating condition. Longer warning intervals mean greater maintenance task intervals. These factors will be considered when assigning life time for a component. In a mechanical system of an aircraft the components can be fail in many deferent ways. But among all these failures statistical experiments have shown that most common failure type is the failure due to fatigue. Even though it's said that the purpose of On-condition is to identify it before fail it may fail before the next scheduled inspection of the part assembly. To prevent it before the next schedule inspection this research introduce a methodology. Using that the inspection schedules can change accordingly.

II. METHODOLOGY

A. Common Failure Modes Arise In Aircraft Structures

Failures can occur in aircraft structural component due to the catastrophic consequences. The results of these failures are loss of life in structural components in aircraft. The purpose of the investigation of defects in aircraft structures is preventing further incidents and reduces its damage minimum. This review of this discusses is what are the failures type can occur in aircraft structures and what is the most common type observed in aircraft structures. The examples drawn from case histories.

In general, failures occur when a components/structures withstand the large quantity of stresses or imposed on it in longer operation time period. Commonly, failures are related with stress which can occur for several reasons including

• Design errors,-e.g. unwanted space in holes, notches, and tight fillets
• The material which made the aircraft structures contain voids
• Corrosion attack of the material- e.g. pitting corrosion. (It can generate a local stress concentration)

This data shows (figure 1) that the incidence of fatigue failure is the most prominent failure mode which occurs in aircraft structures or components in service (Harrison, p. 20). The rectification of corrosion attack in aircraft also shows an effective damage for the aircraft in service. However it needs more effort than the repair of fatigue cracking.

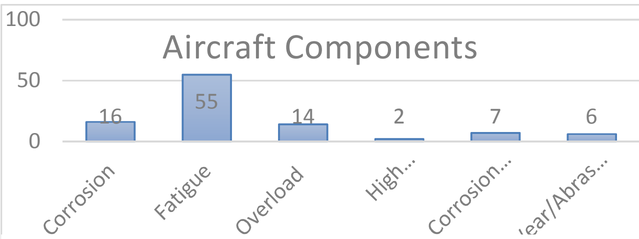

Figure 1 Failure Modes

The metal is subjected to a repetitive or fluctuating stress that will fail at a stress much lower than that required to cause failure on a single application of load.
Failures occurred under conditions of dynamic loading are called **fatigue failures**.
Structural parts or Components that is failed by fatigue usually undergo three separate stages of crack growth. These are:

- Crack Initiation -This can be forced by stresses such as material defects or design.
- Crack Propagation -This is progressive cyclic growth of the crack.
- Final Fracture - When the propagating crack reaches a critical size at which the remaining material cannot support, the rupture occur

Factors affecting Fatigue life

- Loading Conditions
  - Type of stress
  - Stress amplitude, mean value
- Condition Structural Member
  - Stress concentrations
  - Surface finish
- Material
  - Thermal history (e.g. grain size in metals)
- Environmental conditions
  - Temperature
  - Corrosion effects

Fatigue cracks are the most common reason of structural failure in aircraft. Fatigue cracks occurring can be reduced with the help of considering material and their design. But it cannot be reduce or removed 100% completely.

Under this research the main focus going in to the crack initiation of fatigue crack. From all three stages of fatigue crack failure, the initial stage has been considered. Under on-condition components, the components that fail due to fatigue crack has been considered. The on-condition components doesn't have life, due to that it should be identify before it fails completely. This identification is done by the identifying the duration that takes by that components to initiate the crack. Therefor in this research we mainly focus on calculating the duration that takes to initiate the crack.

B. Theory

This research is going to be done on the initiation of the crack and revalidation for the crack that initiated. There are a number of methods that use to calculate the amount of the time that takes to initiation of the crack. This time has been given in fatigue cycle. When consider analysing uniaxial fatigue life evaluation and multi axial fatigue life evaluation there are several procedures used. These procedures can be divided into two sectors as "crack initiation" or "crack propagation approaches". There are researches that has been done and found that there are 3 phases that involve in fatigue failure process. (Stevan Maksimovi, 2005)

Phase 1-Crack initiation
Phase 2- Crack propagation
Phase 3- when the crack reaches critical size the unstable rapid crack growth to fracture components

The modeling of each of these phases has been under intense scrutiny, but the models have not yet been developed in a coordinated way to provide a widely accepted engineering design tool. Different material selection criteria is happen due to fatigue design against crack initiation and structural design from fatigue design against crack propagation. So under this to evaluate the fatigue life Ramber-Osgood theory and Morrow Equation have been used.

1. *Ramberg-Osgood Theory'*

When consider a material that is named as elastic – plastic material is subjected to cyclic loading the strain and stress will initially go through a transient state which same as to a cyclic state. When analyze the behavior of the body in this cyclic state, it can be separated in three alternative regions:
1. Elastic
2. Elastic – plastic and
3. Failure

By linking the tips of stable hysteresis loops for different strain amplitudes of fully reversed strain- controlled, the cyclic stress – strain curve can be possibly obtain. (Tudor SIRETEANU1) When the multi-axial loadings come into play, by using Ramber - Osgood equation, the stable cycle stress – strain curve can be represented in analytical form. (Analytical method for fitting the ramberg osgood model) The Ramber-Osgood equation can be given as follows:

$$\Delta\varepsilon\text{eq} = \frac{\Delta\sigma eq}{E} + 2\left(\frac{\Delta\sigma eq}{2K'}\right)^{\frac{1}{n'}} \text{-----------------------------(1)}$$

$\Delta\varepsilon_{eq}$ – Equivalent of strain for multi-axial loading
$\Delta\sigma_{eq}$ – Equivalent of stress for multi-axial loading
E - Young's Module
K' - Cyclic strength co-efficient
n' - Cyclic hardening exponent

when consider the equation, first part ($\frac{\Delta\sigma eq}{E}$) of the equation represents the linear behavior and the second part ($2\left(\frac{\Delta\sigma eq}{2K'}\right)^{n'}$) represents the non-linear behavior or in other word the plastic part. So when consider the stresses applied on the component, for low stress values the non-linear component is not significant when compared to the linear component.

2. *Morrow Equation*

There two methods used in morrow equation one is stress based approach and other one is strain based approach and strain approach is the vastly used method for life estimation. The σ and ε are estimated to use as the basis of the life prediction in this approach. The analytical method is based on fatigue data of number of low – cycles in terms of the strain–life curve, as they are appropriately used to present the strain cycling resistance of materials by describing the endurance as a function of both elastic and plastic strain amplitude. (GLINKA, 17 February 2011)

$$\frac{\Delta\varepsilon}{2} = \varepsilon_a = \frac{\Delta\varepsilon^e}{2} + \frac{\Delta\varepsilon^p}{2} = \frac{\sigma'_f}{E}(2N_f)^b + \varepsilon'_f(2N_f)^c \text{ --------(2)}$$

In this equation the fatigue life calculation has been done at the zero level mean stress. So it has been modified by adding the mean stress to the elastic part of the equation.

$$\frac{\Delta\varepsilon}{2} = \frac{\sigma'_f - \sigma_m}{E}(2N_f)^b + \varepsilon'_f(2N_f)^c \text{ ----------(3)}$$

This morrow model gives that the mean stress has a significant effect on longer lives, where elastic train is dominant and also it gives it less significant when it's come to shorter lives, where the plastic strain is dominant. The prediction of the morrow mean stress correction model illustrate that mean stress of the component has greater influence at longer lives. Halford and Manson suggested that both terms of the elastic and plastic of the strain-life equation should be improved to account for effect of the mean stress and maintain the independence of the elastic to plastic strain ratio from the mean stress. (GLINKA, 17 February 2011)

$$\frac{\Delta\varepsilon}{2} = \frac{\sigma'_f - \sigma_m}{E}(2N_f)^b + \varepsilon'_f\left(\frac{\sigma'_f - \sigma_m}{\sigma'_f}\right)^{\frac{c}{b}}(2N_f)^c \text{ -------------(4)}$$

Where,

$\Delta\varepsilon$ - equivalent strain
σ'_f – Basquin's fatigue strength coefficient/Fatigue strength coefficient
σ_m - Local mean stress
N_f – Number of fatigue cycles
E – Young's modules / modules of elasticity
ε'_f - Fatigue ductility coefficient
b – Basquin's coefficient / Fatigue strength exponent

c - Fatigue ductility exponent

The modification that had been done by Manson and Halford named as mean stress model, which is tends to give overestimate of life when it's come to short lives, where the plastic strain dominates.
These are the equations that will be used in this document to calculate the fatigue life of on-condition components.
When consider these equations they are depending on several properties. They are named as fatigue properties and monotonic and cyclic properties. These properties are depending on the type of material has been used on the components. Therefore the material of the component should be taken into consideration.

C. Airframe Components And Material

The word Airframe is referred to an aircraft mechanical structure. It includes both inner supporting structures and outer supporting structures. Those outer and inner airframe structures provide the integrity and strength to the aircraft body. To support the main structural components there are many supporting structural components. E.g. - Brackets, Hinges, Truss etc. some of these parts can be critical components of the aircraft that can endanger the safety of the component. But also there are noncritical airframe supporting structures as well such as a hinge of a cargo door and other hinge applications inside the fuselage will mostly be noncritical component.

When these structural components are made its important to identify what are the specific requirements for that part and what are its operating conditions according to that manufactures will select the suitable material for the components. The choice of material not only depends only on Strength/ weight ratio even though it's a significant factor to consider in aviation. It also concerns following factors,

- Fatigue toughness
- Crack propagation rate
- Notch sensitivity
- Stress corrosion resistance
- Exfoliation corrosion resistance

As an example for such supporting structures in an aircraft the Doublers are used to reduce stress concentrations around splices, cut-outs, doors, windows, access panels, etc., and to serve as tear-stoppers at frames and Longerons.

Typical material applications in Aviation and Aerospace are Aluminium and Aluminium alloys. It is widely used because of various positive advantages it has over other metal materials. Other than Aluminium and its alloys in aircraft construction following metals are also used,

- Copper and copper alloys
- Titanium and Titanium alloys
- Stainless steel
- Inconel
- Monel
- Magnesium and Magnesium alloys
- etc.

The application of Copper and its alloys for structural components are limited because of its great weight. But the characteristics such as high electrical and heat conductivity are very important in electrical components in aircraft. Such components as Bus bars, bonding and lockwire are mostly manufactured from copper materials. Also copper is used with other materials such as Tin and Aluminium to extended anticorrosive properties of components.
Inconel is a nickel-chromium-iron alloy closely resembling stainless steel (corrosion resistant steel, CRES) in appearance. Aircraft exhaust systems use both alloys.
The use of Titanium in Aircraft construction is very large comparing to other engineering materials. Mostly the Titanium is used for the aircraft skin structure, engine shrouds, firewalls, Longerons, frames, fittings, air ducts and also for fasteners. Other than that because of its high temperature properties and corrosion characteristics it is used for making,

- Compressor disks
- Spacer rings
- Compressor blades and veins
- Tubing and liners
- Miscellaneous hardware for turbine engines

Some of the aircraft tubing is done from stainless-steel because of its strength to weight ratio and corrosion properties.

For aircraft manufacturing Magnesium and Magnesium alloys also used widely. Some of today's aircraft require in excess of one half ton of this metal for use in hundreds of vital spots. The wing panel fabrication is usually done by using Magnesium alloys. The aircraft components manufactured with magnesium alloys have saved significant amount of weights of the aircraft. The Nose wheel doors, flaps cover skins, wingtips, engine nacelles, hydraulic fluid tanks are also manufactured with this alloy.
Monel is a leading high strength alloy with the combination of high strength and excellent corrosion resistance. Monel are widely used in gears and chains in retractable landing system designs and for structural parts subjected to corrosion.

A leading aircraft hinge manufacture Alarin aircraft hinge cooperation manufactures hinges for cargo doors, air inlet doors, exhaust system hinges and helicopter hinges. As detailed in their website the applications for hinges manufactured by Alarin range from interior applications to flight critical hinges and everything in between. Alarin also has significant experience manufacturing hinges for military applications including the C-130, C-17, F-4 and F-16 platforms. Alarin manufactures aircraft hinges from many materials including,

- Aluminium,
- Aluminium bronze and similar alloys,
- Stainless steel, steel alloys,
- Titanium and other aerospace metals.

Aircraft latches are manufactured mainly from Aluminum, Composite, and Titanium. The reason for selecting these materials for the latch design because of its high strength and fatigue resistance characteristics.

3. Case Study

As our case study we took a cargo door of boing 787 aircraft. The cargo door is designed to be load bearing components of the aircraft. That mean this cargo is subjected to structural loads from the adjacent aircraft structure as well as internal overpressure. (Boris Cecavac, June 12, 2015) The forces, that acting on the cargo door are dynamic and vary depending on whether the aircraft is landing, taking-off or in mid-air fight. The rear cargo door of the Boeing 787 Dreamliner is manufactured by using a variety of materials like, mostly aluminum, titanium and composite. There are several parts that consist in this cargo door.

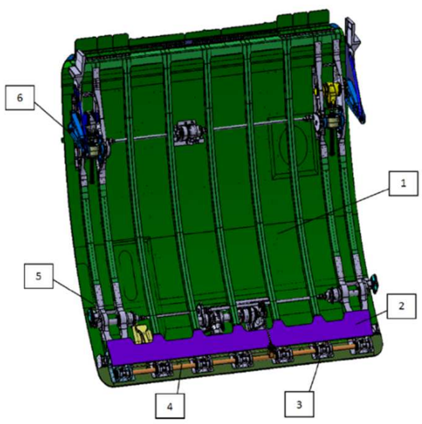

Figure 2 Cargo Door of Aircraft

Annotation	Component name
1	Integrated composite structure
2	Lower strap
3	Latch support(s)
4	Latch torque tube
5	Pull in assembly
6	Lift assembly

Table 1 Component of Cargo Door

From the cargo door configuration, for this case study mainly focus on the latches of the door. Because the latches and the hinges are on-condition components as per the maintenance planning document. Because in the maintenance planning document they have been categorized these component as non-life limited comments. They have given an inspection not a life for the component. There are seven latches available in the cargo door of the aircraft. (Pilatus Time Limit/ Maintence Check) When consider latches, one latch consist with several components. From that we are going to concern about the latch support.

When consider the cargo door there are multiple latch supports in it to distribute the forces evenly and as a safety factor, should any of them fail. This is crucial if an accident would occur in fight. The latch supports hold the latch cam. The latch cam is connected to the torqueing tube that is helps to distribute the torque that is applied when the door is closed (Boris Cecavac, June 12, 2015). There are forces (internal pressure and shear forces) on the door are applied on the upper hinges and the latches, which are located at the lower part of the door.

Titanium fasteners help to keep the connection between the latch support and the composite structure. It is connected with the outer skin, the L-beam and the lower strap. The latch support and the location of its fasteners are designed in such a way that the loads are distributed mainly to the lower strap and the outer skin.

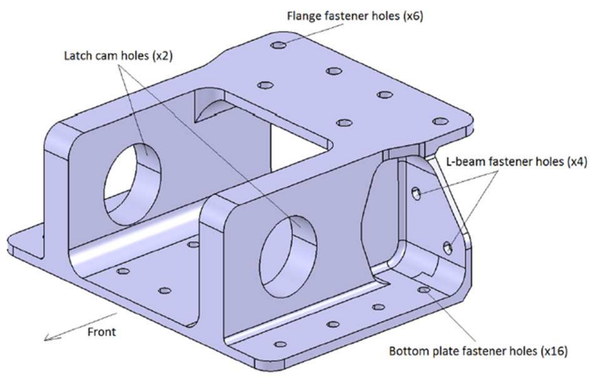

Figure 3 Latch Support Parts

The latch support is made out of Aluminum alloy which is named as 7075-T6. In this alloy Zinc is the main type of alloying material. 7075 aluminum alloy's composition roughly includes 5.6–6.1% zinc, 2.1–2.5% magnesium, 1.2–1.6% copper.

When consider the latch support the stress distribution on the component is not same. The following figure shows the distribution of the stresses action on the latch support. (Boris Cecavac, June 12, 2015)

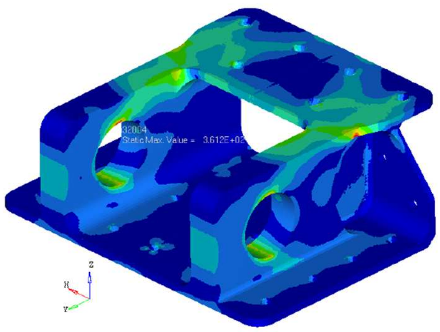

Figure 4 Stress distribution in latch support

100% LL	150% LL	200% LL
361.2	541.8	722.4
321.1	481.7	642.2
281.0	421.6	562.1
241.0	361.4	481.9
200.9	301.3	401.8
160.8	241.2	321.6
120.7	181.1	241.4
80.64	121.0	161.3
40.55	60.83	81.11
47.42	71.14	94.85

Table 2 Magnitude of stresses applied

The highest amount of stress is obtained at the upper part of the cam hole. This is the place which the latch cam is connected. The stress that obtain at there is 361.2MPa.

4. Calculation

When consider these components the manufacturer dose not given time duration to replace the component. These components had been given just only durations to do the inspection and use them until it fails. Therefor these components can be found after it fails. That mean in the next inspection after it fails or sometimes in visual inspections that done routinely. Therefor under this research we are going to calculate the amount of fatigue cycles that the component will take to initiate the fatigue crack.

To do this calculation, the Ramba-Osgood equation and morrow equation had been used. From the Ramba-Osgood equation we are going obtain the equivalent strain for the equivalent stress applied because of the multi-axial loadings. From the results that obtain from this equation can be used in the morrow equation to calculate the amount of the fatigue cycles to initiate the fatigue crack in the component.

In this can calculation we have to use some constants and some data that has been obtain by experimentally. These constants know as monotonic and cyclic properties. These properties had been given as per the material that had been used in the component. The material that used in the latch support is Aluminum alloy 7075-T6. The monotonic and cyclic properties for this material are given below. (Parks, 2004)

	Endurance Limit	Strain-Life Properties			
Material	σ_e MPa ($N_f = 5 \times 10^6$)	σ'_f MPa	b	ϵ'_f	c
Steel					
SAE 1020 (hot rolled)	152	896	-0.12	0.41	-0.51
SAE 1040 (As forged)	173	1540	-0.14	0.61	-0.57
Man-Ten (hot rolled)	262	1089	-0.115	0.86	-0.65
RQC-100 (hot rolled)	403	938	-0.0648	0.66	-0.69
SAE 4340 (Q & T)	492	1655	-0.076	0.73	-0.62
Aluminum					
2024-T351	151	1100	-0.124	0.22	-0.59
2024-T4	175	1015	-0.11	0.21	-0.52
7075-T6	176	1315	-0.126	0.19	-0.52

Table 3 Fatigue properties

	Monotonic Properties							Cyclic Properties		
Material	E GPa	σ_y MPa	σ_{TS} MPa	K MPa	n	σ_f MPa	ϵ_f	σ'_y MPa	K' MPa	n'
Steel										
SAE 1020 (hot rolled)	206	262	441	738	0.19	710	0.96	241	772	0.18
SAE 1040 (As forged)	210	345	621	738	0.22	1050	0.93	386	786	0.18
Man-Ten (hot rolled)	203	322	557	738	0.2	814	1.02	372	786	0.11
RQC-100 (hot rolled)	200	883	931	1172	0.06	1330	1.02	600	1434	0.14
SAE 4340 (Q & T)	200	1172	1241	1579	0.066	1655	0.84	758	1434	0.14
Aluminum										
2024-T351	73	379	469	455	0.032	558	0.28	427	655	0.065
2024-T4	73	303	476	807	0.2	634	0.43	441	655	0.08
7075-T6	71	469	579	827	0.11	745	0.41	524	655	0.19

Table 4 Monotonic and cyclic properties for ductile materials

Obtaining the result for fatigue strain for equivalent stress,

$$\Delta\varepsilon\text{eq} = \frac{\Delta\sigma eq}{E} + 2\left(\frac{\Delta\sigma eq}{2K\prime}\right)^{\frac{1}{n\prime}} \text{-----------------------(5)}$$

Substitute the data to the equation,
The equivalent stress ($\Delta\sigma eq$) has been taken as 361.2MPa. Because that is the highest stress that applied on the latch support.
E = 71.7GPa
K' = 655MPa
n' = 0.19

$$\Delta\varepsilon\text{eq} = \frac{361.2\ MPa}{71.7\ GPa} + 2\left(\frac{361.2MPa}{2*655MPa}\right)^{\frac{1}{0.19}} \text{-----------------(6)}$$

$$\Delta\varepsilon\text{eq} = 7.3082 * 10^{-3} \text{-------------------------------(7)}$$

To calculate the number of cycles the mean stress value should be obtained. The mean stress can be calculated from the data that taken from the table 2.

$$\sigma_m = 185.53MPa \text{------------------------------------(8)}$$

By using above result and other data obtain by the tables, the number of fatigue cycles can be calculated,

$$\frac{\Delta\varepsilon}{2} = \frac{\sigma'_{\mathrm{f}} - \sigma_{\mathrm{m}}}{E}(2\mathrm{N_f})^b + \varepsilon'_{\mathrm{f}}\left(\frac{\sigma'_{\mathrm{f}} - \sigma_{\mathrm{m}}}{\sigma'_{\mathrm{f}}}\right)^{\frac{c}{b}}(2N_{\mathrm{f}})^c \text{-----------(9)}$$

Substitute data to the equation,

$$\frac{7.3082*10^{-3}}{2} = \frac{1315MPa - 185.53MPa}{71.7GPa}(2\mathrm{N_f})^{-0.126} + 0.19\left(\frac{1315MPa - 185.53MPa}{1315MPa}\right)^{\frac{-0.52}{-0.126}}(2N_{\mathrm{f}})^{-0.52} \text{------------------(10)}$$

$$\boldsymbol{N_f = 84014.38} \text{---------------------------(11)}$$

$$\boldsymbol{N_f \approx 84015} \text{-------------------------(12)}$$

This component takes about 84015 fatigue cycles to initiate the crack.

5. *Recommendations*

When consider on-components, to identify its failure we have to do the periodical inspection on that component. When consider the periodical inspection manufacture had been given that time duration between two inspections. This time duration between two inspections is depend upon the component. So the operators have to adhere to the maintenance plan given by the manufacture but maintenance schedule can be change as per the requirement of the operator. There are several factors that the schedule depends on. They are the type of the operation of aircraft, the region of the aircraft operate and etc.

When consider the part it's going to be initiate the failure after about 84015 fatigue cycles. So to identify this failure operators have to carry out the periodical inspection. To identify the failure when it initiate, this amount of flying cycles has to relate with the schedule inspection. As the recommendation, this component's failure initiate at 84015, so to identify this initiation of failure we have to do the inspection after it initiated. So to do that inspection should schedule at 84020, because it give some time to identify it experimentally. But when consider the inspection schedule for this component it's given in flying hours and in calendar years. So flying cycles and flying hours or calendar years should be related to give inspection.

Some component's inspection duration is given flying hours and flying cycles (landings). By using that, there can be build up a relationship between them. So that relationship can be used to evaluate and give the inspection for latches.

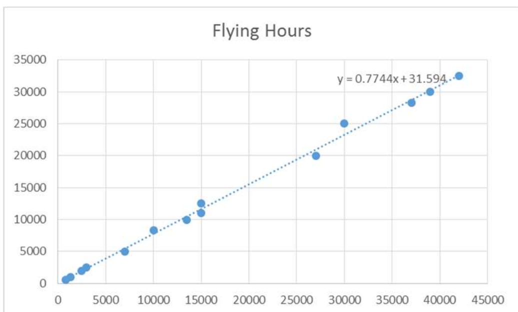

Figure 5 Relationship Between Flying Cycles and Flying Hours

Flying Cycles	Flying Hours
800	600
1300	1000
2500	2000
3000	2500
7000	5000
10000	8300
13500	10000
15000	11000
15000	12500
27000	20000
30000	25000
37000	28300
39000	30000
42000	32500

Table 5 Flying Cycles vs Flying Hous

As per above graph, it can give relationship between flying hours and flying cycles. This can be denoted as in following equation.

y = 0.7744x + 31.594------------------------(13)

Where,

y- Flying hours

x- Flying cycles

From that ground, can build up a relationship between them. When consider the flying cycles which got for the component, there can be given inspection intervals.

For the eddy current inspection for the above component is given as 32,500 flying hours or 42,000 flying cycles and continue in every 12,500 flying hours or 15,000 flying cycles. (Pilatus Time Limit/ Maintence Check). But inspection should do when flying cycles come to 84015. By considering the inspection done as per the flying cycles the crack initiation occurs between 3rd and 4th inspection and it occurs near to 4th inspection. Hence the maintenance schedule should re-arrange according to above calculation for flying cycles to identify the crack.

I. Conclusion

This research is done to find out the time duration that will takes to initiate a crack in an on-condition component. When consider an on-condition component it does not have life limit. That component is used until it fails. So under this research its going be identified before it ultimate failure. It's going to be identified when the failure is initiated. This time has been given in the fatigue cycles. And when consider an aircraft the flying cycles of it is equal to the fatigue cycles. So this time can be obtained by using Ramberg-Osgood equation and Morrow equation. The time can be obtained when supply with the stresses that apply on it and the material is that the component has been made. When those data has been given, by using above equation the time can be calculated. So after obtaining the time duration for a failure the maintenance duration can be updated to identify the crack initiation and do some modifications of any prevention method to increase the time for failure by reducing the crack propagation rate.

Generally the maintenance schedules include Flying Cycles (Landings), Flying hours and calendar years. Therefor if the maintenance is given in flying hours or calendar years, flying cycles should be converted into flying hours or calendar years. Therefor a relationship should be build-up between above parameters as per the requirement

REFERENCES

(1980). *DESIGNING ON-CONDITION TASKS FOR NAVAL - AIRCRAFT.* Navel air systems.

(2015). *Peer review sevice on safe long term operation of Tihange nuclear power plant.* Tihange, Belgium: Internaational atomic energy egency.

Boris Cecavac, D. P. (June 12, 2015). *Structural Analysis of Composite Frame for Aircraft Cargo Door.*

CAAauthority. (2001). *on condition maintenance*. Retrieved from CASA: https://www.casa.gov.au/standard

condition monitoring product and sevices . (n.d.). BMG Technical Resources.

FAA. (2013). *AMT handbooks*.

Gaechter, R. (2002). *Assesment of technology and economic benifits of reciprocating machine condtion monitoring and diagnostic system.* Rhene, Germany: 31st turbomachinery sysmposium.

GLINKA, A. I. (17 February 2011). *A modification of Morrow and Smith–Watson–Topper mean stress.* University of Waterloo, The Department of Mechanical and Mechatronics Engineering, 200 University Avenue West, Waterloo, Ontario,.

Harrison, S. J. (n.d.). *Why Aircraft fail.* Retrieved from ScienceDirect: http://ac.els-cdn.com/S1369702102011380/1-s2.0-S1369702102011380-main.pdf?_tid=bcbca590-1454-11e6-8b0b-00000aab0f27&acdnat=1462626665_94dfe8cee06bf6654df835bb82ef343f

Maab, P. (n.d.). *Corrosion and Corrosion Protection.*

Parks, L. A. (2004). *2.002 MECHANICS and MATERIALS II SPRING* . MASSACHUSETTS INSTITUTE OF TECHNOLOGY DEPARTMENT OF MECHANICAL ENGINEERING CAMBRIDGE, MASSACHUSETTS 02139.

(n.d.). *Pilatus Time Limit/ Maintence Check.*

Shreve, D. H. (n.d.). *Integrated Condition Monitoring Technologies.*

Stevan Maksimovi, P. (. (2005). *Fatigue Life Analysis of Aircraft Structural Components.*

Trey, H., & Schmidt, H.-J. (2001). Programme for life extension and widespread fatigue damage evaluation to ensure continued structural integrety of airbus large transport category airplanes. Manchester: Defence Technical Information Centre.

Tudor SIRETEANU1, A.-M. M. (n.d.). *Analytical method for fitting the ramberg osgood model.* THE PUBLISHING HOUSE PROCEEDINGS OF THE ROMANIAN ACADEMY, Series A.