https://doi.org/10.56452/2022-11-004

ISSN: 0974-5823 Vol. 7 No. 11 November, 2022

International Journal of Mechanical Engineering

An experimental investigation and analysis of quasi-isotropic composite laminate for in-plane and out of plane loading conditions

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Abstract: Research study has been conducted to interpret the failure difference between two quasi-isotropic laminates fabricated for different ply sequences. The material chosen for experimental study is AS_4, the laminates prepared for different ply sequences are [0/90/45/-45] and [0/45/-45/90]. Laminates are quasi-isotropic in nature. Sample specimens examined for compression, tension and flexural test and results revealed that behaviour of specimen in tension and compression is same and shown deviation under flexural loading conditions. CLT (Classical Lamination Theory) and numerical method is used to calculate stresses in specimen for out of plane loading conditions (As deviation has shown for flexural test which meant out of plane loading condition). Experimental results, FEM results and numerical results are studied and compared with each other.

Keywords: CLT, FEM, Flexural strength/Test, CFRP (Carbon Fibre Reinforced Plastics) etc.

1 Introduction

Composite material are widely used in aerospace, automobile, structural application due to their high strength to weight ratio, the versatility of their use increasing day by and thus they are attracting the researchers to investigate them to further depth. The laminate can fabricate for any kind of behaviour such as quasi-isotropic, isotropic, orthotropic, anisotropic etc. strength analysis and behaviour study under applied loading conditions of composite is totally different and quite complicated compared to metal or homogeneous material. In-plane and out of plane loading responses/behaviour for composite material do vary totally. There are different kind of failure criteria's/theory available for composite material/structure, few of them are interactive and few are non-interactive. The maximum stress, strain criteria's are non-interactive and independent failure criteria's where criteria such as Tsai-Hill and Tsai-Wu are interactive and dependant. Numerous studies has been undertaken for out of plane loading and in-plane types of loading.

In research work, aerospace grade carbon is fabricated to achieve two different combinations of laminate, the staking sequence of ply will be vary in thickness direction. The mechanical behaviour of these two differently (For ply sequence) fabricated laminate is studied for tensile, compressive and flexural load. The experimental results compared with numerical and analytical results and they supports each other.

1.1 Material Selection

Industry standard carbon AS_4 was chosen to prepare the laminate. The sequence of laminate layer has kept varied so that difference in mechanical behaviour could have been studied under different loading conditions. Each laminate has 16 layers, for identification purpose they have been named as S_1 and S_2 i.e. specimen one and specimen two respectively. The specimens were manufactured with an Autoclave moulding techniques, the trapped air is forced to flow out, the specimen then cut by using diamond cutter wheel for required length.

The specimen, [0/45/-45/90]: S_1

And, [0/90/45/-45]: S_2

The material for specimen one is, carbon and for specimen two is AS_4. Thickness of each ply is 0.16 mmThe mechanical properties of lamina depicted in the table below,

Sr. No.	Property of the material	Value
1	Longitudinal Young's Modulus, E_1	135 GPa
2	Transverse Young's Modulus, E_2	10 GPa
3	Shear Modulus, G_{12}	5 GPa
4	Poison's ratio in the plane 1-2, μ_{12}	0.25
5	Longitudinal material strength in tension, X_t	1800 MPa
6	Longitudinal material strength in compression, X_c	1130 MPa
7	Transverse material strength in tension, Y_t	50 MPa
8	Transverse material strength in compression, Y_c	210 MPa
9	Shear strength	100 MPa
10	Density of the material	1575 kg/m^3

Table (1.1.1): Material properties measured experimentally.The figure below gives an idea about stacking sequence of the laminate which are named as specimen-one and specimen two i.e. S_1 and S_2.

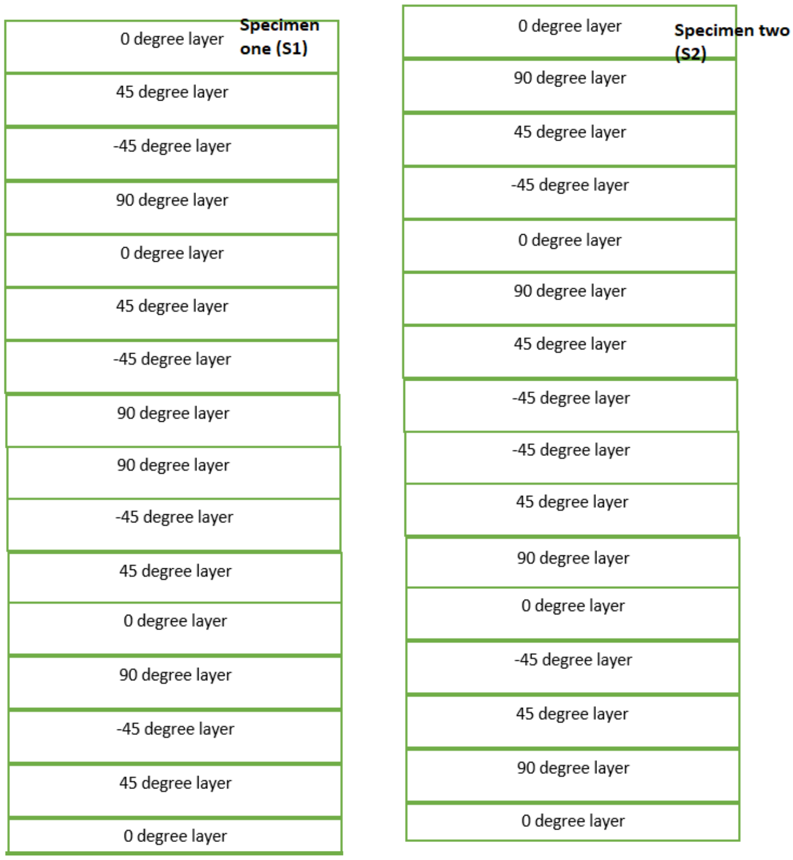

Fig (1.1.1): Laminate/Specimen sequence for specimen one and specimen two Mathematical Modelling: CLT theory is used to stress and strain values. The calculations are done on Microsoft Excel. The stress and strain at global coordinate system is depicted through the equation below. Values can export to any general material axis of the laminate/lamina.The stress obtained through excel calculations is global stress and can transformed to any general material axis of the given laminate by using following transformation equation,

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \sigma_z \end{bmatrix} = [T]^{-1} \begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \end{bmatrix}$$

$[T]^{-1}$ = Transpose or transformation matrix = $\begin{bmatrix} C^2 & S^2 & -2SC \\ S^2 & C^2 & 2SC \\ SC & -SC & C^2 - S^2 \end{bmatrix}$

C = Cosθ and S = Sinθ

The strain can obtained by using following equation of matrix,

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \sigma_z \end{bmatrix} = [T]^{-1} \, [Q] \begin{bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{bmatrix}$$

[Q] is reduced stiffness matrix and can be expressed as follows,

$$[Q] = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix}$$

Reduced stiffness matrix components can be calculated as follows,

$Q_{11} = E_1/(1-\gamma_{21}\gamma_{12})$, $Q_{12} = \gamma_{12}E_2/(1-\gamma_{21}\gamma_{12})$, $Q_{22} = E_2/(1-\gamma_{21}\gamma_{12})$, $Q_{66} = G_{12}$

$$\begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{bmatrix} = \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \varepsilon_{xy}^0 \end{bmatrix} + Z \begin{bmatrix} k_x \\ k_y \\ k_{xy} \end{bmatrix}$$

ε_x^0, ε_y^0, ε_{xy}^0 are the mid plane strain and k_x, k_y, k_{xy} are curvature terms.

The equation above shows linear relationship between strain and curvature. Laminate stresses are the function of "Z". The stiffness components changes from layer to layer. The relationship between laminate stress, stiffness matrix, mid-plane strain and curvature term is given below,

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \sigma_z \end{bmatrix} = \begin{bmatrix} \overline{Q_{11}} & \overline{Q_{12}} & \overline{Q_{16}} \\ \overline{Q_{12}} & \overline{Q_{22}} & \overline{Q_{26}} \\ \overline{Q_{16}} & \overline{Q_{26}} & \overline{Q_{66}} \end{bmatrix} \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \varepsilon_{xy}^0 \end{bmatrix} + Z \begin{bmatrix} \overline{Q_{11}} & \overline{Q_{12}} & \overline{Q_{16}} \\ \overline{Q_{12}} & \overline{Q_{22}} & \overline{Q_{26}} \\ \overline{Q_{16}} & \overline{Q_{26}} & \overline{Q_{66}} \end{bmatrix} \begin{bmatrix} k_x \\ k_y \\ k_{xy} \end{bmatrix}$$

Transformed laminate stiffness matrix coefficients are given by means of following equation,

$\overline{Q_{11}} = Q_{11}\, c^4 + Q_{22}\, s^4 + 2\,(Q_{12} + 2\, Q_{66})\, s^2\, c^2$

$\overline{Q_{22}} = Q_{11}\, s^4 + Q_{22}\, c^4 + 2\,(Q_{12} + 2\, Q_{66})\, s^2\, c^2$

$\overline{Q_{12}} = (Q_{11} + Q_{22} - 4\, Q_{66})\, s^2\, c^2 + Q_{12}\,(c^4 + s^4)$

$\overline{Q_{16}} = (Q_{11} - Q_{12} - 2\, Q_{66})\, c^3\, s - (Q_{22} - Q_{12} - 2\, Q_{66})\, c\, s^3$

$\overline{Q_{26}} = (Q_{11} - Q_{12} - 2\, Q_{66})\, c\, s^3 - (Q_{22} - Q_{12} - 2\, Q_{66})\, c^3\, s$

$\overline{Q_{66}} = (Q_{11} + Q_{22} - 2\, Q_{12} - 2\, Q_{66})\, s^2\, c^2 + Q_{66}\,(s^4 + c^4)$

The resultant forces (In-Plane) and moments (Out of plane) can be written in matrix form as follows,

$$\begin{bmatrix} N_x \\ N_y \\ N_z \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \varepsilon_{xy}^0 \end{bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} k_x \\ k_y \\ k_{xy} \end{bmatrix}$$

$$\begin{bmatrix} M_x \\ M_y \\ M_z \end{bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \varepsilon_{xy}^0 \end{bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{bmatrix} k_x \\ k_y \\ k_{xy} \end{bmatrix}$$

[A], [B] and [D] matrix are called as laminate stiffness matrix. The matrix depicts relations between loads applied on the laminate and hence stress, strains and curvatures produced. Matrix respectively named or identified with following nomenclature,

[A]: Material extensional matrix

[B]: Material coupling and extensional matrix

[D]: Material bending matrix

The matrix can expressed mathematically as follows,

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

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

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

Where, i and j = 1, 2 and 6…….

For symmetric laminates, the matrix [B] = 0 thus load, strain and curvature equation can be re-written as follows,

$$\begin{bmatrix} N_x \\ N_y \\ N_z \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \varepsilon_{xy}^0 \end{bmatrix}$$

$$\begin{bmatrix} M_x \\ M_y \\ M_z \end{bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \varepsilon_{xy}^0 \end{bmatrix}$$

In above matrix, load are in-plane and moments are out of plane entities.

Simulation

Two laminates stacked for different sequence and subjected to tensile, compressive and flexural loading are simulated through FEM. Shells are two dimensional quadrilateral elements used in modelling of the laminate. The size of the mesh was 2*2 mm. It is noted that, location between supports is under constant bending. The part of laminate extended beyond support and hence can be treated as an overhanging beam is subjected to shear. The bottom most layer of laminate subjected to tension and innermost layer to compression. Maximum load is taken by outer layer i.e. 0 degree layer of the laminate. The stress distribution across thickness of 0 degree layer is linear and symmetric about neutral axis of the laminate, such symmetry don't shows by other layers in the laminate and thus contributes towards failure of the laminate. The bottom most layer i.e. 1st layer from bottom i.e. 0 degree layer is subjected to tension and top most i.e. 16th layer or first layer from top subjected to compression. In laminate one uppermost layer is subsequently placed with 45 degree layer and in laminate two uppermost layer is subsequently placed with 90 degree layer. The stress percolated by next layer in specimen will be of the same nature and type but 35% less in value where stress carried by subsequent layer in specimen two will be of opposite nature and yet 20% less in value. Major stress carrying and contributing layer/lamina is 0 degree and it possesses approximately 55 to 60% of the total stress induced or acting on the structure. And this shows how stress baring capacity of structure varies with respect to varying stacking sequence of the ply in laminate. FEM analysis carried help to elaborate load carrying capacity of each layer, stress induced and failure sequence occurred etc.

Experimentation

Test on specimen were conducted by using servo-hydraulic computer connected machine. Three specimens were tested for each type of loading i.e. tensile, compressive and flexural etc. The results were plotted to do the detail analysis and to withdraw the conclusion.

The experimental set up for two point bending test is depicted through following block diagram,

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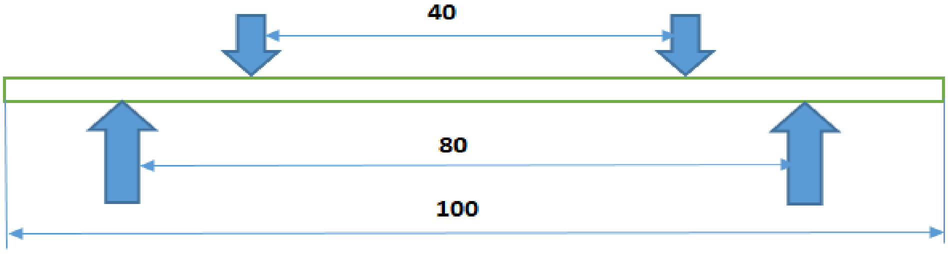


Fig (1.1.2): Specimen and loading conditions

The laminate has width 20 mm and thickness 2.4 mm. the bottom most layer subjected to the tension and top most layer subjected to compression. The stress distribution along length in the specimen is depicted in the table below,

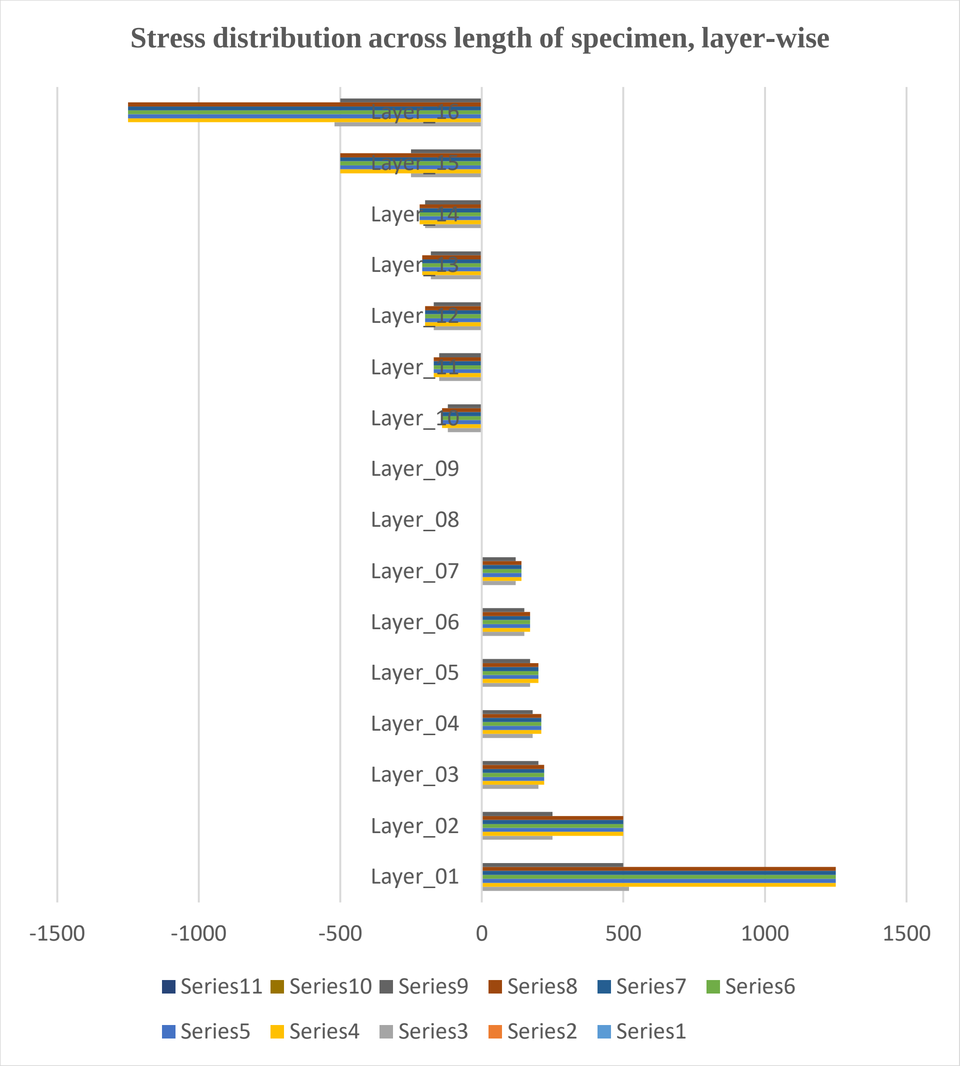


Fig (1.1.3): Stress distribution in laminate, layer wise along length.

The graph above shows flexural stress distribution across length of specimen, the results revealed that, part of the specimen beyond support/overhanging beam subjected to minimum or no stress where middle portion of specimen or part of specimen between load acting is subjected to maximum stress. The magnitude of stress noted zero at one extreme end of specimen and starts increasing while moving at another extreme end of the specimen, it noted highest and yet constant for middle portion or part of the specimen between load acting and then starts decreasing while reaching at another extreme end and finally gets zero while reaching to another extreme end. The magnitude of stress at middle layer/neutral layer is zero. The top and bottom layer of specimen subjected to compression and tension respectively. Layers above neutral axis subjected to compression and below are subjected to tension so the nature of stress in each layer, upper and lower half of the laminate about neutral axis is depicted through graph above.

The flexural stress distribution across thickness of specimen is depicted through graph below,

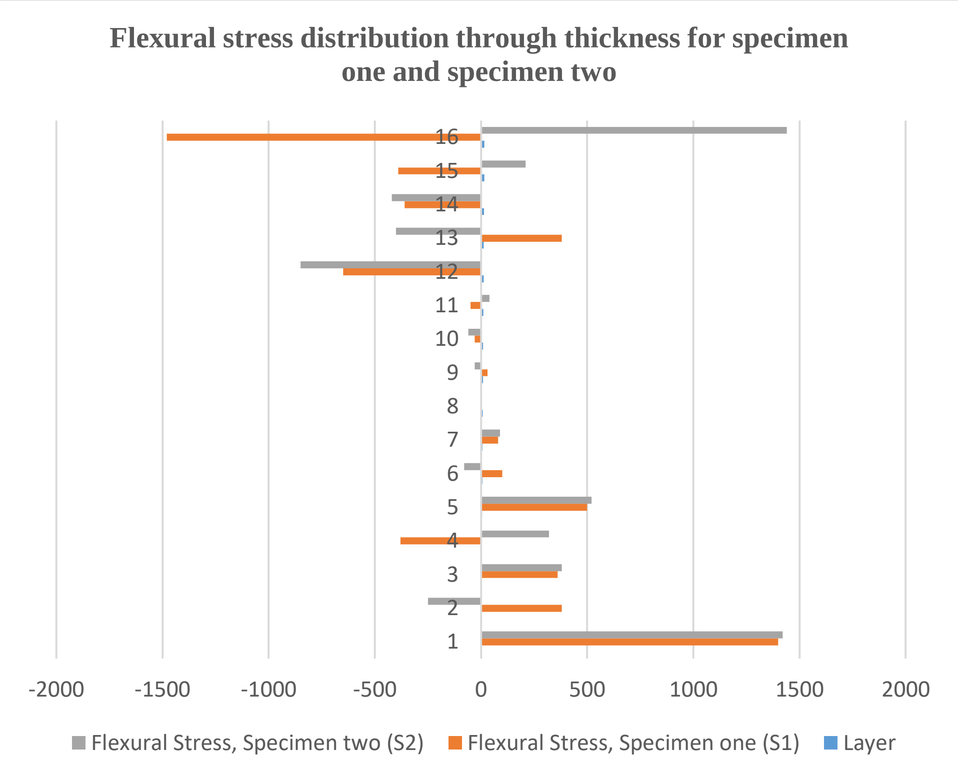


Fig (1.1.4): Flexural strength of specimen through thickness for specimen one and specimen two.

2. Tensile, compressive and flexural properties finding of laminate

Tensile, compressive and flexural tests were carried on laminate specimen as per ASTM standards, so the specimen also prepared remains stick to the standards. The strain gage is attached to the middle of specimen so that movement with respect to loading can be measured. The failure noted in specimen was explosive gauge and few were failed by angle gage when loading nature was tensile. The length of specimens considered in compression was 140 mm. strain gages are mounted back to back to measure and monitor an alignment of the specimen with respect to loading axis. Precaution has taken that failure will be occurred in specimen will be virtue of compression only. The gauge length of specimen between wedge grips in both the cases was 10 mm. In flexural test four point bending configuration with support given to span length to depth ratio chosen 31:1. To avoid failure of the specimen due to delamination distance between upper and lower roller was kept minimum and compact i.e. 20 mm. Top and bottom layer is equipped with strain gages to predict modulus of flexure.

Location 31 and 71 mm comes under upper roller and location 11 and 91 mm do come under bottom roller respectively.

The table below show tensile, compressive and flexural strength for laminate one and laminate two respectively,

Laminate one/Specimen one	Tensile strength, MPa	720
	Compressive strength, MPa	560
	Flexural strength, MPa	1100
Laminate two/Specimen two	Tensile strength, MPa	750
	Compressive strength, MPa	570
	Flexural strength, MPa	1000

Table (2.1): Laminate engineering elastic properties

3. Result comparison between numerical, experimental and FEM model

Laminate/specimen chosen in research work are quasi-isotropic in nature, they has equal extensional stiffness for planer loading conditions i.e. [A] matrix is similar for both the specimen. The components A_{16} and A_{26} are zero for quasi isotropic laminate i.e. shear-extension coupling component do not exists. Also $A_{11} = A_{22}$ and [D] matrix vary for specimen one and specimen two for out of plane loading conditions. Component D_{16} and D_{26} exists in this case i.e. twisting moment exists and thus layup should done in such a way that twisting moment should be avoided. [A], [B] and [D] matrix are obtained from Classical Lamination Theory. [A] Matrix behaviour for specimen one and two behaves similarly for in-plane loading where matrix [D] behaves differently for both specimen for out of plane load acting. Stress for in plane loading i.e. tension and compression for specimen one and two is same but it changes when loading nature changes from in plane to out of plane loading i.e. bending. CLT/Mathematical and FEM/Simulation results matches quite with experimental results for tensile loading and for compression loading it do not matches at all. There noted deviation between compressive and tensile strength of the molded laminate which is subjected to three point bending test. This deviation is noted approximately 20%. Compression strength noted less than tensile strength. Compression failure is noted due to fibre de-bonding, kinking and buckling that educes load carrying capacity of laminate which is subjected to the test as described above. FEA and CLT analysis shows similar strain responses of the specimen, along with reduced material modulus and failure strength, provided effect of bucking, kinking and de-bonding does not take into consideration. In experimentation failure was noted in linear range due to occurrence of the large deflection in bending. Tensile strain is lower than compressive strain for given stress values. In FEM and CLT compressive strain are in good agreement with tensile strain and that reveals CLT and FEM analysis gives similar value for strain in tension and compression.

The values for [A], [B] and [D] matrix for specimen one and specimen two are given as below,

$$[A]_{S1} = \begin{bmatrix} 140 & 42 & 0 \\ 42 & 140 & 0 \\ 0 & 0 & 47.6 \end{bmatrix} \text{MN/m} \quad \& \quad [A]_{S2} = \begin{bmatrix} 140 & 42 & 0 \\ 42 & 140 & 0 \\ 0 & 0 & 47.6 \end{bmatrix} \text{MN/m}$$

$$[B]_{S1} = \begin{bmatrix} 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix} \text{N} \quad \& \quad [B]_{S2} = \begin{bmatrix} 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix} \text{N}$$

$$[D]_{S1} = \begin{bmatrix} 87.38 & 18.6 & 3.5 \\ 18.6 & 46.12 & 3.5 \\ 3.5 & 3.5 & 21.40 \end{bmatrix} \text{N-m} \quad \& \quad [D]_{S2} = \begin{bmatrix} 81.3 & 13.8 & 2.5 \\ 13.8 & 63.8 & 2.5 \\ 2.5 & 2.5 & 16.8 \end{bmatrix} \text{N-m}$$

The laminate elastic constants values for in-plane and out-of-plane loading are summarized in the table below,

Laminate elastic constants for In-Plane loading (Tension and compression)

Laminate	E_X, GPa	E_Y, GPa	G_{XY}, GPa	ϑ_{xy}	ϑ_{yx}
Specimen one (S_1)	52	52	20	0.3	0.3
Specimen one (S_2)	52	52	20	0.3	0.3

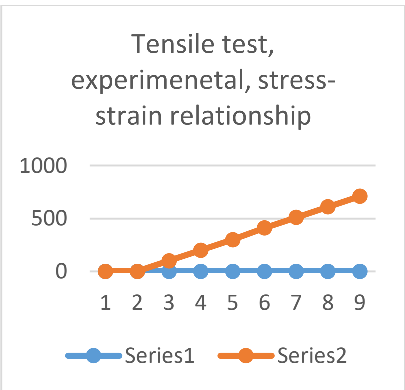
Table (3.1): Laminate elastic properties for in plane loading (Tension and compression)

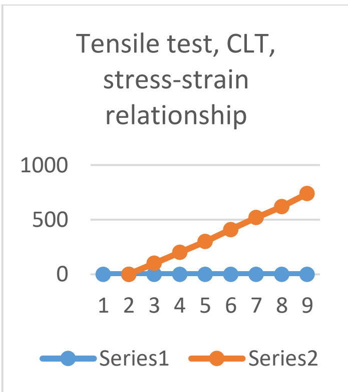
Laminate elastic constants for Out-of-Plane loading (Bending/Flexural)

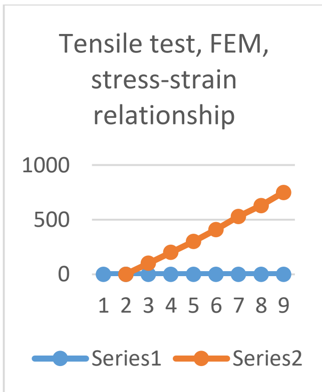
Laminate	E_X, GPa	E_Y, GPa	G_{XY}, GPa	ϑ_{xy}	ϑ_{yx}
Specimen one (S_1)	69	37	19	0.4	0.23
Specimen one (S_2)	68	54	14	0.213	0.178

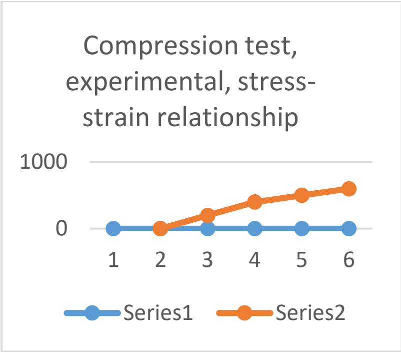
Table (3.2): Laminate elastic properties for out of plane loading (Bending/Flexural)

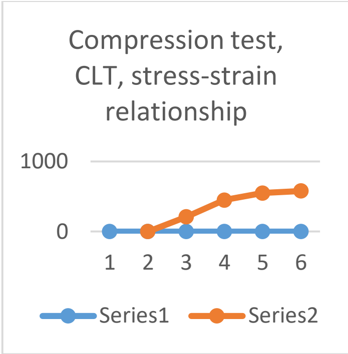
The laminate behavioural analysis for tensile, compressive and flexural loading is depicted through various graphs as follows,

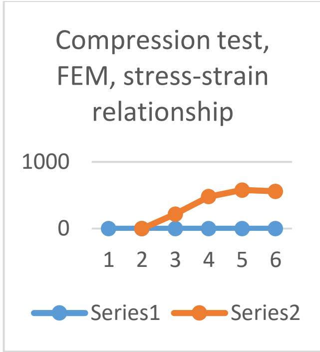






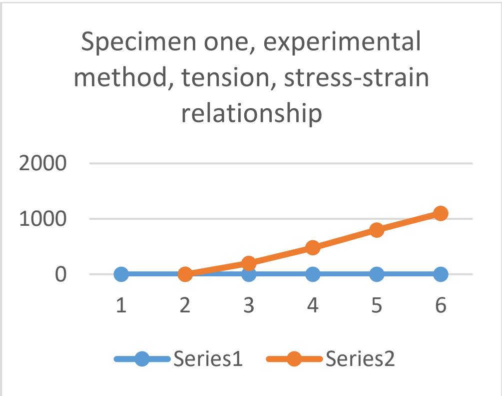


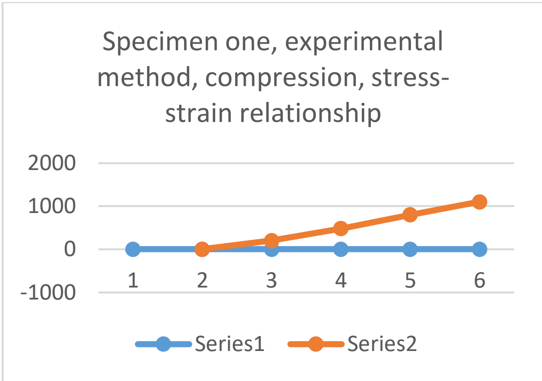


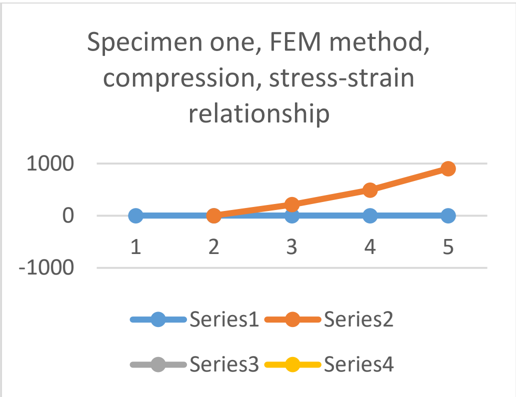


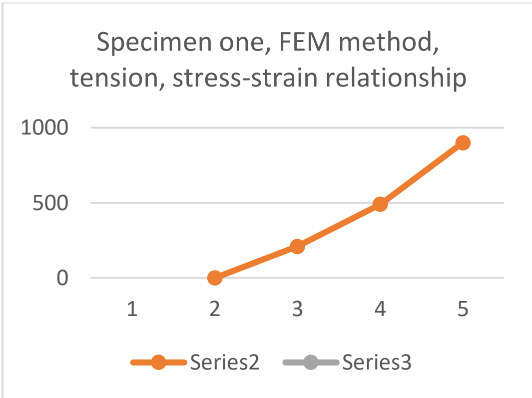
The graph above depicts, experimental, CLT and FEM results shows resemblance with each other for tensile and compression loading.

The following graphs shows, stress-strain behaviour of specimen one and two for tensile and compressive loading, the graphs are plotted and compared to check result accuracy obtained through experimental, CLT and FEM method.









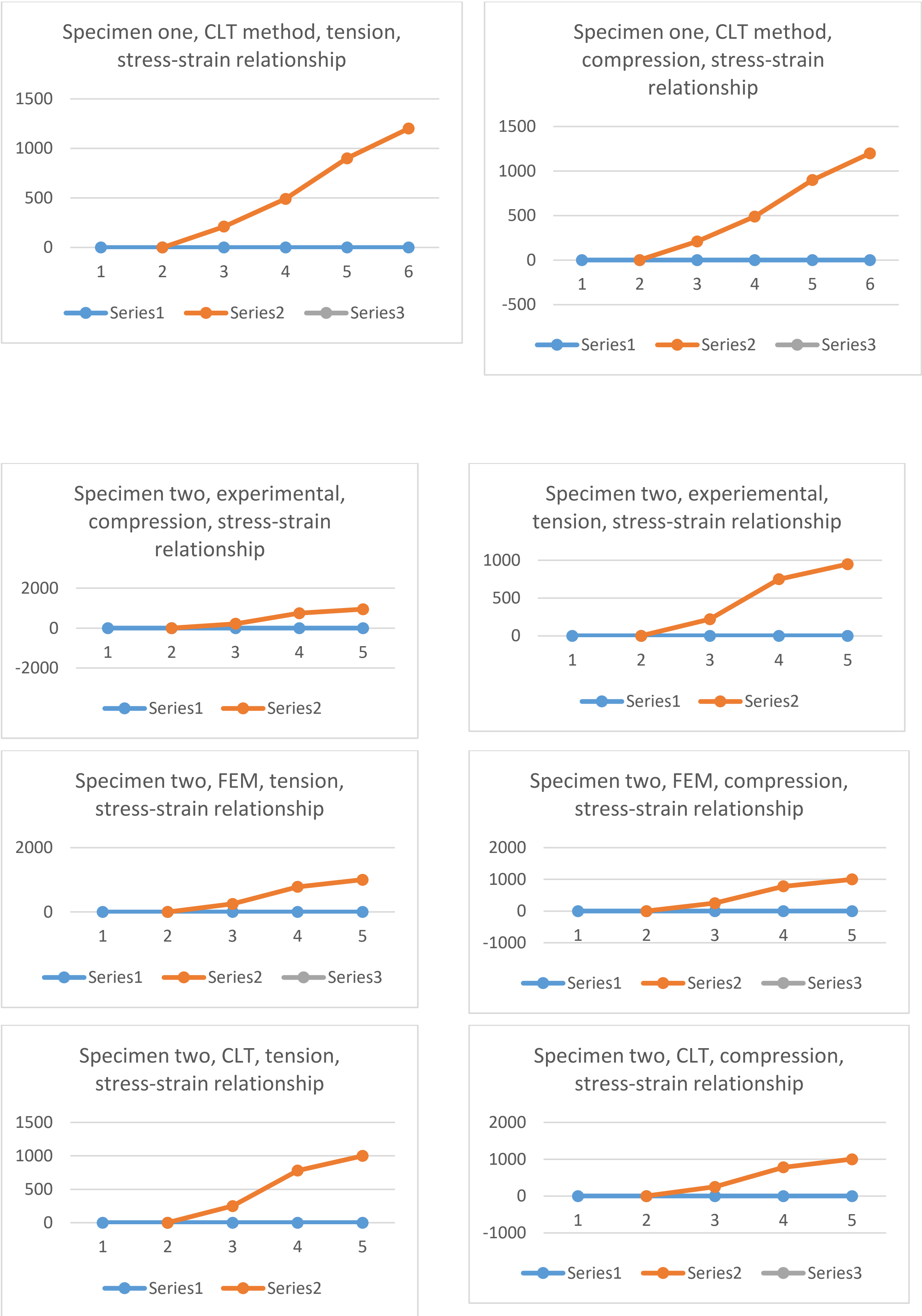


Fig (3.1): Laminate behaviour for tensile, compressive and flexural loading conditions

The table below gives comparative predicted values for compressive strength, tensile strength and flexural strength for laminate one/specimen one and specimen two,

	Laminate one/Specimen one	
Tensile strength	Predicted	650
	Experimental	720
Compressive strength	Predicted	630
	Experimental	560
Flexural strength	Predicted	1000
	Experimental	1100

Table (3.3): Laminate one/Specimen one predicted/FEM vs experimental strength comparison

	Laminate Two/Specimen Two	
Tensile strength	Predicted	650
	Experimental	750
Compressive strength	Predicted	630
	Experimental	570
Flexural strength	Predicted	910
	Experimental	1000

Table (3.4): Laminate one/Specimen one predicted/FEM vs experimental strength comparison

4. Failure strength

Tsai-Wu based failure criteria is used to predict strength of the laminate for in plane and out of plane loadings. Progressive failure of lamina noted for in-plane loading conditions. The failure sequence for first, second and third ply were noted for 90, 45 and 0 degree layer where for out of plane loading the failure sequence for first, second and third ply was 0, 45 and 90 degree layer. Bending strain component produced due to flexural load varies linearly where stress component value is based on individual ply. Stress value which alters with respect to ply also effect the failure behaviour of the ply. In specimen one and two the failure is on compressive side. Huge compressive stress, stress concentration along with deflection/displacement causes the first ply failure. The tensile failure followed by shear failure and this failure propagates through next layers and reaches to +45 and -45 plies junction.

5. Conclusion

- In laminate types i.e. quasi isotropic, for planer loadings for tensile and compression, moduli are un-affected.
- For out of plane loadings, same laminate shows considerable different ability of load carrying capacity, sequence of the failure and arresting capacity of propagated crack.
- Composite laminates fails in linear elastic manner, FEM analysis shows much resemblance with kind of failure of laminate mentioned.
- Strength values predicted through FEM analysis shows close resemblance and in goo agreement with experimental values.

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