Design and Analysis of Carbon Fiber Reinforced Composite Shell Structure Using Classical Laminate Plate Theory

P. V. Gopal Krishna\textsuperscript{1*}, Podila Meghana\textsuperscript{1}, A. Sai Kumar\textsuperscript{1} and K. Kishore\textsuperscript{1}

\textsuperscript{1}Department of Mechanical Engineering, Vasavi College of Engineering (Autonomous), Ibrahimbagh, Hyderabad, Telangana, India.

Authors’ contributions

This work was carried out in collaboration between all authors. Author PVGK designed the study, performed the statistical analysis, wrote the protocol and wrote the first draft of the manuscript. Authors PM and ASK managed the analyses of the study. Author KK managed the literature searches. All authors read and approved the final manuscript.

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ABSTRACT

The main focus of this project is to understand the nature of these laminated composites when subjected to specific damage cases like loads. In order to understand the progression of the failure modes in a laminated composite, models were designed and analyzed using ANSYS 14.5. In this work composite cylinder is designed which can withstand an external pressure of 5 bar. Stress analysis of the structure is done using classical laminate theory theoretically and the result obtained is validated using finite element analysis procedure using ANSYS and the error is very less. The behavior of the composite cylinder is checked by exploring the stresses and strains. The present work includes determination of optimum design parameters like fiber orientation. 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. The results thus obtained were found to be in good agreement in terms of damage size. To check the health of the composite cylindrical shell, failure criteria’s like Tsai-wu and Maximum stress criteria were also calculated in this paper.

*Corresponding author: E-mail: pvgopalkrishna@gmail.com;
NOMENCLATURES

- $E$ : Young's Modulus
- $N$ : Poisons Ratio
- $G$ : Rigidity Modulus
- $\sigma_{ij}$ : Stress Tensor
- $\varepsilon_{ij}$ : Strain Tensor
- $C_{ijkl}$ : Material Property Matrix
- $\Sigma_{\text{axial}}$ : Longitudinal Stress
- $\Sigma_{\text{hoop}}$ : Circumferential Stress
- $Q_{ij}$ : Reduced Stiffness Matrix
- $Q_{ijkl}$ : Transformed Reduced Stiffness Matrix
- $[D]_{ij}$ : Bending Stiffness Matrix
- $[B]_{ij}$ : Coupling Stiffness Matrix
- $[A]_{ij}$ : In Plane Stiffness Matrix
- $N_{x}, N_{y}$ : Normal Force
- $N_{xy}, N_{yy}$ : Force Per Unit Length

1. INTRODUCTION

The use of fiber reinforced laminated composite shells has increased in various fields like piping, remotely operated vehicles like long range missiles, space ships, risers, marine applications like submarines etc. Fiber reinforced composites have been successfully used in underwater vehicles and ocean structures replacing conventional metal counterparts due to their light weight, corrosion resistance, high strength to weight ratio. The use of civil aircraft structures has now expanded to underwater applications. This is because of the advancement in composite material manufacturing; that the small sized under water vehicles can now be manufactured in one piece.

Composite materials (or composites for short) can be defined as engineered materials made from two or more constituent materials which contain significantly different physical or chemical properties and remain separate and distinct on a macroscopic level within the finished structure. There are two types of composite, the first one is short fiber reinforced polymer and the other one is continuous fiber reinforced polymer. Fiber reinforced composites unlike metals are orthotropic in nature. The elastic properties are different in different directions. The strength of the composite structure is dependent on the fiber directions, stacking sequence, thickness of each ply along with the material properties of the material used.

Several researches have investigated the inelastic behavior of unidirectional fibrous composites. Two different approaches have been used: a macro mechanical and micro mechanical approach. In the macro-mechanical approach the composite is regarded as “mixture” of two different materials and approximated as a macroscopically homogeneous medium. Although this approach is usually simpler, it often heavily relies on experimental data. Examples of such approaches are given in [1]. On the other hand the micro mechanical approach regards the composite as a non-homogeneous structure. This approach allows modelling more complicated composite behavior and reducing testing needs by taking internal interactions into account.

In recent literatures [2,3] and [4] a new hyperbolic shear deformation theory was used to find the equilibrium and stability equations of rectangular functionally graded plates. [5] has analyzed thermal buckling behavior of FGM square plates with simply supported edges has been studied in this note using the classic plate theory (CPT). Stability analysis of functionally graded ceramic–metal plates under thermal loads was presented in [6] using the first order shear deformation theory. [7,8] investigated the buckling of FGM plate under thermal loads.

The effect of fiber orientation of laminates has been investigated & experimentation was performed to determine property data for material specifications [9]. A study on a Circular cylindrical thin-walled shell failure made of GRP composite subjected to static internal and external pressure was carried out by [10]. Where as in [11] material characterization of FRP of carbon T300/Epoxy for various configurations as per ASTM standards is experimentally determined using filament winding and matched die mould technique. [12] described the processing and mechanical characterization of two different fibers (glass and carbon) and two different fabric architectures (woven roving and stitch bonded) that were made into composites using Dow Chemical’s Derakane 510A-40, a brominated vinyl ester resin. [13] addressed the linear analysis of axially symmetric structures of thin shell subjected to axisymmetric loads. In [14] filament wound glass fiber reinforced plastic (GRP) tubes were studied. Impact tests were realized at 5 J and 10 J energy levels. [15] presented a methodology for reliability assessment of composite members based on
appropriate limit state functions derived according to fundamental failure criteria, Tsai-Hill and Tsai-Wu, applicable to composite materials. [16] offered an overview of several applications in aeronautics, but it focused on composites application as structural materials and basic knowledge on composite materials and their theory of analysis was followed from standard text books mentioned [17-22]. An exact analytical solution for mechanical buckling analysis of symmetrically cross-ply laminated plates including curvature effects is presented by [23].

The present 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 [24]. This is done by using classical laminate plate theory and verifying the details using FEA Analysis in ANSYS 14.5. Laminate configuration for various stacking sequences are analyzed and plotted.

2. METHODOLOGY

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.

2.1 Geometry, Material and Loading

Consider a long cylinder shown in Fig. 1 having thickness ('t') 6mm, length ('l') of 830 mm and mid surface radius of ('R') of 583 mm. These dimensions are obtained from the ASME. The uniform 5bar 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' [25].

The material selected for the analysis is carbon fiber reinforced epoxy. The cross-ply laminate construction consists of 12 layers with each layer having thickness of 0.5 mm the fiber direction is defined with respect to the longitudinal axis of the cylinder. The elastic and strength properties are defined in the Tables 1 and 2 respectively having fiber volume fraction of 60% and density 1.5 gm/cm3.

![Fig. 1. Methodology adopted for failure analysis](image-url)
Table 1. Material properties of AS4/3501-6

| Material properties in (GPa) [26-29] |
|-------------------------------------|
| $E_1$ | $E_2$ | $E_3$ | $\nu_1$ | $\nu_2$ | $\nu_3$ | $G_{12}$ | $G_{13}$ | $G_{23}$ |
| 143   | 10    | 10    | 0.33    | 0.33    | 0.56    | 8.7      | 3        | 5        |

Table 2. Strength properties of AS4/3501-6

| Strength properties in (MPa) |
|-----------------------------|
| $\sigma_{xt}$ | $\sigma_{xc}$ | $\sigma_{yt}$ | $\sigma_{yc}$ | $\sigma_{zt}$ | $\tau_{xy}$ | $\tau_{yz}$ | $\tau_{xz}$ |
| 143 | 10 | 10 | 0.33 | 0.33 | 0.56 | 8.7 | 3 | 5 |

Fig. 2. Cylinder dimensions

The dimensions are considered from procedure given in ASMEE section vii division [30] (Shells and Pressure Vessels code) for external pressure applications of 5 bar (0.5MPa) uniformly and equivalent load conditions.

The shell considered here in constructed of “symmetric laminates with special orthotropic piles” with even no. of layers (i.e. N=12) and of equal thickness (t) of each 0.5mm. The material selected for analysis is “carbon fiber reinforced epoxy” and the Industrial name of it is AS4/3501-6.

Fiber type = AS4carbon, fiber volume = 62%
Matrix = 3501-6 epoxy, density = 1.5 gm/cm$^3$

2.2 Analysis Using Classical Laminate Theory

This is the simplest laminate plate theory, which is based on laminate plate theories. The principal assumptions made in this theory are the plane sections normal to the mid plane before deformation remain straight and normal to the mid-plane after deformation. The other assumptions made in this theory are 1) the in-plane strains are small when compared to unity, 2) the plates are perfectly bonded, 3) the displacement are small compared to the thickness, although tis assumptions lead to simple constitutive equations. CLT is still a common approach used to get quick and simple
predictions especially for the behavior of thin plated laminated structures. The main simplification in this model is that 3D structural plates (with thickness) or shells are treated as 2D plate or shells located through mid-thickness which results in a significant decrement of the total number of equations and variable, consequently saving a lot of computational time and effort. Since they are present in closed-form solutions they provide better practical interpretation and their governing equations are easier to solve [31].

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

**Circumferential Stress,**

\[\sigma_{ hoop} = \frac{p \times R}{t} = \frac{0.5 \times 583}{6} = 49.583 \text{ MPa} \tag{1}\]

**Longitudinal Stress,**

\[\sigma_{ axial} = \frac{p \times R}{2t} = \frac{0.5 \times 583}{2 \times 6} = 24.291 \text{ MPa} \tag{2}\]

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,

\[N_y = \frac{\sigma_{ axial} x A_1}{\text{thickness of face perpendicular to } x \text{ axis of the local coordinate system}} = 145.746 \text{ N/mm.} \tag{3}\]

\[N_x = \frac{\sigma_{ hoop} x A_1}{\text{thickness of face perpendicular to } y \text{ axis of the local coordinate system}} = 291.498 \text{ N/mm} \tag{4}\]

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,

\[\begin{bmatrix} \sigma_x \\ \sigma_y \\ \sigma_z \end{bmatrix} = \begin{bmatrix} Q_{xx} & Q_{xy} & Q_{xz} \\ Q_{yx} & Q_{yy} & Q_{yz} \\ Q_{zx} & Q_{zy} & Q_{zz} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_z \end{bmatrix} \tag{5}\]

Where,

\[Q_{xx} = m^4 Q_{11} + n^4 Q_{22} + 2 mn^2 Q_{12} + 4 m^2 n^2 Q_{66}\]

\[Q_{yy} = n^4 Q_{11} + m^4 Q_{22} + 2 mn^2 Q_{12} + 4 m^2 n^2 Q_{66}\]

\[Q_{xy} = m^4 Q_{11} + n^4 Q_{22} + 2 mn^2 Q_{12} + 4 m^2 n^2 Q_{66}\]

\[Q_{zz} = m^2 n Q_{11} - mn^2 Q_{22} + (m^3 - n^3) Q_{12} + 2 (m n - mn^2) Q_{66}\]

\[Q_{zy} = mn Q_{11} - 2 mn^2 Q_{22} - 2 m^2 n Q_{12} + (m^3 + n^3) Q_{66}\]

\[Q_{xz} = m^2 n^2 - 2 mn^2 Q_{22} - 2 m^2 n Q_{12} + (m^3 - n^3) Q_{66}\]

\[m = \cos \theta \; n = \sin \theta ; \; \theta = \text{Angle of the ply with respect to longitudinal axis.} \tag{6}\]

So, to calculate this transformed reduced stiffness matrix we need to calculate the reduced stiffness matrix \([Q_i]_k\).

\[\begin{bmatrix} Q_{ij} \end{bmatrix} = \begin{bmatrix} 143905.07 & 3019.001 & 0 \\ 3019.001 & 1063.03 & 0 \\ 0 & 0 & 8790 \end{bmatrix} \text{ MPa} \tag{7}\]

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

\[\begin{bmatrix} N_x \\ N_y \\ N_z \\ M_x \\ M_y \\ M_z \end{bmatrix} = \begin{bmatrix} A_{xx} & A_{xy} & A_{xz} & B_{xx} & B_{xy} & B_{xz} \\ A_{yx} & A_{yy} & A_{yz} & B_{yx} & B_{yy} & B_{yz} \\ A_{zx} & A_{zy} & A_{zz} & B_{zx} & B_{zy} & B_{zz} \\ M_{xx} & M_{xy} & M_{xz} & D_{xx} & D_{xy} & D_{xz} \\ M_{yx} & M_{yy} & M_{yz} & D_{yx} & D_{yy} & D_{yz} \\ M_{zx} & M_{zy} & M_{zz} & D_{zx} & D_{zy} & D_{zz} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_z \end{bmatrix} \tag{8}\]

The above matrix is called \([A][B][D]\) Matrix.

\([A]\) is in-plane stiffness matrix and is related with transformed reduced stiffness matrix as

\[\begin{bmatrix} A_{ij} \end{bmatrix} = \sum_{k=1}^{n} [Q_i]_k (Z_k - Z_{k-1}) \tag{9}\]

\([B]\) is coupling stiffness matrix

\[\begin{bmatrix} B_{ij} \end{bmatrix} = \sum_{k=1}^{n} [Q_i]_k (Z_k^2 - Z_{k-1}^2) \tag{10}\]

\([D]\) is bending stiffness matrix

\[\begin{bmatrix} D_{ij} \end{bmatrix} = \sum_{k=1}^{n} [Q_i]_k (Z_k^3 - Z_{k-1}^3) \tag{11}\]
All the laminates are all arranged in [0/90/0/90/0/90] or [0/90/0/90/0/90/0/90/0/90/0/90/0/90/0/90/0/90] and the kth layer is oriented in the principal x-axis direction. Then, we have the following relations.

For two identical piles we consider two terms of [B] having opposite signs, so they are cancelled.

\[ [B]_{ij} = 0 \]  \hspace{1cm} (12)

Due to bi-axial forces \( N_x \) and \( N_y \) only exists as hoop and longitudinal direction and remaining forces are zero.

\[ N_S = 0 \]  \hspace{1cm} (13)

There is no elongation and bending coupling, bending moment are zero.

\[ m_x = m_y = m_z = 0 \]  \hspace{1cm} (14)

From equation (6) and equation (9) it is evident that

\[ A_{xx} = A_{yy} = 0 \]
\[ D_{xx} = D_{yy} = 0 \]  \hspace{1cm} (15)

Now Substituting Equation (12), (13), (14), (15) in ABD Matrix, it reduces to

\[
\begin{bmatrix}
N_x \\
N_y \\
0
\end{bmatrix} = \begin{bmatrix}
A_{xx} & A_{xy} & A_{xs} & 0 & 0 & 0 & \varepsilon_x \\
A_{xy} & A_{yy} & A_{ys} & 0 & 0 & 0 & \varepsilon_y \\
0 & 0 & A_{xs} & 0 & 0 & 0 & \varepsilon_x \\
0 & 0 & 0 & 0 & 0 & 0 & \varepsilon_y \\
0 & 0 & 0 & 0 & 0 & 0 & \varepsilon_x \\
0 & 0 & 0 & 0 & 0 & 0 & \varepsilon_y \\
0 & 0 & 0 & 0 & 0 & 0 & \varepsilon_x \\
0 & 0 & 0 & 0 & 0 & 0 & \varepsilon_y \\
\end{bmatrix}
\]

Equation is further reduced to

\[
\begin{bmatrix}
N_x \\
N_y \\
0
\end{bmatrix} = \begin{bmatrix}
A_{xx} & A_{xy} & A_{xs} & \varepsilon_x \\
A_{xy} & A_{yy} & A_{ys} & \varepsilon_y \\
0 & 0 & A_{xs} & \varepsilon_x \\
0 & 0 & 0 & \varepsilon_y \\
0 & 0 & 0 & \varepsilon_x \\
0 & 0 & 0 & \varepsilon_y \\
0 & 0 & 0 & \varepsilon_x \\
0 & 0 & 0 & \varepsilon_y \\
\end{bmatrix}
\]

Substituting these equations in equation (5) we get the transformed reduced stiffness matrix as

For 0° laminates,

\[
[Q]_{xy}^0 = \begin{bmatrix}
143905.07 & 3019.001 & 0 \\
3019.001 & 10063.03 & 0 \\
0 & 0 & 8700 \\
\end{bmatrix} \text{MPa} \tag{17}
\]

For 90° laminates,

\[
[Q]_{xy}^{90} = \begin{bmatrix}
10063.03 & 3019.001 & 0 \\
3019.001 & 143905.07 & 0 \\
0 & 0 & 8700 \\
\end{bmatrix} \text{MPa} \tag{18}
\]

Now \([A]_{ij}\) Matrix for entire laminate is

\[
[A]_{xy} = \begin{bmatrix}
\sum_{k=1}^{6} [Q]_{xy}^k \times (Z_k - Z_{k-1})^0 + \sum_{k=1}^{6} [Q]_{xy}^{90} \times (Z_k - Z_{k-1})^{90} \\
\sum_{k=1}^{6} [Q]_{xy}^k \times (Z_k - Z_{k-1})^0 + \sum_{k=1}^{6} [Q]_{xy}^{90} \times (Z_k - Z_{k-1})^{90} \\
\sum_{k=1}^{6} [Q]_{xy}^k \times (Z_k - Z_{k-1})^0 + \sum_{k=1}^{6} [Q]_{xy}^{90} \times (Z_k - Z_{k-1})^{90} \\
\end{bmatrix} \tag{19}
\]

Substituting equation (18),(19) in (20), we get

\[
[A]_{xy} = \begin{bmatrix}
461904.30 & 18114.006 & 0 \\
18114.006 & 461904.30 & 0 \\
0 & 0 & 52200 \\
\end{bmatrix} \text{MPa m/m} \tag{20}
\]

Substituting equations (17), (18), (19), (20) in (16), we get

\[
\begin{bmatrix}
\varepsilon_x \\
\varepsilon_y \\
\varepsilon_z
\end{bmatrix} = \begin{bmatrix}
145.746 \\
291.498 \\
0
\end{bmatrix} \text{MPa}\tag{21}
\]

Solving (22) for mid plane strains we get

\[
\begin{bmatrix}
\varepsilon_x \\
\varepsilon_y \\
\varepsilon_z
\end{bmatrix} = \begin{bmatrix}
0.0002748403 \\
0.0005824700 \\
0
\end{bmatrix}\tag{22}
\]

The stress values calculated for 0° oriented fibers are by substituting equation (22) in equation (5) we get

\[
\begin{bmatrix}
\sigma_x \\
\sigma_y \\
\sigma_z
\end{bmatrix} = \begin{bmatrix}
143905.07 & 3019.001 & 0 \\
3019.001 & 10063.03 & 0 \\
0 & 0 & 8700 \\
\end{bmatrix} \begin{bmatrix}
\varepsilon_x \\
\varepsilon_y \\
\varepsilon_z
\end{bmatrix} - \begin{bmatrix}
-41.30 \\
-6.69 \\
0
\end{bmatrix} \text{MPa}\tag{23}
\]

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

\[
\sigma_{\text{longitudinal}} = -41.30 \text{ MPa} \\
\sigma_{\text{lateral}} = -6.69 \text{ MPa} \tag{24}
\]

Now calculating stress in fibers oriented at 90° in layers 2, 4, 6, 7, 9, 11.
Substituting (22), in (5) using (18) we get,

\[
\begin{bmatrix}
10063.03 \\
3019.001 \\
0 \\
\end{bmatrix}
= 
\begin{bmatrix}
3019.001 \\
143905.07 \\
0 \\
\end{bmatrix}
\begin{bmatrix}
\sigma_{1} \\
\sigma_{2} \\
\sigma_{3} \\
\end{bmatrix}
-
\begin{bmatrix}
-0.0002748403 \\
-6.0005824700 \\
8700 \\
\end{bmatrix}
\]

\[
\begin{bmatrix}
\sigma_{1} \\
\sigma_{2} \\
\sigma_{3} \\
\end{bmatrix}
= 
\begin{bmatrix}
-4.52 \\
-84.65 \\
0 \\
\end{bmatrix}
MPa
\]

These are the stresses in fibers oriented at 90° in longitudinal and lateral directions in layers 2,4,6,7,9,11.

\[
\sigma_{\text{longitudinal}}^{90°} = -4.52 \text{ MPa}
\]

\[
\sigma_{\text{lateral}}^{90°} = -84.65 \text{ MPa}
\]

2.3 Finite Element Analysis

Finite Element Analysis is well accepted for many industrial applications. Complex geometry or highly computationally demanding problems or intermediate structures are solved by FEA. The FEA model is developed considering cylinder as an elastic material model (i.e. orthotropic) loading and constraints. In the present work the material is not a hyper elastic (highly stretchable) material model, hook-elasticity can applied. For hyper elastic materials neo-hookean material model should be applied. One end of the element is fixed in the axial direction of the cylinder while the other end is free. One dimensional axi-symmetric shell 8NODE 281 is used in this element model. This element is suitable for modelling thin to moderately thick axi-symmetric shell structures.

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.

Section command is used to define each layer for giving fiber directions, stacking sequence and integration points. The fibers are oriented with respect to X axis of the element coordinate system and the stacking of the thickness takes place in the positive z direction. Stacking sequence is shown in the Fig. 4.

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 and applying the force of 5000 N in y-direction along with appropriate constraints.

Results were in good conformance with theoretically calculated results by Classical Laminate Theory. These values obtained by the ANSYS 14.5 simulation are compared and validated with the theoretical calculated stress and strain values with the help of various theories.

To calculate the inter laminar shear stresses a sample composite rectangular slab was taken of dimensions 250×45 mm and thickness is same as composite cylinder and analyzed for Max stress criteria and Tsai Wu failure criteria. The layer stacking sequence and no. of layers and the material are same as of the cylinder. Same procedure was followed and same shell element was considered and except for the type of analysis done on the rectangular composite slab and applying the force of 5000 N in y-direction.

These results of failure analysis obtained by the ANSYS simulation are validated based on Max stress and Tsai Wu failure theories and results were in good conformance.
Fig. 4. Orientation of the laminate

Fig. 5. Externally loaded cylinder and restrained at bottom

Fig. 6. Externally loaded Rectangular composite slab in y-direction
3. RESULTS AND DISCUSSION

The result obtained by applying the external stress and desired boundary conditions is seen in Fig. 7. The layer no.1 has zero-degree orientation of the fiber while the layer no. 2 has the 90-degree orientation. The Stress along fiber direction for each layer in tabulated as follows.

The results obtained in Table 3 in column have be compared to that of analytical results mentioned in equation (24).

The deformation occurred in the rectangular composite slab due to the external loading of 5000 N are shown (Fig. 8).

Table 3. Result for Stresses for composite cylindrical shell

| Orientation | Layer numbers | Longitudinal stress (MPa) | Transverse stress (MPa) |
|-------------|---------------|---------------------------|------------------------|
|             |               | CLT | FEA | CLT | FEA |
| 0           | 1,3,5,8,10,12 | -41.30 | -41.69 | -6.69 | -6.65 |
| 90          | 2,4,6,7,9,11  | -84.65 | -84.79 | -4.52 | -4.54 |

Fig. 7. Deformed composite cylinder shell under external loading

Deformation in X and Y direction
The results for stresses and deformations in the rectangular composite slab analyzed for Max stress criteria and Tsai Wu failure criteria are tabulated as follows.
Table 4. Table for stresses and failure criteria

| Sl. no | DOF | STRESS       | Tsai Wu | Maximum stress |
|-------|-----|--------------|---------|----------------|
|       |     | X  | Y  | Z  | X  | Y  | Z  |             |               |
| DMX   |     | 0.0104 | 0.0104 | 0.0104 | 0.0104 | 0.0104 | 0.0104 | 0.0104 |               |
| SMN   |     | -0.108 | -0.104 | -0.444 | -597.4 | -244.07 | 0         | 0         | 0             |
| SMX   |     | 0.8414 | 0.8678 | 0.0631 | 597.06 | 244.07 | 0         | 0         | 0             |

By observing the results obtained in above table in the last column i.e. Maximum stress theory have be compared to that of Tsai Wu theory. Hence this procedure we followed held good and hence this particular FE analysis procedure can be used for any other composite analysis.

4. CONCLUSION AND FUTURE SCOPE

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

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.

Stress analysis was carried out using CLT and validated using finite element analysis. The accuracy and efficiency of the CLT method is examined by comparing the results that were obtained from Finite Element Analysis. The results were found to be in very good agreement which can be seen in Table 3.

Maximum stress and Tsai Wu failure criteria are very much near to each other and safe for desired operating conditions of 0.5 bar. Hence the procedure we followed held good and hence this particular FE analysis procedure can be used for any other composite analysis.

This work can further be extended for the research work to conduct the real time experimental investigation upon the same laminates configuration to find the Failure analysis, and also we can conduct the experiment with inward stress conditions, to find out the inter-laminar shear stresses, for the missile applications. It can also be extended for practical validation by fabricating the prototype and its hydrostatic testing.

COMPETING INTERESTS

Authors have declared that no competing interests exist.

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