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Buckling and Static Analysis of an Aircraft's Fuselage Panel under various load Conditions.

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ABSTRACT: As compared to the ground vehicles the airplane the airplane is made up with the mid lowing structure and this is a artificial flying machine. Here the manufacturers used extremely high technologies to fly this vehicle and maintained some important parameters and guidelines to ensure the passenger safety and flying comfort ability. The main important elements of the structure of an aircraft are to transmit and support the connected loads: to give an aerodynamic shape and to secure passengers. This work is done to check the effect of static and buckling analysis of fuselage panel which is very important structure in the airplane. This is achieved by designing the structure using CATIA V5 and analysis is done by using NASTRAN.
KEYWORDS: Airplane, Static, Buckling, safety.

I. INTRODUCTION

An airplane is a mind-blowing structure, but an extremely efficient artificial flying machine. Airplanes generally have one or more particular capabilities and have been designed to ensure that they can safely perform these capabilities. Any slight disappointment on the part of either of these parties can lead to a debacle resulting in an extraordinary demolition of lives and properties. It's all about finding the ideal weight / load ratio for your vehicle

It must be robust and adequate to withstand the exceptional circumstances in which it is necessary to work. Robustness is a critical element. Similarly, if a section flattens out, it is no coincidence that it causes the disappointment of the entire aircraft. It is still possible for the plane to planar towards a protected landing point only if the shape is aligned with the current. Auxiliary honesty is performed.

The main important elements of the structure of an aircraft are to transmit and support the connected loads: to give an aerodynamic shape and to secure passengers. Payload frames, etc., from the ecological conditions experienced in flight These prerequisites, in most aircraft, offer the upgrade to thin-film structures, in which the outer surface of the skin or carapace is generally reinforced by individuals with fortification longitudinal and transverse ribs to counteract the bending and compression loads without distorting the torsion. These structures are known as semi-monocoque, while the delicate shells that depend entirely on your skin for their carrying capacity are known as monocoque.

II. LITERATURE SURVEY

William L. Ko and Raymond H. Jackson[2] displayed Investigation on buckling performed on cap –stiffened board subjected to uni-axial stacking. Different progressed hot basic board ideas have been researched for hypersonic aircraft wing boards and found that beaded boards and tubular boards were exceedingly effective. It was found that worldwide buckling burden was higher than the nearby buckling load. Compressive nearby buckling burdens were marginally lower than the quality anticipated from FEM buckling analysis and exploratory worth, the reason might be the presence the fortifications at the board edges, which were disregarded in the analysis. The worldwide buckling anticipated compressive buckling load application. Three times more than the qualities anticipated from buckling speculations. In this manner, cap solidified board will clasp locally rather than internationally.

D Quinn1 et al., [3] concentrated on the possibility to present buckling control highlights without corrupting board toughness and to produce trial Data which can be utilized to approve demonstrating procedures to anticipate break

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development through both buckling and split spread regulation elements. This study illustrates, through exploratory analysis, that buckling regulation component board plans, determined by stationary quality and dependability, can likewise yield enhanced split engendering conduct, basically proposing the possibility to adapt board skin-sound geometry for together stationary quality and weakness life.

Mustafa Osaka et al.,[4] researched The (post)buckling execution of boards by sub-solidifying or nearby customizing of skin depth ("covering chiselling") utilizing in lines variable depth finite stripe analysis, (nonlinear-) finite element investigation and tests on solidified boards. The study displayed that Collapse loads and post buckling solidness anticipated by the FE Simulations were in close concurrence with the examination, yet introductory buckling burdens were up to 30% lower than measured furthermore linear finite strip analysis permitted streamlining of the design with cushion ups and a solitary sub stiffener between stiffeners, uncovering possible for additional change of the underlying buckling weight by more than 10%. Plan rules for sub-solidified boards remained determined.

III. OBJECTIVES

- To determine the maximum stress developed for the different design loads considered.
- To decide the buckling factor for the numerous design weights considered in the work.
- To determine the maximum design load at which the panel undergoes buckling to be considered for the current configuration of the panel.

IV. METHODOLOGY

- CAD model of fuselage section of Aircraft is designed by using CATIA V5 software.
- The designed CAD model is then converted into FE model by meshing the CAD model with the help of FE pre-processor Hyper Mesh.
- Once meshing is done then assign the material properties and assign the thickness to the model.
- The boundary conditions and the loads are applied as per the chosen flight condition.
- The final analysis of the model with the given boundary conditions and the load applications is done by using NASTRAN.

By using Hyper View the results are extracted after completing analysis and documentation is also done.

V. FINITE ELEMENT ANALYSIS

In this chapter, we are dealing with geometric model preparation, Finite element model preparation and analysis of the FE model with loads and Boundary conditions.

Geometric modelling:-

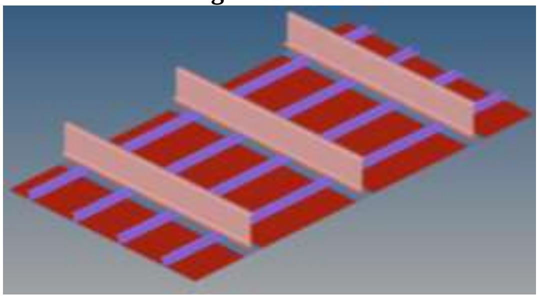

Figure 5.1: Fuselage Skin panel with stiffeners

Figure 5.1 shows the geometric model of a fuselage skin panel. The modelling of Fuselage panel is done by using CATIA V5. It consists of skin, crack stoppers, panel stiffeners and bulkhead assembly. A mouse hole has been inserted in the bulkhead assembly to facilitate the insertion of the panel stiffeners. For model simplification, all fillets and chamfers below 3mm are neglected and the final model is represented in the above figure. Thermal expansion is allowed by dividing the skin into different sections.

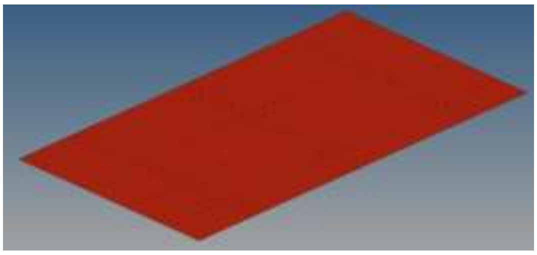

Figure 5.1: Fuselage panel skin

Figure 5.2 shows the CAD model of a fuselage skin panel. The dimensions of the skin panel are 3556 x 1905 x 1.8 mm. All components are connected using rivets and rivet holes are modelled into the panel to facilitate the same. The diameter of the rivet holes is 4.4 mm and is 25.4 mm or 1in apart from each other.

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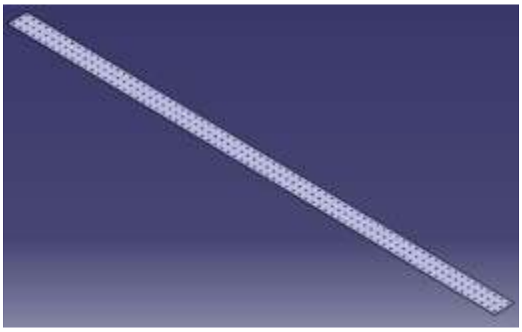

Figure 5.2: Crack Stopper

Figure 5.3 show the CAD model of the crack stopper used in the fuselage panel. The crack stopper is placed in between the skin panel and the bulkhead assembly to arrest the formation of cracks due to repeated loads on the skin and finally fracture. The diameter of the rivet holes is 4.4 mm and is 25.4 mm or 1in apart from each other.

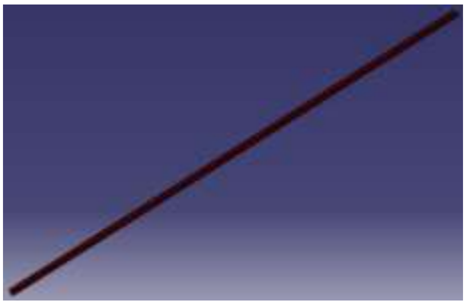

Figure 5.3: Hatch stiffener or panel stiffener

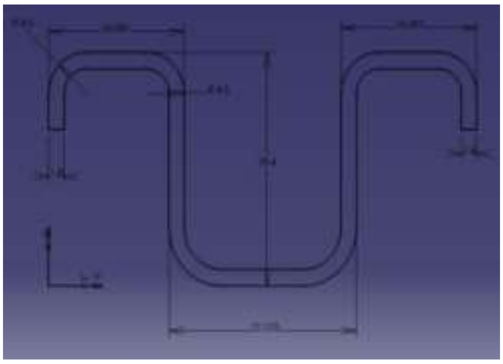

Figure 5.4: Hat section Dimensions

Figure 5.4 show the CAD model of the panel stiffener used to give strength to the skin panel along the length of the skin. The length of the stiffeners is equal to the length of the skin panel. For the current work, the panel length is taken as 3556 mm. The diameter of the rivet holes is 4.4 mm and is 25.4 mm or 1in apart from each other. Figure 5.5 show the geometry of the panel stiffener used in the present work.

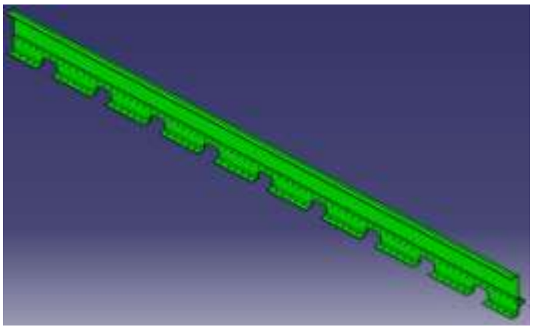

Figure 5.5: Bulkhead

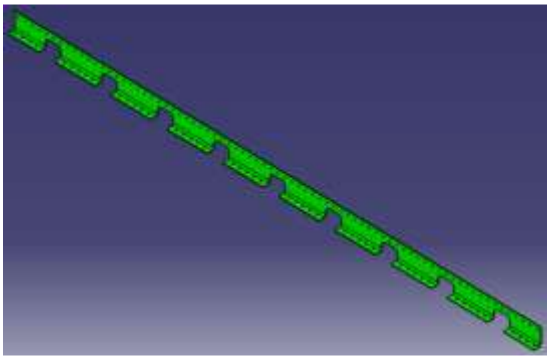

Figure 5.6: L-section used in a bulkhead

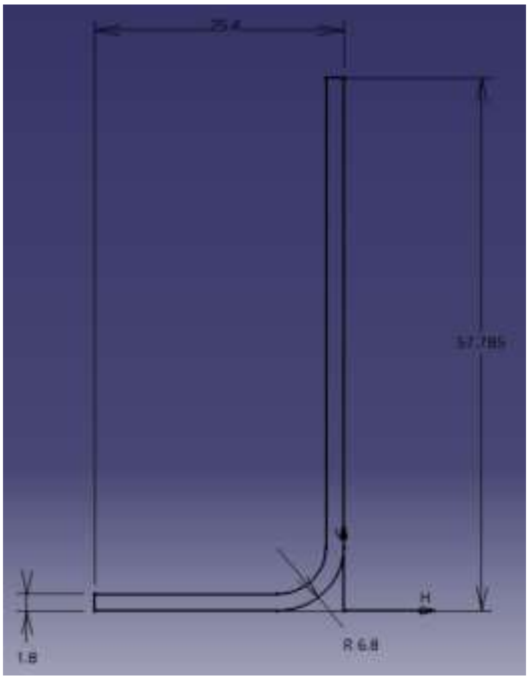

Figure 5.7: L-section dimensions

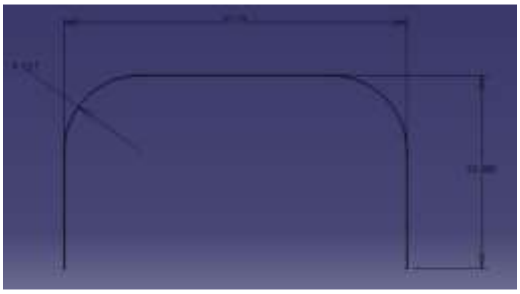

Figure 5.8: Rat hole cutout in the L-section

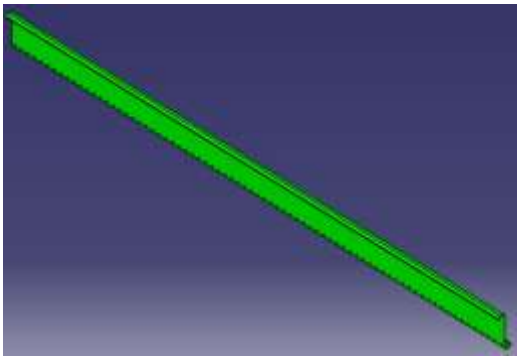

Figure 5.9: Z-section used in a bulkhead

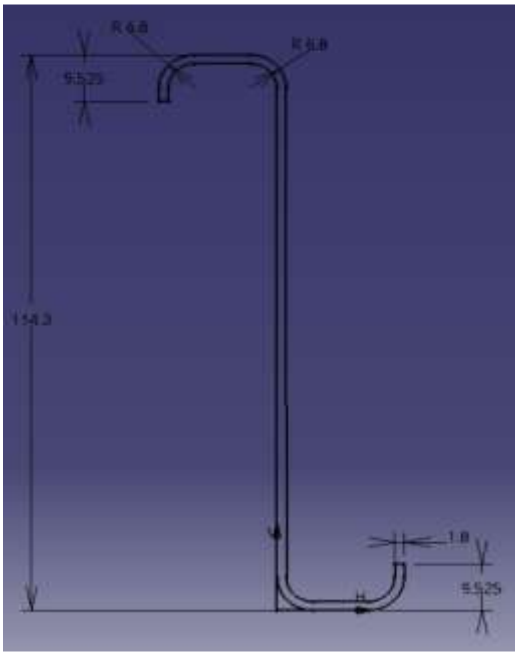

Figure 5.10: Z-section dimensions

VI. FINITE ELEMENT MODELING

Figure 6.1: FE Model of the fuselage pane

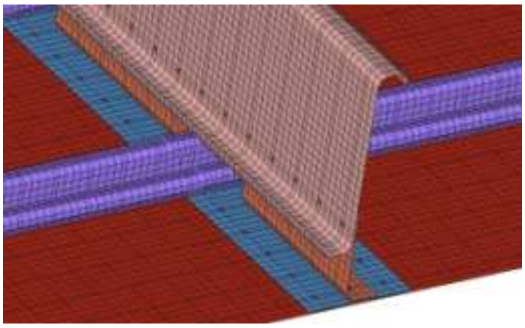

Figure 6.2: close view of the fuselage panel components.

Figure 6.1 shows the Finite element model of fuselage structure generated for the geometric model which is done using Quad 2D elements. These elements are chosen because the large dimensions of the structure and one of the dimensions of all components are relatively smaller than the other two dimensions. Each component is considered a separate entity and then fastened using 1D rigid elements that replace the rivets used to connect all components in the real model.

Table 6.2 gives the material composition of the aluminium alloy which is mainly used to manufacture the airplane parts. The material is aluminium alloy Al7075 and the mechanical properties are shown in table 4-3. The material properties which are required to the current investigation are Young’s Modulus, density, Ultimate strength and Poisson’s ratio. This is because the analysis is of structural nature and other properties are not considered in the analysis.

Below Table gives the number of nodes and elements generated during the discretization.

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Table 6.1: Elements and nodes count

Nodes	115544
Elements	110620

VII.MATERIAL PROPERTIES

Aluminium alloys Al7075 and Al2024-T351 are two materials which are used in manufacturing of Aircrafts because of their wide range of properties. Their material properties are as given below.

Aluminium alloy Al7075-T6

Aluminium with zinc as a main component in it is termed as the Aluminium alloy Al7075-T6. This aluminium alloy is a strong component, with similar strength as compared to various steels. This aluminium alloy has fatigue strength and ordinary machinability however it has fewer struggles to erosion than many other Al alloys. Comparatively it has high rate boundaries and it can be used where low quality alloys cannot be used.

Al7075is mainly consist of 5.6-6.1% zinc, 2.1-2.5% magnesium, 1.2-1.6% copper and other materials like silicon, iron, manganese, titanium, chromium all combined in the range less than half percentage. Aluminium7075 is made in various models; some of those are 7075-0, 7075-T6, 7075-T651.

Chemical composition of Aluminum Al7075-T6 is given in Table 4.2

Table 7.1: Material Composition of Al7075-T6

Component	**Weight (%)**
Aluminium	87.1 - 91.4
Copper	1.2 - 2
Chromium	0.18 - 0.28
Ferrous	Max 0.5
Magnesium	2.1 - 2.9
Manganese	Max 0.3
Titanium	Max 0.2
Zinc	5.1 - 6.1
Silicon	Max 0.4

Table 7.2: Mechanical properties of Aluminum alloy 7075-T6 are as shown below

Young's modulus	71700 N/mm^2
Poisson's ratio	0.33
Density	2.81 X 10^{-6} kg/mm^3
Tensile yield strength	503 N/mm^2
Tensile ultimate strength	572 mm^2

Aluminium alloy Al2024-T351

Aluminium alloy 2024 is an alloy of aluminium and copper as the main component of the alloy. It is used in applications that require a high strength / mass ratio and also good fatigue strength. it is simply welded when friction welded and has medium machinability. Due to the reduction in opposition, a layer of aluminium or Al-1Zn is often applied as a shield, although this can reduce the fatigue strength. In ancient terminology systems, this league was called 24ST. 2024 is usually extruded, also available in aluminum foil and plate. it is generally not fake.

The chemical composition of Aluminium Al2024-T351 is given in Table 7.3

Table 7.3: Material Composition of Al 2024-T351

Component	Weight (%)
Aluminum	90.7-94.7
Copper	3.8-4.9
Magnesium	1.2-1.8
Ferrous	Max 0.5
Chromium	Max 1

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Manganese	0.3-0.9
Titanium	Max 0.15
Zinc	Max 0.25

Table 7.4: Material Properties in Al 2024-T351

Properties	Material
Density	2.78 X 10^{-6} kg/mm^3
Poisson's ratio	0.3
Young's modulus	73100 N/mm^2
Tensile yield strength	345 N/mm^2
Tensile ultimate strength	483 N/mm^2

Below Figure 7.1 shows the loads and boundary conditions applied on the fuselage structure. When the aircraft is flying at the height of 3000m (10,000ft) and moving at a speed of 250km/hr the ends of the centre fuselage are fixed and to simulate the air pressure acting on fuselage pressure load is applied for this analysis.

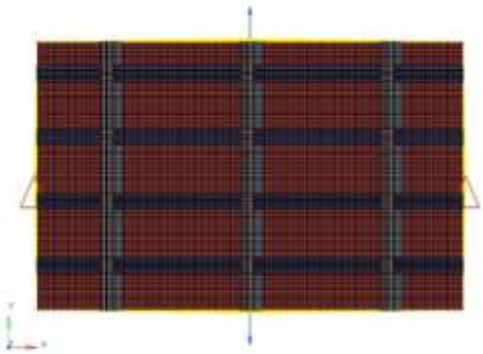

Figure 7.1: Weights and boundary conditions applied on the model

VIII. RESULTS AND DISCUSSION

The designed model was converted into FE model then the boundary conditions are given to the converted FE model and then the loads are given to the test model as mentioned previous section. Then for given conditions the results are obtained for static analysis and buckling analysis. Static analysis results are mainly displacement and stresses, buckling analysis contains buckling mode.

A. FOR LINEAR STATIC ANALYSIS

For Design load – 1g: -

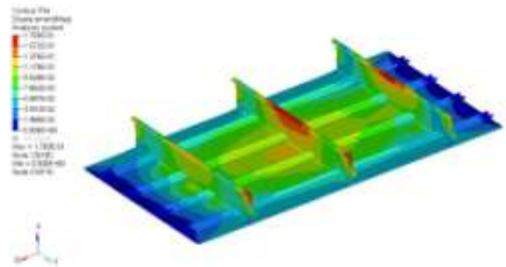

Figure 8.1: Displacement contour plot for fuselage panel for 1g load

Figure 5.1 shows the displacement contour plot for the aircraft fuselage panel under 1g loading condition. The centre stiffeners gets displaced maximum as they are not constrained. The maximum displacement value is 0.1769 mm.

Figure 8.2: Contour stress plot for fuselage panel at 1g load

Figure 8.2 shows the contour stress plot for the fuselage panel under 1g loading condition. The maximum stress recorded here is 163.6 MPa, this stress is recorded near the edges of the panel as there is a circumferential direction. The maximum stress recorded here is less the yield strength of the material used for the fuselage panel.

For Design load – 3g: -

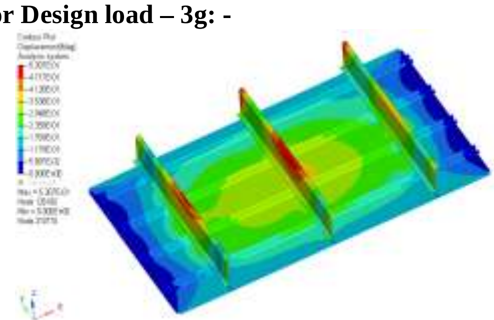

Figure 8.3: Displacement contour plot for fuselage panel at 3g load

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Figure 8.3 shows the displacement contour plot for the fuselage panel under 3g loading condition. The maximum displacement value recorded here is 0.5307 mm. The maximum displacement is found in the stiffeners as they are not constrained.

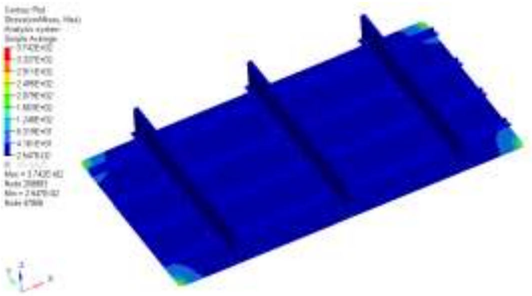

Figure 8.4: Stress contour plot for fuselage panel at 3g load

Figure 8.4 shows the stress contour plot for the fuselage panel under 3g loading condition. The maximum stress recorded here is 374.2 MPa, this stress is recorded near the edges of the panel as there is a circumferential direction. The maximum stress recorded here is less the yield strength of the material used for the fuselage panel.

Design load – 7g: -

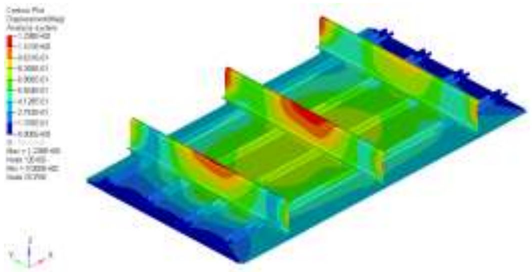

Figure 8.5: Displacement plot for fuselage panel at 7g load

Figure 8.5 shows the displacement contour plot for the fuselage panel under 7g loading condition. The maximum displacement value recorded here is 1.238 mm. The maximum displacement is found in the stiffeners as they are not constrained.

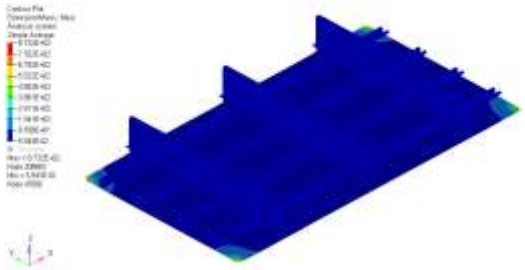

Figure 8.6: Stress plot for fuselage panel at 7g load

Figure 8.6 shows the stress plot for the fuselage panel under 7g loading condition. The maximum stress recorded here is 873.2 MPa, this stress is recorded near the edges of the panel as there is a circumferential direction. The maximum stress recorded here is more the yield strength of the material used for the fuselage panel.

B. RESULTS FOR BUCKLING ANALYSIS

For Design load – 1g: -

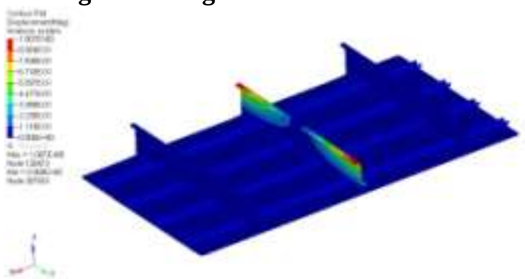

Figure 8.7: Buckling mode contour plot for fuselage panel at 1g

Figure 8.7 shows the buckling mode contour plot for the fuselage panel under 1g loading condition. For 1g load the buckling factor of the panel is seen to be 27.401. The buckling factor got here is above the value 1 and this value is considered as the standard for measuring the buckling of any model.

For Design load – 3g: -

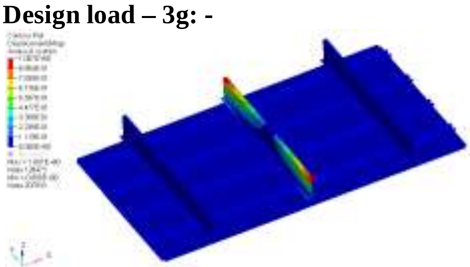

Figure 8.8: Buckling mode contour plot for fuselage panel at 3g load

Figure 8.8 shows the buckling mode contour plot for the fuselage panel under 3g loading

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condition. For 3g load the buckling factor of the panel is seen to be 9.133. The buckling factor got here is above the value 1 and this value is considered as the standard for measuring the buckling of any model.

For Design load – 7g: -

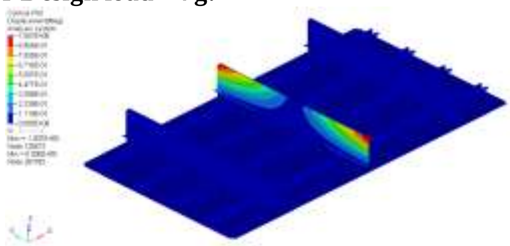

Figure 8.95: Buckling mode contour plot for fuselage panel at 7g load

Figure 8.9 shows the buckling mode contour plot for the fuselage panel under 7g loading condition. For 7g load the buckling factor of the panel is seen to be 3.915. The buckling factor got here is above the value 1 and this value is considered as the standard for measuring the buckling of any model.

The buckling factor which is determined during this buckling analysis determines the safety of the component.

Table 8.1: Buckling factor

Standard Buckling factor	Remarks
< 1	Leads to Failure
= 1	Modification required
> 1	Safe design

C. FATIGUE ANALYSIS

The fatigue life estimation in this work has been carried out using the S-N curve with the Goodman's equation. Life estimation has to be performed for the skin panel and the stiffeners separately as the material for both the components are different. Figure 5.13 shows the S-N curve for the material Al7075 and figure 5.14 shows the S-N curve for Al2024-T3.

From figure 5.14, the fatigue life of the material corresponding to the value of σ_e is below the stress value which implies infinite life for the component.

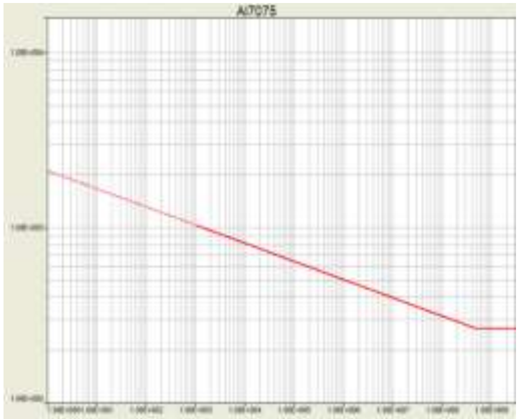

Figure 8.10: S-N curve for Al7075

From figure 5.14, the fatigue life of the material corresponding to the value of σ_e is below the stress value which implies infinite life for the component.

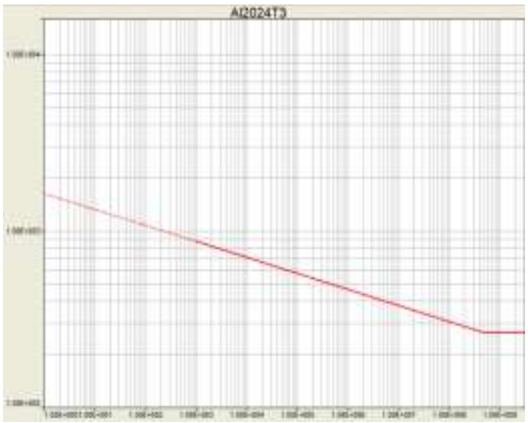

Figure 8.11: S-N curve for Al2024-T3

IX. CONCLUSION

Here in this work the fuselage panel which is a main part of an aircraft is considered. First this fuselage panel is modelled using CATIA V5 and then descretized using meshing and then finally did analysis to get results for static and buckling strength because of varying loads. Based on the analysis results which we get it can be concluded that

- The displacement that occurs in the fuselage panel for all given load conditions is minimal and has no significant effect on the integrity of the panel.
- By conducting linear static analysis we come to know that the under given conditions the panel can withstand the maximum design load of 5g by considering the loads..
- With a load of 5g, the stress level in the panel exceeds the elastic limit of the material considered and small changes of the panel

which can reduce the stress value so that it is within the elastic limit of the material.

- By conducting instability or buckling analysis of an aircraft fuselage panel under the given foundry conditions and load conditions revealed that the fuselage panel has a capacity to withstand more than 7 g of load even if the stress value at that load exceeds the elastic limit of a given material.

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