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Structural Analysis and Material Optimization of Aircraft Fuselage

Mr. Vasudev H[a], Mr. Rangaswamy H[b], Mr. Imran Ali M R[c]

[a]*PG Scholar, Machine Design, S.I.E.T, Tumkur, India.*
[b]*Asst Prof, Machine Design, S.I.E.T, Tumkur, India.*
[c]*Asst Prof, Machine Design, H.M.S.I.T, Tumkur, India.*

ABSTRACT

Presently the Aircraft industries are normally using the fuselage skin made of Aluminum alloy with supporting structural members along longitudinal and circumferential directions. Since fuselage is meant for transportation, hence the strength and safety is the main priority. The fuselage has to be designed in such a way that it should have greater strength and should be light weight. This research work basically focuses on reducing the weight and to increase the strength of the fuselage. In this thesis Aluminum alloy is replaced with Composite materials. The analysis performed for Aluminum alloy and Composite materials using ANSYS 15.0. In this paper the structural behavior and fracture characteristics of the fuselage has been studied. Finally the conclusion has been made that the Aluminum alloy can be replaced with Composite material without affecting the strength of the fuselage.

Keywords *- Displacement, Stress, Crack, Stress Intensity Factor.*

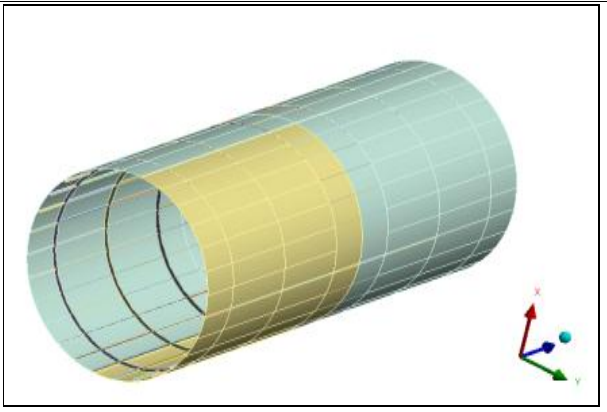
1. INTRODUCTION

Fuselage is an abbreviation derived from the French word fusel (shape of the spindle) is an aircraft main body structure designed for accommodating the crew, passengers, and cargo. The fuselage structure is hollow to reduce weight. Similar to other parts of the aircraft, the shape of fuselage is normally determined by the area of application of the aircraft. It may vary in design and size according to its function and the area of application. . The weight of an aircraft is distributed all along the fuselage structure. The fuselage, along with the crew, passengers and cargo, may contribute a significant portion of the weight of an aircraft. The center of gravity of the aircraft is an average location of the weight and it is usually located inside the fuselage. The fuselage must be designed with enough strength to withstand the torque generated during takeoff, flight and landing. M. Pacchione, J. Telgkamp [1] they investigated about the challenges of the metallic fuselage. W.J.Vankan, B.A.T. Noordman, R. Maas [3] made modeling, analysis and optimization of composite aircraft fuselage structures. Ilhan Sen. [6] investigated aircraft fuselage design study for composite structures. Muniyasamy Kalanchiam and Baskar Mannai [7] investigated the topology optimization of the aircraft fuselage structure. Prof Dr. Ing Wilhelm Rusti and Dipl-Ing M. Kracht [10] they investigated recent experiences in load analysis of aircraft fuselage panels. Padraic. E, O Donoghue, Jinsan Ju [12] they investigated the experimental/numerical techniques for aircraft fuselage structure containing damage. Karthik N, Dr. C Anil Kumar [13] they made investigations on the analysis of the fuselage structure for multi-site damage.

2. RESULTS AND DISCUSSIONS

2.1 Iteration 1: Geometry of Fuselage made of fully Aluminum Alloy

The geometry considered for the analysis is shown below fig 2.1. The analysis is carried out using 2D shell meshing.



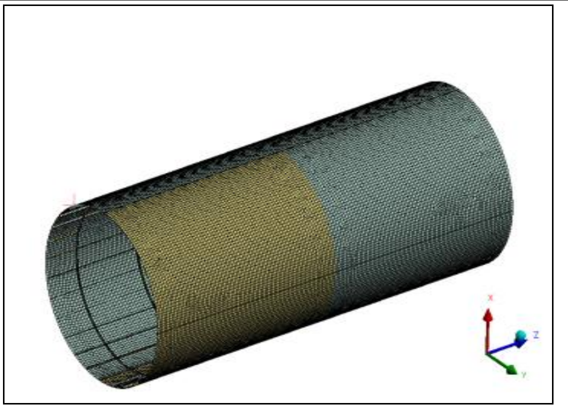


Fig 2.1: Geometry of Fuselage

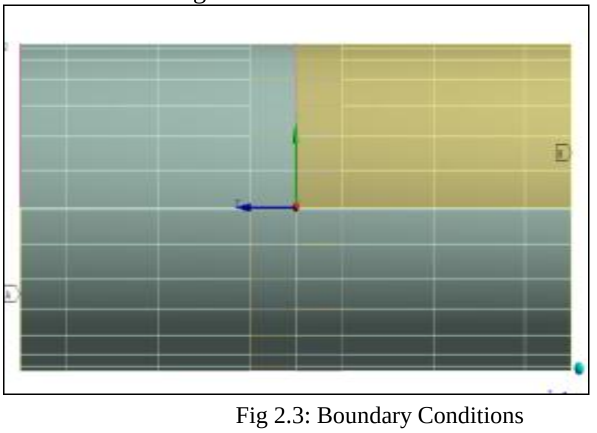
Fig 2.2: FE Model

2.2 FE Model

Geometry was imported to ANSYS and FE model was created as shown in above fig 2.2. The 2D Shell 181 element is used for the analysis.

2.3 Boundary and Loading Conditions

The end of the fuselage is constrained for all 3 translations DOF and all 3 rotational DOF are left free, as shown below in fig 2.3.



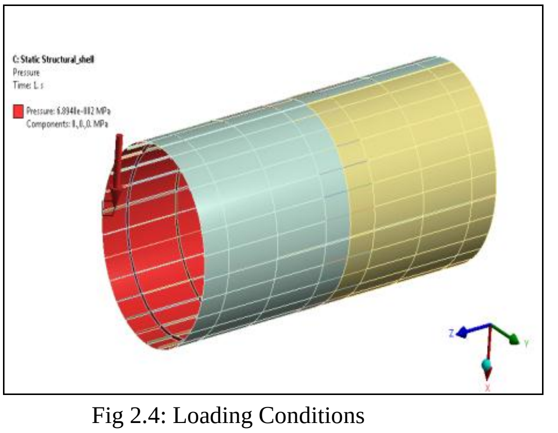


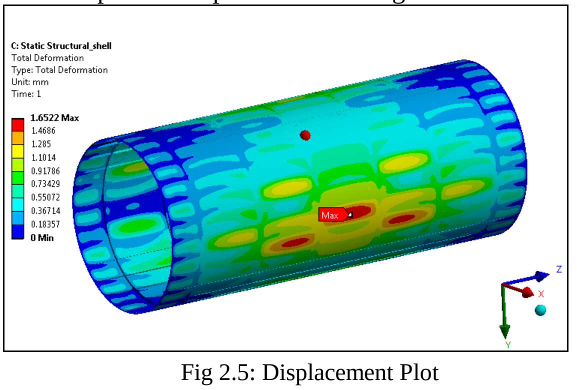
Fig 2.3: Boundary Conditions

Fig 2.4: Loading Conditions

Considering the average flying altitude as 30,000 feet (Max) where the Atmospheric pressure will be around 4.4 psi and atmospheric pressure at sea level will be around 14.7 psi. The pressure variation on the fuselage ranges from 14.7 psi to 4.4 psi. The average the differential pressure on the fuselage cabin is taken as 10 psi (0.0689Mpa). This is shown in above fig 2.4.

2.4 Displacement Plot

The displacement plot of the fuselage is shown below in fig 2.5. The max displacement is 1.65 mm.



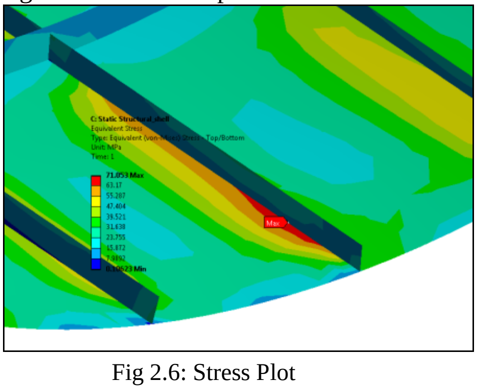


Fig 2.5: Displacement Plot

Fig 2.6: Stress Plot

2.5 Stress Plot

The stress plot for skin of the fuselage is shown above & for frames & stringers are shown below

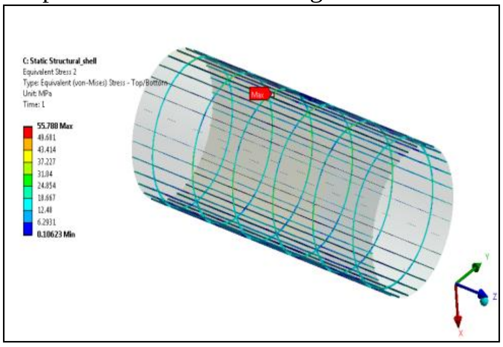


Fig 2.7: Stress plot for frames

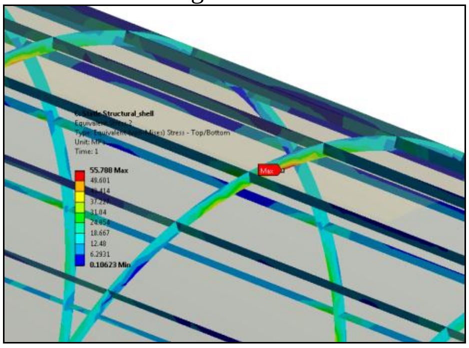


Fig 2.8: Stress plot for stringers

2.6 Geometry – Crack Modeling of Fuselage made of Aluminum Alloy

The stress intensity factor (SIF) for the fuselage is calculated using Solid Model approach. The max stress location from the analysis is identified and that location is considered for the fracture toughness evaluation. The geometry considered for the analysis is shown below in fig 2.9.

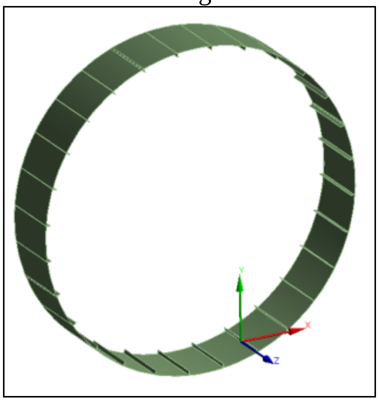


Fig 2.9: geometry of fuselage with crack modeling

The crack is modeled in model where max stress is observed, hence crack is defined at this point and local coordinate is created to crack path definition.

2.7 Analysis set up for Fracture toughness

The analysis setup for the fracture toughness calculation is shown below.

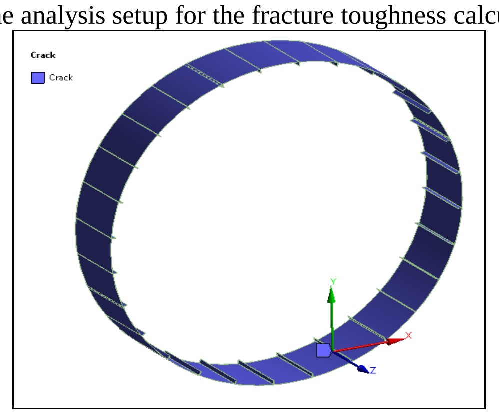


Fig 2.10: Analysis for fracture toughness

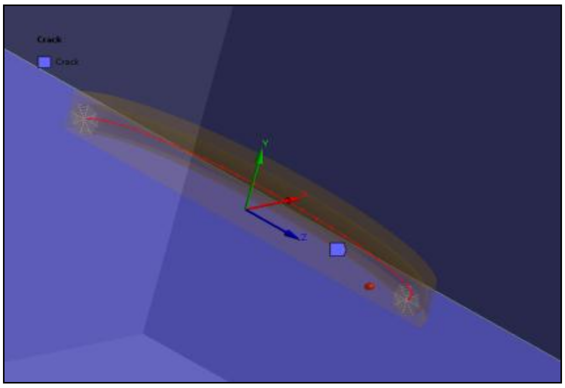


Fig 2.11: View of Crack

2.8 Stress intensity factor for crack length of 1mm

The SIF for the crack length of 1mm is shown below. The max SIF is 28 Mpa.mm^{½}

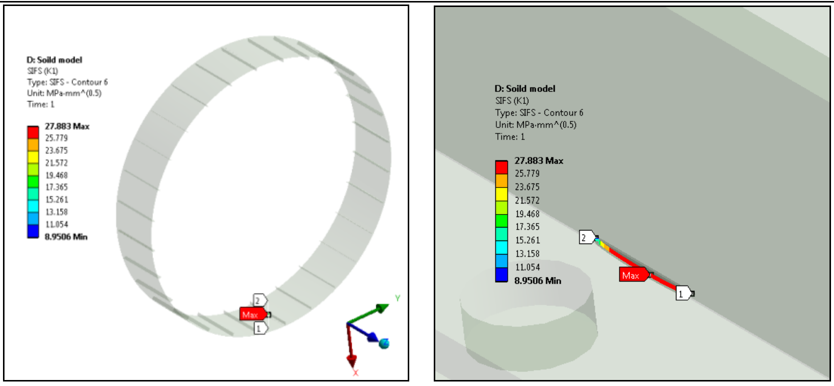


Fig 2.12: SIF for crack length of 1mm.

2.9 Stress plot for crack length of 1mm

The stress plot of the fuselage is shown below. The max stress is 285Mpa at the crack region due to discontinuity (crack growth).

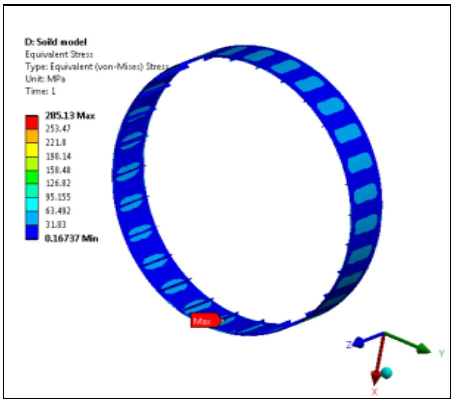


Fig 2.13: Stress plot for crack length of 1mm

2.10 Optimization

The present trend in aero field is making use of light weight material to reduce the overall weight of the vehicle without compromising in strength and durability of the vehicle. To achieve these mostly they use alloys or composite materials as much as possible. In general the fuselage is made of Al alloys and composites material. In this work, Finite element analysis is performed on the Al alloy fuselage and then analysis is carry forwarded with polymer composite material (PMC). PMC material are playing crucial role in automotive field. In this work, selected PMC material is, Carbon Epoxy PMC. Also we studied the effect of lamina orientation on strength and fracture toughness of the fuselage.

2.11 Iteration-2: Geometry of fuselage

The Fuselage skin is made of Carbon epoxy composite material. The ribs are made of Aluminum alloy as shown below.

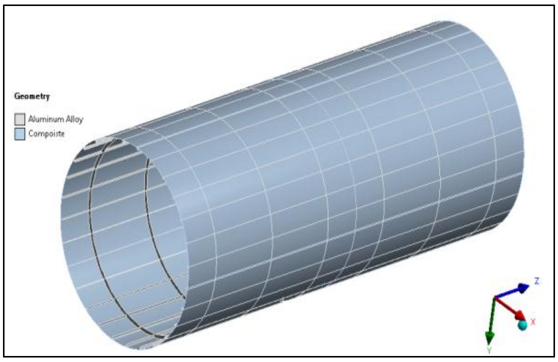


Fig 2.14: Geometry of fuselage for Iteration-2

2.12 Displacement Plot

The displacement plot of the fuselage is shown below. The max displacement is found to be 0.702 mm.

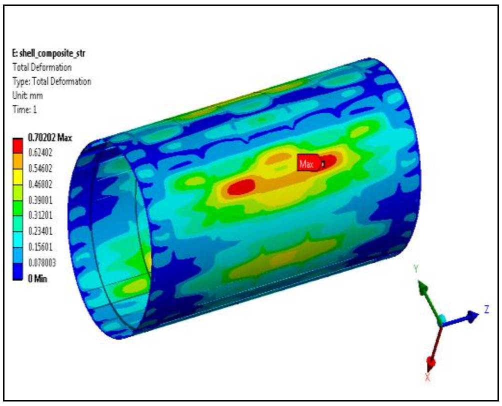


Fig 2.15: Displacement plot for iteration-2

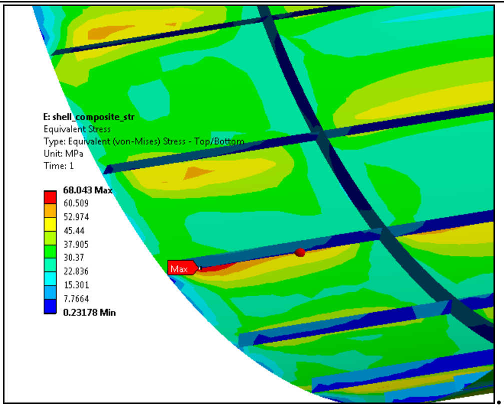


Fig 2.16: Stress plot for iteration-2

2.13 Stress Plot

The stress plot of the fuselage skin is shown above in fig 2.17. The max stress is 68Mpa. The stress plot of the Frames and Stringers is shown below. The max stress is 41Mpa.

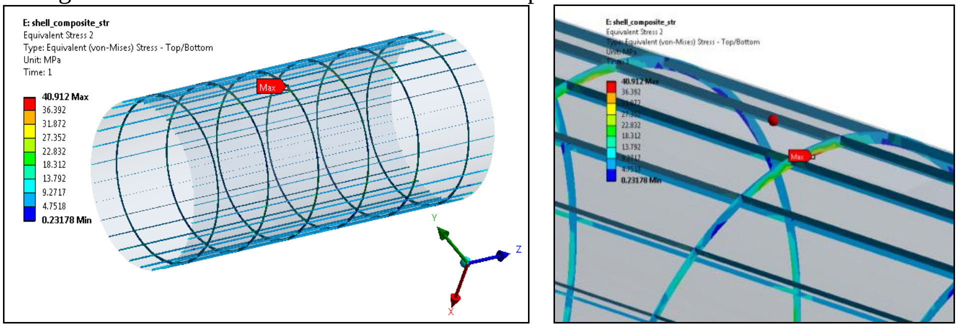


Fig 2.17: Stress plot of Fuselage Frames and Stringers for Iteration-2

2.14 Stress Intensity Factor for Crack Length of 1mm for Iteration-2

The SIF for the crack length of 1mm is shown below. The max SIF is 28 Mpa. mm^{½}.

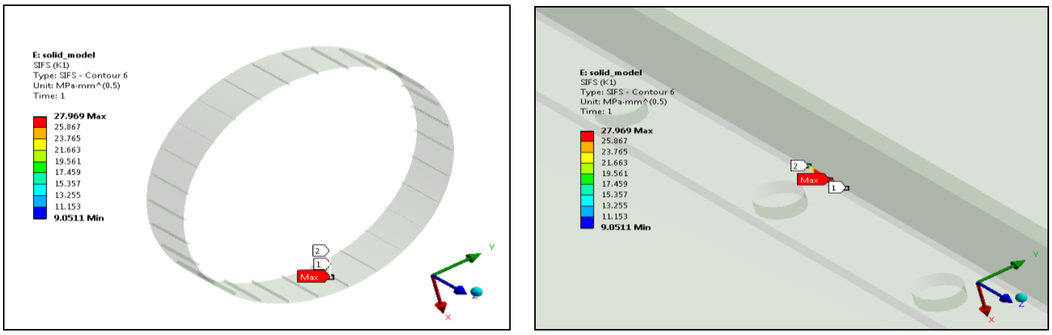


Fig 2.18: SIF of Iteration 2 for crack Length of 1mm

2.15 Stress Plot for Crack Length of 1mm for Iteration-2

The stress plot of the fuselage is shown below. The max stress is 414Mpa at the crack region due to discontinuity (crack growth).

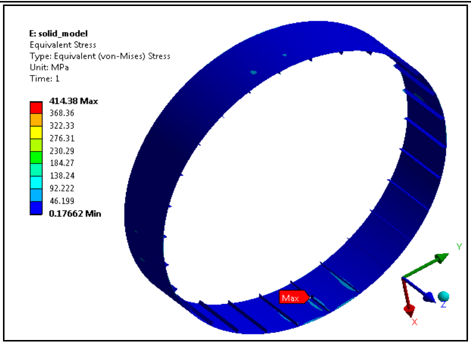


Fig 2.19: Stress Plot for crack length of 1mm for iteration-2

2.16 Iteration-3: Fuselage with Carbon Epoxy resin

The Fuselage skin is made of Carbon epoxy composite material. The ribs are made of Aluminum alloy as shown below. This iteration is similar to iteration 2 but only the material properties of the composites are increased in iteration-3.

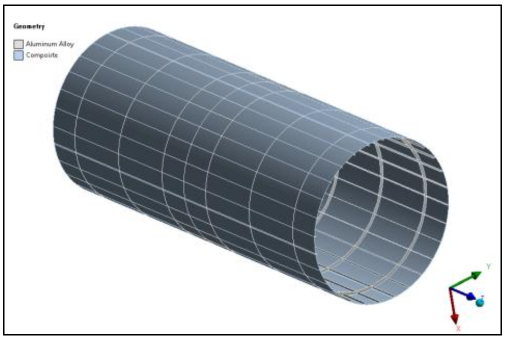


Fig 2.20: Geometry of fuselage for iteration-3

2.17 Displacement Plot

The displacement plot of the fuselage is shown below. The max displacement is 0.531mm.

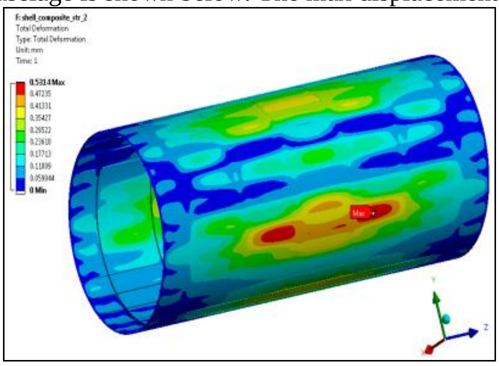


Fig 2.21: Displacement plot for iteration-3 of composite

2.18 Stress plot

The stress plot of the fuselage skin is shown below. The max stress is 66Mpa.

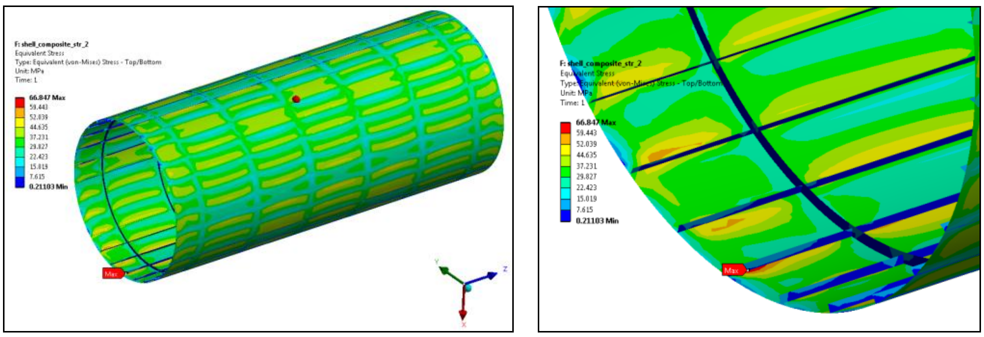


Fig 2.22: Stress plot for iteration-3 of composite

The stress plot of the Frames and Stringers is shown below. The max stress is 35Mpa.

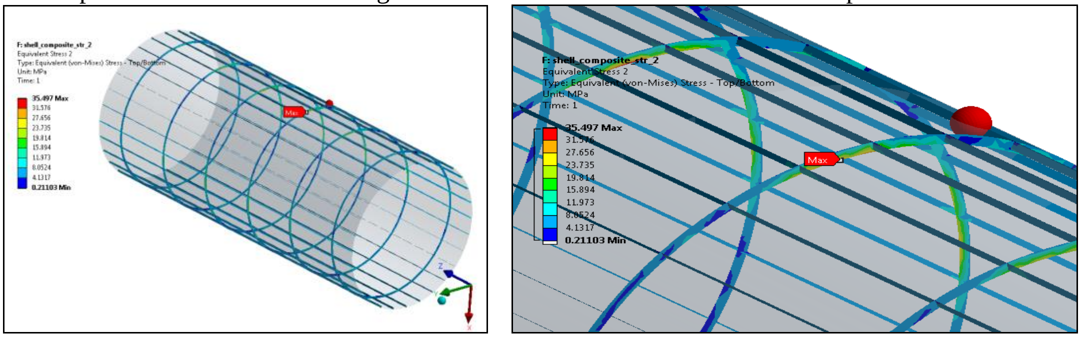


Fig 2.23: Stress plot of Fuselage Frame and Stringers for Iteration-3

2.19 Stress Intensity Factor for Crack Length of 1mm for Iteration-3

The SIF for the crack length of 1mm is shown below. The max SIF is 29Mpa. $\text{mm}^{1/2}$.

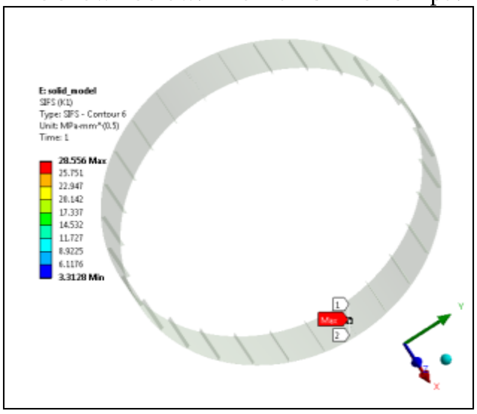


Fig 2.24: SIF of Iteration 3 for crack Length of 1mm

2.20 Stress Plot for Crack Length of 1mm for Iteration-3

The stress plot of the fuselage is shown below. The max stress is 369Mpa at the crack region due to discontinuity (crack growth)

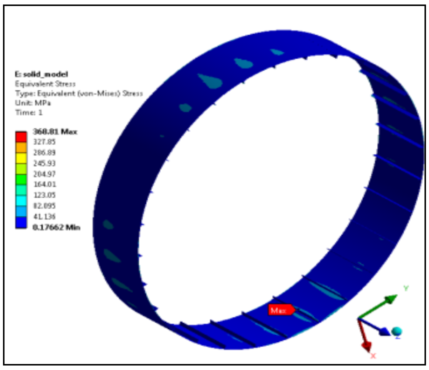


Fig 2.25: Stress Plot for crack length of 1mm for iteration-3

2.21 Summary

The stress, displacement induced in the fuselage for different material is listed below.

Material	Displacement (mm)	Von-Misses Stress MPa)
Al Alloy	1.65	71
Composite Material 1	0.70	68
Composite Material 2	0.53	66

Table 1: Displacement and Von-Misses Stress for Aluminum and Composites.

The Stress Intensity Factor (SIF) and Von-Misses stress at the crack font for different crack length is shown below.

Material/Crack length	Stress Intensity Factor (SIF) (Mpa.mm^0.5)			Von-Misses Stress (MPa)		
	1mm	**2mm**	**3mm**	**1mm**	**2mm**	**3mm**
Iteration1: Al Alloy	25	35	37	285	337	426
Iteration2: Composite	28	29	30	414	437	461
Iteration3: Composite	29	30	33	369	461	475

Table 2: SIF and Von-Misses Stress for different crack length of Aluminum and Composites.

3. CONCLUSION

From the above Finite element analysis we can make the following conclusion:

- It is observed that the Stress Intensity Factor (SIF) is approximately same for all the 3 materials.
- The stress and displacement induced in the fuselage for the composite material is less than the aluminum alloy.
- The optimum stacking of the lamina will helps to reduce weight and stress acting on the aircraft fuselage.
- The stacking sequence of the lamia is playing vital role in stress distribution and the maximum deformation in the fuselage.

From the FEA results, it is clear that aluminum alloy can be replaced with composite material without affecting the strength and Stress intensity factor characteristics of the fuselage.

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