Research Journal of Applied Science
MEDWELL PUBLICATIONS SINCE 2000

Effects of Ply Orientation on Nonlinear Buckling of Aircraft Stiffened Composite Panel

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Key words: Stiffened composite panels, nonlinear buckling analysis, ply orientations, FEM, CFC, FEA

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Page No.: 139-149
Volume: 15, Issue 4, 2020
ISSN: 1815-932x
Research Journal of Applied Sciences

Abstract: In this study, gradual demand for curved, stiffened panels composite emerged from Carbon Fiber Reinforced Plastics (CFRP) these are structures that are frequently used in aerospace engineering. Up to 50% of the primary structure of the Boeing 787 is made up of woven graphite-epoxy. This research emphasizes the buckling analysis of stiffened composite panels using the nonlinear, finite element modeling. The stiffened panel is assumed to be subjected to a uniform axial compression load of 10 kN. For the nonlinear buckling analysis, a stiffened composite panel is made of Carbon Fibre Composite (CFC) in epoxy, Kevlar in epoxy and E-Glass (EG) in epoxy. Due to the fact that there are numerous stiffened composite panels materials this study, therefore uses a numerical approach to present the study and results of the nonlinear static buckling analysis of i-stiffened on different composite panels. This research also presented the effect of the different ply orientations using Abaqus Finite Element Analysis (FEA).

INTRODUCTION

The fuselage is the major component of the aircraft which is made up of stiffened composite panels, the reinforced structures are made by arranging the members and frames in vertical, horizontal, diagonal and longitudinal directions. The study of fuselage sections is very expensive. This research is mainly, focused on reinforced panels, considering the fact that there are two categories of the panel, considering the stated limitation it is important more investigation be carried on this subject, so, as to uncover greater findings which is usually the impact of the different study. Aircraft completion essentially, depends on the capability of the structural panel materials to carry the total weight exerted by the internal components of the aircraft. Therefore, the construction of the fuselage for a commercial airliner is basically influenced by the interaction of its functions and its basic strength, rigidity and durability.

Manufacturing and designs must meet these criteria stated above while considering the goals of low mass and cheaper structure for the future airplane. Similarly, if there is a wealth of metallic materials available to choose from there are bollards to do intelligent design using metals.

Several researchers are working on the replacement of metallic materials with other best performing materials. Therefore, composites are considered to be a superior choice for converting metallic structures in order to obtain a better power and weight which ultimately translates into a better performance of aircraft.

The European aeronautical manufacturing industry is currently demand for a reduction in both the development and operating costs by 20 and 50% in the short and long term, respectively. European Commission POSICOSS (EC) which continued from January 2000-September 2004 is the project of regular quadrennial COCOMAT which extended until September 2008, contributed to this goal[1, 2].

Composite materials are particularly attractive for aeronautical and aerospace applications because of their exceptional strength/density ratio and superior physical properties[3]. Specifically, the application of laminates (Multi-layered composite) has much possible use in various aerospace fields[4].

Despite its wide application, laminates are generally missing in the normal direction of the fibre orientation angles. This is due to their laminated form which consists of stacked interfaces with lower resistivity values.

The divergence between two layers can be explained as a phenomenon, this phenomenon is called 'delamination' and it creates one of the most common modes of weakness in laminated composites. Delamination can dramatically reduce measured weight capacity and stability of the constituent materials thereby increasing the chances of breakage[5]. Logically, the modeling of composite failure behavior has evolved as a primary goal in recent years[6-8].

Delamination is one of the most important factors in the layered composites for structures subjected to compressive loads because this is buckling will take place at the lower load level[9]. Manufacturing defects, collisions with birds and instrument graves are some of the causes of delamination. Delamination mostly recedes to lateral shear and lateral shear at regular constraints, it is hard to sense this inter-laminar fracture due to the toughness of the CFRP composites[10]. However, there is an increasing demand for a more precise and finite method of analysis as a result of recent and rapid development in computational analytical. When a laminate is under compression, the impact of delamination on stiffness and hardness can be determined by the pre-buckling load and post-buckling under submitted loads. Therefore, buckle mode is defined as the cracking mode in which the submitted structure has an abrupt failure due to delamination when subjected to compressive loading[11].

When the structure is subjected to axial compression loading, due to the design method a short deformation will be created on the structure precisely when the load is at a critical level. As a result of the above condition, the structure will suddenly be subjected to severe deformation and as such, lack lift which is co-spreading in pre-buckling.

Zhang *et al*.[12] applied grapheme oxide loading material, arranged with nano-diamond, nano-cluster to complete the strength and flexural modulus and significantly improve the material's resistance to fracture. They observed that the epoxy samples containing 0.1% by weight of hybrid filler material (GN), exhibited enhanced mechanical stability over the epoxy composite having 0.2% by weight of Graphene Oxide (GO). The epoxy composite prepared by the ND/GO filler material also exhibited improved thermal properties such as decomposition temperature and activation energy. The authors argued that the presence of nano-diamonds not only prevented the agglomeration of GO sheets but also acted as a fixing agent in the polymer composites which could improve its breaking strength[12].

Study of linear and non-linear buckling analysis of stiffened panels composite is carried out in the pre-buckling, a detailed study is therefore carried out to determine the buckling and post-buckling responses of stiffened panels composite with central circular defenses. Subject to various combinations of mechanical and thermal loads, the results showed the effects of variations in hole diameter, the aspect ratio of the panel and the position of the fiber at the end of stability[13, 14]. In real cases, the deviation continues even after subsequently taking the critical load, the post-buckling analysis is therefore non-linear and by rule, the non-linear load traversal relationship can be taken from the non-linear stress.

One notable and a probably better solution other than increasing the thickness of the plate could be to increase the stiffeners. Rigid panels are in principle, governed by the stability criterion of resistance. More details and reviews of the literature on laminated composite plates/shells may be found by Leissa[15] who presented a brief overview of the buckle composite panel[16]. Empirical studies in rigid and composite non-rigid panels were carried out by SudhirSastry *et al*.[17]. In principle, with the advent of FEM versatility many researchers around the world are currently working on designing the buckling attitude of the laminate through the FEM Models. SudhirSastry *et al*.[17] studied the buckling behavior of stranded laminated panels subjected to compression by applying a computational formation method[18].

Numerical methods for the modeling of composite laminates are not applicable in the design because of the presence of several variables. The most structural design used in aerospace manufacturing is presented under the configuration of thin curved panels subjected to subject to compressive stresses. In the current study, we considered the buckle strength of the multi-layers for curved stiffened panel composite against the axial load of compression. Therefore, the layout of this work is presented in the following form: mathematical and numerical studies of nonlinear and linear buckle models. Nonlinear buckling analysis. Numerical study of the rigidity of the curved stiffened panels composites reinforced by the different configurations of combinations of the composites. The E-Glass (EG) in epoxy and second composite is Carbon Fiber Composite (CFC) and finally with Kevlar.

Buckling analysis of stiffened composite panels: The objective of this study is to predict the linear and non-linear buckling behavior and resistance of an ideal linear and non-linear elastic structure. Which means that linear buckling analysis of a stiffened panel can be derived from the classic Euler method which is not an exact solution when there are non-linear geometric material and there are any imperfections, moreover, nonlinear buckling analysis is a non-linear static analysis with gradually increasing loads to check the load level at which the structure becomes unstable. Therefore, the non-linear buckling analysis is more explicit and is recommended for design.

Deformations and displacements of stiffened composite panels: We examined a curved composite stiffened panel cross-section as shown in the Fig. 1 with a coordinate system and the displacement is shown in Fig. 2.

Where are the displacement vectors and are the angles of rotation of a line perpendicular to the plane, along the direction x, y and z, respectively. According to the theory of deformation by first-order stress, the displacement is given by the following system:

$$\begin{aligned} u(x, y, z) &= u_0(x, y)\text{-}z\theta_x \\ u(x, y, z) &= v_0(x, y)\text{-}z\theta_y \\ u(x, y, z) &= w_0(x, y) \end{aligned} \quad (1)$$

where, θ_x et θ_y the average rotations of a perpendicular line within each element are can be expressed in terms of the unknown number of nodes using the following functions N_i form:

$$\begin{aligned} u &= \sum_{i=1}^{N} N_i u_i,\ v = \sum_{i=1}^{N} N_i v_i,\ w = \sum_{i=1}^{N} N_i w_i \\ \theta &= \sum_{i=1}^{N} N_i \theta_{xi},\ \theta_y = \sum_{i=1}^{N} N_i \theta_{yi},\ \theta_z = \sum_{i=1}^{N} N_i \theta_{zi} \end{aligned} \quad (2)$$

where, N is the global number of nodes, the Lagrangian method is used in this research. For each node, we define the response vector q as:

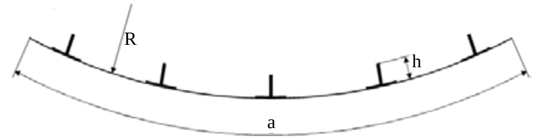

Fig. 1: Panel cross-section

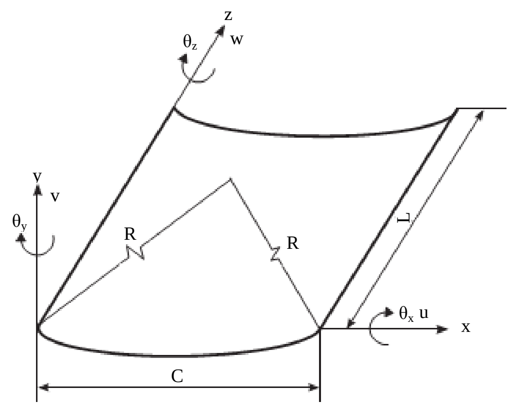

Fig. 2: Coordinate system with the displacement

$$\{q_i\} = \{u_i v_i w_i \theta_{x_i} \theta_{y_i} \theta_{z_i}\}^T \quad (3)$$

The total deformation is written in the following form:

$$\{\varepsilon\} = \{\varepsilon_0\} + \{\varepsilon_1\} = \begin{Bmatrix} u_{,x} \\ v_{,y} \\ u_{,y}+v_{,x} \\ -\theta_{x,x} \\ -\theta_{y,y} \\ -\theta_{x,y}-\theta_{y,x} \\ w_{,y}-\theta_x \\ w_{,y}-\theta_x \\ \theta_z \end{Bmatrix} + \frac{1}{2}\begin{Bmatrix} v_{,x}^2+w_{,x}^2 \\ u_{,y}^2+w_{,y}^2 \\ 2w_{,x}w_{,y} \\ 0 \\ 0 \\ 0 \\ 0 \\ 0 \\ 0 \end{Bmatrix} \quad (4)$$

Where:

$$\{\varepsilon\} = \{\varepsilon_x \varepsilon_y \gamma_{xy} K_x K_y K_{xy} \gamma_{yz} \gamma_{zx} \theta_z\}^T \quad (5)$$

Total potential energy of the panel: The total potential energy Π of the submitted plate in the plane and the transverse load is the summation of the strain energy U and external load potential Ω, expressed as follows:

$$\begin{aligned} \prod &= U+\Omega \\ \prod &= \frac{1}{2}\int \{\sigma\}^T \{\varepsilon\} d\Omega\text{-}\lambda\{q\}^T\{f\} \end{aligned} \quad (6)$$

λ It is assumed that the load factor to increment the load vector {f} and stress vector {σ} can be estimated from the right component:

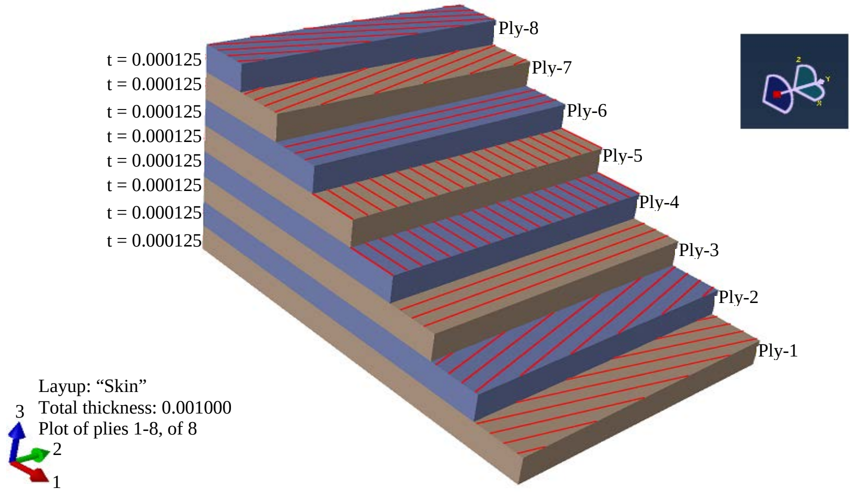

Fig. 3: Layup sequence used in the state is 8 layers with total thickness is 1 mm

$$\{\sigma\} = \{N_x N_y N_{xy} M_x M_y M_{xy} Q_x Q_y M_z\}^T \tag{7}$$

With N_x, N_y and N_{xy} are the stresses and M_x, M_y and M_{xy} are the moments Q_x, and Q_y are the shear stresses, the characteristic law:

$$\{\sigma\} = \{C\}\ \{\varepsilon\} \tag{8}$$

where, C is the material constant matrix, the study presents the evaluation of stresses and strains, deformations for static buckling analysis[19].

The matrices for the stability analysis: Analysis of linear static buckling at the beginning of the analysis, the stiffness matrix can be formulated as:

$$[K_0] = \int_\Omega [B_0]^T [C][B_0] d\Omega \tag{9}$$

where, (B_0) is the strain-displacement matrix. The pre-buckling can be given by this precautionary measure:

$$[K_0]\{q_0\} = \{f_0\} \tag{10}$$

The second phase is the detection of critical states on the fundamental path reason, it is important to calculate the geometry of the stiffness matrix $[K_\sigma]$ this can be done as follows:

$$[K_\sigma] = \int_\Omega [G]^T [\sigma][G] d\Omega \tag{11}$$

where, $[\sigma]$ is the stress vector, stress reorganized in the form of a matrix according to Taylor:

$$[\sigma] = \begin{bmatrix} N_y & 0 & 0 & 0 \\ 0 & N_x & 0 & 0 \\ 0 & 0 & N_x & N_{xy} \\ 0 & 0 & N_{xy} & N_y \end{bmatrix} \tag{12}$$

The linear eigenvalue problem is:

$$([K_0]\text{-}\lambda[K_\sigma])\{x\} = 0 \tag{13}$$

The critical load is determined by Eq. 13.

Non-linear static buckling analysis: Non-linear static buckling usually involves several variables. These variables construct the stability equations and discrete equations of the virtual working equation:

$$F^N(u^M) = 0 \tag{14}$$

Either F^N is the combined force component up to the N^{th} level in the problem and u^M is the value of the M^{th} variable Fig. 3. The essential objective is the solution for an Eq. 14.

Newton’s method is supposed that after iteration i an approximate u^M of the solution was obtained as the exact solution of the discrete stability Eq. 14 this shows that:

$$F^N(u_i^M c_{i+1}^M) = 0 \tag{15}$$

With c_{i+1}^M is the difference between this solution and the exact discrete result. By lengthening the left side of this equation in a series of Taylor’s. A linear system of equations:

Table 1: Mechanical properties of the CFC, the E-glass and Kevlar used in the analysis[20]

			Material		
Quantity	Symbol	Units	CFC	E-glass	Kevlar
Young's modulus 0°	E_{11}	Gpα	164	38	195
Shear modulus in planes 90°	$E_{22} = E_{33}$	Gpα	12.8	8.27	14.6
Shear modulus in planes	$G_{12} = G_{13}$	Gpα	-	-	-
Shear modulus in planes	G_{23}	Gpα	2.5	4	5
Poisson's ratio in planes	$\nu_{12} = \nu_{13}$	None	0.32	0.25	0.3
Poisson's ratio in planes	ν_{23}	None	0.45	0.27	0.45
Density	ρ	Kg/m^2	1800	1900	1400

$$F^N\left(u_i^M\right)+\frac{\partial F^N}{\partial u^P}\left(u_i^M\right)c_{i+1}^P+\frac{\partial^2 F^N}{\partial u^P\partial u^Q}\left(u_i^M\right)c_{i+1}^P c_{i+1}^Q+, ..., = 0 \quad (16)$$

$$K_i^{NP}c_{i+1}^P - F_i^N = 0 \quad (17)$$

K_i^{NP} is the Jacobian matrix. The best way to measure the convergence of Newton's method is to ensure that the entries in F_i^N and the inflows in c_{i+1}^M are small enough. (ABAQUS Documentation).

Modeling static buckling of stiffened panel: We examined a square panel where the total length of the panel and its width with a radius R = 381 mm and thickness of stiffened panels composite t_T = 1 mm as shown in Fig. 3, the thickness of each layer is t = 0.125 this panel consists of eight layers. The width of the stiffeners a = 33 mm. The layers with orientations are 0, ±15, ±30, ±45, ±60, ±75, ±90. Epoxy is treated as a matrix material. Table 1 shows the mechanical properties of the panels.

The mesh generated in this study corresponded to the convergence of 5128 shell nodes and 5104 linear quadrilateral elements of type S4R. For the representation of boundary conditions used in this study, let u, v and w be (the translations) while r_x, r_y and r_z are (the rotations) the slender (or roughly) x, y and z-axes as shown in Fig. 2. The longitudinal (uniform axial compression) edges are simply supported with w = 0 and $r_x = r_y = 0$. The transverse (loaded) edges are simply supported, so, w = and $r_x = 0$, I-shape stiffeners are represented as Fig. 3.

MATERIALS AND METHODS

The mechanical properties of composite panels in different laminas of the CFC, the E-glass and the Kevlar used in the analysis are presented in the following Table 1:

Steps for nonlinear buckling analysis: In post-buckling analysis nonlinear load-deflection curve is produced based on the modified Riks algorithm.

The shape of nonlinear buckling is induced with an initial defect based on the modes of buckling extracted. The buckling analysis is modified to perform a nonlinear load deflection analysis to predict post-buckling behavior. The following steps were followed in performing the static nonlinear buckling analysis in Abaqus FEA Software. Defined for the composite stiffened panel properties module using a composite layup.

The mechanical properties of each lamina are listed in Table 1. For nonlinear buckling analysis, the eigenvalue buckling step is a static Riks step. The axial force is defined along with the boundary conditions. The history output query is added to define the displacement history for the loads applied. The boundary conditions are applied and the job is submitted for the static nonlinear buckling analysis, the progress of the solution is monitored. The results of the static nonlinear analysis buckling analysis can be processed.

The results of the buckling analyses are discussed in detail in Fig. 4. This figure presents the variation of the loads buckling as function displacement with the different layup sequences and combinations of composite material.

Boundary condition of stiffened panel: The stiffened panel is modeled by the finite element method using S4R elements as shown in Fig. 5. This model consists of I-shaped stiffeners as shown in Fig. 4a, b. The panel is completely narrow at the base element, free to move axially along the longitudinal section of the upper edge, and simply along its vertical edges with an axial compression of 10 kN being applied to the stiffened panel.

Compares the buckling loads by numerical model and the experimental results: When a constant load is applied to different materials of layup sequence $(45/\text{-}45/90/0)_s$ there is a lookout for the scenario where the configurations of the composite stiffened panel made of the Kevlar epoxy and the buckling loads of the model analyzed by the Finite Element Method (FEM) are in good agreement with the experimental results this is a model of the experiment carried out in laboratory National School of Arts and Crafts in Meknes-Morocco where they studied the buckling stresses and buckling loads with different configurations of the composite stiffened panel. Their results presented estimates for three sequences,

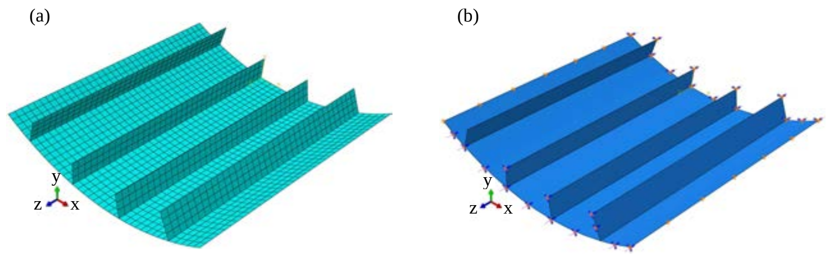

Fig. 4(a, b): (a)Finite element models with the straight stiffeners (b) and Finite element model with configuration boundary conditions and loads

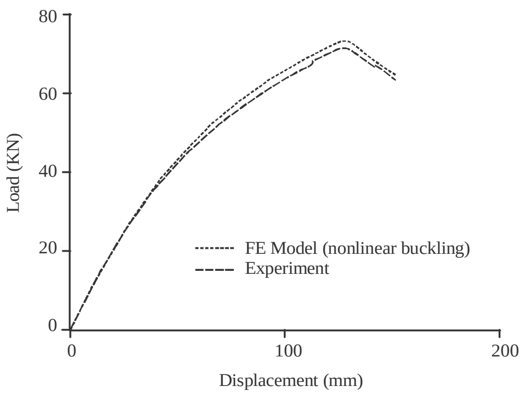

Fig. 5: Comparison of nonlinear buckling loads from finite element model and the experimental model

including linear Eigenvalue analysis loads (pre-buckling) of the numerical model in Abaqus which when compared with existing studies are excellent. However, this found the best results using the static non-linear buckling analysis method. This Nonlinear Method (MNM) for the numerical solution was found to be in very close agreement with the experiment results which is better than the method used by SudhirSastry *et al*.[17].

RESULTS AND DISCUSSION

Highlights:

- A new Stiffened Composite Panels structure, for Aircraft application, consists of carbon epoxy composite (CFC), Kevlar and E-glass (EG) is evaluated
- Static nonlinear post-buckling analysis of stiffened composite panels used in aircraft applications was studied
- Several combinations of the materials for stiffened composite panels

Analysis of the stiffened panel with four i stiffeners: In this research, we studied the nonlinear buckling analysis of the composite stiffened panel with four stiffeners as shown in the Fig. 4a. We eventually considered several cases which are generated by lay-up variation and decided to take the different combinations of materials for all configuration of the composite stiffened panel. In this example we examined several orientations with layup sequences $(45/\text{-}45/90/0)_s$, $(90/0/90/0)_s$and $(60/\text{-}30/90/0)_s$, respectively. We have examined several combinations of materials for stiffened panels, the cases analyzed in this research are shown below: when the panel and the stiffeners are made of the same material, the Carbon Fiber Composite (CFC) for panel and stiffeners, E-Glass (EG) for panel and stiffeners, Kevlar for panel and stiffeners as shown in Fig. 6a, b.

We examined the first case in Fig. 6(a) and (b) with a layup sequence $(45/\text{-}45/90/0)_s$, the stiffeners and the skin are of the same configuration with respect to the composite material. We have noticed that when the panel and stiffeners are made of Kevlar material it has a maximum critical load with a static non-linear buckling P_{cr} = 84 kN of E-glass composite, the epoxy contains the minimum critical buckling loads P_{cr} = 22 kN and when Carbon Fiber Composite (CFC) is placed in the middle of Kevlar and E-glass as shown in Fig. 6(a), the critical load P_{cr} = 68 kN. However, when the panel is made of different materials for instance, the critical load of maximum buckling is observed when the panel is made of Kevlar skin and CFC stiffeners whereas the critical load of the minimum buckling is observed in the E-glass skin with CFC stiffeners as shown in Fig. 6(b).

In case number two, we examined a Layup sequence $(90/0/90/0)_s$ where the stiffeners and the skin are made of the same composite material and this panel holds the four stiffeners, critical buckling loads are plotted in the Fig. 6c, d, respectively.

We looked at a condition, where the skin and the stiffeners are of the same composite material given Kevlar for instance, it has a maximum critical buckling load, E-glass epoxy have the minimum load, the critical loads of the non-linear static buckling of the CFC is placed between the Kevlar and EG as shown in the

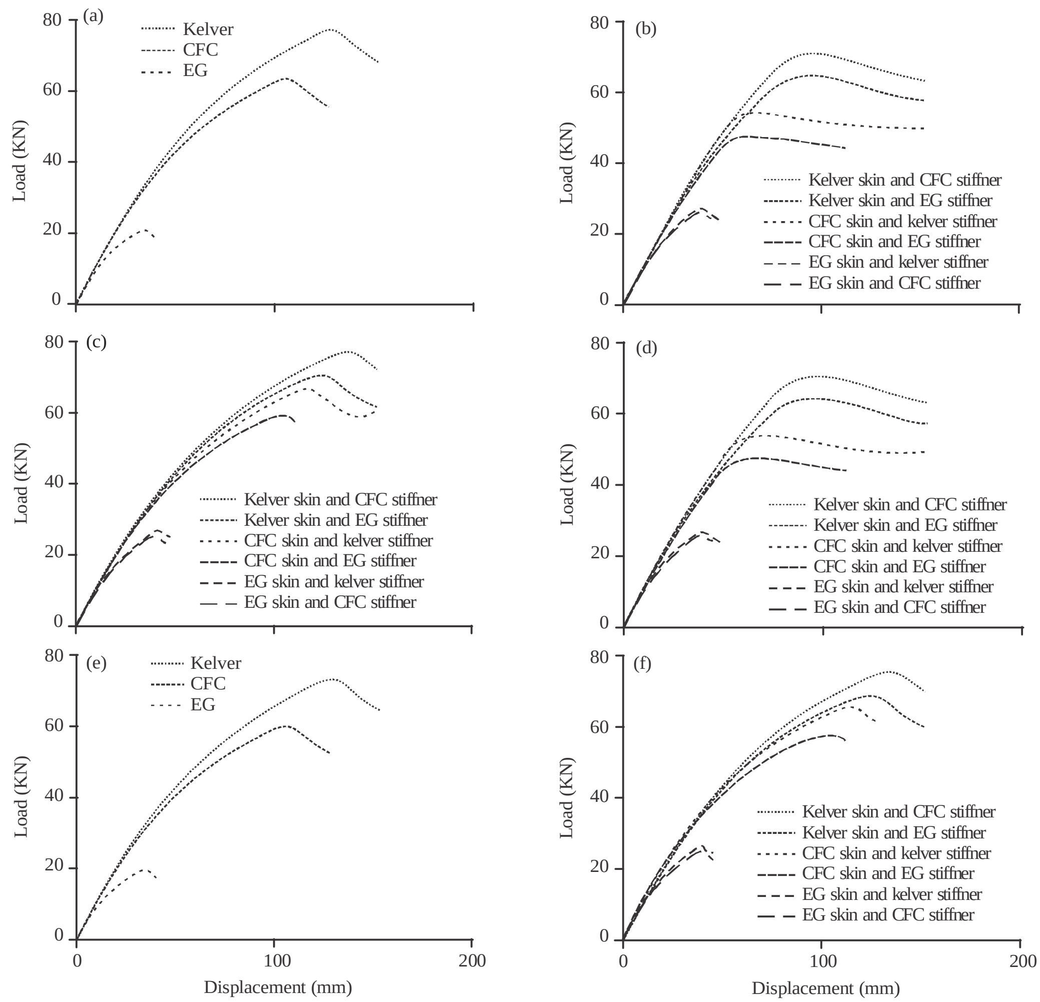

Fig. 6(a-f): Variation of the loads buckling as function displacement with the different layup sequences and combinations of composite material, (a) Layup sequence (45/-45/90/0)s; (b) Layup sequence (45/-90/0)s; (c) Layup sequence (90/0/90/0)s; (d) Layup sequence (90/0/90/0)s; (e) Layup sequence (60/-30/90/0)s and (f) Layup sequence (60/30/90/0)s

curves Fig. 6c. And when the skin and stiffener are made of different composite materials, the maximum critical loads are observed for composite in Kevlar skin and CFC stiffener while the minimum loads are observed in composites with EG skin and CFC stiffener as shown in Fig. 6d.

A layup sequence $(60/-30/90/0)_s$, we have observed that when the skin and stiffeners are made of the same composite material, Kevlar is the most resistant-glass epoxy known for minimum loads and the lowest loads while CFC is placed in the middle as shown in Fig. 6e.

When the skin and the stiffener are made of different composite materials, maximum loads are observed in epoxy Kevlar skin with CFC stiffeners while the minimum load is observed when the composite materials are EG skin and CFC stiffeners. Considering the most severe of the composite stiffened panel configuration with regards to buckling strength along with the different composite materials. The stress distribution at different composite materials and ply orientations are shown in Fig. 7a-f. The stress distribution of different configurations is plotted in Fig. 7a-f. The nonlinear analysis is performed and Von-Mises stress is calculated around the stiffened panel composite with different composite materials and ply sequence.

The von Mises stress distribution at different combinations of composites and ply orientations view are shown in Fig. 7. The maximum von Mises stress is 354.4 MPa on the case layup sequence $(45/-45/90/0)_s$ with composites EG skin and CFC stiffener which is in red color in Fig. 3. The second higher von Mises stress is 175.3 MPa occurs in Layup sequence $(90/0/90/0)_s$ with

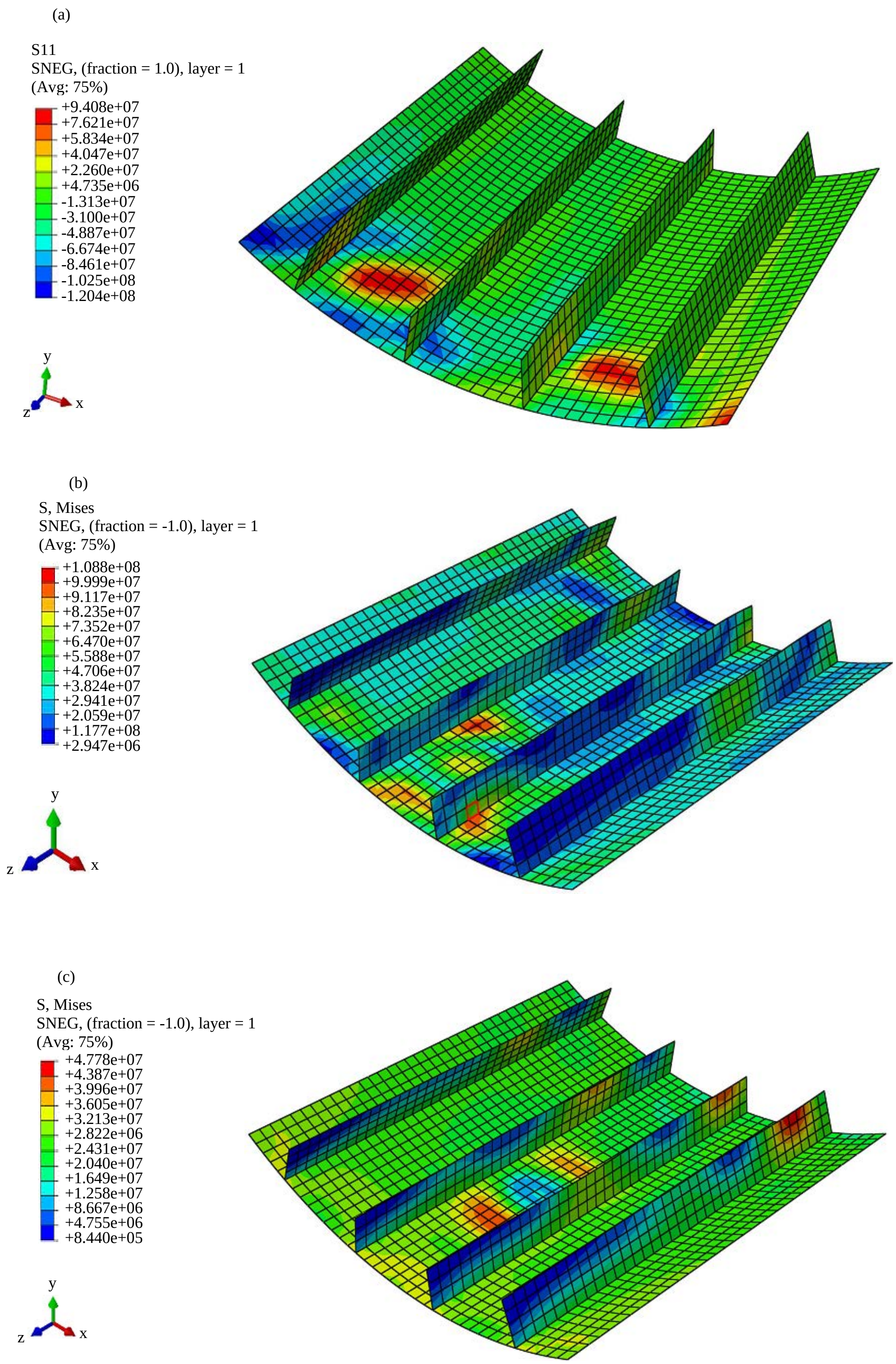

Fig. 7(a-f): Continue

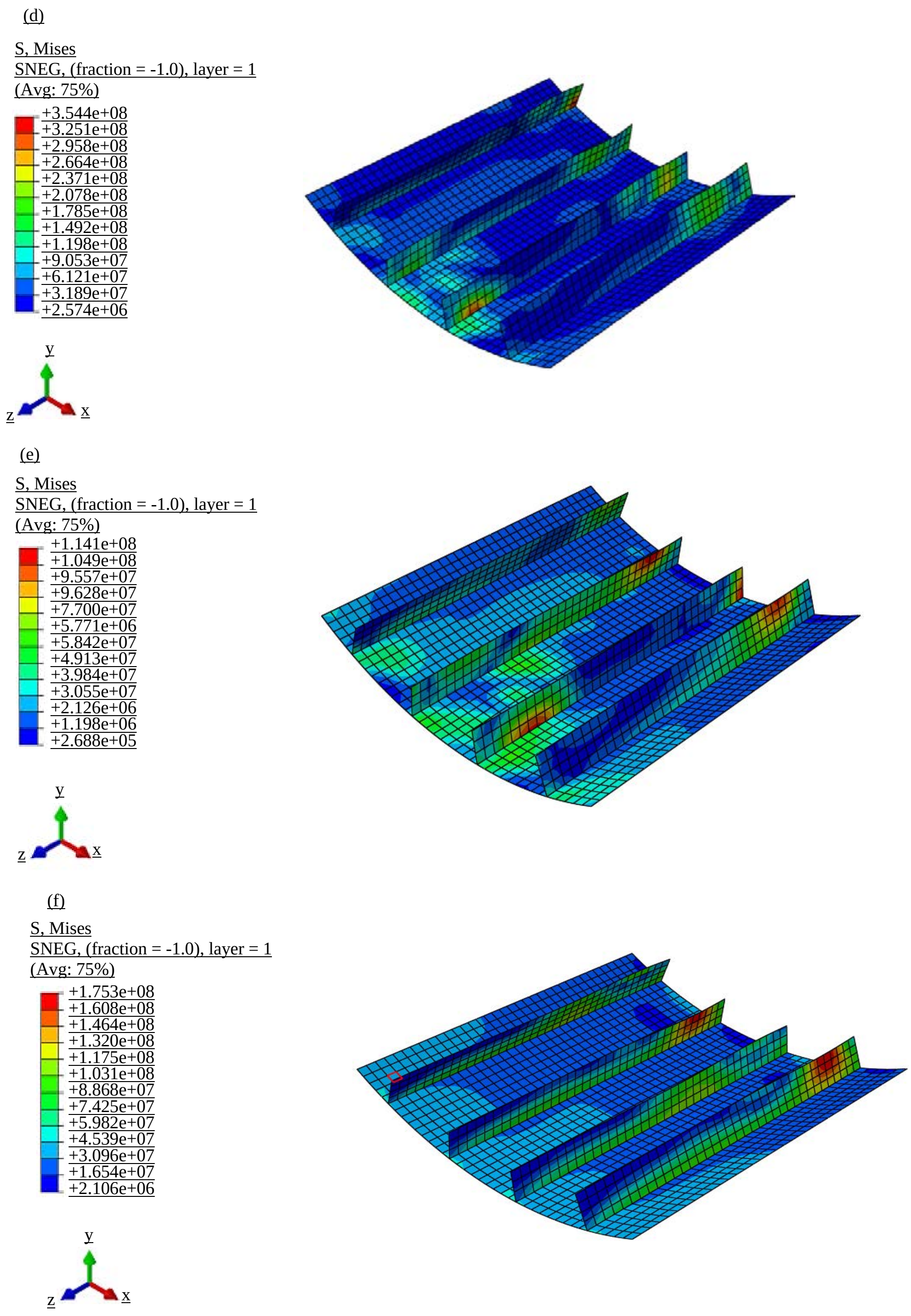

Fig. 7(a-f): The Von-mises stresses at different composite materials and ply orientations of the panel with 4 Straight stiffeners (a) Von-Mises stress with a Layup sequence (45/-45/90/0) s; The stiffeners and the skin are of the same configuration (E-glass (EG)); (b) Von-Mises stress with a Layup sequence (60/-30/90/0)s; The stiffeners and the skin are of the same configuration (E-glass (EG)); (c) Von-Mises stress with a Layup sequence (90/0/90/0) s; The stiffeners and the skin are of the same configuration (E-glass (EG)); (d) Von-Mises stress with a Layup sequence (45/-45/90/0) s; Composites with EG skin and CFC stiffener; (e) Von-Mises stress with a Layup sequence (60/-30/90/0) sand (f) Von-Mises stress with a Layup sequence (90/0/90/0) s; Composites with EG skin and CFC stiffener

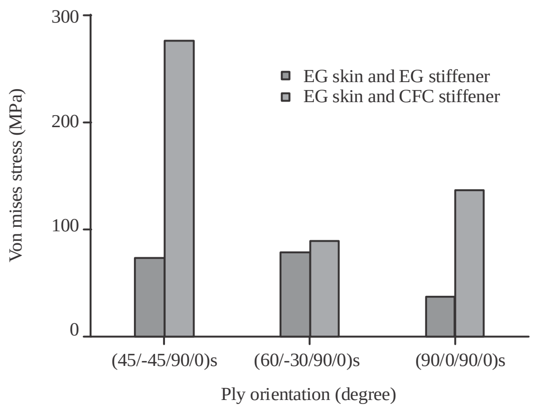

Fig. 8: Von Mises stresses obtained for the various ply layout sequences

composites EG skin and CFC stiffener, the low von Mises stress values below 47.78 MPa. The maximum von Mises stress in the stringer and near the edge of the loading.

The von Mises stresses obtained with various materials and different ply orientation ($(45/-45/90/0)_{s,}$ $(60/-30/90/0)_s$ and $(90/0/90/0)_s$) are plotted on the graph shown in Fig. 8. The results of the simulations also made it possible to identify the most unfavorable that is to say, the one for which the critical stress is the lowest. It corresponds to the combinations with ply orientation $(90/0/90/0)_s$ and Composites with EG skin and EG stiffener, the maximum von Mises stress with ply orientation $(45/-45/90/0)_s$ and composites with EG skin and CFC stiffener.

CONCLUSION

From the study of the three previous cases, we observed that when the skin and the stiffener are made of the Kevlar, the panel has better resistance compared to the other studied composites. The ply sequence $(60/-30/90/0)_S$ has the maximum critical buckling load. When the panel and the stiffener contain different composites, the Kevlar skin and the CFC stiffener has a maximum buckling load with orientation $(45/-45/90/0)_S$. After this research, we would like to study complex aircraft structures with both material and geometric imperfections and validate finite element models through experiments.

ACKNOWLEDGEMENTS

This research is sponsored by department of Industrial and Civil Sciences and Technologies, ENSA Tetouan, Abdelmalek Essaadi University and Structure LGCE, EST Sale-CED of Mohammadia Engineering School Mohammed V University, Rabat. These supports are gratefully acknowledged. The researchers also, wish to thank the anonymous reviewers for their thorough review of the study and their constructive advice.

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