Buckling of Composites Laminate

P. Neelima, B. Meghashyamraju, Thejesh. S & Monisha
Department of Aerospace Engineering, Srinivas Institute of Technology, Valachil, Mangalore, India

Abstract: **Since the early days of aviation the choice of material has been primary concern. This is so because of varied loading conditions. Added to this is the requirement of high strength to weight ratio. In the recent times use of composites which is stack of lamina has been very extensive as it caters to the above mentioned requisites. However, they are susceptible to compressive loading causing delamination. This greatly affects the Behaviour of the structure under compression. The present thesis makes an attempt to study the same by finding the buckling strength before and after delamination.**

Keywords: **Rectangular plate, composite materials, through width delamination, epoxy, buckling.**

I. INTRODUCTION

Composite materials have been widely used in for structural applications. Because of High strength to weight ratio, high tensile strength, high toughness, and flexibility in design, etc.

Some of the properties displayed by these materials that have benefited many industries such as aerospace, automotive and marine. For example, 45% of the Light Combat Aircraft [LCA] Tejas' body is made using composites.

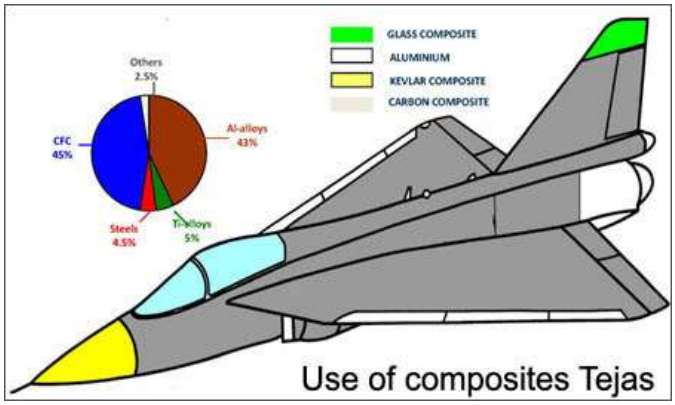


Figure 1.1 Use of composites in LCA airframe

1.1. Composites

Composites are among the most adaptable advanced engineering materials known till date. Composites in general are heterogeneous materials and consist of two or more constituents such as fiber and a matrix. In contrast to metallic alloys, each constituent material retains its separate chemical, physical, and mechanical properties.

The matrix of composites can be metal (Metal Matrix Composite), polymeric matrix (Epoxy), ceramic (Ceramic Matrix Composite) or carbon based (Carbon-Carbon or polymeric). These different matrix give composites to their shape, appearance, environmental tolerance and durability while the fibers carry most of the structural load and make these materials strong and stiff.

One of the biggest benefits of composites is that the properties are adjustable to design parameters. For example, the mechanical properties, content, orientation and fiber architecture, and the properties of the matrix are all materials design parameters.

Composite materials play a key role in industries like aerospace and automobile because of their outstanding strength to weight ratio and modulus to weight ratio figure 1.2.

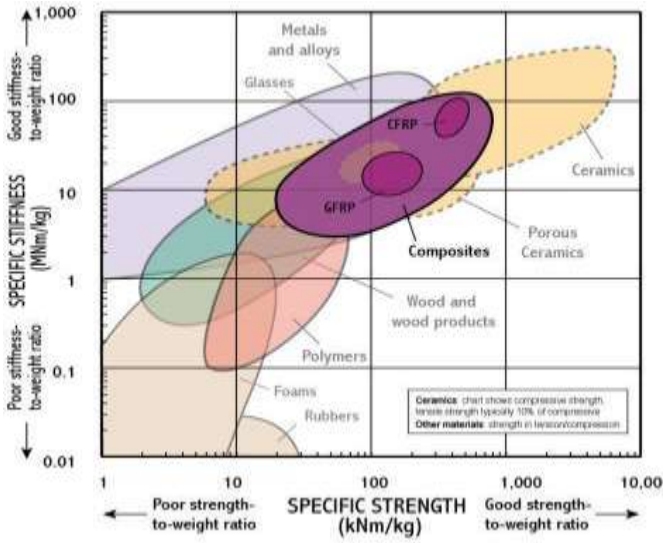


Figure1.2: Strength to weight ratio of composites (©Cambridge University Engineering Department, 2009)

1.2 Composite failure

Despite these benefits composite material will fail due to breaking of fibers, development of micro-cracks in matrix, Deboning between fibers and matrix. Delamination i.e separation of different layers of a laminate. The composite structures have a tendency to delaminate which reduces the strength and stiffness and thus limits the life of a structure.

Common damage of an aircraft can arise from accidental impact, bird strike, low velocity impact, hailstones and

lightning strike or from deterioration caused by the absorption of moisture or hydraulic fluid, because of the laminated layers in composite structures, damage often manifests as delamination between plies.

Delamination is one of the major failure modes and it may cause structural failure leading to catastrophic consequences.

Development of an early damage detection method for delamination is an important requirement for maintaining the integrity and safety of composite structures. Many detection techniques have been proposed for structural health monitoring (SHM) and some of the non-destructive evaluation approaches that utilize advanced technologies, such as X-ray imaging, ultrasonic scans, infrared thermograph, and eddy current, can identify damages.

Delamination due to interlaminar stresses can reduce the failure stress of the laminate that predicted by in-plane failure criteria. Failure by delamination is not necessarily the same as the initiation of delamination, however. The initiation of delamination generally followed by stable delamination growth, which eventually leads to unstable growth and ultimate failure.

The onset of it can be predicted by using either mechanics of material approach or fracture mechanics. In this thesis we discuss the use of fracture mechanics approaches, particularly those involving the use of the SERR, for the prediction of delamination growth and failure.

It is in effect of a separating adjacent laminate and the plane of the delamination lies in the plane of interface b/w laminate. Like a crack in metallic material, a delamination grows in stable manner until it reaches a critical size, where upon further growth occur in an unstable manner. These characteristics make interlaminar fracture a prime candidate for the application of fracture mechanics analysis, interlaminar stresses are part of complex 3D state of stress that leads to delamination.

While such a complex state of stress at the crack tip inhibits the effective use of the stress intensity factor approach, it makes the problem ideally suited for the SERR approach.

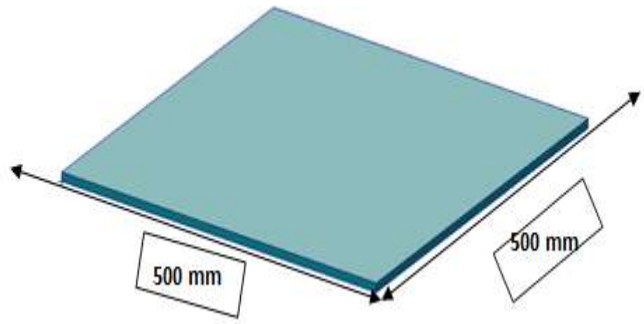
II. RELATED WORKS

2.1 Model description

Consider a model 500*500*3 mm shown in Figure, composite laminate panel of 16 ply 0 degree orientation symmetrically $(0/0/0/0/0/0/0/0)_s$, each ply of thickness 0.1875 mm Composite material is Boron-Epoxy. Boron fibers are characterized by their very high tensile modulus, the range of which is 379-414 Gpa.

Boron fibers have relatively large diameters and due to this they are capable of withstanding large compressive stress and providing excellent resistance to buckling.

Epoxy resins are mostly used in aerospace structures for their high performance application and are extensively used in industries due to ease with which it can be proceeded.



There are two type of delamination namely embedded delamination and Through- width-delamination as shown in figure4, in this thesis considering through-width-delamination, for this model finding SERR, SIF and buckling strength reduction.

Model	Length(mm)	Width(mm)	Thickness(mm)
Composite panel	500	500	3

Table: 2.1.1.model dimension

Properties	Boron/epoxy
Longitudinal modulus, E1 (GPa)	208
Transverse in-plane modulus, E2 (GPa)	25.4
Transverse out of plane modulus, E3 (GPa)	25.4
In- plane shear modulus, G12 (GPa)	7.2
Out of plane shear modulus, G13 (GPa)	7.2
Out of plane shear modulus, G23 (GPa)	4.9
Major in- plane poison's ratio, ϑ12	0.1677
Major out of plane poison's ratio, ϑ13	0.1677
Major out of plane poison's ratio, ϑ23	0.36
Density (g/cm^3)	2
Yield stress (GPa)	1.165
Ultimate strength (GPa)	1.550

Table 2.1.2 .material properties

There are two type of delamination namely embedded delamination and Through- width-delamination as shown in figure4, in this thesis considering through-width-delamination, for this model finding SERR, SIF and buckling strength reduction.

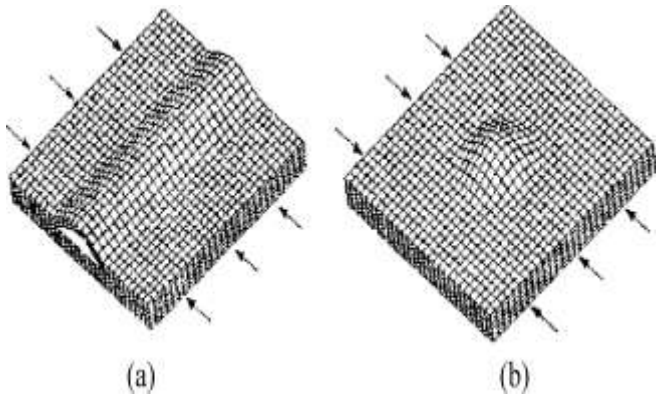


Figure 2.1.1: (a) Through width delamination, (b) Embedded delamination [1]

2.2 Finite Elément Methodology

Usage of composites in aerospace and automotive structures is increasing in present era. Numerical approach on the model is essential to achieve reasonably accurate results. This is essential to ensure the proper design. The numerical method such as finite element technique is a popular method for analyzing composites. Despite of these benefits, composite will fail due to delamination of interlaminar ply. In this thesis study is carried on through width delamination of Composite panel made up of 16 symmetric ply of Born/epoxy, each oriented at 0 degree $(0/0/0/0/0/0/0/0)_s$. This laminate of compressive load is modeled by FE method. Modeling is done with MSC Patran and for analysis using MSC Nastran.

The specimen with through width delamination is shown in figure 3.1 where (a) is the length of delamination and (t) is the depth of delamination. Gauge length is (L) and width is (W) and the thickness is (h) of the panel which are 500 mm, 500 mm and 3 mm, respectively.

Through-width-delamination is modeled with three different delamination length (a/L=0.2, 0.28 and 0.4), two different delamination position (t/h= 0.125 i.e. from top in between 2nd and 3rd layer, t/h=0.25 i.e. from top in between top 4th and 5th layer) and for progressive delamination length (a/L=0.2,0.28,0.36).Position in between 2nd and 3rd (t/h=0.125), 4th and 5th (t/h=0.25).

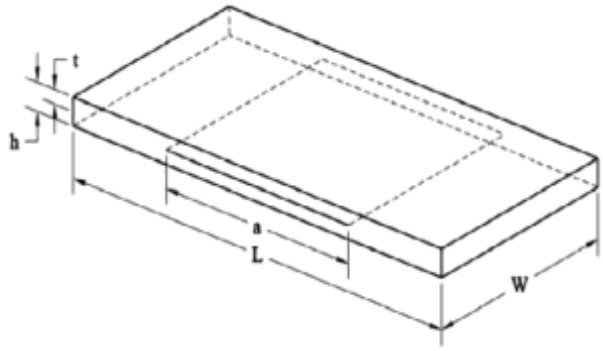


Figure 2.2.1The specimen with through width delamination

2.3. Meshing methodology

The way the meshing is done will affect the results. Here Hexa8 brick element is used to discretize the model to get the accurate SIF and SERR at the delamination front.

Quadrilaterals generalize to hexahedra. A hexahedron is a polyhedron with six faces, eight corners and sixteen edges or sides. It is topologically equivalent to a cube. It is informally known in the finite element literature as brick. Finite elements with this geometry are extensively used in modeling three-dimensional solids.

2.4. Convergence study:

Describes what might be one of the most concentrated issues that affect accuracy, namely; mesh convergence. This refers to the smallness of the elements required in a model to ensure that the results of an analysis are not affected by increasing the mesh or changing the size of the mesh.

The formal method of establishing mesh convergence requires a curve of a critical result parameter in a specific location, to be plotted against some measure of mesh density.

At least three convergence runs will be required to plot a curve which can then be used to indicate when convergence is achieved or, how far away the most refined mesh is from full convergence. Here plotting a curve buckling factor v/s number of element shown in figure 2.4.

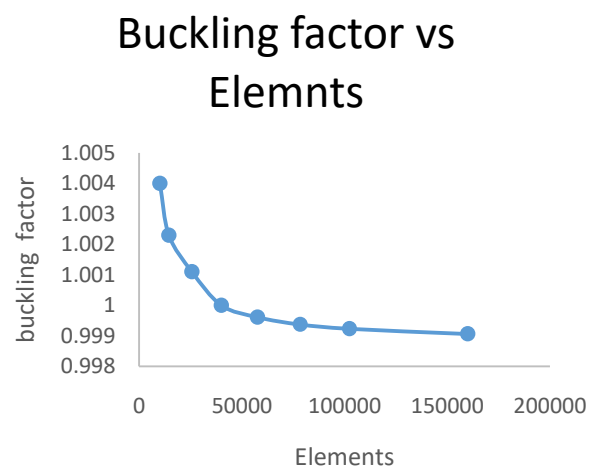


Figure 2.4: convergence study

From this convergence study we can prefer 40000 element, 8 nodded Hexahedron element model of each element size 10*10mm.

The mesh density is adjusted such that a finite mesh is created in the vicinity of the delamination regions.

The total number of elements present in the model is 40000 and the total number of nodes is 83232. Meshed model with initial delamination is shown in below figure.

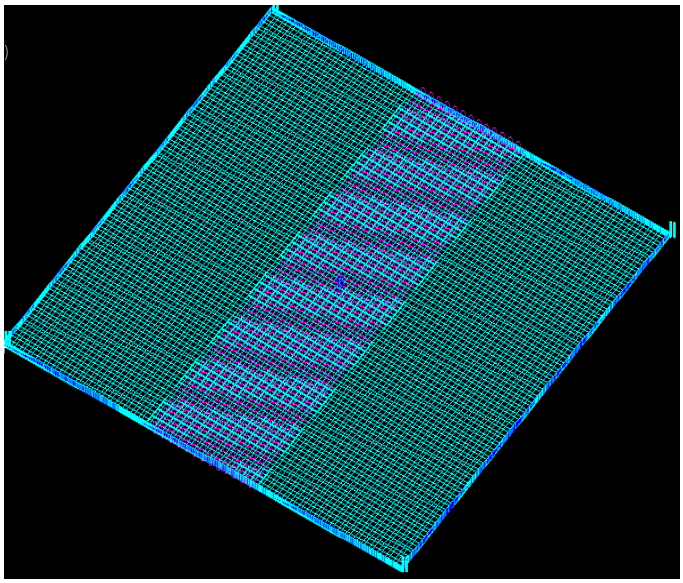


Figure2.4.1: Meshed model with initial delamination

To create contact properties b/w plies, equivalence is used which delete double nodes except the delamination region, initial delamination creating by keeping two nodes in between the laminae, except this region every plies contacted to form a laminate.

2.5. Boundary conditions

The boundary condition is the application of a force and/or constraint.

The model should be restrained otherwise applied load will cause infinite displacement, regardless of loading conditions, the minimum number of constraints that has to be imposed in two dimensions is three i.e. two translations along x and y, and one rotation about z. for in case of three dimension the minimal number of freedoms that have to be constrained is now six and many combinations are possible.

Those six boundary conditions are three translation along x, y and z, three rotation along x, y and z. in my case two simply supported and two side free, as shown in figure.

Displacement in x and z direction is zero along the boundary of left end

Displacement in z direction is zero along the boundary of right end

Displacement and rotation in y is zero along the boundary of bottom end

Displacement and rotation in y is zero along the boundary of top end

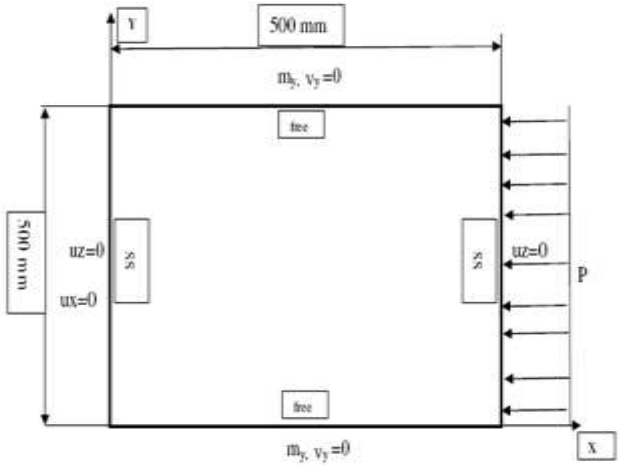


Fig 2.5.Model with boundary condition

2.6. Loading

Compressive load applied to the end as shown in figure 9, this is achieved by applying negative stress (pressure) along the right end of the model. In the left end of x direction translation is constrained.

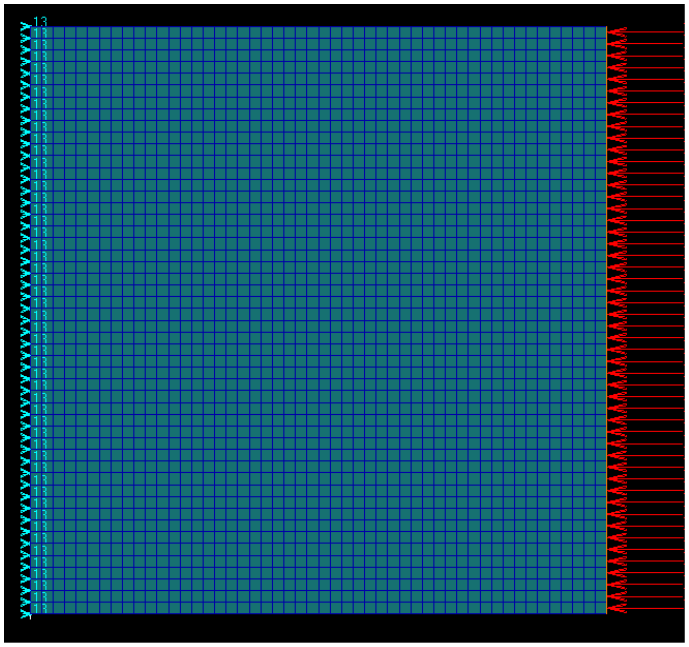


Figure 2.6: loading

III. RESULTS

3.1. Buckling Analysis

3.1.1. Introduction

Composite material lamina will be relatively thin, so occurrence of buckling is a primary concern when specimens are loaded in compression. Known Euler buckling or gross

buckling, this phenomenon is instability. That is, the test specimen carries the applied load as it increases with *no indication* of incipient failure until it suddenly collapses when the load is increased by a small amount.

These structural elements are often subjected to significant forces such as in-plane compressive or shear loading. So understanding and proper application of composite materials have helped to influence the lifetime and stability of these constructions. Thus in the design context buckling analysis plays a crucial role.

Buckling analysis or parametric studies are often conducted by FE analyses. But sometimes these FE analyses are quite complex and make heavy demands on both computer resources and the analyst's expertise. There is a need for simplified but reliable analysis methods that can readily be used for parametric studies.

Laminated composite plates are made up of plies (layers), each ply being composed of straight, parallel fibres (e.g., glass, boron, graphite) embedded in and bonded together by a matrix material (e.g., epoxy resin). Each ply may be considered as a homogeneous, orthotropic material having a value of Young's modulus (E) considerably greater in the longitudinal direction than in the transverse directions. Adjacent plies will have longitudinal axes usually not parallel. Cross-ply laminated plates arise in the special case when the longitudinal axes of adjacent plies are perpendicular, whereas angle-ply laminates occur when adjacent layers are alternately oriented at angles of + Θ and - Θ with respect to the edges of the plate.

A laminated plate is made by using a lamina as the building block. Its stiffness is obtained from the properties of the constituent laminae.

To do this we should know the orientation of the principle material direction of the laminae with respect to the laminate axes.

Therefore a knowledge of the variation of stress and strain through the laminate thickness is necessary. We shell make the following assumption regarding the behaviour of a laminate [6]:

1. It is made up of perfectly bonded laminae.
2. The bonds are infinitesimally thin and no lamina can slip relative to the other. This implies that the displacements are continuous across the lamina boundaries
3. After buckling, a line originally straight and perpendicular to the middle surface of the laminate remains straight and perpendicular to the middle surface $\Upsilon_{xz}=\Upsilon_{yz}=0$.
4. This strain perpendicular to the middle surface of the laminate is ignored.
5. Each lamina is homogeneous.

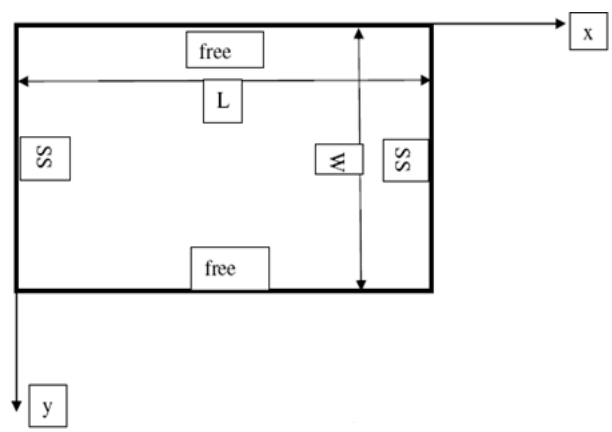
The buckling of plate involves two planes, namely xz, yz, and boundary conditions on each edge of the plate. The basic difference between column and the plate lies in buckling characteristics.

A column once it buckles, can't resist any additional axial load, thus the critical load of the column is failure load. On the other hand, a plate, since it is invariably supported at the edges (e.g., interconnection between two structural plates and web connected to flanges) continues to resist the additional axial load even after it has reached primary buckling and it doesn't fail when the load reaches a value 10-15 times the buckling load.

3.2.1 Analytical solution

Before going to numerical solution analytically derived to find the buckling load from the energy method for the composite laminate without delamination, consider simply supported at the two opposite edges and free at the other two edges as shown in the figure, has sides of length a and b and it is compressed by uniformly distributed load N_x acting along two opposite edges. The boundary conditions are x=0, x=a

w=0, $\frac{dw}{dx}=0$, y=0, y=b w=0, $\frac{d2w}{dy2}=0$



3.2.2. boundary condition

one term solution of the type

$$w=\sum_{m=1}^{\infty}\sum_{n=1}^{\infty}A_{mn}\left(1-sin\frac{m\pi x}{a}\right)\sin\left(\frac{n\pi x}{b}\right)$$

Satisfy the boundary condition, the non-dimensional form of strain energy

$$U=\frac{D}{2}\int_0^a\int_0^b\left(\left(\frac{d2w}{dx2}+\frac{d2w}{dy2}\right)^2-2(1-\mu)\left(\frac{d2w}{dx2}\frac{d2w}{dy2}-\frac{d2w}{dx\,dy}\right)\right)\mathrm{dx\ dy}$$

Non dimensional form of the potential energy

$$V = -\frac{1}{2}\int_0^a \int_0^b N_x \left(\frac{d2w}{dx2}\right)^2 \, dx\, dy$$

$$\frac{d\pi}{dAmn} = 0 \; to \; find \; Nx$$

$$\frac{dw}{dx} = \sum\sum\left(Amn\left(-\cos\left(\frac{m\pi x}{a}\right)\right)\left(\frac{m\pi}{a}\right)\sin\left(\frac{n\pi y}{b}\right)\right)$$

$$\frac{d2w}{dx2} = \sum\sum\left(Amn\left(\frac{m\pi x}{a}\right)^2 sin\left(\frac{m\pi x}{a}\right)\sin\left(\frac{n\pi y}{b}\right)\right)$$

$$\frac{d2w}{dy2} = \sum\sum\left(Amn\left(1-\sin\left(\frac{m\pi x}{a}\right)\right)^2\left(\frac{n\pi}{b}\right)^2\sin\left(\frac{n\pi y}{b}\right)\right)$$

$$\frac{d2w}{dxdy} = \sum\sum\left(Amn\left(-\cos\left(\frac{m\pi x}{a}\right)\right)^2\left(\frac{m\pi}{a}\right)^2\left(\frac{n\pi}{b}\right)cos\left(\frac{n\pi y}{b}\right)\right)$$

$$\left(\frac{d2w}{dx2}+\frac{d2w}{dy2}\right)^2 = \sum\sum((\sin\left(\frac{m\pi x}{a}\right)\sin\left(\frac{n\pi y}{b}\right) + \left(\frac{n\pi}{b}\right)^2\left(1-\sin\left(\frac{m\pi x}{a}\right)\right)\sin\left(\frac{n\pi y}{b}\right))^2$$

$$\pi = U + V$$

$$\pi = \frac{D}{2}\left\{\sum\sum A^2\left[\left(\frac{m\pi}{a}\right)^4\left(\frac{ab}{4}\right)+\left(\frac{n\pi}{b}\right)^4\left(\frac{b}{2}-\frac{ab}{4}\right)\right] - \left(\frac{m\pi}{a}\right)^2\left(\frac{n\pi}{b}\right)^2\right]\} - \sum\sum A^2\left[-\frac{1}{2}N_x\left(\frac{ab}{4}\right)\left(\frac{m\pi}{a}\right)^2\right]$$

By simplifying obtain equation for $\mathbf{N_x}$,L and **W** are substituted as 500mm

$$N_x = \frac{E\,t^3}{12(1-\mu^2)}\left(\frac{9.70*10^{-5}}{1.2337}\right)$$

Therefore N_x= 37.878 N/mm

3.3. Numerical solution

Buckling analysis without delamination of composite panel are found by numerical method using MSC Nastran Patran as shown in this section

For composite laminate by applying the boundary condition and compressive load this achieved by applying negative stress and running Buckling analysis, buckling factor is obtained that multiplied with applied load will give the critical load this is also called buckling load of the composite panel. Buckled composite panel shown in figure 4.2.

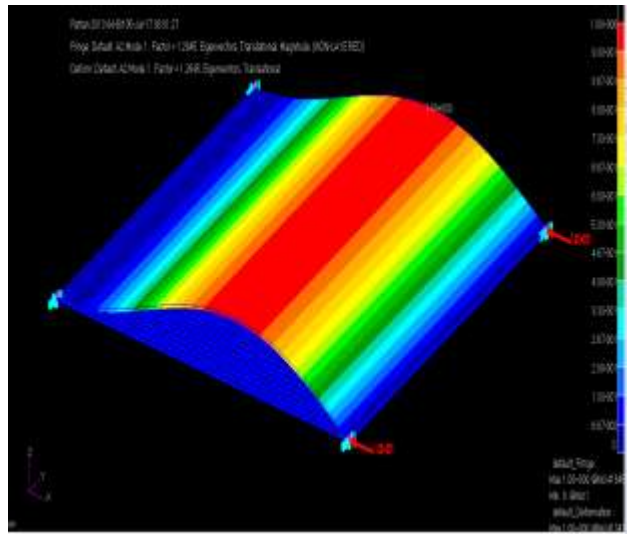


Figure 3.3: buckled composite laminate without delamination

In this case applying a negative stress of 12.644 N/mm^2, getting buckling factor is one therefore,

Buckling load= Applied load*Buckling factor

Buckling load=10*1.2644= 12.644N/mm^2 * 3mm= 37.932 N/mm

% Deviation in analytical to numerical is 0.142%

IV. CONCLUSION

Through width delamination is one of the failure modes in composite laminate. The propagation of delamination front depends on the Strain Energy Release Rate and the Fracture toughness of the material. This in turn depends on the loading, initial delamination width, position of the delamination. Finally, delamination greatly affects the buckling strength.

V. FUTURE ENHANCEMENT

With the increase in use of composites large scale metallic and composite structures such as in aerospace, there are several more issues which need to be resolved so as to increase the confidence in composite structure

- Only unidirectional ply is used in this thesis; different orientation in fiber laminate can be used to find the buckling strength and SERR.
- Study the effect of voids present in the composite structure.
- Effect of interface value.

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