FATIGUE CRACK GROWTH LIFE PREDICTION FOR A STIFFENED PANEL WITH LANDING GEAR OPENING CUTOUT USING MVCCI METHOD.

Amit Kumar Shrivastava[1], Manjunath H S[2], Aravinda D[3], N Shashikanth[4]

[1]Research Scholar, [2,3]Asst. Professor, Department of Mechanical Engineering

[4]Associate Professor, Department of Mechanical Engineering

Dr. Ambedkar Institute of Technology, Bengaluru, India

---***---

Abstract*: Fatigue and Damage Tolerance (F&DT) is procedure for prediction of fatigue life and crack growth. , in which damage is considered at start of analysis and with the lapsed of time how it grows to reduce the life of component. To start with, this requires the stress analysis of a wing under various load distributions that the air frame is going to be subjected to. This identifies the location of high tensile stresses which are the potential sites for fatigue crack initiation. At how many flight hours cracks will initiate at these locations depend on the local stress histories due to the design service loading of the aircraft. Analytically this is done by computing fatigue accumulation under the local stress histories. The critical crack length will be established from the consideration that, when stress intensity factor becomes equal to the fracture toughness the crack will become critical. And the crack arrest capability of the stiffener perpendicular to the cracking direction will be established analytically*

Keywords: Damage tolerance, crack propagation, stress intensity factor, MCCVI.

1. INTRODUCTION.

Starting from 1980 recommendation for damage tolerance for large civil airplane was introduced. It is very rare that a structural failure of an aircraft occurs due to the static overload. During its service operation, fatigue cracks initiate at critical locations of the air frame. These cracks grow under the variable amplitude service loading. Fatigue and Damage Tolerance (F&DT) quantify the air frame structure, therefore are of paramount importance for the long service life of an aircraft. If a fatigue crack at the critical location of the air frame goes unnoticed then it could lead to a catastrophe. Here in the current project work is addressed to a stiffened panel of a landing gear opening cutout of a typical transport aircraft wing.

2. LITERATURE REVIEW.

Boris G. Nesterenko [1], in this paper, the author presents the results of research on some real problems in ensuring damage tolerance aircraft. The first issue considered is the growth of fatigue cracks with random spectra. The second experiment investigates residual strength.

Grigory I. Nesterenko, [2], in this paper, the author presents the results of the analysis of the research evidence on fatigue test safety and damage tolerance of the Russian plane. Tensions occur in the structures of wide body aircraft. Curves fatigue resistance for structures of the wings and fuselage are generated. Residual strength data are presented for these structures having a crack under the skin stiffener broken. Generalized curves for the duration of skin cracks under the broken stiffener are presented.

A Rama Chandra Murthy et al [3], In this paper, the author explains the methodologies for evaluating damage tolerant stiffened panels subjected to fatigue loads. The two main objectives of tolerant, namely damage assessment, life prediction and assessment of the residual strength of unstiffened remaining panels have been discussed. Concentric and eccentric reinforcements have been considered. Stress intensity factor for the rigid panel has been calculated using the parametric equations of the modified comprehensive technical virtual crack closure numerically integrated method.

Saint. Tavares et al [7], in this paper a review of post processing techniques to estimate the stress intensity factors (SIF) using stress fields and displacement is calculated by numerical methods.

JC Newman, Jr. [8], in this article, the author explains some of the advances that have been made in the stress analysis of aircraft components cracked in understanding the process of growth and fatigue cracks fatigue, and waste predicting resistance complex aircraft structures generalized fatigue damage.

A. Brot et al. [9], Israel Aerospace Industries (IAI) has studied the behavior of damage tolerance fully rigid metallic structures as part of a project in international course called Daton.

3. OBJECTIVE.

- How the conveying limit of the aircraft decreases with crack length.
- Crack length versus number of flight hours.

4. METHOLOGY.

1. GEOMETRIC MODELLING.
2. FINITE ELEMENT ANALYSIS IN MSC PATRAN
3. SOLVING THE PROBLEM IN MSC NATRAN.
4. RESULT AND DISCUSSION IN MSC PATRAN.

5. CONCEPTUAL DESIGN.

Design of the landing gear cutout is modeled using modeling software CATIA V5, which is illustrated in fig 1.

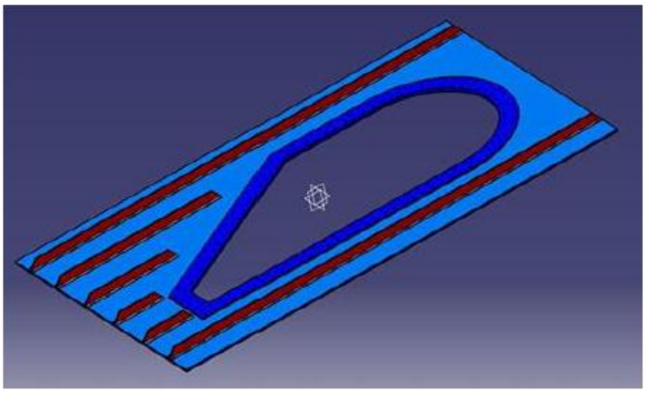

Fig.1. CATIA V5 model of landing gear cutout.

5. FINITE ELEMENT ANALYSIS.

CATIA model is extracted into MSC PATRAN software where fine meshing is carried out. Based on given input load on stiffened panel is calculated. Figure 2.a and 2.b shows fine meshing.

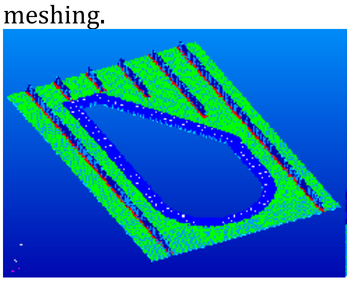

Fig 2.a Meshing of stiffened Panel.

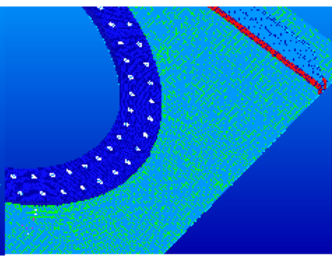

Fig 2.b close up view

5.1CALCULATION NET LOAD ACTING ON STIFFENED PANEL.

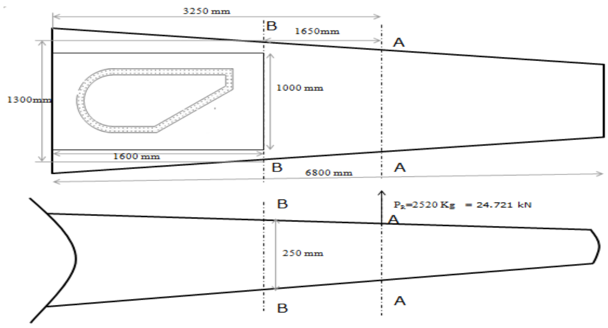

Fig 3 Load distribution on wing structure.

The class of aircraft is 6 seater and laod case level is maximum.

Specification:
Span of wing = 6.8 m
Total weight of aircraft = 1400 kg
Depth of wing = 250mm
Design load = 3g , FOS = 1.5
Material property:
Material : Aluminum alloy 2024-T351
Young's modulus:73.1 GPa
Poisons ratio: 0.3
Density: 27.271 kg/mm^3

Load acting along the width of wing skin calculated is 125.509 kN, which corresponds to 4.5g condition.

The analysis of stiffened panel is done in two stage. In first Stage the analysis is carried out without crack, which gives maximum stress and maximum displacement value 42.2kg/mm^2 and 2.56 mm respectively. In second stage analysis is done at maximum stress area and making a hole at that locations which gives maximum stress value and displacement value as 2.75kg/mm^2 and 2.75 mm respectively.

5.2 LOADS AND BOUNDARY CONDITIONS.

Boundary condition:

- One end of the stiffened panel is restricted to all 6 degrees of freedom.
- z direction is limited by all nodes of the stiffened panel.

Load:

Evenly distributed125.509 kN/m is applied to the other end of the stiffened panel.

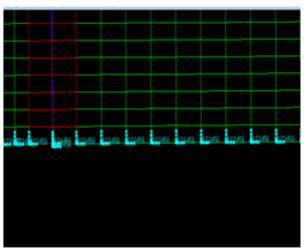

Fig 4.a Boundary conditions

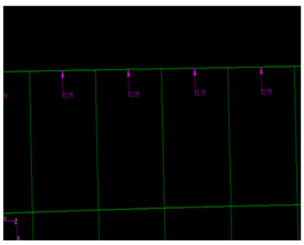

Fig 4.b Load applied

5.3 STRESS AND DISPLACEMENT ANALYSIS OF A STIFFENED PANEL.

Stress and displacement analysis of stiffened panel is carried out in 2 steps. In first step the stress and displacement analysis is done without hole and in second step the location where maximum stress is found is selected and then crack is placed again remeshed at that location and then stress and displacement contour is found.

Results of step 1 : stiffened panel without crack.

Maximum displacement	**42.2 Mpa**
Maximum stress	**2.56 mm**

Figure 5.a and 5.b shows stress and displacement contour respectively.

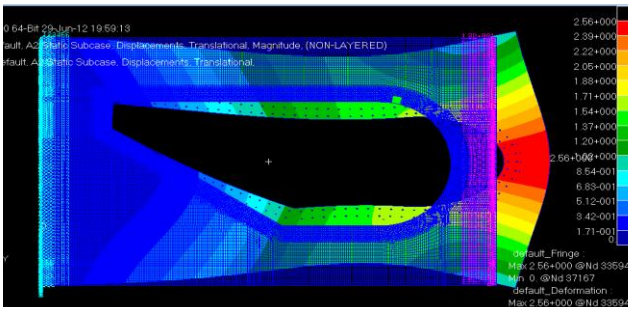

Fig 5.a Stress Analysis in MSC PATRAN

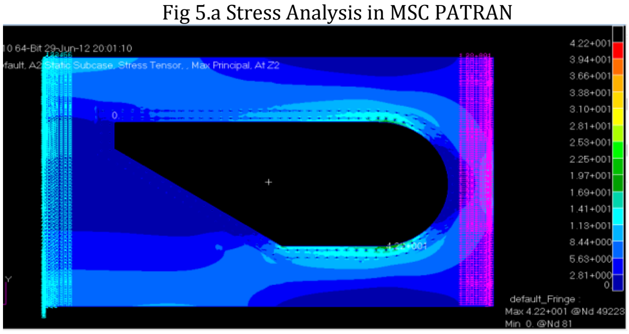

Fig 5.b Displacement Analysis in MSC PATRAN

Result of step 2 for stiffened panel with crack

Maximum stress	**0.275 Gpa**
Maximum displacement	**3.95mm**

The stress and displacement contour is shown in figure 6.a and 6.b respectively.

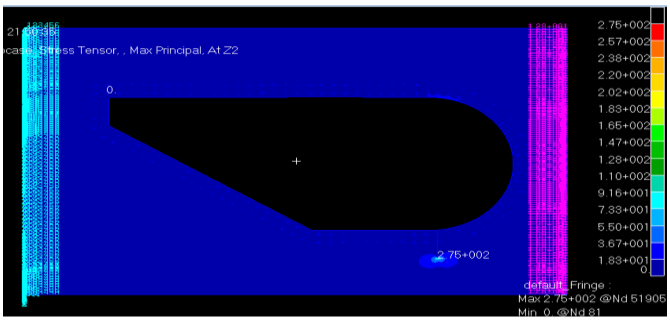

Fig 6.a stress contour for panel with crack.

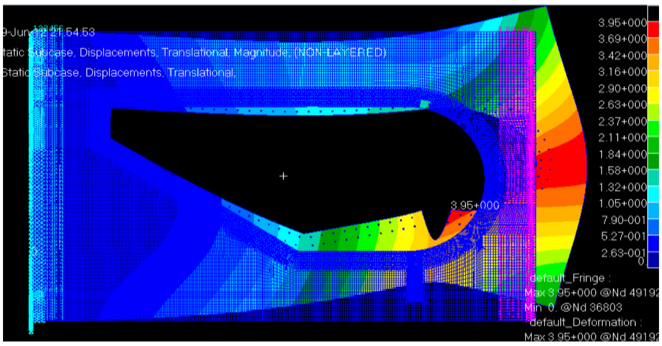

Fig 6.b Displacement contour for panel with crack.

Calculation of stress intensity factor using MVCCI method is done for different crack length (starting from 0.66mm to 134 mm.) we found that stress intensity factor first increases, at 11 mm crack length it shows decrease of SIF again an increase of SIF as crack length increases again it starts decreasing after crack length 18 mm and finally SIF starts decreasing at a crack length of 122 mm which will lead to failure of specimen. The consecutive increase and decrease of SIF can be understood by use of stiffer which ceases the crack growth for a while. The figure 6 shows the distribution of SIF as a function of crack length.

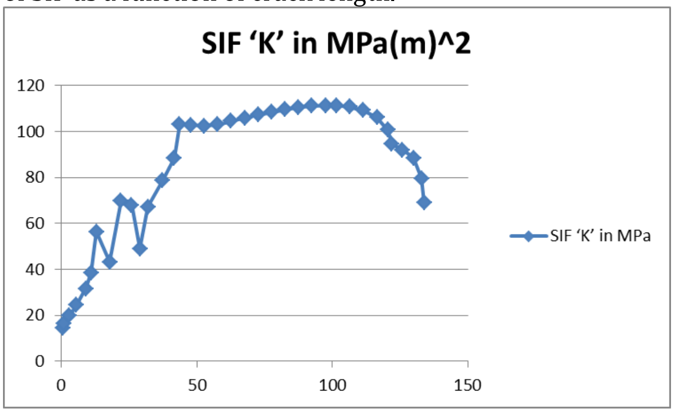

Figure 6. The distribution of SIF as a function of crack length.

6. LIFE PREDICTION.

When aircraft take off the wing experiences variable loading during flight.. Fatigue calculation is done by simplifying the variable spectrum into block loading which is considered as Shown in table 1.

Table 1: block loading.

Number of cycles	g loads
1	0.5 to 0.75
3	0.75 to 1.0
1	1.0 to 1.25
6	1.25 to 1.5
5	0 to 1.75
1	0 to 2.0
1	-0.5 to 1.5

In the above mention cycle g corresponds to acceleration due to gravity. Each block consists of 1823 number of cycles and one representative black of loading is of 100 hours. This analysis is carried out until crack becomes critical or technically we can say when SIF becomes equal to fracture toughness. The fracture toughness of the material Aluminum alloy 2024-T351 / T3 are 99.4 MPa$(m)^{1/2}$.For the critical crack length we will use solver MSC PATRAN to obtain Fatigue crack growth data curve. Fatigue crack growth material data curve is shown in figure 7.

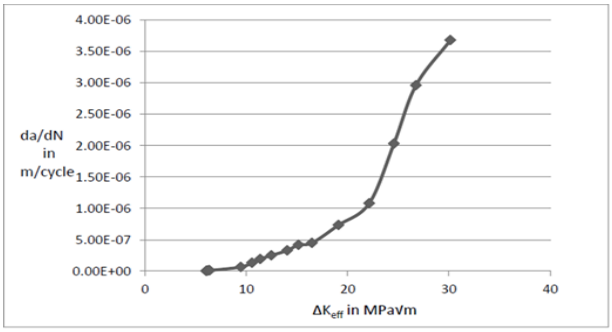

Fig 7: Fatigue crack growth data curve.

Using above fatigue crack data curve we can find total number of blocks required from which the number of hours required for the material to be safe.

6.1 CALCULATIONS TO FIND: K_{eff}

Where $\Delta K_{eff} = K_{max} - K_{opening}$

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

Where K_{max} = Maximum Stress intensity factor

R= Stress ratio

K_{max} for 0.66 mm crack length at 4.5g load = 14.47Mpa (Obtained through FEA results and MVCCI method).

For 0.66 mm crack length, the SIF at 0.75g =0.75 (Linear interpolation)

K_{max} = 2.411 MPa$(m)^{1/2}$

R = 0.6667

On substituting values we get,

$K_{opening}$=1.85MPa$(m)^{1/2}$

$\Delta K_{eff} = K_{max} - K_{opening}$ = 2.411-1.85 = 0.56 MPa$(m)^{1/2}$

As the value of Keff falls in the region that is considered safe, we can say that the crack growth will not take place at 0.75 g for any number of cycles. Similarly calculating ΔKeff 0.66 mm crack length from the edge of the hole in 1g, 1.25 g, 1.5 g, 1.75 g and 2 g. Results has been tabulated in table 2 where NCG denotes no crack growth.

g loads in m/s^2	g_{max} in m/s^2	K_{max} in MPa$(m)^{1/2}$	$K_{opening}$ in MPa$(m)^{1/2}$	Stress ratio R	K_{eff} in MPa$(m)^{1/2}$	da/dN in m/cycle
0.5 to 0.75	0.75	2.411	1.849	0.6667	0.563	NCG
0.75 to 1.0	1.0	3.215	2.572	0.75	0.643	NCG
1.0 to 1.25	1.25	4.019	3.296	0.8	0.723	NCG
1.25 to 1.5	1.5	4.823	4.019	0.833	0.804	NCG
0 to 1.75	1.75	5.627	2.813	0	2.814	NCG
0 to 2.0	2	6.431	3.215	0	3.215	NCG
-0.5 to 1.5	1.5	4.823	1.768	-0.333	3.055	NCG

Table 2: crack growth rate calculation for crack length 0.666 mm

Similarly we calculate da /dN for a crack length of 9 mm, which is tabulated in table 3.

g loads in m/s^2	g_{max} in m/s^2	K_{max} in MPa$(m)^{1/2}$	$K_{opening}$ in MPa$(m)^{1/2}$	Stress ratio R	K_{eff} in MPa$(m)^{1/2}$	da/dN in m/cycle
0.5 to 0.75	0.75	5.244	4.021	0.6667	1.224	NCG
0.75 to 1.0	1.0	6.992	5.594	0.75	1.398	NCG
1.0 to 1.25	1.25	8.74	7.167	0.8	1.573	NCG
1.25 to 1.5	1.5	10.488	8.74	0.833	1.748	NCG
0 to 1.75	1.75	12.236	6.118	0	6.118	5.64E-09
0 to 2.0	2.0	13.984	6.992	0	6.992	2.17E-08
-0.5 to 1.5	1.5	10.488	3.846	-0.333	6.643	1.56E-08

Table 3: crack growth rate calculation for crack length 9 mm

we find that there is no crack growth at 0.75 g, 1 g, 1.25 g and 1.5 g. But we find that the crack growth is carried out by the bottom three 'g'. But the da / dN have is for one cycle in which 'g' in particular.

For 0 to 1.75g, the crack growth is 5.64E-9 m per cycle

For 52 cycles, the crack growth = 52*5.64E-9 = 2.933E-8 m

For 0 to 2.0g, the crack growth is 2.1E-8 m per cycle. For 1 cycle, the crack growth = 2.17E-8 m.

For -0.5 to 1.5g, the crack growth is 1.56E-8 m per cycle.

For 15 cycles, the crack growth = 15*1.56E-8 m =2.34E-7m.

By adding all the 3 crack growths, we get the crack growth of 1 block= (2.93+2.17+2.34)E-8 m = 0 .000549mm.

Therefore the number of blocks needed for the crack growth of 1 mm is = 1821.49 or simply 1821 blocks.

Now for the crack to reach the next crack length of 11 mm should be increased by 2 mm. Therefore the number of blocks required for the crack to grow by 2 mm is =

1821*2 = 3642 blocks.

Likewise by calculating the number of blocks needed for different lengths of crack until the crack reaches critical length of 44 mm.

We know that a block is equal to 100 flight hours. So the total number of flight hours is 7520 blocks is = 7520*100 = 752000 hours of flying, is calculated in table 4.

Table 4: Block calculation.

Crack length in mm to reach from	No of blocks required
9 to 11	3642
11 to 13	904
13 to 18	420
18 to 22	1196
22 to 26	180
26 to 29	147
29 to 32	543
32 to 38	300
38 to 42	136
42 to 44	52
Total	7520

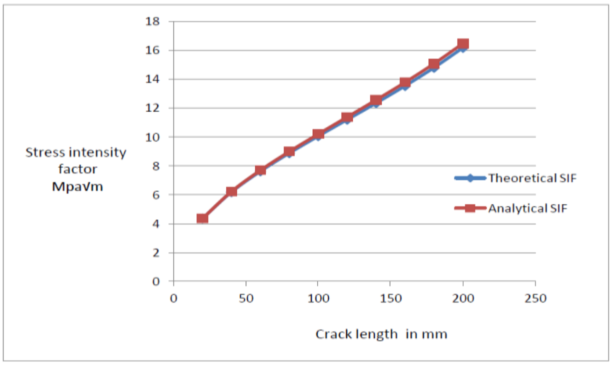

Fig 8: Plot of SIF variation with crack length

7. CONCLUSION.

- Using damage fatigue tolerance concept life estimate of stiffened is carried out, in which firstly location of maximum stress is identified then a hole is incorporated at that location and selecting that location it is remeshed to get stress and displacement contour in FEA software MSC NASTRAN.
- Using MSC PATRAN solver, analytically stress intensity factor is calculated by MVCCI method to get critical crack length which is at point where stress intensity factor is equal to the facture toughness of the material.
- For predicting fatigue crack growth variable load range is simplified to the block loading. A block represents 100 flight hours. And then the number of blocks required for the crack to grow critical is calculated.
- Once the total number of blocks for be critical crack is known, the number of flight hours of the aircraft to fly safe is found. By calculating we find that the aircraft can fly for 752000 numbers of hours before it fails.
- Stress intensity factor calculated using MVCCI method and theoretical method is almost same.
- The validation of the analytical method is carried out by calculating the SIF plate with a crack problem analytically and compared to the theoretical SIF. And we find that the SIF get both the analytical and theoretical method is the same. For this we can conclude that the FEA software used for analysis is valid as shown in figure 8.

8. ACKNOWLEDGEMENT.

I wish to thank to our HOD, **Dr. L. CHANDRA SAGAR**, Professor and HOD, Department of Mechanical Engineering, Dr. Ambedkar Institute of Technology, for his constant support and encouragement throughout the course of the project. I would like to express my heartfelt gratitude to my guide **H.S MANJUNATH**, Assistant Professor, Department of Mechanical Engineering, Dr. Ambedkar Institute of Technology, for his valuable guidance throughout my project work.

9. BIBLIOGRAPHY.

[1] Boris G. Nesterenko, "Analytically experimental study of damage tolerance of aircraft structures", Moscow Institute of Physics and Technology (State University), Department of Aerodynamics and Flight Engineering, Russia, 2002.

[2] Grigory I. Nesterenko, "Service life of airplane structures", Central Aero hydrodynamic Institute (TsAGI), Russia, 2002.

[3] A Rama Chandra Murthy, G S Palani Nagesh R Iyer, "Damage tolerant evaluation of cracked stiffened panels under fatigue loading", Sadhana Vol. 37, Part 1, February 2012, pp. 171–186.

[4] S. M. Beden et al "Review of Fatigue Crack Propagation Models for Metallic Components" European Journal of Scientific Research, ISSN 1450-216X Vol.28 No.3 (2009), pp.364- 397.

[5] Jarkko Tikka and Patria, "Fatigue life evaluation of critical locations in aircraft structures using virtual fatigue test", 2002.

[6] J. C. Newman, Jr., "Crack growth predictions in aluminium and titanium alloys under aircraft load spectra", Department of Aerospace Engineering, Mississippi State University,

Mississippi State, MS, USA.

[7] S.M.O. Tavares et al, "Stress intensity factors by numerical evaluation in cracked structures", Department of Mechanical Engineering and Industrial Management.

[8] J. C. Newman, Jr, "Advances in fatigue and fracture mechanics analyses for aircraft structures", Mechanics and Durability Branch, NASA Langley Research Center, Hampton, VA, USA.

[9] A. Brot et al., "The damage-tolerance behavior of integrally stiffened metallic structures", Israel Annual Conference on Aerospace Sciences, 2008.

[10] Park Jeong Kyu et al., "Finite Element Method analysis and Life Estimation of aircraft structure Fatigue/Fracture Critical Location", Korea Aerospace Industries, Sacheon City, South Korea, 2001.