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FATIGUE CRACK GROWTH LIFE PREDICTION FOR A STIFFENED PANEL WITH LANDING GEAR OPENING CUTOUT OF A TRANSPORT AIRCRAFT WING

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Abstract: Fatigue and Damage Tolerance (F&DT) quantify the air frame structure, therefore are of paramount importance for the long service life of an aircraft 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.

I. INTRODUCTION.

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.

II. 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.

SM Beden et al [4], this fatigue role models as a scientific and engineering knowledge related to the fatigue of materials and structures, including models predicting fatigue life under load variable amplitude and constant revision. Selecting the right model it is generally based on the experience of the analyst and persona1 preference.

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.

III. OBJECTIVE.

- Ensure safety of structure.
- Damage tolerance design of stiffened panel of landing gear cutout.
- Performing stress analysis and maximum stress location and then calculating life of panel using MVCCI method.

IV. STEPS INVOLVED AND SOFTWARE USED.

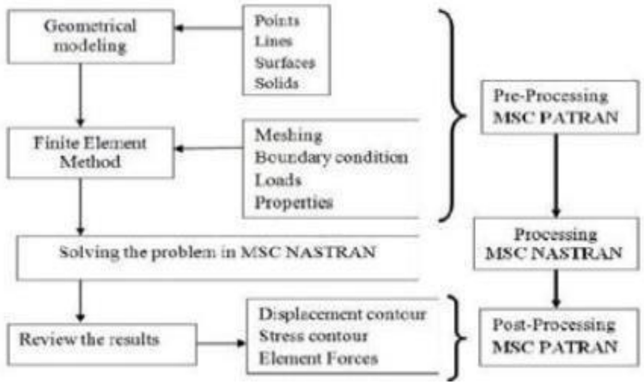


Fig.1. Steps involved.

V. CONCEPTUAL DESIGN

Design of the landing gear cutout is modeled using modeling software CATIA V5, which is illustrated in fig 2.

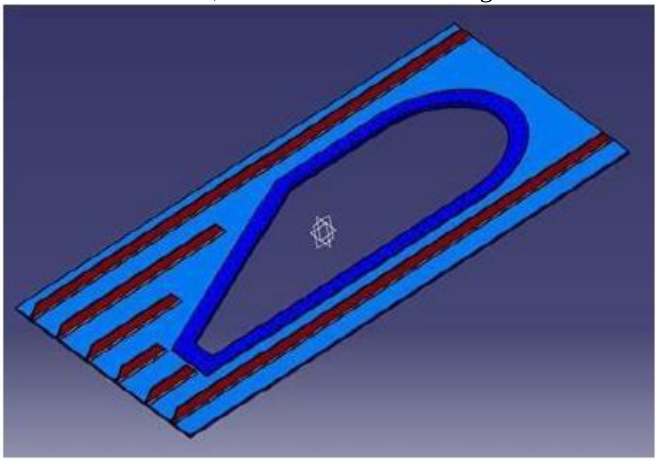


Fig.2. CATIA V5 model of landing gear cutout.

VI. FINITE ELEMENT ANALYSIS.

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. Calculation of stress intensity factor using MVCCI method is done for different crack length(starting from 0.66mm) with is tabulated below:

Table 1 SIF calculated for different crack length.

Crack length 'a' from hole edge in mm	Thic kness 't' in mm	Element length c in mm	Node displacement in mm			Force 'F' in N			Strain energy release rate G in N/m	SIF 'K' in MPa
			1	2		F_1	F_2	F		
0.66	5	0.668	1.28	1.26	0.02	528.3	689.3	1217.6	3051.6	14.5
1	5	0.668	1.28	1.26	0.02	658.7	790.9	1449.6	3943.1	16.5
3	5	0.913	1.28	1.25	0.03	856.6	1152.6	2009.2	5677.9	19.7
5.7	5	0.913	1.28	1.25	0.03	1056.6	1432.9	2489.6	8748.4	24.5
9	5	0.913	1.27	1.23	0.04	1325.3	1847.7	3172.9	14417.5	31.5
11	5	0.913	1.28	1.23	0.05	1601.2	2253.2	3854.4	21439.7	38.4
13	5	1.028	1.29	1.21	0.08	2299.4	3635.2	5934.6	46135.6	56.3
18	5	1.028	1.24	1.20	0.04	1736.6	4738.4	6475.1	27093.3	43.1
22	5	1.028	1.29	1.19	0.1	1185.2	5955.8	7141.1	70971.6	69.8
26	5	0.779	1.27	1.18	0.09	2849.1	2573.2	5422.4	67224.5	67.9
29	5	0.785	1.25	1.18	0.07	1010.8	2589.8	3600.6	34908.9	48.9
32	5	1.037	1.29	1.19	0.1	5122.2	1854.6	6976.8	65700.3	67.2
37.5	5	1.037	1.30	1.19	0.11	4741.5	3537.9	8279.5	90234.4	78.7
41.7	5	1.037	1.32	1.19	0.13	4146.9	5141.7	9288.7	113845.3	88.4
43.7	3	0.996	1.33	1.19	0.14	2980.9	3819.1	6800.1	155201.8	103.2
47.7	3	0.996	1.34	1.20	0.14	2753.1	3679.6	6432.7	153841.7	102.8
52.7	3	0.996	1.35	1.21	0.14	2761.6	3648.5	6410.1	152741.8	102.4
57.7	3	0.996	1.36	1.22	0.14	2806.3	3660	6466.3	154373.5	102.9
62.7	3	0.996	1.38	1.23	0.15	2861.3	3682	6543.3	159051.4	104.5
67.7	3	0.996	1.39	1.25	0.14	2918.9	3707.9	6626.8	163134.2	105.8
72.6	3	0.996	1.41	1.26	0.15	2980.5	3729.1	6709.6	167652.6	107.2
77.6	3	0.996	1.43	1.28	0.15	3032.7	3752.4	6785.1	171183	108.4
82.6	3	0.996	1.45	1.29	0.16	3083.2	3769.7	6852.9	174642.7	109.5
87.5	3	0.996	1.47	1.32	0.15	3128.1	3780.2	6908.2	177495.4	110.4
92.5	3	0.996	1.49	1.34	0.15	3168.5	3778.1	6946.6	179515.1	111
97.5	3	0.996	1.52	1.36	0.16	3193.9	3768.8	6962.7	180325.6	111.3
101.5	3	0.996	1.54	1.38	0.16	3203.9	3753.6	6957.6	180160.7	111.2
106.5	3	0.996	1.56	1.41	0.15	3202.7	3714.2	6916.9	178124.9	110.6
111.5	3	0.996	1.59	1.43	0.16	3179.6	3643.7	6823.3	173445.3	109.1

From the table we can see that with increase in crack length SIF first increases and then starts decreasing after a critical value.

VII. 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 2.

Table 2: block loading.

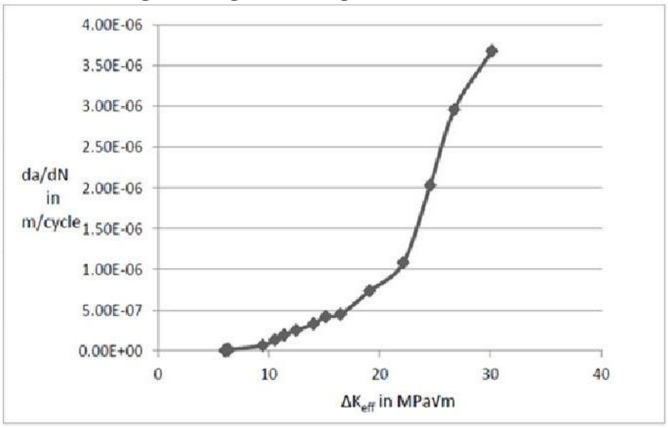
Number of cycles	g loads
1	0.5 to 0.75
3	0.75 to 1.0
1	1.0 to 1.25
6	1.25 to 1.5
5	0 to 1.75
1	0 to 2.0
1	-0.5 to 1.5

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. Fatigue crack growth material data curve is tabulated in table below.

Table 3: Fatigue crack growth material data.

cycle	da/dn(m/cycle)	ΔK_{eff}
30000	2.04E-09	6.032
100000	6.13E-09	6.154
200000	1.32E-07	10.547
240000	4.48E-07	16.471
250000	3.68E-06	30.137

Fig 3: Fatigue crack growth data curve.



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

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,

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

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

Similarly the calculation da /dN for each crack lengths for different 'g'. Similarly we calculate da /dN for a crack length of 9 mm and 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**

VIII. VALIDATION OF RESULT.

The Stress Intensity Factor (SIF) calculated by FEA approach using MSC NASTRAN / PATRAN solver using MVCCI method is verified using theoretical values of SIF using theoretical formulas. The comparison is obtained and tabulated in table 5 and graph is plotted which validates almost equal values in both the cases.

Table 5: Comparison of theoretical and analytical value of Stress Intensity Factor (SIF).

Crack length '2a' (mm)	Theoretical SIF K_{1c} $MPa(m)^{1/2}$	Analytical SIF K in $MPa(m)^{1/2}$
20	4.35	4.36
60	7.61	7.69
100	10.05	10.20
140	12.33	12.56
180	14.78	15.07
200	16.16	16.47

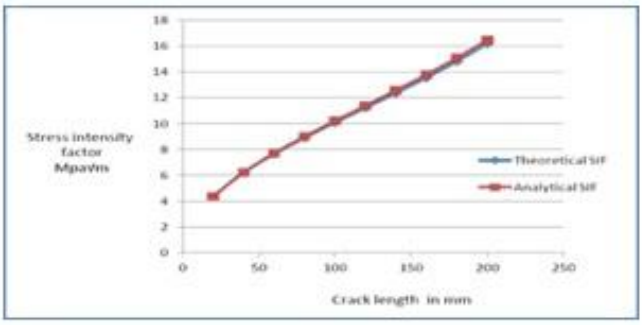


Figure:4 Comparison graph of SIF With crack length.

IX. CONCLUSION.

The main objective of this study is to predict the life of fatigue crack growth. The prediction is necessary because you always have to know when the crack in the structure becomes critical. Here in this study stress analysis of a stiffened panel is performed in MSC NASTRAN / PATRAN software. In performing the stress analysis we found that the maximum tensile stresses acting on the stiffened panel is in the location of the rivet hole on the panel. And cracks always start from the location of maximum tensile stress.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. Predicting fatigue crack growth is by simplifying the variable load range of the load block plane A block represents 100 flight hours. And then the number of blocks required for the crack to grow critical is calculated. Critical crack length is obtained by knowing the fracture toughness of the material. Because when the stress intensity factor at the crack tip is made equal to the resistance to fracture of the structure fails. Once the total number of blocks to be critical crack is known, it is the number of flight hours of the aircraft to fly safe. Thinking that we find that the aircraft can fly for 752000 numbers of hours after the aircraft fail . Once the total number of blocks to be critical crack is known, it is the number of flight hours of the aircraft to fly safe. we find that the aircraft can fly for 752000 numbers of hours after the aircraft fail .

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