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Failure Analysis of Laminated Carbon/Epoxy Composite Using CLT and Altair Optistruct

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Abstract - In the areas like, aerospace, automobiles, construction and marine industries, composite materials are being used in very high proportion as it has unique properties compared to metals and woods such as light weight, high strength and strength to weight ratio.

This project topic deals with the three cases of symmetric carbon/epoxy composite material, having three layers of lamina, i) 0/45/0, ii) 0/60/0 iii) 0/90/0. A square plate of 100mm*100mm is taken and it is subjected to same mechanical loading conditions and analytical model of laminate is calculated using Classical Lamination Theory (CLT) to determine stresses and strain in laminate. Stresses in each laminate are calculated and their limiting values are judged by comparing the obtained results with the Maximum principle stresses. The finite element modelling is carried out in Hypermesh software, loading and boundary conditions are applied along with proper material and thickness assignment and theoretical stresses are validated by Altair-Optistruct solver with proper control cards and the failure analysis is carried out using Maximum principal strėss failurė thėory and Tsi-Hill failurė thėory. The comparison study shows error less than 1%.

The best combination of the lamina is chosen based on obtained results. The same combination is used for the outer door model and the analysis is carried for both conventional material and the composite material. The results for both are obtained in terms of FE results. The composite material proves to be better than the conventional material.

Key Words:

1. Symmetric carbon/epoxy composite material,

2. Classical lamination theory (CLT),

3. Maximum Principle stresses,

4. Hypermesh Software,

5. Altair-Optistruct solver.

1. INTRODUCTION

A Composite material is framed by joining at least two materials – regularly ones that have altogether different properties. The 2 materials cooperate to offer the composite one of kind properties. Be that as it may, inside the composite you'll effectively distinguish the different materials as they are doing not disintegrate or mix into each other. Most composites are produced using only two materials. one is that the network or cover. it encompasses and ties together strands or pieces of the contrary material, which is named the support.

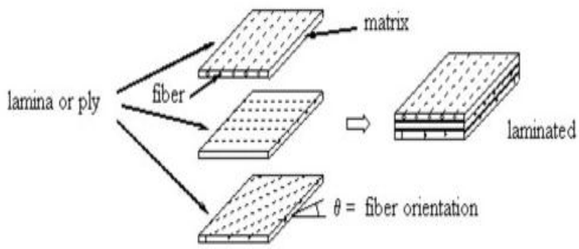


Figure 1 Laminated composite materials

Some composites are presently made utilizing carbon strands rather than glass. These materials are lighter and more grounded than fiberglass however costlier to give .they're used in airplane structures, car building, designing and costly athletic gear like golf clubs, hockey sticks and so forth since they show assortment of the great properties contrasted with metals which incorporate high solidarity to weight proportion, high solidness, consumption opposition, wear obstruction, and so on.[1,2]

The mechanical properties of composite covers rely upon the material of each layer, the amount of layers, the thickness of each layer and subsequently the fiber directions in each layer. The employ thicknesses are frequently foreordained and accordingly the utilize directions are generally limited to somewhat set of points due to assembling limitations.

2. PROBLEM STATEMENT

This project deals with the three cases of symmetric carbon/epoxy material subjected to same mechanical loading conditions and analytical calculations of same is calculated using Classical Lamination Theory(CLT) to find stresses in laminate. These stresses are validated by Altair-OptiStruct solver and therefore the failure analysis is administered using Maximum principal stress failure theory and Tsai-Hill failure theory. The three cases are as fallows[3]

CASE I: A [0/45/0] symmetric cross employ cover is exposed to bi-hub ductile heap of Nx=Ny=1000N/mm. Every lamina is 2mm thick and cross segment of overlay is 100mm×100mm

CASE II: A [0/60/0] symmetric cross employ cover is exposed to bi-hub ductile heap of Nx=Ny=1000N/mm. Every lamina is 2mm thick and cross segment of overlay is 100mm×100mm

CASE III: A [0/90/0] symmetric cross employ cover is exposed to bi-hub ductile heap of Nx=Ny=1000N/mm. Every lamina is 2mm thick and cross segment of overlay is 100mm×100mm.

3. ANALYTICAL CALCULATIONS FOR LAMINATES

The analytical procedure is administered for laminated supported Classical Lamination Theory (CLT) as discussed above.

Table-1: Material Properties of Carbon/epoxy Composite Material

Property	Carbon / Epoxy
Longitudinal Elastic Modulus, E_1	140 GPa
Transverse Elastic Modulus, E_2	10 GPa
In – plane Shear Modulus, G_{12}	5 MPa
Major Poisson's Ratio, ν_{12}	0.3
Ultimate Longitudinal Tensile Strength, $(\sigma_1^T)_{ult}$	1500 MPa
Ultimate Longitudinal Compressive Strength, $(\sigma_1^C)_{ult}$	1200 MPa
Ultimate Transverse Tensile Strength, $(\sigma_2^T)_{ult}$	50 MPa
Ultimate Transverse Compressive Strength, $(\sigma_2^C)_{ult}$	250 MPa
Ultimate In – Plane Shear Strength, $(\tau_{12})_{ult}$	70 MPa

3.1 FOR CASE I:

A [0/45/0] symmetric cross ply laminate is subjected to bi-axial tensile load of N_x=N_y=1000N/mm. Each lamina is 2mm thick and cross section of laminate is 100mm×100mm

0^0
45^0
0^0

Figure 2 laminate orientation for CASE I

Transformed reduced stiffness matrix for each lamina can be calculated by using equations of 3.7 and which gives the results as fallows and in symmetric laminate the stress generated in laminate 3 are equal to laminate 1

FAILURE ANALYSIS FOR CASE I:

1) Maximum Principal Stress Theory:

Failure occurs, if at least one of the following conditions in any layer is not satisfied

$$-\left(\sigma_1^{C}\right)_{ult} < \sigma_1 < \left(\sigma_1^{T}\right)_{ult} \text{ (or)}$$

$$-\left(\sigma_2^{C}\right)_{ult} < \sigma_2 < \left(\sigma_2^{T}\right)_{ult} \text{ (or)}$$

$$-\left(\tau_{12}\right)_{ult} < \tau_{12} < \left(\tau_{12}\right)_{ult}$$

$$-1200\times10^6 < 170.364\times10^6 < 1500\times10^6 ---1$$

$$-250\times10^6 < 120.834\times10^6 < 50\times10^6 ---2$$

$$-70\times10^6 < -44.860\times10^6 < 70\times10^6 ---3$$

In equation 2, the stress is not within the ultimate strength limits i.e;

$$\sigma_2 > \left(\sigma_2^{T}\right)_{ult}$$

Therefore lamina will fail

2) Tsai-Hill Failure Theory:

In its most general form the Tsai-Hill criterion for safe design as

$$\left[\frac{\sigma_1}{\left(\sigma_1^{T}\right)_{ult}}\right]^2 - \left[\frac{\sigma_1\sigma_2}{\left(\sigma_1^{T}\right)^2_{ult}}\right] + \left[\frac{\sigma_2}{\left(\sigma_2^{T}\right)_{ult}}\right]^2 + \left[\frac{\tau_{12}}{\left(\tau_{12}\right)_{ult}}\right]^2 < 1$$

$$\left(\frac{170.364}{1500}\right)^2 - \left(\frac{170.364\times120.834}{1500^2}\right) + \left(\frac{120.834}{50}\right)^2 + \left(\frac{-44.93}{70}\right)^2 < 1$$

$$6.253 < 1$$

It disobeys the Tsai-Hill failure Theory for safe design. So, the lamina will fail.

3.2 FOR CASE II:

A [0/60/0] symmetric cross ply laminate is subjected to bi-axial tensile load of N_x=N_y=1000N/mm. Each lamina is 2mm thick and cross section of laminate is 100mm×100mm

0^0
60^0
0^0

Figure 3 laminate orientation for CASE II

Transformed reduced stiffness matrix for each lamina can be calculated by using equations of 3.7 and which gives the results as fallows[4]

FAILURE ANALYSIS FOR CASE II:

1) Maximum Principal Stress Theory:

$$-1200\times10^6 < 199.838\times10^6 < 1500\times10^6 ---1$$

$$-250\times10^6 < 85.144\times10^6 < 50\times10^6 ---2$$

$$-70\times10^6 < -46.435\times10^6 < 70\times10^6 ---3$$

In equation 2, the stress is not within ultimate strength limits i.e;

$$\sigma_2 > \left(\sigma_2^{\,T}\right)_{ult}$$

Therefore lamina will fail

2) Tsai-Hill Failure Theory:

$$\left(\frac{199.838}{1500}\right)^2-\left(\frac{199.838\times85.144}{1500^2}\right)+\left(\frac{85.144}{50}\right)^2+\left(\frac{-46.435}{70}\right)^2<1$$

2.9091<1

It disobeys the Tsai-Hill failure Theory for safe design. So, the lamina will fail.

3.3 ANALYTICAL PROCEDURE FOR CASE III:

0⁰
90⁰
0⁰

Figure 4 laminate orientation for CASE III

A [0/90/0] symmetric cross ply laminate is subjected to bi-axial tensile load of N_x=N_y=1000N/mm. Each lamina is 2mm thick and cross section of laminate is 100mm×100mm

Transformed reduced stiffness matrix for each lamina can be calculated by using equations of 3.7 and which gives the results as fallows[6]

For Lamina 1 and 3: 0^0 Carbon/Epoxy

$$\left[\bar{Q}\right]_{0^0} = \begin{bmatrix} 140.905 & 3.019 & 0 \\ 3.019 & 10.065 & 0 \\ 0 & 0 & 5 \end{bmatrix} GPa$$

For lamina 2: 90^0 Carbon/Epoxy

$$\left[\bar{Q}\right]_{90^0} = \begin{bmatrix} 10.065 & 3.019 & 0 \\ 3.019 & 140.905 & 0 \\ 0 & 0 & 5 \end{bmatrix} GPa$$

The Stiffness matrices [A], [B] and [D]:

$$[A]=\begin{bmatrix} 583.753 & 18.116 & 0 \\ 18.116 & 322.070 & 0 \\ 0 & 0 & 30 \end{bmatrix}\times10^6\,Pa-m$$

[B]=0

$$[D]=\begin{bmatrix} 2449.063 & 54.342 & 0 \\ 54.342 & 268.397 & 0 \\ 0 & 0 & 90 \end{bmatrix} Pa-m^3$$

The laminate is subjected to bi-axial stresses N_x=N_y=1000 N/mm, which gives the results of mid plane strains and curvatures by using the equations 3.20

The strains in the laminate:

$$\begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix} = \begin{Bmatrix} 1.619 \\ 3.013 \\ 0 \end{Bmatrix}\times10^{-3}$$

The Axial stresses in each lamina:

$$\begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_{0^0} = \begin{Bmatrix} 237.221 \\ 35.213 \\ 0 \end{Bmatrix}\times10^6\,Pa$$

$$\begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_{90^0} = \begin{Bmatrix} 25.391 \\ 429.343 \\ 0 \end{Bmatrix}\times10^6\,Pa$$

The Principal Stresses in each lamina:

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix}_{0^0} = \begin{Bmatrix} 237.221 \\ 35.213 \\ 0 \end{Bmatrix}\times10^6\,Pa$$

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix}_{90^0} = \begin{Bmatrix} 429.434 \\ 25.391 \\ 0 \end{Bmatrix}\times10^6\,Pa$$

3.3(1) ANALYSIS FOR CASE III IN ALTAIR-OPTISTRUCT SOLVER:

For lamina 1&3:0 Carbon/Epoxy

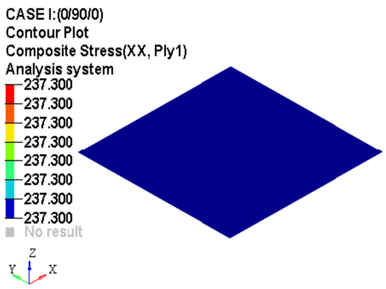


Figure 5 Principal Stress-1(σ1)

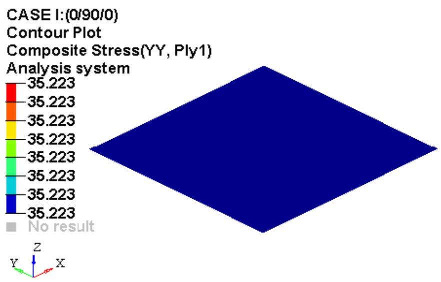


Figure 6 Principal Stress-2(σ2)

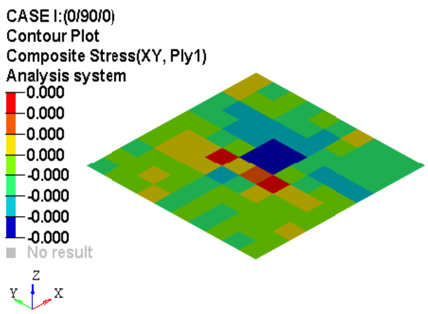


Figure 7 Shear Stress-12(τ12)

For lamina 2: 90⁰ Carbon/Epoxy

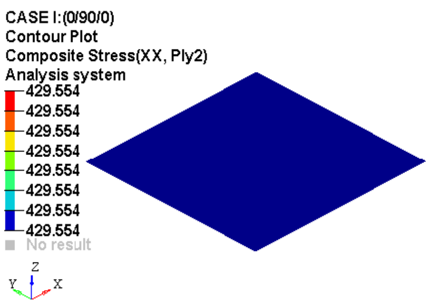


Figure 8 Principal Stress-1(σ1)

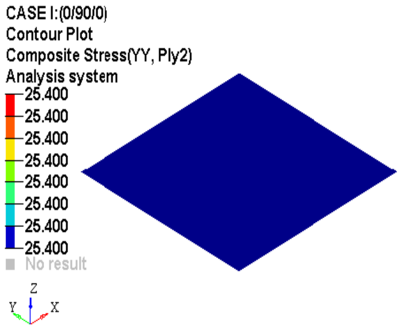


Figure 9 Principal Stress-2(σ2)

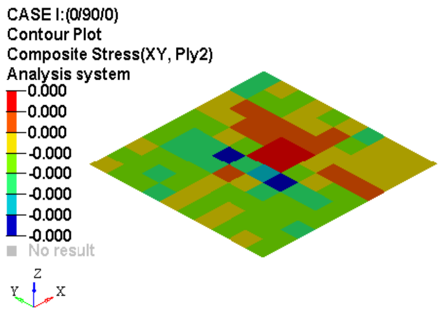


Figure 10 Shear Stress-12(τ12)

FAILURE ANALYSIS FOR CASE III:

1) Maximum Principal Stress Theory:

For lamina 1&3:

$$-1200\times10^{6} < 237.221\times10^{6} < 1500\times10^{6}$$
$$-250\times10^{6} < 35.213\times10^{6} < 50\times10^{6}$$
$$-70\times10^{6} < 0 < 70\times10^{6}$$

All stresses lie within the ultimate strength limits. Therefore lamina 1&3 will safe

Strength Ratio (SR)=1.419

For lamina 2:

$$-1200\times10^{6} < 429.434\times10^{6} < 1500\times10^{6}$$
$$-250\times10^{6} < 25.391\times10^{6} < 50\times10^{6}$$
$$-70\times10^{6} < 0 < 70\times10^{6}$$

All stresses lie within the ultimate strength limits. Therefore lamina 2 will safe

Strength Ratio (SR)=1.969

2) Tsai-Hill Failure Theory:

For lamina 1&3:

$$\left(\frac{237.221}{1500}\right)^2-\left(\frac{237.221\times 35.213}{1500^2}\right)+\left(\frac{35.213}{50}\right)^2+\left(\frac{0}{70}\right)^2<1$$

It obeys the Tsai-Hill failure Theory for safe design. So, the lamina 1&3 will safe

Strength Ratio(SR)=1.390

For lamina 2:

$$\left(\frac{429.434}{1500}\right)^2-\left(\frac{429.434\times 25.391}{1500^2}\right)+\left(\frac{25.391}{50}\right)^2+\left(\frac{0}{70}\right)^2<1$$

It obeys the Tsai-Hill failure Theory for safe design. So, the lamina 2 will safe

Strength Ratio (SR) = 1.728

From the above three cases we observed that keeping outer layer orientation i.e. 00 is constant and increasing the mid-layer orientations from 450 to 900. The principal stresses in x-direction increases, although they're within the range of ultimate strengths in x-direction. But, the principal stresses in y-direction are decreasing and therefore the stresses are quite the last word strengths in y-direction just in case I and Case II so, the laminates are failed in these two cases supported failure theories. just in case III, all the stresses are within the range of ultimate strengths and this one is that the safe and best case from above three cases. So, finally concluded that Case III is that the optimum among the three cases and this material is use for design of radial impeller rather than generally using material i.e. Aluminum alloy.[7]

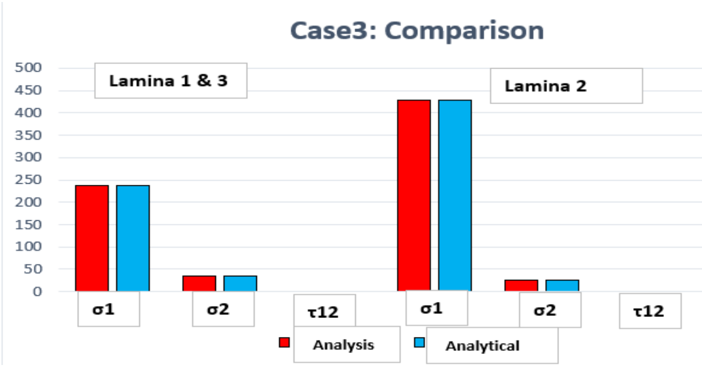
4. RESULTS AND ANALYSIS

The analytical values and analysis values of stresses are tabulated below and evaluate the percentage error to access the capability of Hypermesh Optistruct Solver for analysis of composite materials. The percentage error of analysis value and analytical value is calculated by using following formula

$$\%Error=\frac{\left|\left(Analytical\quad value-Analysis\quad value\right)\right|}{\left|Analytical\quad value\right|}\times 100$$

Table- 2: Comparison of Stresses and Calculate Error for CASE III

Lamina no.	Stress	Analysis Value (MPa)	Analytical Value (MPa)	%Error
Lamina 1&3 (0^0 Carbon/Epoxy)	σ_1	237.30	237.221	0.0332
	σ_2	35.223	35.213	0.0284
	τ_{12}	0	0	0
Lamina 2 (90^0 Carbon/Epoxy)	σ_1	429.554	429.434	0.0279
	σ_2	25.391	25.391	0
	τ_{12}	0	0	0



5. CONCLUSION AND FUTURE SCOPE

Based on the Analytical and Simulation results on laminated composites, it can be concluded that,

- Almost 100% Accuracy in the results is obtained. The results obtained from simulations and calculations are in good agreement with each other with error of less than 1%
- This option can be used to check the various combinations of composites based on their properties to obtain the optimum composite material.
- For various Automotive components these types of composites can be used and good results can be obtained based on their applications.
- Weight reduction, High strength, Good efficiency and good styling can be obtained.
- The use of CAE tool reduces the time of calculation and different experiments can be carried out.
- For the same loading and boundary conditions on the door panel, stresses induced in the composite material(524MPa) are less than that of conventional material(630MPa).
- Even the displacement results are better than that of conventional steel material. And hence proves to be enough stiff to withstand the static and dynamic loading conditions.
- Further same composites can be checked for impact loading, studying crack propagation using LS Dyna.

- Thermal properties are not considered and in future micro level analysis can be done by considering non-local continuum mechanics.

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