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STUDY OF BUCKLING ANALYSIS OF COMPOSITE PLATE.

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Abstract: In this paper buckling analysis of carbon fiber laminated composite plate is performed by using classical laminated plate theory (CLPT) for simply supported plate. The critical buckling load of the composite laminated plate is evaluated by finite element method with help of ABAQUS software. The critical buckling load of plate is affected by the fiber orientation, aspect ratio, plate thickness, mesh size and boundary condition of the stacking sequence and FEA element. The composite laminated plate made of carbon fiber gives maximum critical buckling load at any boundary condition. Some results have been obtained for different parameter and are discussed in detail.

***Keywords*: Buckling, Carbon fiber laminated composite, Critical buckling load, ABAQUS.**

I. INTRODUCTION

Composite plates are used in the infrastructure like aviation, military and waterways etc. because of the high specific stiffness and strength. Therefore, they are very suitable for weight sensitive structures. The composite materials are more thermally stable and more corrosion resistant than conventional metals used for the structures. They are used in the form of rectangular plate. The composite plates are subjected to the various compressive stresses and in plane loads.

Because of their excellent mechanical characteristics, fiber-reinforced composites are becoming increasingly popular. Designing buildings constructed of composite material is a difficult process since both thicknesses, relative orientation of the plies and the number of plies in the laminate must be chosen. The optimum capabilities of the material may be obtained by carefully selecting the composite layup utilized in the composite material. The use of composite laminated plates in aircraft structures, marine structures present design and analysis tasks which require accurate and efficient assessment of the stress and loads in critical regions.

In addition, the ratio of the parent material's composition affects the characteristics of a composite. As a result, the characteristics of the composite may be altered by altering its composition. The two components of a composite are known as the matrix and fiber. The fibers are the primary load carrying components, while the matrix, which has a high elongation and a low modulus, offers the required flexibility while also holding the fibers in place and protecting them from the environment.

Buckling is extremely hazardous to structural components since it generally occurs at a lower applied load and causes significant deformation. This resulted in an emphasis on the investigation of buckling behavior in composite materials. For different orientation the composite plate gives different critical buckling loads.

Generally buckling in real case occurs due to pressure difference at the wing in aircraft wing which tends to buckle, car bonnet buckles due to wind force etc. in this paper, the laminated composite plate is made up of carbon epoxy material with orientation of $[0/90]_2$ and boundary condition taken as simply supported used to evaluate critical buckling load and load is checked with analytical and numerical results.

II. EFFECT OF FIBER ORIENTATION ON BUCKLING

The results of the critical buckling load for different material and different orientation are calculated using ABAQUS software for simply supported condition as shown in following table [5],

Table 2.1 Buckling load comparison for different materials for the SS boundary condition

Material	Properties of material	Buckling load for different orientation
E glass Epoxy	E_1=41130Mpa, E_1=7143.2Mpa, μ_{12}=0.24, G_{12}=2759.6Mpa	$(0)_{10}$=2818.5N $(0/45)_5$=3567N $(0/90)_5$=7546.5N
Aramid Epoxy	E_1=73030Mpa, E_1=7321.5Mpa, μ_{12}=0.347, G_{12}=2564.9Mpa	$(0)_{10}$=2889N $(0/45)_5$=4122N $(0/90)_5$=12012N
Carbon Epoxy	E_1=128030Mpa, E_1=7421.4Mpa, μ_{12}=0.3, G_{12}=2816.6Mpa	$(0)_{10}$=2925N $(0/45)_5$=4891.5N $(0/90)_5$=19498.5N

The orientation for plate used is [0/90] which is cross-ply plate sequence, for this symmetrical plate is preferred to get the maximum buckling load. It is found that for carbon epoxy material with [0/90] orientation which is gives maximum critical buckling load.

The orientation for the symmetrical plate is shown in following fig 2.1

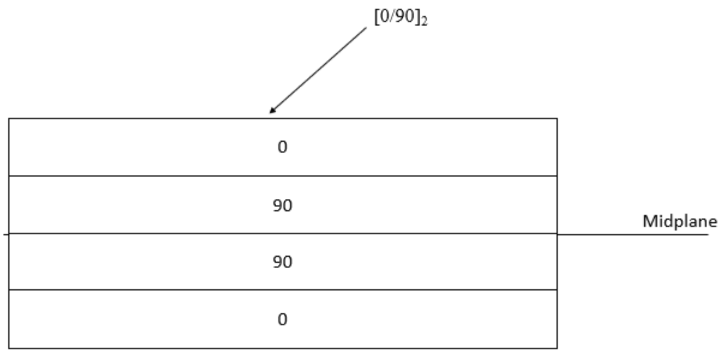

Fig. 2.1 Angular orientation of laminated composite plate

III. ANALYTICAL FORMULATION TO FIND CRITICAL BUCKLING LOAD

3.1 Micromechanics approach

The analytical solutions for the buckling analysis of the laminated composite plate can formulated by using CLPT theory. For CLPT we consider micromechanics approach. By considering micromechanics approach we get following results.

The matrix fiber volume fraction

$$v_m = \frac{A_m}{A_c} = \frac{t_m}{t_c} = 1 - v_f \qquad \text{------------------(1)}$$

Longitudinal Young Modulus

$$E_1 = E_f v_f + E_f v_f \qquad \text{----------------(2)}$$

Transverse Young Modulus

$$\frac{1}{E_2} = \frac{v_f}{E_f} + \frac{v_m}{E_m} \qquad \text{------------------(3)}$$

Major poison ratio

$$\mu_{12} = \mu_f \times v_f + \mu_m \times v_c \qquad \text{------------------(4)}$$

In-plane shear modulus

$$\frac{1}{G_{12}} = \frac{v_f}{G_f} + \frac{v_m}{G_m} \qquad \text{------------------(5)}$$

From the above formulas the material properties for the carbon epoxy material having volume fraction having fiber 55% and epoxy 45% are calculated in following table,

Table 3.1 Mechanical properties of laminated composite plate

Properties	**Carbon epoxy**
Orientation of lamina	0/90
Longitudinal Young Modulus E_1 Gpa	128.03
Transverse Young Modulus $E_2 = E_3$ Gpa	7.421
Modulus of Rigidity $G_{13} = G_{12}$ Gpa	2.816
Modulus of Rigidity G_{23} Gpa	2.816
Poison ratio $\mu_{12} = \mu_{13}$	0.3
Density gm/cm^3	1.44
Volume fraction	55%
Thickness of plate mm	5.2
Thickness of each lamina mm	1.3

Geometry of rectangular laminated plates in simply supported condition is shown below where dimensions are L_x = 300 mm, L_Y = 150 mm, h = 5.2 mm

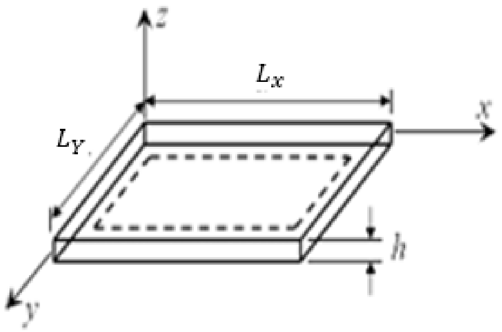

Fig. 3.1 Geometry of rectangular laminated plates in simply supported condition

3.2 Classical lamination plate theory (CLPT) [7]

When the in-plane compressive stress becomes so great that the previously flat equilibrium condition is no longer stable, the plate buckles and deflects into a non-flat (wavy) form. The buckling load is the load that causes the departure from the flat condition. The flat equilibrium state is characterized by solely in-plane forces and is subject to just extension, compression, and shear.

Displacement

Classical Laminate Plate Theory is based on the displacement field:

$$u(x,y,z,t) = u_0(x,y,t) - z\frac{\partial w_0}{\partial x} \qquad \text{------------------(6)}$$

$$v(x,y,z,t) = v_0(x,y,t) - z\frac{\partial w_0}{\partial y} \qquad \text{------------------(7)}$$

$w(x, y, z, t) = w_0(x, y, t)$ ------------------(8)

Where u_0, v_0, $w_0$are the displacement along (x, y, z) coordinates direction respectively of a point on the mid plane (z=0).

3.3 Solution approach:

Consider a rectangular plate with L_x and L_Y dimensions that is simply supported along its four edges. The design is symmetrical. [B] = [0]

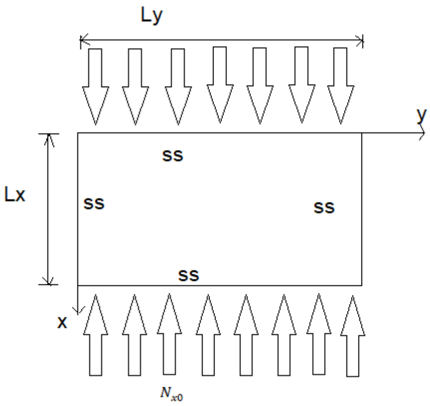

Fig. 3.2 Simply supported rectangular composite plate

Around the edges, the plate is subjected to uniformly distributed in-plane N_{x0}, N_{y0}, and N_{xy0}. These loads are proportionally increased, that is, the loads are λN_{x0}, λN_{y0}, and λN_{xy0} where λ is the load parameter. The load parameter for a buckled plate is denoted by λ_{cr}.

The potential energy can represent by following expression by setting N_{xy} and D_{16} and D_{26} are zero (orthotropy).

$$\pi_p = U + \Omega$$

$$\pi_p = \frac{1}{2}\int_0^{lx}\int_0^{ly}\left[D_{11}\left(\frac{\partial^2 w_0}{\partial x^2}\right)^2 + D_{22}\left(\frac{\partial^2 w_0}{\partial y^2}\right) + D_{66}\left(\frac{\partial^2 w_0}{\partial x \partial y}\right)^2 + 2D_{12}\frac{\partial^2 w_0}{\partial x^2}\frac{\partial^2 w_0}{\partial y^2} + N_x\left(\frac{\partial w_0}{\partial y}\right)^2 + N_y\left(\frac{\partial w_0}{\partial y}\right)^2\right]dydx \quad \text{-----------(9)}$$

We consider an orthotropic plate subjected to N_{x0} and N_{y0} edge loads. The forces inside the plate are

N_x = -λN_x N_y=-λN_{y0} N_{xy}=0

The eigenvalues can be calculated directly. The result is

$$\lambda_{cr} = \frac{\pi^2[D_{11}\left(\frac{i}{Lx}\right)^4 + 2(D_{12}+2D_{66})\left(\frac{i}{Lx}\right)^2\left(\frac{i}{Lx}\right)^2 + D_{22}\left(\frac{i}{Ly}\right)^4]}{N_{x0}\left(\frac{i}{Lx}\right)^2 + N_{y0}\left(\frac{i}{Ly}\right)^2} \quad \text{-----------(10)}$$

By putting the material properties in equation (9) The values obtained from CLPT for chosen laminated plate are as follows,

$$\lambda_{cr} = \frac{\pi^2[1.32\left(\frac{1}{0.3}\right)^4 + 2(0.0261+2*0.032)\left(\frac{1}{0.3}\right)^2\left(\frac{1}{0.3}\right)^2 + D_{22}\left(\frac{1}{0.15}\right)^4]}{\left(\frac{1}{0.3}\right)^2 + \left(\frac{1}{0.15}\right)^2}.$$

$\lambda_{cr} = 606.006\ N/m$

Result calculated analytically = 606.06 N/m

3.3 Numerical analysis

1. ABAQUS software is used for the numerical analysis of the laminated composite plate.
2. While creating part 3D space with deformable type and solid shape is used.
3. Properties given to the material and assembly is created and steps are applied to the part.
4. Mesh of linear quadrilateral elements of type S4R is used for analysis.
5. The simply supported boundary condition and loads are applied to the plate.

Fig 3.3 shows the applied boundary condition and loads to the plate.

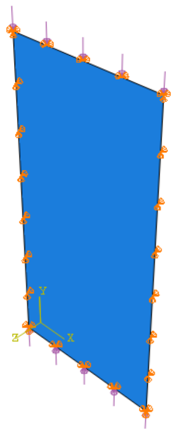

Fig 3.3 Parts with assigned boundary condition

Fig 3.4 shows different mode shapes of the plate,

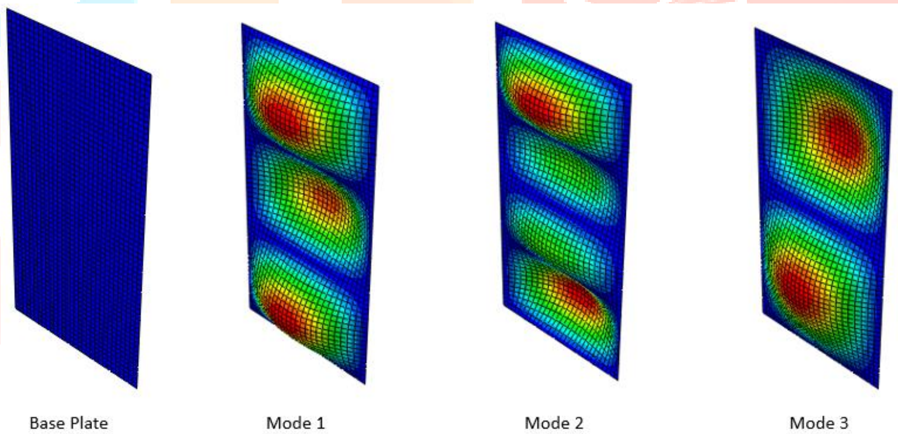

Fig. 3.4 Different mode shapes of laminated composite plate

3.4 Results and Discussion

1. The material fails when the applied load on the material is greater than the critical buckling load and its safe bellow the critical buckling load.
2. The analytical value of the critical buckling load calculated is 606.06 N/m
3. The numerical value of the critical buckling load calculated using ABAQUS software is 631.49 N/m
4. The percentage error in the analytical and numerical result is 4 % which is due to mesh size and boundary condition used to calculate the critical buckling load in ABAQUS software.

IV. CONCLUSION

The results are obtained analytically with the help of classical laminated plate theory (CLPT). CLPT is basic theory and the results obtained by CLPT theory can be enhanced by using the first order shear deformation theory or higher order shear deformation theory. It was found that the results obtained from the ABAQUS software and the analytical results have 4 % error due to mesh size used. The accuracy can increase by increasing the mesh size. When θ was increased, the critical load was found to be increased. For the carbon epoxy material with [0/90] orientation, the critical buckling load were maximum.

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