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LINEAR-BUCKLING ANALYSIS OF CYLLINDRICAL SHELLS SUBJECTED TO EXTERNAL PRESSURE

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ABSTRACT

The main focus of this paper is to understand the nature of these laminated composite shell structures when subjected to external loads. In order to understand the progression of the failure modes in a laminated composite shell structures, models were designed and analyzed using ANSYS WORKBENCH. In the present work composite cylindrical shell is designed which can withstand an external pressure of 5 bar. Stress analysis of the cylindrical structure is done using classical laminate theory and the result obtained is validated using finite element analysis procedure using ANSYS WORKBENCH. The behavior of the composite cylinder is checked by exploring the stresses and strains. The present work includes determination of optimum design parameter like fiber orientation, ply thickness and sequence. Finally, utilizing the finite element modeling of a cylindrical shell specimen, a relative comparison is made between the results of the finite element and the analytical method. To check the health of the composite conical shell structure, failure criteria's like Tsai-wu and Maximum stress criteria were also calculated in this project.

Keywords: *Buckling, composite laminate theory, fiber ply, layup pattern, orthotropic, hoop stress, ANSYS ADPL.*

I. INTRODUCTION

A composite material is made by combining two or more materials – often ones that have very different properties. The two materials work together to give the composite unique properties. However, within the composite you can easily tell the different materials apart as they do not dissolve or blend into each other.

Composite materials (also called composition materials or shortened to composites) are materials made from two or more constituent materials with significantly different physical or chemical properties that when combined, produce a material with characteristics different from the individual components. The individual components remain separate and distinct within the finished structure.

Fiber + Resin =Fiber reinforced composite

A typical composite material is a system of materials composing of two or more materials (mixed and bonded) on a macroscopic scale. BIPIN. P.B et al [1] conducted a study of understanding the buckling response of laminated rectangular plates with clamped-free boundary conditions is carried out. The laminated composite plates have varying L/W ratio, aspect ratio. H.-R. MEYER et al [2] investigated on the buckling loads of cylinders which are imperfection-sensitive under axial loading may not be so sensitive to combined loads. JIABIN SUN et al [3] discussed about a very effective Hamiltonian system constructed within a simplistic space for buckling of cylindrical shells subjected to a combination of pressure, torsion and axial compression is established. GEORGE J SIMLTSES [4] reviewed about the buckling of laminated configuration and discussed that with the advent of fiber-reinforced composites are continuously being replaced by laminated configuration with or without stiffening. Analysis of FRP composite cylinder has been studied by S. BHAVYA et.al [5]. The laminates considered for the analysis are four-layered angle-ply and six layered laminate covered by hoop layers on top and bottom of four angle-ply layers and also an eight layered laminate. T. Y. KAM et.al [6] conducted First-ply failure of laminated composite pressure vessels was studied via both theoretical and Experiments were carried out to verify the accuracy of the analytical

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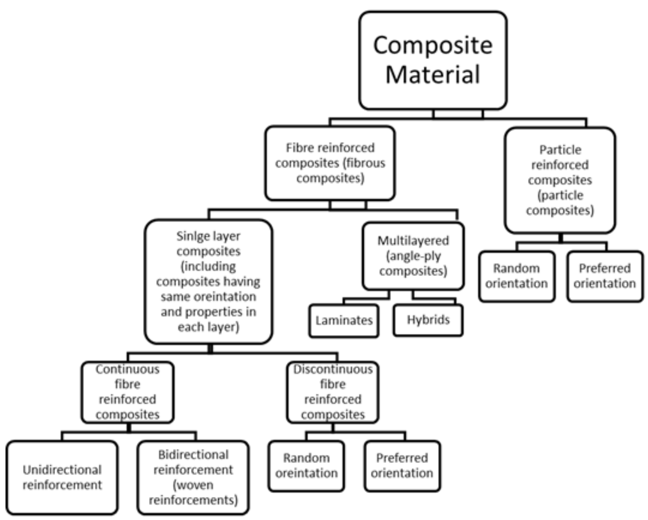
methods. PRASHANTH BANANA et.al [7] conducted experimental investigations used for the analysis of tensile and flexural behavior of carbon fiber reinforced polymer laminates leads to the following conclusions. ATHANSIOS J. KOLIOS et.al [8] has documented a methodology for the efficient reliability assessment of composite structures based on a combination of Finite Element Simulations, Stochastic Response Surface Method and First Order Reliability Methods for the estimation or reliability indices, based on Tsai Hill and Tsai Wu failure criteria. S. GOHARI et.al [9] analyzed the effect of volumetric fiber fraction factor (fV) on failure pressure was investigated for GFRP cylindrical shells subjected to internal and external pressure with various total thicknesses. REN QINGXIN [10] concluded that the enhanced level of buckling load by CFRP sheets descend with increasing columns height. CFRP sheets have higher strength.

The work is much concentrated on detailed study of lamina stresses on fiber reinforced composites cylindrical shell and their position upon applying pressure externally. Among the computational method, finite element method (FEM) is a widely used method due to its flexibility to model and analyze variety of engineering problems [4]. This is done by using classical laminate plate theory and verifying the details using FEA Analysis in ANSYS 16.0. Laminate configuration for various stacking sequences are analyzed and plotted.

Firstly, the study about carbon fiber reinforced composite is one of enormous complexity. In this paper we are prolonging the solutions for the deficiencies from certain reference papers and finding advanced and better solutions for buckling of carbon fiber reinforced cylindrical shell structure subjected to hydrostatic by various orientations for better stiffness. we are using Classical laminate theory for analysis and even a computational model and verify the both.

A. Classification of The Composite Material

The composite materials can be classified in different forms they are clearly in the figure1.1 below:



Classification of composite material

B. Basic assumptions of laminate

Each lamina or ply of the laminate is quasi – homogeneous and orthotropic, but the orientation the fiber may change from lamina to lamina.

All deformations in the laminate are considered to be small.
All displacements are continuous throughout the laminate.

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The laminate is thin and loaded in its plane only. the laminate and its layers are assumed to be in a plane stress conditions except the edges (σz = τxz=τyz).Transverse shear strains γxz and γyz are negligible. This implies that a line originally shear perpendicular to the laminate mid – plane remains straight and perpendicular to the each other.

The bond between plies in a laminate is perfect that is, plies will not slip over each other displacements and strains are continuous across interfaces of plies.

Strain- displacement and stress- strain relations are linear.

C.The Abd Matrix

The ABD matrix is a 6x6 matrix that serves as a connection between the applied loads and the associated strains in the laminate. It essentially defines the elastic properties of the entire laminate. To assemble the ABD matrix.

$$\begin{bmatrix} N_x \\ N_y \\ N_s \\ - \\ M_x \\ M_y \\ M_s \end{bmatrix} = \begin{bmatrix} A_{xx} & A_{xy} & A_{xs} & | & B_{xx} & B_{xy} & B_{xs} \\ A_{yx} & A_{yy} & A_{ys} & | & B_{yx} & B_{yy} & B_{ys} \\ A_{sx} & A_{sy} & A_{ss} & | & B_{sx} & B_{sy} & B_{ss} \\ - & - & - & + & - & - & - \\ B_{xx} & B_{xy} & B_{xs} & | & D_{xx} & D_{xy} & D_{xs} \\ B_{yx} & B_{yy} & B_{ys} & | & D_{yx} & D_{yy} & D_{ys} \\ B_{sx} & B_{sy} & B_{ss} & | & D_{sx} & D_{sy} & D_{ss} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_s \\ - \\ K_x \\ K_y \\ K_s \end{bmatrix}$$

D. Advantages

Light weight, High specific stiffness and strength, Easy moldable to complex forms, Easy bondable, Good dumping, Low electrical conductivity thermal expansion, Part consolidation due to lower overall system costs.

E. Disadvantages

Cost of materials is high, Difficulty manufacturing, Fasteners, Low ductility, Temperature limits, Repair at the original cure temperature requires tooling and pressure, Long-term fatigue characteristics unknown.

F. Applications

The composite used in mechanical industries in various ways is mainly in missiles., Thermal protection against kinetic energy, Thermal protection against combustion in rocket motors., Air frame structures., Aerospace, Military aircraft.

II. SELECTION OF MATERIAL

Selection of material is the most important criteria for a particular application based on several factors specified by performance indices or material indices. Selection of material plays a crucial role in any industry to implement the manufacturing before it undergone.

Performance Indices

Material selection is a step in the process of designing any physical object. In the context of product design, the main goal of material selection is to minimize cost while meeting product performance goals. Systematic selection of the best material for a given application begins with properties and costs of candidate materials.

Utilizing an "Ashby chart" is a common method for choosing the appropriate material. First, three different sets of variables are identified:

Material variable the inherent properties of a material such as density, modulus, yield stress, and many others.
Free variables are quantities that can change during the loading cycle, for example, applied force.
Design variables are limits imposed on the design, such as how thick the beam can be or how much it can deflect.
Performance index M1 for pressure vessels under external pressure is given by M1= ***E***0.5 / ρ (maximize)

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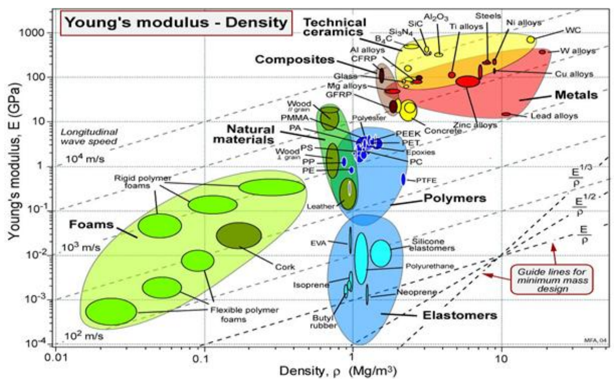
Table 1 design requirements

Function	Pressure hulls
Objective	Maximum safety, stiffness
Constraints	Low wall thickness to reduce mass and cost

Ashby chart plotting

The graph plotted between young's modulus and density is studied carefully. The material considered for our application should have maximum density of 1.5Mg/m3 and young's modulus greater than 100Gpa.

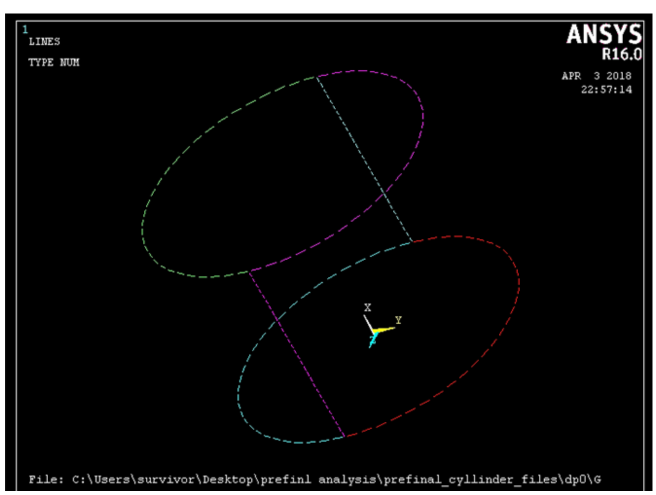


Ashby chart of young's modulus vs density

CFRE (Carbon fiber reinforced epoxy) is being selected from CFRs. AS4/3501-6 is Selected Based on Ref. Ashby Charts 2010.

III. DESIGN & ANALYSIS USING CLASSICAL LAMINATE THEORY

Stresses in the lamina coordinate system are determined analytically using classical laminate plate theory. FEA is conducted using ANSYS 14.5. The overall methodology adopted is summarized in the following flow chart. Consider a long cylinder shown in Fig having thickness ('t') 6mm, length ('l') of 800 mm and mid surface radius of ('R') of 583 mm. These dimensions are obtained from the ASME. The uniform 0.5Mpa pressure ('p') is applied in the cylinder. The shell considered is constructed of cross ply laminate with even number of layers of equal thickness abbreviated as 't'.



cylindrical shell with the prior dimensions modelled in ANSYS 16.0

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An orthotropic material has three mutually perpendicular planes of material symmetry. The present work deals with especially orthotropic materials. It is referred to as especially orthotropic material when the reference system coordinates are selected along the principal planes of material symmetry.

Dimensions of considered composite cylindrical shell

The dimensions are considered from procedure given in ASMEE section vii division [1] (Shells and Pressure Vessels code) for an external pressure application off 5 bar (0.5MPa) uniformly and equivalent load conditions.
The shell considered here in constructed of "symmetric laminates with especially orthotropic piles" or "symmetric cross-ply laminates" with even no. of layers (12) and of equal thickness (t). N=12, t=0.5mm
The material selected for analysis is "carbon fiber reinforced epoxy".
Industrial name of it is AS4/3501-6.
Fiber type =AS4carbon, fiber volume = 62%
Matrix=3501-6 epoxy, density 1.5gm/cm3
Material elastic properties (G Pa):

Table 2 properties of as carbon 3501-6 epoxy

E_1	E_2	E_3	υ_1	υ_2	υ_3	G_1	G_2	G_3
143	10	10	0.33	0.33	0.56	8.7	3	5

The shell is considered as a thin cylindrical pressure vessel, the global principal stress can be calculated as,

$$\text{Circumferential Stress } \sigma_{hoop} = 49.583 \text{ MPa} \quad (2)$$

$$\text{Longitudinal Stress } \sigma_{axial} = 24.291\text{MPa} \quad (3)$$

Where T= total thickness of 12 lamina R= Mean Radius

Let us consider an element from the cylindrical shell. The element is under bi-axial stress. The radial stress in the cylinder and the fiber direction being along the principal direction.

The shear stress in plane lamina is '0'. The element dimensions are 6mm thick, 10mm long and of unit width. Due to applied pressure the force per unit length will be exerted on the lamina in the longitudinal and circumferential direction.

The force per unit length on the mid plane can be calculated as,

$$N_y = 145.746 \text{ N/mm.} \quad N_x = 291.498 \text{ N/mm} \quad (4)$$

There are abrupt changes of slope for the variation of stresses over the depth. Stress varies from lamina to lamina in a discontinuous manner. Stress for kth lamina is derived as follows.
When a lamina is loaded in the reference axis XY, the relationship between stress and strain is as,

$$\begin{bmatrix}\sigma_x\\ \sigma_y\\ \sigma_s\end{bmatrix} = \begin{bmatrix}Q_{xx} & Q_{xy} & Q_{xs}\\ Q_{yx} & Q_{yy} & Q_{ys}\\ Q_{sx} & Q_{sy} & Q_{ss}\end{bmatrix}\begin{bmatrix}\varepsilon_x\\ \varepsilon_y\\ \varepsilon_s\end{bmatrix} \quad (5)$$

Where,

$$Q_{xx} = m^4Q_{11} + n^4Q_{22} + 2m^2n^2Q_{12} + 4m^2n^2Q_{66}$$

$$Q_{yy} = n^4Q_{11} + m^4Q_{22} + 2m^2n^2Q_{12} + 4m^2n^2Q_{66}$$

$$Q_{xy} = m^4Q_{11} + n^4Q_{22} + 2m^2n^2Q_{12} + 4m^2n^2Q_{66}$$

$$Q_{xs} = m^3nQ_{11} - mn^3Q_{22} + (mn^3 - m^3n)Q_{12} + 2(mn^3 - m^3n)Q_{66}$$

$$Q_{ys} = mn^3Q_{11} - m^3nQ_{22} + (m^3n - mn^3)Q_{12} + 2(m^3n - mn^3)Q_{66}$$

$$Q_{ss} = m^2n^2 + m^2n^2Q_{22} - 2m^2n^2Q_{12} + (m^2 - n^2)^2Q_{66}$$

Where, m = cosθ ; n = sinθ; θ = Angle of the ply with respect to longitudinal axis.

$$\begin{bmatrix}\sigma_x\\ \sigma_y\\ \sigma_s\end{bmatrix}_k = \begin{bmatrix}Q_{xx} & Q_{xy} & Q_{xs}\\ Q_{yx} & Q_{yy} & Q_{ys}\\ Q_{sx} & Q_{sy} & Q_{ss}\end{bmatrix}_k \begin{bmatrix}\varepsilon_x\\ \varepsilon_y\\ \varepsilon_s\end{bmatrix}_k \quad (6)$$

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$$[Q_i]_k = [Q_{ij}]_k + [E_j]_k$$

$[Q_{ij}]_k$= Transformed Reduced Stiffness Matrix.
So, to calculate this transformed reduced stiffness matrix we need to calculate the reduced stiffness matrix $[Q_{ij}]$.

$$[Q_{ij}] = \begin{bmatrix} 143905.07 & 3019.001 & 0 \\ 3019.001 & 10063.03 & 0 \\ 0 & 0 & 8700 \end{bmatrix} MPa \qquad (7)$$

Reduced Stiffness Matrix

The relation between entire laminate stiffness and forces and bending moment are given by;

$$\begin{bmatrix} N_x \\ N_y \\ N_s \\ - \\ M_x \\ M_y \\ M_s \end{bmatrix} = \begin{bmatrix} A_{xx} & A_{xy} & A_{xs} & | & B_{xx} & B_{xy} & B_{xs} \\ A_{yx} & A_{yy} & A_{ys} & | & B_{yx} & B_{yy} & B_{ys} \\ A_{sx} & A_{sy} & A_{ss} & | & B_{sx} & B_{sy} & B_{ss} \\ - & - & - & + & - & - & - \\ B_{xx} & B_{xy} & B_{xs} & | & D_{xx} & D_{xy} & D_{xs} \\ B_{yx} & B_{yy} & B_{ys} & | & D_{yx} & D_{yy} & D_{ys} \\ B_{sx} & B_{sy} & B_{ss} & | & D_{sx} & D_{sy} & D_{ss} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_s \\ - \\ K_x \\ K_y \\ K_s \end{bmatrix} \qquad (8)$$

The above matrix is called [A][B][D] Matrix.
Equation is further reduced to

$$\begin{bmatrix} N_x \\ N_y \\ 0 \end{bmatrix} = \begin{bmatrix} A_{xx} & A_{xy} & A_{xs} \\ A_{yx} & A_{yy} & A_{ys} \\ 0 & 0 & A_{ss} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_s \end{bmatrix} \qquad (9)$$

From equation (7) the strains are calculated and also for symmetric laminates of this particular arrangement, there is no necessity to calculate [B] matrix.
Further calculating $[A]_{ij}$ matrix we need to calculate the transformed reduced stiffness matrix.
Now the ‘reduced stiffness matrix” equation (10) is transformed.
Case1:
i) $\theta = 0^o$ for 1,3,5,8,10,12 laminates
ii) $\theta = 90^o$ for 2,4,6,7,9,11 laminates
Substituting these equations in equation (10) we get
The transformed reduced stiffness matrix
For 0^o laminates,

$$[\bar{Q}]_{xy}^{0^\circ} = \begin{bmatrix} 143905.07 & 3019.001 & 0 \\ 3019.001 & 10063.03 & 0 \\ 0 & 0 & 8700 \end{bmatrix} MPa \qquad (10)$$

For 90^o Laminates,

$$\begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_s \end{bmatrix} = \begin{bmatrix} 0.0002748403 \\ 0.0005824700 \\ 0 \end{bmatrix} \qquad (11)$$

The stress values calculated for 0^o oriented fibers are by substituting equation (3.23) in equation (3.5) we get

$$\begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_6 \end{bmatrix}_{0^\circ} = \begin{bmatrix} -41.30 \\ -6.69 \\ 0 \end{bmatrix} MPa \qquad (12)$$

These are the stresses in fibers oriented at 0^oin longitudinal and lateral directions in layers 1, 3,5,8,10,12.

$$\sigma_{longitudinal}^{0^\circ} = -41.30\ MPa$$
$$\sigma_{lateral}^{0^\circ} = -6.69\ MPa$$

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Now calculating stress in fibers oriented at 90 in layers 2,4,6,7,9,11.

$$\begin{bmatrix}\sigma_1\\ \sigma_2\\ \sigma_6\end{bmatrix}_{90^\circ} = \begin{bmatrix}-4.52\\ -84.65\\ 0\end{bmatrix} MPa$$

$$\sigma_{longitudinal}^{90^\circ} = -4.52\ MPa$$
$$\sigma_{lateral}^{90^\circ} = -84.65\ MPa\ .$$

Case2:
i) $\theta = 0^o$ for 1,3,5,8,10,12 laminates
ii) $\theta = 60^o$ for 2,4,6,7,9,11 laminates
Substituting these equations in equation (10) we get
Solving for mid plane strains we get

$$\begin{bmatrix}\varepsilon_x\\ \varepsilon_y\\ \varepsilon_s\end{bmatrix} = \begin{bmatrix}0.0003478\\ -0.008879810\\ -0.0040394\end{bmatrix} \qquad (13)$$

The stress values calculated for 0^o oriented fibers are by substituting equation (3.29) in equation (3.5) we get

$$\begin{bmatrix}\sigma_1\\ \sigma_2\\ \sigma_6\end{bmatrix}_{0^\circ} = \begin{bmatrix}-43.6950\\ 78.850\\ 35.13\end{bmatrix} MPa \qquad (14)$$

These are the stresses in fibers oriented at 0^oin longitudinal and lateral directions in layers 1, 3,5,8,10,12.

$$\sigma_{longitudinal}^{0^\circ} = -43.6950\ MPa$$
$$\sigma_{lateral}^{0^\circ} = 78.850 MPa$$

Now calculating stress in fibers oriented at 60in layers 2,4,6,7,9,11.

$$\sigma_{longitudinal}^{60^\circ} = 81.807\ MPa$$
$$\sigma_{lateral}^{60^\circ} = 8.86\ MPa\ .$$

IV. FINITE ELEMENT ANALYSIS OF COMPOSITE CYLINDRICAL SHELL USING ANSYS

Step 1. Type of Preference
The geometry modelling is done using ANSYS Design modular. The x direction in the element coordinate system represents the direction along the axis, the z represents the thickness or the radial direction while the y axis represents the tangential or the hoop stress direction as shown in fig4.1
Step 2. Defining the element
Type of element as shell 8node281 as shown in Fig 4.

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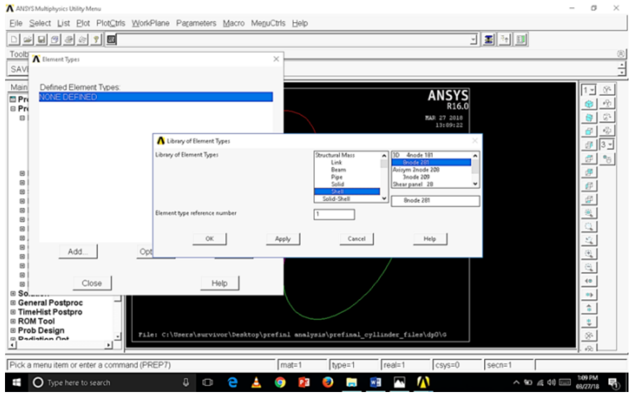
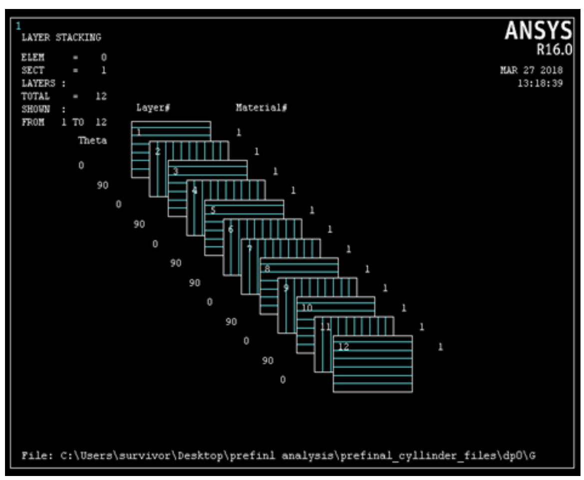
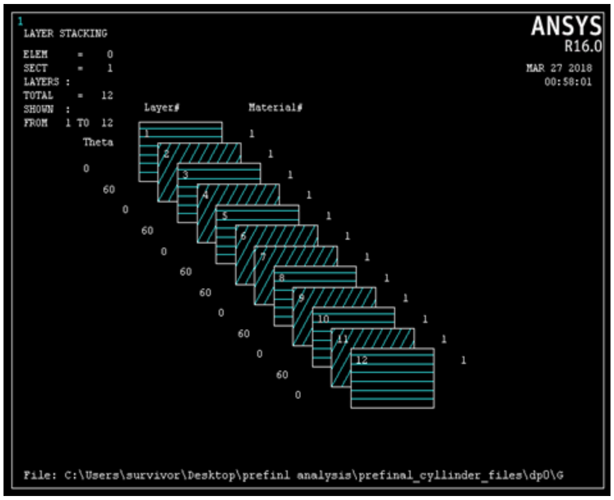


Defining type of element

Step 3. Defining material properties
Select the type of the material, Orthotropic Material. Input the values of Young’s modulus, Poisson’s Ratio, Rigidity modulus.
Step 4. Geometrical Specification:
Giving the geometrical specifications, i.e., the thickness of the lamina, the thickness of each layer.
Step 5. Stacking sequence
Stacking sequence of layers is shown in the figures below

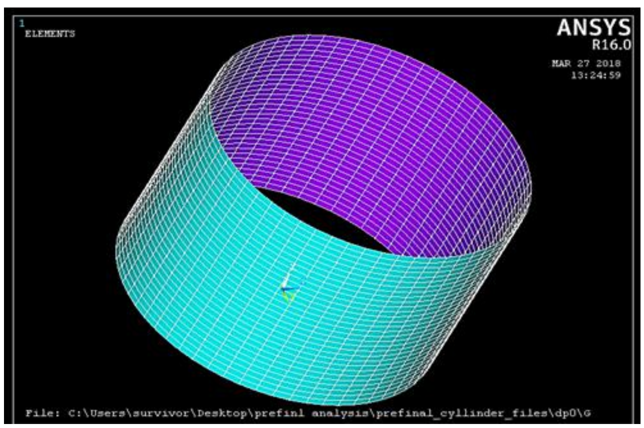


Stacking sequence of Layers of [0,90]



Stacking sequence of Layers of [0,60]

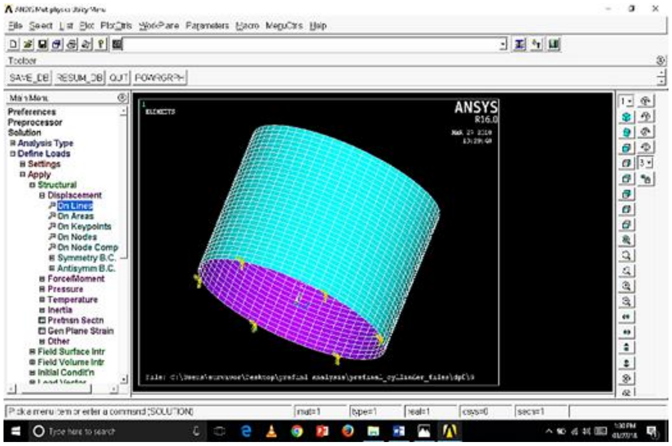
Step 6. Meshing
Meshing of the figure is done as shown in Fig. 7 and obtain the nodal solution by giving the required boundary conditions and Load values



Meshing of Model

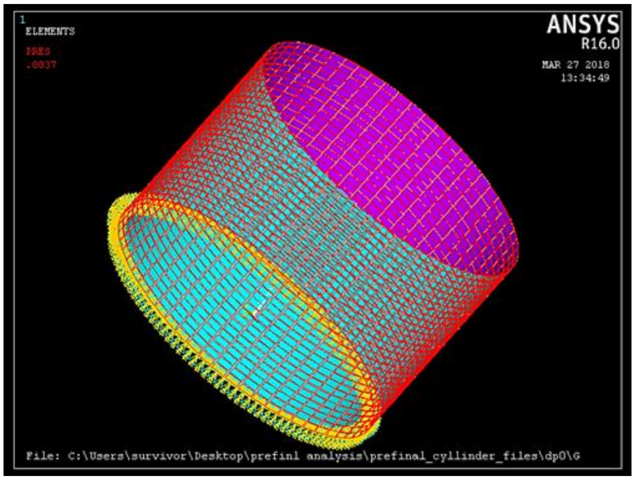
Step 7. Defining the type of analysis
Step 8. Boundary Conditions

One end of the shell is fixed as shown in Fig. 8. The pressure of 5bar is applied on the entire circumferential area.



Applying boundary conditions

Step 9. Solving Current load step
After applying the boundary conditions, the current load step is solved for the solution as shown in fig 4.16



solving current load step

V. RESULTS AND DISCUSSION

Layup pattern [0, 90] & [0, 60]: The result obtained by applying the external stress and desired boundary conditions is seen below.

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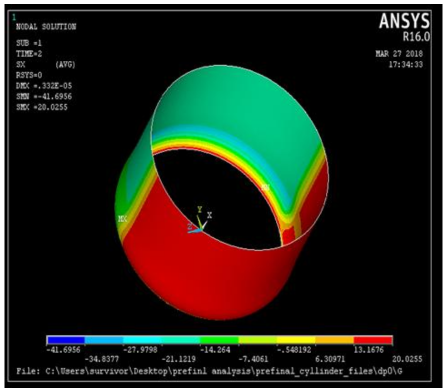
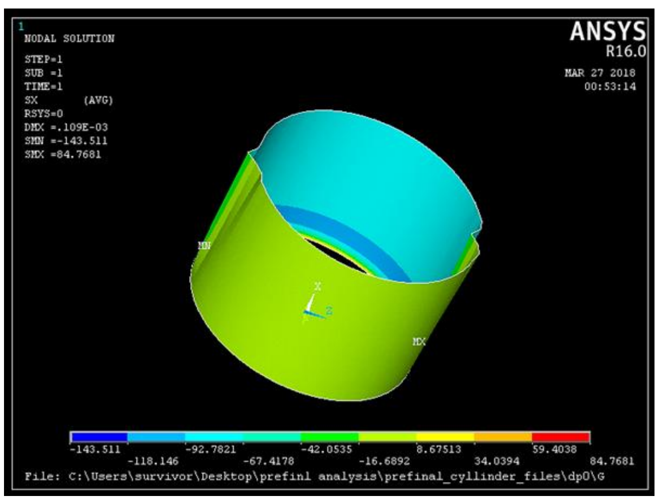


Buckling of layup [0, 90]

The above FEA analysis is verified using conventional shell element SHELL 8 NODE 281, or SHELL41, SHELL181. Modifying the nodal coordinate system to cylindrical coordinate system along with appropriate constraints and results were in good conformance with theoretically calculated results by Classical Laminate Theory. These values obtained by the ANSYS simulation are compared and validated with the theoretical calculated stress and strain values with the help of various theories.

The layers have ply orientation alternatively which are axi-symmetrical from the mid plane of shell.



Buckling of layup [0, 60]

The Stress along fiber direction for each layer in tabulated.

Table 3 results obtained

Lay patterns	Orientation	Layer numbers	Longitudinal stress (MPa)		Transverse stress (MPa)	
			CLT	**FEA**	**CLT**	**FEA**
layup [0, 90]	0	1,3,5,8,10,12	-41.30	-41.69	-6.69	-6.86
	90	2,4,6,7,9,11	-84.6	-84.79	-4.52	-4.54
layup [0, 60]	0	1,3,5,8,10,12	-43.69	-43.51	78.85	80.04
	60	2,4,6,7,9,11	81.80	84.76	8.86	9.64

The results obtained in above table in column have be compared to that of analytical results mention in so and so chapter hence the procedure we followed held good and hence this particular FE analysis procedure can be used for any other composite analysis.

From the above results [0,90] shows best results compared to all other cases studied. As it have less stresses induced and deformation of structure is less.

Hence lay pattern of [0, 90] is best suited and employable for external pressure of 5bar pressure.

VI. CONCLUSION

Stress analysis was carried out using classical laminate theory and validated using FEA methodology. These results were in very mush near to each other. Result obtained for failure criteria are very safe for desired operating conditions of 0.5 bars.

The majority of stress is taken along the fibre axis, resulting very less stress in the direction perpendicular the fibre. This advantage is obtained because of selecting the configuration having orientation of the fiber's in the direction of the principal stresses.

The values obtained using CLT theory in 0 degree -41.30 and -41.69 and 90 degree are -84.65 and -84.79.
This shows us the advantage of the fiber composites as they can be tailored according to the external conditions or loading acting on the structure.

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