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# Design and Stress Analysis of Aircraft Interior Panel for Extreme Loading Conditions

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**ABSTRACT:** Strength of the aircraft interior structural elements has to be evaluated to ensure the safety to the passengers inside the aircraft. Under the severe ‘g’ conditions an interior panel may break and hurt the passengers. Certification requirements indicate that interior panels or parts have to be subjected to severe ‘g’ conditions in all three directions through the stress analysis process. If the margin of safety is positive then it will be accepted from the design certification regulatory authorities. Federal aviation regulations advisory circulars indicate the severe ‘g’ factors to be considered in each direction during the stress analysis of the interior structural components. Fuselage undergoes cabin pressurization and aerodynamic loads during the flight. Considering the crash-landing case, severe ‘g’ conditions in all three directions are arrived at. Interior components of an aircraft are of semi-monocoque construction. Stiffeners in two orthogonal directions reinforce a thin skin. If there is a requirement of higher thickness, then sandwich panels are used with a core material to build the thickness along with face sheets. One of the modes of failure could be static failure due to high stress under the severe ‘g’ loading condition. This project will investigate a simple fuselage interior stiffened panel subjected to linear static load. Linear static stress analysis of the interior stiffened panel will be carried out. Finite element method is adopted for the stress analysis. Different configuration of the stiffened panel will be considered for the stress analysis. The criterion is to keep the stress levels lesser than the strength of the material used with minimum deflection of the panel. Through this approach one can arrive at the minimum weight for the structural components. Both aluminum alloy and composite material will be considered for the stress analysis. A comparative study will be carried out.

**KEYWORDS:** Fuselage, Semi-monocoque construction, Stiffeners, ‘g’ loads, stress, Frames, Stringers and FEA.

## I. INTRODUCTION

The fuselage is the ‘body’ of an aircraft, its primary structure; it is the envelope that houses the cockpit, the space for carrying passengers or for the cargo hold. It is also the framework to which other fundamental parts are attached, such as the wings, tail empennage or landing gear.

Aviation is an industry that is always at the forefront of research and development, so different designs for the same function are constantly emerging. There are therefore several types of aircraft fuselage, depending on force absorption (monocoque/semi-monocoque), size (wide/narrow) or method of manufacture (lattice or tubular).

The interior of an aircraft fuselage can have several divisions, depending on the use for which it is made: the cockpit, several areas for passengers and crew, a part for passenger baggage or the entire interior for cargo. The each of the cabins were separated by the partition panels.

The Partition Panels is one of the most crucial elements in any airplane, and it is placed in various parts of the aircraft cabin. Partition Walls are individually implementable anywhere in the cabin, not only between different travel classes, but also between seat rows and monuments or galleys. A partition for an aircraft is designed to separate a cabin part from another compartment of this aircraft. The partition includes a cellular central structure in contact with a first external wall and a second external wall of this partition. The partition also includes several stiffener elements added onto at least one of the first and second external walls. The invention also relates to an aircraft including at least one such partition.

The load factor is the ratio of the lift of an aircraft to its weight and represents a global measure of the stress ("load") to which the structure of the aircraft is subjected. Load factor is the ratio of two forces, it is dimensionless. However, its units are traditionally referred to as g, because of the relation between load factor and apparent acceleration of gravity felt on board the aircraft. A load factor of one, or 1 g, represents conditions in straight and level flight, where the lift is equal to the weight.

## II. RELATED WORK

Numerical simulation of aircraft interior components under crash loads, this paper developed by S. Heimbs, D. Vogt, R. Hartnack, J. Schlattmann & M. Maier in this work they discuss about This study investigates the structural response of overhead stowage compartments (OHSCs) in commercial aircraft passenger cabins under crash loads using dynamic finite element (FE) simulations using commercial software LS-DYNA. The aim is to establish methods for characterizing and adequate numerical modelling for dynamic simulations of composite aircraft interior components, rather than evaluating the crashworthiness properties of a specific overhead bin design. Different components, such as composite honeycomb sandwich material, load introduction points, and corner L-joints, have a major influence on de-formation and failure behaviour. A detailed knowledge of the respective failure mechanisms and failure loads is necessary for adequate FE modelling, making testing on material and component level inevitable. Baggage items inside the over-head bin also significantly influence the load acting on the structure.

Fail Safe Design of an Aircraft Stiffened Panel by Stress Intensity Factor Determination Through Modified Virtual Crack Closure Integral Method by Shivanandappa N, Guru Kiran they discuss about Damage tolerance is crucial for airplanes to fly safely and comply with airworthiness regulations. A structural component is considered damage tolerant if it remains operational after initial damage is detected. Fatigue crack growth is the main focus of damage tolerance assessment, determining how cracks propagate during service life. Modern aircraft operate in complex environments with loading conditions, human resources, and economic requirements. Designing and verifying damage tolerance and durability are essential for aircraft's continued airworthiness. The design should be designed in terms of damage tolerance to avoid catastrophic failure. A modified virtual crack closure integral method is used for damage tolerance design of a stiffened panel, providing accurate results.

Finite element analysis of foam-filled honeycomb structures under impact loading and crashworthiness design, in this paper This research investigates foam-filled honeycomb sandwich panels under in-plane impact loading and their crashworthiness. The study uses finite element analysis to compare the response of foam-filled honeycomb panels with bare honeycomb panels. Results show that foam-filling significantly affects the response of the honeycomb cores, with heavier foams reducing the plateau region and lower deformations. The study validates the finite element analysis of foam-filled honeycombs under quasi-static compression loading and presents a numerical model for investigating their crashworthiness under low-velocity impact. This research highlights the increasing practical interest in using cellular materials for energy absorption devices, particularly in passive vehicle safety systems.

Design and Optimization of Fuselage of an aircraft for Static and dynamic loads this paper developed by Karthik S and Gurusiddayya they exposing the idea in this project focuses on designing a fuselage structure for a small 50 passenger capacity aircraft using CAD software. The fuselage is crucial for aircraft safety and is subjected to various loads during flight and after landing. The design process considers specific size, performance, competing designs, and common features with existing small capacity aircraft. The project uses CAD/CAM/CAE to evaluate alternative configurations and optimize the fuselage for static and dynamic loads. Finite Element (FE) models are used to predict static and dynamic properties of structures. The project results show that the fuselage structure is rigid and safe, with working stress below the material's yield strength. However, more attention is needed to the critical area of the structure, including assembly parts and joints. The design needs validation through experimental tests and analysis with static and dynamic loads to ensure good and safe results before production.

Optimization of a Fuselage- Stiffened Panel through Stress Analysis in this paper the Finite Element Method (FEM) is used to solve elliptic partial differential equation (PDE) problems, transforming them into algebraic equations. This work focuses on buckling analysis of fuselage stiffened panel structures using NASTRAN & PATRAN software. The analysis identifies necessary modifications to the fuselage cross-section for a safe design. The study also conducts stress analysis of stiffened panels for aircraft fuselage using aluminium alloy plates with grid and orthograde cross-sections. The analysis includes linear and static analysis under specified loading and boundary

conditions. The fuselage, a crucial component in civil transport aircraft, experiences inertia and pressurization loads, causing tension and compression in the upper and lower fuselage stiffened panels. The structural strength and stiffness of the fuselage must be high enough to withstand these loads while maintaining minimal structural weight. The objectives of the study include evaluating load carrying capacity in the presence of cracks and ensuring structural integrity through static and buckling analysis using air load on the fuselage.

Wider parts of the fuselage of a commercial aircraft are subject to shear and compression loads in such a way that this is relevant for dimensioning. First elastic, later plastic buckling can appear. The ultimate load is reached when a combined buckling of skin and longitudinal reinforcements (stringers) leads to failure of a panel. This behaviour is tested physically, but over years CAD-FEM and Airbus Deutschland worked together to simulate the test and analyse further configurations under aircraft-like conditions with ANSYS. The paper describes which difficulties had to be overcome to achieve convergence and obtain reliable ultimate loads. The advantages and disadvantages of different load and solution control features such as force or displacement control, arc-length method, different ways of incrementing are lined out as well as different kinds of imperfections. Furthermore, the use of linear buckling and eigenvalue extraction at some stages of non-linear analysis is considered. It is shown that ANSYS is an appropriate tool for such kind of analysis, but the importance of some missing features becomes evident.

In this paper, Composite is a material consisting of two or more constituents are combined at a ratio of matrix and fibre. Composites are using in most of areas especially in engineering field, in automobile, mechanical, aviation and marine industries. Last few years the scientist is mostly focus on study of composite material and design of components are made with the help of software and experimental analysis because this composite material has high strength to weight ratio. Also, comparative study of Steel, E glass epoxy and FRP is done for stress analysis. These materials are used for vibration analysis to observe the effect of a modal parameters of cantilever beam subjected to free vibration is analysed with the help of analytical method and results obtained from numerical method. finite element method (FEM) is compared. It is concluded that results obtained from FEM have a very good agreement with results obtained from analytically.

A review of some damage tolerance design approaches for aircraft structures this paper discussing about Damage tolerance design is increasingly important in modern aircraft design, with the Air Force requiring a critical review of various approaches. Linear elastic fracture mechanics has been used to predict residual strength and crack growth rates in damaged structures, leading to significant developments in cracked structure analytical methodology. The paper reviews the residual strength analysis methodology, crack propagation laws, and application of fracture mechanics to built-up structural components under spectrum loading conditions. However, the validity of fracture mechanics analytical methodology for complex structures under complex loading conditions remains uncertain. Proper realism factors must be considered in the analysis for high confidence results.

### III. METHODOLOGY

The fuselage's design directly influences the aerodynamics of the aircraft. Streamlined and carefully contoured, it reduces drag and contributes to the airplane's overall efficiency and performance. Inside the fuselage, various compartments are designated for passengers, cargo, cockpit, lavatories, and more. In this project we need only a single section of the fuselage for that we take a reference of transport aircraft. The design of the panel is more important than the design of the fuselage for this project.

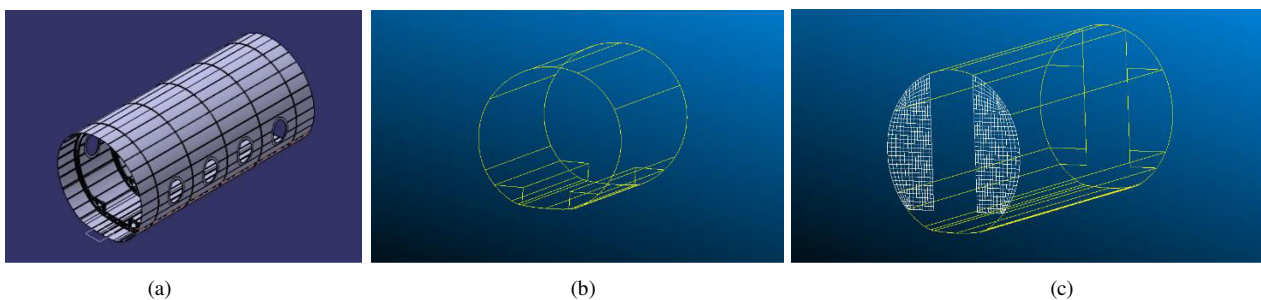


Fig. 1. Design of the panel (a) Isometric view of the fuselage (b) Geometric sketch of the fuselage (c) Meshed view of the panel

Using MSC NASTRAN & PATRAN software the geometry of the panel was created and meshed in 2D. and assigned the properties to the interior panel. Geometry of the Panel, Meshing Details, Boundary conditions and G-conditions are mentioning below;

- Height of the interior panel: 1584mm
- Base width of the interior panel: 390mm
- Mid width of the interior panel: 652mm
  
- Total no. of Nodes: 1800
- Total no. of Elements: 1609
  
- Displacement: Fixed: [0,0,0]
- Inertial load: Force/load: [18,0,0] [0,4.5,0] [0,0,3]
  
- Forward: 18g
- Sideward: 4.5g
- Upward: 3g

Since the forward g condition is high, the forward g condition is considered for the entire analysis.

2024 aluminium alloy is a heat-treatable Al-Cu alloy that is available in a wide variety of product forms and tempers. The properties vary markedly with temper; those in T3- and T4-type tempers are noteworthy for their high toughness, while T6- and T8-type tempers have very high strength. This alloy has excellent properties and creep resistance at elevated temperatures.

A composite material is made up of two or more materials with different chemical and physical properties. When they are combined, they create a material which is specialized to do a certain job, for instance to become stronger, lighter or resistant to electricity. They can also improve strength and stiffness. The reason for their use over traditional materials is because they improve the properties of their base materials and are applicable in many situations.

In materials science, a sandwich-structured composite is a special class of composite materials that is fabricated by attaching two thin-but-stiff skins to a lightweight-but-thick core. The core material is normally of low strength, but its greater thickness provides the sandwich composite with high bending stiffness with overall low density.

Using different thicknesses, the iterations were conducted on the interior panel and analysed at which iteration the deformation of the panel becomes less than 2mm. So that it will not affect the passengers.

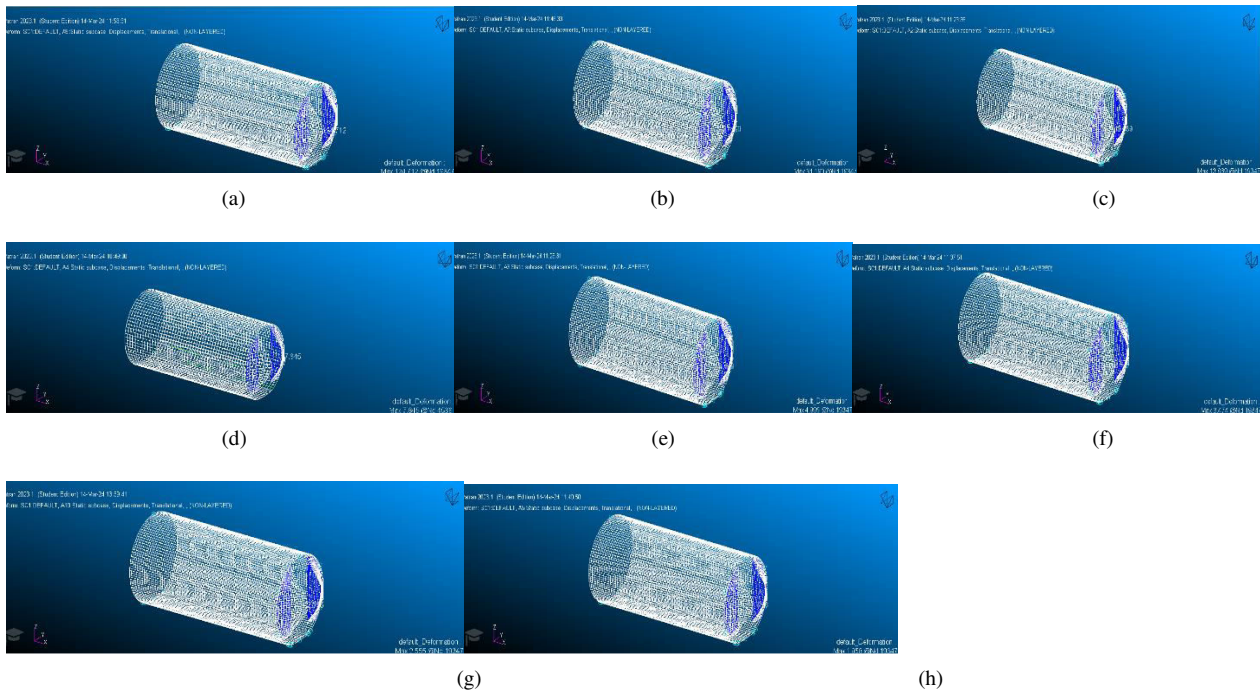


Fig. 2. Iteration by changing thickness of the panel (aluminium alloy 2024) (a) Deformation in 2mm thickness (b) Deformation in 4mm thickness (c) Deformation in 6mm thickness (d) Deformation in 8mm thickness (e) Deformation in 10mm thickness (f) Deformation in 12mm thickness (g) Deformation in 14mm thickness (h) Deformation in 16mm thickness

Now added stiffeners to the panel of thickness 2mm and by keeping the properties of the panel same. C-shaped structure are used as stiffeners in this case and the material is aluminum alloy 2024.

Stiffeners were added to the free ends of the panel in horizontal direction. These two stiffeners were having the properties and geometry as mentioned above. After that, the stiffeners were meshed in 1D meshing and assigned the properties and offsets respectively. Finally assigned the proper fixed edges to the panel and applied the boundary condition and then applied the extreme load (18g) in the forward direction. Then analysed the deformation in this condition.

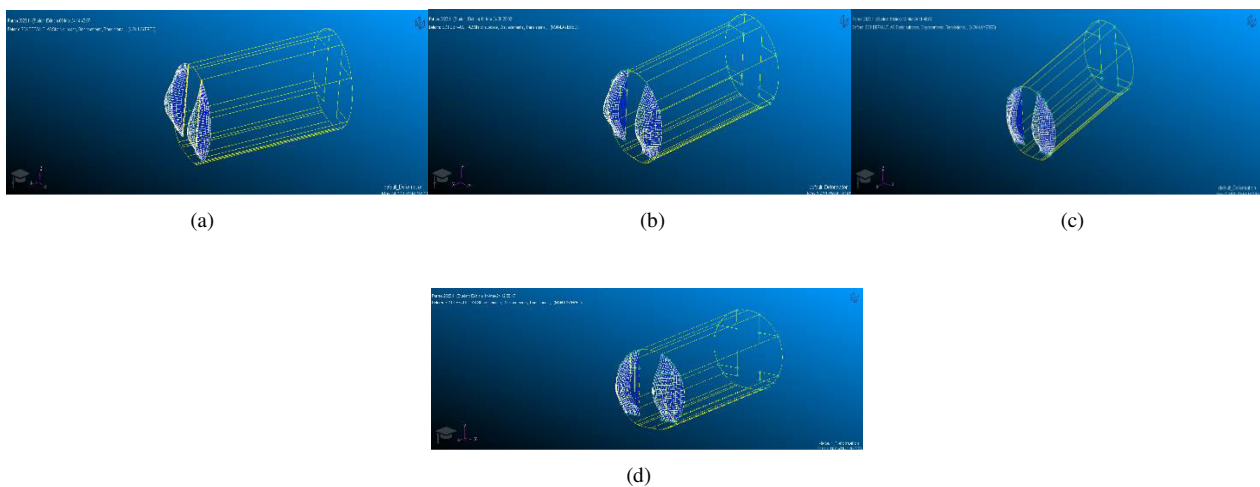


Fig. 3. Iteration by adding stiffeners in panel of thickness 2mm (aluminium alloy 2024) (a) Deformation after adding 2 stiffeners in 2mm panel (b) Deformation after adding 4 stiffeners in 2mm panel (c) Deformation after adding 6 stiffeners in 2mm panel (d) Deformation after adding 8 stiffeners in 2mm panel

Finally, the deflection becomes less than 2mm when 8 stiffeners were added to the 2mm panel. In first analysis got the minimum deflection by increasing the thickness to the panel and in second analysis got the minimum deflections in 2mm thickness of the panel by adding different stiffeners to it. In both cases got the deflections less than 2mm. Now by changing the material to composite material for analysis and got result as minimum deformation with minimum weight. So creating a sandwich structure with the glass fibre and honey-comb composites.

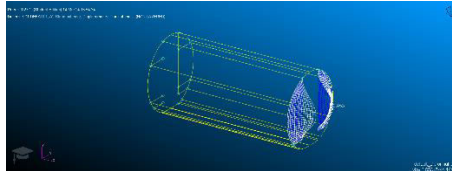


Fig. 4. Iteration by changing the material in interior panel (composite materials)

#### IV. EXPERIMENTAL RESULTS

In this project done several analysis and iterations by the finite element analysis. From each analysis got deformations less than 2mm. It's necessary to check the other values like mass, total load in order to accomplish the objectives of the project. And, the results are:

Table.1. Comparison of mass and total load on different iterations by changing thickness of the panel (aluminium 2024)

Thickness	Mass	Total Load	Deformation	Max Stress
2	8.83	159.246	124.712	5.898
4	17.69	318.484	31.190	2.943
6	26.54	477.740	13.869	1.962
8	35.36	636.4	7.854	1.448
10	44.23	794.210	4.999	1.176
12	53.08	955.480	3.474	0.98
14	61.88	1113.75	2.555	0.827
16	70.78	1273.97	1.958	0.734

When the analysis was done with different thickness of the panel, got the deformation less than 2mm in 16mm thickness of the panel. When the thickness is 16mm the total weight of the panel will be 70.78.

Table. 2.Comparison of mass and total load on different iterations by adding stiffeners in panel of thickness 2mm (aluminium 2024)

No. of Stiffeners	Mass	Total Load	Deformation	Max Stress
2	9.375	168.749	14.072	1.91
4	9.861	177.4949	3.473	0.672
6	10.32	185.7187	2.955	0.608
8	10.83	194.9079	1.93	0.409

Here adding the supporting stiffeners to the 2mm panel. Got the deflection less than 2mm, totally 8 stiffeners were added to the panel. The total weight of the structure was 10.83 when the deflection became less than 2mm.

Table. 3.Mass and total load by changing the material in interior panel (composite materials)

Total Thickness	Mass	Total Load	Deformation	Max Stress
27.4	5.642	101.5595	0.845	0.303

When the material of the 2mm thickness panel was changed to composite material the total weight of the structure becomes 5.642 when the deflection becomes less than 2mm.

## V. CONCLUSION

From the conceptual design of the fuselage an interior partition panel design was required and it has been constructed. The aim of the project is to attain the minimum deflection of the panel at the minimum weight of the structure. For that check with different configurations of analysis using finite element analysis on NASTRAN & PATRAN software. With the reference of the results, the composite material is the best material for the interior partition panel with minimum deflection at minimum weight of the structure when compared to the other material.

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