UAV WING REPLACEMENT DESIGN: DESIGN OF INTERNAL STRUCTURE USING FIBER GLASS

ENGG5508

GROUP NO: III

ABSTRACT

The internal structure, commonly referred to as the spar, holds paramount importance in the design and functionality of Unmanned Aerial Vehicles (UAVs). This structural component, spanning the width of the wing, provides critical support, wing rigidity, and addresses aerodynamic considerations. This project focuses on the design of a spar element for the UAV wing, constructed with Glass Fiber and Epoxy Resin. The study assesses the material's load-bearing capability using Classical Laminate Plate Theory, utilizing the Selig S5010 airfoil as the design reference and a rectangular solid beam with a [0/90] fiber orientation. To the stress values obtained, Maximum Stress Failure Theory is applied and the corresponding Strength Ratios are determined. It is observed that both 0 and 90 degree plies are able to withstand the loads and moments. The entire laminate failure load is also calculated which is found to be 3.085e+05.

1. Introduction

Unmanned aerial vehicles (UAVs) play a crucial role in environmental monitoring and agriculture, relying on efficient wing design for optimal performance. Our design approach focuses on utilizing Glass Fiber, a composite material known for its exceptional mechanical strength and corrosion resistance. This material aligns with aeronautical engineering principles, addressing the intricate requirements of UAV wings. The design rationale, outlined in the following sections, emphasizes the strategic use of Glass Fiber for constructing a rectangular beam internal support structure to enhance structural integrity and overall UAV performance.

2. Design Methodology

2.1. Wing Design and Specifications

The wing design is based on the **Selig S5010** airfoil standard as shown in Figure 1, features a wingspan of **2.83m**. Original specifications and design details are outlined in Table 1. To provide context, the UAV's cruise speed is **17.6m/s**, and the total weight, including payloads, is **8.6 kg**, with the original wing weighing **2 kg**.

Table 1: Wing specifications

Parameters	**Value**
Wingspan (m)	2.83
Length of Wing - L (m)	1.4

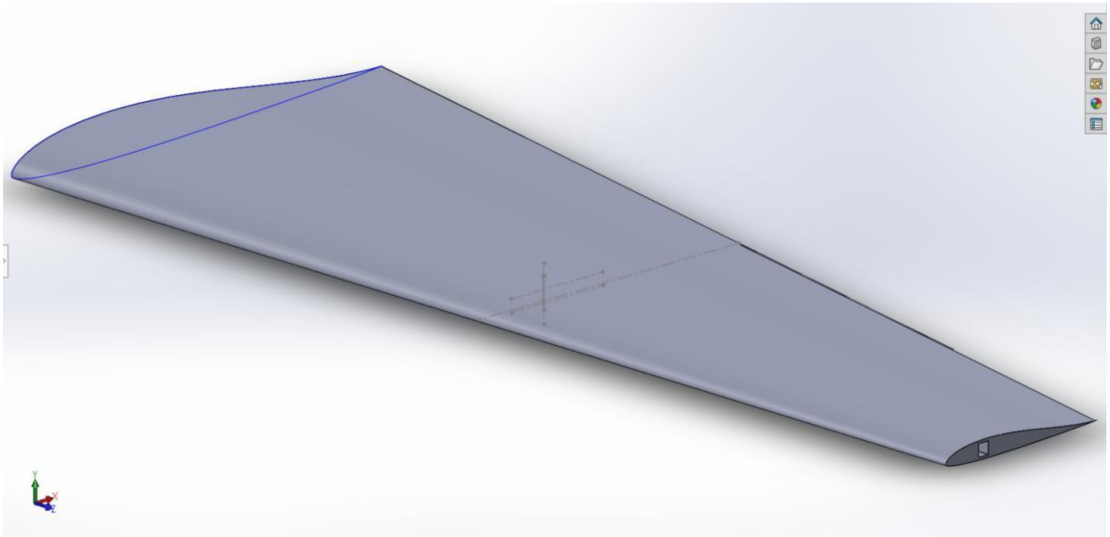

Figure 1: Wing Design

2.2. Load calculations

The expression for lift can be found by the expression q1(x) given below,

$$q_1(x) = \frac{2\text{Wto n}\sqrt{\text{L}^2 - \text{x}^2}}{\text{L}^2\pi} \tag{1}$$

It is assumed that the load caused by the weight of the wing structure is proportional to the chord length (the width of the wing), which is highest at the wing base **(Co = 0.54m)** and tapers off towards the wing tip **(Ct = 0.22 m)**. Thus, the load acting on the wing structure, $q_w(x)$ is calculated by,

$$q_w(x) = kw\left(\frac{Ct - C0}{L}x + C0\right) \tag{2}$$

Now, the total load acting on the wing structure $q_t(x)$ is calculated by adding both equation 1 and equation 2 using the formula,

$$q_t(x) = \frac{2Wto\ n\sqrt{L^2 - x^2}}{L^2\pi} - \frac{Wws\ n\ (C0 - \frac{Ct - C0}{L}x)}{L(C0+Ct)}$$

(3)

Using the load values from the above equations, we can calculate the shear force of the beam. Integrating the shear force, we will obtain the bending moment of the beam. The formula used for calculation in mentioned in Equation 4.

$$\boldsymbol{V(x) = -\int qt(x)dx} ; \tag{4}$$

$$\boldsymbol{M(x) = \int V(x)dx} ; \tag{5}$$

The value of Shear force and Bending moment were found to be **-3.020N** and **44.828Nm** respectively. Detailed calculations for shear force and bending moment are provided in Appendix A.

2.3. Beam specifications and manufacturing process

The spar, a foundational structural component in aircraft wings, is crucial for providing support, strength, and distributing aerodynamic loads during flight. In this project, the focus is on the meticulous design of the spar element for the UAV wing, emphasizing the utilization of Glass Fiber as the primary material. Glass Fiber was the material chosen because of its exceptional mechanical properties, lightweight properties, and resistance to corrosion (given in Table 2). The design specifications are detailed in Table 3. These specifications have been carefully chosen to ensure the designed spar element meets the structural demands of UAV wings, aligning with the principles of aeronautical engineering.

Table 2: Properties of Glass Fiber

Mechanical Properties	**Value**
Longitudinal Elastic Modulus (E_1)	38.6GPa
Transverse Elastic Modulus (E_2)	8.27GPa
Major Poisson's Ratio (V_{12})	0.26
Shear Modulus (G_{12})	4.14GPa
Ultimate Longitudinal Tensile Strength $(\sigma_1^T)_{ult}$	1062MPa
Ultimate Longitudinal Compressive Strength $(\sigma_1^C)_{ult}$	610MPa
Ultimate Transverse Tensile Strength $(\sigma_2^T)_{ult}$	31MPa
Ultimate Transverse Compressive Strength $(\sigma_2^C)_{ult}$	118MPa
Ultimate In-plane Shear Strength $(\tau_{12})_{ult}$	72MPa

Table 3: Beam Specifications

Properties	**Specification**
Fiber	Glass Fiber
Resin	Epoxy Resin

Fiber orientation	$[0/90/0/90/0]_s$
Type of beam	Rectangular
Number of plies	10
Type of structure	Stack Laminates
Individual ply thickness (mm)	1.8
Total beam thickness (mm)	18
Length of beam (mm)	700
Breadth of beam (mm)	14.2

The cross-ply configuration $[0/90/0/90/0]_s$, characterized by layers with fibers oriented perpendicular to adjacent layers, aligns seamlessly with our design objectives. This arrangement significantly enhances the stiffness of the composite material, contributing to the structural integrity of the wing. The alternating layers streamline the layup process, reducing the potential for errors and ensuring a more straightforward and efficient manufacturing process. Also, the multiple fiber orientations in cross-ply laminates enhance impact resistance by dispersing and absorbing impact energy in various directions. This feature minimizes the risk of catastrophic failure, a critical consideration for UAV wings operating in dynamic environments.

The Hand Lay-Up method is used in the fabrication of the beam. The process starts with shaping the material into the required form. For efficient lamination, ten glass fiber sheets are carefully layered, bonded with epoxy resin, and allowed to cure for the entire night. The final product is polished giving it a smooth surface. The extensive cost estimates for manufacturing the beam is as shown in Table 4.

Table 4: Manufacturing cost

Material	Quantity	Cost ($CAD)
Glass Fiber Prepreg	1sqm sheet x 2	150
Epoxy Resin	1Kg Kit x 2	65.96
Labor (30$ per hour)		240
Miscellaneous		100
Total		555.96

It is observed that the airfoil has a maximum thickness at 25% camber. Hence, the beam is placed in that area. Figure 2 shows the airfoil and the beam structure. The internal structure designed is split into 2 sections: Root to Mid and Mid to Tip. From the calculations, (Appendix) it was clearly observed that, the moment at the section from the middle to the tip was more than the other section, with a value of **35.641Nm**.

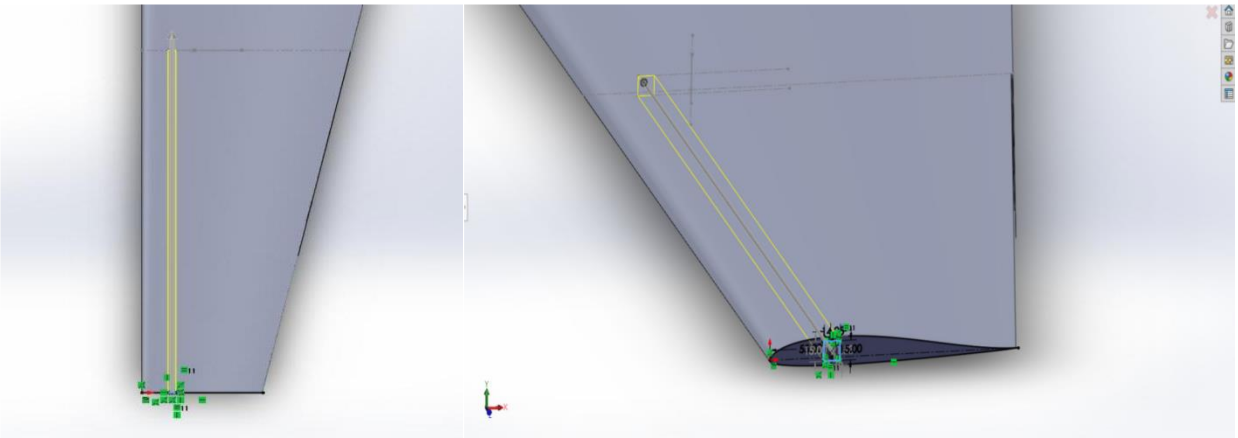

Figure 2: Position of Beam

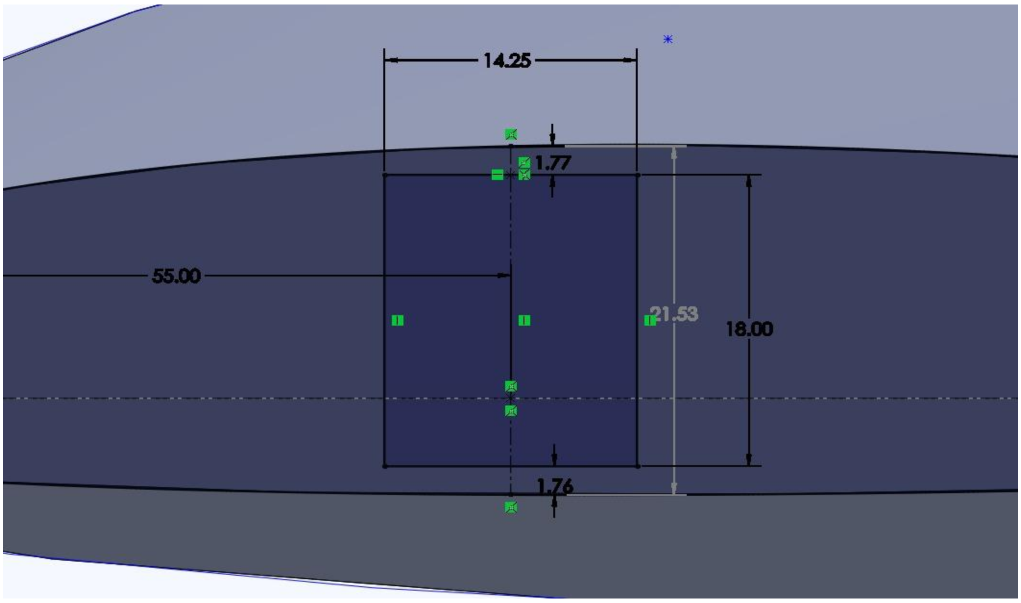

Figure 3: Dimensions of beam

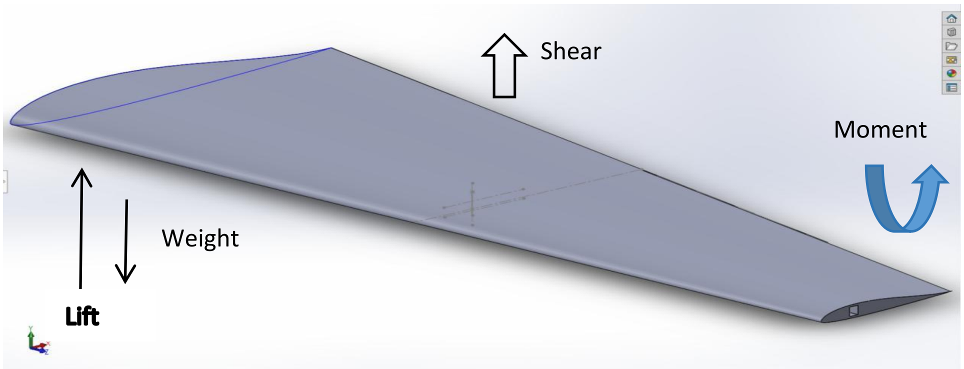

Figure 4: Free Body Diagram

3. Results and Discussions

It is assumed that N_x and N_y is 0 since mid-plane shear is 0 for a rectangular solid beam. From Appendix, we know that **M_x is 2.501Nm**. By the concept of Classical Laminate Plate Theory, we get the local and global stresses and strains with the help of MATLAB. The results of stress and strain are shown below in Table 5.

<u>ABD Matrix</u>

	1	2	3	4	5	6
1	4.2301e+08	2.3563e+07	5.3185e-24	-2.4928e+05	0	1.0268e-12
2	2.3563e+07	9.0629e+07	-5.3185e-24	0	2.4928e+05	1.4237e-11
3	5.3185e-24	-5.3185e-24	4.4712e+07	1.0268e-12	1.4237e-11	0
4	-2.4928e+05	0	1.0268e-12	1.2212e+04	680.2386	1.2510e-28
5	0	2.4928e+05	1.4237e-11	680.2386	2.6163e+03	-1.2510e-28
6	1.0268e-12	1.4237e-11	0	1.2510e-28	-1.2510e-28	1.2908e+03

Figure 5: ABD Matrix

<u>Inverse ABD Matrix</u>

	1	2	3	4	5	6
1	4.2301e+08	2.3563e+07	5.3185e-24	-2.4928e+05	0	1.0268e-12
2	2.3563e+07	9.0629e+07	-5.3185e-24	0	2.4928e+05	1.4237e-11
3	5.3185e-24	-5.3185e-24	4.4712e+07	1.0268e-12	1.4237e-11	0
4	-2.4928e+05	0	1.0268e-12	1.2212e+04	680.2386	1.2510e-28
5	0	2.4928e+05	1.4237e-11	680.2386	2.6163e+03	-1.2510e-28
6	1.0268e-12	1.4237e-11	0	1.2510e-28	-1.2510e-28	1.2908e+03

Figure 6: Inverse of ABD Matrix

Table 5: Stress-Strain Data for Glass Fiber

Ply Group	ε_1	ε_2	γ_{12}	σ_1 **(Pa)**	σ_2 **(Pa)**	τ_{12}
0	1.6657e-06	-1.8904e-07	0	6.4827e+04	2.0478e+03	0
90	-1.8904e-07	1.6657e-06	0	-3.7699e+03	1.3565e+04	0

According to Maximum Stress Failure Theory (MSFT) it is known that,

$$-(\sigma_1^C)_{ult} < \sigma_1 < (\sigma_1^T)_{ult}$$
$$-(\sigma_2^C)_{ult} < \sigma_2 < (\sigma_2^T)_{ult}$$
$$-(\tau_{12})_{ult} < \tau_{12} < (\tau_{12})_{ult}$$

- Applying MSFT for the 0 degree ply we get,

$$-610e{+}06 < \mathbf{6.4827e{+}04} < 1062e{+}06$$
$$-118e{+}06 < \mathbf{2.0478e{+}03} < 31e{+}06$$
$$-72e{+}06 < \mathbf{0} < 72e{+}06$$

Therefore, the Strength Ratios (from Appendix) for the 0 degree ply will be,

$$\mathbf{SR0_1 = 1.6382e{+}04} \text{ (and) } \mathbf{SR0_2 = 1.5138e{+}04}$$

-Applying MSFT for the 90 degree ply we get,

$$-610e{+}06 < -3.7699e{+}03 < 1062e{+}06$$
$$-118e{+}06 < 1.3565e{+}04 < 31e{+}06$$
$$-72e{+}06 < \mathbf{0} < 72e{+}06$$

Therefore, the Strength Ratios for the 90 degree ply will be,

$$\mathbf{SR90_1 = -1.6181e{+}05} \text{ (and) } \mathbf{SR90_2 = 2.2853e{+}03}$$

Based on Fully Discounted Method, when all the 90 degree plies fail, the strength ratio of 0 degree ply in 1_T will be **2.4303**. The failure strength ratio of the 90 degree ply is **2.2853e+03** (from MATLAB). So, the moment at which the entire laminate fails is,

$$M_x = 2.4303 * 2.2853e{+}03$$
$$\mathbf{M_x = 5.553e{+}03\ Nm}$$

Hence, the corresponding stress at which the laminate fails is,

$$\boldsymbol{\sigma_X = \frac{Mx}{h}}$$

$$\boxed{\boldsymbol{\sigma_X = 3.085e{+}05\ Pa}}$$

4. Conclusion

The Glass Fiber/Epoxy laminate proves to be an excellent material for the internal structure and is able to withstand the resultant load acting on it. The stresses and strains are calculated individually for the 0 and 90 degree plies and are compared with their corresponding ultimate tensile and compressive strengths (Table 2). As a result, it can be seen that the plies are within the stress limits as seen in Table 6. The strength ratios are also calculated and the final failure load is determined.

Table 6: Stress Results for plies

Ply Group	σ_1 (Pa)	σ_2 (Pa)	Within limits?
0	6.4827e+04	2.0478e+03	YES
90	-3.7699e+03	1.3565e+04	

APPENDIX

LOAD CALCULATIONS:

The given values:

W_{to} = 8.6 Kg

C_0 = 0.54m

C_t = 0.22m

W_{ws} = 2kg

n = 1.5 (Load factor for Takeoff)

Let's substitute the given values in the Total load equation:

$$q_t(x) = \frac{2W_{to}n\sqrt{L^2 - x^2}}{L^2\pi} - \frac{W_{ws}n\left(C_o - \frac{x(C_o - C_t)}{L}\right)}{L(C_o + C_t)}$$

Then let's calculate the integral of V(x) and M(x):

$$V(x) = -\int q_t(x)\mathrm{d}x$$

After integrating qt(x) we'll get a function:

$$\int_0^L -\frac{x}{2}\left(\sqrt{L^2 - x^2} + \frac{L^2}{2}\arcsin\left(\frac{x}{L}\right) + \frac{4.536x - 0.96x^2}{2.297}\right)\mathrm{d}x$$

This is the equation of M(x).

First, let's integrate the equation of M(x) with limits of 0 to 1.4. (Total length of the wing)

$$\int_0^{1.4} -\frac{x}{2}\left(\sqrt{L^2 - x^2} + \frac{L^2}{2}\arcsin\left(\frac{x}{L}\right) + \frac{4.536x - 0.96x^2}{2.297}\right)\mathrm{d}x$$

The total moment acting on the whole Wing is M(x) = 44.828.

Now let's integrate M(x) with Limit values: 0 to 0.7 and 0.7 to 1.4, So that we can calculate the moment values from the Root of the UAV wing to the mid-section of the wing and from the mid-section to the tip of the wing to get the actual moment distribution.

$$\int_0^{0.7} -\frac{x}{2}\left(\sqrt{L^2 - x^2} + \frac{L^2}{2}\arcsin\left(\frac{x}{L}\right) + \frac{4.536x - 0.96x^2}{2.297}\right)\mathrm{d}x$$

So from 0 to 0.7 (root to mid), the M(x) = 9.812.

Now let's calculate the Moment from mid-section to the tip of the wing:

$$\int_{0.7}^{1.4} -\frac{x}{2}\left(\sqrt{L^2-x^2}+\frac{L^2}{2}\arcsin\left(\frac{x}{L}\right)+\frac{4.536x-0.96x^2}{2.297}\right)\mathrm{d}x$$

Now from 0.7m to 1.4m (mid-section) the M(x) = M_{beam} =35.641.

Now we can conclude that the Moment acts more towards the tip.

So,

$$M_{\text{beam}} = bM_x$$

$$\boxed{M_x = 2.501}$$

Strength Ratio for a given material is given by the formula,

$$\mathbf{SR} = \frac{\boldsymbol{Maximum\ Load\ that\ can\ be\ applied}}{\boldsymbol{Load\ Applied}}$$

REFERENCES

[1] P.-H. Chung, D.-M. Ma, and J.-K. Shiau, "Design, Manufacturing, and Flight Testing of an Experimental Flying Wing UAV," Applied Sciences, vol. 9, p. 3043, Jan. 2019.

[2] D. Doherty, "Analytical Modeling of Aircraft Wing Loads Using MATLAB and Symbolic Math Toolbox," 2009.

[3] C Kong, "Structural Design on Wing of a Small Scale Wig Vehicle with Carbon/Epoxy and Foam Sandwich Composite Structure", 16th International Conference of Composite Materials, Jan 2007.