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Fatigue Crack Growth Prediction on Su-30MKM Horizontal Stabilizer Lug Using Static Analysis

Arvinthan Venugopal[1,*], Roslina Mohammad[1], Md Fuad Shah Koslan[2]

[1] Razak Faculty of Technology and Informatics, Universiti Teknologi Malaysia, 54100 Kuala Lumpur, Malaysia
[2] Pusat Perkhidmatan Kejuruteraan Aerospace TUDM (PUSPEKA), Pangkalan Udara Subang, 40000 Shah Alam, Selangor, Malaysia

ABSTRACT

The critical aircraft structure is the load-bearing members are a vital component for any aircraft. The effect of fatigue loading, operating conditions, and environmental degradation has caused the structural integrity of the airframe to be assessed for its airworthiness requirement. Using the fatigue design concept of Safe Life, the RMAF adopts the fatigue life assessment and crack growth prediction to monitoring its critical components' structural integrity. Various methods were used, and for this analysis, the Crack Growth Prediction method was used to determine the crack growth behavior and its ultimate failure point in case of any crack occurrences. The horizontal stabilizer lug was chosen since it has the highest possibility of fatigue failure. The analytical methods discussed are Crack Growth Analysis and Low Cycle Fatigue. For the numerical method, Nastran was used to simulate crack growth. The result from the crack growth analysis was validated with the numerical result. The conclusion is that, based on the fatigue life cycle, the structure condition is not affected by severe damage, and its failure is approximately around 1 million cycles, and the crack growth location on the bottom of the lug is the critical location. The research outcome will be on the extension of the structure life of the lug.

Keywords:
Sukhoi Su-30MKM, horizontal stabilizer lug, crack growth analysis, fatigue

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1. Introduction

Structural integrity, by definition, is the condition when a structure is sound and unimpaired in providing the desired level of structural safety, performance, durability, and supportability [1]. The integrity of structure effect in terms of thermal, environmental, and mechanical deterioration of materials and structures fabricated using the selected manufacturing processes and joining methods need to be identified with acceptable quality, and cost-effective preventive and in-service repair methods are either available or can be developed on time.

The Sukhoi Su-30MKM aircraft horizontal stabilizer comprises spars, stringers, ribs, and skin. The structural elements of aircraft wing spars and stringers experience cyclic loading during the flight cycle from take-off to landing [2]. The loading creates stress concentration at the counter-sunk rivets from crack nucleation sites. When the stress intensity factor exceeds the threshold value, corrosion pits become fatigue crack growth. Fatigue is based upon repetitive loading below the yield stress, leading to premature failure [3] [4].

* *Corresponding author.*
E-mail address: arvinthanvenugopal@yahoo.com

In the early days of aircraft fatigue studies, the safe life approach was used to ensure the integrity of airframe structures from fatigue failure. In this approach, the mean fatigue life of the structure was estimated and then divided by a scatter factor to give a safe operating life (safe life). The safe life must ensure that the cumulative probability of failure is less than 1 in 1000 over the safe operation life [5] [6]. The structural integrity of the stabilizer lug is determined by its fatigue life and crack growth analysis. This approach allowed the structure to remain in service until a planned inspection procedure detected fatigue cracks before they reached a dangerous extent and that the structure must have sufficient residual strength to provide safety until cracks are detected from routine inspections [7] [8].

Fracture mechanics-based fatigue approaches may be used to model fatigue crack propagation from an initial size to the final dimensions responsible for the fracture of the component. This approach may be used to complement the local strain-life approaches, modeling the crack propagation from the initial crack to the critical dimensions leading to component collapse [9] [10] [11]. Fracture mechanics-based fatigue approaches may be further extended to simulate the global fatigue life of components, using the Equivalent Initial Flaw Size (EIFS) concept based on cyclic J-integral. The material is assumed to include defects acting like initial flaws. Therefore, the component's fatigue life is assumed as the number of cycles required to propagate these initial defects until critical dimensions [12] [13].

2. Methodology

2.1 Fatigue Analysis

The horizontal stabilizer includes two panels with each panel rotated around the pivot rigidly attached to two points of the fuselage frame No.45. For Finite Element Analysis (FEA) and fatigue analysis, the horizontal stabilizer lug structure has been chosen among the critical locations due to its function as an attachment between the horizontal stabilizer and the rear fuselage [14] [15]. Figure 1 shows the location of the horizontal stabilizer lug.

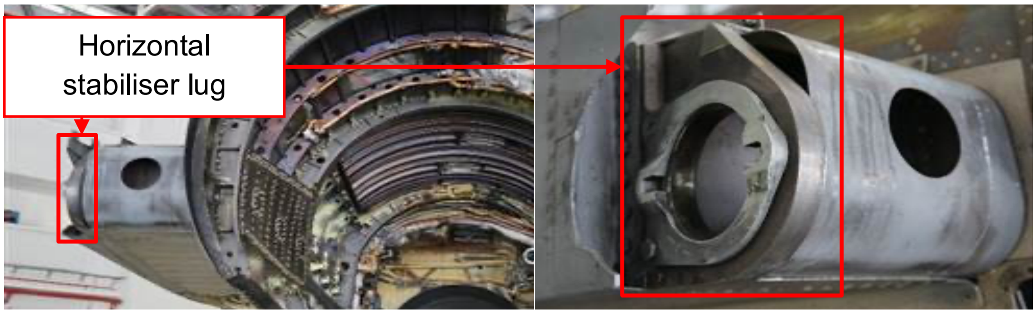

Fig. 1. Location of the horizontal stabilizer lug

The fatigue life calculation of Su-30MKM was performed on the horizontal stabilizer lug joint structure. The structure is made by using the material Al7075-T6 (Aluminium alloy). Fatigue life calculation used in Unigraphics (NX) requires the spectrum loading (g-loading), Material properties, fatigue characteristics, Stress Amplitude vs Number of Cycles to Failure graph or Total Strain vs Number of Cycles graph formulation, and the component to be analyzed. The component should have actual physical geometry, which has been analyzed to obtain its displacement, stress, and strain result. This is usually performed using static linear analysis, but other solutions can be used if needed.

Ten years of history (2008 – 2017) of flights obtained from the ARM-TSV have been used to represent future usage analysis [15]. A safe life approach or crack initiation method was assumed, which uses E-N (Strain life) formulation. Furthermore, the Low Cycle Fatigue (LCF) approach was adopted because the Loading profile is the g-acceleration data.

2.2 Crack Growth Analysis

Pre-processing of Crack Growth Analysis starts by obtaining the Computer-Aided Design model, and mesh models from Figure 2 show the obtained Computer-Aided Design model of the horizontal stabilizer lug, while figure 3 shows the mesh on the computer-Aided Design model.

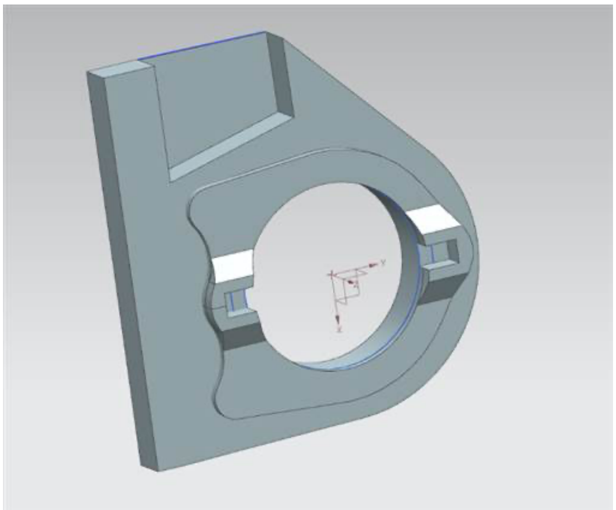

Fig. 2. Obtained horizontal stabiliser lug model

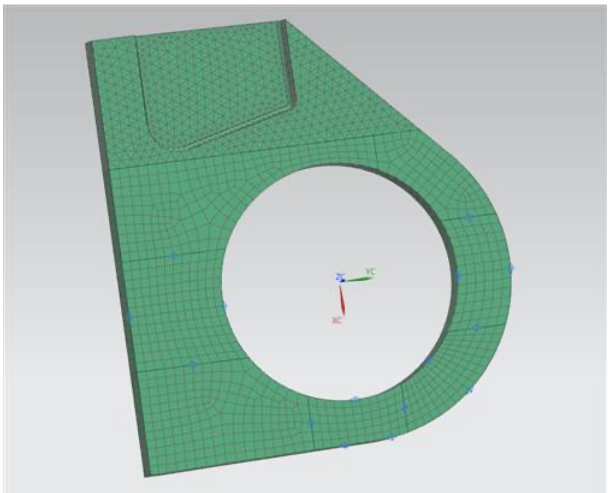

Fig. 3. Obtained horizontal stabiliser lug mesh model.

Material identification and additional material properties were obtained from the Metallic Material Properties Development and Standardization (MMPDS) Handbook. The material identified by Science and Technology Research Institute for Defence (STRIDE) for the horizontal stabilizer lug is the AL7075 aluminium alloy. Basic material properties are shown in Table 1.

Paris Law Theory will be used as the proven mathematical solution for crack growth analysis with a basic equation of:

$$\frac{da}{dn} = CK^N \quad (1)$$

The variables defined in the equation are C and N for material coefficients, $\frac{da}{dn}$ represents the rate of crack length change to cycle, and K will signify the material intensity factor. C and N material coefficients for AL7075 are shown in Table 2.

Table 1
Material Properties of Al7075

No	Property	Value
1.	Young's Modulus	71.016 Giga Pascal
2.	Poisson Ratio	0.33
3.	Yield Strength	413.69 Mega Pascal
4.	Ultimate Tensile Strength	468.84 Mega Pascal

Table 2
Crack Material Properties

No	Property	Value
1.	*C*	1.8×10^{-13} MPa $m^{0.5}$
2.	*N*	3.6

The historical data for the Air Combat Manoeuvre sortie is used for the fatigue input as a multiplier to simulate multi-gravity movements on the Finite Element Analysis strain gauge load placed on the horizontal stabilizer lug was obtained from ARM-TSV.

A mapped force area was obtained from the Finite Element Analysis. Static analysis using the strain gauge obtained data was used as an initial load input of the structure. Figure 4 shows the mapped force and boundary conditions used as a load input for the fatigue Crack Growth Analysis. Table 3 shows the input value.

Table 1
Loading Value Extracted from Finite Element Analysis

No	Type of Loading	Load Value (N)		
		Fx	Fy	Fz
1.	Load from FEA global model and FEA static analysis strain gauge data.	750	-502.85	2200

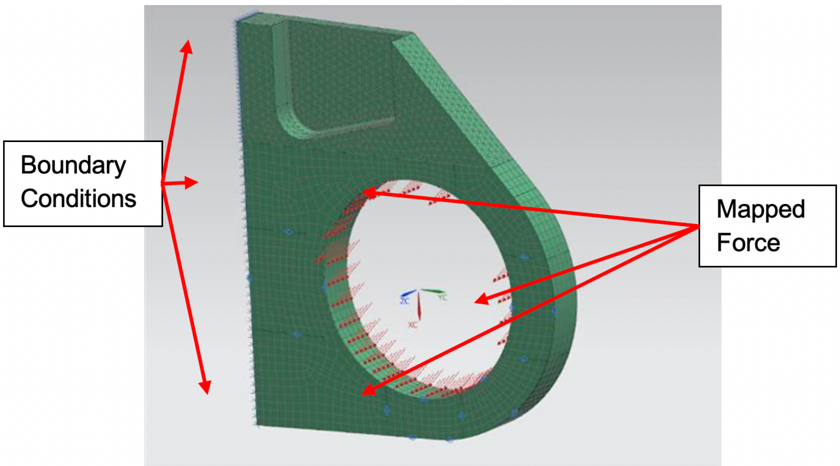

Fig. 4. Mapped loading on the horizontal stabilizer lug structure part model

Finite Element Analysis result data was then transferred into the fracture mechanic software ZENCRACK version 6.1. The parameters that are needed for a comprehensive ZENCRACK model (Figure 5) will be:

a. The completed Computer Aided Design model that needs to be transferred must not have any required clean-up or correction.

b. Assigned mesh on the desired structure model. For the area assigned for the crack model, Hexadecimal Elements must be used in the model. This is required for the crack block system to be implemented.

c. All boundary conditions and loading system has been assigned to the model.

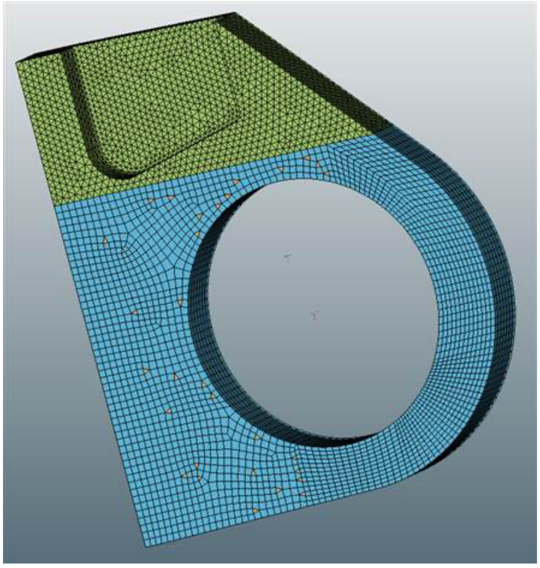

Fig. 5. ZENCRACK horizontal stabiliser lug structure part model

2.3 Crack Modelling

Two cracks were initiated on the horizontal stabilizer lug. Figures 6 (bottom crack) and 7 (top crack) show the two different crack locations modeled on the horizontal stabilizer lug. The simulated cracks will go through the Ny history data of the sortie Air Combat Maneouver.

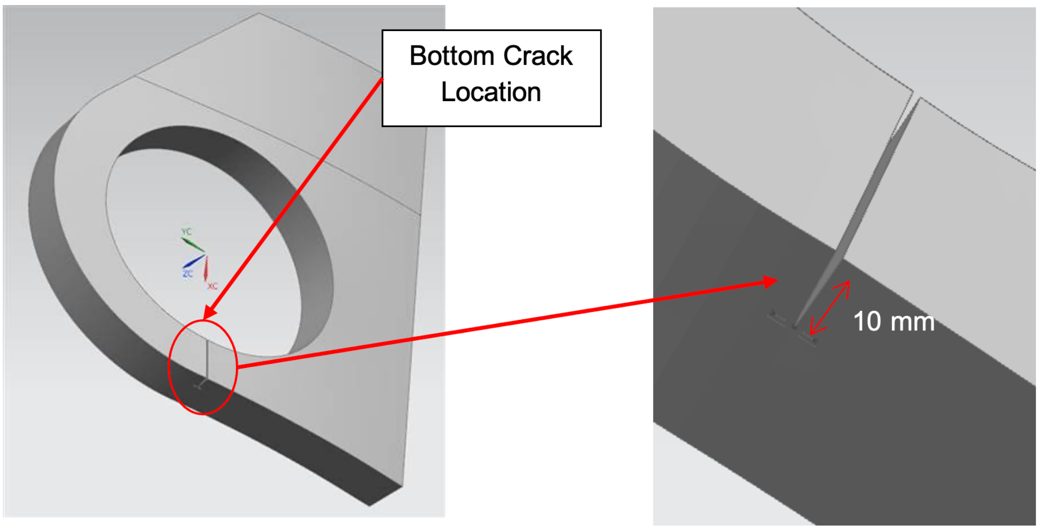

Fig. 6. Bottom crack location on the horizontal stabilizer lug structure part model

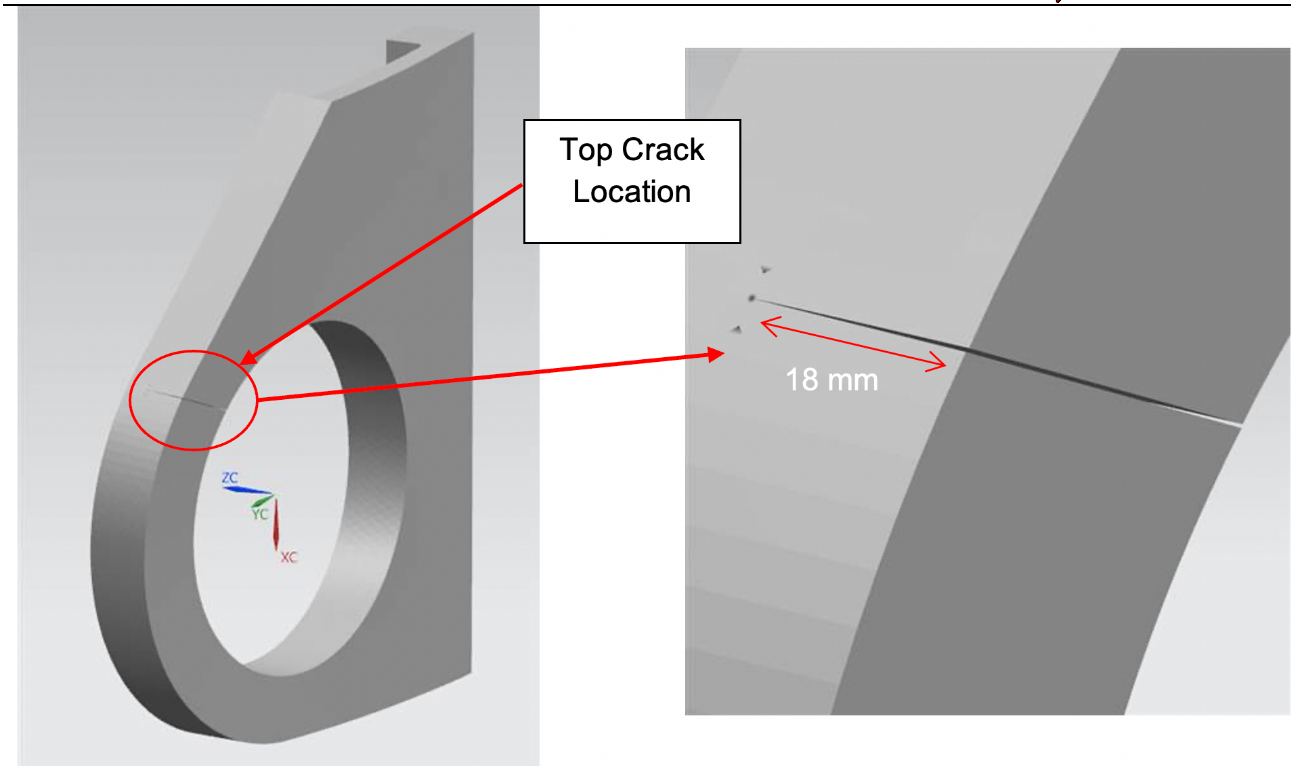

Fig. 7. Top crack location on the horizontal stabilizer lug structure part model

CGA was then conducted using ZENCRACK on the transferred model to obtain crack behaviour, critical crack length, and cracked part lifespan. Figure 8 shows the required procedures for CGA.

2.4 Fatigue Analysis

CAD clean-up processes focus on removing unnecessary lines/surfaces/solids of the 3D scan model. Figure 8 shows the methodology adopted in FEA analysis. The original 3D CAD model was produced from the 3D scanning process. The scanned 3D CAD model went through a CAD clean-up process before it could mesh for FEA. Besides that, it also indicates the difference between CAD model before the clean-up process (on the left) and CAD model after the clean-up process (on the right). In the end, only the existing section will remain for other FEA processes.

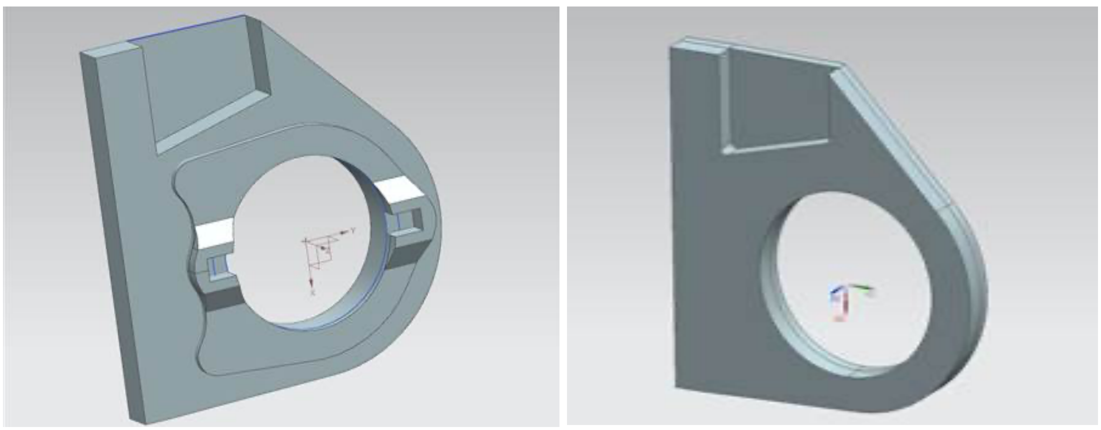

Fig. 8. Scanned horizontal stabilizer lug (left) and after clean-up process (right)

The part meshed with 3D Hexadecimal mesh with a mesh size of 3mm. The quality of the mesh was checked. If the mesh's quality does not meet the requirement set, it will go through the CAD clean-up process and re-mesh until its quality meets the requirement. Figure 9 shows the mesh model (on the left).

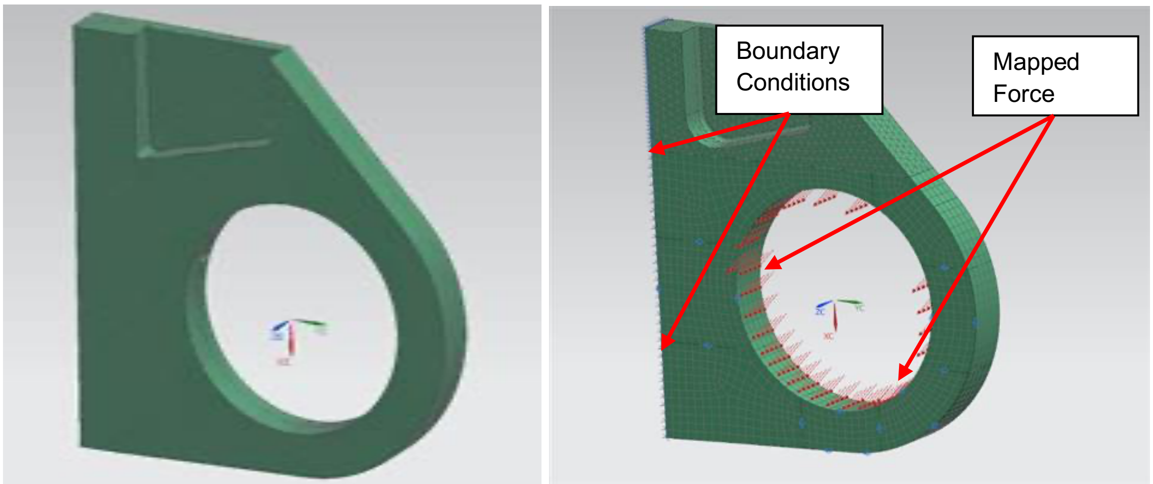

Fig. 9. Mesh model (left) and mapped loading on the horizontal stabilizer lug model (right)

The load in this analysis is extracted from FEA analysis on Su-30 M.K.M. global model. This global model was developed during the Sukhoi PRW program. Table 4 shows loading data extracted from the global model in X, Y, and Z directions. Figure 9 (on the right) shows the mapped force and boundary conditions that were used in the analysis.

Table 4
Loading applied in the analysis

No	Axis	Force Value
1.	X	1481.3 N
2.	Y	-502.85 N
3.	Z	2200 N

Fatigue analysis requires five types of input. First, it requires geometry input, also known as the CAD model, which will be meshed to produce a finite element model (FEA model). Second, it requires stress data from finite element analysis (FEA). The first and second inputs are already created from previous FEA analyses. Therefore, the previous FEA model and stress data were used for fatigue analysis.

There are two types of fatigue analysis. First is the High Cycle Fatigue (HCF) analysis, which utilizes the stress life method or S-N method. The second type of fatigue analysis is Low Cycle Fatigue (LCF) analysis, which utilizes the strain life or E-N method. The third input is defining fatigue analysis parameters, also known as fatigue criterion.

The fourth input is fatigue material data, and the fifth input is the loading profile. The loading profile is the g-acceleration data. This data was obtained from FDR (Flight Data Recorder) and compiled manually for each year. Original data was per flight mission and typically lasted up to 3 hours.

The method of processing the g-acceleration data was using the Rainflow Counting Algorithm, performed automatically by NX. The missions were compiled for each year (from the year 2008 until 2019). The missions had been added one after another according to the date the mission was flown.

Material data needed for FEA analysis are mass density, modulus of elasticity (Young's Modulus), Poisson's ratio, yield strength, and tensile strength. The material used in this analysis was identified as V95 aluminium alloy, whose western equivalence is AL7075. Table 5 shows mechanical material data for AL7075.

Table 5
Material data for AL7075

CHARACTERISTIC	VALUE
Mass density (kg/m3)	2850
Modulus of Elasticity (N/mm2)	71016
Poisson's Ratio	0.33
Yield Strength (N/mm2)	413.68
Tensile Strength (N/mm)	468.84

For fatigue analysis, material data needed for fatigue analysis are fatigue strength coefficient, fatigue strength exponent, fatigue ductility coefficient, fatigue ductility exponent, cyclic strength coefficient, and cyclic strain hardening exponent. Table 6 shows material fatigue data for AL7075 aluminium alloy.

Table 6
Fatigue material data for AL7075

Stress Life (S-N curve)	
Fatigue Strength Coefficient (MPa)	886
Fatigue Strength Exponent	-0.076
Strain Life (E-N curve)	
Fatigue Ductility Coefficient	0.446
Fatigue Ductility Exponent	-0.759
Cyclic Value	
Cyclic Strength Coefficient (MPa)	913
Cyclic Strain Hardening Exponent	0.088

3. Results and Discussions
3.1 Critical Crack Length

The figure will represent the data for the SHM history profile. Figure 10 shows the dn vs da (mm) graph, individually showing the top crack and bottom crack data. Figures 11 represent the graph *dn* vs *ki* (MPa√ mm) with comparable configurations. The crack behavior for the bottom crack is shown in Figure 12, while the top crack behavior is seen as a retardation type of crack without a significant behavior.

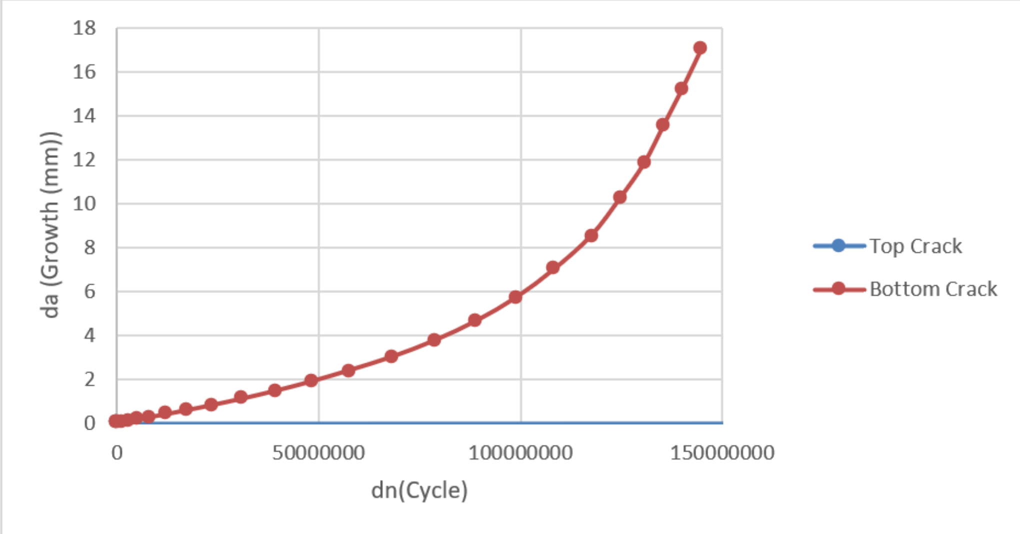

Fig. 10. *dn* vs *da* for top and bottom crack

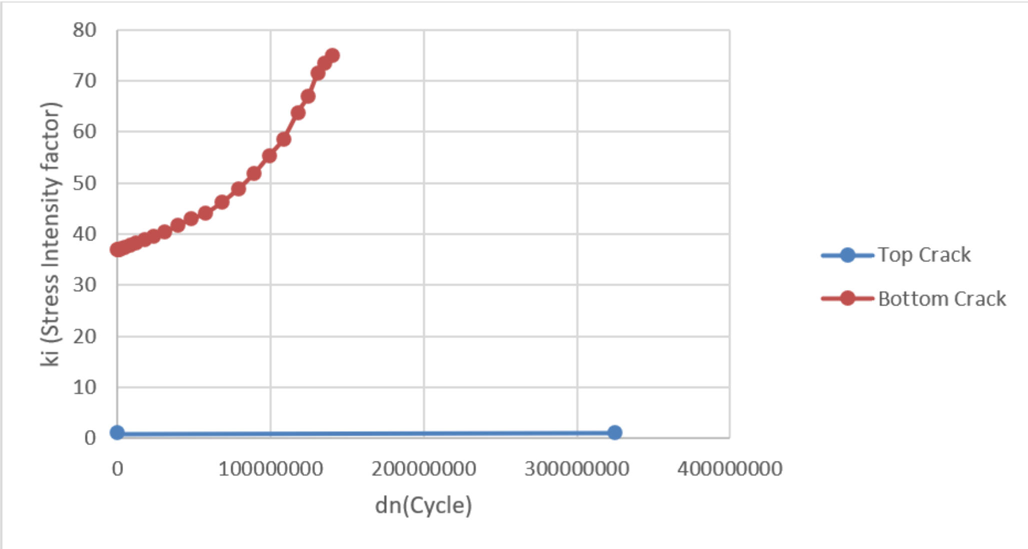

Fig. 11. *dn* vs k_i for top and bottom crack

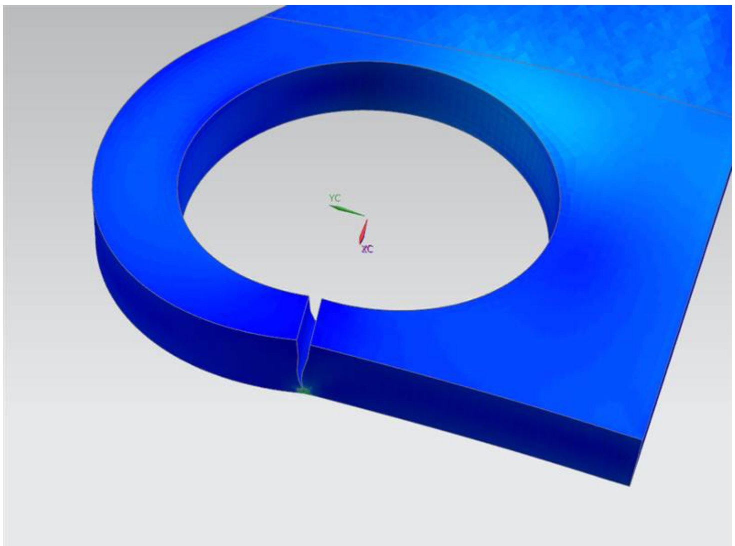

Fig. 12. Bottom Crack Behavior

Table 7 shows the result value for each initiated found crack model based on the SHM history profile as the fatigue multiplier.

Table 7
Summary of Result

No	Crack Type	Properties	Value
1.	Top crack	Critical crack length	Crack retardation.
		Crack part full cycle.	1325009229 cycles of A.C.M. sortie.
2.	Bar Crack	Critical crack length	17.01 mm
		Crack part lifespan.	108294671 cycles of A.C.M. sortie.

Due to the design structure and the loading applied on the horizontal stabiliser lug, the crack growth for the bottom crack will reach its critical stage while the top-initiated crack shows signs of a crack retardation occurrence. Thus, the final crack growth location on the bottom of the horizontal stabiliser lug structure part has been selected as a critical crack length. The critical crack length point will be translated as the limiting value for the crack part lifespan.

3.2 Fatigue Damage

Figure 13 shows the displacement result of the horizontal stabilizer lug. It shows that the maximum displacement is 0.287mm. Figure 14 shows the stress result. It shows that the maximum stress is 26.62 MPa and occurs at the part's top section.

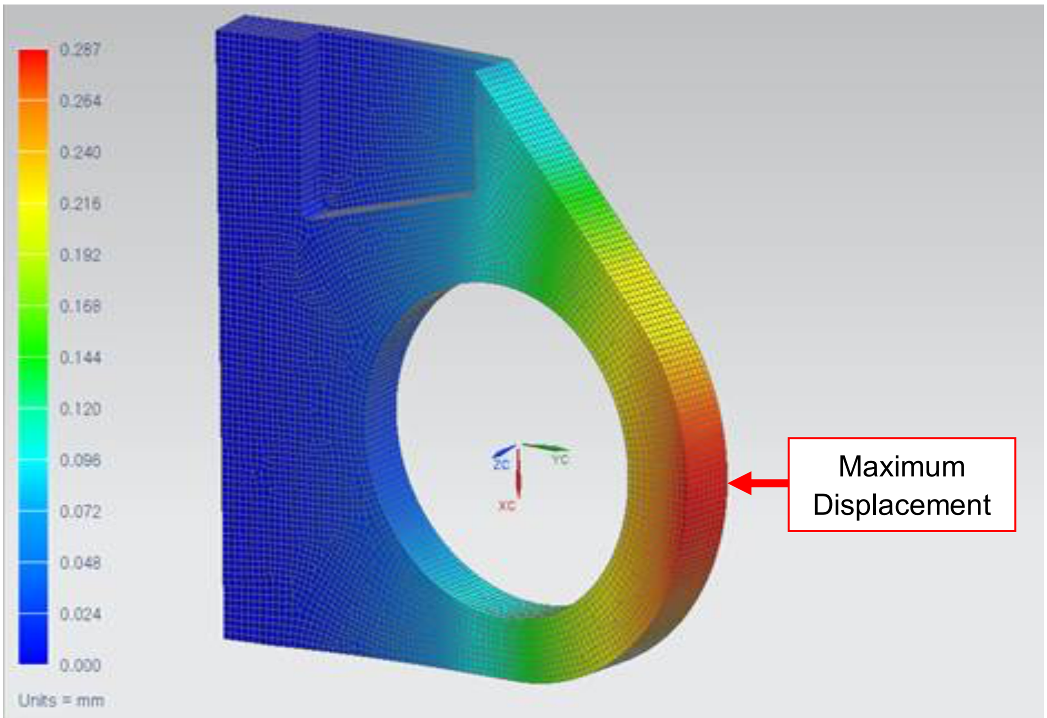

Fig. 13. Displacement result of the analysis part

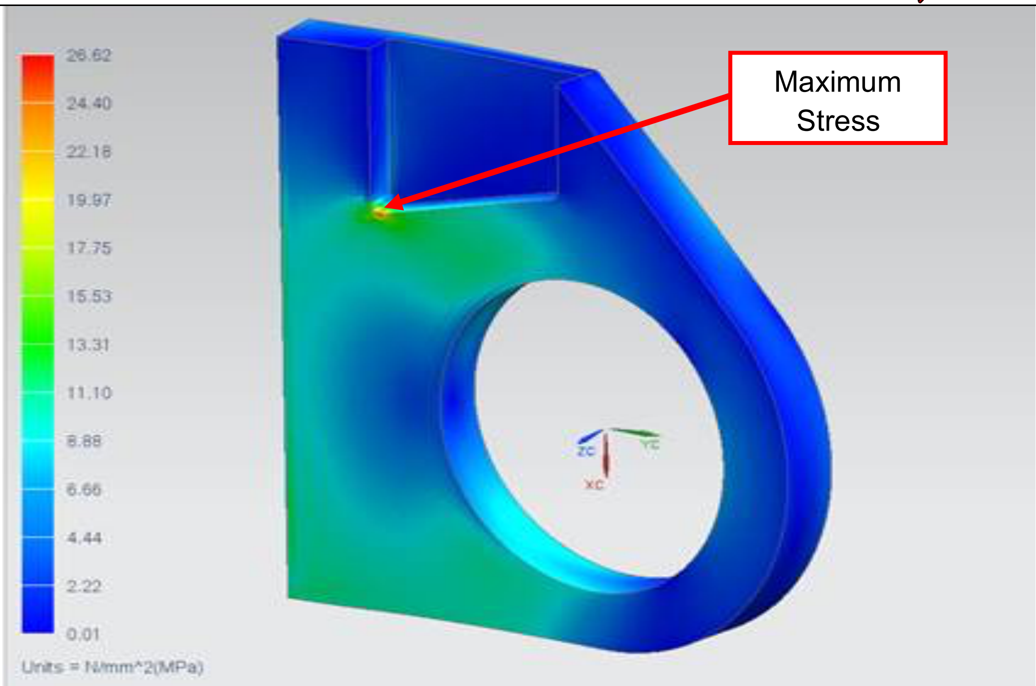

Fig. 14. Stress result of the analysis part

The stress value above is the stress experienced by the part when the flight is steady and level. The stress value must be multiplied by the g value for high-speed maneuvers. From 2008 to 2019, the maximum g experienced by the aircraft was 8.85g in 2015. During that 8.85g maneuvers, the maximum stress experienced by the part is 235.587 MPa. Therefore, during that 8.85g manoeuvres, the maximum stress is 57% of its yield stress. The yield stress for AL7075 is 413.68 MPa.

The loading profile was manually entered into NX from 2008 until 2019. This loading data was obtained from the aircraft and accumulated from its first day until the last day it operated in 2019. Figure 15 shows an example of a graph for the loading profile.

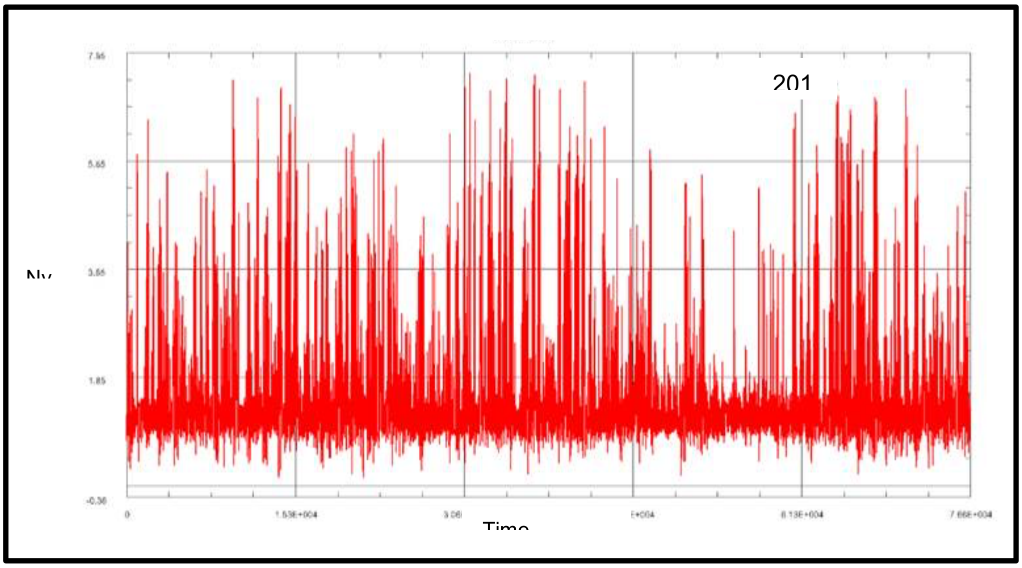

Fig. 15. Loading profile for 2019 input into NX

Fatigue analysis for the horizontal stabilizer lug used the S-N method as this is Low Cycle Fatigue (LCF) analysis. The maximum stress at 8.85 g is only 57% of its yield strength, far from the material yield point. Therefore, the S-N method was used as it was a more suitable method for fatigue analysis. The fatigue analysis result was divided into fatigue damage and life cycle.

For fatigue damage, total fatigue from 2008 until 2019 is 1.069 x 10^{-11}. Table 8 shows fatigue damage results per year. Fatigue damage value shows the accumulative damage in the structure due to the application of cyclic loadings for all the years combined. The structure will fail due to fatigue when the fatigue damage value equals one. Figure 16 shows the fatigue damage result.

Table 8
Damage by Year

Year	Damage	Year	Damage
2008	4.271e-14	**2014**	-
2009	9.555e-13	**2015**	5.616 x 10^{-12}
2010	1.837e-14	**2016**	-
2011	8.574e-13	**2017**	2.005 x 10^{-13}
2012	1.771e-13	**2018**	-
2013	2.821e-12	**2019**	5.027 x 10^{-13}
		Total	**1.069 x 10^{-11}**

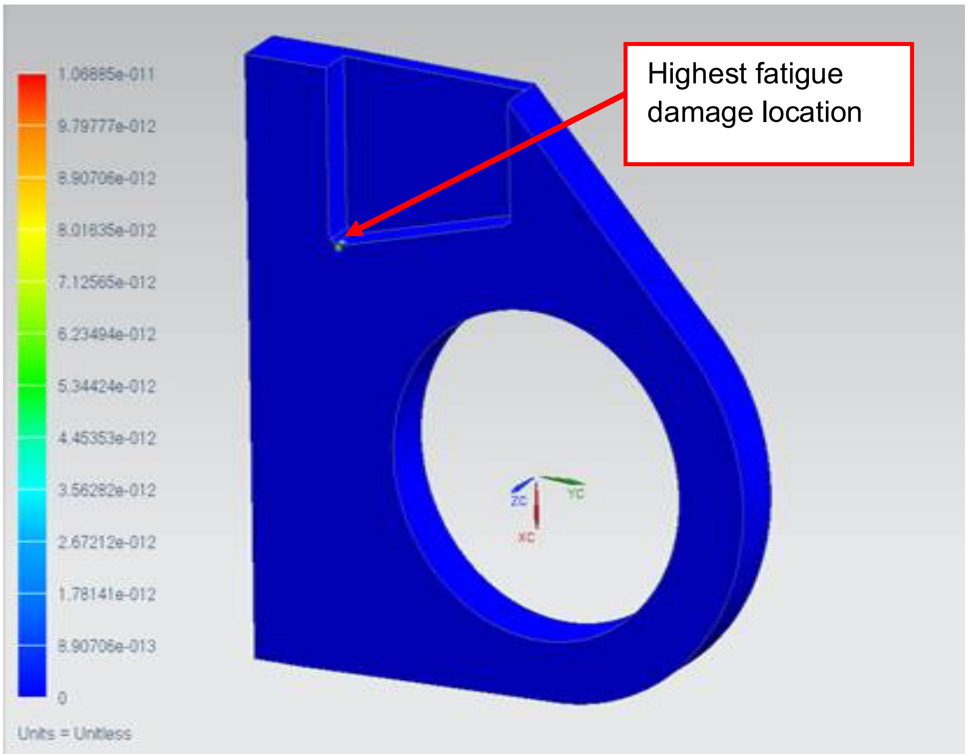

Fig. 16. Fatigue Damage Result

In terms of the fatigue life cycle, the minimum cycle to failure is 9.356x10^{10} cycles. The fatigue life cycle is the value that predicts how many times the structure can withstand the same loading profile and stress history. In this fatigue analysis, one cycle was equal to the loading profile from 2008 until 2019. Figure 17 shows the fatigue life cycle result.

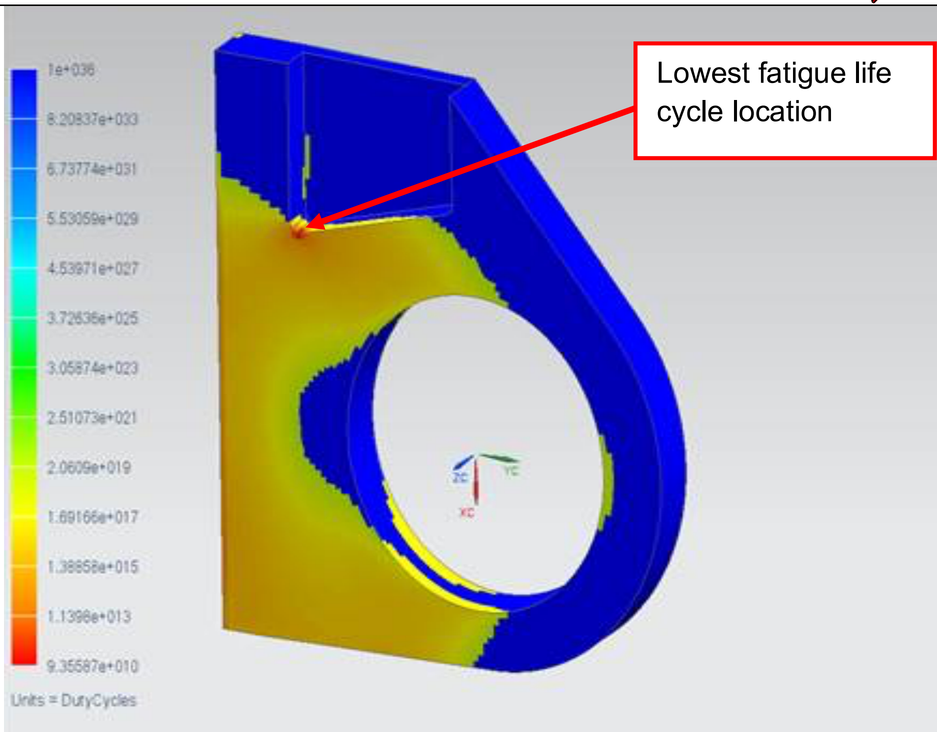

Figure 17. Fatigue Life Cycle Result

4. Conclusion

This study has identified the critical crack length, crack behaviour, and the crack part lifespan of the horizontal stabilizer lug. Using the mapped loadings, a crack propagation analysis was subsequently carried out. In conclusion, based on the FEA, the highest stress result is 26.62 MPa. Besides that, the aircraft experienced a maximum g of 8.85g. The equivalent maximum stress at 8.85g was 235.587 MPa. This value is around 56.95% of its yield stress. From a fatigue perspective, the part is well designed for a fatigue scenario with cycle to failure above the standard $1x10^6$ (1 million) cycle, and damage is well below 1%. It is also recommended to install a strain gauge for continuous monitoring of the structure's health.

Acknowledgement

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