Stress Analysis of Fuselage Stiffened Panel with and Without A Door Cutout

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Abstract - Aircraft is a complex engineering structure. The safety of the structure and the weight of the structure are the two important aspects to be kept in mind while designing the aircraft structure. It is an uphill task for an aircraft structural designer to bring out a safest structure with minimum weight. The current project deals with the stress analysis of the fuselage-stiffened panel with and without a large cutout for an emergency exit door. Cabin pressurization load case, which is one of the critical load cases used for the stress analysis. When the aircraft is flying above 40000 ft. altitude pressure is applied to fuselage inside cabin to create sea level atmospheric pressure . Stress analysis of the fuselage segment will be carried out to capture the global response of the fuselage under pressurization. A skin panel with fuselage frame and stringers representing the fuselage features will be considered for the local analysis to capture global response. An emergency exit door cutout will be introduced in the stiffened panel and analyzed for stress distribution around the cutout. Stress concentration factor and variation of stress from cutout edge towards the free edge of the panel will be obtained from the stress analysis results.

Key Words: Transport aircraft, Fuselage, Bulkhead, Orthogonal stiffening, Finite element method, stress analysis, emergency exit door cutout, Stress concentration factor

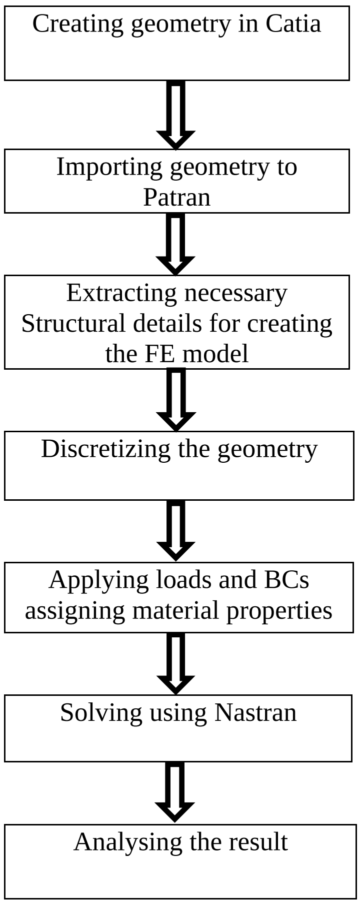
1. INTRODUCTION

Fuselage and wing are the main structural components of the airframe. Fuselage is a cylindrical structure, which houses passenger seats and cargo at the rear end. Normally fuselage is a built-up structure with stiffening members along longitudinal and circumferential directions. The skin used for the structure is a thin member with stiffening. Fuselage consists of external skin, longerons, and stringers. Stringers are attached to frame used to resist bending when it subjected to loads it is responsible for aerodynamics loads acting on the skin. Longerons are the members which run through the length of the fuselage it is responsible for carrying the loads and transmitting the loads imposed on fuselage. The fuselage is able to resist bending moment, torsional load, and cabin pressurization.

2. Objectives

The main aim of this project is Stress analysis of the fuselage. A skin panel with fuselage frame and stringers representing the fuselage features will be considered for the local analysis to capture global response under pressurization

3. Methodology



4. Geometric model

The CAD model of fuselage structure was designed by using Catia V5 as shown in figure 4.

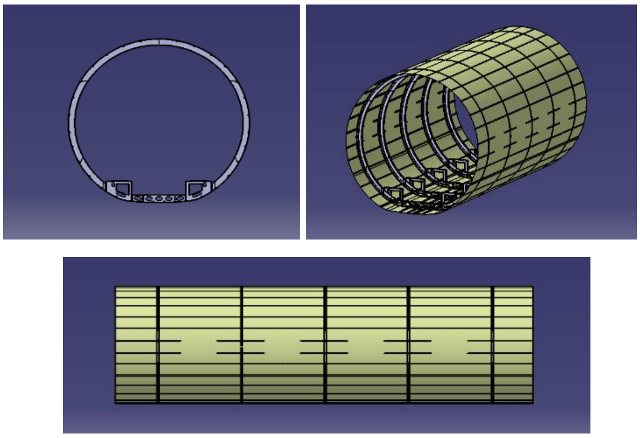


Figure 4 Shows fuselage Structure

5. Finite element analysis of fuselage

For FE analysis MSC Nastran and Patran is used.

5.1 Material properties.

The material used for analysis is Aluminium 2024 material. It is most commonly used in aircraft industries.

Table 1. shows aluminium 2024 alloy material properties

Material properties	Al 2024
Ultimate Tensile strength	469MPa
Yield strength	360MPa
Shear strength	283MPa
Fatigue strength	138MPa
Modulus of elasticity	73.1GPa
Thermal conductivity	121w/m-k
Melting point	502-638 ^{o}C
Electrical resistance	5.82x10^{-6} ohm-cm
Hardness	150GPa
Shear modulus	28GPa

5.2 Theoretical Calculation

The fuselage is a built-up structure with stiffening members along longitudinal and circumferential directions. From pressurized thin cylinder

Hoop stress = $\frac{Pi \times r}{t}$

Longitudinal stress = $\frac{Pi \times r}{2 \times t}$

Where Pi = Internal pressure

r = Radius of the cylinder

t = Thickness of cylinder

Let us consider Flight flying at 40,000 ft. altitude

Sea level pressure = 14.7 PSI

At 40,000 ft. altitude pressure = 2.7 PSI

When the aircraft is flying above 40000 ft. altitude pressure is applied to fuselage inside cabin to create sea level atmospheric pressure.

Internal pressure = 14.7-2.7 = 12 PSI = 0.0084 kg/mm^2

Radius fuselage skin r = 973 mm

Thickness of fuselage t = 2.2 mm

Hoop stress = $\frac{Pi \times r}{t}$ = = $\frac{0.0084 \times 973}{2.2}$ = 3.715 kg/mm^2

Longitudinal stress = $\frac{Pi \times r}{2 \times t} = \frac{0.0084 \times 973}{2.2 \times 2}$ = 1.857 kg/mm^2

This calculation only for Fuselage skin not for entire model

5.3 Boundary Conditions

Condition	Value
End condition	Both end fixed
Internal pressure	0.0084 kg/mm^2
material	Aluminium 2024

5.4 Stress analysis

The analysis was performed using Nastran Patran

5.4.1 Analysis of Fuselage skin

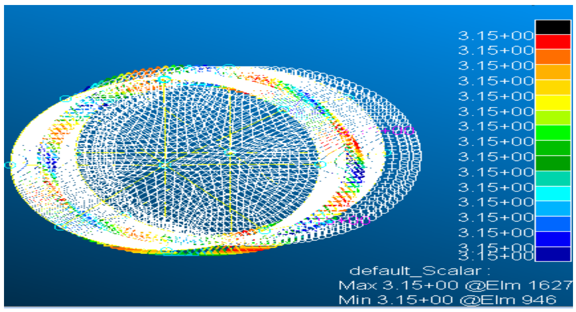


Figure 5.1 Shows analysis of Fuselage skin(Hoop Stress)

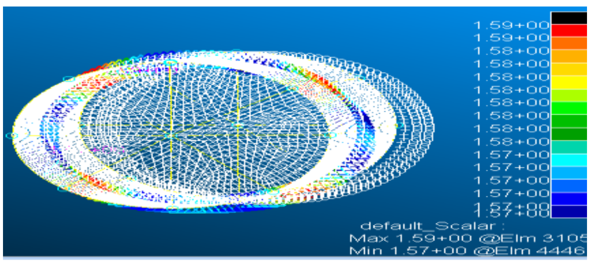


Figure 5.2 Shows analysis of Fuselage skin (Longitudinal Stress)

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5.4.2 Analysis of Fuselage structure (Global model)

Fuselage structure with Frames and Stringers

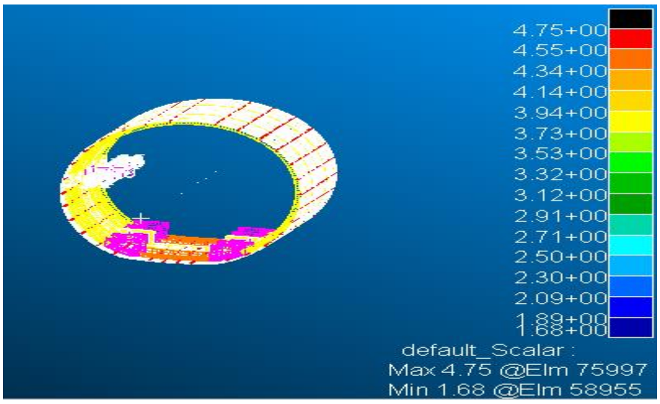

Figure 5.3 shows global model of fuselage (Hoop Stress)

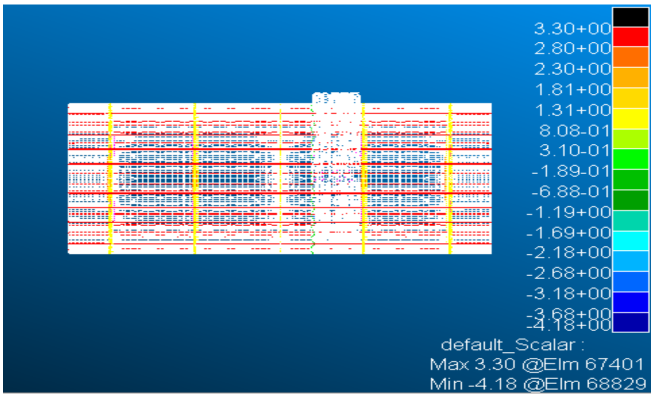

Figure 5.4 shows global model of fuselage (Longitudinal Stress)

We are taking segment of fuselage and representing it to local model such the curved stiffened fuselage panel is now represented as flat stiffened panel in single plane now for this load and boundary conditions are given such that it experience some stress as in global model.

5.4.3 Local model of fuselage structure

Creating local model to capture global model response

Calculation of local model

Taking average stress σ = 3.69 kg/mm^2

$\sigma = \frac{F}{A}$

Area = width $\times$ thickness

Area = 2.2 $\times$1600 = 3520 mm^2

F = σ $\times$ A = 3.69$\times$2.2$\times$1600 = 12988.8 kg

Total load = 12988.8 kg applied to local model

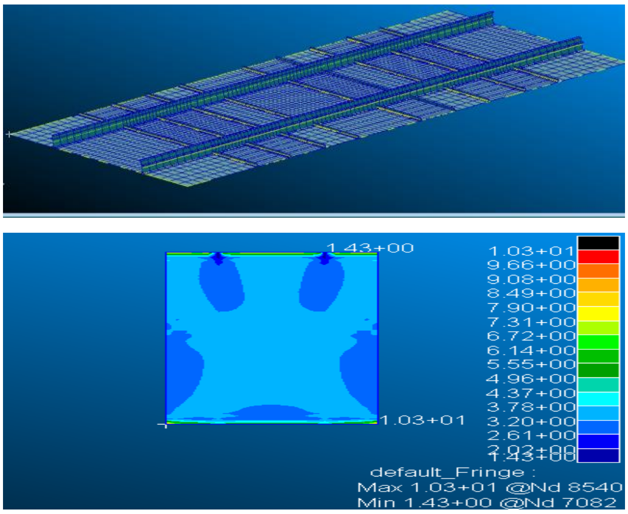

Figure 5.5 shows local model of fuselage structure

Introducing the emegency door cutout in local model and creating a rivet hole over emergency door cutout for this we are checking maximum stress location

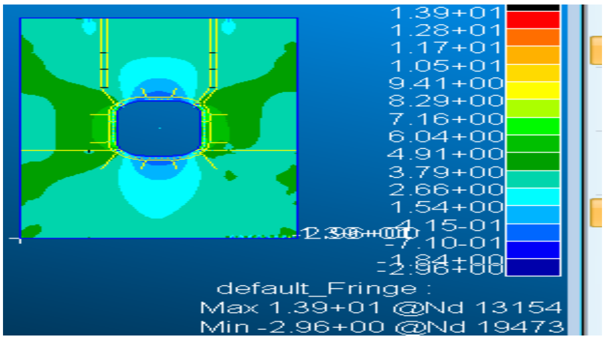

Figure 5.6 shows local model of fuselage structure with emergency door cut out

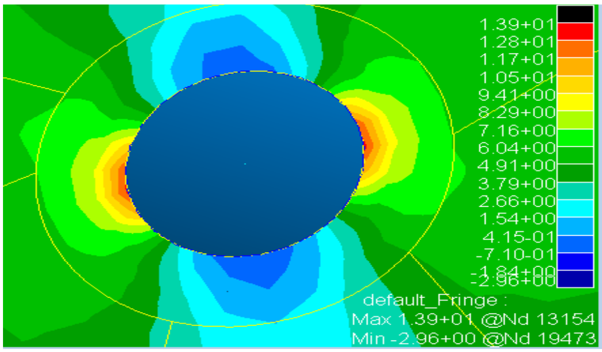

Figure 5.7 shows closest view of rivet hole over emergency door cutout

The maximum stress occurs at rivet hole local is 13.9 kg/mm^2

6. Margin of safety

$\frac{Strength}{Stress} - 1$

$\frac{36}{13.9}$ -1 = 1.589

This shows our structure is safe

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7. CONCLUSIONS

The Aircraft fuselage was designed and analyzed by using Catia and Nastran Patran. The stress analysis of stiffened fuselage structure in presence of emergency door cutout the maximum stress are observed at rivet hole location. The maximum stress at rivet hole are 13.9 kg/mm^2 and yield strength of aluminium 2024 material is 36 kg/mm^2 and for this condition margin of safety observed is 1.589 which concludes that our structure is safe.

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