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ANSYS as a Tool to Perform FEA and Failure Analysis of Hybrid Laminated Composites

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Abstract – *With the increase in the use of composite materials in aerospace, automotive, marine and construction industries it is very important to perform the analysis of composite laminates using available CAE tools. This paper deals with stress and failure analysis of hybrid composite laminates using ANSYS APDL 15.0. Two cases are considered one symmetric and another anti-symmetric laminate with different stacking sequence and loading conditions. The stresses obtained by ANSYS are validated analytically using Classical Lamination Theory (CLT). The failure analysis is performed using failure theories to evaluate First Ply Failure (FPF) load and Last Ply Failure (LPF) load of the laminate which also specifies the mode of failure. The stresses obtained by ANSYS are compared with CLT.*

***Key Words*: Stress Analysis, Failure Analysis, ANSYS APDL, CLT, FPF Load, LPF Load**

1. INTRODUCTION

Composite material is the combination of two or more materials at macroscopic level which are insoluble into each other. It consists of matrix and reinforcements in form of fibres. The fibers can be oriented at any angle to suit the application. Both fibers and matrix combine to form a single lamina which is a building block of laminate as shown in fig - 1 below. These laminas (plies) can be of same material or different, oriented at different angles. Hybrid composite laminate is one which consists of plies made of different materials oriented at different angles. A lamina is orthotropic material having different material properties in different directions.

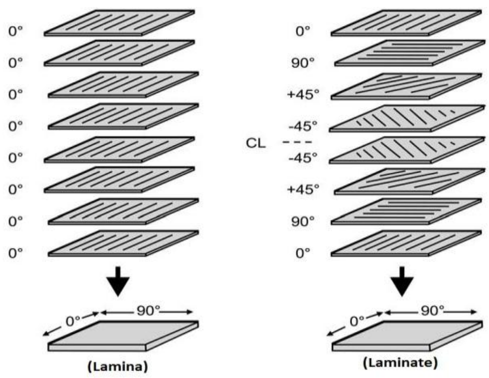


Fig - 1 Lamina and laminate

The stress–strain and failure analysis of composite laminate is very important be carried before it is converted into any structural element to know the behavior of laminate. The stresses are useful to perform failure analysis which gives the failure loads of the laminate. J. Zhang et al. [1] performed investigation on the effect of stacking sequence on the strength of hybrid composite laminate by varying the fiber volume fraction of carbon and glass fibers in epoxy matrix and concluded that specimen with 50% of carbon fibers at the exterior of the laminate showed good flexural properties and alternate layers of carbon epoxy and glass epoxy gave high compressive strength. C. Elanchezhian et al. [2] performed flexural, tensile and impact tests on carbon epoxy (CFRP) and glass epoxy (GFRP) laminates separately and observed that tensile and flexural strength of CFRP was more than GFRP also during impact test the maximum energy was absorbed by CFRP as compared to GFRP laminates. GuruRaja M. N et al. [3] investigated the tensile properties of hybrid carbon–glass epoxy laminates with three different orientations 0°/90°, 45°/45°, 30°/60° and concluded that the laminate with 0°/90° orientation showed good tensile properties as compared to other orientations. N. Nayak et al. [4] performed analytical and experimental study on vibration and bucking analysis of Glass-Carbon/epoxy woven fiber hybrid composite laminates and concluded that maximum buckling load resulted in the laminate with material of higher elastic modulus at the exterior and thereby increasing effectiveness of hybridization. Irina M.M.W. et al. [5] investigated mechanical properties by performing flexural and tensile tests on of hybrid composites made from plain woven glass fiber, stitched bi-axial $\pm 45^0$ glass fiber and plain-woven carbon fiber. The experimental results showed that mechanical properties for $[\mathrm{CWW}]_6$ arrangement was superior compared to other arrangements, where C is carbon fiber (weaved) and W glass fiber (weaved) M. Nayeem Ahmed et al. [6] performed study to calculate the flexural strength by using different types of matrix material on different thickness of lamina to optimize the thickness. They concluded that matrix material LY 5052 showed high bending strength as compared to LY 556 as matrix material, irrespective of their thickness. This study also concluded that bending strength of composites was enhanced with its thickness and therefore thickness should be more for the static or dynamic loading application.

2. PROBLEM STATEMENT

This paper deals with the study of two cases of hybrid composite laminates one angle ply and other symmetric laminate with different loading conditions to determine stresses in the laminate and perform failure analysis to evaluate FPF load and LPF load of the laminate.

Case 1: A $[0_{CE}/45_{GE}/\text{-}45_{GE}/90_{CE}]$ angle ply hybrid composite laminate made of plies of carbon/epoxy (CE) and glass/epoxy (GE) is subjected to a biaxial tensile load of N_x=N_y=1000 N/m. Each lamina is 2mm thick and cross section of laminate is 100mm×100mm.

Case 2: A $[0_{CE}/90_{GE}/0_{CE}]$ symmetric cross ply hybrid laminate made of plies of carbon/epoxy (CE) and glass/epoxy (GE) is subjected to a uniaxial tensile load per unit length of N_x=500 N/mm and temperature change of -75 °C (Curing temp. = 120°C and Room temp. = 45°C). The thickness of CE lamina is 1mm and GE is 2 mm. The cross section of laminate is 100mm×100mm.

Analytical modeling is performed using Classical Lamination Theory (CLT) to validate the stresses obtained by ANSYS. Maximum Stress Failure theory is used to evaluate the mode of failure in the lamina.

3. ANALYTICAL MODELING OF LAMINATE

The stress strain relation is given by Generalized Hooke's Law

$$\sigma_{ij} = Q_{ijkl}\,\epsilon_{kl} \tag{3.1}$$

Where, σ_{ij} is Stress matrix, Q_{ijkl} Stiffness matrix and ϵ_{kl} is Strain matrix.

For 2D unidirectional lamina

$$\begin{Bmatrix}\sigma_1\\ \sigma_2\\ \tau_{12}\end{Bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & 0\\ Q_{12} & Q_{22} & 0\\ 0 & 0 & Q_{66}\end{bmatrix}\begin{Bmatrix}\epsilon_1\\ \epsilon_2\\ \gamma_{12}\end{Bmatrix} \tag{3.2}$$

For angle lamina

$$\begin{Bmatrix}\sigma_x\\ \sigma_y\\ \tau_{xy}\end{Bmatrix} = \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16}\\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26}\\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66}\end{bmatrix}\begin{Bmatrix}\epsilon_x\\ \epsilon_y\\ \gamma_{xy}\end{Bmatrix} \tag{3.3}$$

3.1 Classical Lamination Theory (CLT)

The relation between the strains and displacement is given by CLT and variation can

$$\begin{Bmatrix}\epsilon_x\\ \epsilon_y\\ \gamma_{xy}\end{Bmatrix} = \begin{Bmatrix}\epsilon_x^0\\ \epsilon_y^0\\ \gamma_{xy}^0\end{Bmatrix} + z\begin{Bmatrix}K_x\\ K_y\\ K_{xy}\end{Bmatrix} \tag{3.4}$$

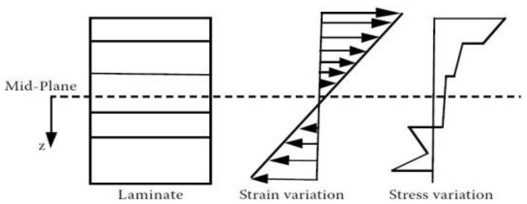


Fig - 2 Stress and strain variation through thickness of laminate

3.2 Resultant Laminate Forces and Moments

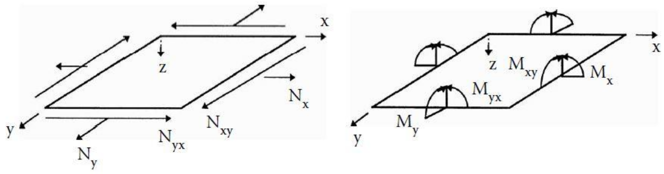


Fig - 3 Resultant forces and moments

The relation between resultant forces and moments with mid plane strains and curvatures

$$\begin{Bmatrix}N_x\\ N_y\\ N_{xy}\\ M_x\\ M_y\\ M_{xy}\end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} & B_{11} & B_{12} & B_{16}\\ A_{12} & A_{22} & A_{26} & B_{12} & B_{22} & B_{26}\\ A_{16} & A_{26} & A_{66} & B_{16} & B_{26} & B_{66}\\ B_{11} & B_{12} & B_{16} & D_{11} & D_{12} & D_{16}\\ B_{12} & B_{22} & B_{26} & D_{12} & D_{22} & D_{26}\\ B_{16} & B_{26} & B_{66} & D_{16} & D_{26} & D_{66}\end{bmatrix}\begin{Bmatrix}\epsilon_x^0\\ \epsilon_y^0\\ \gamma_{xy}^0\\ K_x\\ K_y\\ K_{xy}\end{Bmatrix} \tag{3.5}$$

In which,

$$A_{ij} = \sum_{k=1}^{n}\left[\bar{Q}_{ij}\right]_k (Z_k - Z_{k-1}) \tag{3.6}$$

$$B_{ij} = \frac{1}{2}\sum_{k=1}^{n}\left[\bar{Q}_{ij}\right]_k (Z_k^2 - Z_{k-1}^2) \tag{3.7}$$

$$D_{ij} = \frac{1}{3}\sum_{k=1}^{n}\left[\bar{Q}_{ij}\right]_k (Z_k^3 - Z_{k-1}^3) \tag{3.8}$$

3.3 Thermal Stresses and Strains

Due to change in temperature of a composite laminate from processing temperature to room temperature thermal loads are generated which induces thermal stresses in the laminate

$$\begin{Bmatrix}\epsilon_x\\ \epsilon_y\\ \gamma_{xy}\end{Bmatrix} = \left(\begin{Bmatrix}\epsilon_x^0\\ \epsilon_y^0\\ \gamma_{xy}^0\end{Bmatrix} + z\begin{Bmatrix}K_x\\ K_y\\ K_{xy}\end{Bmatrix}\right) - \begin{Bmatrix}\epsilon_x^T\\ \epsilon_y^T\\ \gamma_{xy}^T\end{Bmatrix} \tag{3.9}$$

In which,

$$\begin{Bmatrix}\epsilon_x^T\\ \epsilon_y^T\\ \gamma_{xy}^T\end{Bmatrix} = \begin{Bmatrix}\alpha_x\\ \alpha_y\\ \alpha_{xy}\end{Bmatrix}\Delta T \tag{3.10}$$

The total inplane forces acting on the laminate

$$\begin{Bmatrix}\bar{N}_x\\ \bar{N}_y\\ \bar{N}_{xy}\end{Bmatrix} = \begin{Bmatrix}N_x\\ N_y\\ N_{xy}\end{Bmatrix} + \begin{Bmatrix}N_x^T\\ N_y^T\\ N_{xy}^T\end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16}\\ A_{12} & A_{22} & A_{26}\\ A_{16} & A_{26} & A_{66}\end{bmatrix}\begin{Bmatrix}\epsilon_x^0\\ \epsilon_y^0\\ \gamma_{xy}^0\end{Bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16}\\ B_{12} & B_{22} & B_{26}\\ B_{16} & B_{26} & B_{66}\end{bmatrix}\begin{Bmatrix}K_x\\ K_y\\ K_{xy}\end{Bmatrix} \tag{3.11}$$

Where forces due to change in temperature are

$$\begin{Bmatrix}N_x^T\\ N_y^T\\ N_{xy}^T\end{Bmatrix}_k = \Delta T\sum_{k=1}^{n}\begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16}\\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26}\\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66}\end{bmatrix}_k\begin{Bmatrix}\alpha_x\\ \alpha_y\\ \alpha_{xy}\end{Bmatrix}_k (Z_k - Z_{k-1}) \tag{3.12}$$

The total moments acting on the laminate

$$\begin{Bmatrix}\bar{M}_x\\ \bar{M}_y\\ \bar{M}_{xy}\end{Bmatrix} = \begin{Bmatrix}M_x\\ M_y\\ M_{xy}\end{Bmatrix} + \begin{Bmatrix}M_x^T\\ M_y^T\\ M_{xy}^T\end{Bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16}\\ B_{12} & B_{22} & B_{26}\\ B_{16} & B_{26} & B_{66}\end{bmatrix}\begin{Bmatrix}\epsilon_x^0\\ \epsilon_y^0\\ \gamma_{xy}^0\end{Bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16}\\ D_{12} & D_{22} & D_{26}\\ D_{16} & D_{26} & D_{66}\end{bmatrix}\begin{Bmatrix}K_x\\ K_y\\ K_{xy}\end{Bmatrix} \tag{3.13}$$

Where moments due to change in temperature are

$$\begin{Bmatrix} M_x^T \\ M_y^T \\ M_{xy}^T \end{Bmatrix} = \frac{1}{2}\Delta T \sum_{k=1}^{n} \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix}_k \begin{Bmatrix} \alpha_x \\ \alpha_y \\ \alpha_{xy} \end{Bmatrix}_k (Z_k^2 - Z_{k-1}^2) \tag{3.14}$$

On combining equation 3.11 and 3.13 in compact form

$$\begin{Bmatrix} \epsilon^0 \\ K \end{Bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix}^{-1} \begin{Bmatrix} \mathrm{N} \\ \mathrm{M} \end{Bmatrix} \tag{3.15}$$

3.4 Effective Elastic Properties of the Laminate

$$\bar{E}_x = \frac{1}{h \times {A_{11}}^{-1}},\ \bar{E}_y = \frac{1}{h \times {A_{22}}^{-1}},\ \bar{G}_{xy} = \frac{1}{h \times {A_{33}}^{-1}} \tag{3.16}$$

Flexural modulus

$$(\bar{E}_x)_{flex} = \frac{12}{h^3 \times {D_{11}}^{-1}},\ (\bar{E}_y)_{flex} = \frac{12}{h^3 \times {D_{22}}^{-1}} \tag{3.17}$$

3.5 Failure Analysis

The stresses in the local axes of the lamina are evaluated using transformation matrix as shown below. These stresses are used in failure theories to determine FPF load, LPF load and mode of failure.

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} u^2 & v^2 & 2uv \\ v^2 & u^2 & -2uv \\ -uv & uv & u^2 - v^2 \end{bmatrix} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix} \tag{3.18}$$

3.5.1 Maximum Stress Failure Theory

The lamina will fail if

$$\sigma_1 = (\sigma_1^T)_{ult} \text{ or } (\sigma_1^C)_{ult},\ \sigma_2 = (\sigma_2^T)_{ult} \text{ or } (\sigma_2^C)_{ult},\ \tau_{12} = (\tau_{12})_{ult} \tag{3.19}$$

3.5.2 Tsai-Wu Failure Theory

The lamina will fail if

$$H_1\sigma_1 + H_2\sigma_2 + H_6\tau_{12} + H_{11}\sigma_1^2 + H_{22}\sigma_2^2 + H_{66}\tau_{12}^2 + 2H_{12}\sigma_1\sigma_2 > 1 \tag{3.20}$$

The strength ratio (S.R) is evaluated using the above failure theories to check which lamina fails first.

To perform failure analysis in ANSYS the following flowchart is used

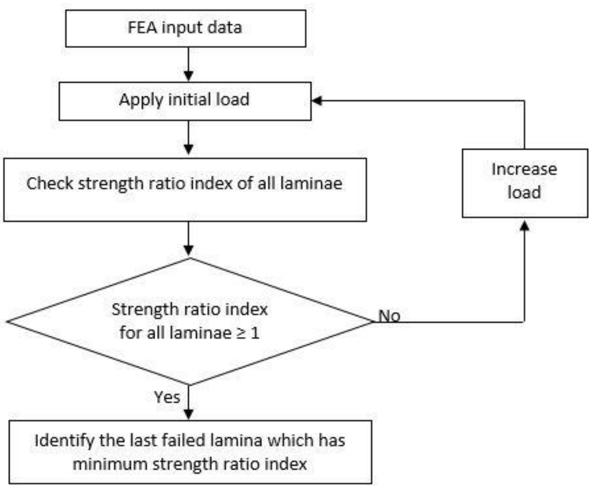


Fig - 4 Process flowchart for failure analysis

4. ANALYTICAL AND ANSYS PROCEDURE APPLIED FOR LAMINATE

Table -1: Material properties [7]

Property	High Strength Carbon / Epoxy	E-Glass / Epoxy
E_1	140 GPa	40 GPa
E_2	10 GPa	8 GPa
G_{12}	5 GPa	4 GPa
ν_{12}	0.3	0.25
$(\sigma_1^T)_{ult}$	1500 MPa	1000 MPa
$(\sigma_1^C)_{ult}$	1200 MPa	600 MPa
$(\sigma_2^T)_{ult}$	50 MPa	30 MPa
$(\sigma_2^C)_{ult}$	250 MPa	150 MPa
$(\tau_{12})_{ult}$	70 MPa	40 MPa
α_1	-0.9×10^{-6}m/m/°C	6.30×10^{-6}m/m/°C
α_2	24×10^{-6}m/m/°C	20×10^{-6}m/m/°C

4.1 Analytical modeling of Case 1

The stacking sequence of the laminate is as shown in fig - 5 below

Fig–5 Stacking sequence

Using the equations from section 3 above the stresses in the each lamina are evaluated.

4.2 Case 1 in ANSYS APDL

Shell 4 node 181 element is used to model the laminate in ANSYS. The shell lay-up when plotted in ANSYS looks as shown in fig-6 below

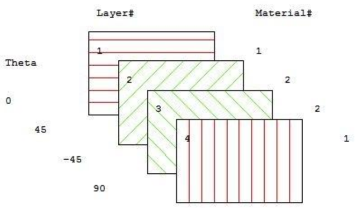


Fig-6 Shell lay-up

The laminate model after performing meshing is as shown in fig-7. The loads and boundary conditions are applied to the laminate

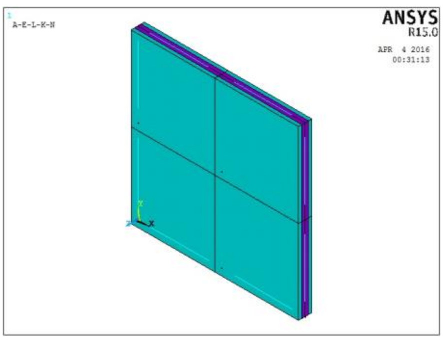


Fig-7 Laminate after meshing

The stresses obtained using ANSYS are plotted for each lamina and entire laminate

For 0° Carbon/Epoxy lamina

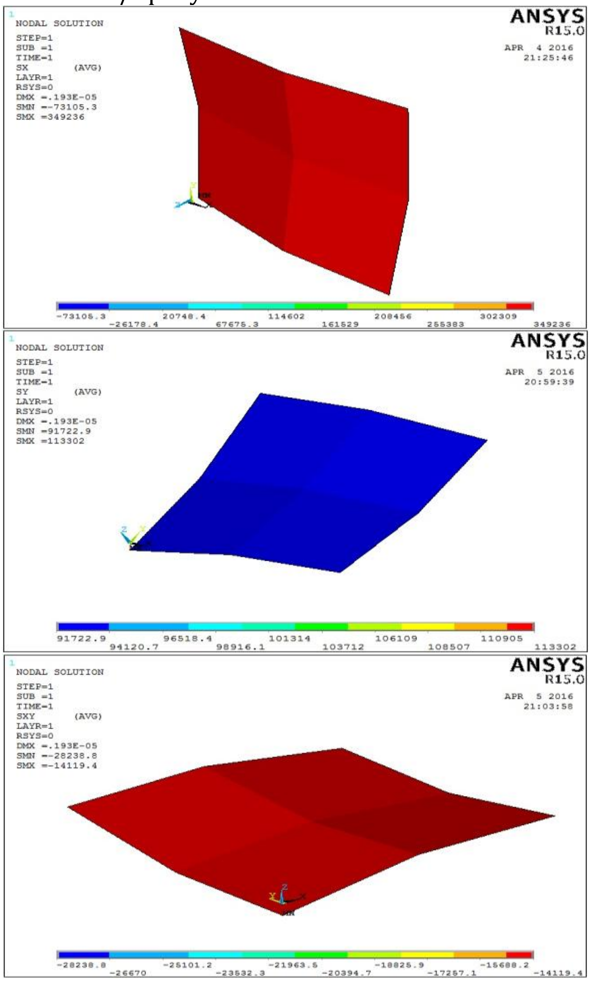
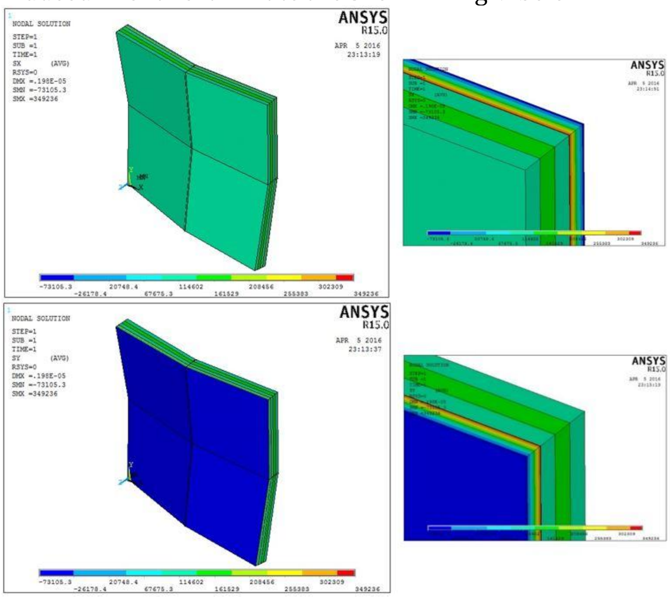


Fig-8 Stress in x, y and xy shear for lamina 1

Similarly the stresses are plotted for all laminae. The stress induced in entire laminate are shown in fig-9 below



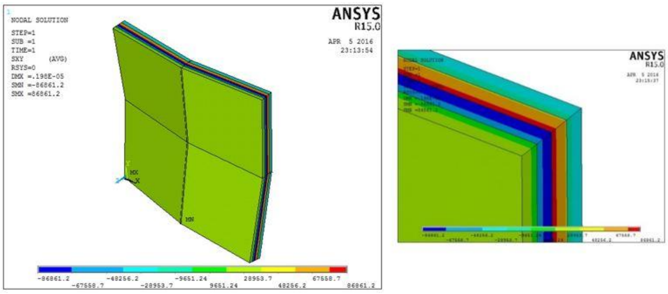


Fig-9 Stresses in entire laminate

4.3 Failure Analysis of Case 1

Using Maximum Stress Failure Theory and Tsai-Wu Failure Theory strength ratio for all the laminae are calculated

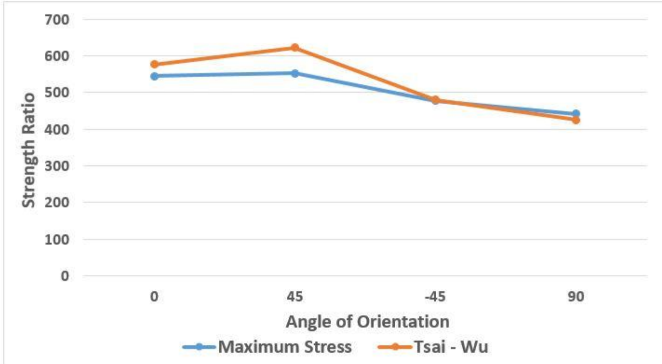


Chart -1: Strength ratio v/s Angle of Orientation

The lamina with least value of strength ratio will fail first which gives the FPF load. The strength ratio in ANSYS is inverse of strength ratio calculated analytically. The variation of strength ratio w.r.t load obtained by ANSYS is as shown below

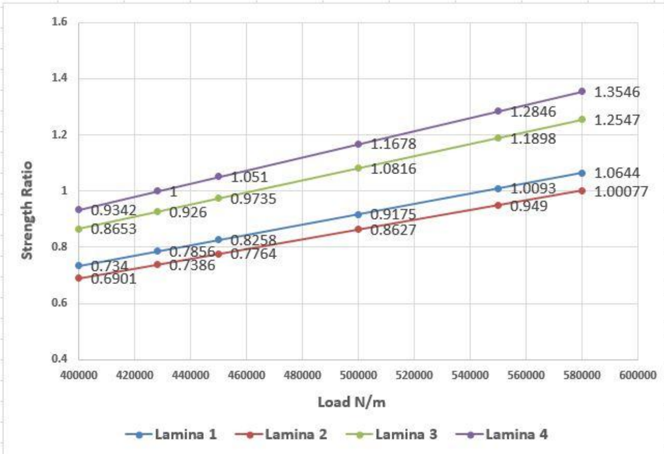


Chart -2: Strength ratio v/s Load

4.4 Analytical modeling of Case 2

The stacking sequence of the laminate is

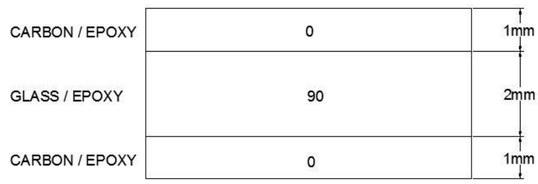


Fig-10 Stacking sequence

Using the equations from section 3 above the stresses in each lamina are evaluated.

4.5 Case 2 in ANSYS APDL

Shell 4 node 181 is used to model the laminate. The shell lay-up when plotted in ANSYS is as shown below

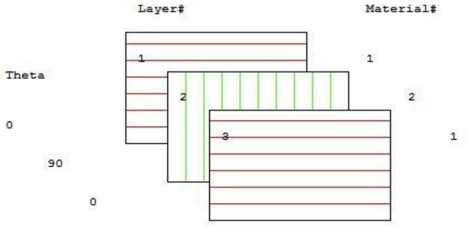


Fig-11 Shell lay-up

The laminate model after meshing is as shown in fig 12

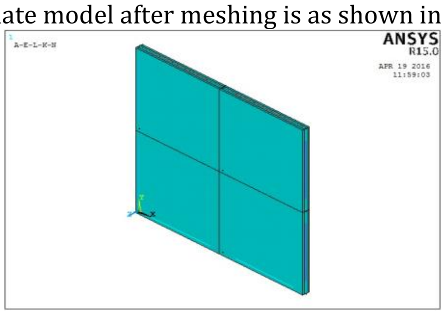


Fig-12 Laminate after meshing

The stresses for 0° Carbon/Epoxy lamina

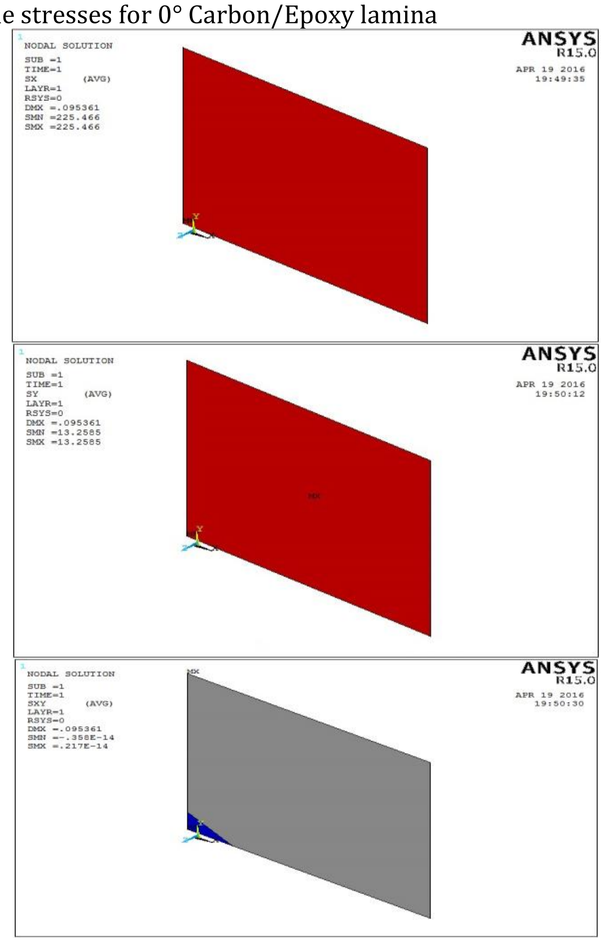


Fig-13 Stress in x, y and xy shear for lamina 1(CE 0°)

The stresses for entire laminate

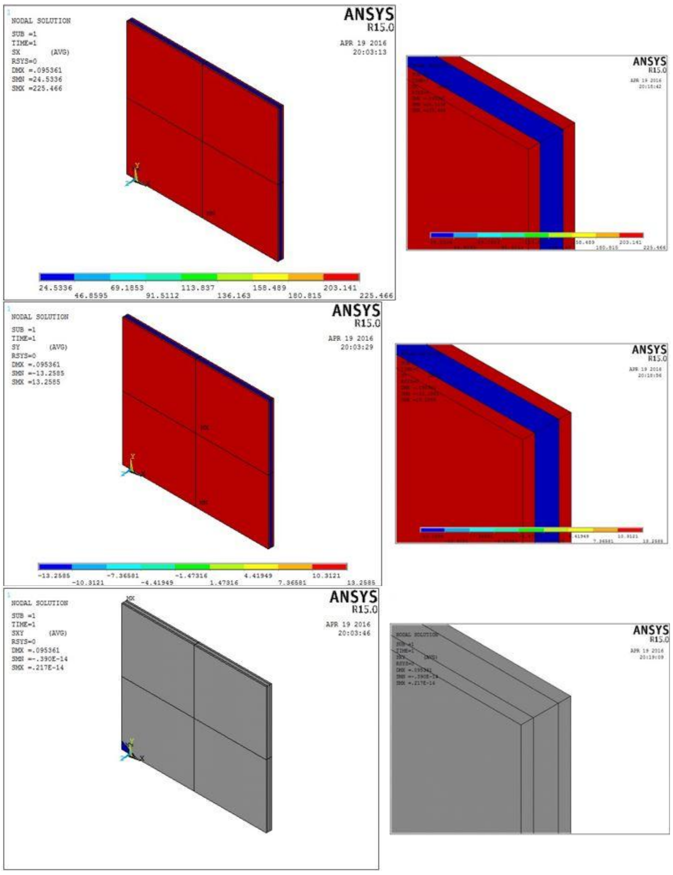


Fig-14 Stress in x, y and xy shear for entire laminate

4.6 Failure Analysis of Case 2

The plot of strength ratio v/s angle of orientation for comparison of failure theories is plotted

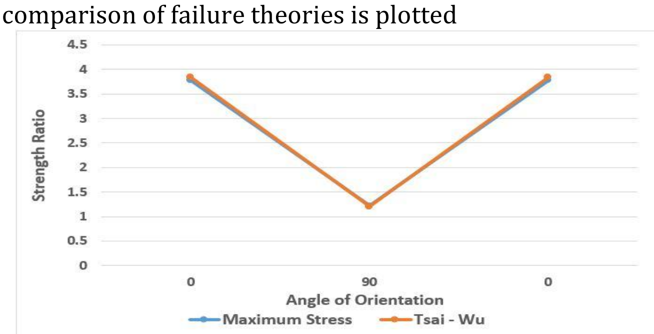


Chart-3 Comparison of failure theories

The variation of strength ratio w.r.t load for Case 2 obtained by ANSYS is as shown below

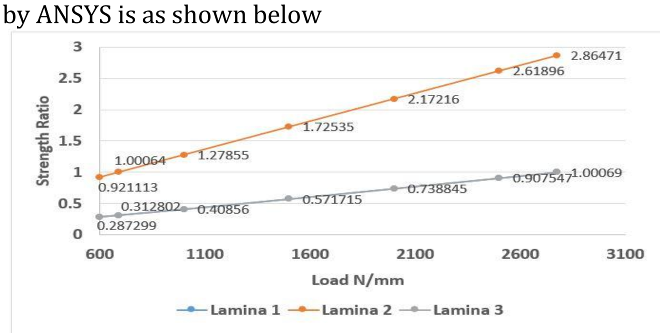


Chart-4 Strength ratio v/s Load

5. RESULTS

The stresses obtained by ANSYS are validated by stresses evaluated analytically using CLT and % Error is calculated for both the cases. The failure theories are used to calculate FPF and LPF load using the stresses obtained and by comparing the strength ratios for both the cases.

5.1 Comparison of Stresses for Case 1

Stresses generated in each laminae

Lamina No.	Stress		ANSYS (Pa)	CLT (Pa)	% Error
1 0^0 Carbon / Epoxy	σ_x	Min	-73105.3	-73198	0.126
		Max	349236	349310	0.021
	σ_y	Min	91722.9	91700	0.024
		Max	113302	113300	0.0017
	τ_{xy}	Min	-28238.8	-28246	0.025
		Max	-14119.4	-14123	0.025
2 45^0 Glass / Epoxy	σ_x	Min	93768.7	93768	0.00074
		Max	141149	141160	0.0077
	σ_y	Min	141149	141160	0.0077
		Max	142776	142790	0.0098
	τ_{xy}	Min	55405.3	55419	0.024
		Max	86861.2	86884	0.026
3 -45^0 Glass / Epoxy	σ_x	Min	141149	141160	0.0077
		Max	142776	142790	0.0098
	σ_y	Min	93768.7	93768	0.00074
		Max	141149	141160	0.0077
	τ_{xy}	Min	-86861.2	-86884	0.026
		Max	-55405.3	-55419	0.024
4 90^0 Carbon / Epoxy	σ_x	Min	91722.9	91700	0.024
		Max	113302	113300	0.0017
	σ_y	Min	-73105.3	-73198	0.126
		Max	349236	349310	0.021
	τ_{xy}	Min	14119.4	14123	0.025
		Max	28238.8	28246	0.025

Stresses generated in entire laminate

Stress		ANSYS (Pa)	CLT (Pa)	% Error
σ_x	Min	-73105.3	-73198	0.126
	Max	349236	349310	0.021
σ_y	Min	-73105.3	-73198	0.126
	Max	349236	349310	0.021
τ_{xy}	Min	-86861.2	-86884	0.026
	Max	86861.2	86884	0.026

Effective elastic properties of hybrid composite laminate are

Effective elastic properties	Value
$\bar{E}_x$	45.52 GPa
$\bar{E}_y$	45.52 GPa
$\bar{G}_{xy}$	8.07 GPa
$(\bar{E}_x)_{flex}$	67.98 GPa
$(\bar{E}_y)_{flex}$	67.98 GPa

5.2 Failure Analysis of Case 1

Load (N/m)	Lamina	Mode of failure	Remarks
4.28×10^5	Carbon / Epoxy 90^0	Matrix failure in transverse direction	First Ply Failure
5.8×10^5	Glass / Epoxy 45^0	Matrix failure in transverse direction	Last Ply Failure

5.3 Comparison of Stresses for Case 2

Stresses generated in each laminae

Lamina No.	Stress	ANSYS (Pa)	CLT (Pa)	% Error
1 0^0 Carbon / Epoxy	σ_x	225.466	225.465	0.00017
	σ_y	13.2585	13.2462	0.092
	τ_{xy}	0	0	0
2 90^0 Glass / Epoxy	σ_x	24.5336	24.5342	0.0024
	σ_y	-13.2585	-13.2462	0.092
	τ_{xy}	0	0	0
3 0^0 Carbon / Epoxy	σ_x	225.466	225.465	0.00017
	σ_y	13.2585	13.2462	0.092
	τ_{xy}	0	0	0

Stresses generated in entire laminate

Stress		ANSYS (Pa)	CLT (Pa)	% Error
σ_x	Min	24.5336	24.5342	0.0024
	Max	225.466	225.465	0.00017
σ_y	Min	-13.2585	-13.2462	0.092
	Max	13.2585	13.2462	0.092
τ_{xy}	Min	0	0	0
	Max	0	0	0

Effective elastic properties of hybrid composite laminate are

Effective elastic properties	Value
$\bar{E}_x$	74.25 GPa
$\bar{E}_y$	25.19 GPa
$\bar{G}_{xy}$	4.5 GPa

5.2 Failure Analysis of Case 2

Load (N/mm)	Lamina	Mode of failure	Remarks
689	Glass / Epoxy 90^0	Matrix failure in transverse direction	First Ply Failure
2775	Carbon / Epoxy 45^0	Fracture of fiber (longitudinal tensile fracture)	Last Ply Failure

6. CONCLUSIONS

The use of ANSYS APDL to perform analysis of composite laminate can be justified by results obtained. The stresses generated in laminate for both the cases using ANSYS APDL are in good agreement with stresses evaluated analytically using CLT with error of less than 1%. The failure analysis of laminate for both the cases gave FPF and LPF load of the laminate along with the mode of failure of the lamina.

Based on the results obtained the following conclusions are drawn

1) A laminate with any number of laminae, different orientations, and different materials can be analyzed using ANSYS.
2) Any type of loads can be applied such as mechanical and/or thermal.
3) The stresses obtained are used to perform failure analysis and determine material properties of the laminate.
4) The exact stacking sequence can be determined which reduces the time required to perform experimental procedure to evaluate material properties of the laminate for different stacking sequence.

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