NUMERICAL ANALYSIS OF BUCKLING AND POST BUCKLING BEHAVIOR OF SINGLE HAT STIFFENED CFRP PANEL

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ABSTRACT

The present work study will be carried out to investigate the buckling and post-buckling behavior of hat stiffened Carbon Fiber Reinforced Polymer (CFRP) composite panel under uniform in-plane axial compressive loading. Finite element study has been carried out using commercial finite element software ABAQUS 6.13 version. A 4-node doubly curved thin or thick conventional shell, reduced integration, hourglass control and finite membrane strains (S4R), having 6 degree of freedom per node is chosen for performing the buckling and post buckling analysis of composite panels. Conventional shell elements are used for analyzing moderately thick shell structures and well suited for large rotation and large strain non linear applications. Comparing to continuum shell (solid elements) which provides similar result, the conventional shell elements has great advantage in saving the computational effort and time. Finite element study using Eigen buckling followed by non linear analysis to obtain the buckling behavior of CFRP panel including mode shapes, critical buckling load, end shortening (axial displacement) and out of plane deflection.

Keywords: *Buckling behavior, Eigen approach, Hat stiffened composite panel, Mode shapes, Post buckling analysis.*

1. INTRODUCTION

A composite material consists of two or more chemically distinct constituents having a distinct interface separating them. Composite helps us to attain superior properties in our desired way, compared to the each constituent consist in the composites separately. Composite materials have a wide range of applications because of its high strength/stiffness to weight ratio, excellent fatigue and tailor-ability properties. Starting from traditional application areas such as military aircraft, now composite has grown rapidly to make an impact on various engineering and industrial fields including automobile, civil structures and even marine structures.

Carbon fiber reinforced polymer (CFRP) is lightweight strong material used in the manufacture of countless products. CFRP has application in aerospace engineering, sports goods and lot of high end products that requires stiffness and low weight. The most interesting property of CFRP is its high strength to weight ratio. So structure manufactured using CFRP can have less weight than the metal counterparts while maintaining the same load carrying capacity. They have good corrosive resistance than metals, which enables it to be used in bad weather conditions and having longer life. The stacking sequence of layers in CFRP can be changed according to its applications, which allows flexibility in design.

An aircraft manufacturer such as Boeing has started manufacturing most parts of fuselage and wings with CFRP panels. Fifty percent of the Boeing 787 was made using CFRP [1]. Fig. 1 shows the usage of composites in Boeing 787.The maintenance of the parts made of CFRP is also reduced.

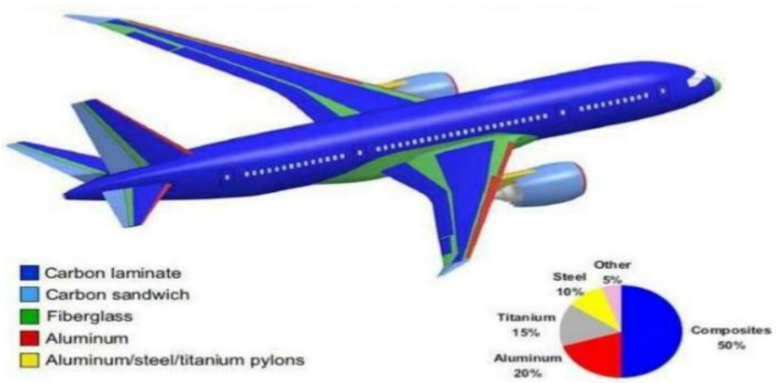


Fig. 1: Materials used in different parts of Boeing 787 [1]

Aircraft industries are able to achieve considerable amount of advantages in terms of fuel efficiency, low emission and noise reduction by replacing composites in place of aluminium in different structures including primary structures namely fuselage and wing [2].

Since CFRP has high strength to weight ratio, the thickness of the structure tends to be lower which increases the risks of buckling failure. Stiffeners are used to make these structures more reinforced against buckling. They increase the area moment of inertia of the cross section and torsional rigidity of the cross section. Stiffeners are classified according to its cross sectional shape. They are named as alphabets as they resemble which shows in Fig. 3. “L”, “C”, “Z”, “T”, “I”, “J”, “Hat” are examples of stiffeners [3].

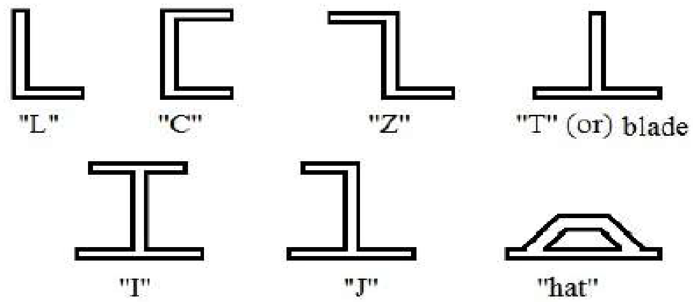


Fig. 2: Stiffeners of various cross sections [3]

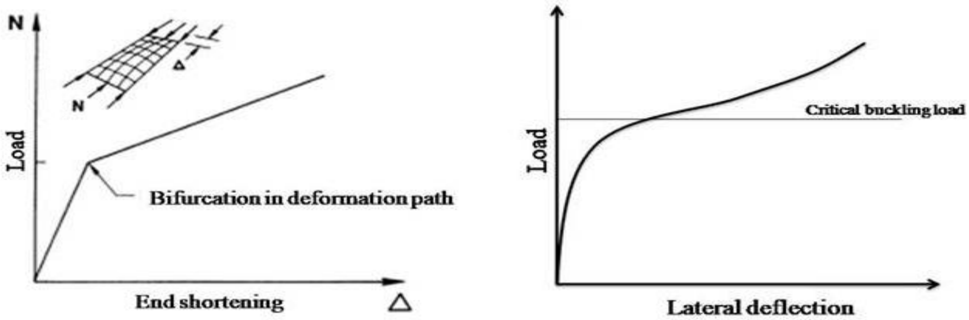
1.1 IMPORTANCE OF STIFFENER

Stiffeners perform various functions in an aircrafts. These functions include transferring bending loads in skin panels, stiffening and strengthening the skin panels so that panels don't buckle under compressive loading.

The stiffeners and skin panels may be made of fiber composites such as Carbon Fiber Reinforced Polymer (CFRP). CFRP is being used in replace of metal, especially in applications where posses high mechanical strength and low weight [4]. A composite stiffener is fabricated from multiple plies of reinforcing carbon fibers. Some plies have reinforcing fibers oriented at 0 degrees which transfer uniaxial loads and the fibers which having orientation ±45 degrees and 90 degrees to transfer shear, transverse and bearing loads with respect to the stiffener's axis of primary loading.

1.2 INTRODUCTION TO PLATE BUCKLING

Buckling is simply the geometrical instability of a structure and characterized by sudden sideways failure of the structural member subjected to high compressive stress, where the compressive stress at the point of failure is lesser than the ultimate compressive strength of the material. Buckling occurs due to the presence of imperfections in the geometry of the structure. When the compressive load is increasing, at a certain point further load is able to sustained in one or two states of equilibrium such as purely compressed stage or a laterally deformed stage, which is called as buckling.



(a) **(b)**

Fig. 3: Pre buckling and post buckling behaviour of simply supported plate under compressive loading: (a) Load vs. End shortening (b) Load vs. Out of plane deflection [3]

When the plate undergoes compressive loading, initially it shortens in the load direction while remaining flat. Then at critical buckling load, the deformation path bifurcates to buckled shape. Even after buckling, plate can support an increased load over the buckling load with a lesser stiffness, shown in above Fig. 3(a). This is because of the side edges which are still under pure compression by the help of side supports. Most of the entire load in the post buckling region is taken by this side edges. Out-of-plane deflection behaviour in the pre buckling and post buckling, shown in the above Fig. 3(b)

1.3 SCOPE AND OBJECTIVE:

Chia and Prabhakara [5], (1974) have studied the post buckling behavior of unsymmetrically layered rectangular anisotropic plates considering each layer having different elastic properties, arbitrary thickness and orientation with respect to the plate axes.

Nemeth [6], (1986) has studied the importance of anisotropy on the buckling of symmetric composite panels under compressive loading. A finite element has been carried out by him to analyze the importance of the fiber orientation, boundary conditions, stacking, sequence, aspect ratio and thickness in anisotropic bending stiffness.

Chiara Bisagni et al. [7], (2010) the present work with the response of single-stiffener compression specimen with a co-cured hat-stiffener, as shown in Fig. 5. Aeronautical panels are stiffened with stiffeners in the axial direction and with frames in the circumferential direction. A typical fuselage is shown in below figure where stiffeners with hat cross section and this stiffener configuration influences both the buckling and post buckling behavior as well as damage mechanisms. For computational efficiency and also for experimental considerations, a single stiffener hat specimen was developed based on the geometrical nonlinear analysis of a multi-stiffener panel.

Fig. 4: Composite fuselage with hat cross section [7]

The effect of torsional stiffness is the main difference in the open vs. closed section. The torsional stiffness of the closed section resists the panel from the buckling deformations. The stiffener twisting resistance is very important to control the buckling deformations. In case of closed stiffeners, all parts of the stiffeners are connected to one another as against open stiffeners, which are not inter-connected at the periphery. In hat type stiffeners, shear stiffness of the web is important parameter because it transfer loads from crown of the stiffener to the panel.

The aim of the present work is to design a single hat stiffened Carbon Fiber Reinforced Polymer (CFRP) composite flat panel for numerical analysis by using commercial FEA software ABAQUS 6.13 version. Conventional shell element is used to find out the buckling and post-buckling behavior of single hat stiffened composite panel.

2. MODELLING

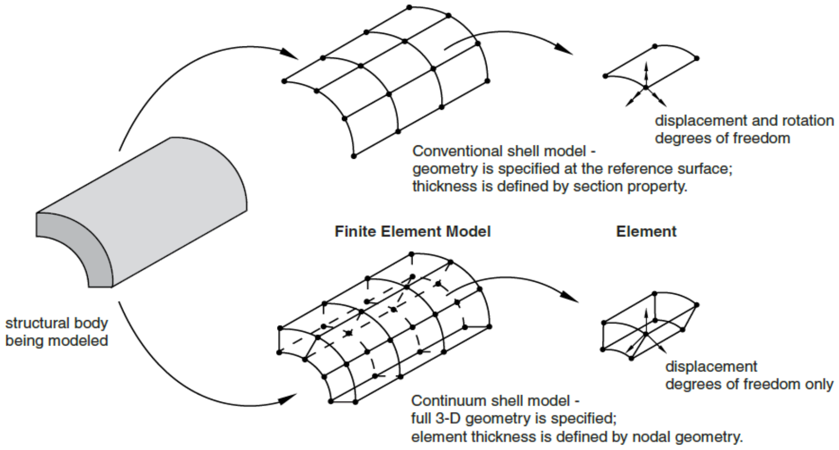


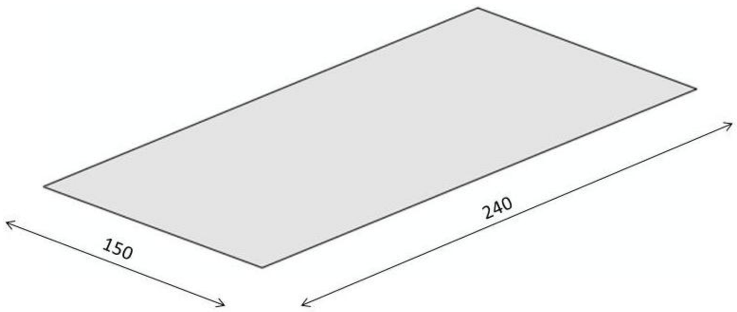
Fig. 5: Conventional shell vs. Continuum shell [9]

Finite element study has been carried out using commercial finite element software ABAQUS 6.13 version [8]. Conventional shell elements are typically planar element, used to model structures where thickness is negligible compared to its length & width and which experience large bending. Comparing to continuum shell (solid elements) which provides similar result, the conventional shell elements has great advantage in saving the computational effort and time. The geometry of conventional and continuum shell are shown in above Fig. 5.

2.1 SKIN PART

Finite element studies are carried out for 10 layered Carbon Fiber Reinforced Polymer (CFRP) rectangular flat plate, having dimensions of 240 mm length, 150 mm width as shown in Fig. 6. Thickness of each layer is 0.22 mm while the plate having 2.2 mm thickness.

Now, the skin part behaves as base part which is like the main panel in the components like aircrafts. The structure of skin parts are flat and curved shape. A flat rectangular panel is used, which is suitable for this requirement.



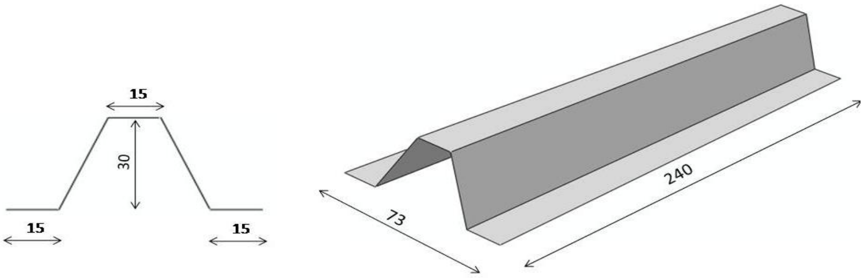
*All dimensions are in mm

Fig. 6: Schematic diagram showing isometric view of skin panel

2.2 STIFFENER PART

Finite element studies are carried out for 8 layered Carbon Fiber Reinforced Polymer (CFRP) hat shape stiffener, having dimensions of 240 mm length, 73 mm width, 30 mm height, 15 mm flange width, 15 mm crown top width as shown in Fig. 7. Thickness of each layer is 0.22 mm while the stiffener having 1.76 mm thickness.

Stiffeners are the composite panels which are used to increase the stiffness, strength and buckling load capacity for the base skin plates. There are various types of stiffeners are in use, but now the hat type stiffener is selected on the basis of requirement.

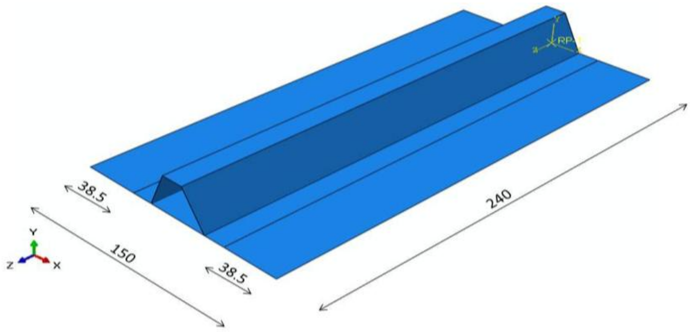


*All dimensions are in mm

Fig. 7: Schematic diagram showing isometric view of hat stiffener

2.3 ASSEMBLY

All the parts of required stiffened panel are assembled by creating instance in between them. With the help of Rotation, parts will be rotated to required position and with Translate, the parts are arranged at required specific point. By the interaction process, a Tie constraint is created, and the surfaces of both skin and stiffener are selected for interaction, this referred as surface-surface constraint. Fig. 8 shows assembly configurations of hat stiffened panel.



*All dimensions are in mm

Fig. 8: Schematic diagram showing isometric view of hat stiffened panel

2.4 MATERIAL PROPERTIES

Table 1 shows orthotropic material properties of UD-CFRP laminate. For the current study, CFRP consists of unidirectional carbon fiber mat as the reinforcement and ARALDITE® CY 230-1 IN epoxy resin mixed with ARADUR® HY 951 IN hardener is used as the matrix [10].

Table 1: Material properties of UD-CFRP laminate

Material properties		**Value**
Longitudinal modulus	$\mathbf{E_{11}}$	105.68 GPa
Transverse modulus	$\mathbf{E_{22}}$	4.64 GPa
Transverse modulus	$\mathbf{E_{33}}$	4.64 GPa
In-plane shear modulus	$\mathbf{G_{12}}$	3.34 GPa
In-plane shear modulus	$\mathbf{G_{13}}$	3.34 GPa
Out-plane shear modulus	$\mathbf{G_{23}}$	1.55 GPa
In-plane Poisson's ratio	$\mathbf{v_{12}}$	0.36
In-plane Poisson's ratio	$\mathbf{v_{13}}$	0.36
Out-plane Poisson's ratio	$\mathbf{v_{23}}$	0.49

2.5 LAYUP SEQUENCE

In the composite layup, the stacking sequence of each layer is initialized. The Eigen buckling loads are varying for multi stacking sequences. The following layup counts which are in symmetrical order of quasi-isotropic & cross-ply layup sequence which shows in below table are used. Quasi-isotropic layup stacking sequence is selected because it gives best buckling behavior among all layups on considering pre and post buckling stiffness.

Table 2: Lay-up count and stacking sequences

Composite lay-up	**Part**	**No. of plies**	**Stacking sequence**
Conventional shell	Skin	10	[45/90/-45/0/0]s
Conventional shell	Stiffener	8	[45/0/-45/0]s

2.6 MESHING:

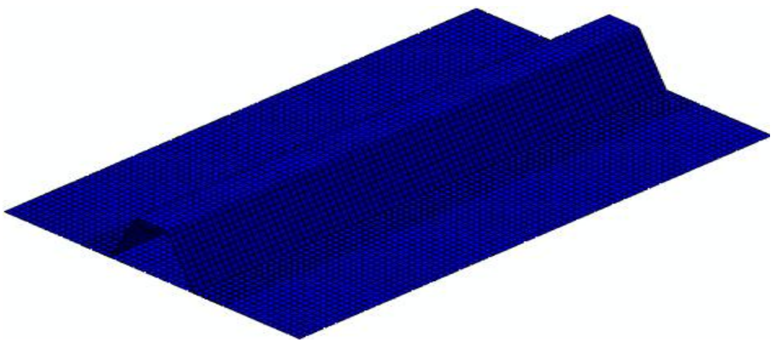


Fig. 9: Isometric view of meshing elements

The meshed model of hat stiffened panel modeled in ABAQUS 6.13 as per the problem dimensions described in section 2.1 & 2.2. Here S4R (A 4-node doubly curved thin or thick shell, reduced integration, hourglass control, finite membrane strains) shell element type has been used. Sweep technique used to assign mesh control for the curved object.

For the stiffened panel, the element size is chosen to be 3mm x 3mm, with skin having 4000 elements over the dimension of 240mm x 150mm and stiffener having 2960 elements over the dimension of 240mm x 73 mm, with a total number of 6960 elements for complete hat stiffened panel after a mesh convergence study, as shown in Fig. 9.

2.7 LOADING AND BOUNDARY CONDITIONS

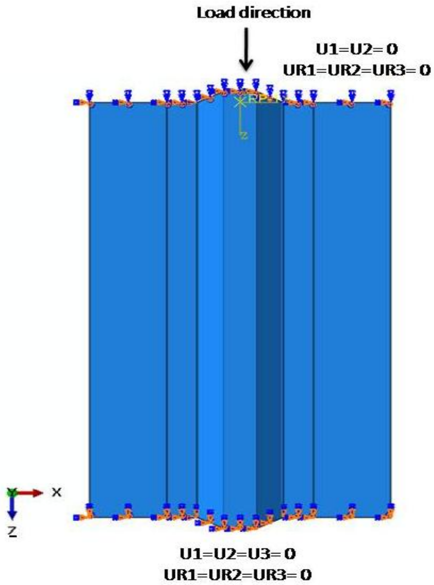


Fig. 10: Loading and boundary conditions

Boundary conditions consists simply supports along the straight edge of the panel and fixed boundary along the bottom edge of stiffened panel and clamped boundary on top edge of the stiffened panel, as shown in Fig. 10.

A group of nodes are selected to form as a set by using Set Manager. A master node is created for the nodes on the top. By using Equation Constraint in Constraint Manager at Interaction module the set of nodes are joined to master node so that a cocentrated load is applied on master node of the force of 1N in z-direction, because it indicates longitudinal direction. Normally a concentrated load of 1N is applied and then the eigen value multiplied to applied load to obtain critical buckling load.

3. RESULTS AND DISCUSSION

3.1 EIGEN BUCKLING

Linear buckling is the most basic form of buckling analysis in FEA. The Eigen buckling analysis produced the critical buckling loads and the corresponding mode shapes for all the specimens. Eigen value buckling analysis predicts the theoretical buckling strength of an ideal linear elastic structure. Eigen buckling analysis also

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predicts the mode shape for corresponding critical buckling loads. In this analysis non-linearity and particular initial geometric imperfections have not considered.

3.1.1 MODE SHAPES

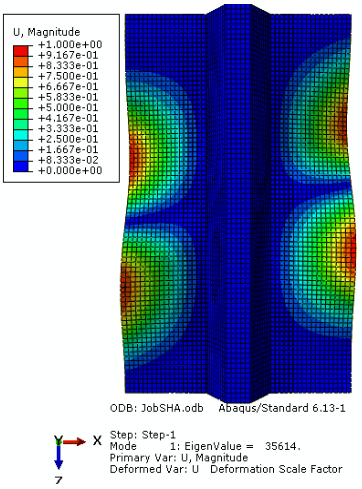

Fig.11: Eigen value 1= 35614

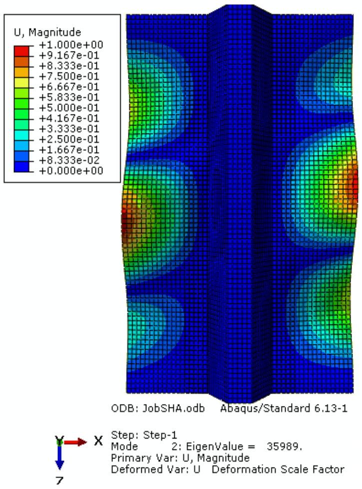

Fig.12: Eigen value 2= 35989

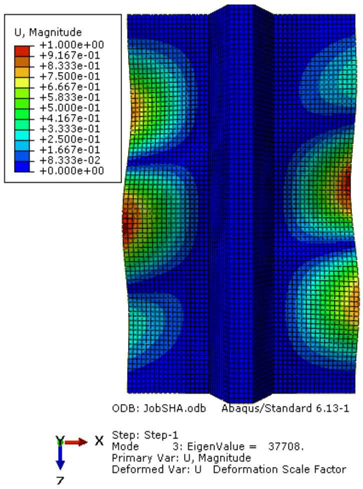

Fig.13: Eigen value 3= 37708

The Fig. 11, 12 and 13 gives the first, second and third Eigen values and they shows mode shapes in symmetrical order, having color indications for showing deformation levels, which comes from buckling analysis which is useful to know the critical buckling load value, while consider U, Magnitude as primary variable.

Critical Buckling load = First Eigen value x applied load in N

Buckling load from FEA analysis = 35.614 kN

3.2 POST BUCKLING ANALYSIS

Nonlinear post buckling analysis is carried out on the composite panels to study the reduction of stiffness of the hat stiffened panel after buckling and also the out-of-plane displacement behavior of the panels. An initial imperfection in geometry or a small lateral load is necessary to initiate the instability of the structure which leads to buckling. The first mode shape from Eigen buckling analysis has been used as the initial imperfection with a scaling factor of 10% of the thickness of the plate. In the post buckling regime, the strain displacement relationship is non-linear and requires non-linear solvers to solve the resulting finite element matrix equations.

Eigen buckling load for the hat stiffened panel is 35.6 kN, from load vs. end shortening (axial displacement) plot, it can be observed that after the buckling, then also panel is able to carry further load, but there is significant reduction in axial stiffness of the hat stiffened panel.

From out-of-plane displacement plot, the drastic increase in slope in post buckling process has been observed. It is because once stiffened panel is perturbed from initial flat shape state, lesser load is required to bend the panel further. Rik's analysis method is used for post buckling analysis. In this analysis, initial geometric imperfection and non-linearity considered. The imperfection file has been generated by Eigen value analysis.

3.2.1 AXIAL DISPLACEMENT MODE SHAPES

These are mode shapes, which comes from non-linear analysis. These mode shapes indicates axial displacement after buckling process. Consider U as Deformation Scale Factor and U, Magnitude as primary variable for knowing end shortening (axial displacement). There are various colors on panel, which shows intensity levels of deformation. Fig. 14, 15 and 16 shows the first, second and third mode shapes with symmetrical order of axial displacement at 3.56 kN, 7.12 kN and 12.46 kN

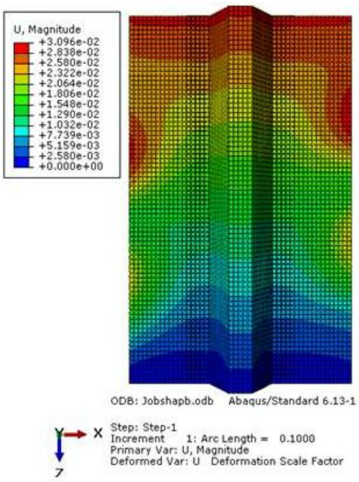

Fig. 14: Increment 1 = 3.56 kN

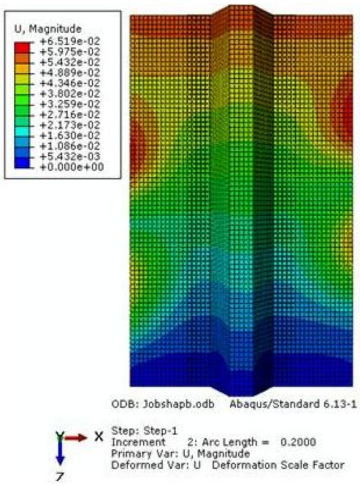

Fig. 15: Increment 2 = 7.12 kN

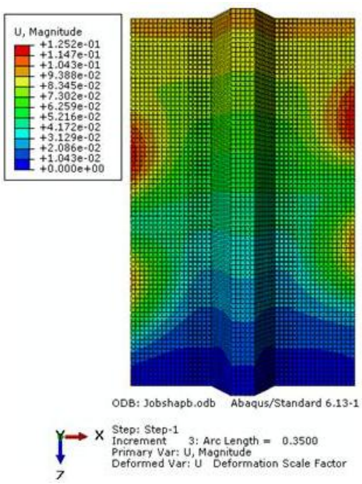

Fig. 16: Increment 3 = 12.46 kN

3.2.2 OUT-OF-PLANE DEFLECTION MODE SHAPES

These are mode shapes, which comes from non-linear analysis. These mode shapes indicates out of plane deflection after buckling process. Consider U as Deformation Scale Factor and U, U2 as primary variable for knowing out of plane deflection, because U2 is the transverse direction to axial compressive load. Fig. 17, 18 and 19 shows the first, second and third mode shapes with symmetrical order of out-of-plane deflection at 3.56 kN, 7.12 kN and 12.46 kN

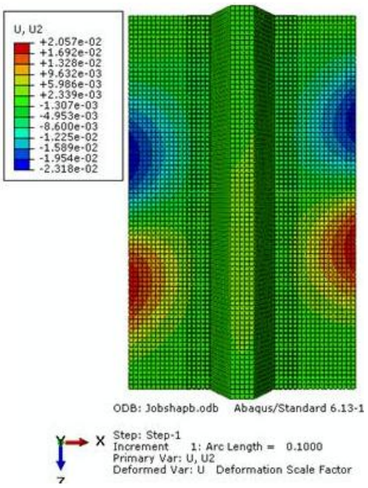

Fig.17: Increment 1 = 3.56 kN

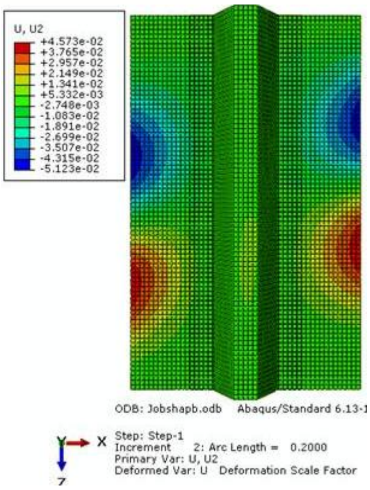

Fig.18: Increment 2 = 7.12 kN

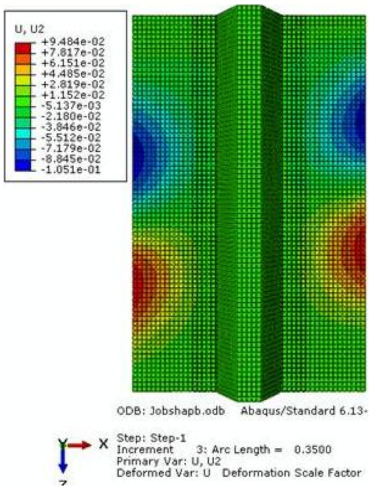

Fig.19: Increment 3 = 12.46 kN

3.3 GRAPH PLOTS

3.3.1 LOAD VS. END SHORTENING

In the post buckling process, assigned number of iteration steps, are obtained with respect to the time. The resultant force from the first Eigen value with the unit load value is applied on each time step and is calculated by multiplying with the time. The end shortening is calculated by extracting the z-displacement of the node at which force is applied at every time steps. This plot represents end shortening (axial displacement) in longitudinal direction with respect to applied axial compressive load.

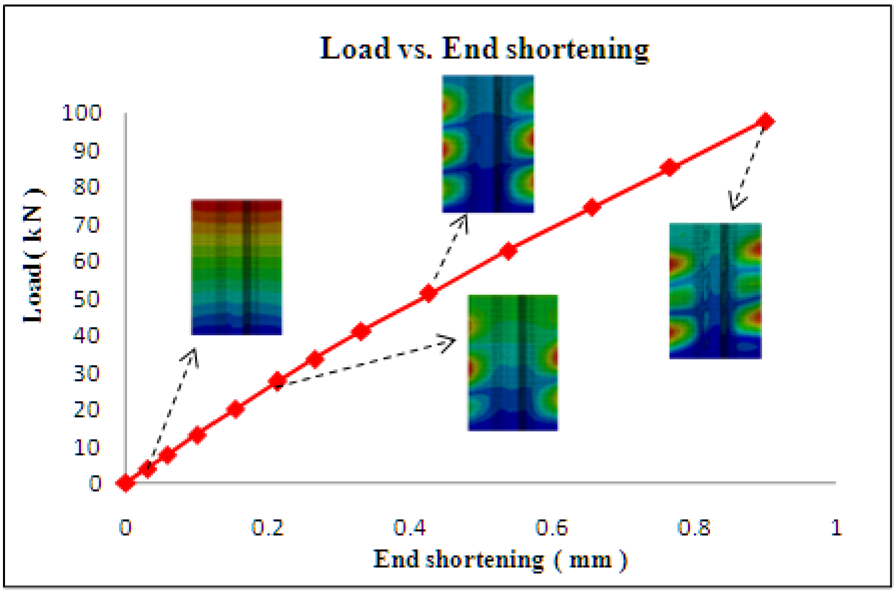

Fig. 20: Load vs. End shortening

3.3.2 LOAD VS. OUT OF PLANE DISPLACEMENT

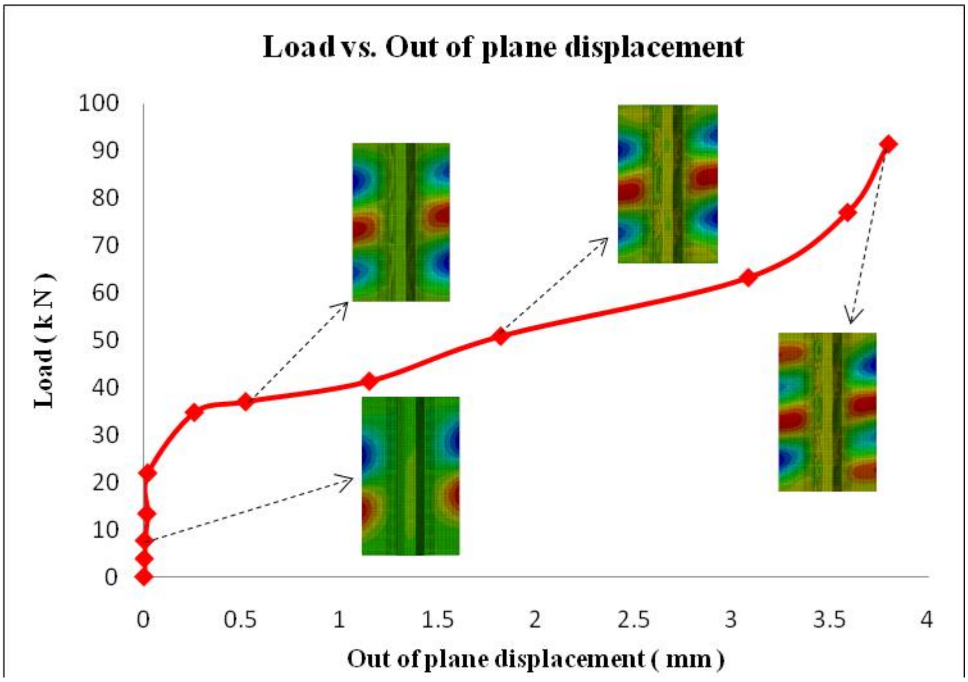

Fig. 21: Load vs. Out of plane displacement

The force applied on each time step is calculated by multiplying the time with maximum applied force. The out-of-plane displacement is calculated by extracting the y-displacement of the node where the plane is deflected in perpendicular to the load direction, which force is applied in z-direction at every time steps.

4. CONCLUSION

Finite element model was created for single stiffened hat structured panel. Buckling behavior of simply supported hat stiffened CFRP panel under axial compressive loading has been analyzed numerically. Non linear analysis of the model is also carried out. Pre and post buckling behavior of end shortening and out-of-plane displacements are obtained by Eigen buckling followed by non linear buckling analysis. The main objective of the analysis is to obtain out-of-plane displacements of hat stiffened panel and it was successfully determined.

[45/90/-45/0/0]s layup stacking sequence for skin & [45/0/-45/0]s layup stacking sequence for stiffener possess good buckling load and post buckling stiffness and recommended for experimental studies.

REFERENCES

[1].Christos Kassapoglou. *Design and Analysis of Composite Structures: With Applications to Aerospace Structures*, 2nd Edition, Aerospace series, May 2013

[2].http://arch5541.wordpress/2013/01/08/the_great_metal_tube_inthe_sky

[3].Robert M. Jones; *Mechanics of Composite Materials*, Second edition, Taylor & Francis Group, 1999

[4].https://patents.google.com/patent/US8720825/en, Composite stiffeners for aerospace vehicles

[5].Chia, C.Y., Prabhakara, M.K., *A General Mode Approach to Nonlinear Flexural Vibrations of Laminated Rectangular Plates. Journal of Applied Mechanics – Transactions of the ASME, Volume 45*, 1978, Page no: 623-628

[6].Michael P. Nemeth, *Importance of Anisotropy on Buckling of Compression- Loaded Symmetric Composite Plates, AIAA JOURNAL, Volume 24, NO.11*, 1986, Page no: 1831-1835

[7].Chiara Bisagni, Ricardo Vescovini and Carlos G. Davila, *Assessment of the Damage Tolerance of Post buckled Hat-Stiffened Panels using Single-Stringer Specimens*, *AIAA 2010-2696*, 2010, Page no: 1-11

[8].Commercial finite element software ABAQUS/CAE 6.13 version for Finite Element Analysis (FEA).

[9].Abaqus 6.13 Documentation; Abaqus/CAE Analysis User's Guide http://dsk.ippt.pan.pl/docs / Abaqus/v6.13/index.html

[10].Naresh Reddy Kolanu, Gangadharan Raju, M. Ramji, *Experimental and numerical studies on the buckling and post-buckling behavior of single blade-stiffened CFRP panels Composite Structures 196*, 2018, Page no: 135–154