Modelling and Finite Element Analysis of an Aircraft Wing using Composite Laminates

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Abstract. The use of finite element analysis is increasingly gaining popularity in the design and analysis of aircraft structures. Also, composite materials in the manufacturing and design of aircraft structures are increasing due to their exceptional properties of high stiffness to weight ratio. Applications include Composite structures are made up of laminates with different fibre orientations, which adversely affects the property of the composites formed. For this study, the three main parts of the aircraft wing are considered for designing: the spars, the ribs, and the aircraft's skin. Two different wing models are designed. For Model 1, the skin is assigned Titanium alloy as a material, and for Model 2, CFRP (Carbon Fibre Reinforced Polymer) with resin epoxy is used as the skin material. The ribs are made of structural steel and the spars of titanium alloy for both models. This paper aims to compare the materials mentioned above to find the optimum strength to weight ratio. It was found that, for a given uniform load on the bottom of the aircraft's skin, a weight reduction of 2.37% was observed in favour of Model 2, the deformation was reduced by 51 per cent, and the von-Mises stress was reduced by ≈ 85%. This paper's results can be used in manufacturing structural components of UAV, especially the wings due to lighter weight and more flexibility and durability than standard titanium alloys and the significant role played by the orientation of the plies in deciding the strength of the composite.

1. Introduction

Aircraft wings contribute a significant amount to the payload of any aircraft. It consists of many panels, and it is designed simultaneously to achieve an optimum design [1, 2]. A good wing design involves the aerodynamic, structural design, as well as material selection [3]. The aerofoil wing shape is responsible for the generation of lift caused by the aircraft's forward velocity [4, 5, 6]. To increase the efficiency of an aircraft, it is essential to reduce the wing's weight by a significant amount, and it is a well-explored topic in the field of aircraft. Weight is a critical factor in aircraft design, and increasing stiffness, although it may reduce flutter onset, it does so at the expense of weight [7, 8, 9]. High strength and lightweight are two primary attributes of any aircraft wing [10, 11]. Materials used for aircraft components impact structural efficiency, payload energy consumption, cost, and reliability [1, 7, 12]. It has been observed that the most efficient way to increase the structural efficiency is by reducing the density, which in turn implies using lightweight materials [13, 14]. Although certain alloys like Aluminium and Titanium help in a significant weight reduction of aircraft materials and
have seen application in the aerospace industry for years, using composites helps with a massive weight reduction, thus facilitating the increased efficiency \[7, 13, 15, 16\]. A composite structure has laminates stacked together with different fibre orientation angles \[13\]. It consists of two or more constituent materials such as the elastic modulus or strength of each constituent is different \[17, 18, 19, 20, 21\]. The use of composites also provides better thermal stability and resistance to corrosion.

The use of composite minimizes the structural weight, although it increases the manufacturing cost as a drawback \[22\]. There are various kinds of laminates; a unidirectional laminate is one with all the filaments arranged in one direction, a cross-ply laminate has layers of unidirectional laminates arranged at right angles with each other \[23\]. Composites like Carbon fibre are gaining popularity in aerospace, although, Aluminium is still leading the aircraft market \[14, 24\]. For example, A380 is a super-sized aircraft whose composition is 64\% made of aluminium alloys. Similarly, for Boeing 777, another super-sized aircraft constitutes 70\%. Aluminium in composition \[7\]. Analysis of laminate structures requires the comprehension of individual stresses and strains of the laminate, and therefore, the analysis of laminates requires the understanding of individual laminates \[25, 26\]. Kakur N et al. study the ILSS and flexural properties of CFRP and GFRP for five different laminate orientations and conclude through both experimental and theoretical investigation that modulus and flexural strengths are more significant 0° laminates. In contrast, the flexural strain is greater for \(45°/45°/45°\) laminate layup \[27\]. Ernie I Basri et al. talk about composite ply orientation in the NACA 4415 wing, the structural analysis studies the layer-wise FEM for a Carbon Fiber Reinforced Polymer laminated composite in the different orientation of ply combination. The paper discusses how a specific ply orientation increases an aircraft wing \[28\]. Muhammad Amir Mirza et al. performed structural analysis over a three-dimensional wing of NACA 23015, which is identical to A320. The paper concludes that a wing with ribs at the root and the tip of the wing shows lesser deformation as compared to a wing that contains spar and wings \[2\].

For the current research, we have decided to optimize the weight of the wings with a NACA 23015 wing profile due to its similarities with A320 by changing the material composition and ply orientation. Airbus A320 is one of the most prominent and successful aeroplanes from the jetliner's family. Constructed after A300 and A310, A320 is the most commercially popular narrow-body airliner \[29\]. This aircraft was chosen because it is one of the most used commercial planes. This study compares the uses of Carbon fibre (395 GPa) with resin epoxy (CFRP) and Titanium alloy on the wing skin. A weight reduction was observed with the composite used compared to the alloy. Modelling of the aircraft wing was done using SolidWorks 2016. The composite layup is done on the FEA software ANSYS ACP PrePost module. Various layups of the composite material (CFRP) are done to find a configuration with the least total deformation and equivalent (Von-Mises) stress. The layups are done following the Classical Laminate Theory (CLT).

The following study employs a definite structure employing sections and subsections. The methodology is discussed in the next section, in which the geometrical configuration, material specifications, mesh generation, loading conditions, and finally, the layup of composite materials are discussed followed by the results and discussions in section 3 followed by the conclusion in section 4.

2. Methodology

2.1 Geometrical Configuration

The aircraft wing design is a complicated iterative procedure. Besides, the calculations are commonly repeated quite a lot of times. For this wing's design, two spar wing design is considered suitable as this configuration is used for most commercial aircraft construction. The NACA profile 23015 is used, referencing the Airbus A320. The model is primarily composed of two spars, ten ribs, and one part of the skin. The design was sketched and assembled in SolidWorks 2016, and the FEM process was done in ANSYS Workbench. The wing's span is set to 4.5 m, and the chord length of the wing is 1 m. The
root and tip chord thickness is 0.04 m, and all other rib thickness is kept to 0.02 m. The design specifications are depicted in Figure 1 and Figure 2.

![Figure 1. Internal structure of the wing](image1)

![Figure 2. External Structure of the wing](image2)

### 2.2 Material Specifications

The specifications of the different parts of the model are mentioned below. The construction of composite material is done using Carbon Fibre (395 GPa) and Resin Epoxy as the global drop-off material. The details of the properties of the materials used are given in the tables below:

| Properties            | Values  | Unit   |
|-----------------------|---------|--------|
| Density               | 4620    | Kg/m³  |
| Young’s Modulus       | 9.6E+10 | Pa     |
| Poisson’s ratio       | 0.36    |        |
| Tensile Yield Strength| 9.3E+08 | Pa     |
| Tensile Ultimate Strength| 1.07E+09 | Pa    |

| Properties            | Values  | Unit   |
|-----------------------|---------|--------|
| Density               | 7850    | Kg/m³  |
| Young’s Modulus       | 2E+11   | Pa     |
| Poisson’s ratio       | 0.3     |        |
| Tensile Yield Strength| 2.5E+08 | Pa     |
| Tensile Ultimate Strength| 4.6E+09 | Pa    |
Table 3. Carbon Fibre 395 GPa (fibre only)

| Properties                        | Values | Unit  |
|-----------------------------------|--------|-------|
| Density                           | 1800   | Kg/m³ |
| Young's Modulus X direction       | 3.95E+11| Pa    |
| Young's Modulus Y direction       | 6E+09  | Pa    |
| Young's Modulus Z direction       | 6E+09  | Pa    |
| Poisson's Ratio XY                | 0.2    |       |
| Poisson's Ratio YZ                | 0.4    |       |
| Poisson's Ratio XZ                | 0.2    |       |
| Shear Modulus XY                  | 8E+09  | Pa    |
| Shear Modulus YZ                  | 2.1429E+09| Pa |
| Shear Modulus XZ                  | 8E+09  | Pa    |

Table 4. Mechanical Properties of Model 1 and 2, Elements and Materials

| Sr. No | Model | Ribs            | Spars | Skin                        | Weight (in kgs) |
|--------|-------|-----------------|-------|-----------------------------|-----------------|
| 1      | Model 1 | Structural Steel | Titanium | Titanium Alloy            | 540.38         |
| 2      | Model 2 | Structural Steel | Titanium | Carbon Fibre(395 GPa) Epoxy | 527.56         |

2.3 Mesh generation

Mesh generation is of prime importance in FE analysis considering the results generated are dependent on the quality of the mesh generated. Derived quantities such as stresses and strains are calculated at the element integration points and extrapolated to the nodes. The value described in ANSYS Workbench is the average value at the node, which is based on all elements connected to that node, and the difference in averaged and un-averaged results is a mesh discretization error indication. As mentioned by Kanesan et al., the aspect ratio of the mesh generated should also be less. Since the loading will affect the skin directly, meshing is much more refined for the skin as compared to other parts of the model to save computational time. Also, the mesh should accurately represent the geometry and computational loads and domain. [29]. Some important steps were undertaken to ensure a better mesh generation for the study:

1. Face meshing was applied to the top and bottom half of the skin, and mesh defeaturing was set to yes for the whole mesh to remove unwanted parts in the model, thereby reducing calculation time.
2. Body Sizing was applied to the spars and the ribs
3. Capture proximity and capture curvature were set to Yes

The total number of nodes and elements generated was 131978 and 54019.
2.4 Boundary and Loading Conditions

2.4.1 Boundary Conditions: As far as the loading conditions are concerned, the fuselage was not included in the model. The boundary conditions consider that an aircraft's wing model is connected to the fuselage, which is not included in the study. The wing acts as a cantilever beam connected to the fuselage [23]. Hence the face attached to the fuselage is constrained in all the DOFs i.e. UX = UY = UZ = ROTX = ROTY = ROTZ =0.

2.4.2 Loading Conditions: The loading condition for this study is assumed to be of a wing torque box under the influence of aerodynamic loading [30]. The loads’ calculation is done by assuming a uniformly distributed pressure load applied in the vertical direction on the bottom part of the wing. A vertical pressure of 1963.770304 Pa is applied to the bottom part of the skin. The aerodynamic loading conditions can be calculated using the formula given in eq. (1).

\[ \frac{L}{(0.4 \times W)} = FOS \times \eta_{\text{max}} \]  
\[ L/(0.4 \times W) = FOS \times \eta_{\text{max}} \]  \( L \) = lift capacity, \( W \) = weight of the aircraft.

Referencing from [2] FOS is taken to be 1.5

W=540.38×9.81=5301.1278N  

\[ W = 540.38 \times 9.81 = 5301.1278 \text{N} \] (2)
In general, 80% of the total lift load is assumed to be uniformly distributed on the bottom part of the wing with the rest of the 20% being attached to the fuselage. This factors in the value of 0.8 in eq. (3).

\[ \therefore L_{\text{wing}} = 3.5 \times 1.5 \times 0.4 \times 5301.1278 \times 0.8 = 8905.894704 \text{N} \]  
(3)

\[ \therefore P = (L_{\text{wing}} \div S) = (8905.894704 \div 4.5351) = 1963.770304 \text{Pa} \]  
(4)

(S = Surface Area of the wing part where the load is applied)

The loading and boundary conditions are shown in the fig. 5 below:

![Figure 5. Boundary and Loading Conditions on the wing](image)

2.5 Modelling of composite layers for the Skin

The composite layup is done on the FEA software ANSYS ACP Pre module. Various layups of the composite material (CFRP) are done to find a configuration with least total deformation, equivalent stress. The layups are done by the classical laminate theory (CLT). Using CLT, we can find the structural response of a composite laminate given an arbitrary set of loads and a stacking sequence [31].

Classical Lamination Theory predicts the behavior of the laminate within the framework of the following assumptions [32]:

1. Each laminate layer is assumed to be orthotropic as well as quasi-homogeneous.
2. The laminate is thin with the lateral dimensions significantly larger than the thickness and that the loads applied on the laminate is only in in-plane directions.
3. Out of plane displacements are assumed to be insignificant compared to the laminate thickness, and in-plane displacements vary linearly throughout the laminate thickness.
4. All displacements are insignificant compared to the laminate thickness, and perpendicular distances from the middle surface stay constant.
5. The relations between applied forces and moments at an arbitrary point (x, y), and the resultant mid-plane curvatures and strains, can be summarized as follows in fig. 6. The A/B/B/D matrix in brackets is the laminate stiffness matrix, and its inverse is the laminate compliance matrix.

Relation between applied forces and moments, and the resultant mid-plane curvatures and strains:

\[
\begin{bmatrix}
N \\
M
\end{bmatrix} =
\begin{bmatrix}
A & B \\
B & D
\end{bmatrix}
\begin{bmatrix}
\phi_1 \\
\phi_2 \\
\kappa
\end{bmatrix}
\]
Figure 6. Ply arrangement of various arrangements in a composite layup made in Solidworks [33]

3. Methodology

3.1. Validation of the paper with previously published results

The model in this paper is validated with Zakuan, M.A.M.B.M [2]. For this validation, aluminium alloy is used in the ribs and wing skin, and titanium alloy is used for primary and secondary spars.
As seen from the table, the results in total deformation, equivalent stress between the obtained results and the existing data are relatively close to each other, and hence the said model can be used for further optimization process using composites. The slight differences in the results are due to differences in meshing parameters and the design of the wing used for this study being optimized to reduce deformation values. Hence as can be inferred from the table, this design can be further used to optimize the wing model using CFRP with different ply orientations.

### 3.2 Static Structural Analysis of Model 1 and Model 2

Static Structural analysis of Model 1 and 2 is done to find the total deformation and the Von-Mises stress plots.

#### 3.2.1 Analysis of Model 1:

Model 1 has titanium alloy as the added material for the skin of the wing as well as the spars and structural steel as the material for the ribs.
As seen in the figure, the wing acts as a cantilever beam with the root attached to the fuselage. The spar primarily carries the model's load. They are the primary members that provide structural support for the wing against twisting (torsion) and upward bending forces while the wing generates lift. On the other hand, ribs, the internal structural parts running fore and aft in the wing, provide the airfoil shape to the structure. The maximum deformation, in this case, occurs with the value of 0.050676 m at the tip of the wing structure, while the minimum deformation is at the root part of the wing structure.

Most of the stress concentration occurs at the ribs’ tip, which means these parts are most prone to damage and have a reduced life compared to the other components. Also, the stress values gradually increase as we move away from the part attached to the fuselage. This is to be expected as the wing behaves similar to a cantilever beam. The maximum stress value is 2.9099e+8 Pa.

3.2.2. Analysis of Model 2: Model 2 employs the use of composite layups with the intent to reduce deformation, stress values, and reducing the weight of the wing. A laminate code is generally used to identify the lamina stacking sequence in a laminate. Knowing the code assists with comparing with the form of [A], [B], and [D] before analysis [8]. This study has employed different types of lamina structures such as symmetric, anti-symmetric, and quasi-isotropic laminates to find which laminate orientation is best suited in terms of strength, stiffness, and bending matrices.

Model 2.A: A ply sequence of [0/90/0/90/0]s is applied on the skin of the wing made up of Carbon-fibre (395 GPa)-epoxy material. In such a symmetric layup, the [Bi,j] matrix becomes zero which means bending and extension are uncoupled, allowing simplified solution procedures.
Table 6. Laminate Engineering Constants of [0/90/0/90/0]

| Constant                        | Value     |
|---------------------------------|-----------|
| Flexural Laminate Shear Stiffness | 8000      |
| Flexural Laminate Stiffness E1   | 314262.4  |
| Flexural Laminate Stiffness E2   | 86960.25  |
| Laminate Shear Stiffness        | 8000      |
| Laminate Stiffness E1           | 239536.6  |
| Laminate Stiffness E2           | 161692.2  |
| Out of Plane Shear              | 1715.785  |
| Out of Plane Shear              | 2923.015  |
| Shear Correction Factor k44     | 0.3825    |
| Shear Correction Factor k55     | 0.516695  |

Figure 12. Polar Coordinates of [0/90/0/90/0]

Figure 13. Total Deformation of [0/90/0/90/0]

Figure 14. Von-Mises stress of [0/90/0/90/0]
The total deformation plot is similar to what was seen in Model 1. The maximum deformation value is 0.035096 m which occurs at the root of the wing. Here, the maximum stress value is 5.9917e+7 Pa and acts on the skin of the model. The minimum value is 9.8927e-5 Pa at the root of the wing.

Model 2.B: A ply sequence of [0/-35/45/90/-45/0/45/90/-45/35/0] is applied on the skin. This layup is a general type of laminate which is antisymmetric about the middle ply and not the midplane. Each $\Theta$ is associated with a $-\Theta$. The $[B_{i,j}]$ matrix is not zero.

Table 7. Laminate Engineering Constants of [0/-35/45/90/-45/0/45/90/-45/35/0]

| Constant                        | Value   |
|---------------------------------|---------|
| Flexural Laminate Shear Stiffness| 48297.99|
| Flexural Laminate Stiffness E1  | 227833.6|
| Flexural Laminate Stiffness E2  | 60384.85|
| Laminate Shear Stiffness        | 56073.39|
| Laminate Stiffness E1           | 162564.5|
| Laminate Stiffness E2           | 109438.3|
| Out of Plane Shear              | 1890.714|
| Out of Plane Shear              | 2840.326|
| Shear Correction Factor k44     | 0.408973|
| Shear Correction Factor k55     | 0.514573|
The maximum total deformation of the model in this arrangement of plies is 0.024841 m. Looking at the stress plot, we see that the maximum stress value is 5.1548e+7 Pa while the minimum value is 0.00046795 Pa on the root of the wing.

Model 2. C: This model has a ply sequence of [0/90/45/-45/0/45/90/45/0] which is a quasi-isotropic laminate layup. A laminate is known as a quasi-isotropic laminate when its extensional-stiffness matrix \([A_{i,j}]\) behaves like an isotropic material. The reason it is called “quasi-isotropic” is that \([B_{i,j}]\) and\([D_{i,j}]\) may or may not behave like an isotropic material. For this ply sequence, the stiffness is independent of the rotation angle of the laminate and that the extension and shear matrices are uncoupled.

| Constant                        | Value       |
|---------------------------------|-------------|
| Flexural Laminate Shear Stiffness| 22625.9     |
| Flexural Laminate Stiffness E1   | 225004.5    |
| Flexural Laminate Stiffness E2   | 135257.1    |
| Laminate Shear Stiffness        | 48760.26    |
| Laminate Stiffness E1           | 168748.9    |
| Laminate Stiffness E2           | 128535.7    |
| Out of Plane Shear              | 2070.831    |
| Out of Plane Shear              | 3008.249    |
| Shear Correction Factor k44     | 0.436329    |
| Shear Correction Factor k55     | 0.557411    |
As seen in the deformation plot, the maximum deformation value is 0.025436 m. The maximum value of stress is found to be 4.2801e+7 Pa.

Table 9. Results of the analysis on the wing model

| Model Types | Skin Material | Ply orientation | Total Deformation (m) | Von-Mises Stress (Pa) |
|-------------|---------------|-----------------|----------------------|----------------------|
| Model 1     | Titanium Alloy | Nil             | 0.050676             | 2.9099e+8            |
| Model 2     | CFRP A        | [0/90/0/90/0/0] | 0.035096             | 5.9917e+7            |
|             | CFRP B        | [0/-35/45/90/-45/0/45/90/-45/35/0] | 0.024841 | 5.1548e+7 |
|             | CFRP C        | [0/90/45/-45/0/-45/45/90/0] | 0.025436 | 4.2801e+7 |

4. Conclusion
A comparison was drawn between two models of the aircraft wing with similar loading and boundary condition and similar design with change in materials of the skin. Titanium alloy in Model 1 and Carbon-Fibre Epoxy Composite material (CFRP) in Model 2 and the effect of different ply orientations were evaluated. Reading the results of the different ply orientations in Model 2, the lowest deformation of 0.024841 m is observed in Model 2.B which has the stacking sequence of [0/-35/45/90/-45/0/45/90/-45/35/0]. On comparing the deformation in Model 1 and Model 2.B, a weight reduction of 2.37 % is observed in the wing favouring Model 2.B, and the total deformation is reduced by ≈ 51%. Similarly, the von-Mises stress is reduced by ≈ 85%. This underscores the fact that the
application of composite materials on the skin of the wing compared to standard titanium alloys provide a higher stiffness to weight ratio. The orientations of plies are of prime importance in deciding the strength of the composite stack up. Various stack-up orientations are used through iterative procedures and compared. It is found that model 2.B is the optimum choice for composite layups among the laminate orientations in the study because it showcased better strength to weight ratio than others because of lesser deformation and stress value. Further study can be focused on making cost-effective materials and optimising the design of the wing to achieve optimum strength to weight ratio. Better and more accurate results can be obtained by generating a finer mesh, and composite failure analysis can be done as an extension to this study.

5. Nomenclature

| N | Traction at point (x, y) |
| M | Moment at point (x, y) |
| [Aij] | Extensional-stiffness matrix |
| [Bi,j] | Extension-bending coupling stiffness matrix |
| [Di,j] | Bending-stiffness matrix |
| ϵ0 | The mid-plane strain |
| κ | The vector comprising of second derivatives of the displacement, known as the curvature. |

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