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Simulation of Wing-Fuselage Attachment Bracket Lug for Fighter Aircraft

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The typical fighter aircraft undergoes complicated maneuvering during its flight. As a result, the aircraft will require immediate change in acceleration. The combination of high level of acceleration and complicated maneuvers will introduce high magnitude of loads on the wings. Wings attached to the fuselage structure through wing fuselage attachment brackets lug. The loads are transferred from the wing to the fuselage through the attachment brackets lug.

For the continued airworthiness of an aircraft during its entire service life, static analysis together with fatigue and damage tolerance analysis plays a vital role. The objective is to validate the wing-fuselage attachment bracket from static, fatigue and damage tolerance point of view and estimate the inspection procedure/maintenance of the Wing-fuselage attachment bracket lug.

For the validation of wing-fuselage attachment bracket, initially, the static load carrying capability of the wing-fuselage attachment bracket is calculated. Analysis for the strict safety experiments are carried out to check the structural integrity before usage. It is to ensure the static load carrying capability of the wing-fuselage attachment bracket.

The fatigue analysis is carried out to determine the fatigue life of the part before initiation of a potential crack. The DT analysis is carried out to determine the crack propagation life for the part using the finite element method and Modified Virtual Crack Closure Integral (MVCCI) method.

Keywords *- Modified Virtual Closure Integral (MVCCI) Method, F&DT, Stress Intensity Factor (SIF), Crack Propagation and Fatigue Life.*

1. INTRODUCTION

Lugs are the main part of the primary structural in airframe structure that are vitally used in connecting different components of the airframe such as aircraft engine-pylon support fittings, wing, fuselage attachment, wing to control surfaces and landing gear links etc. are some of the typical applications where attachment lugs of various configurations can be found to achieve the required design intent.

The catastrophic failure is occurred to the lug joint bracket may lead to the separation of the aircraft structure. Therefore, Finite element analysis, observational and numerical data helps the architect to plan a safe life structure from catastrophic failure. Aircraft attachment bracket lugs are most critical components in terms of fracture which are attached to the aircraft wing fuselage structure and the consequences of structural lug failure can be very disastrous that the fuselage and wings of an aircraft get separated and leads to failure or crash of the aircraft.

Therefore, it is necessary to establish design criteria and analysis methods to ensure the damage tolerance of aircraft attachment lugs.

Literature survey shows that numerous papers have been found that dealt with Lug connections with various different methods have been applied for crack propagation study of various components. The MVCCI method offers an advantage over other method, as it is dealing with a static analysis method to find the fatigue life through various crack lengths. However, a limited number of papers have been found that dealt with lug analysis of wing fuselage attachment bracket, with even fewer studies employing MVCCI method to study the crack propagation. Therefore, it took the author to involve in this method.

1.1 Modified Virtual Crack Closure Integral (MVCCI)

The MVCCI method makes use of reaction force at crack tip node & displacements of nodes adjacent to crack tip node. A typical finite element mesh in the vicinity of the crack-tip is as shown in Fig 1.

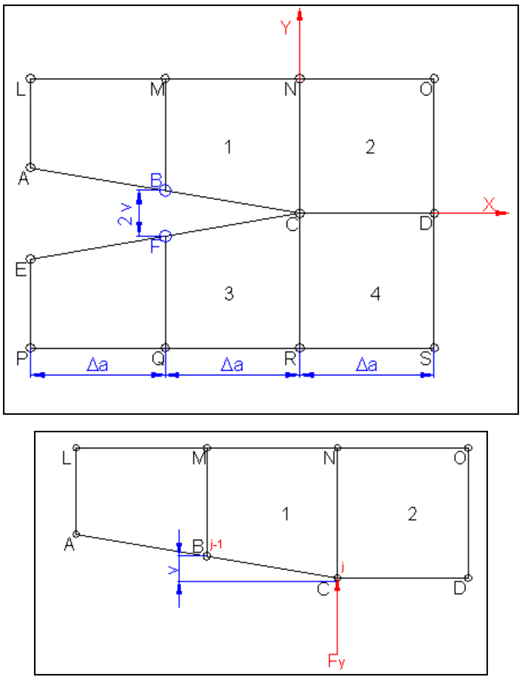

Fig 1: MVCCI Model Showing Displacement Adjacent to Crack Tip and Reaction Force at Crack Tip.

B and F are the first set of nodes on the crack face adjacent to the crack tip C. Fy is the reaction force along y-axis at the node 'C' exerted by the bottom part of the model.

So for Mode I, SERR is expressed as

$$\boxed{G_I = \left(\frac{1}{2\Delta a}\right)\left[\frac{F_y}{t}(2v)\right]} \qquad (1.1)$$

Where, Δa = Crack Increment (distance between crack tip node and adjacent node).

F_y = Reaction force acting at crack tip node.

t = Thickness of plate.

v = displacement of node (B) adjacent to crack tip node (C) along Y direction.

The value of F_y is the crack opening force, shown in Fig 1. Based on the principle described by calculating G_I from MVCCI it is possible to calculate the SIF. The crack tip reaction force F_y is obtained by post processing FEA results.

Irwin shown that the energy release rate, G, is uniquely related to the stress intensity factor as follows:

$$\boxed{G = \frac{K_I^{\,2}}{E}} \quad \text{(Plane stress)} \quad (1.2)$$

$$\boxed{G = \frac{K_I^{\,2}}{E(1-\upsilon^2)}} \quad \text{(Plane strain)} \quad (1.3)$$

1.1.1 Advantages of MVCCI Technique

1. The energy release rates are calculated directly using crack opening displacement and nodal forces and at ahead of crack-tip and hence does not require the computation of the element stresses and singular stress field in the vicinity of the crack-tip.
2. It permits separation of G into its components, G_I and G_{II} associated with Mode I (opening) and Mode II (in plane sliding) deformations respectively in a mixed mode crack problem.
3. It gives higher accuracy even with a coarse mesh of conventional elements. Both lower and higher order elements can be used.
4. The method requires only one analysis per crack length and can be applied to both isotropic as well as laminated anisotropic materials.
5. The elements available in general purpose finite element programs can be used without any modification.

MVCCI Technique is used throughout the present work for estimation of SIF.

1.2 Geometry Configuration

The wing-fuselage attachment bracket considered in the study is shown in Figures 1.2 to 1.3 in different views which is modeled in CATIA V5.

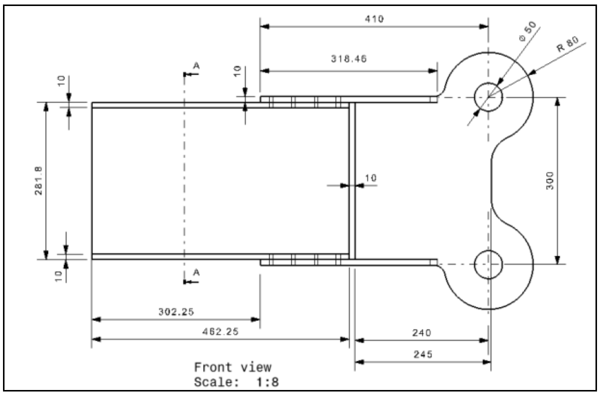

Fig 2: Front View of Assembly.

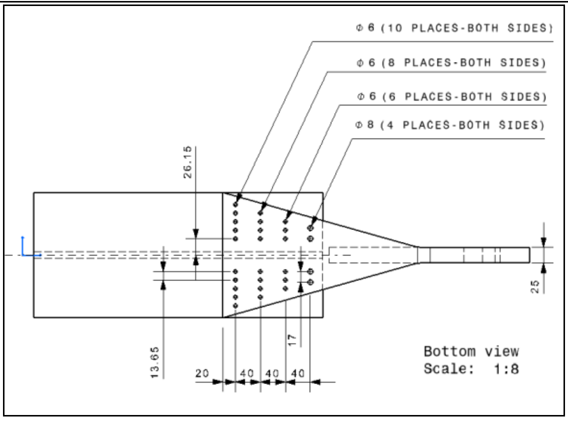

Fig 3: Top View of Assembly.

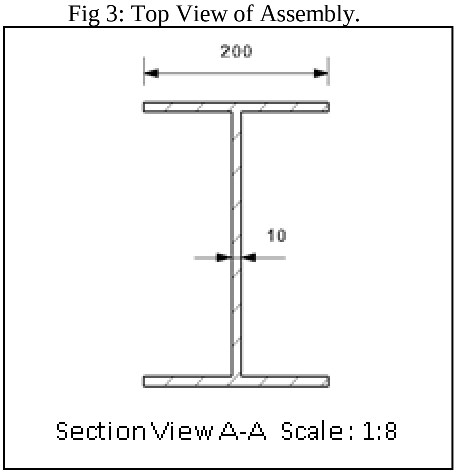

Fig 4: Side view of I – Spar.

1.3 Material Specification

The different structure members in Wing Fuselage attachment bracket lug are

1. Lug - Steel AISI-4340
2. I-Spar -- Aluminum 7075-T6:7075-T651)
3. Top and bottom flanges of lug attachment bracket -- -Aluminum 7075-T6:7075-T651
4. Rivets -Aluminum 7075-T6:7075-T651

Properties of Alloy Steel, Heat Treated AISI-4340:

- Young's Modulus. $E = 211000 N/mm^2$
- Poison's Ratio, $\mu = 0.3$
- Ultimate Strength, UTS. = $2200 N/mm^2$
- Density, $\rho = 7.85 g/cm^3$
- Fracture toughness KIc = 89 $MPa.m^{1/2}$

Properties of Aluminum 7075-T6; 7075-T651:

- Young's Modulus. $E = 71700 N/mm^2$
- Poison's Ratio, $\mu = 0.33$
- Ultimate Strength, UTS. = $572 N/mm^2$
- Density, $\rho = 2.81 g/cm^3$
- Fracture toughness KIc = 29 $MPa.m^{1/2}$

1.4 Finite Element Model

The finite element model of the wing fuselage attachment bracket was developed in HYPERMESH 12.0. It has 19762 elements, 20362nodes and 56 MPC's. A finite element mesh for complete assembly is shown in Figure 5 and RBE2 is used to distribute the loads equally on attached members.

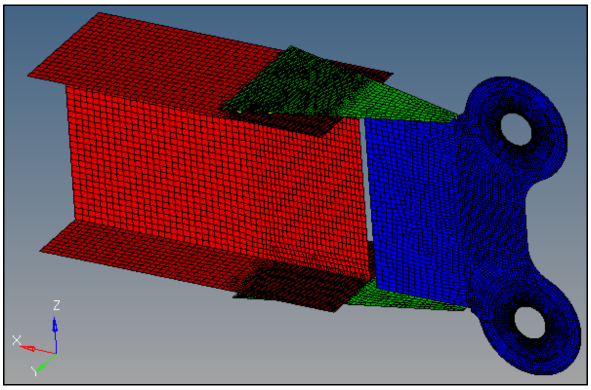
Fig 5: 2D FE Model of the Wing Fuselage Attachment Bracket.

1.5 Application of Loads & Boundary Conditions

The application of loads and boundary conditions along with the finite element model are shown in the below figure 6. A load of 90584.62N [refer ref. (3)] is equally introduced to all nodes at one end of the spar beam. This load will essentially create the required bending moment at the root. Since, it is a lift load; the loads are applied in the Z direction as shown in figure 6. The both lug holes of the wing fuselage attachment bracket Lugs are constrained with all six degrees of freedom (u, v, w, θ_x, θ_y & θ_w) at the semicircular circumferential region. Therefore, that it will act as a cantilever beam.

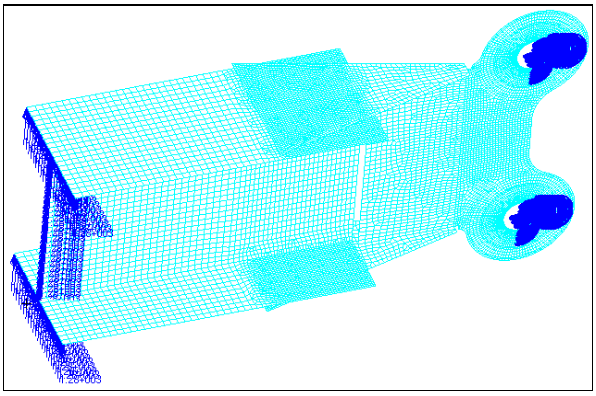
Fig 6: Loads and Boundary Conditions.

2. ANALYSIS

2.1 Linear Static Analysis

The linear static analysis solution 101 is done in the MSC.NASTRAN solver. And the post processing will be done in the MSC.PATRAN. The following results are observed in the post processing of the bracket assembly.

A maximum stress of 1220N/mm^2 is observed at the Lug section of the bracket as shown in fig 7 and the maximum displacement of 7.13mm at the free end of the cantilever structure can be observed from the displacement contour as shown in the fig 8. The maximum stress value obtained from the static analysis is used as the input for the fatigue calculations.

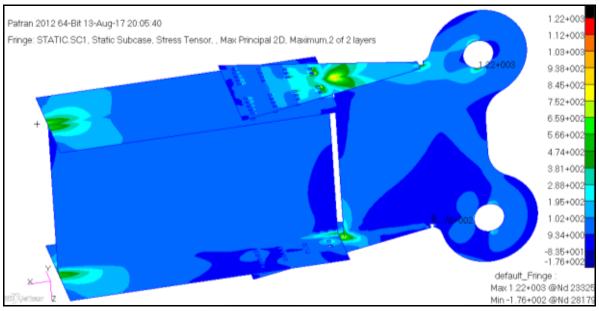

Fig 7: Stress Contour.

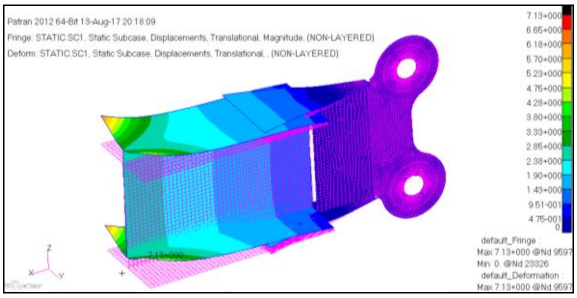

Fig 8: Deformed Shape Contour.

	$\sigma_{max.}$ (N/mm^2)	$\Delta_{Max.}$ (mm)	TUS (MPa)	ρ (Kg/m^3)
I-Spar	496	7.13	572	2810
Lug	1220	1.67	2200	7850

Table 1: Static Stress Analysis Result.

2.2 Crack Propagation Analysis

Assuming an initial crack of 1.242mm in the simplified lug plate where the highest stress concentration found in the bracket, the crack propagation study has been carried out to find out the crack growth rate through stress intensity factor approach. The locally calculated nodal load of 4035.207N is applied for all the 56 nodes of the semicircle portion of the lug in the longitudinal direction, such a way that to propagate the crack in the transverse direction. In addition, edge of the plate is constrained in all the six degrees of freedom.

There are several iterations done to study the crack propagation and the results are shown in table 2.

Iteration	‘a’ (mm)	COD Δu (mm)	‘F_y’ (N)	‘G’ (N/mm)	SIF (Mpa. $\sqrt{m}$)
1	1.242	0.0216	20606	5.94	35.33
2	2.507	0.0377	17102	7.89	40.69
3	3.696	0.0516	26188	8.02	41.04
4	4.909	0.0635	28734	8.41	42.02
5	7.440	0.0842	32525	9.82	45.42
6	10.006	0.1009	43427	10.52	47.00
7	13.856	0.1604	52138	11.92	50.03
8	16.424	0.1925	51487	12.94	52.14
9	18.990	0.2531	55209	14.54	55.26
10	21.559	0.6412	54543	16.02	58.00
11	24.127	0.9535	26929	16.63	59.10
12	26.699	1.2568	21040	18.28	61.96
13	29.267	2.5643	19415	38.79	90.25
14	31.835	2.6483	22134	49.79	102.25
15	34.404	2.7589	29926	56.84	109.25
16	36.973	2.8946	35438	59.84	112.10
17	39.541	2.9785	38214	70.89	122.01

Table 2: ‘G’ & SIF Results of Various Iterations.

2.2.1 Comparison of Results

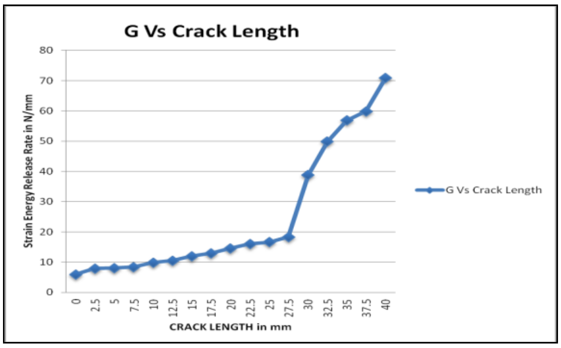

Fig 9: ‘G’ versus Crack length.

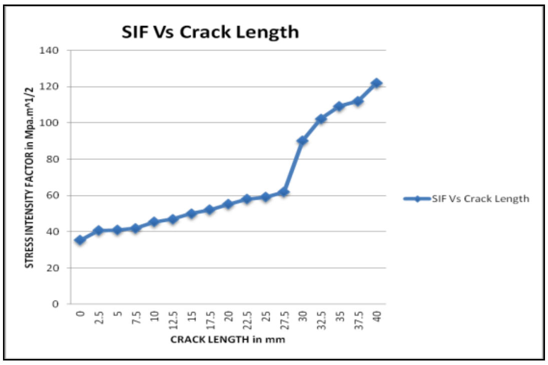

Fig 10: SIF versus ‘a’.

2.3 Analysis of Life for the Fatigue Crack Propagation of Lug Plate

To predict the fatigue life of the crack structural parts “Paris’s Law’ is more widely used in the Fracture Mechanics Field which is given by below equation;

$$\left(\frac{da}{dN}\right) = C\Delta K^{m} \quad (2.1)$$

Elber's law [refer ref. (8)] is based on crack closure concept which is given as below;

$$\left(\frac{da}{dN}\right) = C\Delta K_{eff}^{m} \quad (2.2)$$

To find the crack life of the structural parts simplified equation of Elber's law is given as below;

Numbers of cycle after crack Initiation

$N_i = 1 / \{C_{eff} \times (A + B \times R) \times (SIF)_i)^{m}\}$ (2.3)

Total numbers of cycle $N = \sum (N_{i+1} + N_i)$ (2.4)

Life estimation of Crack lug plate is performed with the help of crack propagation data table 2 and Elber's law is given in table 3,

For stress ratio R =0.10

Ceff = 68.3×10^{-9}

A = 20.67

B = 0.33

m = 2.5

K_{IC} =89Mpa $\sqrt{m}$

Note:

1.Crack detection during aircraft maintenance/ inspection.

2. In order to avoid the catastrophic failure of the aircraft, the maintenance and inspection programs are defined for each aircraft.

3. During these maintenance and inspection programs various techniques are used for early detection of the potential cracks in various parts.

4. Some of such techniques are listed below

HFEC High Frequency Eddy Current Method

LFEC Low Frequency Eddy Current Method

XRAY XRAY difraction method

US Ultra Sonic Method

ROTOTEST Roto test method.

SIF (Mpa $\sqrt{m}$)	dn	N
35.33	4762	4762
40.69	3346	8108
41.04	3275	11383
42.02	3087	14470
45.42	2541	17011
47.00	2333	19344
50.03	1996	21340
52.14	1800	23140
55.26	1557	24697
58.00	1379	26076
59.10	1316	27392
61.96	1169	28561
90.25	457	29018
102.25	334	29352
109.25	283	29635
112.10	266	29901
122.01	215	30116

Table 3: Crack Propagation Life.

Assuming HFEC method for crack detection, from HFEC method a minimum crack length of 2.00mm can be detected. So, the Detectable Crack Length (a_DET) is assumed as 2.00mm and the life corresponding to a_DET is determined (N_DET).

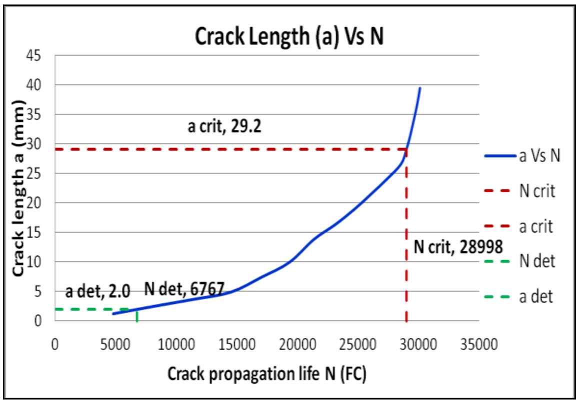

Fig 11: 'a' Vs Crack Propagation Life.

Therefore, from above graph, it is evident that the repair is required before or on reaching the detectable number of cycles.

The life from N_DET to N_CRIT is the life available for repair of the potential cracks which may otherwise grow to critical crack length and there by leading to catastrophic failure of the part.

3. CONCLUSION

Wing-fuselage attachment bracket lug is examined for static and fatigue strength. Further, the crack propagation analysis of the wing-fuselage attachment lug is done through various iterations.

Once the material reaches the mentioned fracture toughness (89Mpa$\sqrt{m}$) then the material degradation begins. This 1.242mm initial crack starts propagates once we apply the loads and boundary conditions. Due to this applied load the crack length is getting increased correspondingly SERR 'G' is also getting increased (refers Fig 9:). Moreover, once the crack length is reached 29mm then the SERR 'G' is increasing drastically. So then the crack propagates very rapidly.

Similarly, when the crack length is increasing then the stress intensity factor also gets increased. Then finally once the crack length reaches the critical value then the stress intensity factor increases very rapidly (refers Fig 10). When the crack length reaches the value of 29.267mm then the stress intensity factor observed is 90.25Mpa$\sqrt{m}$. This is called critical SIF and is directly proportional to the fracture toughness. So once the material reaches the value of SIF then the material leads to fracture.

At the iteration 14 when the crack length becomes near to 31mm then we reached the SIF more than the material fracture toughness value. Though we reached the SIF value more than the fracture toughness we further continued the analysis for the crack length of more than 29mm to know the behavior of the lug plate. It clearly shows that, once the SIF reaches more than the material fracture toughness then the crack propagation will be more severe.

So the above study shows that the behavior of the crack propagation for the lug attachment bracket due to the lift load. According to the current study, the lug plate considered for the analysis is safe for the crack length up to 29mm, beyond which it leads to fracture. Results the detachment of the aircraft wings from the fuselage. It leads to the catastrophic failure. Concluded that the major loss for the aircraft as well for the passengers.

From section 2.3, an inspection procedure /maintenance is required before or on the reaching the defined number of 6767 cycle (refers Fig 11) to ensure that the fight will fly without any failure.

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