

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

Safe Life Estimation of an Aircraft Fuselage with Window Cut Outs

Prof. Avil Allwyn Dsa¹, Helga V Lobo²

Assistant Professor, Department of Mechanical Engineering, Don Bosco College Of Engineering, Fatorda, Goa, India¹

P.G. Student, Department of Computer Science Engineering, Centre for P.G Studies, VTU, Belagavi, India²

ABSTRACT: The estimation of safe life is vital in aircraft structure to use it safely and economically for the maximum duration, while keeping the weight minimum. This paper describes the estimation of safe life of an airplane fuselage with a passenger window cut-out region. This is the location where failure is likely to initialize due to very high stress concentration. The fatigue life of the fuselage structure under a given sample loading block is studied using Palmgren-Miners rule. Cracks become critical and cause high velocity propagating fracture when the Stress intensity factor reaches its critical value. The proposed method uses FEM approach for linear static stress analysis to locate the points of high stress concentration. The stiffened panel is analysed for different crack lengths in these regions of high stress concentration. Crack growth rate is evaluated for different crack lengths. The safe life is estimated by combining the fatigue life and time taken by crack to grow critical after the crack initiation.

KEYWORDS: Safe Life Estimation, aircraft fuselage

I. INTRODUCTION

Window cut-outs in fuselage serve to reduce weight and relieve stresses, if any, formed inside the aircraft structure. But discontinuities also serve as locations of high stress concentration where failure occurs under fatigue loading. Estimating this life span becomes important as this will allow the maintenance personal to inspect the structure to critical damage. A small crack in the structure caused by local stresses is not an enough reason to ground the aircraft. Hence an estimation of the time for the crack to grow to its critical length predicts the airworthiness of the aircraft. Keeping the economic factors in mind, damage tolerance design is used to put the structure for maximum usage after the crack initiation.

To ensure safety of the aircraft structure, it is very essential to clearly identify critical locations in airframe. These critical locations are the ones with very high stress concentration due to the presence of window cut-outs, mouse holes or rivets. The critical locations may be identified using stress analysis. The aircraft is unlikely to fail due to static load. Critical locations are the candidates to initiate failure, where the fatigue cracks creep in due to dynamic service loads. This paper describes the method to estimate safe life of an airplane fuselage with window cut-out, where the safe life is given by summing up the fatigue life and the life of the structure until the crack grows critical after the crack initiation. The rest of the paper is laid out as follows. Section II presents the methodology. Section III presents pre-processing. Section IV presents finite element analysis. Section V explains the total life estimation and Section VI brings up the conclusion.

II. METHODOLOGY

The method proposed for safe life estimation of aircraft fuselage consists of four steps namely, pre-processing, global analysis of the structure, local analysis of structure, and finally the safe life estimation. The pre-processing involves material selection and performs finite element meshing of geometrical model of aircraft fuselage. Further the global analysis locates the region of high stress concentration, which are analysed in detail during local analysis. Finally the

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

safe life is estimated by summing up the fatigue life and life after crack initiation. The Figure 1 shows the steps involved in proposed methodology.

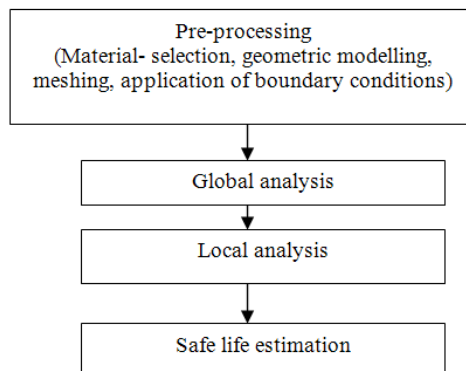


Fig 1: Steps involved in finding safe life

III. PRE-PROCESSING

A. Material selection

The choice of material used for aircraft structure is dictated by material properties like density, stiffness (young's modulus), strength (ultimate and yield strength), durability (fatigue), damage tolerance (fracture toughness and crack growth), corrosion etc. An alloy of aluminium is the best choice for the material due to its high strength to weight ratio. It is corrosion resistant, light weight and easy to fabricate. The material choice is Al2024-T351. The material properties are listed in Table 1.

Table 1: Material properties of Aluminium.

MATERIAL	ELASTIC MOD (GPa)	POISSON RATIO(Γ)	ULTIMATE STRENGTH σ_t (MPa)	YIELD STRENGTH σ_y (MPa)	DENSITY (g/cm ²)
Al 2024 T351	72	0.3	483	338	2.78

B. Geometrical Modelling

The aircraft mainly consists of three main regions namely,

- Fuselage
- Wings
- Empennage (tail).

Fuselage

The fuselage contains the cockpit or flight deck, passenger compartment and cargo compartment. A minimal lift is produced by fuselage, whereas it contributes to lot of drag. To minimize the drag the fuselage is streamlined. The fuselage with window cut-out is considered for the present work and modelled using CATIA V5. The components of global fuselage considered in global stress analysis are shown in Figure 2.

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

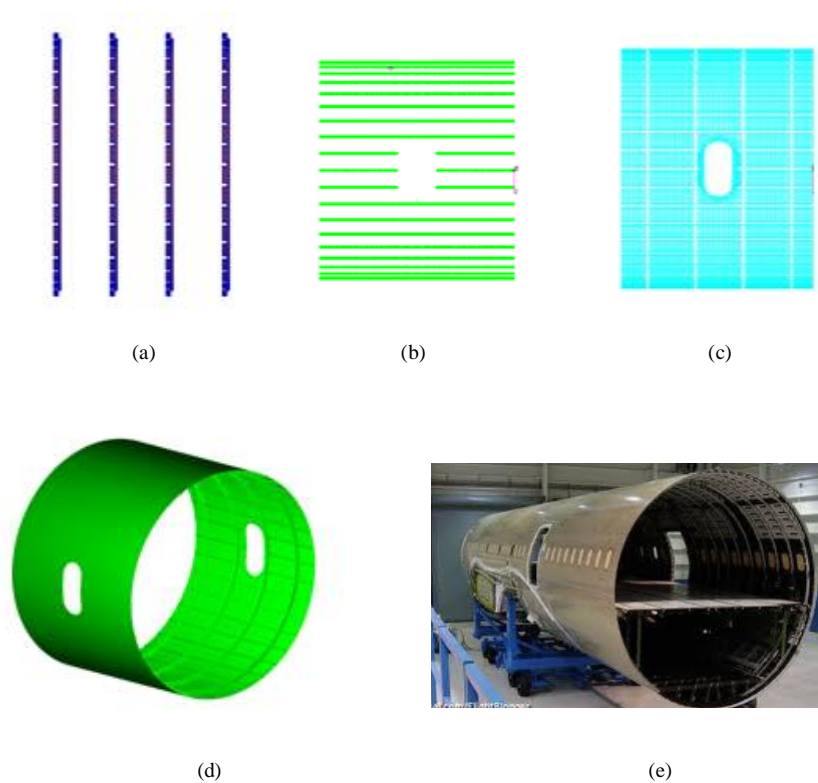


Fig 2: (a) Bulk heads (b) Longirons or stringers (c) skin with cut-out (d) global assembly (e)BOEING 777 fuselage

Bulkhead

Bulkhead is also known as frame. The bulkhead has Z cross-section. The bulkheads are placed on top of the skin and fastened onto the skin. These bulkheads run in circumferential direction. The dimension of the global fuselage is given in Table 2.

Table 2: Dimensions of the global fuselage

Parts	Dimensions (mm ²)	Thickness (mm)	Material	No's
Skin with window cut-out	R1398*2000(L)	2	Al	1
Z-Section(Bheads)	(40+58.3+23.3)*2000	2	Al	4
L-Section(Stringers)	(30+20)*2000	2	Al	40

The stress analysis is carried out on global fuselage to locate the regions of high stress concentration. Further these regions of high stress concentration are locally analysed considering the stiffened panel, which is the part of the global fuselage.

Following Figure shows the stiffened panel. A stiffened panel with window cut out is chosen for local analysis, as the global analysis points this region as region of high stress concentration.

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017



Fig 3: Stiffened panel

Below given are the dimensions of the stiffened panel.

Table 3: Dimensions of the global fuselage

Parts	Dimensions (mm ²)	Thickness (mm)	Material	No's
Skin with window cut-out	1750(b)*2000(L)	2	Al	1
Z-Section(Bheads)	(40+58.3+23.3)*2000	2	Al	4
L-Section(Stringers)	(30+20)*2000	2	Al	7

C. Mesh Generation

All the components of the fuselage with window cut out are meshed using FEM. Critical sections are finely meshed where the stress is expected to be high. Following figure shows the meshing of a longiron.

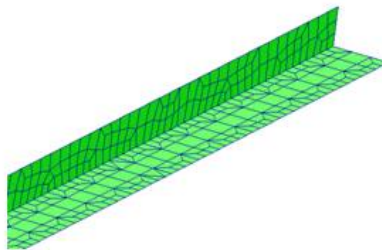


Fig 4: Meshing of a longiron

The mesh is carefully generated such that there is a node present at the point.

D. Loading condition

Loads and boundary conditions to which the fuselage is subjected is given below.

Table 4: Loads and boundary conditions

Name of the part	Displacement(Fixed ends)	Loads (Differential pressure)
Skin	X0,Y0,Z0,R(0,0,0)	0.041MPa
Bulkhead	X0,Y0,Z0,R(0,0,0)	0.041MPa

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

IV. FINITE ELEMENT ANALYSIS

A. Global Analysis

When the fuselage is subjected to internal pressurization 2 kinds of stresses are induced.

- Longitudinal stress-responsible increase in diameter of fuselage.
- Hoop stress or tangential stress-responsible of increase in length of fuselage.

The hoop stress is given by [1]

$$\sigma_{\text{hoop}} = \frac{\text{pressure}(p) * \text{radius}(r)}{\text{thickness}(t)}$$

The hoop stress values for different pressurization loads are tabulated in Table 5.

Table 5: Hoop stresses under different pressure loads

Pressure(p) MPa	Radius (r)mm	Thickness (t)mm	σ_{hoop} MPa
0.041	1398	2	28.65
0.044	1398	2	30.75
0.048	1398	2	33.55
0.051	1398	2	35.64
0.054	1398	2	37.74
0.058	1398	2	40.54
0.061	1398	2	42.63

The global fuselage is analysed in FEM. The values of the hoop stresses are averaged over the region. The results are comparable to the theoretical results. For an internal pressure of 0.41MPa the theoretical value of hoop stress obtained is 28.65MPa and the value obtained through analysis is 28.3MPa. The maximum value of 107MPa is observed around the window cut-out. Hence this is the region that requires detailed analysis.

B. Local Analysis

A detailed analysis is of region of window cut-out is done considering the stiffened panel with window cut-out. Fine meshing is done unlike in global analysis. Dimensions of the stiffened panel are given in below table.

Table 6: Dimensions of stiffened panel

Parts	Dimensions (mm ²)	Thickness (mm)	Material	Nos
Skin with cutout	1750(B)*2000(L)	2-4	Al	1
Z-section(B heads)	(40+58.3+23.3)*2000	2	Al	4
L Section (Stringers)	(30+20)*2000	2	Al	7

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

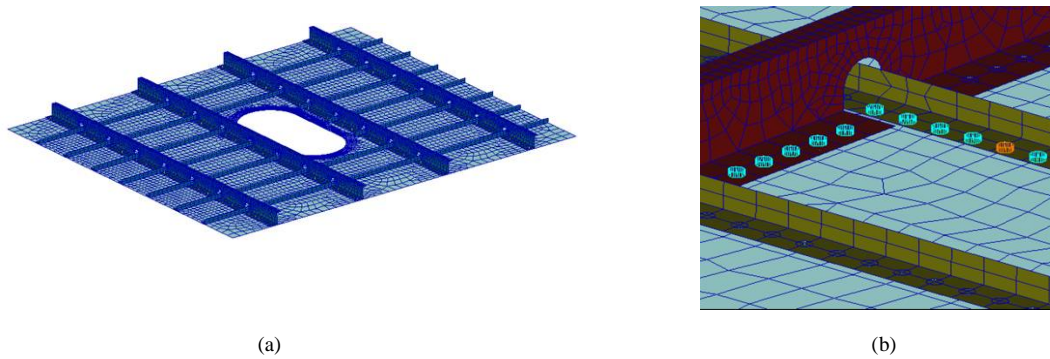


Fig 5: (a) Assembly of the stiffened panel meshed (b) Rivets used to assemble all components of stiffened panel

Fine meshing is done at the mouse hole cut-out region. The 2D shell CQUAD4 and Tria3 elements are used for meshing skin, frame, and stringers. Rivets are simulated using IDCBEAM elements. Mesh is generated using MSC PATRAN.

Table 7: Number and type of elements used in mesh generation

Components	Type of elements	No of elements
Skin	CQuad4 & Tria3	8960
Bulk heads	CQuad4 & Tria3	5580
Stringers	CQuad4 & Tria3	2176
Rivets	CBeam	738
Total numbers of element		17462

Stress Analysis

Linear stress analysis of the stiffened panel is done using the tensile load at the edge corresponding to the pressurization.

$$P = \sigma_{hoop} * A$$

Uniformly distributed tensile load (UDL) is calculated as [4],

$$\text{Uniformly distributed Load (UDL)} = \text{Load (P)} / \text{Length}$$

Accordingly the loads are calculated and tabulated in Table 8.

Table 8 : Pressure and respective load on skin

Pressure (p) MPa	σ_{hoop} MPa	Area A_{skin} mm^2	Load P_{skin} mm^2	UDL_{skin} kg/mm^2
0.041	28.65	4000	11682	5.84
0.044	30.75	4000	12539	6.26
0.048	33.55	4000	13680	6.84
0.051	35.64	4000	14532	7.32
0.054	37.74	4000	15380	7.82
0.058	40.54	4000	16530	8.3
0.061	42.63	4000	17382	8.8

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

Table 9 :Pressure and respective load on bulkhead

Pressure (p) MPa	σ_{hoop} MPa	Area $A_{bulkhead}$ mm^2	Load $P_{bulkhead}$ mm^2	UDL $_{bulkhead}$ kg/mm^2
0.041	28.65	243.2	711	5.84
0.044	30.75	243.2	763	6.26
0.048	33.55	243.2	832	6.84
0.051	35.64	243.2	884	7.32
0.054	37.74	243.2	936	7.82
0.058	40.54	243.2	1005	8.3
0.061	42.63	243.2	1057	8.8

Stress analysis is carried out by applying boundary conditions and UDL obtained. Figure 6 shows the analysis result for pressure of 0.041 MPa.

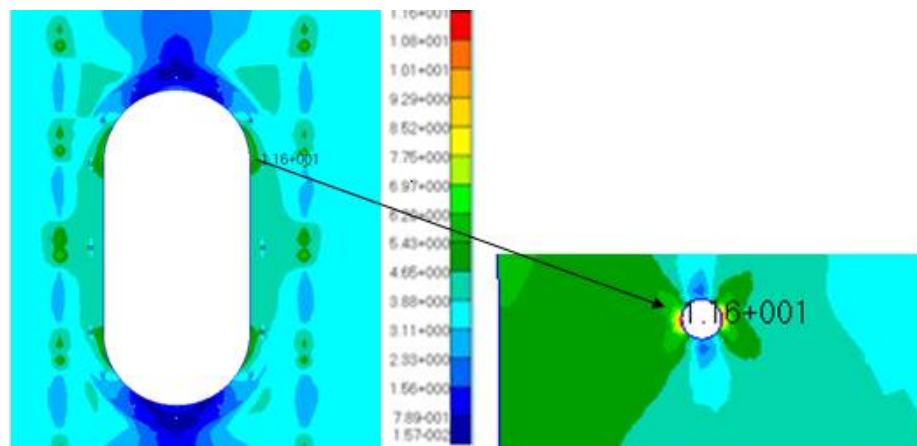


Fig 6: Maximum stress location in the skin window cut-out.

The magnitude of maximum tensile stress is observed in the loading direction. The location of maximum stress in the skin is seen to be near the window cut out exactly near the riveting hole. The maximum stress locations are probable locations for crack initiation. Invariably these locations will be at stringer cut-out locations in bulkhead.

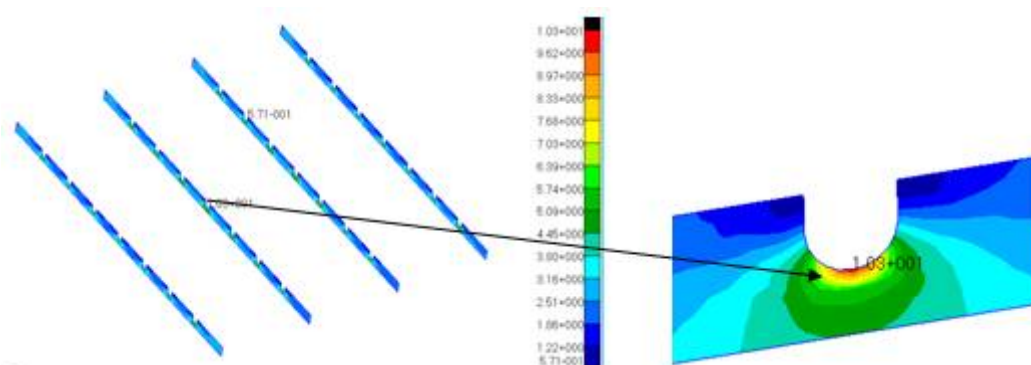


Fig 7: maximum stress location in bulkheads

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

Table 10: Maximum stress induced for various pressure loads

Pressure (p) MPa	σ_{hoop} MPa	Max stress window cut out kg/mm ² (MPa)	Max stress Bulkheads kg/mm ² (MPa)
0.041	28.65	11.6(13.79)	10.3(101.0)
0.044	30.75	12.6(123.600)	11.2(109.87)
0.048	33.55	13.6(133.41)	12.0(117.7)
0.051	35.64	14.6(143.22)	12.9(126.54)
0.054	37.74	15.5(152.0)	13.7(134.4)
0.058	40.54	16.59(161.86)	14.6(143.2)
0.061	42.63	17.5(171.67)	15.5(152.0)

The regions of maximum stress are likely areas where the crack would initiate. These regions are near rivet hole in the window frame with maximum value of 171.67 MPa at pressure of 0.061 and in a bulk head regions are near mouse hole with a maximum stress of 152MPa at 0.061 MPa.

V. TOTAL LIFE ESTIMATION

A. Fatigue life estimation

A typical flight load spectrum is considered for the fatigue analysis of stiffened panel. For performing the fatigue calculations constant amplitude loading is preferred. Typical constant life diagram [2] for un-notched fatigue behaviour of 2024-T3 Aluminium alloy is shown below. If loading is constant amplitude, then it represents the number of cycles until the part will failure due to fatigue. Calculation of fatigue life to crack initiation is carried out by Miners rule[3]. The reference test condition R=0 used for obtain fatigue properties. For this condition $\sigma_{min}=0$ is called 'pulsating tension' under constant amplitude loading or zero tension loading. The following figure shows the mean stress level below which the material has infinite life. For most engineering purposes infinite is taken to be 1 million cycles.

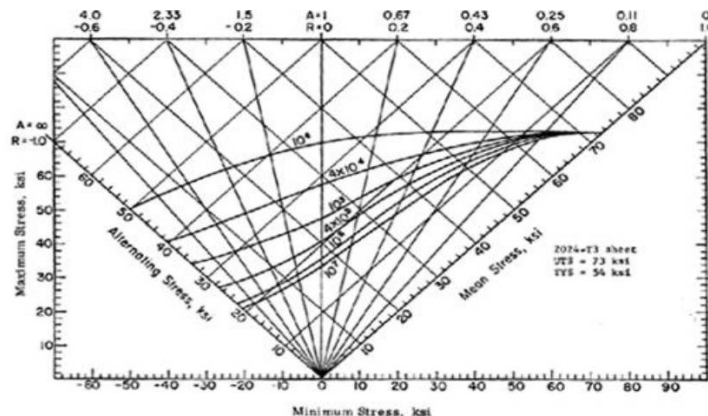


Fig 8: Typical constant life diagram for un-notched fatigue behavior of 2024-T351Aluminium alloy.

The above shown graph is for un-notched plate subjected to fatigue loading. Stress concentration factor(K) for notched plate is given by [4]

$$K = \sigma_{max} / \sigma_{nominal}$$

For a plate with very small circular hole $K=3(\text{max})$. A value of 2 is considered for present analysis.

$$\sigma_{max} = 2 * \sigma_{nominal}$$

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

For variable spectrum loading is simplified as block loading, where each block consists of load cycles corresponding to 1500 flight hours and each block consists of 67450 cycles. Damage calculation is carried out using Miners rule for complete service life of air craft.

The value of the maximum stress for window is obtained using stress analysis. A correction factor of 2 is used to account for the variation in shape called shape factor .The results are tabulated in Table 11.

Table 11: Maximum and mean stress values for window frame.

Pressure (psi)	Max stress skin (Mpa)	Corrected Max stress skin (Mpa)	Mean stress (Mpa)	Mean stress (Ksi)
6.0	113.8	227.6	113.8	16.55
6.5	123.6	247.2	123.6	17.98
7.0	133.41	266.82	133.41	19.41
7.5	143.22	286.44	143.22	20.83
8.0	152.0	304.0	152.0	22.11
8.5	161.86	323.72	161.86	23.55
9.0	171.67	343.34	171.67	24.97

Damage calculation for window frame
Applying miners rule for one block of loading we have,

$$D_i = \sum_{i=1}^{i=n} \frac{n_i}{N_i}$$

$$D_i = \frac{3000}{1.0E+7} + \frac{17500}{1.0E+7} + \frac{12800}{1.0E+7} + \frac{2450}{1.0E+7} + \frac{1850}{1.0E+7} + \frac{1650}{1.0E+7} + \frac{1200}{1.0E+7}$$

$$D_i = 6.75E - 03$$

Table 12: Damage calculation for window cut-out

Pressure (p) MPa	Mean stress ksi	No of cycles (n _i)	Cycles to failure (N _i)	Damage fraction(d _i)
0.041	16.55	30000	1.0E+07	3.0E-03
0.44	17.98	17500	1.0E+06	17.5E-03
0.48	19.41	12800	6.0E+05	21.3E-03
0.51	20.83	2450	4.0E+05	6.12E-03
0.54	22.11	1850	3.5E+05	5.28E-03
0.58	23.55	1650	1.5E+05	11E-03
0.061	24.97	1200	1.00E+05	12E-03
Total Damage accumulated (d _i)				76.2E-03

A similar calculation follows for bulkheads, which gives total accumulated damage d_i=6.75E-03. It is observed that the damage accumulation in skin is more than damage accumulation in bulkhead. Hence the crack initiation is more likely to occur in skin first rather than the bulk head.

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

Calculation of service life based on skin

The number of loading blocks (B) required to cause failure is given by,

$$B = \frac{1}{\overline{D_i}}$$

$$B = \frac{1}{76.2E - 03}$$

$$B=13.12\text{Blocks}$$

The total service life can be calculated as given below,

$$T_{\text{hrs}}=B*1500$$

$$T_{\text{hrs}}=13.12*1500$$

$$T_{\text{hrs}}=19800\text{hrs}$$

Therefore fuselage structure has a service life of 19800 flight hours under pressurization loads.

B. Life post crack initiation

The life of the fuselage after the crack initiation can be obtained by knowing the crack growth rate. The crack growth rate is calculated from da/dN V/s $\Delta K_{\text{effective}}$ curve for the respective material. Where K is the stress intensity factor calculated using FEM. A plot of da/dN V/s $\Delta K_{\text{effective}}$ for Al 2024 T351 which is provided by aerospace material manufacturer ALCOA [5] is shown in Figure 9.

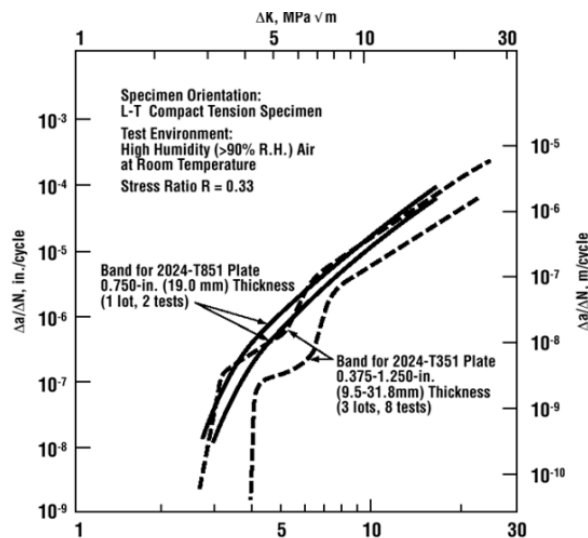


Fig 9: da/dN V/s $\Delta K_{\text{EFFECTIVE}}$ plot for AL 2024 T351

Hence to obtain crack growth rate the $\Delta K_{\text{effective}}$ has to be evaluated, which is given by [6]

$$K_{\text{effective}} = K_{\text{max}} - K_{\text{opening}}$$

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

Where K_{max} is Maximum SIF for given crack length and $K_{opening}$ is given by,

$$K_{opening} = K_{max} (0.5+0.4R)$$

The plot is produced for a stress ratio of $R=0.33$.

$$\Delta K_{effective} = 0.368 K_{max}$$

Similarly the $\Delta K_{effective}$ is calculated for different crack lengths and the crack growth rate is found from the graph in Figure 9. Number of cycles for the crack to reach the nearest bulkhead (ΔN) is found using the relation,

$$\Delta N = (dn/da)_{avg} * (\text{crack increment})$$

Table 13: Crack growth rate

Crack length 2a mm	K max MPa \sqrt{m}	$\Delta K_{effective}$ MPa \sqrt{m}	da/dN m/cyc	dN/da cyc/mm	(dn/da) _{avg}	ΔN
111	28.55	10.50	5E-07	2000	3500	14000
107	22.93	8.43	2E-07	5000		
102.6	24.3	8.94	2E-07	5000	3750	15000
98.6	25.04	9.21	4E-07	2500		
94.5	25.03	9.21	4E-07	2500	2500	11250
90.5	25.3	9.31	4E-07	2500		
86.4	25.41	9.35	4E-07	2500	2500	10000
82.4	25.77	9.48	4E-07	2500		
78.4	25.5	9.38	4E-07	2500	2500	10000
74.4	25.88	9.52	4E-07	2500		
70.4	25.91	9.53	4E-07	2500	2500	11000
66	25.88	9.52	4E-07	2500		
62	25.87	9.52	4E-07	2500	2500	10000
58	26.7	9.82	4E-07	2500		
54	28	10.3	5E-07	2000	3500	14000
50	22.71	8.35	2E-07	5000		
42	14.12	5.19	1E-07	10000	55000	467500
33.5	10.98	4.04	1E-08	100000		
25.25	9.28	3.41	-	-		
17	7.3	2.68	-	-		
11	5.8	2.13	-	-		
7	4.5	1.65	-	-		

- It is seen that the crack growth rate is negligible below crack lengths of 33.5 mm.
- Hence the crack has to be monitored when and after it reaches a value of 33.5 mm due to fatigue loading
- It is seen that the crack growth rate is negligible below crack lengths of 33.5 mm.
- Hence the crack has to be monitored when and after it reaches a value of 33.5 mm due to fatigue loading.



ISSN(Online): 2319-8753
ISSN (Print): 2347-6710

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 6, Issue 10, October 2017

The total number of cycles the aircraft could fly before the crack could reach the nearest bulkhead is 562750 cycles. This equals to 8.34 loading blocks or equal to 12514 flight hours.

VI. CONCLUSIONS

For a given loading block consisting of 64450 cycles which equals 1500 flight hours, it takes 13.12 blocks or 19800 flight hours for the initiation of crack under fatigue load. Once the crack initiates, the number of cycles the aircraft could fly before the crack could reach the nearest bulkhead is 562750 cycles. This equals to 8.34 loading blocks or equal to 12514 flight hours. Therefore the total flight hours the aircraft could be operated safely is equal to 32314 hours.

REFERENCES

- [1] V.B Bhandari, Design of Machine elements, Tata Mc Graw Hill, 2010
- [2] Elmer Franklin Brunhns, Analysis and Design of airplane structures, Tristate offset company, 1949
- [3] Ralph I Stephens, Metal fatigue in engineering, A Wiley: Interscience publication, 1980
- [4] James Gere, Mechanics of Materials, PWS Publication Company, 1997.
- [5] ALCOA mill products, Alloy 2024 sheet and plates, Fatigue properties data chart.
- [6] David Broeks, Practical use of Fracture mechanics, Springer, 1989