

Design optimization of airframe in aircraft fuselage structure under static loading conditions

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Abstract: The fuselage is one of the main component in any aircraft and its function is to hold all parts together and carries passengers. This fuselage part experience a different loads like static, fatigue, dynamic, buckling during landing, flying and take-off conditions. Now a day's aircraft undergo different type of failure modes, due to improper design, pilot error, weather conditions etc. In the present work, fuselage component with airframes can be optimized in design in order to check which design of airframes in fuselage structure can withstand static loading conditions with minimum deflection and minimum induced stress. The result shows that fuselage component with airframes containing three vertical truss members under roof can able to withstand static loading conditions with minimum deflection, fuselage component with airframes containing two inclined and one vertical truss member located at centre under roof can able to withstand static loading conditions with minimum induced stress and fuselage component with airframes don't containing any truss members under roof cannot able to withstand static loading conditions.

Index Terms: Fuselage, airframe, truss members, static loads.

I. INTRODUCTION

An aircraft is a machine that is able to fly by gaining support from the air and driven by jet engines or propellers. The main sections of an aircraft, the fuselage, tail and wing, determine its external shape. The load-bearing members of these main sections, those subjected to major forces, are called the airframe. Fuselage is based on French word fuseler, which means "to streamline". The fuselage, or body of the airplane, is a long hollow tube, which holds all the parts of an airplane together. The fuselage is hollow to reduce weight.

In order for an airplane to fly straight and level, the following relationships must be true [1]:

- Thrust = Drag
- Lift = Weight

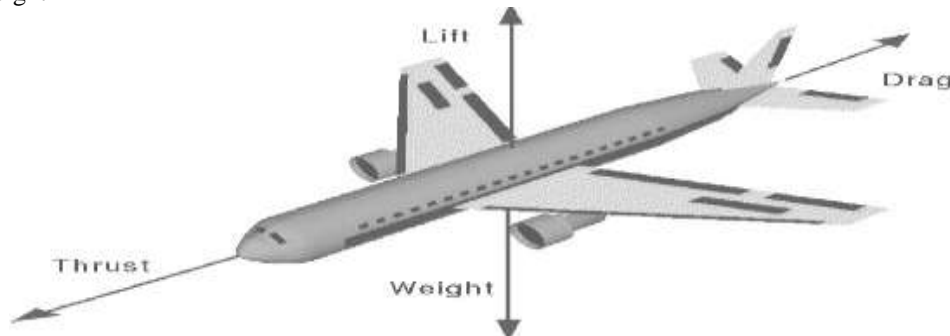


Fig. 1 The forces acting on aircraft

For analysis purpose Airbus A321 is used. It is a largest member of A320 family's. The Airbus A321 single-aisle medium range- airliner is the largest aircraft in the A320 range.



Fig. 2 Airbus A321

Airbus A321 Specifications [2]
Dimensions

Length	44.5m
Wingspan	34.1m
Height	11.8m
Wing area	122.4m ²
Weight	
Maximum take-off weight	83000-93500kg
Maximum landing weight	73500-77800kg
Operating empty weight	48100kg
Maximum zero fuel weight	71500kg
Maximum payload	23400kg
Standard fuel capacity	23700-29680Litres
Performance	
Range with max payload	5000-5500km
Cruise speed	840km/h
Maximum speed	890km/h
Maximum operating altitude	11900m
Take-off field length	2180m
Landing field length	1580m
Engines	CFMI CFM56-5A/5B, 2*30000-33000 lb IAE V2500-A5, 2*30000-33000 lb
Fuel efficiency	18.2g/pass*km
Fuel flow rate	3200kg/h
Cabin Data	
Passengers	220(1-class)
Passengers	185(2-class)
Cabin width	3.7m

Many researchers have worked on designing this part through various techniques like finite element method, experimental method and analytical method. The researchers have carried out different analysis related to aircraft fuselage structure such as static, buckling, dynamic fracture, fatigue analysis etc., The buckling analysis can be made by different ways such that post buckling response behavior of stiffened panels under compression [3] and post buckling response of stiffened panels under shear [4]. The dynamic fracture analysis can be made by different ways such that dynamic fracture analysis of aircraft fuselage with damage due to two kinds of blast loads [5], blast response of metal composite laminate fuselage structures with two material configurations such as aluminium and GLARE [6]. The researchers are also made analysis related to predicting the service durability of aerospace components [7], residual strength pressure tests analysis of stringer and frame stiffened aluminium fuselage panel with longitudinal cracks [8], weight comparison analysis between a composite fuselage and an aluminium alloy fuselage [9], impact of engine debris on fuselage skin panel [10], damage analysis of aircraft structure due to bird strike [11], damage prediction in airplane flap structure due to bird strike [12], and analysis of high energy impact on a sheet metal aircraft structures [13]. The fatigue analysis can be made by different ways such that damage tolerance analysis of aircraft reinforced panels [14], fatigue cracks at many rivet locations in the skin panel [15], and fatigue analysis for upper and lower folding beams on the rear fuselage [16]. The static analysis can be made by different ways such that different conceptual designs that included as frames spacing was smaller compared to stringers spacing, frames spacing was larger compared to stringers spacing, frames and stringers spacing was approximately equal [17] and laminate constructions for stiffened fuselage panels in aircraft design [18]. The review of their work are they developed numerical model satisfied with test results, but the structure reacts very sensitively to the modelling of the boundary conditions. Hence, the scope of this work reported in this paper is to design optimization of airframe in aircraft fuselage structure under static loading conditions.

II. GEOMETRY OF THE MODELS

The below figures show the 3D CAD model of different cases airframes in fuselage structure. The model is built by using NX CAD v7.5 modeling software. While preparing the model different features like extrude, pattern, revolve etc. are used. The models are created in .prt format, in order to use this models in Hypermesh v11.0 Pre-processing software it is converted into .IGES file format, which can be now opened in any other FEA software also. Figure 3 shown the case 1 model don't have any supporting members under the roof. Figure 4 shown the Case 2 model, the 50mm width of 2 vertical truss members are placed at equal distance under roof in fuselage of aircraft. Figure 5 shown the Case 3 model, the 50mm width of 2 truss members are placed in 75° inclined position under roof in fuselage of aircraft. Figure 6 shown the Case 4 model, due to maximum deflection at centre, a single truss member of width 100mm is placed at centre under roof in fuselage of aircraft. Figure 7 shown the Case 5 model, due to weight constraints and stress concentration, a single truss member of width 35mm with taper ends is placed at centre under roof in fuselage of aircraft. Figure 8 shown the Case 6 model, due to weight constraints and deflection, the 3 truss members of width 35mm are placed in equal distance with combination of inclination and vertical under roof in fuselage of aircraft. Figure 9 shown the Case 7 model, due to

weight constraints and deflection, the 3 truss members of width 35mm are placed in equal distance with all are in vertical position under roof in fuselage of aircraft.

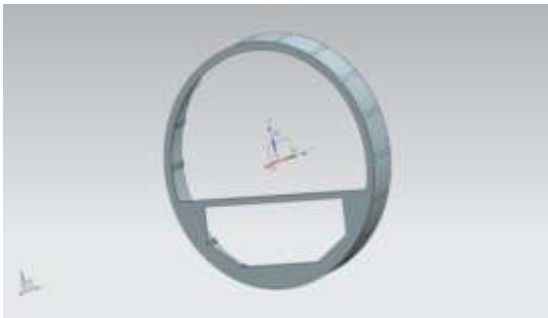


Fig. 3 Geometry of case 1 model

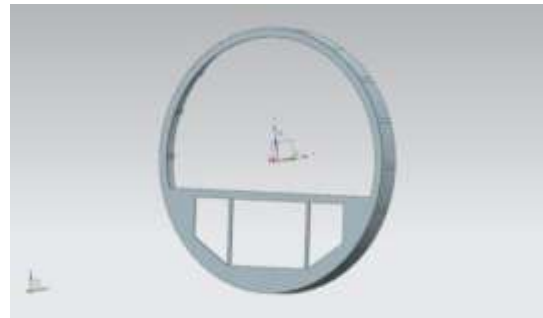


Fig. 4 Geometry of case 2 model

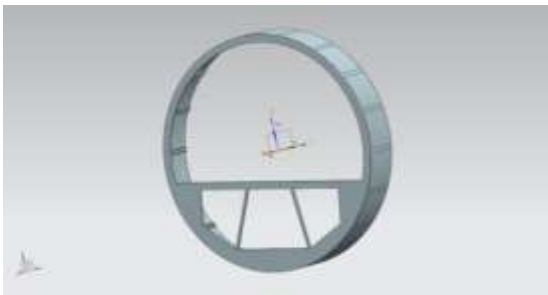


Fig. 5 Geometry of case 3 model

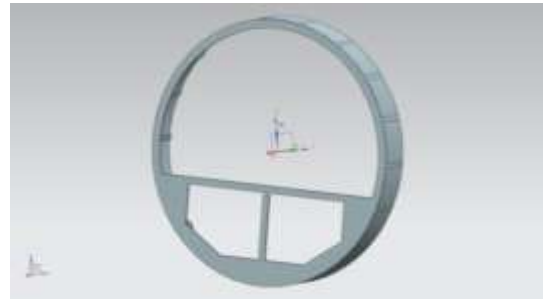


Fig. 6 Geometry of case 4 model



Fig. 7 Geometry of case 5 model



Fig. 8 Geometry of case 6 model



Fig. 9 Geometry of case 7 model

III. MESHING OF THE MODELS

Meshing is the process of converting infinite Degrees of freedom (DOF) to finite Degrees of freedom. The .IGES file format of the model is imported in Hypermesh v11.0 software. The first step is to do the geometry clean up. The imported geometry has a free edge, shared edge and suppressed edges, all of them are removed and geometry is modified so that the finite element mesh can be created efficiently.

The modified structure is Hexa meshed by using 8 noded Hexahedron elements. The manual mesh method was used to generate a continuous mesh. At critical cross section of the geometry, more number of elements and fine mesh was used in order to obtain more accurate results. The figures from 10 to 16 shows meshed models for different cases of airframes in aircraft fuselage structure.

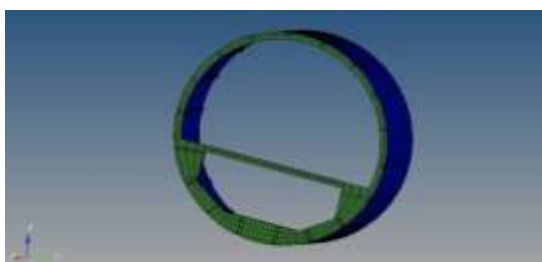


Fig. 10 Meshed case 1 model

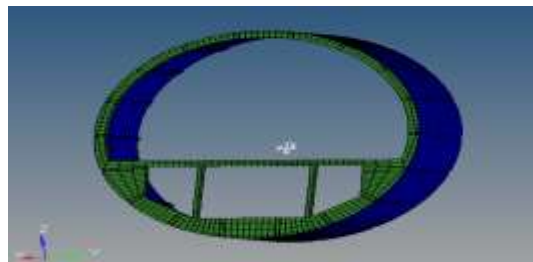


Fig. 11 Meshed case 2 model

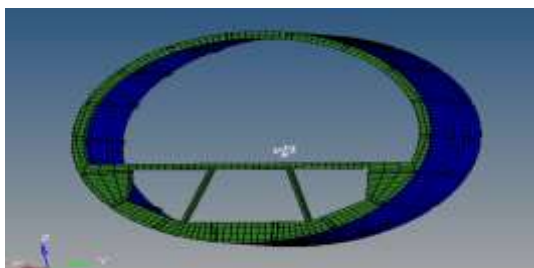


Fig. 12 Meshed case 3 model

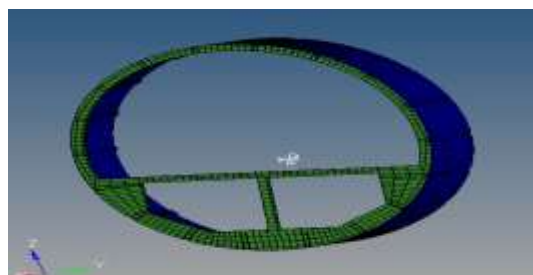


Fig. 13 Meshed case 4 model

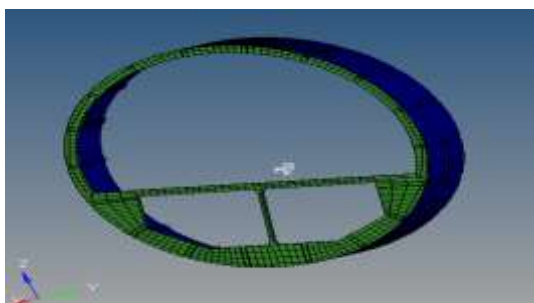


Fig. 14 Meshed case 5 model

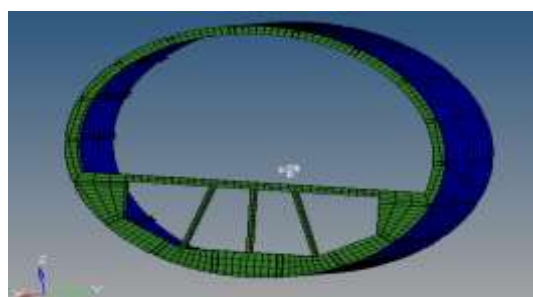


Fig. 15 Meshed case 6 model

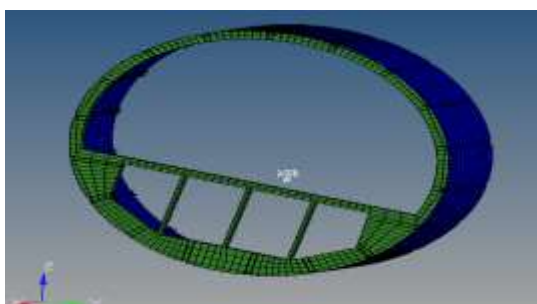


Fig. 16 Meshed case 7 model

Elements used:- 8 noded hexahedron element

The 8 noded hexahedron element is a three dimensional element with 8 nodes at its corners. The element is defined by 8 nodes having three degrees of freedom at each node: translations in the nodal x, y and z directions (UX, UY, UZ). Hexahedron element is also called Brick element. The figure 17 shows a 8 noded hexahedron element.

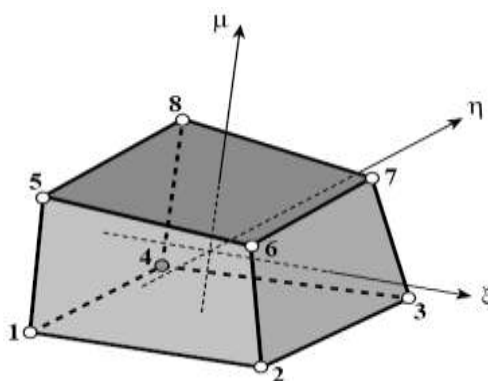


Fig. 17 8 noded hexahedron element

IV. MATERIAL SELECTED

After the meshing process next step is to assign the material properties and its behaviour. The two image cards PSOLID and MAT1 are used in software. PSOLID defines solid element and MAT1 defines the material properties for linear elastic, temperature-independent, isotropic material.

Selection of materials in aircraft construction is rather complex and is based on trade off amongst conflicting requirement of high strength, low density and easy of fabrication or processing. The material used in various parts of vehicle structures generally are selected by different criteria. The material used in the fuselage structure is Aluminium alloy 2024-T351 and its composition as shown in table 1.

Table 1 Composition of Aluminium alloy 2024-T351

Composition	Wt. %
Al	90.7-94.7
Cr	Max 0.1
Cu	3.8-4.9
Fe	Max 0.5
Mg	5.2-5.8
Mn	0.3-0.9
Si	Max 0.5
Ti	Max 0.15
Zn	Max 0.25
Others	Max 0.15

The following mechanical properties of the material are used in the analysis.

Young's Modulus, $E=70,000 \text{ N/mm}^2$

Poisson's Ratio, $\mu = 0.3$

Ultimate Tensile Strength, $\sigma_u = 420 \text{ N/mm}^2$

Yield Stress, $\sigma_y = 350 \text{ N/mm}^2$

Density = 2780 kg/m^3

V. LOAD AND BOUNDARY CONDITIONS APPLIED

The figure from 18 to 24 shows different models with load and boundary conditions. The boundary conditions are the application of a force, pressure and constraints. In Hypermesh boundary conditions are stored within the load collectors. One of the assumption here is the load acting on roof is uniformly distributed. In the software, to apply a uniformly distributed load on a roof a load collector of name FORCE is used, to apply a cabin pressure in a fuselage a load collector of name PLOAD is used and to apply a constraints a load collector of name SPC (single point constraint) is used.

The static analysis is carried out by taking a maximum take-off weight of Airbus A321 as 93500kg is applied on roof of fuselage, a cabin pressure of about 56kPa is applied around the skin panel in fuselage structure. The fixed support that is constraints are applied at frame and skin panel connected locations in a fuselage structure. The next step in the analysis is deck preparation that means preparing final model for solving. By using the control card STATICS, the analysis type is set to the linear static analysis and similarly using SOL, PARAM, and GLOBAL_OUTPUT_REQUEST control cards, the required output parameters like displacements, stress and strains are clearly defined. Then this final FEA model which is ready for solving is fed to the solver. Before that the FEA model which is in .hm file format is converted to .bdf or .dat file format because it is the required input file format for MSC NASTRAN solver.

The solver takes around 10 to 15 minutes of time for solving in a Pentium dual core processor, 2GB RAM equipped PC. The solving time can be minimized by using high configured computer. A number of output files were generated after solving, among which a file of .bdf and .op2 format are used to generate the contour plots of stress and deflection in the Hyperview v11.0 software, which is the post processor used in the analysis.

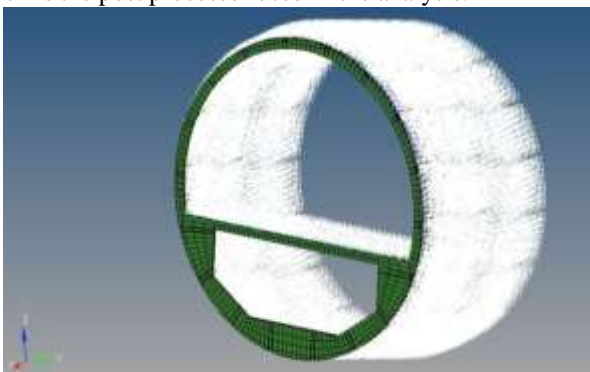


Fig. 18 Load and boundary conditions for case 1

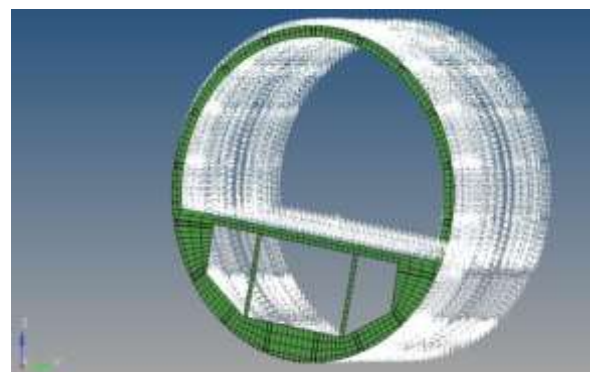


Fig. 19 Load and boundary conditions for case 2

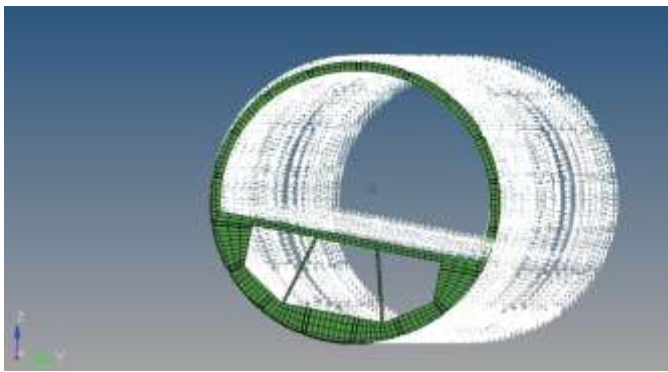


Fig. 20 Load and boundary conditions for case 3

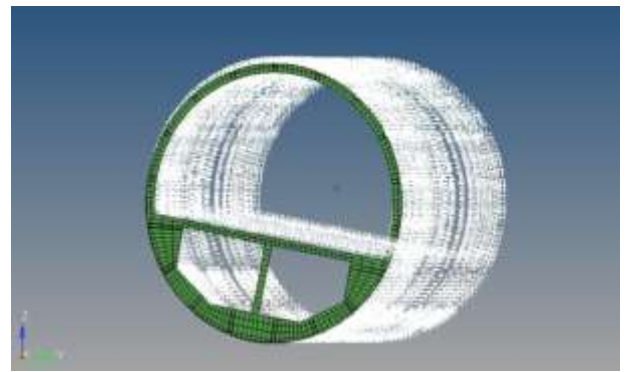


Fig. 21 Load and boundary conditions for case 4

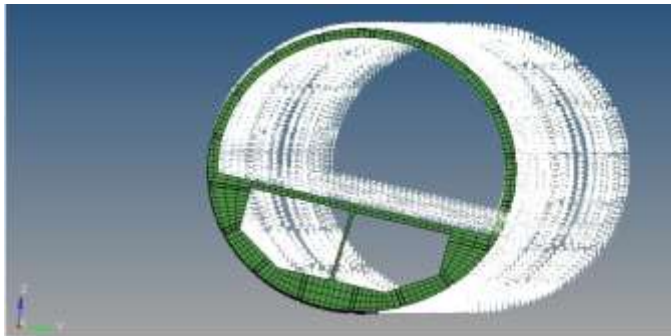


Fig. 22 Load and boundary conditions for case 5

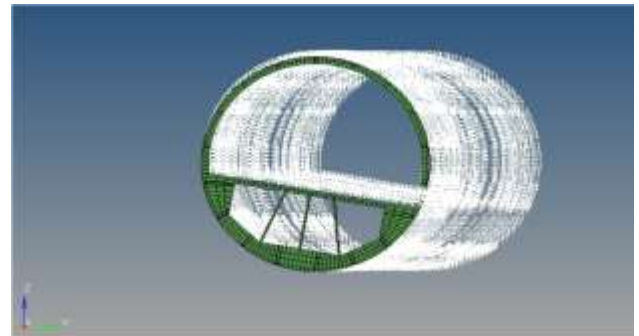


Fig. 23 Load and boundary conditions for case 6

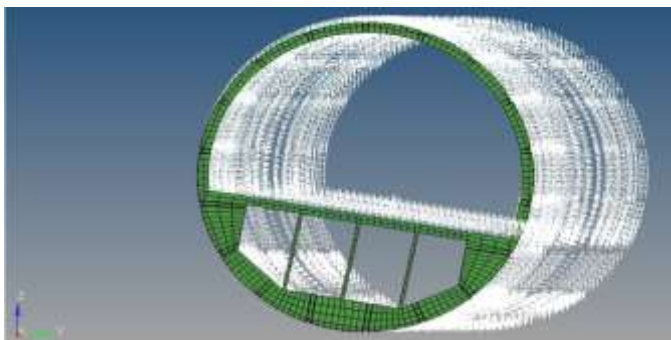


Fig. 24 Load and boundary conditions for case 7

VI. RESULTS

The generated .bdf and .op2 file format for different cases were loaded in the Post-processor. The contours of deflection and Von-Mises stresses for different cases were plotted as shown in below figures from 25 to 38.

Calculation of allowable stress the material

For Aluminium alloy 2024-T351

We know that Yield strength = 350MPa

Factor of safety considered = 2

Hence,

$$\begin{aligned} \text{Allowable stress} &= (\text{yield strength of the material}) / (\text{Factor of safety}) \\ &= 350/2 \end{aligned}$$

$$\text{Allowable stress} = 175\text{MPa.}$$

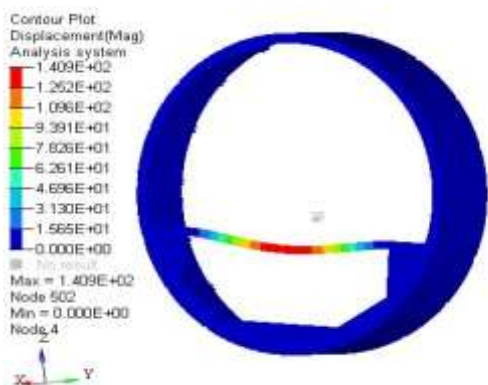


Fig. 25 Deflection plot for case 1

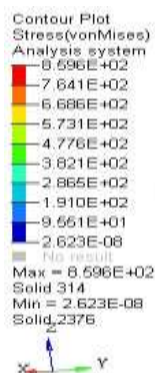


Fig. 26 Stress plot for case 1

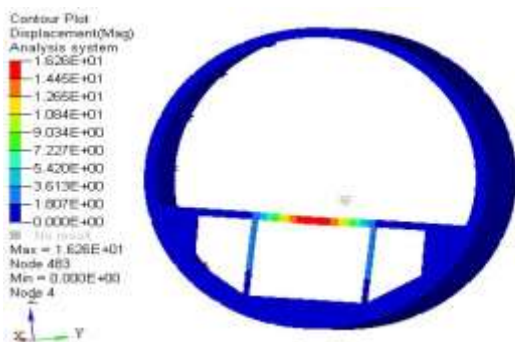


Fig. 27 Deflection plot for case 2

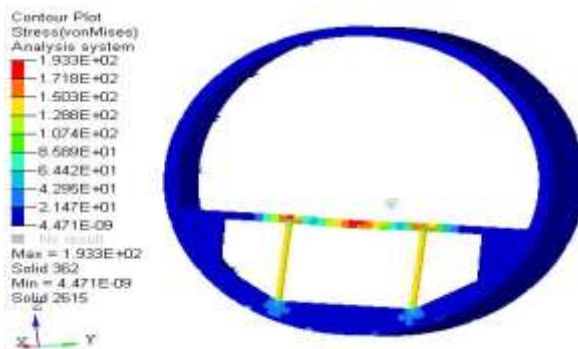


Fig. 28 Stress plot for case 2

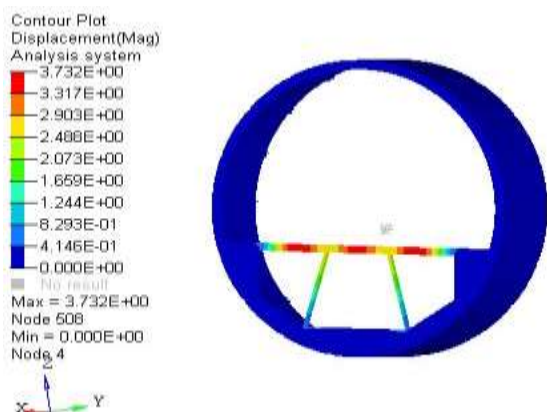


Fig. 29 Deflection plot for case 3

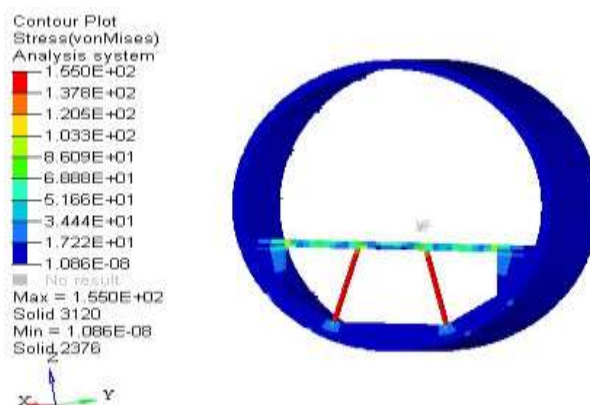


Fig. 30 Stress plot for case 3

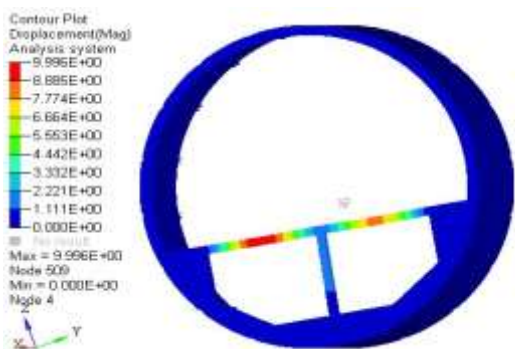


Fig. 31 Deflection plot for case 4

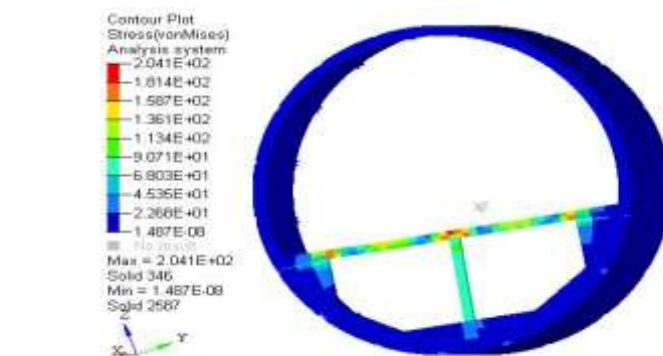


Fig. 32 Stress plot for case 4

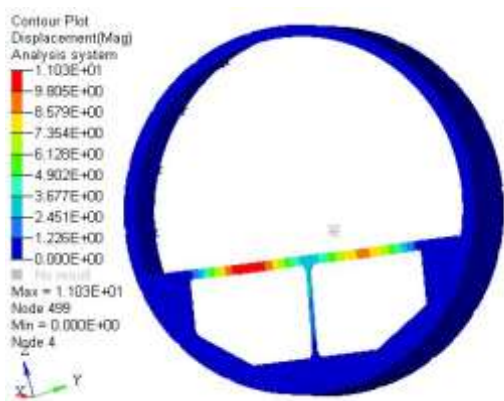


Fig. 33 Deflection plot for case 5

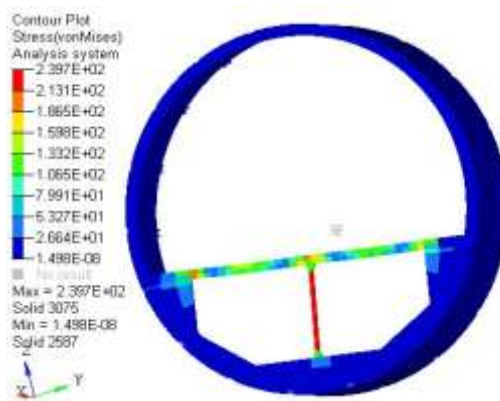


Fig. 34 Stress plot for case 5

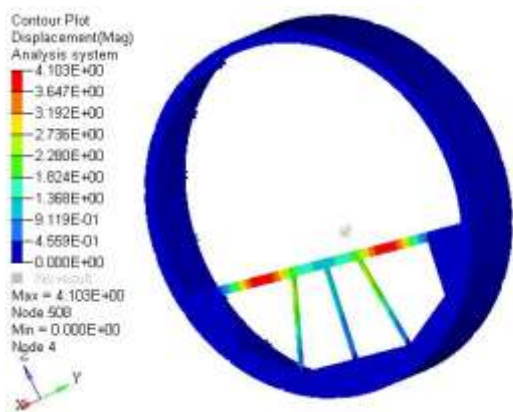


Fig. 35 Deflection plot for case 6

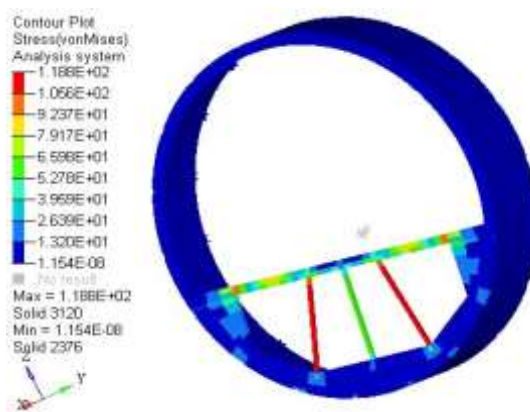


Fig. 36 Stress plot for case 6

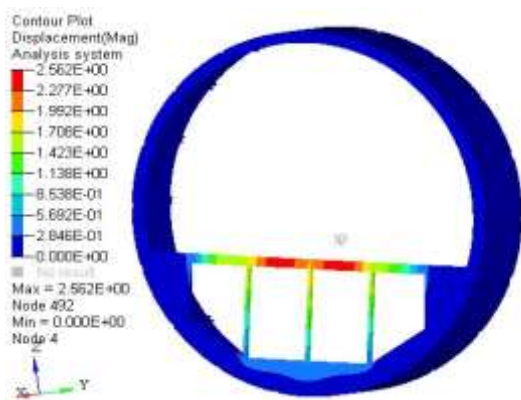


Fig. 37 Deflection plot for case 7

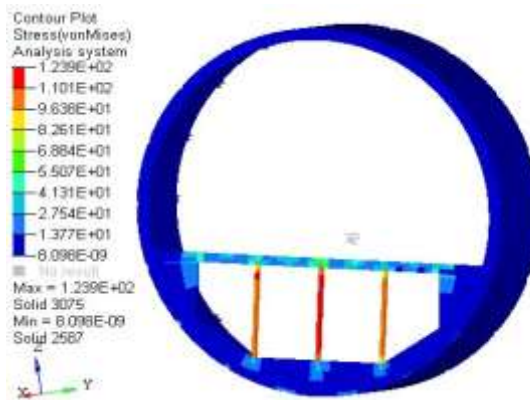


Fig. 38 Stress plot for case 7

The results are tabulated as follows.

Table 2 Results for different optimized models

Optimized model Cases	Mass in *10 ^{^10} kg	Max. Deflection in mm	Induced Von-Mises Stress in MPa
1	47.98	140	859
2	49.33	16	193
3	48.92	3.7	155
4	48.62	9.9	204
5	48.52	11	239
6	49.43	4.1	118
7	49.43	2.5	123

VII. CONCLUSIONS

From the results of static analysis, the case 1 model have fail under static loading conditions because induced stress is more compared to allowable stress of material that is $859\text{MPa} > 175\text{MPa}$. Hence Optimization in geometry of model of fuselage is necessary to withstand static loading conditions. Among different models of fuselage Case 3, Case 6 and Case 7 models have safe design compared to obtained model of fuselage because induced stress is less compared to allowable stress of the material, that is for Case 3, Case 6 and Case 7 optimized models have induced stress values are 155, 118, 123MPa respectively is less than allowable stress of material that is 175MPa for Aluminium alloy 2024-T351. The Case 6 optimized model of fuselage have less induced stress that is 118MPa, Case 7 optimized model of fuselage have less deflection that is 2.5mm as compared to other optimized models of fuselage.

REFERENCES

- [1] David F. Anderson & Scott Eberhardt "Understanding flight" McGraw-Hill (2001).
- [2] www.Airbus.com
- [3] Lynch C, Murphy A, Price M, Gibson A, "The computational post buckling analysis of fuselage stiffened panels loaded in compression", Journal of Thin-walled structures, vol., 42 2004, pp. 1445-1464.
- [4] Murphy A, Price M, Lynch C, Gibson A, "The computational post buckling analysis of fuselage stiffened panels loaded in shear", Journal of Thin-walled structures, vol., 43 2005, pp. 1455-1474.
- [5] Jinsan Ju, Xiaochuan You, "Dynamic fracture analysis technique of aircraft fuselage containing damage subjected to blast", Journal of Mathematical and Computer modeling, vol., 58 2013, pp. 627-633.
- [6] Kotzakolios T, Vlachos D. E, Kostopoulos V, "Blast response of metal composite laminate fuselage structures using finite element modeling", Journal of Composite structures, vol., 93 2011, pp. 665-681.
- [7] Aleksandar Grbovic, Bosko Rasuo, "FEM based fatigue crack growth predictions for spar of light aircraft under variable amplitude loading", Journal of Engineering failure analysis, vol., 26 2012, pp. 50-64.
- [8] Richard D. Young, Marshall Rouse, Damodar R. Ambur and James H. Starnes, Jr "Residual strength pressure tests and nonlinear analysis of stringer and frame stiffened aluminium fuselage panels with longitudinal cracks".
- [9] Marco Aurelio Rossi, Sergio Frascino Muller de Almeida, "Design and analysis of a composite fuselage", vol., 2009, pp. 14-16.
- [10] Norman F. Knight Jr, Navin Jaunky, Robin E. Lawson, Damodar R. Ambur, "Penetration simulation for uncontained engine debris impact on fuselage-like panels using LS-DYNA", Journal of Finite elements in analysis and design, vol., 36 2000, pp. 99-133.
- [11] Smojver I, Ivancevic D, "Bird strike damage analysis in aircraft structures using Abaqus/Explicit and Coupled Eulerian Lagrangian approach", Journal of Composites science and Technology, vol., 71 2011, pp. 489-498.
- [12] Smojver I, Ivancevic D, "Numerical simulation of bird strike damage prediction in airplane flap structure", Journal of Composite structures, vol., 92 2010, pp. 2016-2026.
- [13] MacDonald B. J, "A computational and experimental analysis of high energy impact to sheet metal aircraft structures", Journal of Materials processing technology, vol., 124 2002, pp. 92-98.
- [14] Carta F, Pirondi A, "Damage tolerance analysis of aircraft reinforced panels", vol, 16 2011, pp. 34-42.
- [15] Karthik N, Anil Kumar C, "Analysis of the fuselage structure for multisite damage", International Journal of Innovative Research in Science, Engineering and Technology, vol., 2 2013, ISSN: 2319-8753.
- [16] Giglio M, "FEM submodelling fatigue analysis of a complex helicopter component", International Journal of Fatigue, vol., 21 1999, pp. 445-455.
- [17] Khairi Yusuf, Nukman Y, Dawal S Z, Devi Chandra, N.Sofia (2010), "Conceptual design of fuselage structure of very light jet aircraft", ISSN: 1792-4359.
- [18] Linde P, Schulz A, Rust W, "Influence of modeling and solution methods on the FE-simulation of the post-buckling behaviour of stiffened aircraft fuselage panels", Journal of Composite structures, vol., 73 2006, pp. 229-236.