Indian Institute of Technology Madras

Aerospace Engineering

Course:

Structural Design and Fabrication of UAV

AS5100

Design of UAV for Forest Surveillance

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Chapter 1

Specifications

1.1 Mission Specifications

Our UAV has a fixed wing and a twin propeller design. The aim is to perform forest surveillance focusing on identifying events that could lead to wildfires.
After completing the aerodynamic design, the UAV is estimated to weigh 5.47 kg. The aerodynamic parameters of the mission are summarised below.

MISSION DETAILS	
Cruise Speed, V_{cr}	20 m/s
Cruise Altitude	100 m
Maximum Cruise Speed, V_{max}	25 m/s
Takeoff speed, V_{TO}	12.86 m/s
Stall Speed, V_{stall}	12.2 m/s
Corner Velocity, V_{corner}	19.29 m/s
Red Line Speed	30 m/s
Positive Limit Load Factor n_+	2.5
Negative Limit Load Factor n_-	-1
Lift Coefficient at zero AOA, C_{L_0}	0.2
Cruise Lift Coefficient, $C_{L_{cr}}$	0.53
Maximum Lift Coefficient	1.5
Ostwald's efficiency, e	0.82
Parasitic Drag, C_{D_0}	0.023

Table 1.1: Aerodynamic Parameters of the UAV

1.2 Dimensional Specifications

The dimensional specifications of different structural components of the UAV is summarised below.

- DTOW of UAV = 5.47 Kg, Wing Loading = $127.76 N/m^2$.

WING	
Wingspan, b	1.84 m
Chord, c	0.23 m
Aspect Ratio, AR	8
Planform Area, S	0.42 m^2
Weight, W_w	1.081 Kg
Airfoil	NACA2412
Twist	0 deg
Taper Ratio	1
Sweep	0 deg
Dihedral	0 deg

Table 1.2: Wing Parameters

FUSELAGE	
Total Length	169 cm
Width	16.81 cm
Height	13.66 cm
Weight, Wf	1.605 Kg
Length of upswept part	109 cm

Table 1.3: Fuselage Parameters

HORIZONTAL TAIL	
Span	60.6 cm
Chord	11.37 cm
Planform Area	0.07 m^2
Aspect Ratio, AR_h	5.33
Airfoil	NACA0012
Sweep	0 deg
Twist	0 deg
Taper Ratio	1
Dihedral	0 deg
Weight, W_h	0.183 Kg

Table 1.4: Horizontal Tail Parameters

VERTICAL TAIL	
Span	34.64 cm
Root Chord	15.4 cm
Tip Chord	7.7 cm
Planform Area	0.04 m^2
Aspect Ratio, AR_v	3
Airfoil	NACA0012
Sweep	0 deg
Twist	0 deg
Taper Ratio	0.5
Dihedral	0 deg
Weight, W_h	0.107 Kg

Table 1.5: Vertical Tail Parameters

LANDING GEAR	
Wheel Base, B	55 cm
Wheel Track, T	20.1 cm
Landing Gear Height, H_g	21.56 cm
Clearance Angle	16 deg
Tire Diameter	6 cm
Tire Width	2 cm
Weight, W_{LG}	0.385 Kg

Table 1.6: Landing Gear Parameters

CONTROL SURFACES	
Aileron (x2)	
Chord	4.6 cm
Span	23 cm
Planform Area	0.011 m^2
Elevator (x1)	
Chord	4.55 cm
Span	26.66 cm
Planform Area	0.022 m^2
Rudder (x1)	
Root Chord	4.31 cm
Tip Chord	2.16 cm
Span	29.44 cm
Planform Area	0.0096 m^2
Flap (x2)	
Chord	6.9 cm
Span	36.8 cm
Planform Area	0.021 m^2

Table 1.7: Control Surfaces Parameters

1.3 Torque Requirement for Servos

Torque requirement for Servos can be estimated using the following empirical relation.

$$Torque(oz-in) = 8.56 \times 10^{-6}(C^2V^2Lsin(S1)tan(S1)/tan(S2) \tag{1.1}$$

where,

$$C = \text{Control Surface Chord in cm}$$
$$V = \text{Speed in mph}$$
$$L = \text{Control Surface length in cm}$$
$$S1 = \text{Maximum Control Surface deflection in deg}$$
$$S2 = \text{Maximum Servo deflection in deg}$$

Since the maximum chord and length are for Rudder, maximum torque will also be required for the rudder. hence if rudder requirement is satisfied, it automatically will work for other control surfaces too. substituting all the values, we get required torque for the servo to be,

$$\text{Torque} = 5.438 \text{ oz-in}$$

Converting it to units used by the vendor (kg-cm), we get,

$$\text{Torque} = 0.39 \text{ Kg-cm}$$

Hence the minimum torque requirement for control surfaces would be 0.4 kg-cm. Nearest commercially available specification is chosen for the UAV, which is found to be **0.6 kg-cm** at 6V rating.

Chapter 2

Shear Force Diagram and Bending Moment Diagram

Shear Force and Bending Moment Diagram are crucial in the structural design of load-carrying members. They give us an idea of where the maximum load or bending moment can occur so that we can design the component to withstand that load or bending moment within tolerable limits. While designing an Aircraft wing, it can be assumed to be a cantilever beam supported at the root and free end at the tip.
Until this part of the design, we only have the overall lift requirement and not how it is distributed. In this section, we will see how lift is distributed in the wing, and then we will arrive at the Shear Force and Bending Moment calculation along with the diagrams.

2.1 Lift Distribution

For an elliptical wing (where Oswald Efficiency factor e=1), The lift distribution in the span-wise direction follows a similar curve as the chord distribution in the span-wise direction. However, for rectangular wings, the lift distribution is not as intuitive as elliptical wings. There is one empirical method to estimate the Lift distribution along the span wise direction for rectangular wings, known as Schrenk's Method.
In this method, we take the chord distribution of the actual wing across the span wise direction and an equivalent elliptical wing (which has the same aspect ratio and span as that of the rectangular wing). The arithmetic average of the two chords at each point along the span is known as Schrenk's Chord. The empirical relation states that, for any rectangular wing, the lift distribution curve is similar to the Schrenk's chord variation in the spanwise direction.
Using the specs of UAV, we split the half-wing (since it is symmetric, analyzing one half is enough) into 100 segments, and Schrenk's chord at each segment is found numerically. The variation of Actual wing chord, Equivalent wing chord, and Schrenk's chord is shown in the plot below.

Now, to find distribution along the wing, we need the proportionality factor k, where k multiplied by the chord at any given segment will give us the lift per unit length for that segment. k is found by using the overall lift requirement of the UAV. Since we know that Total lift = $k\times$ Area under the curve. Using the following

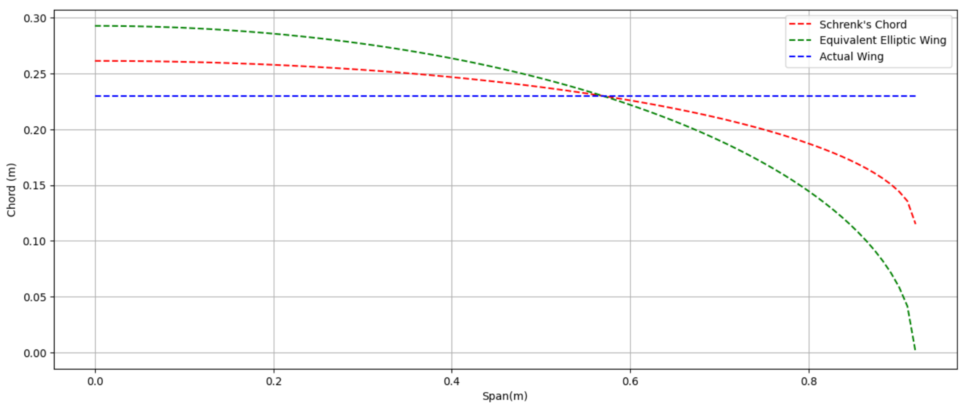

Figure 2.1: Variation of Chord along span-wise direction

formula, k is calculated.

$$k = \frac{w/2}{\int_0^{b/2} C_{Schrenk}(y)dy} \tag{2.1}$$

After finding k, it is multiplied with Schrenk's chord of each segment to find out the Lift Distribution. The lift Distribution of the wing is shown in the following plot.

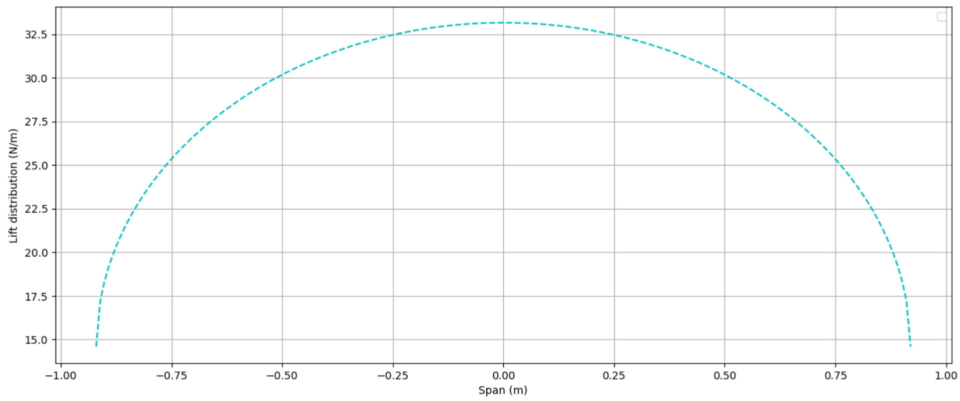

Figure 2.2: Variation of Lift in the wing

Now we have the lift at each segment of the wing. from here, Shear Force is calculated by taking the cumulative forces acting on each segment of the wing from tip. similarly bending moment is calculated by taking the cumulative effect of bending moment caused by loads from tip to that particular segment. The Shear Force and Bending Moment diagram are as follows:

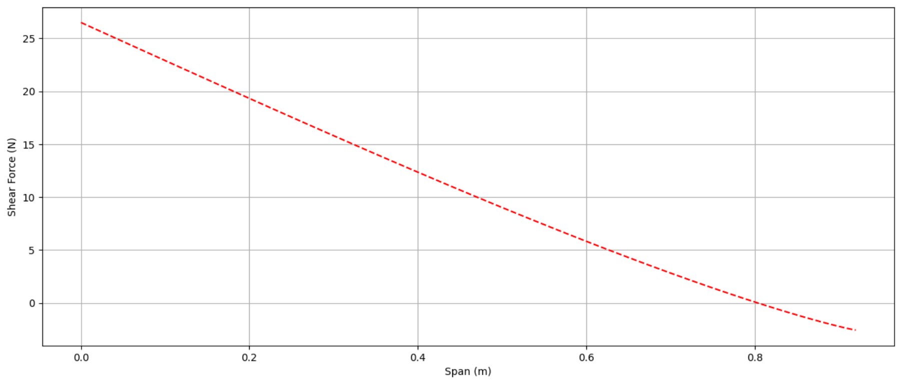

Figure 2.3: Shear Force Diagram

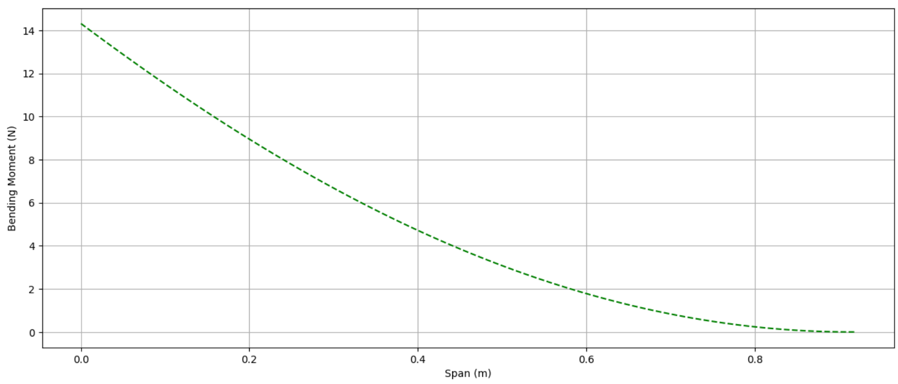

Figure 2.4: Bending Moment Diagram

From the above diagram, we can find that, Maximum bending moment occurs at the root.

$$BM_{max} = 14.315\, Nm \tag{2.2}$$

Note: Please refer Appendix for all calculations.

Chapter 3

Spar Calculation

3.1 Spar Dimensions

We estimate the allowable stress from the yield stress by the approximation,

$$\sigma_{\text{Allowable}} \leq \frac{\sigma_{\text{Yield}}}{n \cdot \eta_{\text{fatigue}} \cdot k \cdot f}$$

Where,

- σ_{Yield} = Yield Stress of sheet metal, measured in Pascals. For Al 6061, we have it as 270 MPa.
- $n = 2.5$ is the positive limit load factor, $\eta_{fatigue} = 1.5$ is the fatigue factor.
- $k = 2$ is the stress concentration factor, $f = 1.4$ is the factor of safety.

Substituting values,

$$\sigma_{\text{Allowable}} \leq \frac{\sigma_{\text{Yield}}}{2.5 \cdot 1.5 \cdot 2 \cdot 1.4} \equiv \frac{\sigma_{\text{Yield}}}{10.5}$$

$\sigma_{\text{Allowable}}$ because of Maximum Bending Moment (M_{max}) generated can be expressed as,

$$M_{\text{max}} = 14.6\,\text{N-m}, \quad t_{\text{max}} = 0.12 \times 230 = 27.6\,\text{mm}, \quad \sigma_{\text{allowable}} = 25.71\,\text{MPa},$$

$$\sigma_{\text{Allowable}} = \frac{M_{\text{max}} \cdot \left(\frac{t_{max}}{2}\right)}{I_{total}}$$

$$I_{\text{total}} = 8778\ mm^4$$

$$\frac{\sigma_{\text{Yield}}}{10.5} = \frac{M_{\text{max}} \cdot \left(\frac{t_{\text{max}}}{2}\right)}{I_{\text{Spar}} + I_{\text{Skin}}}$$

$$I_{\text{Spar}} + I_{\text{Skin}} = \frac{5.25 \cdot M_{\text{max}} \cdot t_{\text{max}}}{\sigma_{\text{Yield}}}$$

3.2 I_{Skin} Calculations

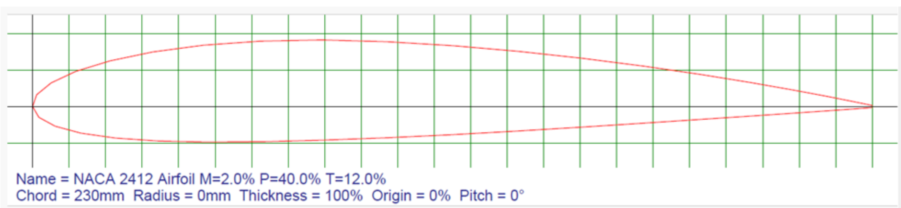

Figure 3.1: NACA 2412 Aerofoil Section

For the NACA 2412 aerofoil, with a chord length $c = 230\,\text{mm}$, the upper and lower surfaces are divided into 20 equal segments, and the corresponding x and y coordinates are generated. Each individual segment is treated as a small, almost horizontal plate with a thickness $t = 0.5\,\text{mm}$.
The location of the center of gravity (CG) for the thin aerofoil surfaces is calculated. The second moment of area, I_{Skin}, is then calculated by summing up the second moments of area for each individual segment.
For the calculations, the assumed thickness of skin is 0.3mm. The following values were obtained. Refer

$$I_{skin} = \Sigma[((Y_{cg} - y)^2)A] = 24352.84mm^4$$

$$Ycg = [\Sigma(A \times y)/\Sigma A] = 3.10mm$$

So, putting the values,

$$I_{\text{Skin}} = 24352.84\,\text{mm}^4, \quad M_{\max} = 14.6\,\text{N-m}, \quad t_{\max} = 0.12 \times 230 = 27.6\,\text{mm}, \quad \sigma_{\text{Yield}} = 270\,\text{MPa},$$

$$\left(I_{\text{Spar}} + 24.35 \times 10^{-9}\right) = \frac{5.25 \times 14.6 \times 0.0276}{270 \times 10^6} = 7.835 \times 10^{-9}$$

$$I_{\text{Spar}} = -15.57 \times 10^{-9}\,\text{m}^4$$

Negative value of I_{spar} omits the requirement of any spar in the wing as the skin is sufficient to carry the bending loads. So, we will use a wing-to-bulkhead coupler to attach the wing to the bulkheads through a 3D-printed adapter.

AIRFOIL TOP SURFACE							
X (mm)	Y (mm)	Y-mid= [y2+y1]/2 [y] (mm)	Strip length (x2-x1) [b] (mm)	Strip area [A=bt] (mmˆ2)	[A(y)] (mmˆ3)	$[Y_{cg} - y]$ (mm)	$[((Y_{cg} - y)^2)\text{A}]$ (mm^4)
0.00	0.00	1.62	1.11	0.56	0.90	1.49	1.22
1.11	3.23	4.85	3.96	1.98	9.60	-1.75	6.06
5.07	6.47	8.05	6.73	3.37	27.07	-4.95	82.28
11.80	9.62	11.06	9.36	4.68	51.74	-7.96	296.16
21.16	12.49	13.72	11.75	5.88	80.58	-10.62	661.98
32.91	14.94	15.86	13.85	6.93	109.83	-12.76	1127.51
46.76	16.78	17.34	15.59	7.80	135.13	-14.24	1579.54
62.35	17.89	18.06	16.93	8.47	152.84	-14.96	1893.21
79.28	18.22	17.98	17.76	8.88	159.66	-14.88	1966.16
97.04	17.74	17.20	18.10	9.05	155.61	-14.10	1797.95
115.14	16.65	15.87	18.07	9.04	143.39	-12.77	1473.36
133.21	15.09	14.14	17.59	8.80	124.32	-11.04	1070.98
150.80	13.18	12.11	16.69	8.35	101.06	-9.01	677.45
167.49	11.04	9.92	15.38	7.69	76.28	-6.82	357.68
182.87	8.80	7.70	13.68	6.84	52.63	-4.60	144.42
196.55	6.59	5.57	11.67	5.83	32.47	-2.47	35.45
208.22	4.54	3.67	9.37	4.69	17.17	-0.57	1.50
217.59	2.79	2.11	6.85	3.43	7.23	0.99	3.36
224.44	1.43	1.01	4.18	2.09	2.10	2.10	9.17
228.62	0.58	0.44	1.40	0.70	0.30	2.67	4.97
230.02	0.29						
AIRFOIL BOTTOM SURFACE							
0.00	0.00	-1.48	1.72	0.86	-1.27	4.58	18.00
1.72	-2.95	-4.17	4.47	2.24	-9.31	7.27	117.96
6.19	-5.38	-6.33	7.07	3.54	-22.38	9.43	314.35
13.26	-7.28	-7.96	9.51	4.76	-37.83	11.06	581.12
22.77	-8.63	-9.03	11.69	5.85	-52.78	12.13	860.02
34.46	-9.43	-9.59	13.59	6.80	-65.13	12.69	1093.38
48.05	-9.74	-9.68	15.18	7.59	-73.47	12.78	1239.66
63.23	-9.62	-9.41	16.42	8.21	-77.22	12.51	1283.84
79.65	-9.19	-8.87	17.33	8.67	-76.86	11.97	1241.53
96.98	-8.55	-8.13	17.88	8.94	-72.64	11.23	1126.45
114.86	-7.70	-7.20	17.91	8.96	-64.48	10.30	950.04
132.77	-6.70	-6.17	17.50	8.75	-53.99	9.27	751.91
150.27	-5.64	-5.11	16.66	8.33	-42.52	8.21	560.79
166.93	-4.57	-4.07	15.39	7.69	-31.28	7.17	395.04
182.32	-3.56	-3.11	13.76	6.88	-21.36	6.21	264.89
196.08	-2.65	-2.25	11.78	5.89	-13.25	5.35	168.59
207.86	-1.85	-1.52	9.49	4.74	-7.21	4.62	101.28
217.35	-1.19	-0.95	6.95	3.48	-3.28	4.05	56.86
224.30	-0.70	-0.55	4.25	2.13	-1.16	3.65	28.23
228.55	-0.39	-0.34	1.43	0.71	-0.24	3.44	8.46
229.98	-0.29	0.00	0.00				

Table 3.1: Idealization of airfoil surface

Chapter 4

Shear Calculation

4.1 Idealization of Wing Cross section

Idealization of structures is used to simplify calculations where we assume the booms to carry the bending loads and the stiffeners carry only the shear loads. Refer Appendix for details of calculation.

$$B_2 = \frac{t \cdot b_{12}}{6}\left(2 + \frac{\sigma_1}{\sigma_2}\right) + \frac{t \cdot b_{32}}{6}\left(2 + \frac{\sigma_3}{\sigma_2}\right)$$

$$\frac{\sigma_1}{\sigma_2} = \frac{Y_{CG} - Y_1}{Y_{CG} - Y_2} \quad \text{(Since, stress is proportional to strip length)}$$

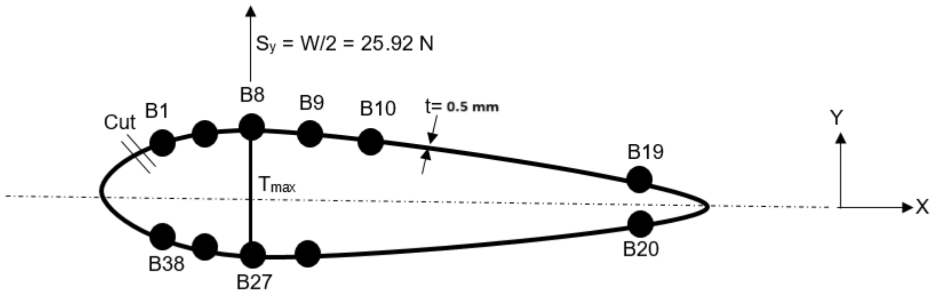

Figure 4.1: Structural idealisation

Flange area = 5 mm^2, and flanges will be located at a distance of $\frac{t_{\max}}{2}$, at $X = 79.28\,\text{mm}$.

$$Y_{\text{Flange}} = \frac{t_{\max}}{2} = \frac{27.6}{2} = 13.8\,\text{mm}$$

Total Moment of Area of Idealized section will be:

$$I_{XX} = I_{\text{CG}} = 26278.67\,\text{mm}^4$$

4.2 Shear Flow of Wing Cross section

The shear flow of a closed section can be calculated as:

$$q_s = -\left(\frac{S_x I_{xx} - S_y I_{xy}}{I_{xx} I_{yy} - I_{xy}^2}\right)\left(\int_0^s t_D x\, ds + \sum_{r=1}^{n} B_r x_r\right) - \left(\frac{S_y I_{yy} - S_x I_{xy}}{I_{xx} I_{yy} - I_{xy}^2}\right)\left(\int_0^s t_D y\, ds + \sum_{r=1}^{n} B_r y_r\right) + q_{s,0}$$

Assuming:

$$\begin{aligned} S_x &= 0, && \text{(Only Vertical Loads)} \\ t_D &= 0, && \text{(Skin is carrying only shear stress and no direct stress)} \\ I_{xy} &= 0, && \text{(Symmetric loading)} \end{aligned}$$

The above equation reduces to:

$$q_s = -\left(\frac{S_y}{I_{xx}}\right)\left(\sum_{r=1}^{n} B_r y_r\right) + q_{s,0}$$

AIRFOIL TOP SURFACE									
Yr=[$Y_{cg} - y$] (mm)	Top Surface Idealization	Idealized Areas [Br] (mm^2)	$q = [-\frac{S_y}{I_{xx}}]\Sigma[B_r Y_r]$ (N/mm)	Shear Flow in Open Section [qb] (N/mm)		X(mm)	ds=[x2-x1] (mm)	qds (N)	Shear Flow in Closed Section q=[qb+qs0] (N/mm)
-0.13	B1	7.19	0.0010	q1,2	0.0010	1.11	3.96	0.00	-0.63
-3.37	B2	2.88	0.0103	q2,3	0.0113	5.07	6.73	0.08	-0.62
-6.52	B3	4.09	0.0284	q3,4	0.0398	11.80	9.36	0.37	-0.59
-9.39	B4	5.29	0.0530	q4,5	0.0928	21.16	11.75	1.09	-0.54
-11.84	B5	6.38	0.0805	q5,6	0.1732	32.91	13.85	2.40	-0.46
-13.68	B6	7.31	0.1066	q6,7	0.2798	46.76	15.59	4.36	-0.35
-14.79	B7	8.06	0.1271	q7,8	0.4069	62.35	16.93	6.89	-0.22
-15.12	B8	13.59	0.2190	q8,9	0.6259	79.28	17.76	11.12	0.00
-14.64	B9	8.90	0.1389	q9,10	0.7648	97.04	18.10	13.84	0.13
-13.55	B10	8.99	0.1298	q10,11	0.8946	115.14	18.07	16.16	0.26
-11.99	B11	8.88	0.1134	q11,12	1.0080	133.21	17.59	17.73	0.38
-10.08	B12	8.55	0.0919	q12,13	1.0999	150.80	16.69	18.36	0.47
-7.94	B13	8.03	0.0679	q13,14	1.1678	167.49	15.38	17.96	0.54
-5.70	B14	7.33	0.0445	q14,15	1.2123	182.87	13.68	16.58	0.58
-3.49	B15	6.49	0.0241	q15,16	1.2364	196.55	11.67	14.43	0.61
-1.44	B16	5.70	0.0087	q16,17	1.2452	208.22	9.37	11.67	0.62
0.31	B17	2.15	-0.0007	q17,18	1.2445	217.59	6.85	8.52	0.61
1.67	B18	2.47	-0.0044	q18,19	1.2401	224.44	4.18	5.18	0.61
2.52	B19	1.29	-0.0035	q19,20	1.2366	228.62	1.38	1.71	0.61
AIRFOIL BOTTOM SURFACE									
6.05	B20	1.63	-0.0105	q20,21	1.2261	1.72	4.47	5.48	0.60
8.48	B21	2.91	-0.0263	q21,22	1.1998	6.19	7.07	8.48	0.57
10.38	B22	4.14	-0.0458	q22,23	1.1540	13.26	9.51	10.97	0.52
11.73	B23	5.28	-0.0659	q23,24	1.0881	22.77	11.69	12.72	0.46
12.53	B24	6.29	-0.0839	q24,25	1.0042	34.46	13.59	13.65	0.37
12.84	B25	7.15	-0.0979	q25,26	0.9063	48.05	15.18	13.76	0.28
12.72	B26	7.87	-0.1066	q26,27	0.7997	63.23	16.42	13.13	0.17
12.29	B27	13.41	-0.1756	q27,28	0.6240	79.65	17.33	10.81	-0.01
11.65	B28	8.77	-0.1089	q28,29	0.5151	96.98	17.88	9.21	-0.11
10.80	B29	8.93	-0.1027	q29,30	0.4124	114.86	17.91	7.39	-0.22
9.80	B30	8.85	-0.0924	q30,31	0.3200	132.77	17.50	5.60	-0.31
8.74	B31	8.55	-0.0796	q31,32	0.2404	150.27	16.66	4.01	-0.39
7.67	B32	8.04	-0.0657	q32,33	0.1747	166.93	15.39	2.69	-0.46
6.66	B33	7.33	-0.0520	q33,34	0.1227	182.32	13.76	1.69	-0.51
5.75	B34	6.43	-0.0394	q34,35	0.0833	196.08	11.78	0.98	-0.55
4.95	B35	5.37	-0.0283	q35,36	0.0550	207.86	9.49	0.52	-0.57
4.29	B36	4.17	-0.0190	q36,37	0.0360	217.35	6.95	0.25	-0.59
3.80	B37	2.85	-0.0115	q37,38	0.0245	224.30	4.25	0.10	-0.61
3.49	B38	1.45	-0.0054	q38	0.0191	228.55	1.45	0.03	-0.61

Table 4.1: Shear flow in airfoil

We have to consider the bending stress caused by the bending moment along with the shear stress. We have two panels with lengths $b_1 = 114.97mm$ and $b_2 = 230.7mm$. We note that the surface of the airfoil above the CG will have compressive bending stress and that below will have tensile bending stress due to the bending moment on the wing. We take the maximum bending moment from the BMD of our wing to get the limiting case. We calculate the net effect due to these two and that will be compared to the critical buckling load found from the formula. We have the shear flow at each location. So, net load per unit area is given by,

$$F_{net} = \frac{q}{t} + \frac{M_{wing_{max}} y}{I_{skin}}$$

Critical stress for simply supported conditions is given by the formula,

$$N_{cr} = \frac{K\pi^2 E}{12(1-\nu^2)}\left(\frac{t}{b}\right)^2 \tag{4.1}$$

where,

- K = $5.34 + \frac{4b^2}{a^2}$
- a = distance between two consecutive ribs,
- b = width of the panel,
- E = Young's Modulus of Elasticity,
- t = thickness of the panel = 0.5 mm

We have to divide the N_{cr} with limit load factor n, factor of safety f and an additional factor of 2. Considering these, the values calculated are shown below.

Panel	F_{max} (N/mm^2)	N_{cr} (N/mm^2)
1	8.11	9.805
2	7.07	9.11

Refer Appendix for the code used to arrive at the above result.

Since we get the required condition to avoid buckling without any requirement for stringers, we won't be needing stringers.

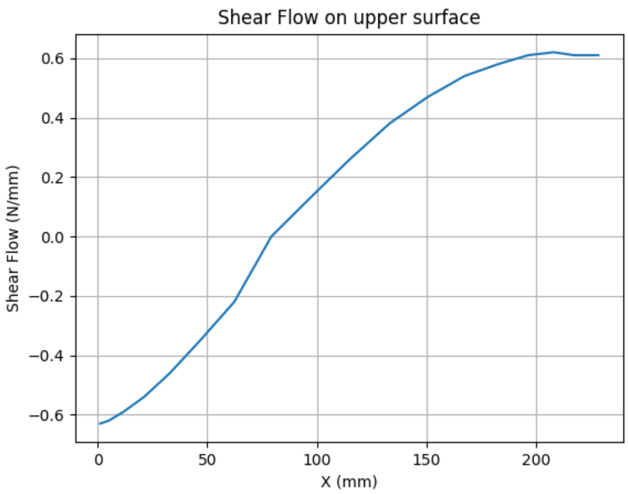

Figure 4.2: Shear flow in upper surface

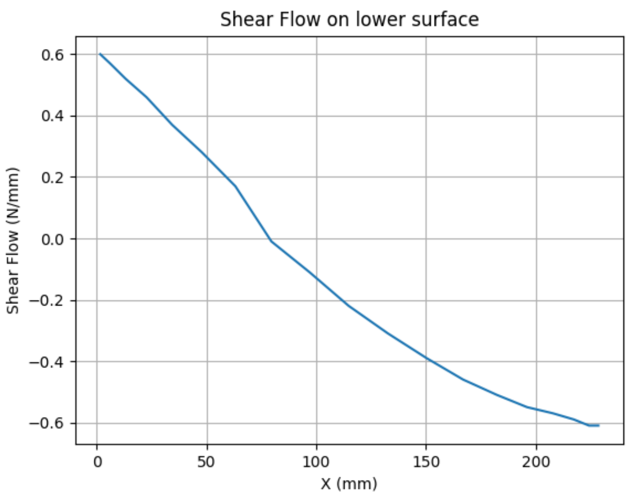

Figure 4.3: Shear flow in lower surface

4.3 Ribs

The Wing span is 1840 mm, we have taken the the rib spacing to be 40 % of wing chord (with the advised spacing to be between 25 - 40 % of the chord).

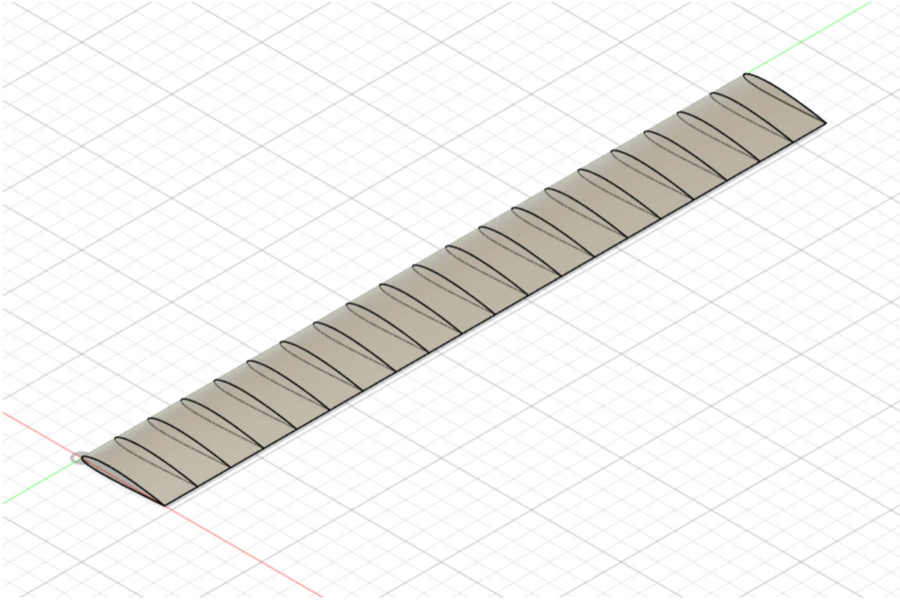

Figure 4.4: Wing isometric view

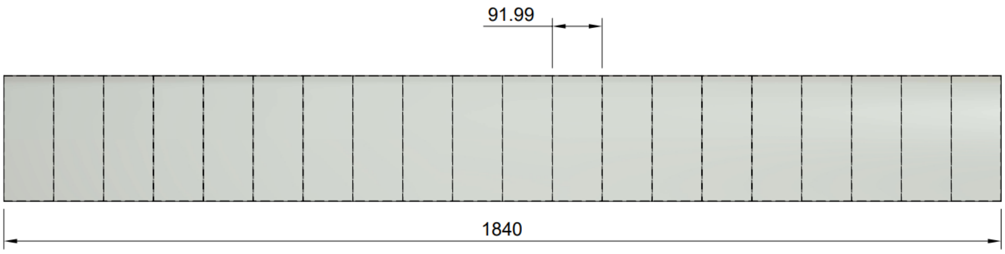

Figure 4.5: Ribs spacing

Hence the ribs are placed at a distance of 92 mm. **The total number of ribs to be 20**. We can rivet the ribs onto the fuselage skin by making sure that there is enough spacing to prevent stress concentration.

Chapter 5

Fuselage Design

5.1 Review of fuselage parameters and components

The fuselage is the main structural part of the Aircraft, as a wing, horizontal tail, and vertical tail are all connected to this structure at different attachments. Ensuring fuselage working is essential for operation.
Aircraft fuselage has adopted the semi-monocoque philosophy from ships. The fuselage structure includes numerous structural parts such as longerons, bulkheads, and outer skin.
The fuselage constructed from $0.5mm$ thick aluminium sheets. We have selected a rectangular cross-section fuselage. The front of the fuselage is of half ellipsoidal shape for a streamlined shape.
Dimensions of the fuselage are given below.

1. Total fuselage length: 1.69 m
2. Width: 16.81 cm
3. Height: 13.66 cm

The following components are loaded into the fuselage

- Battery
- Flight controller
- Receiver
- Wiring etc.

For structural analysis of the fuselage, a load of the aircraft is considered along with a referential distance from the nose of the aircraft. Refer Appendix A.7 for the data.

5.2 SFD & BMD of Fuselage

For generating the below presented SFD and BMD plots, we consider each component placed inside as point masses and considered their weights as loads acting on the fuselage. Please refer to Appendix A.8 for the code used.

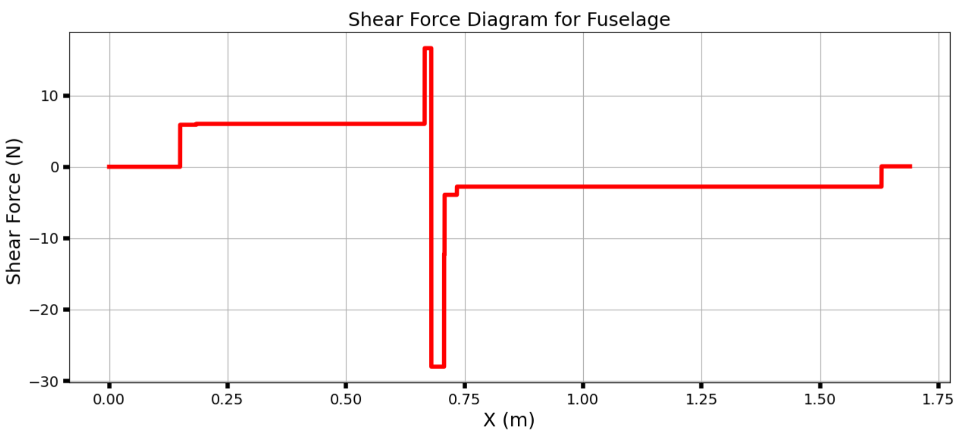

Figure 5.1: SFD of Fuselage cross-section

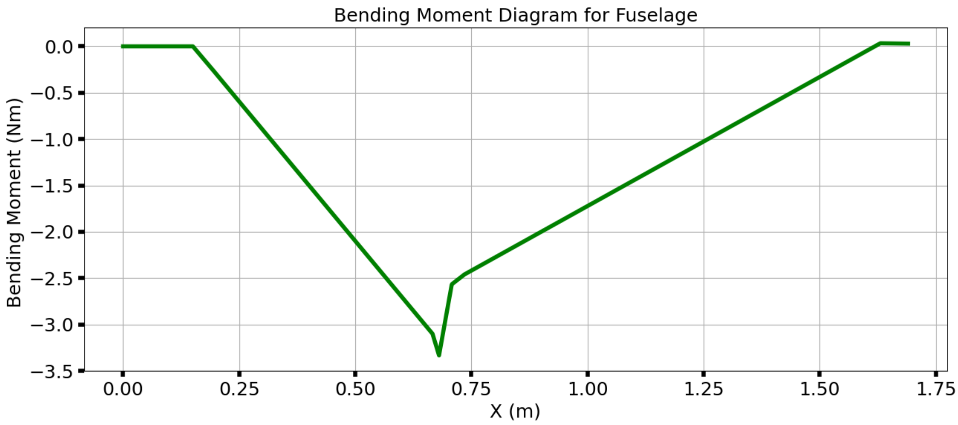

Figure 5.2: BMD of Fuselage cross-section

The maximum bending moment on the fuselage is close to 4 Nm.

5.3 Longeron Requirement

We will find the shear flow distribution in the fuselage cross-section. Using this and the bending stresses, we find the maximum stress on the fuselage skin. If that comes out to be lesser than the allowable stress ($\sigma_{allowable}$), we need to use longerons to carry the bending stress.

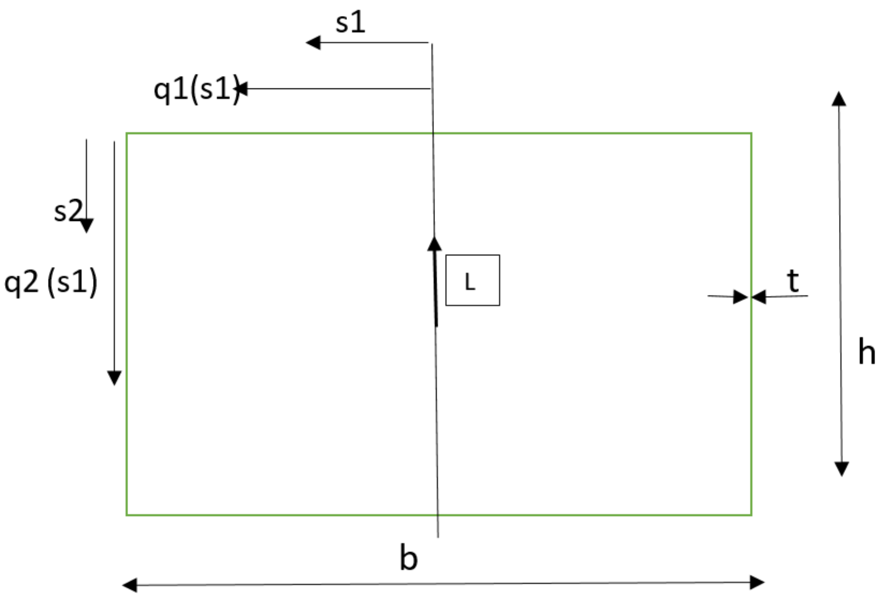

Figure 5.3: Notations for fuselage shear flow calculation

We use the above notations for calculating shear flow along the fuselage. The Load acting will be the lift on the aircraft during cruise, $L = 53.66N$. h and b are the height and width of the fuselage. $h = 13.66cm$ and $b = 16.81cm$. The moment of inertia about the horizontal axis in the above figure is given by,

$$I_{xx} = 2\left[\frac{bt^3}{12} + bt\left(\frac{h}{2}\right)^2 + \frac{th^3}{12}\right] = 996579.5mm^4$$

Due to symmetry, we know that the shear flow in the middle of the horizontal side will be 0 [$q_1(s_1) = 0$]. By considering a small element ds_1 at a distance of s_1 from the starting point ($q_1 = 0$) for the horizontal part respectively, can write,

$$q_1(s_1) = \frac{Lhs_1t}{2I_{xx}}$$

for $0 < s_1 < b/2$ Similarly, if we consider s_2 from top of the vertical section to the center-line, $0 < s_2 < h/2$, we can write,

$$q_2(s_2) = q_1(b/2) + \frac{L(h-s)st}{2I_{xx}}$$

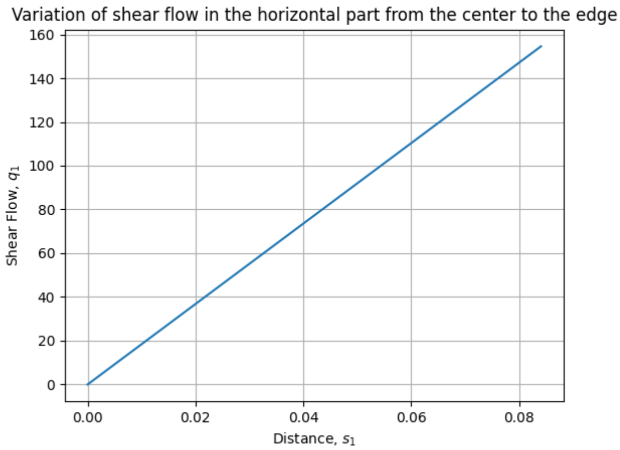

Figure 5.4: Shear Flow in horizontal section

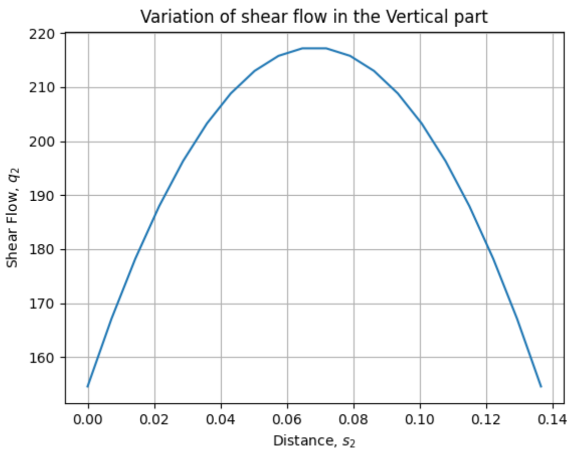

Figure 5.5: Shear Flow in Vertical section

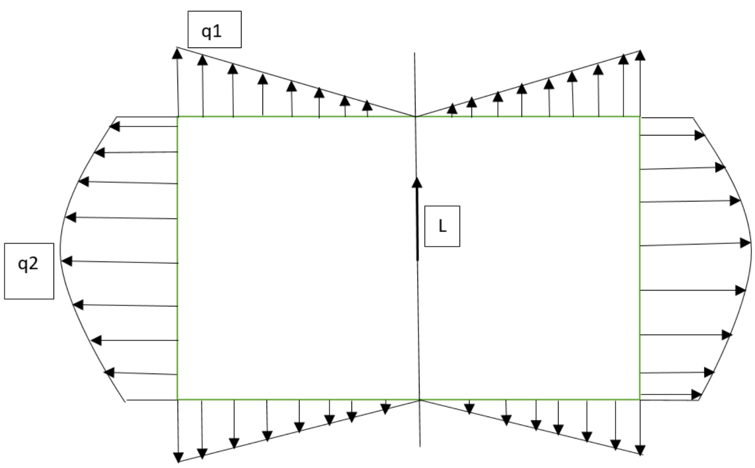

Figure 5.6: Shear Flow variation in the fuselage

Now we have to check if the maximum of the combination of shear stress and the bending stress due to bending of fuselage is lesser than the critical buckling load of fuselage.

$$F_{net} = \frac{q}{t} + \frac{M_{max_{fus}} y}{I_{xx}}$$

The maximum value of F_{net} comes out to be,

$$F_{max} = 0.583 N/mm^2$$

and this occurs at the top corner of the fuselage.
This should be less than the $\sigma_{allowable}$ of the material. $\sigma_{allowable} = 25.71 N/mm^2$. So, we have $F_{max} < \sigma_{allowable}$. So, we won't be needing longerons to carry the bending moment of the fuselage. However, we need longerons to attach the bulkheads. So, we will be using longerons of dimensions 10 mm x 10 mm x 0.5 mm.

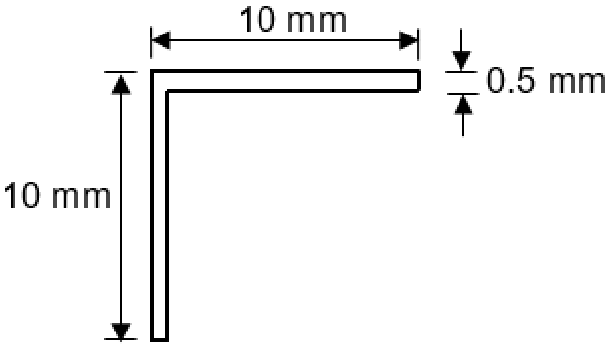

Figure 5.7: Longeron cross-section

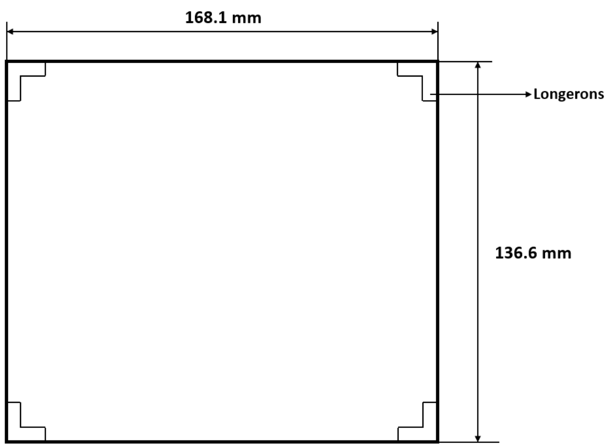

Figure 5.8: Fuselage sizing with longerons

Chapter 6

Bulkhead Sizing

6.1 FBD and equilibrium equations

The aim is to find a cross section of bulkhead needed. The idea is to use the shear flow in the fuselage found earlier to analyse the shear force, axial force and the bending force acting on one half of the bulkhead. The FBD is such that we attach the spar onto the the center of the bulkhead by riveting because the half the lift and bending moment due to wing is shown to be acting at the center of the half-bulkhead. This essentially means that there will be rivets concentrated onto the center of the bulkhead which we will not do in actual fabrication. We will ensure that there is enough spacing which takes care of stress concentration. This assumption is such that it simplifies calculation.

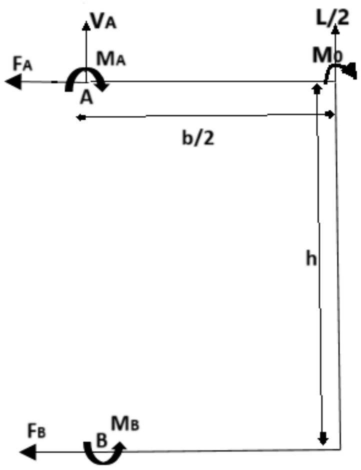

Figure 6.1: Half of bulkhead

6.1.1 Top Member

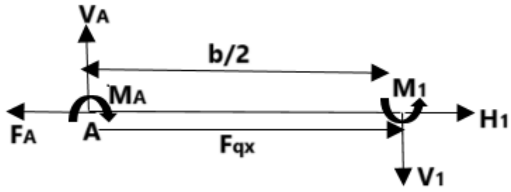

Figure 6.2: FBD of the top horizontal member

V is the shear force, H is the axial force and M is the bending moment for all further analysis of bulkhead. Moment balance will be taken about A.

We can write the equilibrium equations with moment taken about A,

$$V_1 = V_A$$
$$H_1 = F_A - F_{q_x}$$
$$M_1 = M_A + V_1 x$$

where,

$$F_{q_x} = \int_0^x q_1 dx = \frac{Lhx^2 t}{4I}$$

as found from the shear flow analysis.

6.1.2 Vertical Member

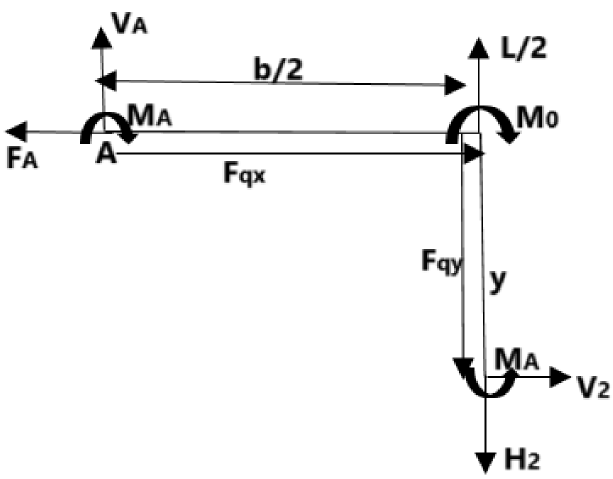

Figure 6.3: FBD of the vertical member

The equilibrium equations can be written with moment taken about the top right corner,

$$V_2 = F_A - F_{q_x}(b/2)$$
$$H_2 = \frac{L}{2} - F_{q_y} + V_A$$
$$M_2 = M_A + M_0 - V_2 y + \frac{V_a b}{2}$$

where,

$$F_{q_x}(b/2) = \frac{Lhb^2 t}{16I}$$

and,

$$F_{q_y} = \int_0^y q_2 dy = \frac{Lhbty}{4I} + \frac{Lt}{2I}\left[\frac{hy^2}{2} - \frac{y^3}{3}\right]$$

6.1.3 Bottom Member

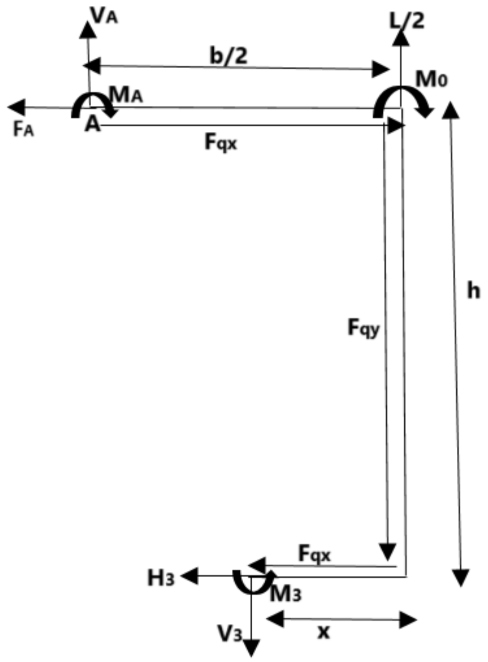

Figure 6.4: FBD of the bottom horizontal member

We can write the equilibrium equations with moment taken about the top right corner,

$$V_3 = \frac{L}{2} - F_{q_y}(h) + V_A$$
$$H_3 = -F_A + \frac{Lht}{4I}\left(\frac{b^2}{4} - x^2\right)$$
$$M_3 = M_A + M_0 + F_{q_x}h + H_3h - V_3x + \frac{V_ab}{2}$$

where,

$$F_{q_y}(h) = \frac{Lh^2bt}{4I} + \frac{Lh^3t}{12I}$$

from the shear flow calculation.
Now, we have to use Castigliano's second theorem to estimate the energy of the beam.

$$U_{bending} = \frac{1}{2}\int_0^L \frac{M^2}{EI}dx$$
$$U_{axial} = \frac{1}{2}\int_0^L \frac{H^2}{EA}dx$$
$$U_{shear} = \frac{1}{2}\int_0^L \frac{V^2}{GA}dx$$

We have to do this for all three members and sum them up.

$$U_{total} = U_{bending} + U_{axial} + U_{shear}$$

We can do a failure analysis. To find the reaction forces and moments M_A and F_A, we can use the energy method. We first assume a virtual displacement and a virtual rotation due to reaction forces and moments which is directly related to the partial derivatives of total strain energy with respect to the forces and moments. Then we can say that the displacements and rotations are zero because the ends are fixed. Therefore,

$$\frac{\partial U_{\text{total}}}{\partial M_A} = 0$$

$$\frac{\partial U_{\text{total}}}{\partial F_A} = 0$$

After calculating using a Python program (refer Appendix), we find the values, $F_A = 12.99N$, $M_A = -28.09Nm$ and $V_A = -13.415N$. We substitue these back into the equilibrium equations and find the bending moment of bulkhead.

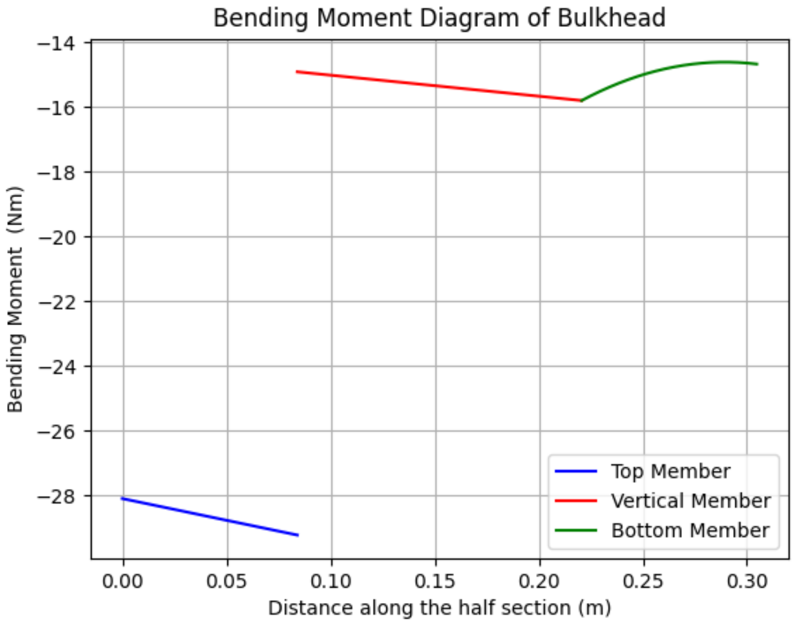

Figure 6.5: BMD of bulkhead

The magnitude of maximum bending moment comes out to be $-29.375Nm$. We can use this bending moment to calculate the moment of inertia required for the cross section of bulkhead.

$$\frac{M_{max}y_{max}}{I_{req}} \leq \sigma_{allowable}$$

We choose C-section to be the cross-section of bulkhead. We have, $M_{max} = 29.375Nm$, $\sigma_{allowable} = 25.71MPa$.

$$\frac{I_{req}}{y_{max}} \geq 1142.55mm^3$$

If we choose a C-section of dimensions, 35 mm X 35 mm X 1 mm, we get, $I = 23230.6mm^4$ and $y_{max} = 17.5mm$ and hence, $\frac{I_{req}}{y_{max}} = 1327.46mm^3$, which satisfies the requirement.
The cross section of bulkhead is shown below.

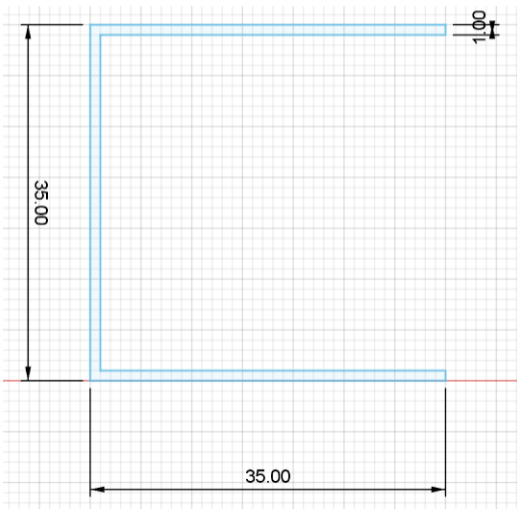

Figure 6.6: bulkhead cross-section, units are in mm

The CAD model of the bulkhead is shown below.

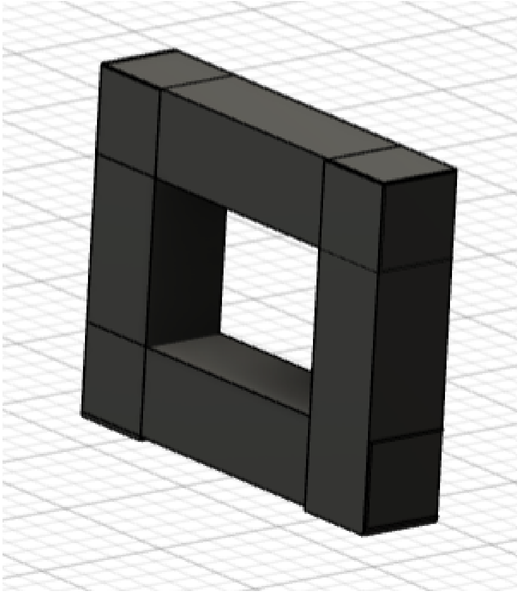

Figure 6.7: bulkhead CAD model

Now, we need to figure out the number and locations of the bulkheads. We will keep one at each landing gear locations apart from the one joined to the spar of the wing.

6.2 Assembly of Components

We don't need spars as per our calculations. So, we are using rods for rotating the control surfaces. These rods will pass through the ribs and we will connect the control surfaces to it. We need one rod for each aileron and a single rod connecting both flaps. The ribs will be designed in such a way that there will be holes for the rods to pass. The assembly of structural components are shown below.

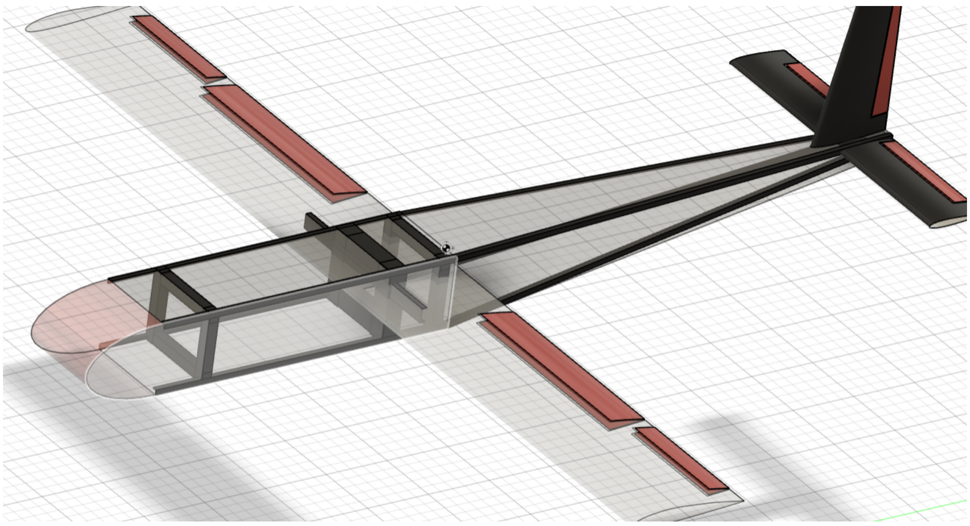

Figure 6.8: Structural components

The weights of the structural components are given below:

Component	Weight	Number	Total Weight (g)
Spar	22.22	1	22.22
Longerons	60.49	4	241.96
Bulkhead (to connect spar)	169.5	1	169.5
Bulkhead (to connect LG)	48.54	2	97.08

Chapter 7

Landing Gear Design

7.1 Configuration

No.		Single main	Bicycle	Tail-gear	Nose-gear	Quadricycle	Multi-bogey	Human leg
1	Cost	9	7	6	4	2	1	10
2	Aircraft weight	3	4	6	7	9	10	1
3	manufacturability	3	4	5	7	9	1	10
4	Take-off/landing run	3	4	6	10	5	8	2
5	Stability on the ground	1	2	7	9	10	8	5
6	Stability during taxi	2	3	1	8	10	9	–

10: best, 1: worst.

Figure 7.1: Table for comparing different configurations, from Sadrey

The preferred choice is the tricycle landing gear configuration. Some features are:

- Tricycle is the most widely used landing gear configuration.
- The wheels aft of the aircraft CG are very close to it compared to the forward gear and carry 80-90% of the aircraft weight.
- Both main and nose gears have the same height, so the aircraft is level on the ground, although the main gears tend to have larger wheels.
- A nose-gear configuration aircraft is directionally stable on the ground as well as during taxiing.

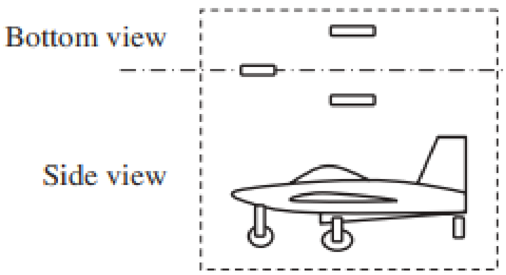

Figure 7.2: Tricycle Landing gear configuration, taken from Sadrey

7.2 Notations

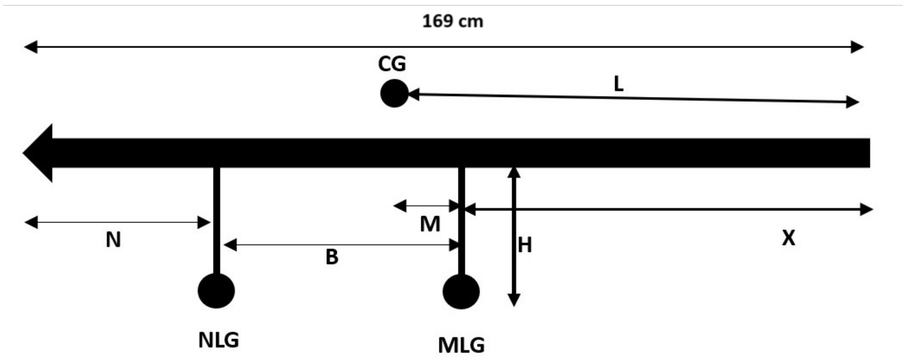

Figure 7.3: Rough Sketch of Landing Gear Position

From tail and fuselage design, we found the Fuselage length = 169cm. We are assuming the main landing gear to carry 85% load and the nose landing gear to carry the remaining 15%load as given in Sadrey.

The other parameters are as follows:

- Location of CG from the aft end = L
- Landing gear height = H
- Wheel Base = B
- Distance of Main Landing Gear (MLG) from tail = X
- Distance of Nose Landing Gear (NLG) from Nose = N
- Distance between CG and MLG = M
- Take off attitude of aircraft = α

As an initial estimate we take L = 1.2 m from the aft end to be the location of CG, which is slighly higher than the Tail moment arm.

Taking moment about CG,

$$0.15W \times N = 0.85W \times M$$
$$N + M = B$$

We get, distance between CG and MLG =$M = 0.15B$.
After estimating Wheel base, B, we can use these relations to estimate the values of the parameters. We can later use these values for stability analysis to finalise them.

7.3 Landing Gear Height

Landing gear height is estimated so that none of the components strike the surface while the aircraft is taxing. The components which can strike the ground during ground roll are the camera and the propeller. We are using a 15 inch diameter propeller. propeller is attached on the wing.

- Height of fuselage = 13.66 cm.
- Height of camera = 13.9 cm.
- Radius of propeller = 18.9 cm.

So, propeller reaches 18.9 - 13.66 = 5.24 cm below the fuselage. So, the camera is the longest component below the fuselage extending to about ¡camera height ¿cm. We take the landing gear height to be 20cm which gives enough clearance for the camera and propeller.

$$H = 21.56cm$$

7.4 Main Landing gear position

Sadrey mentions that 10–15 deg, so the tipback angle must be equal to or greater than 15–20 °. We take that angle as 16°. With this and landing gear height, $H = 21.56cm$ and fuselage height of 13.66 cm, we get distance from tail to position of main landing gear, $X = 99cm$.

7.5 Ground Clearance and Takeoff Angle

Sadrey mentions that the takeoff angle, α_{TO} should be 5° less than the tipback angle. Hence, we get $\alpha_{TO} = 11°$.

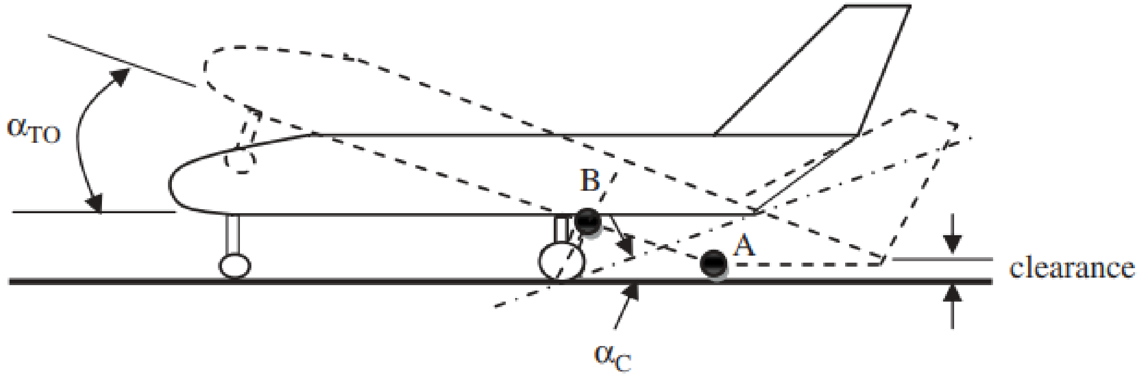

Figure 7.4: Ground clearance, from Sadrey

7.6 Wheel Base

Wheel base (B) plays an important role in the load distribution between primary (i.e., main) gear and secondary (e.g., nose or tail) gear. This parameter also influences the ground controllability and ground stability.Due to the ground controllability requirement, the nose gear must not carry less than about 5% of the total load and also must not carry more than about 20% of the total load (e.g., aircraft weight). Thus, the main gear carries about 80–95% of the aircraft load. To meet this requirement, it is decided that the nose gear should carry 15% of the total load and the main gear 85% of the total load.

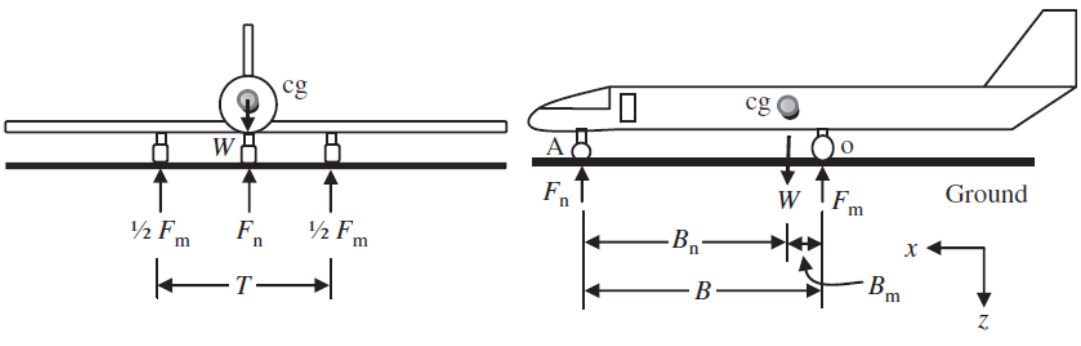

Figure 7.5: Wheel Base

The nose landing gear would be kept where the nose ends and fuselage starts, which is kept at 15cm, from the nose end. Since, $X = 99cm$ and total length of plane, $L = 1.69m$ we get the wheel base, B as ,

$$L = B + 99cm + 15cm$$
$$B = 55cm$$

Therefore,

1. Wheel Base (B) = 55 cm.
2. Distance of nose gear from CG (B_n) = 0.81 B = 44.55 cm.
3. Distance of main gear from CG (B_m) = 0.19 B = 10.45 cm.

7.7 Wheel Track

Wheel Track (T) is the distance between the right and leftmost wheels when looking at the front view. Three main requirements which drive the magnitude of wheel track are : 1) Ground lateral control, 2) Ground lateral stability, and 3) Structural integrity.
To determine the wheel track, the overturn angle is introduced. When looking at the front view the angles between the vertical line passing through the cg and line between aircraft cg and main wheel is known as overturn angle.

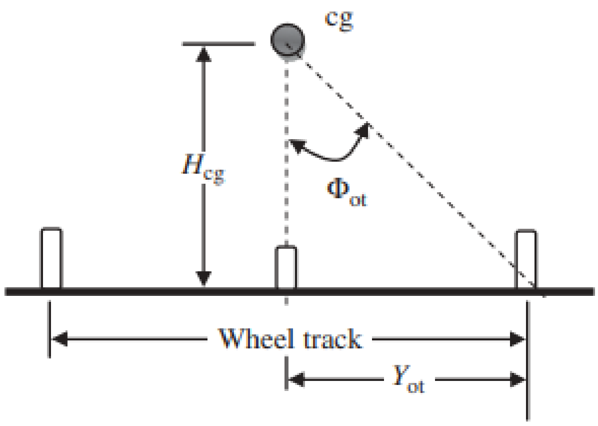

Figure 7.6: Overturn Angle

The overturn angle ϕ_{ot} is inside the following limit:

$$\phi_{ot} \geq 25^o$$

As seen from the figure,

$$\tan \phi_{ot} = \frac{Y_{ot}}{H_{cg}}$$

$$Y_{ot} \geq H_{cg} \tan 25^o$$

$$Y_{ot} \geq (14.73 + 0.5 \times 13.66)(0.46630)$$

$$Y_{ot} \geq 10.05\ cm$$

Also,

$$T = 2Y_{ot}$$

Hence,

$$T \geq 20.1\ cm$$

We keep the wheel track, $T = 20.1$ cm for now.
The overturn angle and wheel track can be calculated more precisely after detailed design & sizing.

7.8 Tire sizing

The tires are sized to carry the weight of the aircraft. For a small fixed wing UAV, a solid, non-inflatable tire made of foam or plastic is generally used.

7.8.1 Diameter and Width of Tire

The diameter and width of main tires can estimated using,

$$d = A{W_w}^b$$

where d is diameter(cm), W_w is weight on the wheel, $A = 5.1$ and $b = 0.349$ for a general aviation aircraft. Since, for a smaller fixed wing UAV, we need considerably lower diameter of tire, we take $A = 2.75$ **and** $b = 0.25$. we can get W_w by,

$$2W_w = \frac{B_n}{B} W$$

Weight of our aircraft is 6.51 kg from second weight estimate, B = 0.55 m and $B_n = 0.1045$ m.

$$W_w = 21.68N$$

This is also the **max static load of the main wheel.**

$$d = 2.75 \times (21.68)^{0.25}$$

$$d \approx 6cm.$$

Examining tire data from Raymer, we get that the ratio of width to diameter of tires is roughly 0.33. So using this ratio as a preliminary estimate, we get width, w of tire = $0.33 \times d = 1.98cm$.

Therefore the diameter and width of main tire is 6 cm and 1.98 cm respectively.

The nose tires can be about 60-100 % of the main tire size. So the diameter and width of nose landing gear can be from 3.6 - 6 cm and 1.19 - 1.98 cm respectively.

7.8.2 Static and Dynamic Loading

The static loads on each main wheel and nose gear wheel was already found to be 27.145 N and 9.58 N respectively.

Sadrey gives the following forumlas for calculation of dynamic loads.

For nose landing gear,

$$F_{n_{dyn}} = \frac{a_L W H_{cg}}{gB}$$

for main landing gear,

$$F_{m_{dyn}} = \frac{a_T W H_{cg}}{gB}$$

where, a_T and a_L are the magnitudes of takeoff and landing accelerations respectively, H_{cg} is the height of the cg from the ground and W is the weight of the aircraft.

For our UAV, we know W = 5.47g.

The cg can be assumed to be at the centre of the overall height of the aircraft for the time being.So,

$$H_{cg} = H + 0.5 \times H_{fuselage} = 21.56 + 0.5 \times 13.66 = 28.39cm$$

We can assume a typical braking coefficient of 0.3, which gives a deceleration of 0.3g. So, we can assume an acceleration while takeoff as $4m/s^2$. These are an upper limit. So, we get an upper bound of dynamic load. So,

$$F_{n_{dyn}} = 0.3 \times 5.47 \times 9.81 \times \frac{28.39}{55} = 8.31N$$
$$F_{m_{dyn}} = 4 \times 5.47 \times \frac{28.39}{55} = 11.29N$$

Net load is a summation of static and dynamic loads.

$$F_n = 10.19 + 8.31 = 18.5N$$
$$F_m = 43.46 + 11.29 = 54.75N$$

7.9 Compilation of Parameters

Parameter	**Value**
Wheel Base	55 cm
Wheel Track	20.1 cm
Landing Gear Height	21.56 cm
Clearance Angle	16°
Tire Diameter	6 cm
Tire Width	1.98 cm

7.10 Landing Gear Design

7.10.1 Bending Moment Diagram

The landing gear is used to keep the UAV stable on ground, and able to move it while takeoff, and safe while landing. For designing the landing gear we first plot the Bending Moment Diagram (BMD) of the landing gear.

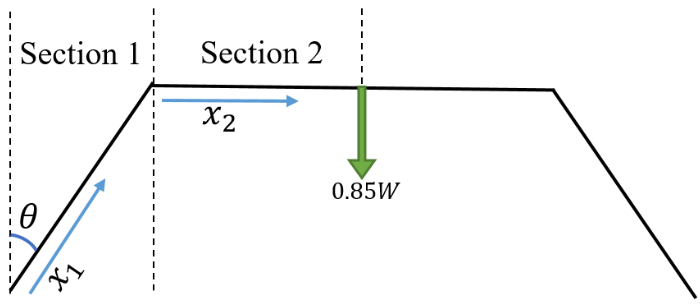

Figure 7.7: Landing Gear Schematic Diagram

We now plot the the BMD, using two equations for section 1 and section 2. These sections are shown in the Figure 7.7. For section 1,

$$M_1 = M_0 + x_1 F \sin(\theta); \quad x_1 \in [0, L] \tag{7.1}$$

For section 2,

$$M_2 = M_0 + F(x_2 + L\sin(\theta)); \quad x_2 \in \left[0, \frac{T}{2} - L\sin(\theta)\right] \tag{7.2}$$

where,

$$F = 0.85 \times \left(\frac{W}{2}\right)$$
$$M_0 = F \times \left(\frac{w_t}{2}\right)$$
$$T = \text{Wheel Track} = 20.1cm$$
$$W = \text{weight of UAV} = 5.47kg = 53.6607N$$
$$w_t = \text{width of tire} = 1.98cm$$

For the other half of the landing gear, the same equations follows. Using these equations, coded and plotted the BMD. It is shown in Figure7.8. The code for the plot of this BMD is available in Appendix A.10

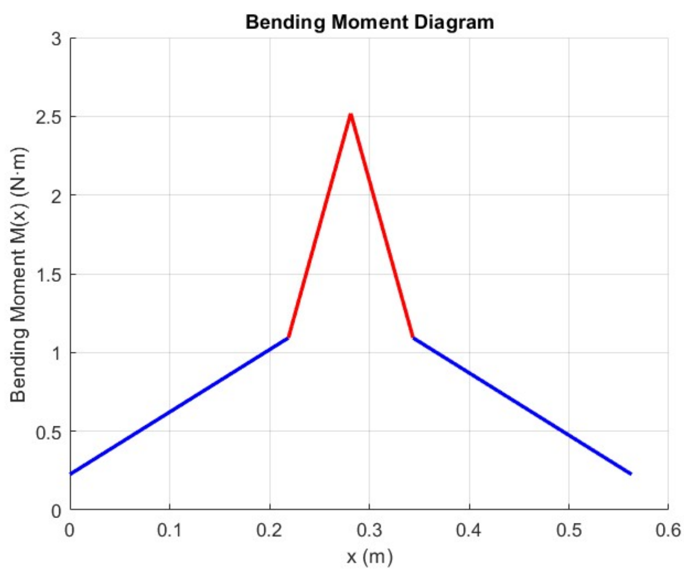

Figure 7.8: Landing Gear Bending Moment Diagram

Using this BMD plot, the maximum Bending moment is found to be M_{max} = **2.5177 N.m**, occurring at the mid point along lateral direction.

7.10.2 Longitudinal analysis

Now, we have the maximum Bending moment, using the below equation we will find the dimenions of the cross section of the landing gear. We are using a hollow square section.

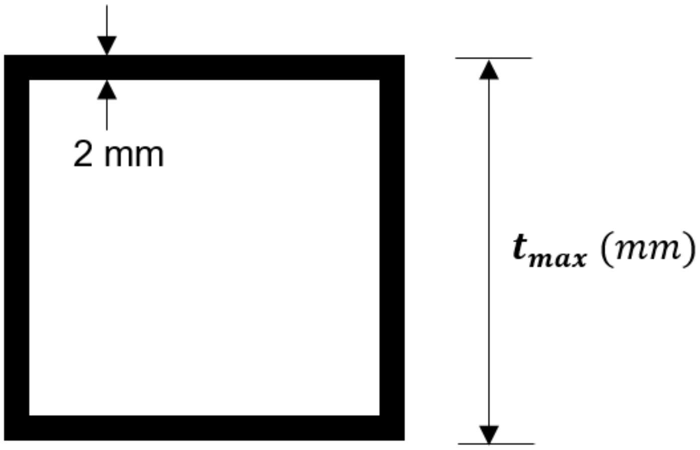

Figure 7.9: Possible Main Landing Gear Cross section

$$I_{required} \geq \frac{M_{max} \cdot y}{\sigma_{allowable}} \tag{7.3}$$

where,

$$\begin{aligned}
M_{max} &= 2.5177\text{N.m} = 2517.76\text{N.mm} \\
y &= \frac{t_{max}}{2} \\
\sigma_{allowable} &= \frac{\sigma_{yield}}{\eta_{\text{fatigue}} \cdot k \cdot f} = \frac{270}{1.5 \cdot 2 \cdot 1.4} = \frac{270}{4.2} \\
I_{required} &= \frac{1}{12}\left(b_{\text{outer}}^4 - b_{\text{inner}}^4\right) \\
&= \frac{1}{12}\left(t_{\text{max}}^4 - (t_{\text{max}} - 4)^4\right)
\end{aligned}$$

From equation[7.3],

$$\begin{aligned}
\frac{1}{12}\left(t_{\text{max}}^4 - (t_{\text{max}} - 4)^4\right) &\geq \frac{2517.76 \cdot \frac{t_{max}}{2}}{\frac{270}{4.2}} \\
\implies t_{max} &\geq 6.204 \text{ mm}
\end{aligned}$$

7.10.3 Lateral analysis

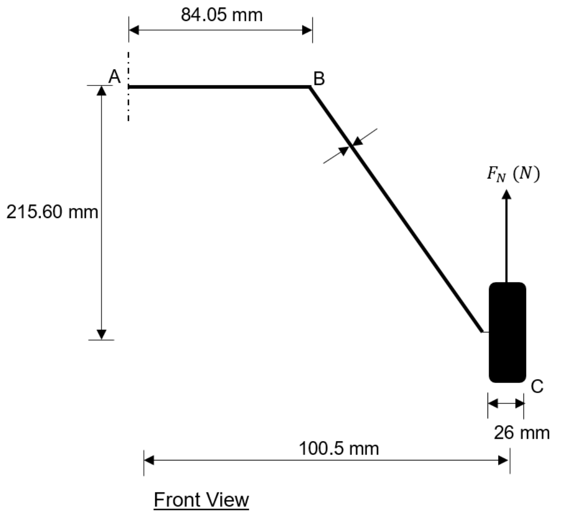

Figure 7.10: Front View of Main Landing Gear

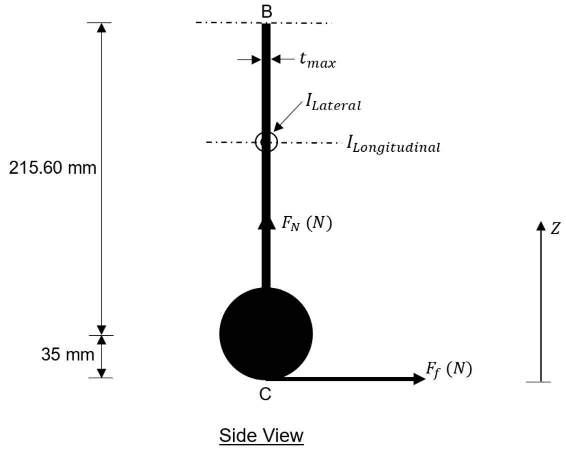

Figure 7.11: Side View of Main Landing Gear

Moment acting about Lateral axis, along z-direction can be expressed as,

$$M_{\text{Lateral}} = F_f \cdot z \quad \text{(N-mm)}$$
$$M_{\text{Lateral}} = [\mu \cdot F_N] \cdot z$$

where,

$$M_{\text{Lateral}} = \text{Moment about } I_{\text{Lateral}}, \text{ (N-mm)}$$
$$F_f = \text{Frictional Force on tire, (N)}$$
$$\mu = \text{Kinetic friction coefficient, (Dimensionless)}$$
$$F_N = \text{Normal Weight on tire, (N)}$$

Here both MLGs are taking 85% of total weight, divided equally. For worst case, UAV lands on only one MLG with total weight. So,

$$F_N = W$$
$$F_N = 5.47 \times 9.81 = 53.66\,\text{N}$$

Polyurethane material is to be used for suitable tire dimensions. The reason behind choosing Polyurethane, is the commercial availability and the mechanical properties being almost same as that of rubber. So, kinetic friction co-efficient (μ) to be kept as 0.75.

$$M_{\text{Lateral}} = [0.75 \times 53.66] \cdot z = 40.25z \quad \text{(N-mm)}$$
$$\text{At, } z = 215.60 + 35 = 250.60\,\text{mm}$$

Maximum $M_{Lateral}$,

$$M_{\text{Lateral,max}} = 40.25 \times 250.60 = 10085.53\,\text{N-mm}$$

$I_{Lateral}$ of MLG Strut

Governing Equation for Maximum stress developed in the MLG Strut cross section,

$$\sigma_{\text{Allowable}} \leq \frac{\sigma_{\text{Yield}}}{\eta_{\text{fatigue}} \cdot k \cdot f} \tag{7.4}$$

where, $k = 2$ is the stress concentration factor, $\eta_{fatigue} = 1.5$ is the fatigue factor and $f = 1.4$ is the factor of safety. $\sigma_{Allowable}$ because of Maximum Bending Moment ($M_{\text{Lateral,max}}$) generated, can be expressed as,

$$\frac{\sigma_{\text{Yield}}}{4.2} \geq \frac{M_{\text{Lateral,max}} \cdot \left(\frac{t_{\max}}{2}\right)}{I_{\text{Lateral}}}$$
$$I_{\text{Lateral}} \geq \frac{M_{\text{Lateral,max}} \cdot \left(\frac{t_{\max}}{2}\right)}{\left(\frac{\sigma_{\text{Yield}}}{4.2}\right)}$$

For AL-6063-T6, $\sigma_{\text{Yield}} = 270\,\text{MPa}$. And considering $M_{\text{Lateral,max}} = 10085.53\,\text{N-mm}$,

$$I_{\text{Lateral}} \geq \frac{2.1 \cdot 10085.53 \cdot t_{\max}}{270} \left(\text{mm}^4\right),$$
$$I_{\text{Lateral}} \geq 78.443\, t_{\max} \left(\text{mm}^4\right)$$

The landing gear strut is made from AL-6063-T6 grade metal. $I_{Lateral}$ of cross-section can be calculated to be as of a simple hollow square beam having thickness of 2mm and depth. Hence,

$$I_{\text{Lateral}} \geq 78.443 t_{\max}$$
$$\frac{1}{12}\left(t_{\max}^4 - (t_{\max} - 4)^4\right) \geq 78.443 t_{\max}$$
$$t_{\max} \geq 10.306\,\text{mm}$$

So, the minimum width of the hollow square cross-section strut (thickness 2 mm) required is 11 mm, so that the strut design can withstand bending along I_{Lateral}.

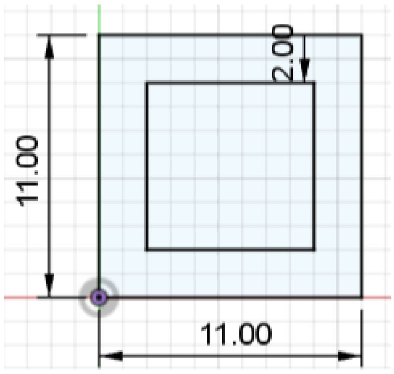

Figure 7.12: Main Landing Gear Cross Section

7.11 Buckling Analysis

Compressive load acting on the landing gear struts can cause it to buckle under a specific load (F_{cr}). This load can be identified using the Euler buckling formula.

7.11.1 Nose Landing Gear

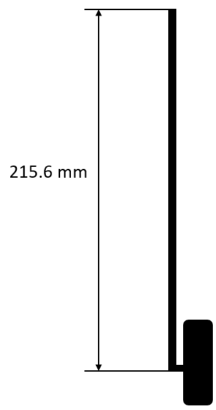

Figure 7.13: Nose landing gear dimension (one-half)

Nose landing gear is designed to take 15 % of weight. Hence

$$\sigma_{NLG} = \frac{0.15 \times W}{A}$$

W = Weight, A = Cross-sectional area of landing gear

From Euler Buckling formula,

$$\sigma_{cr} = \frac{\pi^2 EI}{(kL)^2}$$

Where,

k = 0.707 (For One end Fixed- One end pivot),
I = Least Moment of Inertia = $\frac{bt^3}{12}$,
E = Young's Modulus = 68.9×10^9 GPa,
L = Length of the landing gear = 216 mm (Precision of 1mm).

Design should consider a factor of magnitude higher then the buckling stress. Hence,

$$\sigma_{design_{cr}} = \sigma_{cr} \times k \times f \times \eta_{fatigue} \times \text{Impact Factor}$$

Where, k (Stress concentration factor) = 2, f (Factor of safety) = 1.4, $\eta_{fatigue}$ (Fatigue factor) = 1.5, Impact Factor = 1.5
For the above critical load, we find the required I.

$$I = \frac{\sigma_{design_{cr}}(kL)^2}{\pi^2 EI}$$

$$I = 1.7391 \text{ mm}^4$$

As,

$$I = \frac{bt^3}{12}$$

This is satisfied by an L-section of dimensions 15mm x 15mm x 2mm. However, due to ease of availability, we are using an L-section of dimensions 19mm x 19mm x 2mm as available in the inventory.

7.11.2 Main Landing Gear

We find the dimensions of main landing gear by Euler Buckling formula. As longitudinal and lateral analysis of main landing gear is done considering it to be **thin square cross-section**, the same is considered for this calculation.

$$\sigma_{MLG} = \frac{F}{A} = \frac{5.47 \times 9.81 \times 0.9848}{A}$$

$$\sigma_{cr} = \sigma_{MLG} \times \eta_{impact} \times k \times f \times \eta_f$$

Where, $\eta_{impact} = 1.5$ is the impact factor, $k = 2$ is the stress concentration factor, $f = 1.4$ is the factor of safety, $\eta_{fatigue} = 1.5$ is the fatigue factor
Now,

$$\sigma_{cr} = \frac{\pi^2 EI}{A\ l_e^2} \tag{7.5}$$

$$I = \frac{\sigma_{cr} l_e^2}{\pi^2 E} = \frac{5.47 \times 9.81 \times 0.9848 \times 5.0625 \times 4 \times (0.2156^2)}{(3.14^2) \times 68.9 \times (10^9)}$$

$$b^4 - (b-2t)^4 = \frac{12 \times 5.47 \times 9.81 \times 0.9848 \times 5.0625 \times 4 \times (0.2156^2)}{(3.14^2) \times 68.9 \times (10^9)}$$

considering t = 2mm (Also considered for longitudinal & lateral bending case), solution is $b = 6$ mm. Which is less than previous cases for bending. Hence the landing gear will not buckle.
The weight estimation of the main and nose landing gears are shown below:

Component	Weight (g)
MLG	286.75
NLG	41.7

Chapter 8

Fabrication

8.1 Introduction

The fabrication process was done in parts. The aim was to manufacture all the components separately and assemble them as efficiently as possible by sustaining modularity.

8.2 Fuselage Fabrication

In order to make the fuselage, we first made the longerons and the bulkheads. Then we made the sheet metal drawing of the entire fuselage and made the skin cutout.

8.2.1 Bulkheads

Figure 8.1: Bulkheads

Three bulkheads were made - one each for the landing gears and one for the wing attachment. The wing bulkhead is of the dimensions 3.5mm x 3.5mm x 1 mm and the landing gear bulkheads have the dimensions 2mm x 2mm x 0.5mm.

8.2.2 Longerons

The longerons were simple L-sections and are four in number. However we had to fabricate the longerons running along the upswept part separately for ease.

Figure 8.2: Longerons along the main fuselage

These were 600mm of length, made of 0.5mm sheets and had a cross section of 10mm x 10mm.

Figure 8.3: Longerons along the upsweep

These were 942mm and 947mm in length for the top and bottom respectively, made of 0.5mm sheets and had a cross section of 10mm x 10mm.

8.3 Skin

The CAD drawing of the fuselage was spread out onto a sheet metal in Fusion 360.

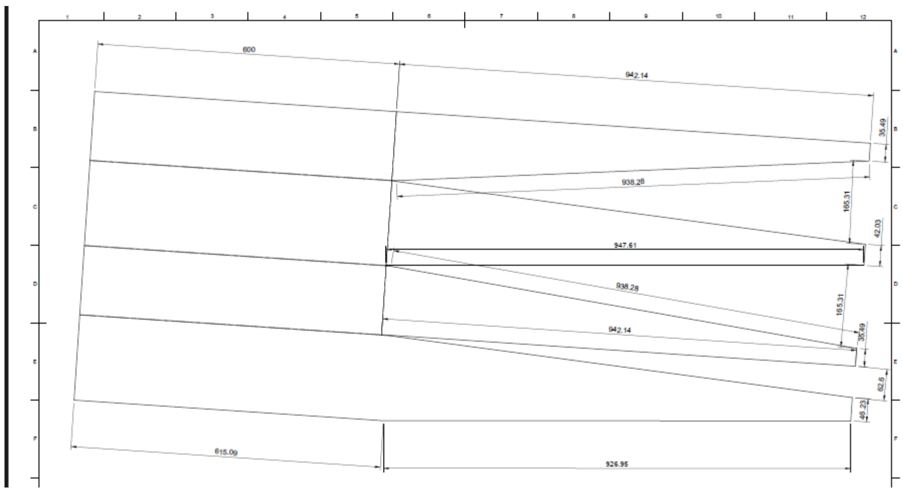

Figure 8.4: Fuselage skin drawing

8.4 Assembly

The parts were assembled in an efficient manner according to our calculations.

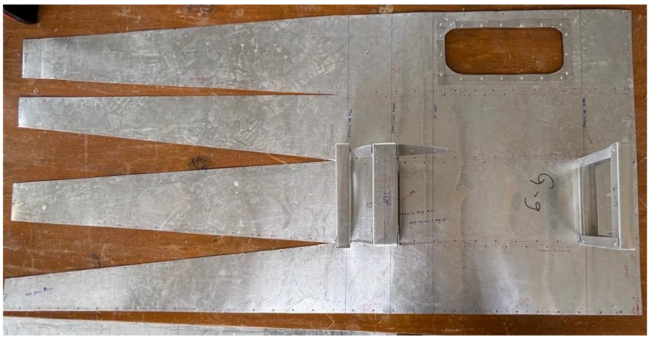

Figure 8.5: Overall assembly of components with the fuselage

The bulkheads after folding the skin is shown below.

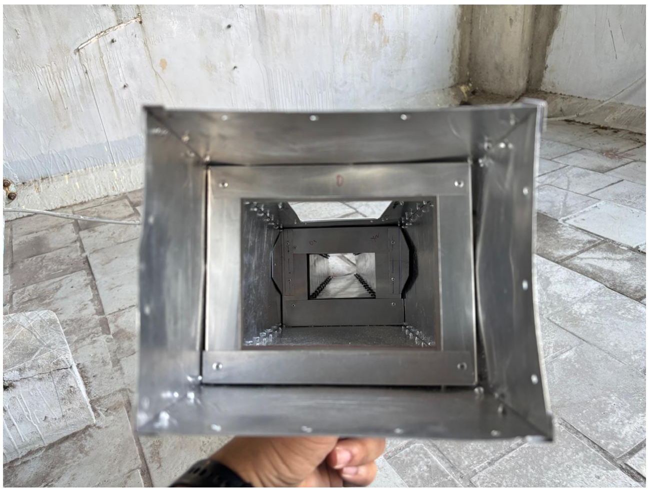

Figure 8.6: Bulkheads after folding the skin

The final finished fuselage is shown below.

Figure 8.7: Finished fuselage after folding the skin

The nose mount was 3D printed and covered with sheet metal as shown below.

Figure 8.8: 3D printed model of the nose mount

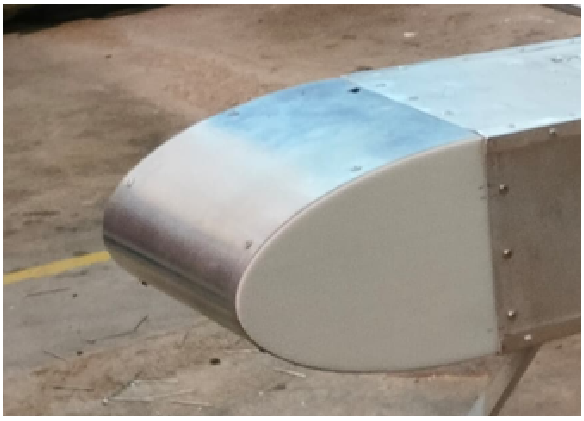

Figure 8.9: Nose mount with metal cover

Figure 8.10: Nose mount

8.4.1 Challenges Faced

- The fuselage door left the structure weak and we had to provide extra strength by attaching a rectangular strip of Aluminium around it.

- Cutting the sheet to match the dimensions of the flat pattern was a challenge and had to be done carefully.

- Folding the sheet carefully by making sure that the bulkheads remained undisturbed was a challenge.

- Hammering the longerons to bend it caused it to bend out of plane. We had to repeat it by providing a wooden block for support while hammering.

8.5 Wing Fabrication

The wing was designed using aluminium metal and 3D printed parts. The ribs, control surfaces, motor mounts and wing-to-bulkhead adapter is 3D printed. The skin and the wing-to-bulkhead coupler is of metal. Since there is no requirement of a spar, we decided to keep a coupler to connect the wing to the bulkhead through an adapter.

8.5.1 3D Printed Parts

Figure 8.11: Main Ribs of the wing

Since we have twin propeller configuration, we need ribs modified to hold the motors. These 3D-printed motor mounts are shown below.

Figure 8.12: Motor mount

The wing and the bulkhead in the fuselage are attached through a wing adapter.

Figure 8.13: Wing Adapter

8.5.2 Aluminium parts

The layout of the wing skin is shown below. The spaces cut out for the flaps, ailerons and the fuselage door are shown.

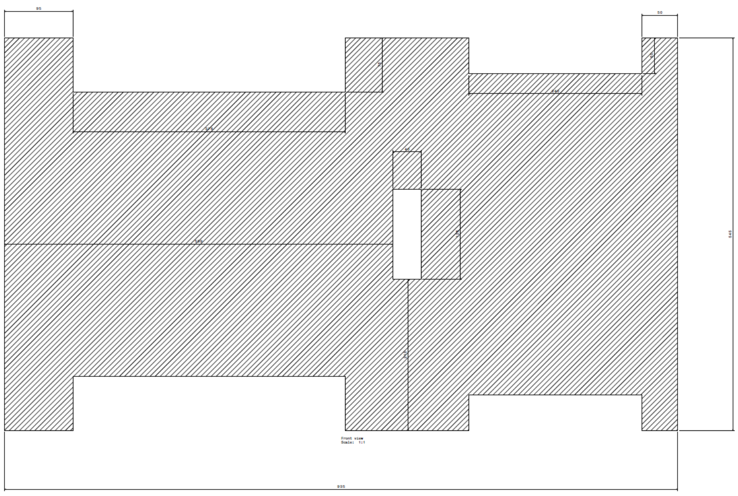

Figure 8.14: Wing skin sheet metal layout

The wing-bulkhead coupler used for joining the wing to the fuselage is as shown.

Figure 8.15: Wing-bulkhead coupler

The finished wing is shown below.

Figure 8.16: Structurally completed wing

8.5.3 Challenges Faced

- The 3D printed parts had delays in printing as well as the small, careless yet time-taking-to-rectify errors we made in designing cost us a lot of time during the fabrication process.
- Wrapping the skin around the ribs was a difficult task as the riveting had to be done beforehand to make sure the process is smooth. The trailing edge had to be sharp and this required some extra effort.
- We made the mistake of not keeping enough space for the slots for the control surfaces - the flaps and the ailerons. We had to melt away some of the material to enable the motion of the control surfaces.
- Since we have twin propeller configuration, we had to leave space for the motors before folding the wing skin. This required some careful design of the motor mounts for 3D printing as well as precise measurements for cutting the slots.

8.6 Tail Fabrication

The horizontal and vertical tails were manufactured separately and attached to the fuselage by a tail adapter which was 3D-printed.

8.6.1 Horizontal Tail

The flat sheet pattern for the Horizontal tail is given below.

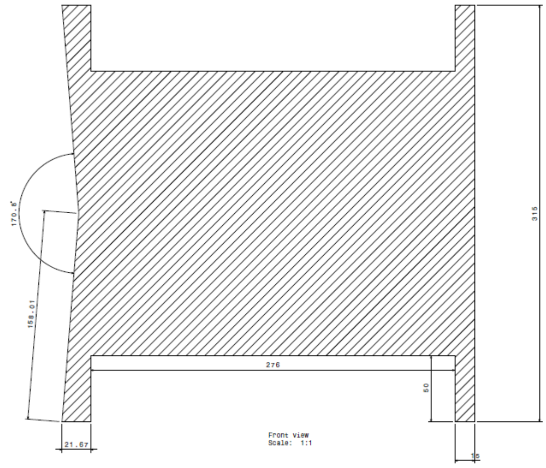

Figure 8.17: Horizontal Tail Layout

8.6.2 Vertical Tail

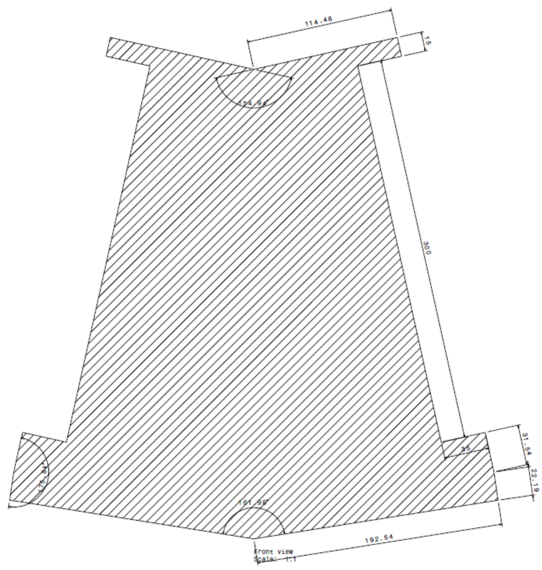

Figure 8.18: Vertical Tail Layout

A 3D-printed tail adapter is used to attach the horizontal and vertical tails onto the fuselage.

Figure 8.19: Tail Adapter

The final finished tail assembly is shown below.

Figure 8.20: Manufactured Tail Assembly

8.6.3 Challenges Faced

- The 3D prints were an essential part of the fabrication of the tail - for the tail adapter and the control surfaces. Unforeseen delays in getting the prints and unexpected changes in dimensions after getting the prints delayed the fabrication process.
- Inserting the tail adapter to the fuselage body was challenging as the holes for rivets had to be precisely placed. In some cases, we had to drill holes after aligning the adapter with the fuselage which proved to be cumbersome.
- It was seen that there was a tilt in the tail after the finish. It leads to reason that there was an issue with inserting the tail adapter despite the care taken to minimize as much as possible.

8.7 Landing Gear Fabrication

The Main and Nose landing gear after fabrication are shown here.

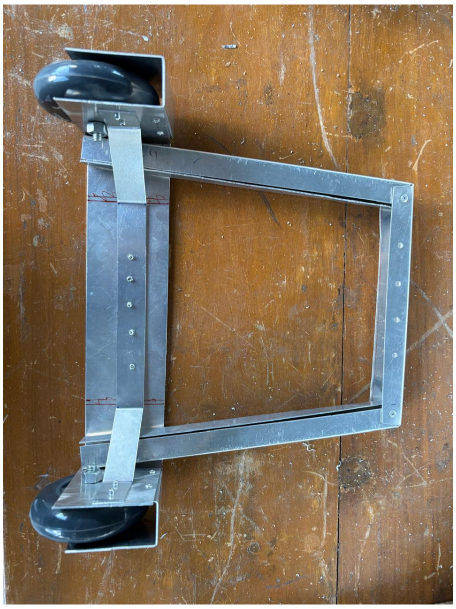

Figure 8.21: Main Landing Gear

Figure 8.22: Nose Landing Gear

8.7.1 Challenges Faced

- The Main Landing Gear after the first design had less strength and was susceptible to bending. So, we added a flat plate connecting both the sides above the wheels which arrested this bending. This will increase the drag. A modification could be to make this addition using an aerodynamic shape which will not add as much drag as a flat plate.
- The nose landing gear had severe bending problems which is yet to be addressed. It was seen that after giving a load on the UAV, the nose gear keeps bouncing which suggests some additional support needs to be added.

8.8 Final Assembly

The overall MTOW comes out to be around 6 Kg without the electronics.
The final fabricated UAV is shown below:

Figure 8.23: Structurally completed UAV

Chapter 9

Bill Of Materials

9.1 Avionics Components

AVIONICS COMPONENTS							
Sr No.	**Component**	**Vendor**	**Specification**	**Quantity (kg or pc)**	**Total (Rs.)**	**Lead time (days)**	**Status**
1	Battery	Robu	11.1 V, 10000 maH, 3S	1 pc	2,699	7 days	Available
2	PixHawk	Robu	Supply Voltage: 7V Firmware: Mission planner. Sensors: Gyrometer, Accelerometer, Barometer & Magnetometer	1 pc	11,296	7 days	Available
3	GPS	Robu	Input Voltage: 3.5 ∼5.5 V Position Accuracy: 2,2.5 meter Acceleartion: <4g Navigation update rate: 5Hz Max altitude: 18 km Weight: 26 grams	1 pc	1658	7 days	Available
4	ESC	Robu	Burst Current: 60A (up to 10Sec) Input Voltage(V): up to 16.8V Constant Current: 40A BEC: 5V /3A	2 pc	1832	7 days	Available
5	PixHawk Power Module	Robu	Operating Voltage:6∼28 VDC Max Input Voltage:28 V DC Max Current Sensing: 90 A Connecting Type:XT-60	1 pc	523	7 days	Available
6	Reciever	Robu	No. of channels: 10 RF Range: 2.4055-20475 GHz RF Channel: 140 Operating Voltage: 4.0∼6.5 V Net weight: 15 grams Control Distance: 400m+	1 pc	1600	7 days	Available
7	Telemetry	Robu	Air data rates up to 250kbps 500 mW maximum output power	1 pc	9512	7 days	Available
8	Servos	Robu	Weight: 9 gm Operating voltage: 3.0V∼7.2V Servo Plug: JR Stall torque @4.8V : 1.2kg-cm Stall torque @6.6V : 1.6kg-cm	5 pc	2250	5 days	Available
9	Velcro Straps	Flipkart	Dimensions: 3m x 25mm Self-Adhesive Hook and Loop Tape Roll	1 pc	199	4 days	Available
10	Connecting Wires	Amazon	Silicone Wire Length: 5m	1 pc	299	1 week	Available

9.2 Structural Components

STRUCTURAL COMPONENTS							
Sr No.	**Component**	**Vendor**	**Specification**	**Quantity (kg or pc)**	**Total (Rs.)**	**Lead time (days)**	**Status**
2	Aluminiuim Sheet	Inventory	Standard: 6061 Thickness : 0.5 mm, 1mm Dimensions : atleast 2 m x 50 cm	4 Kg (EW=3.4Kg)	1980	1 week	Available
6	Aluminium Square section	Inventory	Size: 11 x 11 x 2 mm Main Landing Gear	1	-	-	Available
8	Aluminium Angle Section	Inventory	Nose Landing Gear Dimensions: 19mm x 19mmx 2mm	1	-	-	Available
8	Landing Gear Tyres	Amazon	Size: 7.2 cm diameter	3	1400	3 weeks	Available
9	Control surface hinges	Robu	Material: Nylon Dimensions: 30 x 40mm	2 sets (8 pcs)	398	5 days	Available
10	Pull Push Rods	Robu	Material: Steel 1.2x840 mm Z type	5 pc	450	2 weeks	Available

9.3 Propulsion Components

PROPULSION COMPONENTS							
Sr No.	**Component**	**Vendor**	**Specification**	**Quantity (kg or pc)**	**Total (Rs.)**	**Lead time (days)**	**Status**
1	Motor	Robu	RPM: 62500 Thrust: 2.85Kg 400KV LiPO cells: 3-4 S Shaft dia: 4 mm	2 pc	22578	1 week	Available
2	Propeller	Robokits	15.5 inch Propeller Carbon Fiber	1 pc (2 propellers)	6835	1 week	Available

9.4 Tools

TOOLS							
Sr No.	**Component**	**Vendor**	**Specification**	**Quantity (kg or pc)**	**Total (Rs.)**	**Lead time (days)**	**Status**
1	Cutting Pliers	-	-	1 pc	-	-	Available
2	Steel cutting Pliers	-	-	1 pc	-	-	Available
3	Mallet	-	-	1pc	-	-	Available
4	Drilling Machine	-	-	1 pc	-	-	Available
5	Riveter	-	-	1 pc	-	-	Available
6	3D printer	-	-	1 pc	-	-	Available
6	Shearing Machine	-	-	1 pc	-	-	Available
6	Bending Machine	-	-	1 pc	-	-	Available
6	Drill Bit	-	-	as per size	-	-	Available
6	Rivets	-	-	as per number	-	-	Available
6	PLA	-	-	as per need	-	-	Available

Appendix A

Calculations

A.1 SFD, BMD

```
# %%
import numpy as np
import matplotlib.pyplot as plt

# %%
plt.figure(figsize=(15,6))
span = 1.84/2
area = 0.42
chord = 0.23
ceqroot = chord * 4 / 3.14159
cellipse = []
crect = []
cschrenk = []
xaxis = []
lx=[]
Lift = []
Liftdist=[]
SF = []
BM = []
Width = []
Iskin = 0
sigallow = 100
t = 0.12 * chord

ww = 0.95*9.81
wwf = ww/100
Weight = 5.47*9.81
for i in range(101):
    crect.append(chord)
    x = i*span/100
    xaxis.append(x)
    ce = ceqroot*(1-(x/span)**2)**0.5
    cellipse.append(ce)
    cschrenk.append((ce+chord)/2)

plt.plot(xaxis,cschrenk,'r--',label= "Schrenk's Chord")
plt.plot(xaxis,cellipse,'g--', label = "Equivalent Elliptic Wing")
plt.plot(xaxis,crect,'b--', label="Actual Wing")
plt.xlabel("Span(m)")
plt.grid(True)
plt.ylabel("Chord (m)")
plt.legend()
plt.show()
```

```
AreaUC = np.trapz(cschrenk,xaxis)
k = Weight / (2*AreaUC)
for i in cschrenk:
    Lift.append(i*k/100)
    Liftdist.append(i*k)

for x in xaxis:
    lx.append(x)
    lx.append(-x)
lx.sort()
Liftdist.sort()
for i in cschrenk:
    Liftdist.append(i*k)
for i in range(len(Lift)):
    if(i==0):
        SF.append(-Lift[i]+(Weight)/2)
        BM.append(0)
    else:
        SF.append(SF[i-1] - Lift[i] )
        product = [a * b for a,b in zip(Lift[0:i][::-1],xaxis[0:i])]
        BM.append(sum(product))
plt.figure(figsize=(15,6))
plt.xlabel("Span (m)")
plt.ylabel("Shear Force (N)")
plt.grid(True)
plt.plot(xaxis,SF,'r--')
plt.show()
BM = BM[::-1]
plt.figure(figsize=(15,6))
plt.grid(True)
plt.xlabel("Span (m)")
plt.ylabel("Bending Moment (N)")
plt.plot(xaxis,BM,'g--')
plt.show()
max(BM)

# %%
plt.figure(figsize=(15,6))
plt.xlabel("Span (m)")
plt.ylabel("Lift distribution (N/m)")
plt.grid(True)
plt.legend()
plt.plot(lx,Liftdist,'c--', label = 'Lift Distribution')
# %%
```

A.2 Shear Flow and Buckling load in airfoil surface

```
import numpy as np
import matplotlib.pyplot as plt

b1 = 0.114967 # m
b2 = 0.23069 # m
a = 0.032 # m
K1 = 5.34 + 4*((b1/a)**2)
K2 = 5.34 + 4*((b2/a)**2)
nu = 0.33
E = 69e09 #Pa
t = 5e-04 # m
N_cr1 = (K1*((np.pi)**2)*E/(12*(1 - nu**2)))*((t/b1)**2)  # Pa
N_cr2 = (K2*((np.pi)**2)*E/(12*(1 - nu**2)))*((t/b2)**2)  # Pa
print('N_cr1',N_cr1)
print('N_cr2',N_cr2)
M_wing_max = 14.315 #Nm
I_skin = 24352.84e-12 # m^4

y1 = [-0.13, -3.37, -6.52, -9.39, -11.84, -13.68, -14.79, -15.12, 6.05,
    8.48, 10.38, 11.73, 12.53, 12.84, 12.72, 12.29]
q1 = [-0.63, -0.62, -0.59, -0.54, -0.46, -0.35, -0.22, 0.00, 0.60, 0.57,
    0.52, 0.46, 0.37, 0.28, 0.17, -0.01]
y2 = [-14.64, -13.55, -11.99, -10.08, -7.94, -5.70, -3.49, -1.44, 0.31,
    1.67, 2.52, 11.65, 10.80, 9.80, 8.74, 7.67, 6.66, 5.75, 4.95, 4.29, 3.80,
     3.49]
q2 = [0.13, 0.26, 0.38, 0.47, 0.54, 0.58, 0.61, 0.62, 0.61, 0.61, 0.61,
    -0.11, -0.22, -0.31, -0.39, -0.46, -0.51, -0.55, -0.57, -0.59, -0.61,
    -0.61]
x1 = [1.11, 5.07, 11.80, 21.16, 32.91, 46.76, 62.35, 79.28, 1.72, 6.19,
    13.26, 22.77, 34.46, 48.05, 63.23, 79.65]
x2 = [97.04, 115.14, 133.21, 150.80, 167.49, 182.87, 196.55, 208.22, 217.59,
     224.44, 228.62, 96.98, 114.86, 132.77, 150.27, 166.93, 182.32, 196.08,
    207.86, 217.35, 224.30, 228.55]
y1_up = [-0.13, -3.37, -6.52, -9.39, -11.84, -13.68, -14.79, -15.12]
y2_up = [-14.64, -13.55, -11.99, -10.08, -7.94, -5.70, -3.49, -1.44, 0.31,
    1.67, 2.52]
q1_up = [-0.63, -0.62, -0.59, -0.54, -0.46, -0.35, -0.22, 0.00]
q2_up = [0.13, 0.26, 0.38, 0.47, 0.54, 0.58, 0.61, 0.62, 0.61, 0.61, 0.61]
q_up = q1_up + q2_up
x_up = [1.11, 5.07, 11.80, 21.16, 32.91, 46.76, 62.35, 79.28, 97.04, 115.14,
     133.21, 150.80, 167.49, 182.87, 196.55, 208.22, 217.59, 224.44, 228.62]
x_down = [1.72, 6.19, 13.26, 22.77, 34.46, 48.05, 63.23, 79.65, 96.98,
    114.86, 132.77, 150.27, 166.93, 182.32, 196.08, 207.86, 217.35, 224.30,
    228.55]
q_down = [0.60, 0.57, 0.52, 0.46, 0.37, 0.28, 0.17, -0.01, -0.11, -0.22,
    -0.31, -0.39, -0.46, -0.51, -0.55, -0.57, -0.59, -0.61, -0.61]

F1 = []
F2 = []

for i in range(len(q1)):
    q1[i] = abs(q1[i])*1000
    F1.append(q1[i]/t + 0.001*M_wing_max*y1[i]/I_skin)
print('max_F1',max(F1))

for i in range(len(q2)):
    q2[i] = abs(q2[i])*1000
    F2.append(q2[i]/t + 0.001*M_wing_max*y2[i]/I_skin)
```

```python
print('max_F2',max(F2))


plt.plot(x_up, q_up)
plt.grid(True)
plt.title('Shear Flow on upper surface')
plt.xlabel('X (mm)')
plt.ylabel('Shear Flow (N/mm)')
plt.show()

plt.plot(x_down, q_down)
plt.grid(True)
plt.title('Shear Flow on lower surface')
plt.xlabel('X (mm)')
plt.ylabel('Shear Flow (N/mm)')
plt.show()
```

A.3 Components Placement

Component	Weight (kg)	Location (m) from nose
Payload	-0.6	0.1503
Nose Landing Gear	-0.15	0.1845
Lift	5.47	0.55975
Avionics	-0.85	0.7075
Wing	-0.952	0.666
Main Landing Gear	-0.3	0.7345
Fuselage	-2.3569	0.707
Tail	-0.261	1.63

Table A.1: Component Weights and Locations

A.4 SFD, BMD of Fuselage

```python
import matplotlib.pyplot as plt
import numpy as np
import pandas as pd
L = 1.69
SFD = []
BMD = []
Load = []
x = []
for i in range(10001):
    if(i/5000 > L):
        break
    x.append(i/5000)
    SFD.append(0)
    BMD.append(0)
    Load.append(0)
columns = ["Component", "Weight", "Location from Nose"]
df=pd.read_csv("location.csv")
df['Location'] = np.round(df["Location"],3)
SFD=[]
BMD=[]
for j in range(len(df['Location'])):
    index = x.index(df['Location'][j])
    Load[index] = df['Weight'][j] * 9.81
for i in range(len(x)):
```

```
    if(i==0):
        SFD.append(-Load[i])
        BMD.append(0)
    else:
        SFD.append(SFD[i-1] - Load[i] )
        product = [a * b for a,b in zip(Load[0:i][::-1],x[0:i])]
        BMD.append(sum(product))
plt.figure(figsize=(15,6))
plt.title("Shear Force Diagram for Fuselage")
plt.xlabel("X (m)")
plt.ylabel("Shear Force (N) ")
plt.grid(True)
plt.plot(x,SFD,"r",linewidth=4)
plt.tick_params(width=4,length=6,labelsize=14)
plt.show()

plt.figure(figsize=(15,6))
plt.title("Bending Moment Diagram for Fuselage")
plt.xlabel("X (m)")
plt.ylabel("Bending Moment (Nm) ")
plt.grid(True)
plt.plot(x,BMD,"g",linewidth=4)
plt.tick_params(width=4,length=6,labelsize=14)
plt.show()
```

A.5 Fuselage shear flow and bulkhead sizing

```
import numpy as np
import matplotlib.pyplot as plt
from scipy.optimize import fsolve

b = 0.1681 # m
h = 0.1366 # m
t = 5e-04 # m
L = 5.47 * 9.81 # N
I = 2*(b*(t**3)/12 + b*t*((h/2)**2) + (h**3)*t/12) # m^4
M_max_fus = 4 # Nm

s1 = np.linspace(0, b/2, 20)

q1 = []
for i in range(len(s1)):
    q1.append(L*h*s1[i]*t/(2*I))


s2 = np.linspace(0, h, 20)
q2 = []

for i in range(len(s2)):
    q2.append(q1[-1] + L*(h-s2[i])*s2[i]*t/(2*I))

q2 = np.array(q2)

#I = 1
A = 152.1e-06
E = 69e09
G = 26e09
F_qx = L*h*t*(s1**2)/(4*I)
F_qy = L*h*b*s2*t/(4*I) + L*t*(h*(s2**2)/2 - s2**3/3)/(2*I)
```

```
F_qx_b = L*h*b*t*s1/(4*I) - L*h*(s1**2)*t/(4*I)
M_0 = 14.315

def top_member(x):
    F_A, V_A, M_A = x
    V = V_A
    H = F_A - F_qx
    H = np.full_like(s1, H)
    M = M_A + V*s1
    M = np.full_like(s1, M)

    dU_MA = np.trapz(M/(E*I), s1)
    dU_FA = np.trapz(H/(E*A), s1)
    dU_VA = np.trapz(M*s1/(E*I), s1) + np.trapz(np.ones_like(s1), s1)*(V/(G*
    A))

    return [dU_FA, dU_VA, dU_MA]

def vertical_member(x):
    F_A, V_A, M_A = x
    V = F_A - F_qx[-1]
    H = L/2 - F_qy + V_A
    H = np.full_like(s2, H)
    M = M_A + M_0 - V*s2 + V_A*b/2
    M = np.full_like(s2, M)

    dU_MA = np.trapz(M/(E*I) , s2)
    dU_FA = np.trapz((-M*s2)/(E * I), s2) + V/(G*A)*np.trapz(np.ones_like(s2
    ),s2)
    dU_VA = np.trapz(-(M*b/2)/(E*I), s2) + np.trapz(H/(E*A), s2)
    return [dU_FA, dU_VA, dU_MA]

def bottom_member(x):
    F_A, V_A, M_A = x
    V = L/2 - F_qy[-1] + V_A
    H = L*h*t/(4*I) * (b**2/4 - s1**2) - F_A
    H = np.full_like(s1, H)
    M = M_A + M_0 + F_qx_b*h + H*h - V*s1 + V_A*b/2
    M = np.full_like(s1, M)
    dU_MA = np.trapz(M/(E*I) , s1)
    dU_FA = np.trapz(-h*M/(E*I), s1) + np.trapz(-H/(E*A), s1)
    dU_VA = np.trapz(M*s1/(E*A), s1) + np.trapz(np.ones_like(s1), s1)*(V/(G*
    A))
    return [dU_FA, dU_VA, dU_MA]

# Initial guess for F_A and M_A
initial_guess = [1e-05, 1e-05, 1e-05]  # Initial guess for [F_A, M_A]

# Solve the system of equations
sol_top = fsolve(top_member, initial_guess)
sol_vert = fsolve(vertical_member, initial_guess)
sol_bottom = fsolve(bottom_member, initial_guess)

# Sum the results to get the final solution
F_A_sol = sol_top[0] + sol_vert[0] + sol_bottom[0]
V_A_sol = sol_top[1] + sol_vert[1] + sol_bottom[1]
M_A_sol = sol_top[2] + sol_vert[2] + sol_bottom[2]

print("F_A = ", F_A_sol)
print("M_A = ", M_A_sol)
print("V_A = ", V_A_sol)
```

```python

def M_top(F_A,V_A, M_A):
    V = V_A
    H = F_A - F_qx
    H = np.full_like(s1, H)
    M = M_A + V*s1
    M = np.full_like(s1, M)

    return M

def M_vertical(F_A, V_A, M_A):
    V = F_A - F_qx[-1]
    H = L/2 - F_qy + V_A
    H = np.full_like(s2, H)
    M = M_A + M_0 - V*s2 + V_A*b/2
    M = np.full_like(s2, M)

    return M

def M_bottom(F_A, V_A, M_A):
    V = L/2 - F_qy[-1] + V_A
    H = L*h*t/(4*I) * (b**2/4 - s1**2) - F_A
    H = np.full_like(s1, H)
    M = M_A + M_0 + F_qx_b*h + H*h - V*s1 + V_A*b/2
    M = np.full_like(s1, M)

    return M

M1 = M_top(F_A_sol, V_A_sol, M_A_sol)
M2 = M_vertical(F_A_sol, V_A_sol, M_A_sol)
M3 = M_bottom(F_A_sol, V_A_sol, M_A_sol)
print(min(M1))

plt.plot(s1, M1, 'b', label = 'Top Member')
plt.plot(s2+s1[-1], M2, 'r', label = 'Vertical Member')
plt.plot(s2[-1]+s1[-1]+s1, M3, 'g', label = 'Bottom Member')
plt.title("Bending Moment Diagram of Bulkhead")
plt.xlabel("Distance along the half section (m)")
plt.ylabel('Bending Moment  (Nm)')
plt.grid(True)
plt.legend()
plt.show()

plt.plot(s1, q1)
plt.title('Variation of shear flow in the horizontal part from the center to
    the edge')
plt.grid(True)
plt.xlabel('Distance, $s_{1}$')
plt.ylabel('Shear Flow, $q_{1}$')
plt.show()

plt.plot(s2, q2)
plt.title('Variation of shear flow in the Vertical part')
plt.grid(True)
plt.xlabel('Distance, $s_{2}$')
plt.ylabel('Shear Flow, $q_{2}$')
plt.show()

F1_max = M_max_fus*h/(2*I) + q1[-1]/t
F2 = []
for i in range(len(s2)):
```

```
    F2.append(q2[i]/t + M_max_fus*(h-s2[i])*s2[i]*t/(2*I))
F2_max = max(F2)
print(F1_max)
print(F2_max)
print(I)
```

A.6 BMD of Landing Gear

```
% Given constants
W = 53.6607;
wt = 1.98 / 100;
theta = 10;
T = 201 / 1000;
L = (215.6/cos(deg2rad(10))) / 1000;

% Calculations
F = 0.85 * (W / 2);
M0 = F * (wt / 2);
theta_rad = deg2rad(theta);

% Section 1 (x in [0, L])
x1 = linspace(0, L, 100);
M1 = M0 + x1 * F * sin(theta_rad);

% Section 2 (x in [0, T/2 - Lsin(theta)])
x2_limit = (T / 2) - L * sin(theta_rad);
x2 = linspace(0, x2_limit, 100);
M2 = M0 + F * (x2 + L * sin(theta_rad));

% Section 3 (Reversed Section 2, x3 in [x2_limit, 0])
x3 = linspace(x2_limit, 0, 100);
M3 = (M0 + F * (x3 + L * sin(theta_rad)));

% Section 4 (Reversed Section 1, x4 in [L, 0])
x4 = linspace(L, 0, 100);
M4 = (M0 + x4 * F * sin(theta_rad));

% Plotting
figure;
hold on;

% Plot Section 1
plot(x1, M1, 'b', 'LineWidth', 2);

% Plot Section 2,
plot(x2 + L, M2, 'r', 'LineWidth', 2);

% Plot Section 3,
plot(x2 + L + x2_limit, M3, 'r', 'LineWidth', 2);

% Plot Section 4,
plot(x1 + L + 2*x2_limit, M4, 'b', 'LineWidth', 2);

% Labeling the plot
xlabel('x (m)');
ylabel('Bending Moment M(x) (N.m)');
title('Bending Moment Diagram');
grid on;
hold off;
```