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Buckling Analysis Of Aircraft Stiffened Panel Cylindrical Shells

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Abstract: **For past decades devolopment of composite material had brought tremendous change in AirFrames and Missiles. Due to composite stiffened panels implementation speed, distance and life of the Airframes and missiles had increased. During integration the composite cylindrical shell structure is provided with cutouts which reduce the strength of composite cylindrical shell and are porne to buckling. By adding Reinforcement around cutout will lead to improvement of strength. Generally T-section and I-section stiffened panel are used, but there is a disadvantage of using T-section, it can't resist to deformation. So we designed an I-section stiffened panel (because I-section is more resistant to deformation) in CATIA and analysis in ANSYS. We considered three types of analysis in ANSYS, Modal analysis, Static analysis, Harmonic analysis respectively and also we considered two materials, one is aluminum and the other is carbon fiber. Aluminum is the common element used in the design of aircraft, but Carbon fiber is recently being used in aircrafts.**

Key words: **1st Principle Stress; 2nd Principle Stress; Von Mises Stress;**

I. INTRODUCTION

Generally in aircrafts there are two types of structures Monocoque and Semi Monocoque. Monocoque structure is a structural approach that supports loads through an object's external skin, where as the semi Monocoque system uses a substructure to which the airplane's skin is attached. The substructure, which consists of bulkheads and formers of various sizes and stringers, reinforces the stressed skin by taking some of the bending stress from the fuselage. The semi Monocoque is the most often used construction for modern, high-performance aircraft. Hence in the aircrafts today semi Monocoque structure is used. In these semi Monocoque structure components like bulk heads, formers, stringers, stiffeners, ribs, spars, etc are present. Among these components we have selected stiffener component as it carries the maximum load, in fuselage the stiffener is called as stringer also. We have selected the fuselage stiffener for our project.

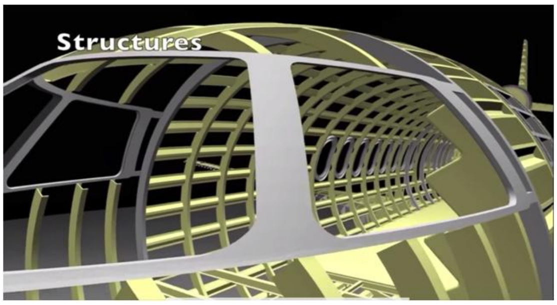

Figure:1 Structure

Literature:

E. Stein et.al(1); gone through analysis of nonlinear elastic shells often the stability and postbuckling behaviour governs the response. And also discused problems which also include contact constraints. In their study a nonlinear cylindrical shell element is derived directly from the associated shell theory using one point integration and a stabilization technique. J.c.simo et.al(2); concerned with the numerical solution of large deflection structural problems involving finite strains, subject to contact constraints and unilateral boundary conditions, and exhibiting inelastic constitutive response. For this a three-dimensional finite strain beam model is summarized, and its numerical implementation in the two-dimensional case is discussed by them. A. E. Mohmedet.al(3); In their study, Lagrangian formulations for geometric nonlinear plane stress/strain problems based on different stress measures are evaluated based on the exact

Engineering strains is developed. Geometric nonlinear Total Lagrangian formulations applied on two-dimensional elasticity using 4-node plane finite elements were used. The formulations were implemented into the finite element program (NUSAP) and nonlinear equations was obtained by the Newton-Raphson method. Oludele Adeyefa et.al(4); Their research work focussed on flexural-torsional buckling of beam-column supports of field fabricated spherical pressure vessels using finite element analysis. This research has therefore considered the total potential energy equation for the flexural-torsional buckling of a beam-column element. The energy equation was formulated by summing the strain energy and the potential energy of the external loads. The final finite element equation obtained was in the form of an eigenvalue problem were de-termined by solving for the eigenvalue of the equation. The resulting eigenvalue equation from the finite element analysis was coded using FORTRAN 90 programming language to aid in the analysis process. R. Santhanam et.al(5); has been carried out Analysis of monocoque and semi-monocoque cylindrical flight vehicle structures by using finite element method. Shell elements are used for idealizing skin portions and end rings and beam elements are used for idealizing stiffeners. The behaviour of these structures is compared in terms of mass, deformation, stress and buckling under structural and thermo-structural loads to study the effect of number of longitudinal stiffeners. The study shows that semi-monocoque structures give higher factor of safety and buckling load factor when only structural loads.

II. PROBLEM DEFINITION

Aircrafts are the fastest means of transportation for several years, capacity of carrying people and goods are increased from the beginning this is because of introduction of light weight materials with great strength. Among all the material Aluminum served for long period, development of Nano technology begins the new era of composite material which is more suitable and more lighter then the material with-in the economic. But, now days the aircrafts are losing their strength and are getting deformed very easily due to various loads that act on the aircraft. Hence main motto of our project is to reduce the deformation caused due to application of various loads like the pressure loads, gust loads, compressive loads, buckling loads etc. We are first designing a stiffened panel model in CATIA and then analyzing the same model in ANSYS.

The main objective our paper is to reduce the deformation , deflection , and buckling caused due to application of various loads and also to increase the stability and strength by changing materials, sections, adding an additional part or removing a part. Stiffeners are secondary plates or sections which are attached to beam webs or flanges to stiffen them against out of plane deformations. In aircraft construction, a longeron or stringer or stiffener is a thin strip of material, to which the skin of the aircraft is fastened. In the fuselage, stringers are attached to formers (also called frames) and run the longitudinal direction of the aircraft. They are primarily responsible for transferring the loads (aerodynamic) acting on the skin onto the frames/ formers. In the wing or horizontal stabilizer, longerons run span wise and attach between the ribs. The primary function here also is to transfer the bending loads acting on the wings onto the ribs and spar. Sometimes the terms "longeron" and "stringer" are used interchangeably. If the longitudinal members in a fuselage are few in number and run all along the fuselage length (usually 4 to 8), then they are called "longerons". If the longitudinal members are numerous (usually 50 to 100) and are placed just between two formers/frames, then they are called "stringers". Longerons often carry larger loads than stringers and also help to transfer skin loads to internal structure. Stiffeners are to control buckling.

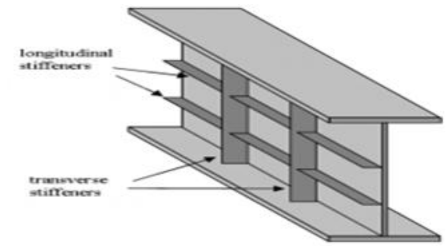

Figure 2: (a) Stiffeners on I-section grid

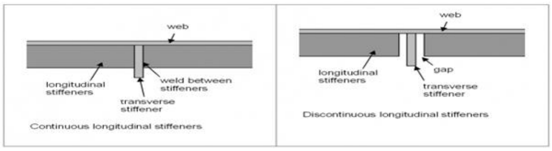

Figure2: (b) Continuous and Discontinuous longitudinal stiffeners

III. DISADVANTAGES OF EXISTING STIFFENED PANEL

Aluminum is the most commonly used material for many aircrafts. But pure aluminum is completely unsuitable as structural materials for airframes, because they have very low strength. However, when alloyed (chemically mixed) with other metals include zinc, copper, manganese, silicon and lithium….etc. and may be used singly or in combination metals, their strength is vastly improved, and they form the most widely used group of airframe materials. But Aluminum alloys are more prone to corrosion than pure aluminum, so pure aluminum is often rolled onto the surfaces of its alloys to form a protective layer. The process is known as cladding, and sheets of alloy treated like this are known as clad sheets or Al-clad. Another common means of protecting aluminum alloys is anodizing - conversion of the surface layer to a form which is more corrosion-resistant by an electro-chemical process. Their use is limited because they are around three times as expensive.

Selecting the optimum material for a specific application meant analyzing every area of the airframe to determine the best material, given the operating environment and loads that a component experiences over the life of the airframe. For example, aluminum is sensitive to tension loads but handles compression very well. On the other hand, composites are not as efficient in dealing with compression loads but are excellent at handling tension. The expanded use of composites, especially in the highly tension-loaded environment of the fuselage, greatly reduces maintenance due to fatigue when compared with an aluminum structure. This type of analysis has resulted in an increased use of carbon fiber as well. Where loading indicates metal is a preferred material system but environmental considerations indicate aluminum is a poor choice, carbon fiber is an excellent low-maintenance design solution. It can withstand comparable loads better than aluminum, has minimal fatigue concerns, and is highly resistant to corrosion.

The properties of carbon fibers, such as high stiffness, high tensile strength, low weight, high chemical resistance, high temperature tolerance and low thermal expansion, make them very popular in aerospace. Here are some researches made on stiffened panel

<u>1st Principle Stress</u>

The 1st principal stress gives you the value of stress that is normal to the plane in which the shear stress is zero. The 1st principal stress helps you understand the maximum tensile stress induced in the part due to the loading conditions.

<u>2nd Principle Stress</u>

The Stress in second principal direction result shows the second principal stress in the part at the selected layer through the cross-section, after ejection. Positive values correspond to tension in the part, and negative values to compression

<u>Von Mises Stress</u>

Von Mises stress is a geometrical combination of all the stresses (normal stress in the three directions, and all three shear stresses) acting at a particular location. Since it is a stress, it is measured in Pascal's, just like any other type Von Mises stress is useful for materials which classify as ductile. If the Von Mises stress at a particular location exceeds the yield strength, the material yields at that location. If the Von Mises stress exceeds the ultimate strength, the material ruptures at that location. For brittle materials, the Von Mises stress concept isn't applicable. Instead, maximum principle stress (normal stress on the plane at which it is maximum) is what is used to predict failure.

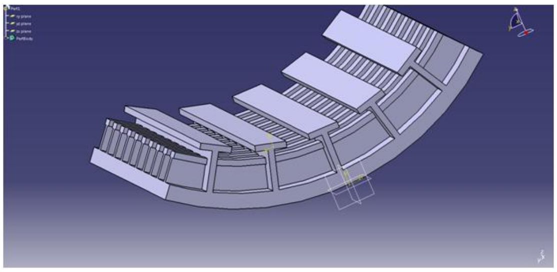

Figure 3: T-section model

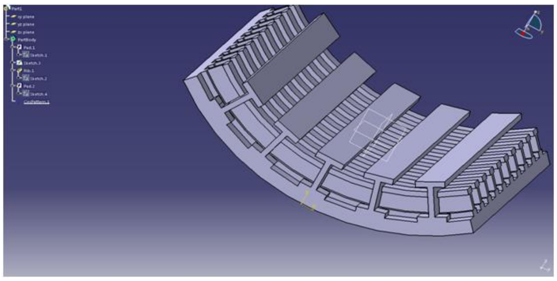

Figure 4: I- section model

Repeat the whole process with another material i.e. carbon fiber reinforced plastic with EX=150 e3, PRXY=0.25 and density=1.72e-6

<u>1st Principle Stress</u>

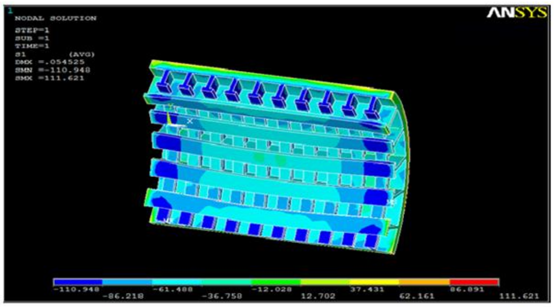

Figure.5 Result of 1st principle stress of T-section (Aluminum)

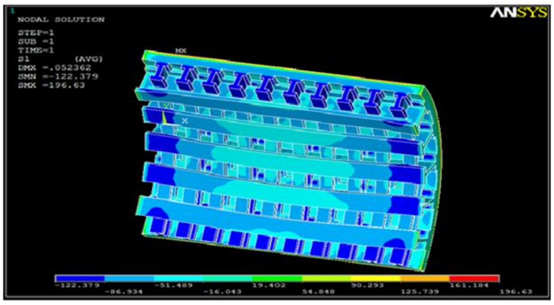

Figure.6 Result of 1st Principle Stress of I- section (Aluminum)

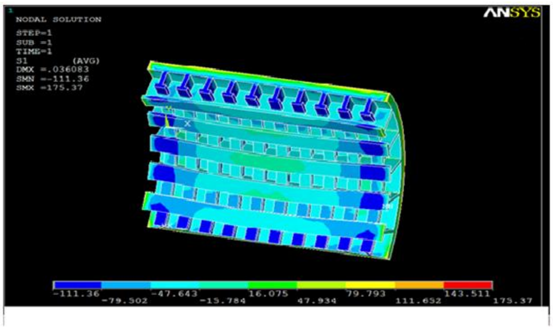

Figure 7: Result of 1st Principal Stress of T-section (Carbon Fiber)

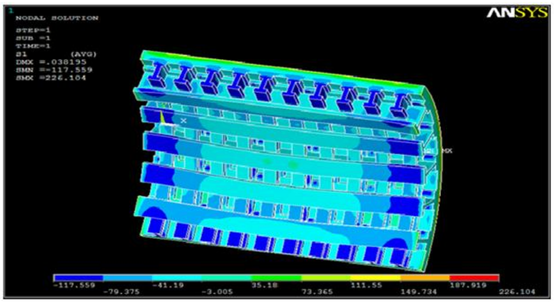

Figure 8: Result of 1st Principal Stress of I-section (Carbon fiber)

Repeat the whole process with another material i.e carbon fiber reinforced plastic with EX=150 e3, PRXY=0.25 and density=1.72e-6

2nd Principle Stress

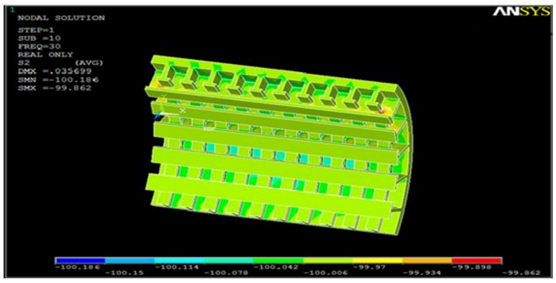

Figure 9: Result of 2nd Principle Stress of T – Section (Aluminum Material)

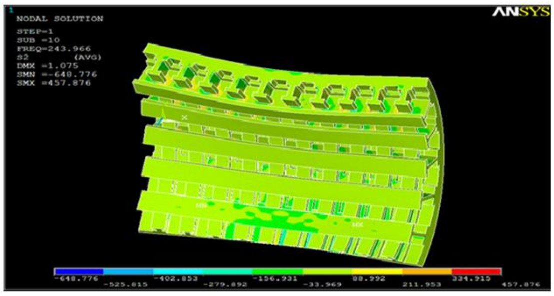

Figure 10: Result of 2nd Principle Stress of I–Section (Aluminum Material)

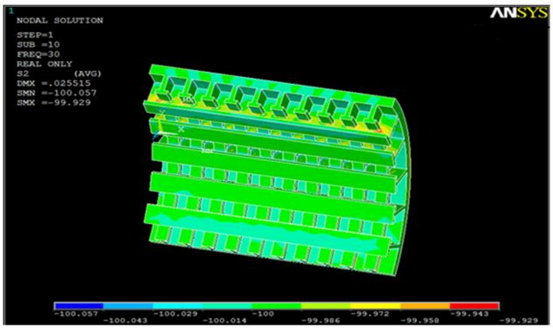

Figure 11: Result of 2nd Principle Stress of T–Section (Carbon Fiber Material)

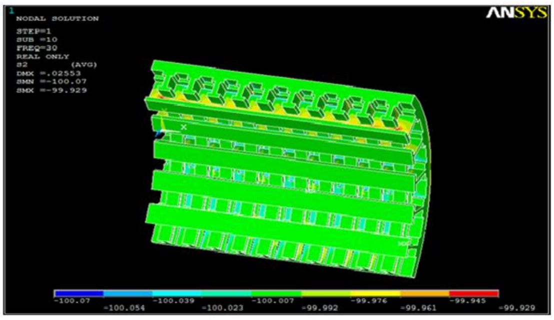

Figure 12: Result of 2nd Principle Stress of I–Section (Carbon Fiber Material)

5. Von Mises Stress

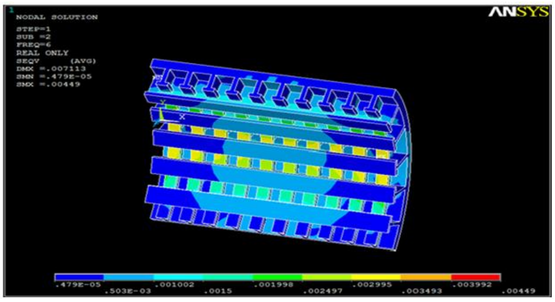

Figure 13: Result of Von mises stress of T-Section (Aluminum Material)

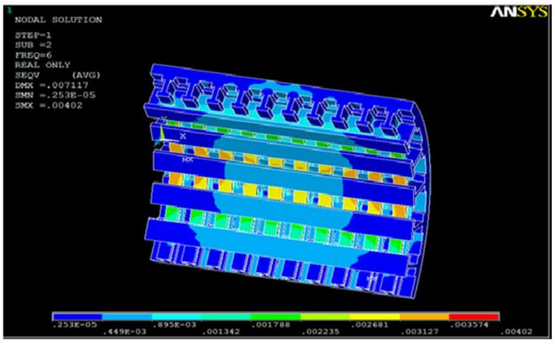

Figure 14: Result of Von misses stress of I-Section (Aluminum Material)

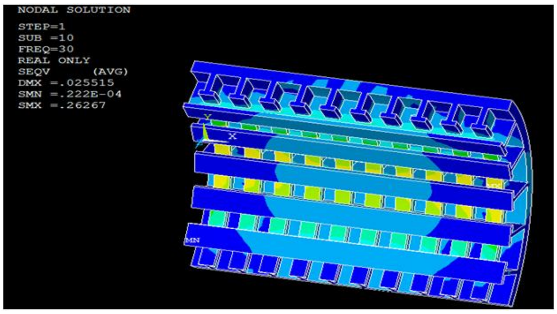

Figure 15: Result of Von mises stress of T-Section (Carbon Fiber Material)

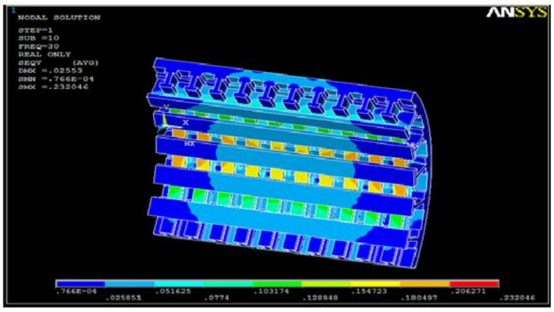

Figure 16: Result of Von misses stress of I-Section (Carbon Fiber Material)

IV. RESULT

The table below shows the analytical values of I-section and T-section of aluminum and carbon fiber

Sections	I-section (Aluminium material)			T-section (Aluminium material)			I-section (Carbon Fiber material)			T-section (Carbon Fiber material)		
	Static	Modal	Harmonic	Static	Modal	Harmonic	Static	Modal	Harmonic	Static	Modal	Harmonic
Deflection Result (mm)	0.052	1.08	0.007	0.054	1.086	0.007	0.038	1.059	0.025	0.036	0.02	0.025
Maximum Stress Result (N/mm²)	196.6	1389	0.004	111.62	2237	0.00449	226.104	88.016	-99.892	175.37	-99.89	-99.85
Minimum Stress Result (N/mm²)	-122	-521	$2.50E^{-6}$	-110.9	-433.6	$4.79\ E^{-6}$	-117.59	-13.48	-110.07	-111.3	-100.5	-100.025

Table 1(a) Comparison of different sections and materials

V. CONCLUSION

Generally, the material that is used in the construction of aircraft is aluminum. But now the bigger aircraft companies like Boeing Airbus have already started using carbon fiber material also for their aircraft. So we tried to compare the two materials i.e. aluminum and Carbon fiber and through ANSYS found out the results that which material can withstand the loads applied and have less deformation. So below is the comparison theoretically and analytically.

Aluminum gets deformed easily with some amount of loads where as Carbon fiber doesn't get deformed easily with less loads. At high temperatures Aluminum strength decreases unlike that carbon fiber is heat resistant and when the temperature is above 100, aluminum gets very much affected. Physical strength, toughness and light weight are the features of carbon fiber. Carbon fiber also has good vibration damping, chemical conductivity compared to aluminum. The properties of carbon fibers, such as high stiffness, high tensile strength, low weight, high chemical resistance, high temperature tolerance and low thermal expansion, make them very popular in aerospace. Aluminum has some disadvantages like they are Prone to corrosion, so need protective finishes, particularly magnesium alloys Many alloys have limited strength, especially at elevated temperatures .

When we compare density, then aluminum is denser than carbon fiber, aluminum density is about 2700 kg/m3 and carbon fiber density is 1500kg/m3. Therefore carbon is much lighter and young's modulus for aluminum is around 70-79 mpa and where as for carbon fiber it is 150 mpa. We know that young's modulus measures the resistance of a material to elastic (recoverable) deformation under load. So the material with high young's modulus changes its shape slightly under elastic loading. Poisson's ratio for aluminum is 0.33 and where as for carbon fiber it is 0.25. The ratio of lateral strain by longitudinal strain is Poisson's ratio. So material with less possion's ratio has less deformation.

From the analysis we found that carbon fiber is more robust than the aluminum material; also found that I-section gives less deformation than that of T-section

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