Effect of Layers Arrangement on the response of Sandwich Composite Cantilever Plate

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Abstract:
A numerical study regarding stress, strain, and deflection of a composite plate is presented. The plate, consisting of three layers of Carbon-, Boron-, and Graphite-Epoxy, was fixed at one end and loaded at the other end in a conventional cantilever configuration. Six arrangements were examined and the spatial distribution of stress, strain, and deflection of the upper surface were calculated. Generally, it was found that the order, by which the three layers are arranged, has a great effect on the response of the plate and the maximum stiffness (in terms of deflection) is achieved when using Epoxy with Graphite-Carbon-Boron as the top-central-bottom layers of the plate.

Keywords: Sandwich, Composite, Plate, Stress, Strain, Deflection.

1. Introduction

Sandwich structures with laminated polymer matrix composite face sheets and lightweight core materials are being used increasingly as primary load-carrying components in aircraft and aerospace structures. In practical engineering design, deflections and stresses are very important criteria in reliability and serviceability evaluations of structures [1].

The word "composite" in composite material means that two or more materials are combined on macroscopic scale to form a useful material. Different materials can be combined on microscopic scale, such as in alloying but the resulting material is macroscopically homogenous [2].

A composite can be broadly defined as a combination of two or more materials, each of which has its own distinctive properties in which one of the materials, called the "reinforcing phase", is in the form of fibers, sheets or particles, and is embedded in the other material called the "matrix phase". The reinforcing material and the matrix material can be metal, ceramic or polymer. Typically reinforcing materials are strong with low densities, while the matrix is usually a ductile or tough material. If the composite is designed and fabricated correctly, it combines the strength of the reinforcement with the toughness of the matrix to achieve a combination of desirable properties not available in any single conventional material [3].

Basic ply or lamina of a composite structure can be considered as orthotropic with two principal material directions or natural axes–parallel and perpendicular to the direction of the filaments. By bonding these laminas to form a multi-lamina composite laminate, the designer has a material in which he can change the directional properties by changing the orientation of the various laminas, thus he is able to design a structure with a material that precisely matches the directional loading requirements at the considered point of the structure [3].
Composites are used to increase stiffness, toughness, compact strength and strength or dimensional stability, increase heat deflection and temperature mechanical damping, reduce permeability to gases and liquids and reduce costs, modify electrical properties (e.g. Increase electrical resistivity) and decrease thermal expansion and increase chemical and corrosion resistance [4].

Fiber composite offers many superior properties. Almost all high-strength / high stiffness materials fail because of the propagation of flaws. A fiber of such materials is inherently stronger than the bulk form because the size of a flaw is limited by the small diameter of the fiber [5].

The laminated composite plates used in the construction of aircraft, automobiles, ships and chemical vessels are sometimes provided with circular holes to meet functional or design requirements. In course of their service, laminates get fractured and developed cracks. These cracks and holes act as stress raisers [6].

One of the first investigations in this field was reported by Mawenya and Davies [7] who have developed a general formulation for a quadratic, isoparametric, multi layer plate element which permits the layers to deform locally and imposes no restriction upon the relative properties of the constituent layers of the plate. The formulation incorporates the effects of transverse shear deformation in each layer which is applicable to any arbitrary layered plate.

The finite element analysis of micromechanics treats a few of the fibers with the surrounding matrix [8]. The fibers and matrix are divided into many finite elements, and each element is homogenous and isotropic. Since the micromechanics finite element analysis requires the modeling of fibers and matrix separately, it requires an astronomical number of finite elements for a fiber-reinforced composite structure of real size.

Chang et al. [9] examined the feasibility of enhancing damage tolerance and durability of fiber composites through the design of microstructure by using three woven fabric-reinforced composite systems (carbon , Kevlar and Carbon Kevlar in Epoxy matrix). Enhancement in notched strength has been demonstrated by comparing the performance of composite with drilled and molded-in circular holes. Specimens with molded-in holes exhibited failure strengths, which were (2.7 –38.3 %) higher than those of drilled specimens.

The present work deals with composite sandwich plate of different arrangements of Carbon-, Boron-, and Graphite-Epoxy layers loaded as a cantilever beam.

2. Stress-strain Relation for Anisotropic Material

The elasticity matrix in the existence of an elastic potential or strain energy density function (Castigliano's theorem) implies that [10]:

$$
\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \\ \sigma_4 \\ \sigma_5 \\ \sigma_6 \end{Bmatrix} =
\begin{bmatrix}
C_{11} & C_{12} & C_{13} & C_{14} & C_{15} & C_{16} \\
 & C_{22} & C_{23} & C_{24} & C_{25} & C_{26} \\
 & & C_{33} & C_{34} & C_{35} & C_{36} \\
 & & & C_{44} & C_{45} & C_{46} \\
 & \text{symmetric} & & & C_{55} & C_{56} \\
 & & & & & C_{66}
\end{bmatrix}
\begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \varepsilon_3 \\ \varepsilon_4 \\ \varepsilon_5 \\ \varepsilon_6 \end{Bmatrix}
\tag{1}
$$

Where: $\varepsilon_1 = \dfrac{\partial u}{\partial x}, \quad \varepsilon_2 = \dfrac{\partial v}{\partial y}, \quad \varepsilon_3 = \dfrac{\partial w}{\partial z},$ (2)

$$\gamma_{12} = \frac{\partial v}{\partial x} + \frac{\partial u}{\partial y}, \quad \gamma_{23} = \frac{\partial v}{\partial z} + \frac{\partial w}{\partial z}, \quad \gamma_{13} = \frac{\partial u}{\partial z} + \frac{\partial w}{\partial x}$$

If there is one plane of material property symmetry, the stress-strain relation reduced to

$$
\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \\ \sigma_4 \\ \sigma_5 \\ \sigma_6 \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} & 0 & 0 & C_{16} \\ & C_{22} & C_{23} & 0 & 0 & C_{26} \\ & & C_{33} & 0 & 0 & C_{36} \\ & & & C_{44} & C_{45} & 0 \\ & \text{symmetric} & & & C_{55} & 0 \\ & & & & & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \varepsilon_3 \\ \varepsilon_4 \\ \varepsilon_5 \\ \varepsilon_6 \end{Bmatrix} \tag{3}
$$

If there are two orthogonal planes of material property symmetry for a material, symmetry will exist relative to a third mutually plane. The stress-strain relations in coordinates aligned with principal material directions, (parallel to the intersections of the three plans of material symmetry) are:

$$
\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \\ \sigma_4 \\ \sigma_5 \\ \sigma_6 \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} & & & \\ C_{12} & C_{22} & C_{23} & & \text{zeros} & \\ C_{13} & C_{23} & C_{33} & & & \\ & & & C_{44} & & \\ & \text{zeros} & & & C_{55} & \\ & & & & & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \varepsilon_3 \\ \varepsilon_4 \\ \varepsilon_5 \end{Bmatrix} \tag{4}
$$

and are said to define an orthotropic material. Note that there is no intersection between normal and shearing stresses which occurs in anisotropic materials (by virtue of the presence of, for example, C_{14}). Similarly, there is no intersection between shearing stresses and normal strains as well as none between shearing stresses and shearing strains in different planes. Note also that there are now only nine independent constants in the elasticity matrix.

If the laminate is thin, a line originally straight and perpendicular to the middle surface of the laminate which is assumed to remain straight and perpendicular to the middle surface when the laminate is extended and bent. Requiring the normal to the middle surface to remain straight and normal under deformation is equivalent to ignoring the shearing strains in planes perpendicular to the middle surface, that is, $\gamma_{xz} = \gamma_{yz} = 0$ where z is the direction of the normal to the middle surface in **Fig.(1)** [10]. In addition, the normal are presumed to have constant length so that the strain perpendicular to the middle surface is ignored as well, that is, $\varepsilon_z = 0$. The foregoing collection of assumptions of the behavior of the single layer that represents the laminate constitutes the familiar Kirchhoff-Love hypothesis for shells. Note that no restriction has been made to flat laminates.

The implications of the Kirchhoff or the Kirchhoff-Love hypothesis on the laminate displacements u, v, and w in the x-, y-, and *z*- directions are derived by using the laminate cross section in the *x-z* plane shown in **Fig.(1).** The displacement in the *x*-direction of point B from the undeformed to the deformed middle surface is (u_o).

Since line ABCD remains straight under deformation of the laminate,

$$
u_c = u_0 - z_c \beta \tag{5}
$$

But since, under deformation, line ABCD further remains perpendicular to the middle surface, β is the slope of the laminate middle surface in the x-direction, that is,

$$\beta = \frac{\partial w_o}{\partial x} \tag{6}$$

Then, the displacement, u, at any point z through the laminate thickness is

$$u = u_o - z\frac{\partial w_o}{\partial x} \tag{7}$$

By similar reasoning, the v, in the y-direction is

$$v = v_o - z\frac{\partial w_o}{\partial y} \tag{8}$$

The laminate strains have been reduced to $\varepsilon_x, \varepsilon_y$, and γ_{xy} by virtue of the Kirchhoff-Love hypothesis. That is, $\varepsilon_z = \gamma_{xz} = \gamma_{yz} = 0$. For small strains (linear elasticity), the remaining strains are defined (after substituting the displacements u and v from Esq. (7 and 8)) as follows:

$$\varepsilon_x = \frac{\partial u}{\partial x} = \frac{\partial u_o}{\partial x} - z\frac{\partial^2 w_o}{\partial x^2}$$
$$\varepsilon_y = \frac{\partial v}{\partial y} = \frac{\partial v_o}{\partial y} - z\frac{\partial^2 w_o}{\partial y^2} \tag{9}$$
$$\gamma_{xy} = \frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} = \frac{\partial u_o}{\partial y} + \frac{\partial v_o}{\partial x} - 2z\frac{\partial^2 w_o}{\partial x\partial y}$$

Thus, the Kirchhoff or Kirchhoff-Love hypothesis has been readily verified to imply a linear variation of strain through the laminate thickness. By substitution of the strain variation through the thickness, Eq. (9), in the stress-strain relations, the stresses in the kth layer can be expressed in terms of the laminate middle surface strains and curvatures as :

$$\begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{12} & Q_{22} & Q_{26} \\ Q_{16} & Q_{26} & Q_{66} \end{bmatrix} \left\{ \begin{Bmatrix} \varepsilon_{ox} \\ \varepsilon_{oy} \\ \gamma_{oxy} \end{Bmatrix} + z \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} \right\} \tag{10}$$

Since the Q_{ij} can be different for each layer of the laminate, the stress variation through the laminate thickness is not necessary linear, even though the strain variation is linear.

The resultant laminate forces acting on a laminate are obtained by integration of the stresses in each layer or lamina through the laminate thickness, as:

$$N_x = \int_{-h/2}^{h/2} \sigma_x dz \tag{11}$$

Actually, N_x is a force per unit length (width) of the cross section of the laminate as shown in **Fig. (2)** [2]. The entire collection of force resultants for an N-layered laminate is defined as:

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \int_{-h/2}^{h/2} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_k dz = \sum_{k=1}^{N} \int_{z_{k-1}}^{z_k} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_k dz \tag{12}$$

Where z_k and z_{k-1} are defined in **Fig. (3)** [10]. Note that $z_0 = -h/2$, these forces resultants do not depend on z after integration, but are functions of *x* and *y*, the coordinates in the plane of the laminate middle surface.

The integration indicated in Eq. (12) can be rearranged to take advantage of the fact that the stiffness matrix for a lamina is constant within the lamina. Thus, the stiffness matrix goes outside the integration over each layer, but is within the summation of force and moment resultants for each layer. When the lamina stress-strain relations, Eq. (10), are substituted,

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \sum_{k=1}^{N} \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{12} & Q_{22} & Q_{26} \\ Q_{16} & Q_{26} & Q_{66} \end{bmatrix}_k \left\{ \int_{z_{k-1}}^{z_k} \begin{Bmatrix} \varepsilon_{ox} \\ \varepsilon_{oy} \\ \gamma_{oxy} \end{Bmatrix} dz + \int_{z_{k-1}}^{z_k} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} zdz \right\} \tag{13}$$

However, it is noted that $\varepsilon_x^o, \varepsilon_y^o, \gamma_{xy}^o, k_x, k_y$, and k_{xy} are not functions of z but are middle surface values so can be removed from under the summation signs. Thus, Eq. (13) can be written as:

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_{ox} \\ \varepsilon_{oy} \\ \gamma_{oxy} \end{Bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} \tag{14}$$

Where: $\quad A_{ij} = \sum_{k=1}^{N} (Q_{ij})_k (z_k - z_{k-1}) \quad$ and: $\quad B_{ij} = \frac{1}{2} \sum_{k=1}^{N} (Q_{ij})_k (z_k^2 - z_{k-1}^2) \tag{15}$

3. Stress-Strain Relation for Principal Directions:

A laminate is two or more than two laminate bonded together to act as an integral structure element. The lamina principal directions are oriented to produce a structural element capable of resisting load in several directions. The stiffness of such a composite material configuration is obtained from the properties of constituent lamina. The procedure enabled analysis of laminates that have individual lamina with principal material directions oriented at arbitrary angles to the chosen or natural axes of the laminate consequence of the arbitrary orientations; the laminate may not have definable principal directions.

Since each lamina is a thin layer, one can treat a lamina as a plane stress problem as shown in **Fig. (4)**. Also, since each lamina is constructed by unidirectional fibers bounded by a metal or polymer matrix, it can be considered as an orthotropic material. Thus, the stress-strain relations on the principal axes can be expressed such that:

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_{12} \end{Bmatrix} = \begin{bmatrix} \dfrac{E_1}{1-\nu_{12}\nu_{21}} & \dfrac{\nu_{12}E_2}{1-\nu_{12}\nu_{21}} & 0 \\ \dfrac{\nu_{21}E_1}{1-\nu_{12}\nu_{21}} & \dfrac{E_2}{1-\nu_{12}\nu_{21}} & 0 \\ 0 & 0 & G_{12} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix} \qquad (16)$$

It is noteworthy to mention that only four of E_1 , E_2 , G_{12} , υ_{12} , and υ_{21} are independent material properties. Again, the shear modulus G_{12} corresponds to the engineering shear strain γ_{12} which is twice the tensor shear strain ε_{12}.

4. Formulation of the Problem

The rectangular sandwich flat plate studied in this research is composed of top, central, and bottom sheets or layers of thickness (1 mm). The composite plate is assumed to have a length of (L = 400mm), a width of (W = 200mm), and a total thickness of (H = 3mm), as well as loaded with a loading of (w = 10N/m), as shown in **Fig. (5)**, where coordinates are also shown. The sheets are considered as ordinary thin plates. The types of arrangements of the plate is shown in **Table (1)**, while the properties of the three types of layers are listed in **Table (2)** . All numerical work was achieved by the well-known software (ANSYS).

5. Results and Discussion

The normal stress (σ_x) was calculated at the upper surface of the plate and graphed as a function of the distance from the fixed end, see **Fig. (6)**. In this figure, the maximum value of the normal stress occurs at the fixed end for all arrangements of layers in the plate. Moreover, the plate of type (4) exhibits a maximum stress of (0.7MPa); a result which can be speculated by approximating the plate to a typical case of cantilever beam.

Away from the constrained end of plate, the normal stress decreases almost along the whole of plate length as shown in **Fig. (6)**. It is noteworthy to point out the occurrence of local maximum normal stress nearby the free end in almost all the types of the plate. The concentration of load, the viscoelastic effects and lateral effects are some of the dominating factors at the free end.

The variation of the normal stress (σ_y), calculated again at the upper surface of the plate, along the plate length is shown in **Fig. (7)**. It can be shown that this kind of stress fluctuates between tensile and compressive levels of relatively negligible values. However, the normal stress (σ_y) near the loaded end exhibits even more drastic changes than the normal stress(σ_x) does. This suggests that within the vicinity of the concentrated load, the plate is subject to conflicting types of behavior, namely viscoelasticity, orthotropy, and composite lamination. The maximum compressive stress occurring right under the load is well predictable.

The normal strain (ε_x) calculated at the same location, i.e. at the upper surface of the plate, is shown in **Fig. (8)**. In this figure, the plate type (3) exhibits a maximum strain of value (2.85×10^{-6}) at the constrained end. The trend of normal strain (ε_x) variation, **Fig. (8)**, is somewhat similar to that of the normal stress (σ_x), **Fig. (6)**.

Similar trends of (σ_y) and (ε_y) are also apparent in **Figs. (7** and **9)**. This suggests that the nondiagonal element in Eq. (19) is of minor effect on the strain.

The longitudinal displacement in x-direction is plotted in **Fig. (10)** , where it can be shown that maximum extension of $(8 \times 10^{-7} \text{m})$ in the fibers of the upper surface, occurs at the loaded end of the plate type(3).

Finally, the deflected plate is shown in **Fig. (11)** where the end deflection reaches up to (4×10^{-6}m) in plate type(3). That the maximum extension and deflection are occurring in the same type of plate, type (3), is attributed to the Boron-Epoxy layer being the central layer in this type.

6. Conclusions
It may be concluded that:
1- The maximum tensile stress, and hence the more likely critical, occurs in the plate type (4) at the fixed end.
2- The maximum compressive stress and strain, and hence the more likely critical, occurs in plate types (3 and 4) right under the load.
3- The maximum tensile strain occurs in plate types (3 and 4) at the fixed end.
4- The maximum stiffness of the plate is achieved in type (6).

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Nomenclature

A_{ij}	Extension stiffness elements
B_{ij}	Bending-extension coupling stiffness elements
C_{ij}	Element of elasticity matrix
N	Number of laminate layers
Q_{ij}	Transformed stress-strain relation from principal to laminate coordinates
Q_x, Q_y	Shear forces
N_i	Stress resultants
u,v,w	Displacements
u_o, v_o, w_o	Middle surface displacements
x,y,z	Rectangular coordinates
β	Slope of laminate middle surface
γ_{ij}	Shearing strain
γ_o	Middle surface shear strain
ε	Normal strain
σ	Normal stress
τ	Shearing stress
G_{ij}, E_{ij}	Higher – order laminate stiffnesses
K_{ij}	Transverse shear correction factors.
υ_{ij}	Poisson ś ratio giving the strain in j direction caused by a strain in the i direction.

Table (1): Types of sandwich plates.

Plate Type	Top Layer	Central Layer	Bottom Layer
1	Boron-Epoxy	Carbon-Epoxy	Graphite-Epoxy
2	Boron-Epoxy	Graphite-Epoxy	Carbon-Epoxy
3	Carbon-Epoxy	Boron-Epoxy	Graphite-Epoxy
4	Carbon-Epoxy	Graphite-Epoxy	Boron-Epoxy
5	Graphite-Epoxy	Boron-Epoxy	Carbon-Epoxy
6	Graphite-Epoxy	Carbon-Epoxy	Boron-Epoxy

Table (2): The properties of sandwich layers.

Layer Type	E_1 (GPa)	E_2 (GPa)	G_{12} (GPa)	G_{13} (GPa)	G_{23} (GPa)	υ_{12}
Boron-Epoxy[11]	211	21	6.6	--	--	0.21
Carbon-Epoxy[12]	54.5	54.5	3.1	--	--	0.08
Graphite-Epoxy[13]	132.3	10.7	5.6	3.6	3.6	0.24
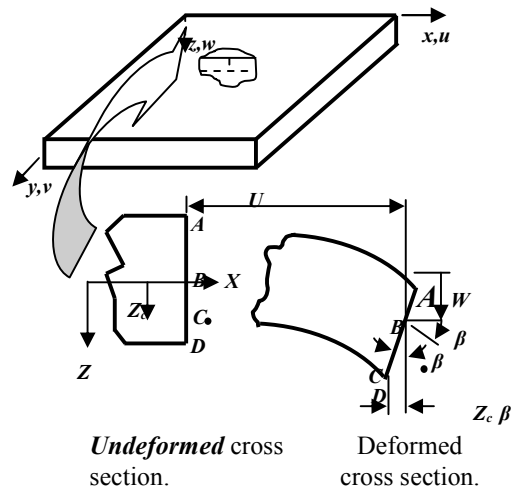
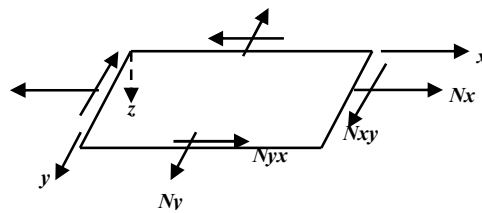Figure (1): Geometry of deformation in x-z

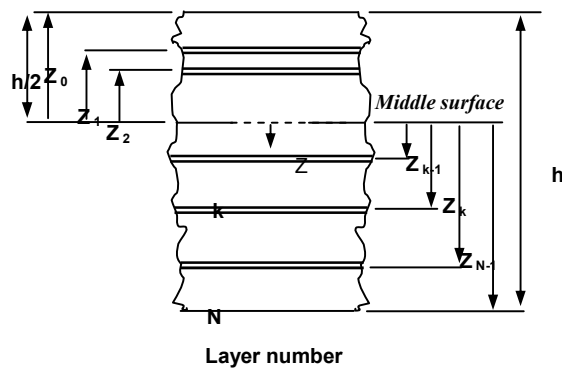Figure (2): In-plane on a flat laminate [2].

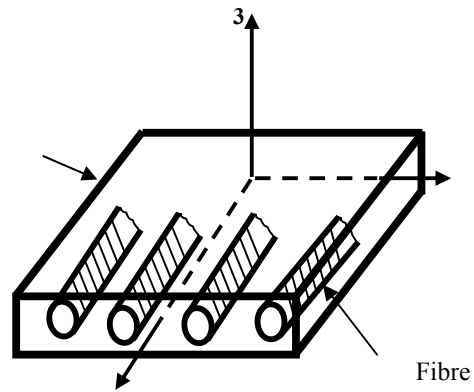Figure (3): Geometry of an n-layered laminate [10].
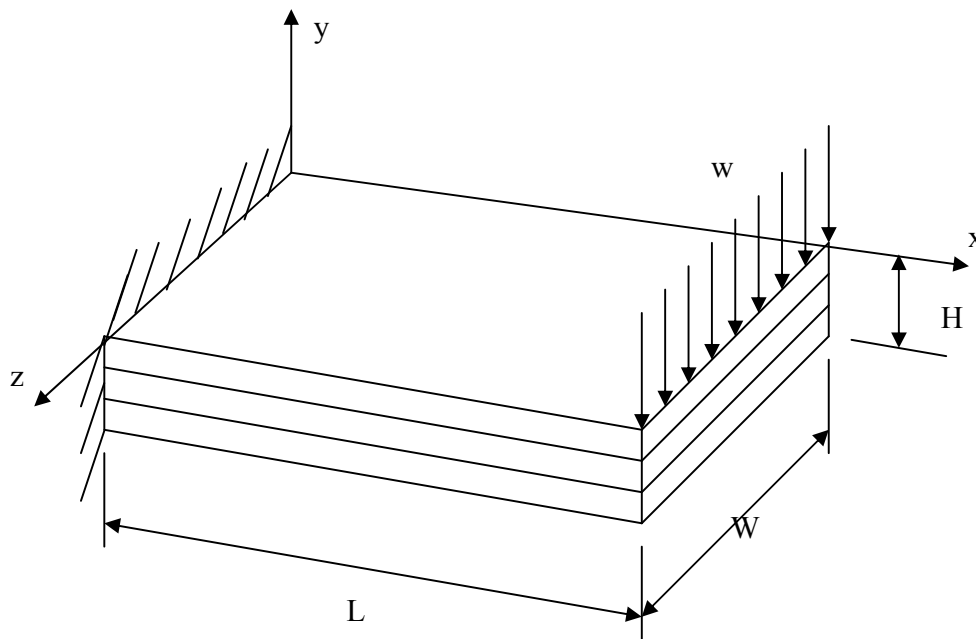
Figure (4): Unidirectional Reinforced Lamina

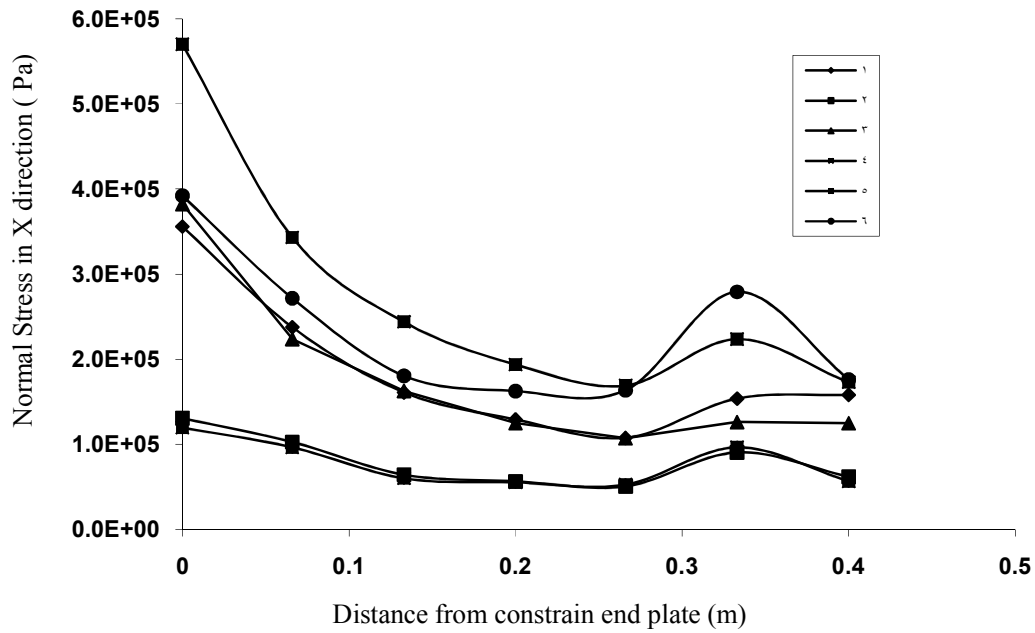Figure (5): Cantilever plate subjected to end load.

Figure (6): The relationship between the normal stress with the distance from the constrained end of the plate.
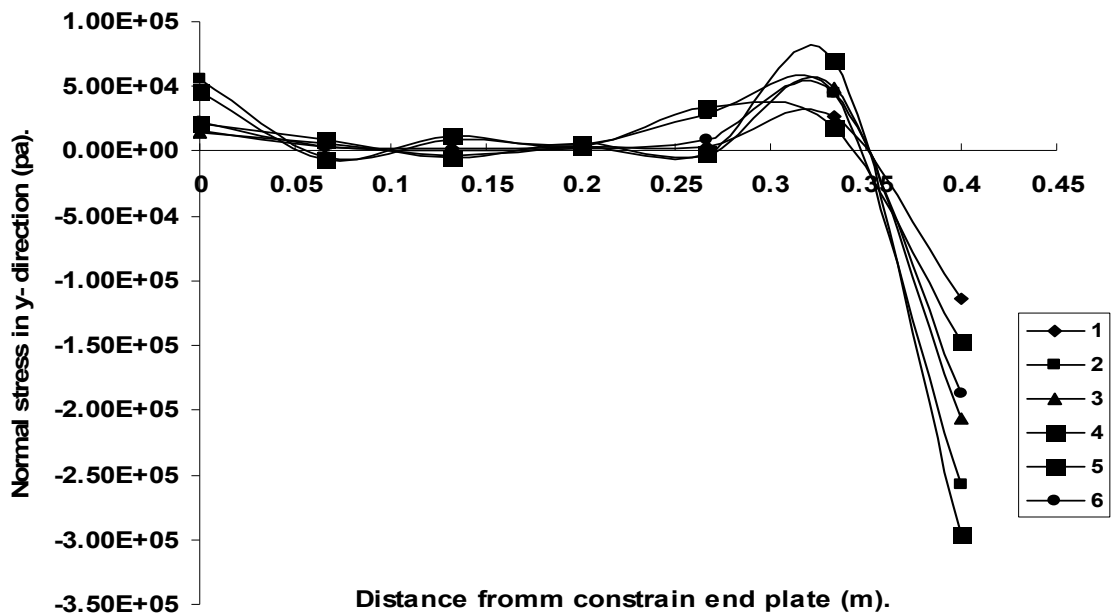

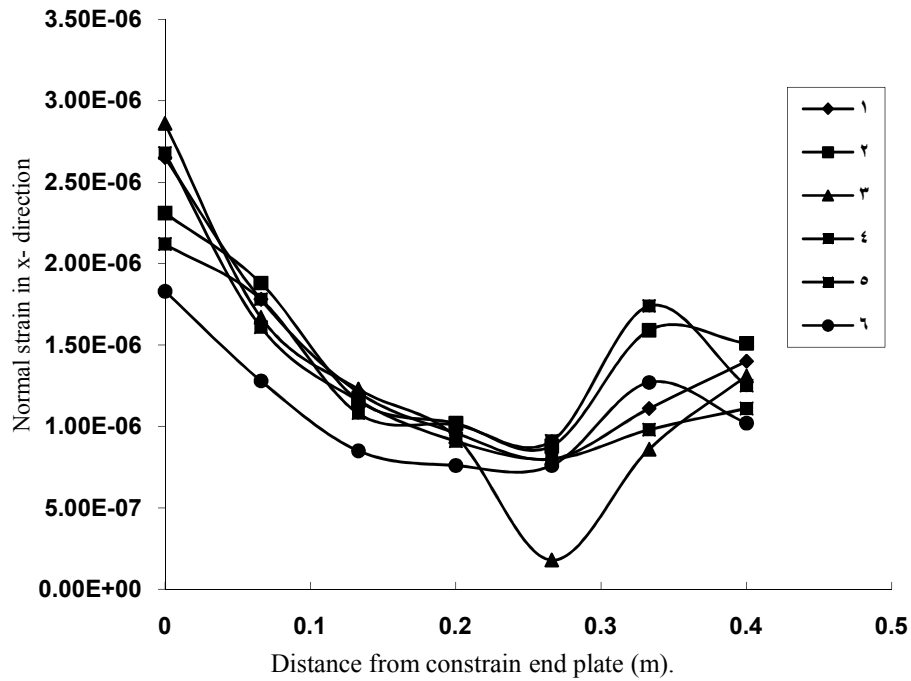Figure (7): The relationship between the normal stresses with the distance from the constrained end of the plate.

Figure (8): The relationship between the normal strain with the distance from the constrained end of the plate.
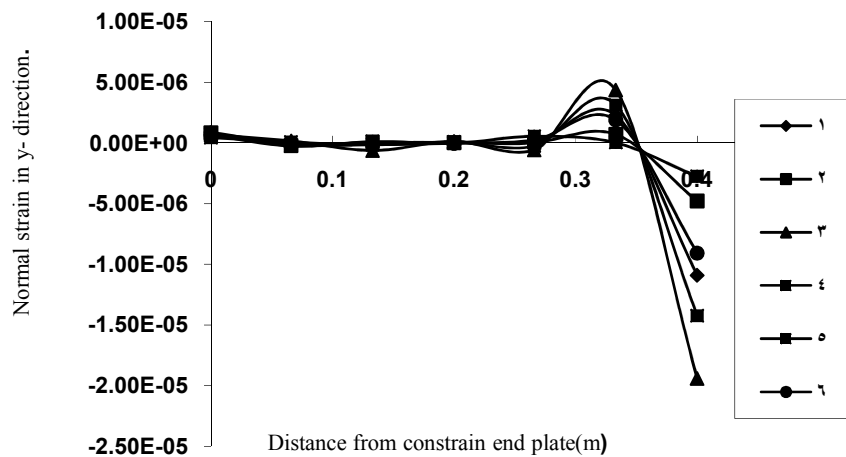

Figure (9): The relationship between the normal strain with the distance from the constrained end of the plate
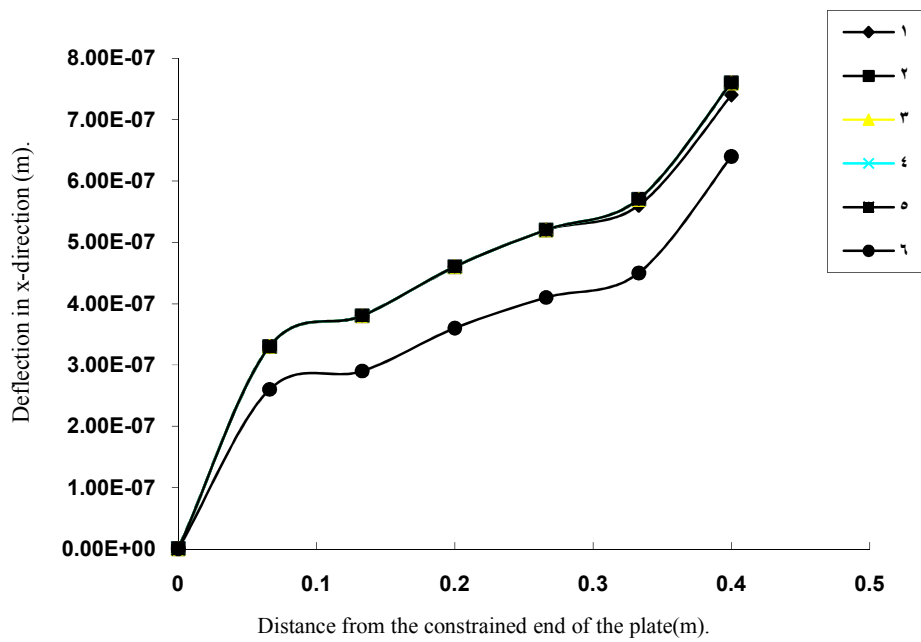
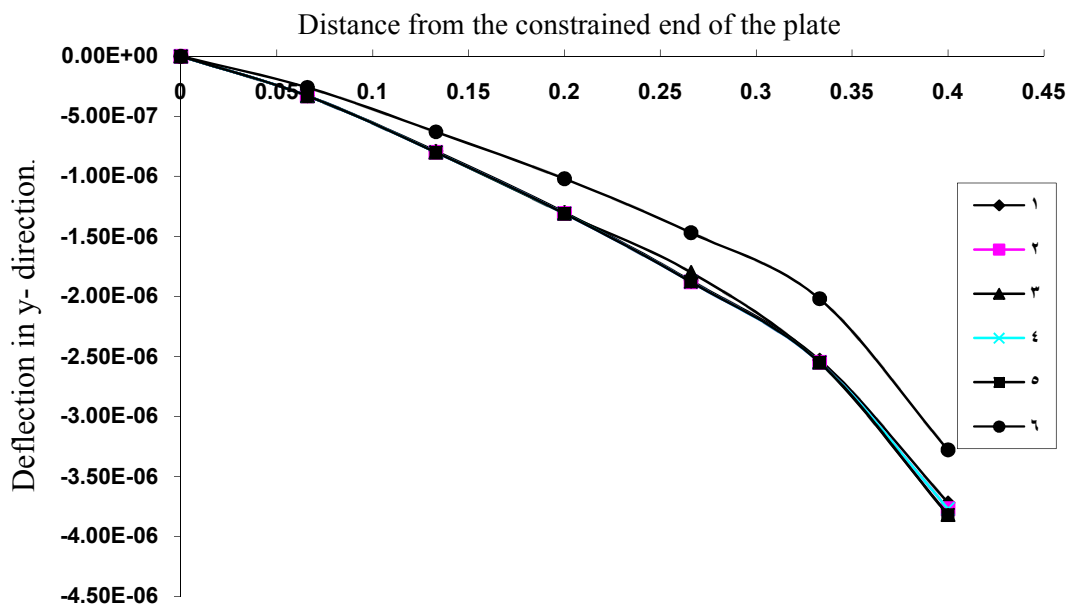Figure (10): The relationship between the deflection with the distancefrom the constrained end of the plate.

Figure (11): The relationship between the deflection with the distance from the constrained end of the plate.

تأثير ترتيب الطبقات على أستجابة الصفائح المركبة الرقائقية الكابولية

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الخلاصة:

يقدم البحث دراسة عددية لحساب الاجهاد، الانفعال والانحراف الحاصلة في الصفائح المركبة. تتألف الصفيحة من ثلاث طبقات من الكاربون _ايبوكسي ،البورون _ايبوكسي والكرافيت_ايبوكسي. وقد ثبتت الصفيحة من احدى جوانبها وحملت من الجانب المقابل بحمل مركز. تم اختبار ستة ترتيبات للطبقات وتم حساب الاجهاد، الانفعال والانحراف على سطح السطح العلوي للصفيحة. بصورة عامة وجد بان ترتيب الطبقات الثلاث يؤثر بشكل كبير على أستجابة الصفائح المركبة وأن أعظم جساءة (بدلالة الانحراف) تم الحصول عليها عندما استخدم ترتيب الايبوكسي مع الكرافيت-الكاربون-البورون كطبقات علوية-وسطى-سفلية للصفيحة. أجريت الدراسة العددية بأستخدام البرنامج المعروف (ANSYS) .