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Finite Element Analysis of Composite Rectangular Panel with Different Stacking Sequences and Ply-Orientations

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Abstract - *Composite materials are widely used in aerospace industry due to their high performance and characteristics like strength to weight ratio, stiffness to weight ratio etc. In this paper, finite element analysis for flat rectangular panel is done with different stacking sequences (symmetric) and different ply orientations, the ply orientations considered for the analysis are 0^0, $\pm45^0$ and 90^0 with a total number of 28 plies. Roark's formula is considered for the calculation of critical buckling stress when the panel is under equal uniform compression on two opposite short edges and is simply supported on all edges. After the analysis, buckling factor and buckling mode shapes are observed for the different stacking sequences and ply orientations. ABD matrix as well as elastic modulus in longitudinal direction E_{11}, modulus in transvers direction E_{22}, Poisson's ratio Nu_{11} and shear modulus G_{12} are tabulated for each stacking sequence. From the analysis best stacking sequence for the applied loading condition is summarized.*

***Key Words*: Composite, Stacking sequence, Ply-orientations, Aerospace composite, Finite element analysis, Flat panel, Carbon composite.**

1. INTRODUCTION

The use of composite materials is rapidly increasing in domains such as aerospace, automobile, military, marine etc. This paper is mainly concentrated on aerospace composites, according to Airbus the use of composite material has increased over the past twenty years. The maximum use of composite material is made in Airbus A350 XWB which has fifty three percent of composite and mainly used in parts such as wings, center wing box, keel beam, tail cone, skin panels, frames, stringers, doublers and the passenger as well as cargo doors [1]. Composites have the advantage of high strength to weight ratio and high stiffness to weight ratio [2], composites can give better performance when it is compared with metallic materials such as aluminum and steel but the maximum performance of a composite laminate can be achieved only when the laminas are oriented in a suitable way. Thickness of laminates, fiber orientation angles and stacking sequence are usually taken as common design variables in the design of the laminates. Single laminate consists of various number of laminas depending upon the required thickness, these laminas are usually stacked in 0^0, $\pm45^0$ and 90^0 to make manufacturing easy but in some complex design 0^0, $\pm30^0$, $\pm60^0$ and 90^0 are also considered. The stacking sequence optimization should be treated as a discrete optimization problem to which the conventional optimization methodologies are difficult to consider. The main agenda of this paper is to make a finite element analysis for composite skin panel of an aircraft with a suitable dimension and thickness where buckling of the panel with respect to global-local mode and Eigen values are observed for different number of stacking sequence and ply orientations. Composite panel buckling verification includes three main methods which are numerical, experimental and computational. In buckling of composite plate paper by A W Leissa, the author has gone through three segments to verify the composite plane which involved classical bifurcation buckling analysis, discusses the relevant plate equations and their solutions, and considers shapes, edge conditions and loadings which may arise [3]. Classical complicating effects including elastic foundation, variable thickness, shear deformation, hygrothermal effects and in-plane heterogeneity followed by post buckling, geometric imperfections, parametric excitation, follower forces and inelastic material. In the article design of composite laminated plates for maximum buckling load with stiffness and elastic modulus constraints by Kazemi M and Verchery G deals with the four different layers of ply orientation where the composite plate was modeled with the help of Finite Element Method [4].

The load cases chosen for this model is compression load, shear load and combined load where buckling of each panel is observed and, in the results, the main takeaway is plies which are not properly arranged gave extra thickness, which is, more number of plies and critical buckling loads were increased more than five times. Since there are many ways to analyze the composite panel, in this paper, finite element analysis method is followed, which is also known as computational method where initially suitable cross section of skin panel of an aircraft is considered for which compressive load is calculated by keeping metallic panel as a reference. With the help of software, a finite element model is created and loads are applied in the compressive direction on the shorter edge of the panel for eight different stacking sequence and ply orientation where composite panel thickness is kept constant for all the eight stacks. Buckling factor, buckling mode shapes (global-local), Elastic modulus E_{11}, Elastic modulus E_{22}, Poisson's ratio Nu_{11} and shear modulus G_{12} as well as ABD matrix are observed and tabulated for each stacking sequence. The composite material used for the finite element analysis is carbon fiber composite T-300 (CFC T-300). The main purpose of observing ABD stiffness matrix is to check what kind of effect can be seen on extensional-shear stiffness, extensional-bending stiffness and bending-torsional stiffness for each and every stacking sequence with different ply orientations. In analogy with classical laminate theory we write the 6 × 6 matrix relating the two sets of variables as an ABD stiffness matrix [5],

$$\begin{bmatrix} N_x \\ N_y \\ N_{xy} \\ M_x \\ M_y \\ M_{xy} \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} & B_{11} & B_{12} & B_{16} \\ A_{12} & A_{22} & A_{26} & B_{12} & B_{22} & B_{26} \\ A_{16} & A_{26} & A_{66} & B_{16} & B_{26} & B_{66} \\ B_{11} & B_{12} & B_{16} & D_{11} & D_{12} & D_{16} \\ B_{12} & B_{22} & B_{26} & D_{12} & D_{22} & D_{26} \\ B_{16} & B_{26} & B_{66} & D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \epsilon_{xy} \\ k_x \\ k_y \\ k_{xy} \end{bmatrix}$$

In the above matrix loads are represented by N and moment by M where ϵ and k are mid plane strain and shear strain respectively. Aij, Bij and Dij are extensional-shear stiffness, extensional-bending stiffness and bending-torsional stiffness respectively. According to the classic laminate theory A_{16}, A_{26}, and Bij [B_{11}-B_{66}] should be zero if not then the laminate is not balanced.

2. METHODOLOGY

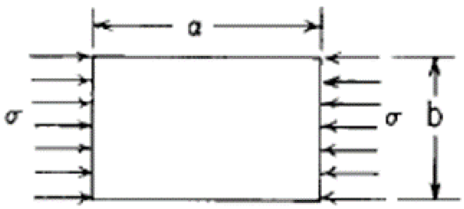

Fig-1: Compressive load on skin panel on shorter edges 'b'

Initially skin panel of a vertical tail with dimensions of 500mm in 'a' direction and 300mm in 'b' direction as shown in the Fig-1, with a panel thickness of 3mm is considered. To calculate reference value of load, initially, panel is considered as metallic material (Al2024) and critical unit compressive stress is calculated by using Roark's formula [6] which is as shown below.

$$\sigma' = K \frac{E}{(1-\upsilon^2)}(t/b)^2 \tag{1}$$

Where,
σ' is critical compressive stress = 25.72
K can be found from the $\frac{a}{b}$ ratio = 1.66
E is modulus of elasticity = 70000 N/mm^2
t is the thickness of the plate = 3mm
υ is the Poisson's ratio = 0.3
a is the horizontal dimension of the plate = 500mm
b is the vertical dimension of the plate = 300mm

From the calculation, critical unit compressive stress obtained for the metallic panel is 26.5 N/mm^2 and total load can be calculated by multiplying critical stress, thickness and length of the 'b' which is 23850N. This load is distributed in the compressive direction on the edges of 'b'. Using software skin panel is created as per the dimension and panel is meshed with a quad mesh with an element size of ten. The loads are distributed as shown in the Fig-2.

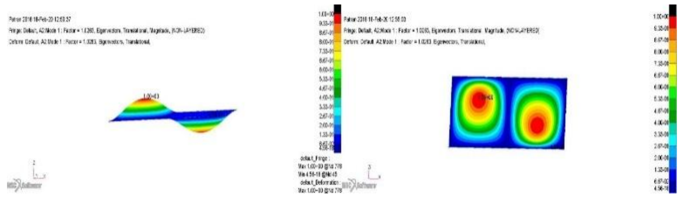

Fig-2: Application region of the loads and displacement on the metallic panel.

In Fig.2, load and displacement distribution is shown on the shorter edge of the metallic skin panel in a compressive direction, when the buckling analysis is carried out buckling mode and buckling factors are observed which is as shown in the Fig.3.

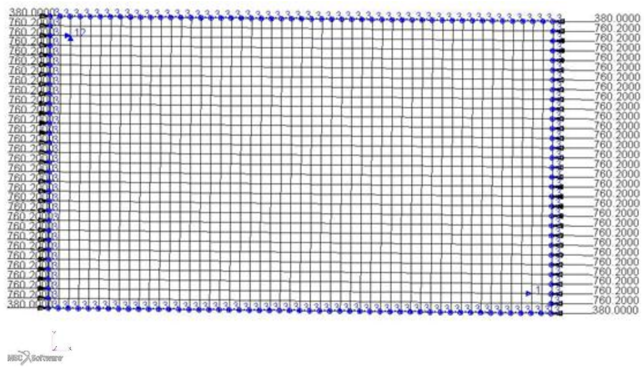

Fig-3: Analysis result for rectangular metallic skin panel.

In the Fig-3, Eigenvector for the rectangular aluminum skin panel is 1.026 and a very clear local modes can be seen, which indicates that the panel is stiffer for the applied load. The total mass of the metallic panel is 1215gm, the load acting on the metallic panel is considered as a reference load for the composite laminate and buckling analysis is carried out for different stacking sequence and ply orientation.

Laminate is made up of different number of lamina or the plies, in this paper unidirectional tape CFC-T300 [7] lamina is used where its elastic modulus in the longitudinal direction is 130000N/mm^2 and it is denoted as E_{11}, elastic modulus in the transvers direction is 10000N/mm^2 and it is denoted by E_{22}, the Poisson's ratio is 0.35 and it is denoted by Nu_{11} and in-plane shear modulus is 5000N/mm^2 where it is denoted by G_{12} all the values which are mentioned belongs to single composite ply. When the different number of plies are placed in a suitable direction then best performance of a laminate can be observed. The thickness of the laminate depends on the use of plies where each ply has a thickness of 0.15mm with a density of

0.001754g/mm^3. In this paper nine different stacking sequence is followed and they are in symmetric condition. To fix the thickness of the composite laminate of dimension 500mm in 'a' direction 300mm in 'b' direction initially analysis is carried out for 3mm thickness with a twenty plies which are oriented in 0^0, $\pm45^0$ and 90^0 but eigenvector is not above one by which it can be consider as panel is not stiffened enough to take the applied load and even the local buckling modes where absent because the thickness of the laminate is too less for the applied load. Composites main advantage is less mass and higher efficient for the applied load but its main disadvantage is it has the higher volume, keeping this in mind different iterations is carried out to find the suitable thickness for the applied load and finally 4.2mm gives the good eigenvector value as well as local buckling mode. By keeping 4.2mm as a final thickness for the composite panel analysis is carried out for different stacking sequence which is as mentioned further.

Table -1: Stacking sequences and ply orientations with their obtained properties

Stacking Number	1	2	3	4	5	6	7	8	9
Number of plies	28	28	28	28	28	28	28	28	28
Ply thickness. (mm)	0.15	0.15	0.15	0.15	0.15	0.15	0.15	0.15	0.15
Laminate thickness. (mm)	4.2	4.2	4.2	4.2	4.2	4.2	4.2	4.2	4.2
Stacking sequence with different ply orientation in 0^0. (Symmetric)	0	45	45	45	45	45	45	45	45
	90	-45	-45	-45	-45	-45	-45	-45	-45
	0	0	90	45	90	0	0	0	0
	90	0	45	-45	45	45	45	45	45
	0	45	90	45	90	90	45	90	0
	90	90	-45	-45	-45	-45	90	-45	0
	0	-45	90	45	90	0	45	-45	-45
	90	0	45	-45	45	0	0	0	90
	0	0	90	45	0	45	-45	45	45
	90	45	-45	-45	0	90	90	0	90
	0	0	0	45	-45	-45	45	-45	-45
	90	-45	45	-45	90	90	0	90	0
	0	0	90	45	45	0	45	45	0
	90	0	-45	-45	0	45	45	0	45
Number of 0^0	14	14	2	0	6	8	6	8	10

Number of $+45^0$	0	6	8	14	8	8	14	8	8
Number of -45^0	0	6	8	14	6	6	4	8	6
Number of 90^0	14	2	10	0	8	6	4	4	4
E_{11}(N/mm²)	70489	70500	31533	17624	46827	54611	46633	54400	62048
E_{22}(N/mm²)	70489	35900	61002	17624	54611	46827	39507	39400	38826
Nu_{11}	0.499	0.409	0.277	0.762	0.293	0.342	0.456	0.435	0.4104
G_{12}(N/mm²)	4999	17500	21322	33566	19283	19283	23364	21300	19283

In the Table-1, representation of the nine different stacking sequence is made where each stack consists of 14 plies with symmetric configuration (with a total of 28 plies in each laminate) which makes single laminate thickness of 4.2mm. E_{11}, E_{22}, Nu_{11} and G_{12} are also represented for each stacking sequence which helps to decide what load is predominant in the laminate. In the first stack only 90^0 and 0^0 are used equally without plies of $\pm45^0$ due to which we can see the greater value in E_{11} and E_{22} of 70489N/mm² with a maximum Nu11 of 0.499 but least G_{12} which is 4999N/mm². Since the elastic modulus are observed in the longitudinal and transvers direction, E_{11} and E_{22} respectively are showing higher in a laminate where as in-plane shear modulus is very less. This type of laminate is not suitable for aircraft industry though it is good in bending stress but not in shear stress since in an aircraft shear stress plays an important role. In the second stacking sequence $\pm45^0$ plies are introduced and number of 90^0 plies are reduced gradually by keeping 50% of 0^0 plies, 6% of $\pm45^0$ each and 7% of 90^0 plies. In this stacking sequence after the introduction of the $\pm45^0$ plies it can be seen that in-plane shear modulus increased up to 17500N/mm² (G_{12}), but in this stacking sequence the laminate is beginning with 00 plies which is not recommended because 0^0 plies can take highest bending stress and if they are placed at the beginning of the stack then there is a chance of lamina damage. In the third stacking sequence 7% of 0^0 plies, 29% of ±450 each and 36% of 90^0 plies are stacked, where in this laminate 90^0 ply is predominant due to which elastic modulus in transvers (E_{22}) is more than the longitudinal direction (E_{11}) but due to increase of $\pm45^0$ ply in the laminate it can be observed that in-plane shear modulus is increased compared to stacking sequence number two. The main observation can be made in the third stacking sequence is the 0.277 Poisson's ratio (Nu_{11}), low Poisson's ratio indicates that the laminate is easy to fracture, so third stacking sequence is not suitable. In the fourth stacking sequence 0^0 plies and 90^0 plies are eliminated and whole laminate is created by using only $\pm45^0$ and the observation made is G_{12} is maximum of 33566N/mm² with a highest Poisson's ratio of 0.762 but E_{11} and E_{22} are most least when they are compares to previous stacking sequence, since it doesn't have a good E_{11} and E_{22} value the chosen stacking sequence is not suitable. In the fifth stacking sequence 21% of 0^0 plies as well as -45^0 plies are stacked with 29% of 90^0 plies as well as 45^0 plies is used, in this laminate E_{11} and E_{22} have 46827N/mm² and 54611N/mm² respectively, though the E_{11} and E_{22} have the better value the panel is not so much stronger and it can be determined by Nu_{11} value and even the in-plane shear modulus value is less when it is compare to other stacking sequence. In the fifth stacking sequence number of 90^0 plies are more than the 0^0 plies due to which E_{22} is greater than the E_{11} which means the laminate can receiving more load in the transvers direction than the longitudinal direction which is not suitable for the compressive loading condition and even the value of the Nu_{11} is lower than the previous stacking sequence, by this it is clear that less number of 90^0 plies should be used in a laminate when the loading condition is compression. In the sixth stacking sequence 29% of 0^0 plies as well as 45^0 plies are stacked with a 21% of -45^0 plies as well as 90^0 plies due to which it can be observed that in E_{11} is greater than the E_{22} and even the Poisson's ratio as well as G_{12} has a better value but the percentage of 90^0 plies can be even reduced so that more number of 0^0 plies can accommodated and get a better performance from the laminate. In the seventh stacking sequence more number of 45^0 plies(50%) are used due to which it can be observed that Poisson's ratio can be seen with a higher value up to 0.456 and the in-plane shear modulus of 23364N/mm² but the value of E_{11} and E_{22} are not satisficing for the chosen number of plies, even better performance can be observed by re-arranging the stacking sequence with a more number of 0^0 plies. In the eight-stacking sequence eight plies of each 0^0 and $\pm45^0$ is stacked with the four 90^0 plies which gave an E_{11} and E_{22} value of 54400N/mm² and 39400N/mm² respectively with a Nu11 value of 0.435 and in-plane shear modulus of 21300N/mm².

This type of stacking sequence will make the laminate to perform better for the compressive loading condition with very good E_{11}, E_{22}, Nu_{11} and G_{12} values, this type of stacking sequence can be considered as best for the manufacturing. If it is necessary to achieve maximum bending stress then it is also necessary to increase the 0^0 plies in the laminate. Stacking number nine consist of 36% of 0^0 plies with a 29% of 450 plies and 21% of -450 plies as well as 14% of 90^0 plies due to which we can see a better vale in the E_{11} which consist of 62048N/mm^2 with a Poisson's ratio of 0.4104 and an in-plane shear of 19283N/mm^2. All the values obtained in the stacking sequence is only for the composite material CFC-T300 unidirectional tape for the panel dimension 500mm cross 300mm, but these values can vary for different tapes as well as different thickness and dimension of the panel.

Table-2: ABD matrix for nine different stacking sequences.

ABD matrix for stacking sequence 1	2.97E+05	1.48E+04	-3.16E-04	-2.89E-01	-1.21E-02	0.00E+00
	1.48E+04	2.97E+05	-1.08E-02	-1.21E-02	-2.79E-01	0.00E+00
	-3.16E-04	-1.08E-02	2.10E+04	0.00E+00	0.00E+00	-2.25E-02
	-2.89E-01	-1.21E-02	0.00E+00	4.76E+05	2.18E+04	-4.14E-04
	-1.21E-02	-2.79E-01	0.00E+00	2.18E+04	3.96E+05	-1.42E-02
	0.00E+00	0.00E+00	-2.25E-02	-4.14E-04	-1.42E-02	3.09E+04
ABD matrix for stacking sequence 2	3.54E+05	6.63E+04	0.00E+00	-3.12E-01	-5.47E-02	-2.93E-03
	6.63E+04	1.36E+05	-1.95E-03	-5.47E-02	-1.08E-01	-2.93E-03
	0.00E+00	-1.95E-03	7.24E+04	-2.93E-03	-2.93E-03	-5.37E-02
	-3.12E-01	-5.47E-02	-2.93E-03	4.43E+05	1.21E+05	1.51E+04
	-5.47E-02	-1.08E-01	-2.93E-03	1.21E+05	2.32E+05	1.51E+04
	-2.93E-03	-2.93E-03	-5.37E-02	1.51E+04	1.51E+04	1.30E+05
ABD matrix for stacking sequence 3	1.56E+05	8.34E+04	0.00E+00	-1.40E-01	-6.45E-02	-2.93E-03
	8.34E+04	3.01E+05	-9.77E-03	-6.45E-02	-2.69E-01	-1.95E-03
	0.00E+00	-9.77E-03	8.96E+04	-2.93E-03	-1.95E-03	-6.93E-02
	-1.40E-01	-6.45E-02	-2.93E-03	2.00E+05	1.36E+05	1.88E+04
	-6.45E-02	-2.69E-01	-1.95E-03	1.36E+05	4.45E+05	1.88E+04
	-2.93E-03	-1.95E-03	-6.93E-02	1.88E+04	1.88E+04	1.45E+05
ABD matrix for stacking sequence 4	1.77E+05	1.35E+05	0.00E+00	-1.59E-01	-1.11E-01	-3.91E-03
	1.35E+05	1.77E+05	0.00E+00	-1.11E-01	-1.59E-01	-3.91E-03
	0.00E+00	0.00E+00	1.41E+05	-3.91E-03	-3.91E-03	-1.20E-01
	-1.59E-01	-1.11E-01	-3.91E-03	2.60E+05	1.98E+05	2.00E+04
	-1.11E-01	-1.59E-01	-3.91E-03	1.98E+05	2.60E+05	2.00E+04
	-3.91E-03	-3.91E-03	-1.20E-01	2.00E+04	2.00E+04	2.07E+05
ABD matrix for stacking sequence 5	2.19E+05	7.48E+04	9.09E+03	-1.87E-01	-6.05E-02	-1.17E-02
	7.48E+04	2.55E+05	9.09E+03	-6.05E-02	-1.90E-01	-1.07E-02
	9.09E+03	9.09E+03	8.10E+04	-1.17E-02	-1.07E-02	-6.54E-02

	-1.87E-01	-6.05E-02	-1.17E-02	2.29E+05	1.33E+05	1.97E+04
	-6.05E-02	-1.90E-01	-1.07E-02	1.33E+05	4.21E+05	1.97E+04
	-1.17E-02	-1.07E-02	-6.54E-02	1.97E+04	1.97E+04	1.42E+05
ABD matrix for stacking sequence 6	2.55E+05	7.48E+04	9.09E+03	-2.37E-01	-6.45E-02	-1.07E-02
	7.48E+04	2.19E+05	9.09E+03	-6.45E-02	-1.83E-01	-1.07E-02
	9.09E+03	9.09E+03	8.10E+04	-1.07E-02	-1.07E-02	-6.54E-02
	-2.37E-01	-6.45E-02	-1.07E-02	3.75E+05	1.31E+05	1.68E+04
	-6.45E-02	-1.83E-01	-1.07E-02	1.31E+05	2.80E+05	1.68E+04
	-1.07E-02	-1.07E-02	-6.54E-02	1.68E+04	1.68E+04	1.40E+05
ABD matrix for stacking sequence 7	2.38E+05	9.20E+04	4.54E+04	-1.90E-01	-8.01E-02	-4.20E-02
	9.20E+04	2.01E+05	4.54E+04	-8.01E-02	-1.59E-01	-4.39E-02
	4.54E+04	4.54E+04	9.81E+04	-4.20E-02	-4.39E-02	-8.11E-02
	-1.90E-01	-8.01E-02	-4.20E-02	3.49E+05	1.45E+05	5.47E+04
	-8.01E-02	-1.59E-01	-4.39E-02	1.45E+05	2.77E+05	5.47E+04
	-4.20E-02	-4.39E-02	-8.11E-02	5.47E+04	5.47E+04	1.54E+05
ABD matrix for stacking sequence 8	2.65E+05	8.34E+04	0.00E+00	-2.22E-01	-7.23E-02	-1.95E-03
	8.34E+04	1.92E+05	-3.91E-03	-7.23E-02	-1.59E-01	-1.95E-03
	0.00E+00	-3.91E-03	8.96E+04	-1.95E-03	-1.95E-03	-7.71E-02
	-2.22E-01	-7.23E-02	-1.95E-03	3.57E+05	1.42E+05	5.72E+03
	-7.23E-02	-1.59E-01	-1.95E-03	1.42E+05	2.76E+05	5.72E+03
	-1.95E-03	-1.95E-03	-7.71E-02	5.72E+03	5.72E+03	1.51E+05
ABD matrix for stacking sequence 9	2.91E+05	7.48E+04	9.09E+03	-2.53E-01	-6.45E-02	-9.77E-03
	7.48E+04	1.82E+05	9.09E+03	-6.45E-02	-1.51E-01	-8.79E-03
	9.09E+03	9.09E+03	8.10E+04	-9.77E-03	-8.79E-03	-6.15E-02
	-2.53E-01	-6.45E-02	-9.77E-03	4.38E+05	1.27E+05	2.17E+04
	-6.45E-02	-1.51E-01	-8.79E-03	1.27E+05	2.25E+05	2.17E+04
	-9.77E-03	-8.79E-03	-6.15E-02	2.17E+04	2.17E+04	1.36E+05

ABD matrix is used to observe wither the laminate is balanced or not and it can be confirmed by looking at the values of each ABD matrix of a laminate. According to the classic laminate theory A_{16}, A_{26}, and B_{ij} should be zero if not then the laminate is not considered as balanced, from the Table-2, it can be observed that ABD matrix for the first stacking sequence is balanced since the values in the B_{ij} is completely zero but in the values in the Aij position represents the extensional shear stiffness and B_{ij} values represents the extensional bending stiffness where as in the D_{ij} position the values represents the bending and torsional stiffness. When each lamina is changed with its orientation and position then the values in A_{ij}, B_{ij} and D_{ij} will also change and if the Bij values is not zero then it is said to be laminate is not balanced and if the values in of A_{16} and

A_{26} are not zero then the laminate is not balanced and not symmetric.

3. RESULTS AND DISCUSSIONS

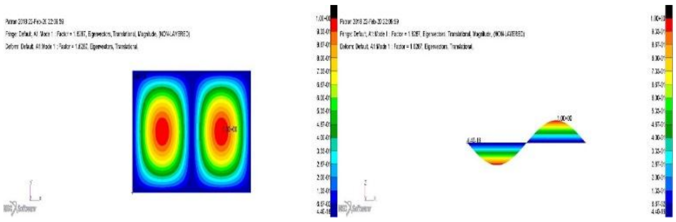

Fig-4: Buckling analysis result for the first stacking sequence.

As it can be seen in the Fig-4, the eigenvector is 1.628 which is for the first stacking sequence where only 0^0 and 90^0 plies are considered. Local buckling modes can be observed which means the panel is stiffer, though the panel is having local buckling modes it is not suitable to consider due to its poor in-plane shear values which can be observed from the Table-1.

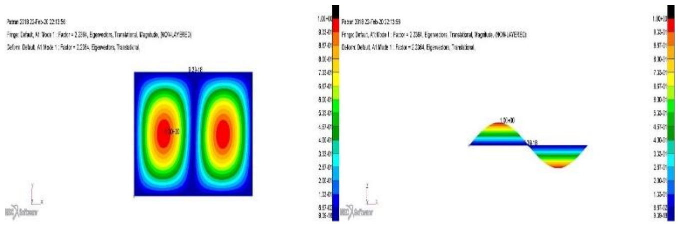

Fig-5: Buckling analysis result for the second stacking sequence.

In the Fig-5, representation of the analysis result is made for the second stacking sequence, the eigenvector of the second stacking sequence laminate is 2.236 which is much better than the first laminate and even the local buckling modes are visible which means that the panel is stiffer for the applied load but the values obtained for the second stacking sequence is not satisfying in the in-plane shear modulus.

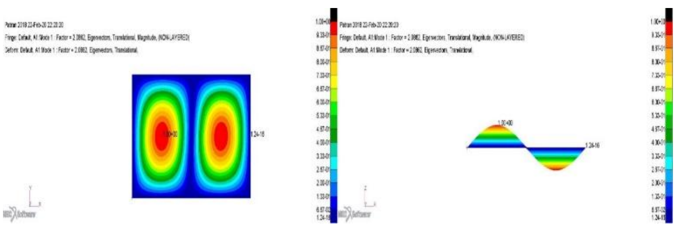

Fig-6: Buckling analysis for the third stacking sequence.

For the third stacking sequence the eigenvector is 2.071 which is represented in the Fig-6. Eigenvector became less than the second stacking sequence in the third stacking sequence and even the Poisson's ratio is very low in this stacking sequence when it is compared with other stacking sequences so this type of stacking sequence is not considerable for the applied load.

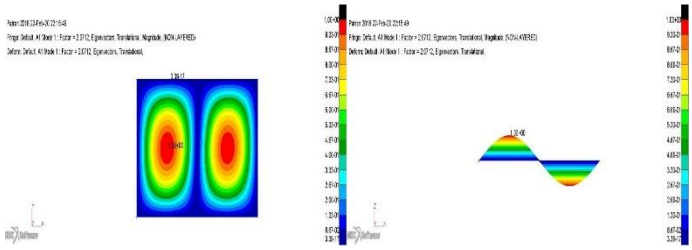

Fig-7: Buckling analysis result for the fourth stacking sequence.

In the Fig-7, representation of the analysis result for fourth stacking sequence is made where the eigenvector value is 2.542 and it has the local buckling mode shape. In this stacking sequence all the plies which are used are $+45^0$ and -45^0 plies due to which shear stress and Poisson's ratio are maximum than any other stacking sequence with a very low bending stress due to absence of 0^0 and 90^0 plies

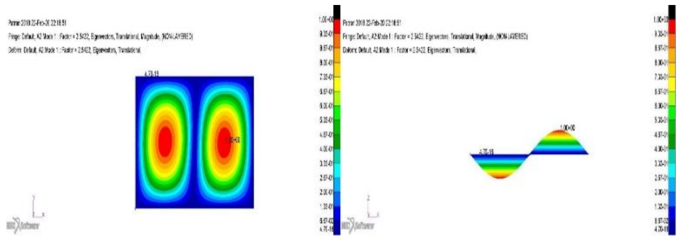

Fig-8: Buckling analysis for fifth stacking sequence.

In the Fig-8, buckling analysis result for fifth stacking sequence is shown in which the eigenvector is 2.086 with a clear local buckling mode. In this stacking sequence a greater number of 90^0 plies are used than the 0^0 plies due to which E_{22} is greater than the E_{11} which is not suitable for the compressive loading condition and even the Poisson's ratio is lower when it is compared with other stacking sequence (excluding stacking sequence number 3).

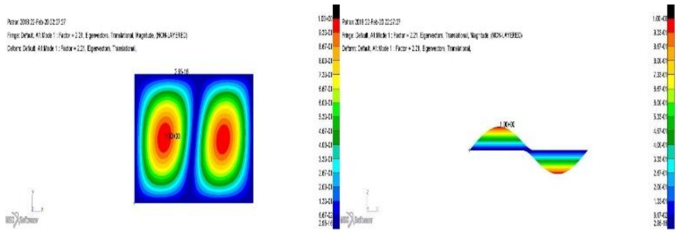

Fig-9: Buckling analysis result for sixth stacking sequence.

In the sixth stacking sequence 29% of 0^0 plies as well as 45^0 plies are stacked with a 21% of -45^0 plies as well as 90^0 plies. In the Fig-9, buckling analysis result for the stacking sequence number six is represented, eigenvector for this stacking sequence is 2.228 with a local buckling mode.

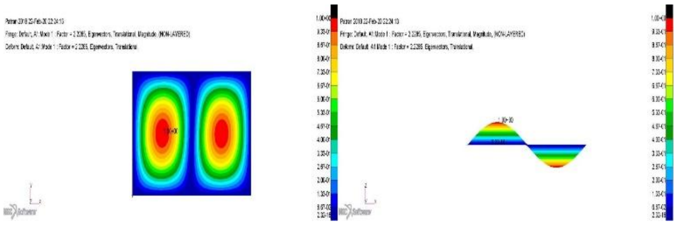

Fig-10: Buckling analysis result for the stacking sequence seven.

Fig-10, represents the buckling analysis result for the stacking sequence number seven with a eigenvector of 2.21, in the seventh stacking sequence a greater number of 45^0 plies (50%) are used due to which it can be observed that Poisson's ratio is with a higher value of 0.456 and change of buckling mode shape can be observed in this stacking sequence (tilted oval) which is due to a greater number of 45^0 plies.

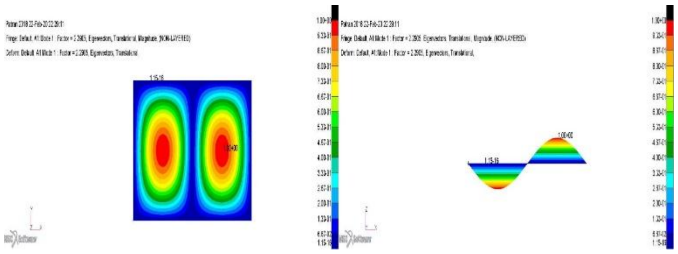

Fig-11: Buckling analysis result for the stacking sequence eight.

In the Fig-11, representation of the buckling analysis result for the eighth stacking sequence is made where the eigenvector is 2.290 with a local buckling mode. In the eight-stacking sequence eight plies of 0^0 and $\pm45^0$ each is stacked with the four 90^0 plies.

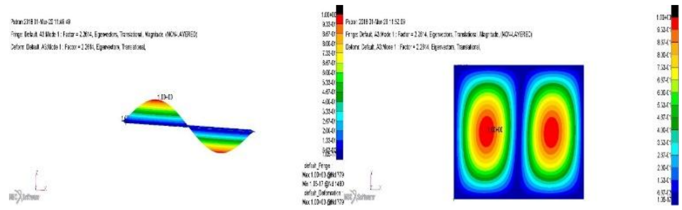

Fig-12: Buckling analysis result for the stacking sequence nine.

In the Fig- 12 buckling analysis result for the ninth stacking sequence is represented where the eigenvector is 2.261 with a very good local buckling mode. Stacking number nine consist of 36% of 0^0 plies with a 29% of 45^0 plies and 21% of -45^0 plies as well as 14% of 90^0 plies.

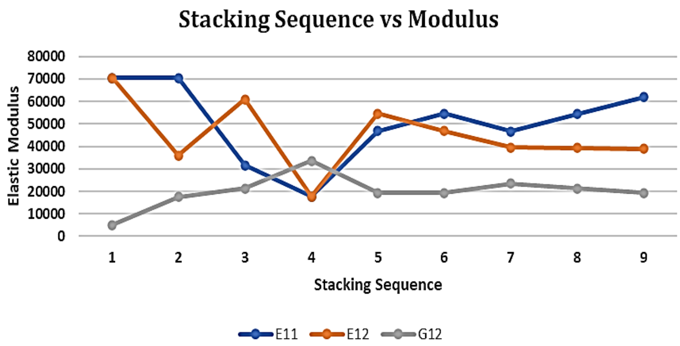

Chart-1: Stacking Sequence vs Elastic modulus.

In the Chart-1, representation of nine different stacking sequence with respect to their modulus is plotted. Blue line in the graph represents the elastic modulus in the longitudinal direction which is denoted by E_{11}. Orange line represents the elastic modulus in the transvers direction which is denoted by E_{22} whereas gray line represents the in-plane shear modulus which is denoted by G12. From the line graph it is clear that stacking sequence four is having higher shear modulus of 33566N/mm^2 whereas first stacking sequence has the highest elastic modulus of 70489N/mm^2 in E_{11} and E_{22} but both of the stacking sequences are not suitable, because in the first stacking sequence only 0^0 and 90^0 plies are used due to which shear modulus is very low which can be observed in the graph where as in the fourth stacking sequence as the plies which are used is 45^0 due to which elastic modulus E_{11} as well as E_{22} are very low which can be seen in the above graph. In the graph it can be observed that in the stacking sequence eight and nine has the best values in all the three moduli. In the stacking sequence number eight 0^0, $+45^0$ and -45^0 consist of 29% of each ply in the stack and 14% of 90^0 ply due to which it can be observed that in all the modulus E_{11}, E_{22} and G_{12} has the better value than any previous stacks, E_{11} has the value of 54400N/mm^2 whereas E_{22} has the value of 39400N/mm^2 and G_{12} has the value of 21300N/mm^2. In the ninth stacking sequence a greater number of 0^0 plies are used due to which E_{11} (62000N/mm^2) can be observed higher when it is compared with the E_{11} of eight stacking sequence but G_{12} (19283N/mm^2) and E_{22} (38826N/mm^2) has reduced in the ninth stacking sequence when it is compared with eight stacking sequence because of a smaller number of -45^0 plies. Though the number of 90^0 plies are same in eight and ninth stacking sequence shear has dropped down in the ninth stacking sequence, by this it can be said that each and every ply and its orientation as well as its position in the stack impacts the three modulus values as well as laminate Poisson's ratio.

4. CONCLUSION

This paper carried out panel buckling analysis for the composite rectangular panel with a nine different stacking sequence under uniaxial compression loading. All the nine stacking sequences have the panel dimension of 500mm by 300mm with a thickness of 4.2mm. From the analysis it is observed that if a laminate is having a greater number of zero-degree plies then the laminate can give better elastic modulus in the longitudinal direction which means that the laminate can take more load which is acting in the longitudinal direction. If the more number of ninety-degree plies are used then the elastic modulus is predominant in the transvers direction which is not suitable for the load which is acting in the longitudinal direction but from the analysis it can be said that if the load acting in the transvers is also higher then more number of ninety degree plies can be stacked in a laminate in the same way from the analysis it can be stated that if the laminate has to be prepared for the shear predominant loading condition then more number of forty-five degree plies have to be used since it is observed that forty-five degree plies can give better values for in-plane shear modulus. Finally it can be stated that if a laminate has to sustain grater amount of bending stress in the longitudinal direction for the compressive loading condition then the laminate has to consist of more number of zero-degree plies and if the laminate has to sustain grater amount of shear stress then laminate should consist more number of forty-five degree plies and always number of ninety degree plies can be kept in the minimal but it cannot be excluded, at least ten percent of the total plies should be dedicated to ninety degree ply so that it sustains the load which is acting in the transvers direction of the laminate when the laminate is under compression loading, finally it can be concluded that stacking sequence number eight and nine are suitable for the chosen loading condition. In the future this study can help in the comparative study of metallic panels with stringer to the composite panel with composite stringer and this analysis method can help for the further study of composite panel with unsymmetrical and unbalanced laminates.

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