

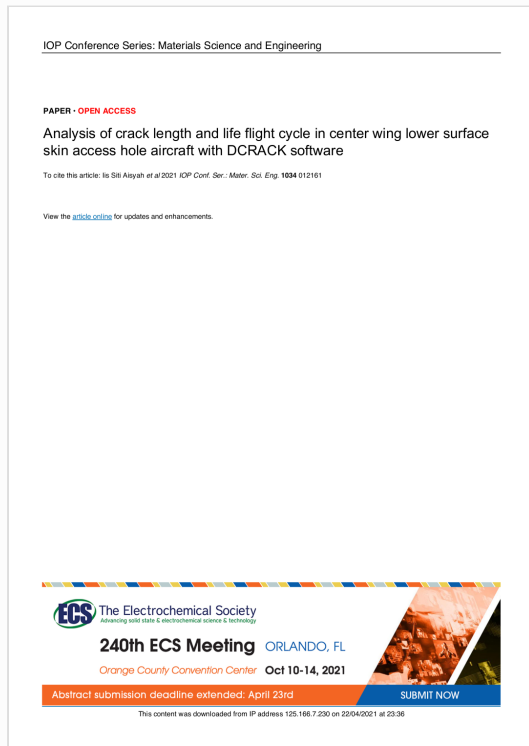


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# Analysis of crack length and life flight cycle in center wing lower surface skin access hole aircraft with DCRACK software

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## Analysis of crack length and life flight cycle in center wing lower surface skin access hole aircraft with DCRACK software

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**Abstract.** One of the problems in the aviation world is that accidents are increasing. The increase in incidents and accidents can be caused by several factors including human error, weather conditions, and failure of the structure / component of the aircraft. Accidents that occur from failure factors, one of which is caused by cracks that occur in the structure due to continuous loading which does not become more attention until the failure. In this final project an analysis of damage tolerance is carried out at the Wing Centre, Lower Surface and Skin Access Hole locations. The choice of location is because aircraft wings are the most important component in aircraft control while operating. Damage tolerance analysis is carried out as an effort to minimize failures in aircraft structures due to cracks that arise. Damage tolerance analysis is done by giving load in the form of an aircraft wing, then an analysis of the growth of crack length that occurs due to loading. The result of damage tolerance analysis will be obtained crack length, aircraft flight life cycle, and critical crack value. And in this study will also bring up a recommendation that is the right time to do the inspection interval.

**Keywords:** Crack Length, Flying Life Cycle, Critical Crack Length, Damage Tolerance Analysis.

### 1. Introduction

The criteria of structural design dictates the speeds, manoeuvres, useful load, and weights of designed aircraft need to be considered for structural design and sizing [1]. Any failure, eventough small one, in any of these components may bring to a catastrophic condition which lead to disaster causing huge loss of lives and property [2]. The level of failure, especially catastrophe one, of a structural part of an airplane can have disastrous consequence, with may cause in shortening of lifecycle of the airplane [3]. Aircraft wing is very crucial for aircraft operation during its flight cycle started from taxiing, taking off, climbing, cruising, descending, approaching, landing, and then taxiing again. It provides the lifting forces while other part such as tail-plane provide directional control of aircraft. To be able to control the flight of airplane, others components such as elevators, ailerons, and rudder are equipped to enable manoeuvring and maintaining stability of the aircraft during it flight by the pilot, while other component, i.e. flaps provide the additional increase of lift during take- off and landing [4]. In addition, the aircraft wing plays a role in the process of receiving



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the forces that occur (thrust, drag, lift, and weight) on the plane. In the process of manoeuvring the wing is very influential and will certainly get a large enough load. The wing will experience highest bending moment when the maximum lift is generated, which occur during the flight phase of aircraft [5]. In this condition, the root of the wing experience highest stress as bending moment will be maximum at this location [5]. Inspection and maintenance of engine components and structures is very important.

In a design process, a step of design must be considered carefully to avoid failure. Some factors that must be considered include environmental factors, working load factors, the type of material used, and so forth. Other conditions that can cause the failure of a structural component are yielding, buckling, fatigue, deflection, creep, resonance, fracture, impact and influence of environmental condition of operation [6]. If material fatigue occurs, it will be deformed or change in shape and size. Per definition, fatigue is related to decrease of load carrying ability of structures when subjected to fluctuating loads [7]. It can be easily detected in the form of propagated cracks [7]. The cracks which initiated in the critical location, when unnoticed, could lead to the catastrophic failure of the aircraft [7]. Crack growth occurs due to cracks that grow little by little until the crack size reaches its critical value and results in the value of the stress intensity factor being the same as the fracture toughness price. Measuring characteristics of crack growth of a material under cyclic stresses are related to its ability to contain a crack and prevent it from rapidly attaining critical length [8].

The structure of an aircraft is designed based on the concept of damage tolerance which is its ability to resist fracture from preexistent damage for a given period of time. It is an essential attribute of components when failure could result in catastrophic loss of life or property [9]. In this research the material used is aluminum 2024-T3. The choice of material is due to the unique material properties for weight, fatigue strength, corrosion resistance, and durability. This process of damage tolerance analysis will obtain the growth of crack length, critical crack length, flight cycle, and recommendations for determining the appropriate inspection interval time.

## **2. Methods and Materials**

### **2.1 PSE Selection & Geometry**

Principal Structural Element (PSE) is a grouping of structural parts that contribute to the flight load, flight, or pressurization load. Principal Structural Element (PSE) will classify structures that are likely indicated experiencing cracks that cause fatigue failure. In the part that is grouped in the Principal Structural Element there needs to be more attention so that the structure can be carried out at inspection intervals before the structure experiences fatigue failure. In determining the Principal Structural Element (PSE) there are several criteria (Criteria of PSE Selection) that need to be considered which consist of typical structure, tension stress level, margin of safety, allowable stress, and inspect ability.

#### **2.1.1 Typical structure**

There are several components - aircraft components that allow fatigue to occur causing the plane to cause disaster (catastrophic). These components consist of:

Wing and Empennage Components, Fuselage component, Landing Gear and Devices, Engine Mounts.

#### **2.1.2 Tension stress level**

In flying position, the aircraft will experience several forces. The aircraft can fly because of the lift (lift) and weight (weight) that occurs on the wings of the aircraft. This force causes the aircraft wings to experience shear stress and bending moment. As a result of that there will be a tensile stress at the bottom of the plane and compressive stress at the top of the plane. Under these conditions there will be a point where the shear stress and bending moment are at the highest and lowest points. The level of stress intensity will be considered to determine the components that will be included in the Principal Structure Element criteria.

### 2.1.3 Margin of safety

This is a parameter used to determine the allowable loading limit for a structure in the hope that the structure will not suffer damage to the specified Margin of Safety limits. If the load that occurs is greater than the margin of safety, the crack that occurs will be faster and failure will occur quickly. Conversely, if the loading is lower than the margin of safety, the crack that will occur will be slower in intensity and failure will be longer.

### 2.1.4 Allowable stress

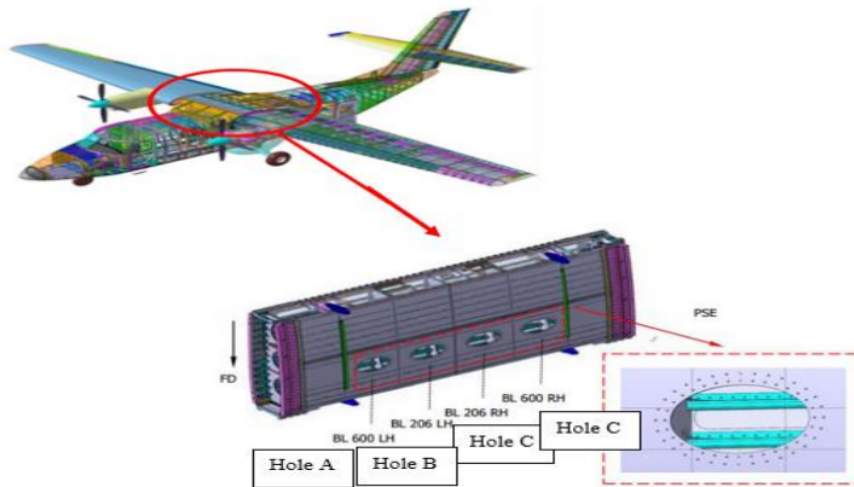
Allowable stress is a parameter used to indicate the maximum amount of stress that is allowed to be applied to a material structure. The applied voltage must be under allowable stress so that an aircraft structure can meet the design service goal.

### 2.1.5 Inspect Ability

Inspect ability is the detection carried out on a structure before the occurrence of fracture failure. The inspect ability step is to do maintenance during the inspection interval on a structure. The inspection interval is obtained from half the age of the structure before it is damaged.

In this experiment the determination of PSE Seletion was chosen on the wing component, namely the center box lower surface skin access hole. This component selection is due to the wing area is an important main structure for aircraft in the control system from taxi, take off, climb, cruise, decent, approach, landing, and taxi. besides that in that area there are 4 inspection holes which will allow high voltage to occur which causes cracking and even failure. PSE (Principal Structure Element) Selection Center Wing Lower Surface Skin Access Hole and geometry as shown in Figure 2.

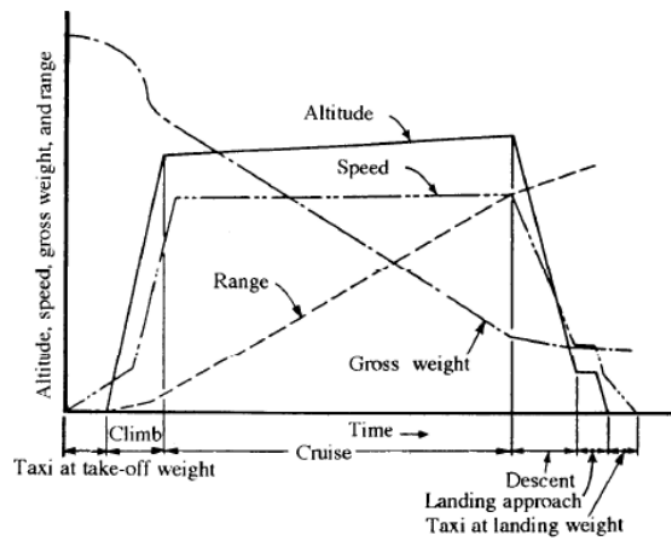
Figure 1 shows the location of the Principal Structure Element (PSE) Selection and its geometry. PSE Selection is a document that selects certain parts that are prone to failure to fly on an aircraft. In this case the part analyzed is the center wing lower surface skin access hole.



**Figure 1.** PSE Selection & Geometry

## 2.2 Load and Stress

Calculation of damage tolerance analysis is very dependent on the voltage spectrum generated from the load analysis that works. The value of the load used is what happens cyclic. The analysis process can be done by means of an assumption or a simplification of calculations. Simplification can be done in one flight cycle, only on the main load amplitude caused by the aircraft operating from the time of taxi, take off, climb1, climb 2, cruise, descent, approach, landing, taxi. The flight profiles of transport aircraft during operation can be seen in Figure 2.



**Figure 2.** Transport Aircraft Flight Profiles [1]

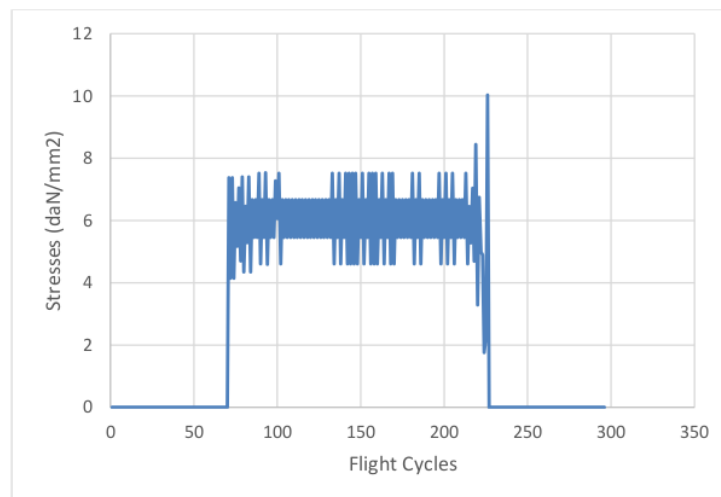


Spectrum development must be based on realistic stresses that occur. Analysis of spectrum development can be done with a particular approach. The initial stress spectrum development was that the load was changed to stress with an analysis approach through the Finite Element Method (FEM) program. There are three main outputs of the voltage spectrum that are very influential on the results of damage tolerance analysis

- a) Number of cycles that work.
- b) The magnitude of the working voltage.
- c) The sequence of the voltage cycle.

The load that is charged on the PSE Center Wing Lower Surface Skin Access Hole includes 4 data, namely short range, medium range, long range, and training range. Stresses that occur on the PSE Center Wing Lower Surface Skin Access Hole are expressed in the form of spectrum. Description of the load spectrum flight by flight, stress by stress [10]. This particular spectrum takes into account of the stress/loading levels during a flight of particular duration [11].

4 data short range, medium range, long range, and training range applied to the center wing lower surface skin access hole. The results of the load spectrum are shown in figure 3. Figure 3 is a load spectrum of 1G load conditions at medium range. The load spectrum will contribute in determining the critical crack area that occurs at the center wing lower surface skin access hole. In addition, the load spectrum graph also contributes in determining the damage tolerance analysis process.



**Figure 3.** Maximum and minimum stresses cycle of the severe flight in PSE Stress Spectrum

### 2.3 Material Properties

Aluminum (Al) alloy structures have found wide applications in defense, aerospace, utensil industries, transport industry, etc [12]. Its unique properties such as high strength to weight ratio, corrosion resistance, fatigue strength and, ductility with high structural efficiency, durability, and workability [12] make it extensively used. Both in military transport and civilian (commercial) aviation aircraft, it counts for about 80% of the structural material, so that the material and its cost become major economic problems [8]. The popularity of Aluminum 2014 (aluminum-copper-magnesium alloy) for aircraft structures makes it widely used since 1920 [8].



**Table 1.** Material Properties of Al 2024-T3

Name	Properties Name	Value
<i>Principal Structural Element W02: Center Wing, Lower Surface, Skin Access Hole</i>	$K_c$	$35,000 \frac{N}{mm} \sqrt{m}$
	$K_{Ic}$	$80,000 \frac{N}{mm} \sqrt{m}$
	Yield Stress	29,00 daN/mm <sup>2</sup>

Type of material used in center wing lower surface skin access hole is AL 2024-T3. This type of aluminum makes use of copper as the primary alloying element and found its primary use in structures of aircraft [7]. The 2024-T3 alloys plates are often used for the structures which need good strength and fatigue performance such as wing tension members, fuselage structures, shear webs and other such structural members [7]. Also, the material found its application in member body which work under tension such as wing and fuselage structures [13]. The recent technique such as electron probe micro-analysis together with structure–property relationships makes us able to conclude that the alloy possesses excellent tensile strength therefore suitable for aerospace applications such as the skin of aircrafts [14]. Table 1 describe material properties of aluminium 2024-T3 alloy.

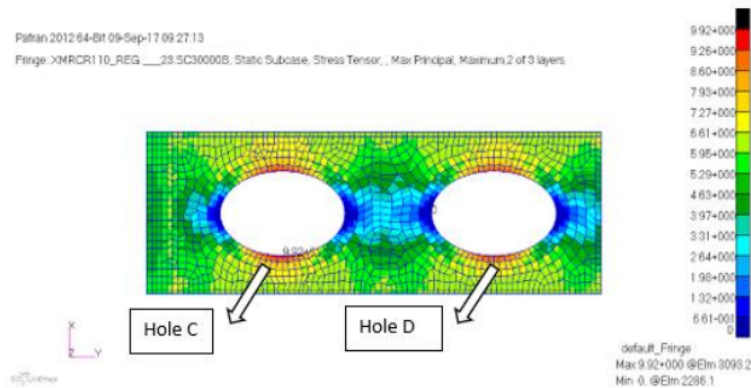
#### 2.4 Finite Element Methods

FEM when viewed from the very basic concept is a system-a-body of a structure which can be divided into elements called “finite elements [15]. Finite Element Method (FEM) or Finite Element Method is a concept used to solve a problem by dividing the object of analysis into finite small parts. The sections are then analyzed and the results will be combined again to get the overall completion of the analyzed area. Dividing the analysis part into small parts is called “discretizing”. These small parts are called elements, which consist of angular points (called nodals, or nodes) and regions of elements formed from these points.

Finite Element Model (FEM) that is by describing the material or structural model tested using a CAD application. Then the described model is analysed using FEM software such as ansys, patran-nastran, etc. After going through the analysis process, it can be seen which areas experience the highest stresses and nominal stresses.

A finite element modelling and analysis approach will be used for simulation of bulkheads and crack stopper straps realistically for its capability to arrest crack in the two-bay strap of the aircraft [2]. In order to analyse internal load and internal sizing of aircraft, separated external load and data was created by adequate solver to activate the finite element analysis [16]. The entire process of the simulation of crack growth by the finite element method proven to be successful and reliable since the simulation values in accordance with the experiment [17].

In this case the results of the analysis on the centre wing lower surface skin access hole with FEM software are shown in Figure 4. Critical crack analysis is performed using NASRAN PATRAN software. Figure 4 shows that critical cracking occurs in medium range conditions when the cruise is loaded 1G. Critical cracks at high stress levels occur in hole C. Factors that cause cracking occur in these parts due to several factors including the influence of the load and high stress concentrations.



**Figure 4.** Maximum and minimum stresses cycle of the severe flight in PSE Stress Spectrum

### 2.5 Damage Tolerance Analysis

Employing damage tolerance approach, means that an initial cracks was assumed to exist in structures caused by defect during manufacturing or handling during maintenance and the fatigue life of the structure in the mode of slow crack growth up to the critical point when crack length will initiate failure because the load applied is beyond the design limit load [18]. In airworthiness, the key factors which need to be followed are to discover all damage that could lead to failure through a rigorous inspection program, and to design structures which resistant to damage by appropriate materials selection and structural design features [9]. Damage tolerance analysis is carried out to predict and type crack propagation from the result of the analysis using the Finite Element Method (FEM) software PATRAN NASTRAN. Prediction of crack growth scenario can be illustrated in figure 5 and figure 6.

In determining the crack propagation scenario, the stress intensity factor must be considered. Stress Intensity is one of the parameters used to perform a crack analysis that describes the stress that occurs at the crack tip. The intensity of stress at the tip of the crack will increase as the crack propagates slowly under a static or cyclic applied stress, or as the applied stress increases at a constant crack length [8]. Stress intensity is often used as a design parameter to determine the nature of the crack process and can be used to determine the remaining life of structures damaged by fatigue. Stress Intensity formula can be seen in equation (1):

$$K = \Delta\sigma \sqrt{\pi a} \quad (1)$$

Where:

$\Delta K$  = Stress Intensity Factor ( $\text{Mpa}\sqrt{\text{m}}$ )

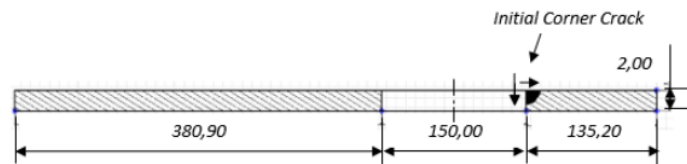
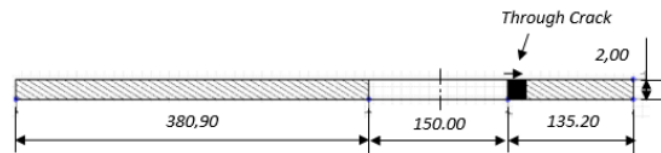
$\Delta\sigma$  = Stress Range (Mpa)

$a$  = Crack Length (m)

The value of K for the above equation is defined for an infinite plate, for other geometric shapes the geometry factor ( $\beta$ ) needs to be added so that it can be written using the equation (2):

$$K = \sigma \sqrt{\pi a} \beta \quad (2)$$

$\beta$  factor is one of the factors that influence the fast or slow rate of crack propagation. To get  $\beta$  obtained from the results of theoretical testing or analysis. With the development of existing technology,  $\beta$  factors can be analyzed using the finite element model program.

**Stage 1***Corner Crack***Figure 5.** Crack Growth Direction Stage 1**Stage 2***Through Crack***Figure 6.** Crack Growth Direction Stage 2

The Stress Intensity Formula in the crack growth direction stage 1 can be seen in the equation (3):

$$K = \Delta\sigma \sqrt{\pi a} \quad (3)$$

Where:

$\Delta K$  = Stress Intensity Factor

$\Delta\sigma$  = Stress Range

$a$  = Crack Length

$\beta_1$  = Corner Crack in a Finite Plate

$\beta_{41}$  = Tabular Look Up in Length  
Direction

$\beta_1$  = Corner Crack in a Finite Plate

Initial Crack Depth, bmm)=1.27 mm

Plate Thickness, t (m) = 2 mm

Plate Width, W (mm)135.200 mm

Bending Load Factor= 0

$\beta_{41}$  = Tabular Look p in Depth

The Stress Intensity Formula in thecrack growth scenario can be seen in the equation (4):

$$\Delta K = \Delta\sigma \sqrt{\pi a} \beta_{41} \quad (4)$$

Where:

$\Delta K$  = Stress Intensity Factor

$\Delta\sigma$  = Stress Range

$a$  = Crack Length

$\beta_{41}$  = Tabular Look Up in Length

### 2.6 DCRACK Software

DCRACK software is one of the programs used to analyze damage tolerance. Points 2 until 6 are the data needed in the damage tolerance analysis. all are included in the DCRACK software for data processing. the results of the run of the DCRACK software in the form of data containing numbers including the number of flights, flight cycles, cracks, Beta factors, and residual strength.

## 3. Result and Discussion

### 3.1 Crack Growth Analysis

Crack growth analysis is done to predict the age of crack propagation that occurs in an aircraft structure. From the analysis of crack propagation, the relationship of crack propagation ( $a$ ) to the cycle that occurs ( $N$ ) will be obtained. The crack growth function is derived from a linear fracture mechanics analysis for each mission that is to be considered in the analysis [19]. This function is used to revise the c-rack length probability distribution function to represent the crack population at a given number of flight hours [19].

Figure 7 shows the results of the crack length obtained for the total flight obtained from the damage tolerance analysis of the centre wing lower surface skin access hole using DCRACK software. At stage 1 occurred during 5188 flights with the last crack growth of 2.00 mm in the long direction ( $a$  direction). At stage 2 it occurred during 11882 flights with the last crack growth of 8,920 mm in the long direction ( $a$  direction). On the residual stress graph the crack data ( $a$ ) critical crack is 2,281 mm. The crack data ( $a$ ) the critical crack is then drawn constant to the Y axis on the crack growth curve until the crack line ( $a$ ) critical crack intersects the crack length curve towards flight. From the intersection of the crack line ( $a$ ) critical crack with the curve it will get ( $N$ ) critical flight. The graph shows that ( $N$ ) critical flight gained 6557 flights. Individual marks identification on the surface of fracture enables a complete fractographic reconstitution of the fatigue crack growth in the main beam of aircraft fracture. The results were summarised as a two-dimensional description of fatigue process which correlated to the number of simulated flying hours [20].

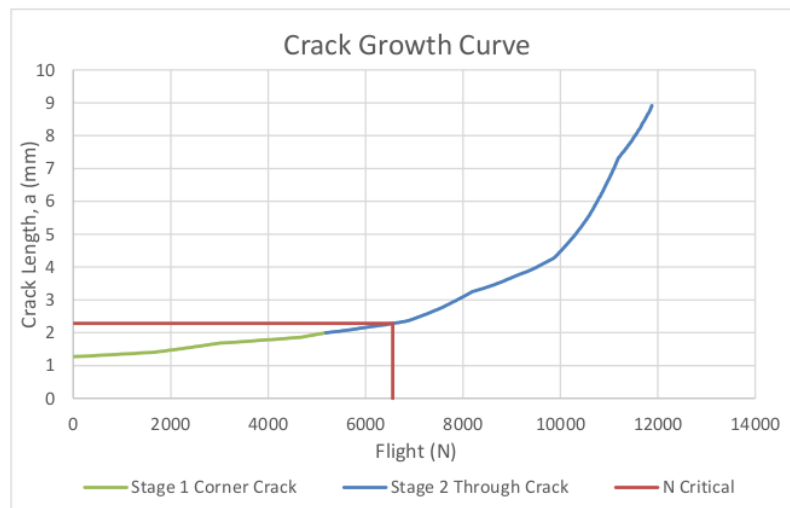
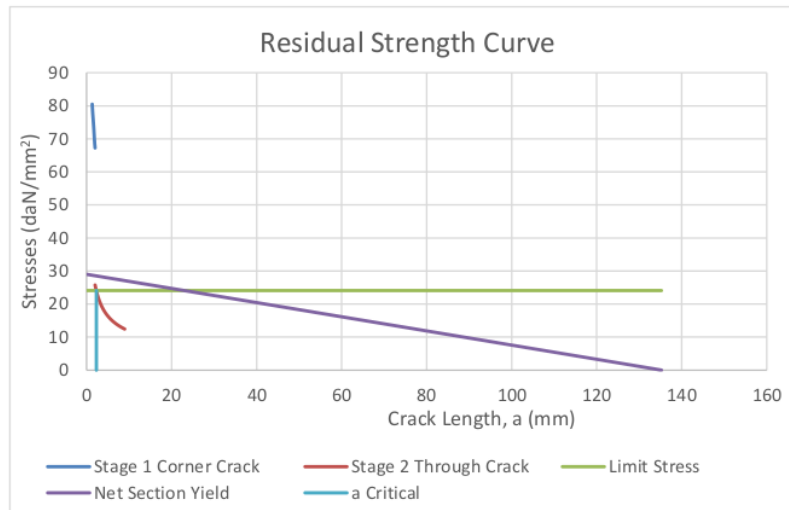


Figure 7. Crack Growth Curve

### 3.2 Residual Strength Analysis

Residual Strength will decrease when the material is no longer able to withstand high loads exceeding  $\sigma_{yield}$ . When the material receives a high load exceeds the yield, the residual strength that is able to work to hold the load is not strong and the residual strength will decrease, causing crack growth to rise or lengthens and cause failure.



**Figure 8.** Residual Strength Curve

The residual strength, in units of pressure, of a component is defined as the strength of the one to endure certain sized crack length [21]. As the crack length grows through the component, the residual strength decreases, although one of the fundamental functions of crack growth is to ensure structural integrity of the component even as the crack is growing [21]. Figure 8 shows the results of the residual stress in the crack length of the damage tolerance analysis process at the center wing lower surface skin access hole.

At stage 1, residual stress at the initial crack length of 1.27 mm residual stress is worth 80.50 daN / mm<sup>2</sup> and crack length at stage 1 ending at 2.00 mm has a residual stress of 67.18 daN / mm<sup>2</sup>. At stage 2, the initial crack length starts from 2.00 mm with a residual stress of 25.75 daN / mm<sup>2</sup> and at the end of the crack length of stage 2 is 8.912 mm with a residual stress of 12.47 daN / mm<sup>2</sup>. From the residual stress graph, data will be obtained (a) critical crack. The value of (a) critical crack length is obtained from the intersection point between the net section yield line or residual strength against the stress limit line. The chosen point is the intersection first. On the line curve that cuts the stress limit first is the residual strength line at a stress value of 24.10 daN / mm<sup>2</sup> of crack length of 2,281 mm. So the value of 2.281 mm is used as (a) critical crack. It was observed that the residual strength of skin decreases when crack length increase [7].

#### 4. Conclusion

In conducting this analysis with DCRACK software there are 5 stages that need attention. First determine PSE Selection and Geometry, second determine Load and Stress, third Material Properties, fourth Finite Element Method, and fifth Damage Tolerance Analysis.

PSE Selection is a document that determines which part will be analyzed. In this study is the center wing lower surface skin access hole. Then after determining the location to be analyzed, it enters the second and third stages, namely the determination of the Load and Stress at work and the Material Properties used in the structure. The Fourth Finite Element Method identifies the location of high stress occurring in the third hole (C).

The fifth stage of Damage Tolerance Analysis where the identification results of the Finite Element Method are analyzed for crack propagation will propagate in such a way by making a damage tolerance analysis scenario. The beginning of the crack is known to have a crack length of 1.27 mm and a crack depth of 1.27 mm. From the data the damage tolerance analysis scenario is carried out in 2 stages. the first stage is corner crack and the second stage is through crack. In doing damage tolerance analysis scenario things that need to be considered are determining the stress



intensity factor formula, because stress intensity factor is very influential in the process of damage tolerance analysis scenario.

After going through 5 stages then the data obtained will be run on the DCRACK software. The results of crack length analysis performed at the center wing lower surface skin access hole locations obtained the final crack length value at stage 1 of 2.00 mm and stage 2 of the crack length of 8.920 mm. The results of the analysis of the life cycle conducted at the location of the center wing lower surface skin access hole on stage 1 lasted for 5188 flight cycles and on stage 2 lasted for 11882 flight cycles. With a critical flight cycle (N critical) 6557 flight cycle. The critical crack length value is obtained from the point of intersection between the net section yield line or residual strength against the limit stress line. The chosen point is the intersection first. The results of the analysis show that the intersecting point is the stress limit line with the residual strength line at the point of stresses 24.10 daN / mm<sup>2</sup> and the crack length is 2,281 mm, so the value of a critical crack length is 2,281 mm.

## 5. Acknowledgement

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## 6. References

- [1]Niu M, 1999 *Aircraft Structures - Airframe Stress Analysis And Sizing* (New York: Adaso/Adastra Engineering Center)
- [2]Peram P, Pamidimukkala K, Student M T, Pradesh A and Pradesh A 2015 *International Journal of Research in Advanced Engineering and Technology* 1 3 1–6
- [3]Zakuan M A M B M, Aabid A and Khan SA 2019 *Int. J. Eng. Adv. Technol.* 9 1 6820–6828, doi: 10.35940/ijeat.A2983.109119.
- [4]Megson T H, 1999 *Aircraft Structures 3rd edition - For Engineering Students* (London: Edward Arnold) pp. 610
- [5]Sriranga B K, 2014 *Int. J. Res. Eng. Technol.*, 3 15 818–822, doi: 10.15623/ijret.2014.0315154.
- [6]Vashisht K S V, 2016 *International Journal of Research in Engineering and Technology (IJRET)* 5 5 510–516
- [7]George S P, 2014 *Int. J. Adv. Eng. Res. Dev.*, 1 11 160–169, doi: 10.21090/ijaerd.011126.
- [8]Niu M C Y, 1988 *Airframe Structure Design* (Hongkong: Conmilit Press Ltd)
- [9]Grandt F 2011 *Procedia Eng.* 17, 236–246, 2011, doi: 10.1016/j.proeng.2011.10.025.
- [10]Mckeighan PC, 2005 *Fatigue Testing and Analysis Under Variable Amplitude Loading Conditions* (West Conshohocken: ASTM)
- [11]Anwar W, Zubair Khan M, Israr A, Mehmood S and N. A. Anjum 2017 *Aerosp. Sci. Technol.*, 64 39–51, doi: 10.1016/j.ast.2017.01.012.
- [12]Arora P K, Srivastava SC and Kumar H, 2017 *Eng. Solid Mech.* 5 3 185–198, doi: 10.5267/j.esm.2017.5.001.
- [13]Diganth Kumar B N and Mahesh Dutt K, 2013 *Int. J. Innov. Res. Sci. Eng. Technol.*, 2 9 5048–5053
- [14]Huda Z, Taib N I and Zaharinie T, 2009 *Mater. Chem. Phys.*, 113 2–3 515–517, doi: 10.1016/j.matchemphys.2008.09.050.
- [15]Venkatesan S P, Beemkumar N, Jayaprabakar J and Kadiresh P N, 2018 *Int. J. Ambient Energy*, 0 0, doi: 10.1080/01430750.2018.1531262.
- [16]Jang J H and Ahn S H, 2019 *Int. J. Aviat. Aeronaut. Aerosp.*, 6 2, 1-6 doi: 10.15394/ijaaa.2019.1301.
- [17]Grbovic A and Rasuo B, 2012 *Eng. Fail. Anal.*, 26 50–64, doi: 10.1016/j.engfailanal.2012.07.003.
- [18]Zhang X and Li Y, 2005 *AIAA J.*, 43 7 1613–1623, doi: 10.2514/1.10275.
- [19]Federal Register, *Widespread Fatigue Damage*, vol. 1, no. December. 1995.
- [20]Siegl J, Nedbal I and Kunz J, 2009 *Int. J. Fatigue* 31 6 1062–1067, doi: 10.1016/j.ijfatigue.2008.05.006.
- [21]Goksel LS, 2013 *Fatigue and Damage Tolerance Assessment of Aircraft Structure under Uncertainty Master Thesis* Georgia Institute of Technology, Atlanta, USA.

# Analysis of crack length and life flight cycle in center wing lower surface skin access hole aircraft with DCRACK software

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