AS5100 Mini Project

Team SARAS - Group 3

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UAV CAD Model

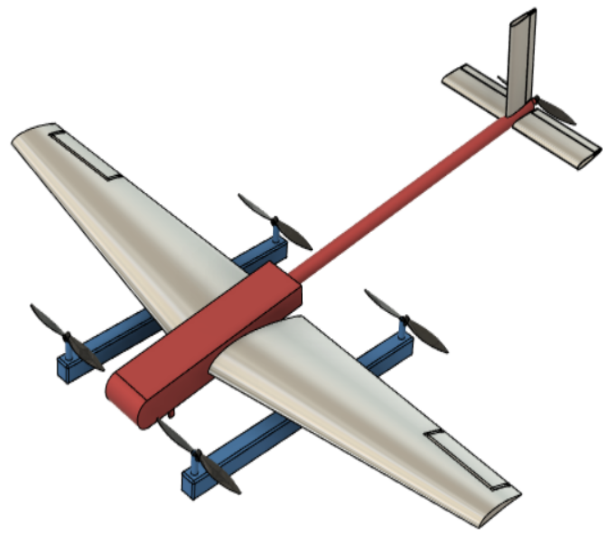

Figure 1: Isometric View of the UAV

S. No.	Primary Specification	Value
1	Cruise Velocity	20 m/s
2	Stall Speed	15 m/s
3	Endurance	97 min
4	Absolute Ceiling	4200 m above MSL
5	Payload Weight	1.309 kg
6	Max height from ground	45.81 m

(a)

S. No.	Derived Specification	Value
1	MTOW	10.47 kg
2	Battery weight	3.81 kg
3	Empty weight	5.28kg
4	Wing span	1.9475 m
5	Fuselage length	1700.5 mm
6	Horizontal tail span	0.5734 m
7	Vertical Tail span	0.2376 m

(b)

1 Mission Profile

Segment No.	**Mission Segment**	**Propulsion System**	**Height (from ground)**	**Speed (m/s)**
1	Take-off	VTOL	0 to 40 m	5 m/s
2	Cruise Search	Forward thrust	40 m	20 m/s
3	Descend	VTOL	40 to 0 m	2.5 m/s
4	Take-off	VTOL	0 to 40 m	5 m/s
5	Cruise Back	Forward thrust	40 m	20 m/s
6	Descend at Base	VTOL	40 to 0 m	2.5 m/s

Table 2: Mission Segment specifications

2 Wing Sizing

Tapered wing
Area: 0.5268 m^2
Span: 1.9475 m
Mean Aerodynamic Chord: 0.2835 m
Aspect Ratio: 7.2
Taper Ratio: 0.45
Dihedral: 6°
Sweep: 0 (Half-Chord)
Setting: 5.3705°
No High lift devices are used

2.1 Control Surface Sizing - Aileron

Area = 0.0275 m^2
Span = 0.4869 m
Chord = 0.0567 m
Location = 0.6861 m from wing root

2.2 Airfoil Parameters

Airfoil: GOE 802
Max thickness= 9.8% at x = 26.51% chord
Max camber= 6.2% at x = 37.73% chord

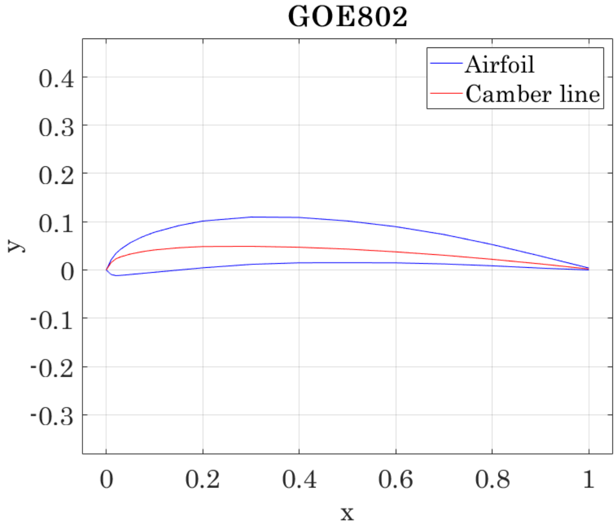

Figure 2: GOE802 airfoil

2.3 Wing Diagram

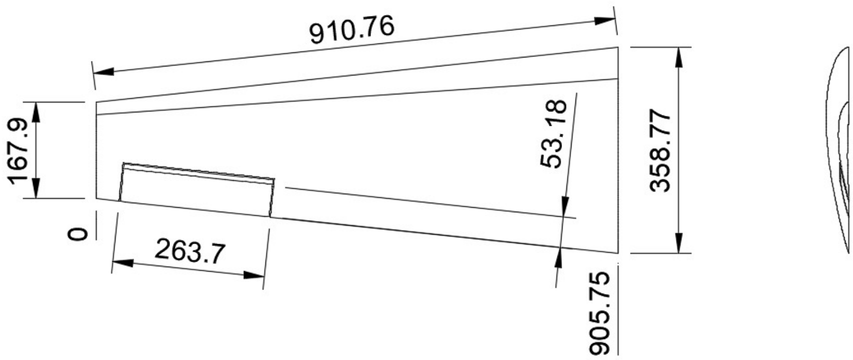

Figure 3: Wing 2D Diagram

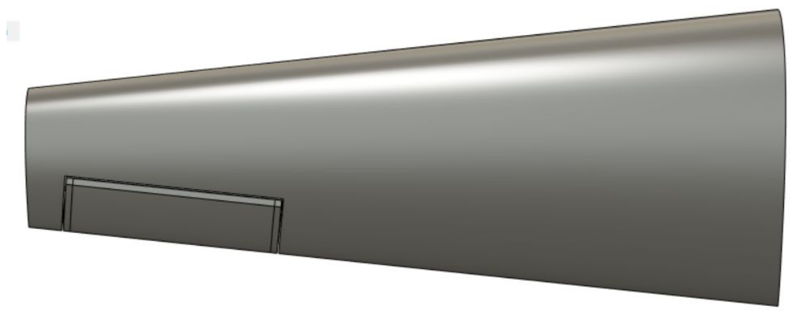

Figure 4: Top view of wing

3 V-n diagram

The V-n diagram is as shown in Figure 5.

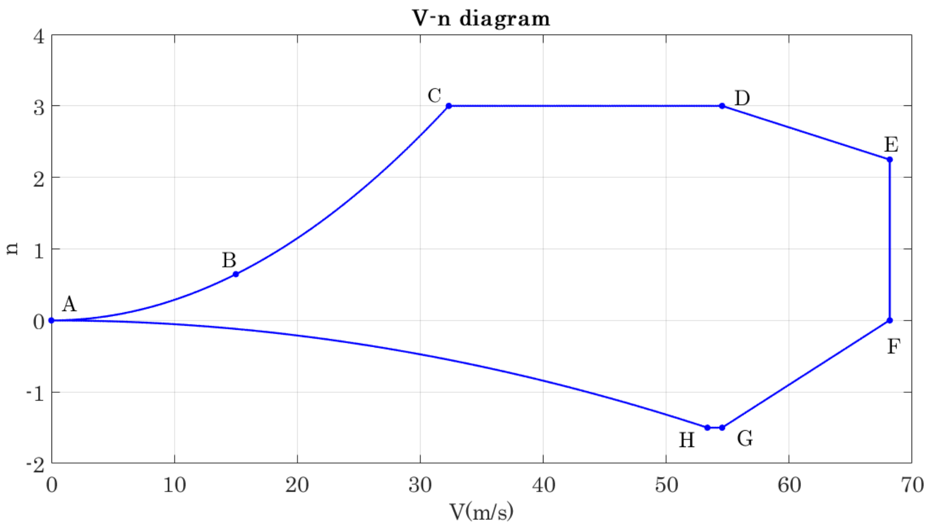

Figure 5: V-n diagram for the mission. The point B indicates the normal stall speed, C and H are the maneuvering speeds corresponding to maximum positive ($n_{max} = 3$) and negative load factors ($n_{max_{neg}} = -1.5$) respectively, D and G correspond to maximum structural cruise speed, and E and F correspond to never exceed speed.

4 Bill of Materials(Tentative)

A rough estimate of the bill of materials is shown below. This gets updated as design progresses.

4.1 Structural Elements

Component	**Rate**	**Time req. for procurement**	**Vendor**	**Qty**	**Price**
Aluminium 6082, 6mm(kg)	550	N/A	Arihant Aluminium Agencies, Parrys-600001	4	2200
Aluminium Sheet, 0.5mm(sq. m)	400	2 days to 1 week	The Kosmos Aluminium and Alloys, Parrys-600001	3	1200
				Total	3400

4.2 Propulsion

Component	**Rate**	**Time req. for procurement**	**Vendor**	**Qty**	**Price**
VTOL Motors	8400	8 days	Robokits	4	33600
Forward Motor	3900	8 days	Robokits	1	3900
VTOL Propeller	320	N/A	T-motor	4	1280
Forward Propeller	450	8 days	indianrobostore.com	1	450
VTOL motor ESC	10620	8 days	Robokits	4	42480
Forward motor ESC	3717	8 days	Robokits	1	3717
UAV Battery(10000 mAh)	13850	5 days	lipobattery.in	1	13850
				Total	99277

4.3 Avionics

Component	**Rate**	**Time req. for procurement**	**Vendor**	**Quantity**	**Price**
Pixhawk + GPS + Telemetry	27000	3 days	thinkrobotics.in	1	27000
Servos	975	8 days	Robokits	5	4875
Pitot tube	5760	8 days	Robokits	1	5760
				Total	37635

Grand Total: ₹140312

5 Testing of Aluminum samples(Tentative)

The aluminum sample used is Aluminum 6082. This sample is used for testing in a UTM. The standard used is ASTM D638 Type 1*.

Property	**Value**
Young's Modulus	84.66 GPa*
Tensile Strength	320 MPa*
Yield Stress	71 MPa*
Density	2,777.78 kg/m^3

Table 3: Calculated Properties of the sample

* We are going to perform a UTM test for another sample (as the failure surface is not at 45 deg to the horizontal) so these properties are subject to change. Plus as the extensometer slipped, some measurements were not taken properly hence the small depression in the stress-strain curve obtained from the UTM data.

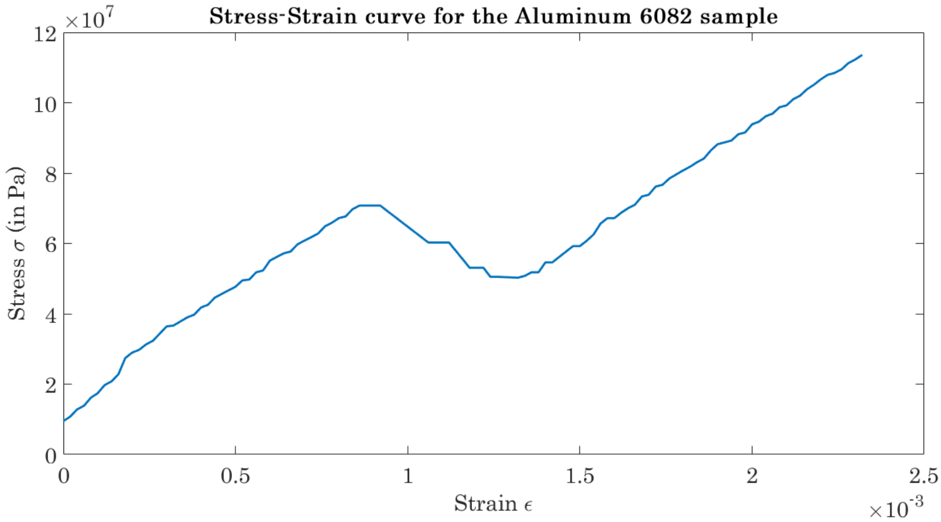

Figure 6: Stress-Strain Curve obtained from the UTM. As seen from the plot, there is a small depression at nearly $(\sigma, \epsilon) \approx (7.077 \times 10^7\, Pa, 9.2 \times 10^{-3})$ due to extensometer slippage.

6 Schrenk's Approximation

To find the lift distribution over the span for a finite non-elliptic wing we use Schrenk's approximation[1], [2]. As our aspect ratio is high enough($AR = 7.2$) and it's not a swept/twisted wing, we follow the below approach.
Please note that this method is an approximate result because the effect of fuselage is neglected.
We consider our actual wing to have area S, span b and chord $C(y)$ and having taper ratio λ.

1. The first step is to construct an "equivalent" elliptical wing planform having same area S and wingspan b has the load distribution similar to wing. The chord length is given by the relation(1)

$$c_{\text{ellipse}} = \frac{4S}{\pi b}\sqrt{1 - \left(\frac{2y}{b}\right)^2} \tag{1}$$

2. The actual wing chord variation(i.e., for a tapered wing) across the span is given by the relation(2)

$$c_{\text{actual}} = \frac{2S}{(1+\lambda)b}\left(1 + \frac{2y}{b}(\lambda - 1)\right) \tag{2}$$

3. The Schrenk's approximate chord length $\bar{c}_{\text{Schrenk}}$ is given by the average of the above 2 chords. Note that this excludes the presence of fuselage.

$$c_{\text{Schrenk}} = \frac{c_{\text{actual}} + c_{\text{ellipse}}}{2} \tag{3}$$

4. The total lift generated by the wing should be equal to the weight of the aircraft provided there are no lifting surfaces other than the wing(i.e., assuming the lift produced by the horizontal tail is almost small compared to that of a wing).

   The lift generated by an infinitesimal element of span dy is proportional to the area of that element.

$$dL = \kappa \, c_{\text{Schrenk}} \, dy$$

   Thus,

$$\frac{W}{2} = \int_0^{\frac{b}{2}} \kappa \, c_{\text{Schrenk}} \, dy \tag{4}$$

   From the above equation, the proportionality constant κ can be found and hence the load(i.e, the lift) on the wing per unit length can be found.

Following the procedure outlined above, κ is found out to be 194.97 Pa.
A plot is drawn in MATLAB showing the chords of the actual wing, equivalent ellipse and Schrenk's chord as shown in figure(7).

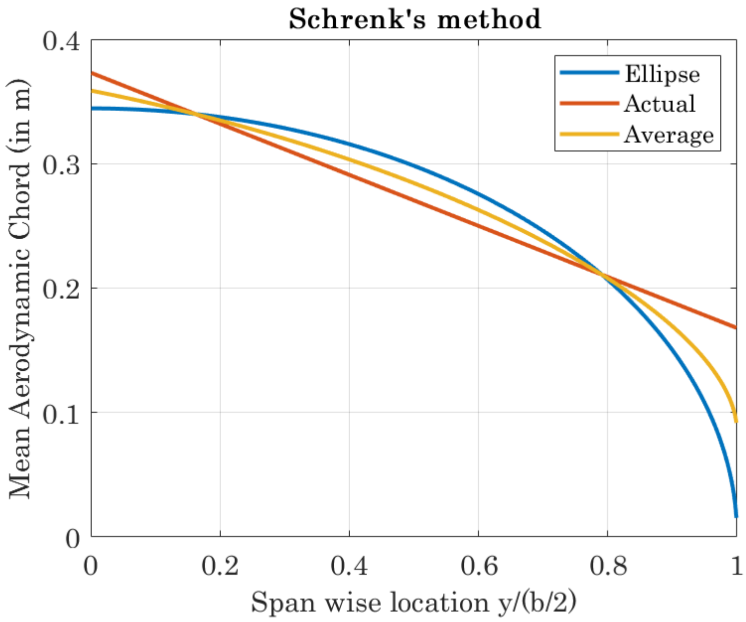

Figure 7: Schrenk's chord obtained from averaging actual wing chord and elliptic chord distribution. The lift per unit span distribution is κ times Schrenk's chord.

7 Shear force and Bending moment due to Lift

The shear force and bending moment due to lift forces at each wing sections are calculated. The weight of the wing and drag induced are ignored. The shear force $V(y)$ is obtained by summation of all the elemental lift forces from the tip all the way to the root of the wing.

$$V = \int_0^{\frac{b}{2}} \frac{dL}{dy'} dy'$$

where y' co-ordinate is measured from tip. The shear force diagram(SFD) is plotted as shown in figure(8).

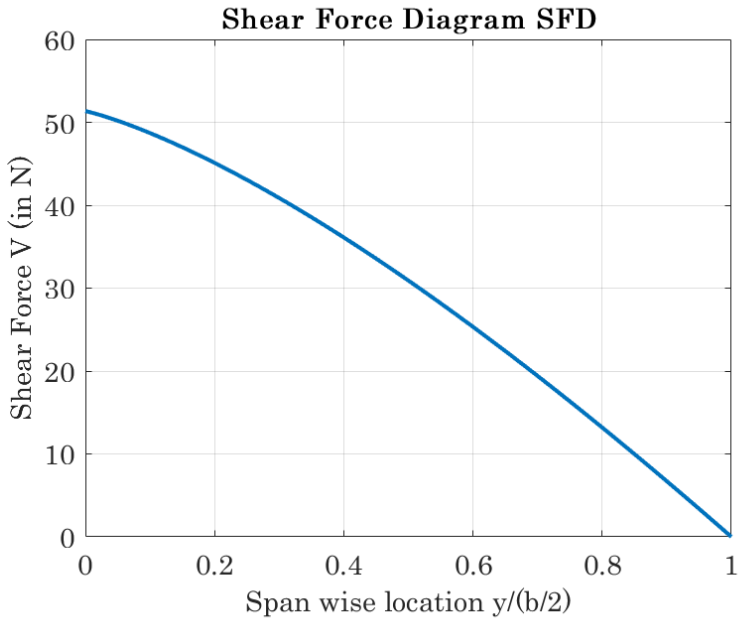

Figure 8: Shear Force Diagram

Bending moment $M(y)$ can be calculated by calculating the area under the shear force diagram or sum of moment of forces from the elemental section considered from tip $y' = 0$ to y'-station. The lift force distribution cause sagging moment on the wing. It is because the upper part of the wing will be under compression and lower part in tension.

$$M = \int_0^{\frac{b}{2}} \frac{dL}{dy'} y' dy'$$

The bending moment diagram(BMD) is plotted as shown in figure(9).

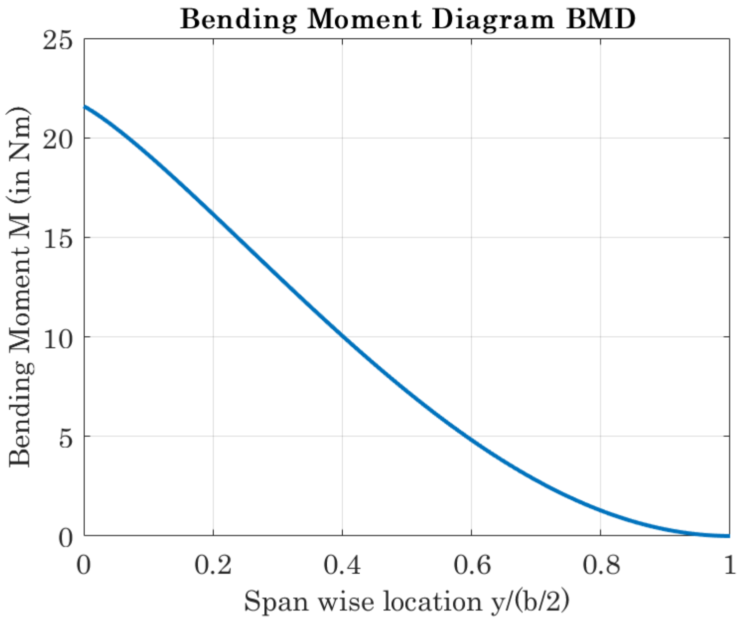

Figure 9: Bending Moment Diagram

8 Moment of Inertia Distribution

8.1 I-section spar

From unidirectional bending theory,

$$\frac{\sigma_y}{y} = \frac{M}{I_{xx}} \tag{5}$$

where
M = bending Moment at that section
y = Semi-height of the web
σ_y = Bending stress
I_{xx} = Area moment of Inertia about centroid.
Considering the Factor of safety, stress concentration, fatigue factor and load factor, the maximum allowable stress is roughly 1/10th of yield stress. i.e, $\sigma_{max} = 26MPa$.
Neglecting web contribution to the moment of inertia and if both rectangular flanges are having same area, $I = \frac{Ah^2}{2}$ where A is the area of a flange and h is the length of the web. We wish to place the spars at the max. thickness location of the airfoil so that they can take the maximum bending moment.

The Max. thickness distribution is given by:

$$h(y) = \left(c_{\text{root}} - \frac{2y}{b}(c_{\text{root}} - c_{\text{tip}})\right)\frac{t_{max}}{c}$$

To find the moment of area distribution, we rearrange the terms of equation(5) to get:

$$I_{xx} = \frac{Mh}{2\sigma_{\text{max}}} \tag{6}$$

Considering rectangular flanges of equal thickness(and width), we have $A = w_{\text{flange}}t_{\text{Al}}$ where w_{flange} is the width of the flange used and t_{Al} is the thickness of the flange. From this and equation(6) as $I_{xx} = \frac{Ah^2}{2}$, we get the flange width distribution as

$$w_{\text{flange}} = \frac{M}{\sigma_{\text{max}}t_{\text{Al}}h} \tag{7}$$

The thickness of Aluminum sheet used to make flange is 0.3mm. Using equation(7) we find flange width distribution obtained theoretically. But because of manufacturing difficulties, a linear variation of flange width is considered. The linear envelope is above the theoretically obtained flange width distribution (i.e.,linear flange width at a given span wise location should be greater than the theoretically obtained(or the minimum) flange width at a given location). For this, 3 mm offset is given for flange width at root and flange width at tip is 1/10th of root chord(from linear envelope). This is shown in figure(10). The moment of area distribution is plotted in figure(11).

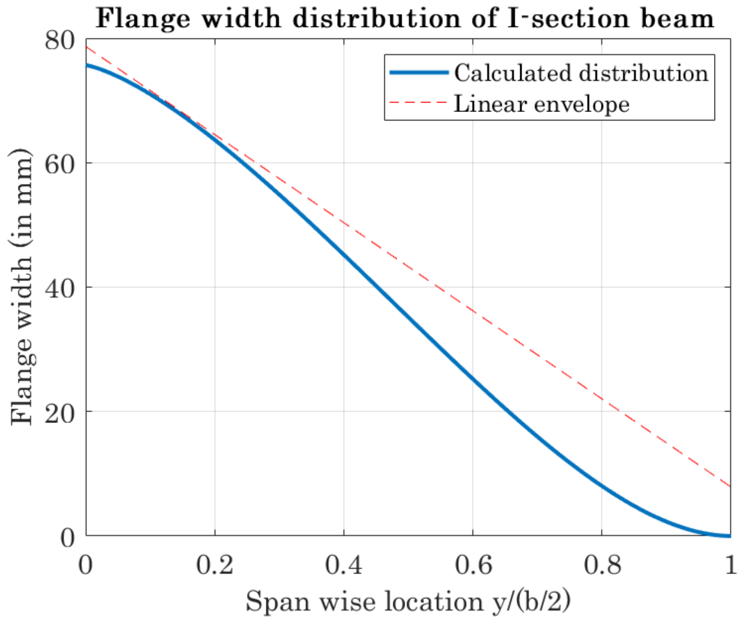

Figure 10: Spanwise Flange width distribution of I section spars

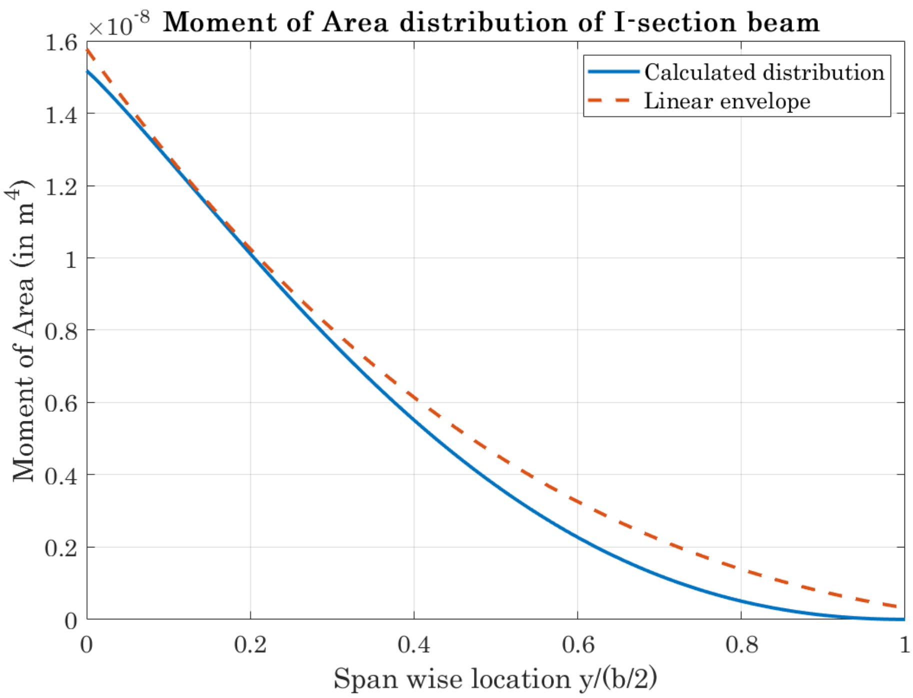

Figure 11: Spanwise Moment of Inertia distribution of I-section spars. Linear Width envelope is also considered in the plot

8.2 Skin

To find the moment of inertia of the skin, the skin is divided into 34 panels based on the coordinates available (per unit chord).
The following steps are followed for calculating I_{xx} of the skin:

1. The area of the panels are assumed to be "concentrated" at the midpoints of the panels. Using this, the centroid is calculated using the equations.

$$\overline{x} = \frac{\sum_i A_i x_i}{\sum_i A_i} \tag{8}$$

$$\overline{z} = \frac{\sum_i A_i z_i}{\sum_i A_i} \tag{9}$$

   where A_i is the area of the i^{th} panel and (x_i, z_i) are the coordinates of the center of i^{th} panel.

2. The moment of area is calculated using the equation

$$I_{xx} = \sum_i A_i (z - \overline{z})^2 = t_{Al} \sum_i l_i (z - \overline{z})^2 \tag{10}$$

   where t_{Al} is thickness of Aluminum sheet used for the skin and l_i is the length of the panel.

The aluminum sheet used for skin is 0.3mm thick. The moment of area distribution is calculated spanwise and is plotted. The plot is shown in figure(12)

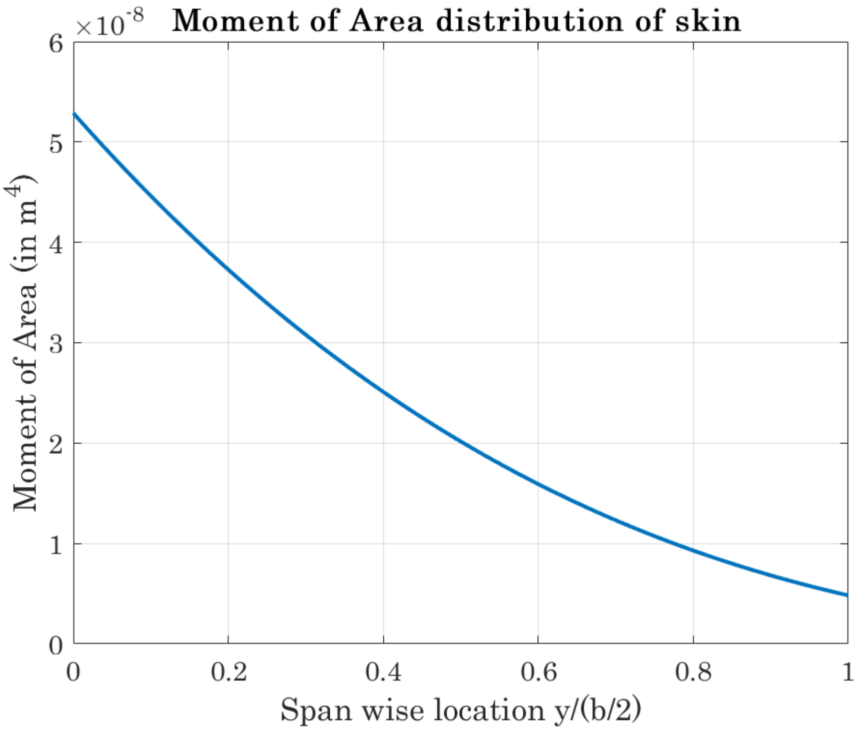

Figure 12: Spanwise Moment of Inertia distribution of wing skin

The moment of inertia of skin and spars are shown in same plot for comparision as shown in figure(13). As shown in the plot, the moment of area distribution of the skin is significantly larger than that of the spar(linear envelope). This implies that the bending stress is smaller for skin compared to spar for same moment. Thus, skin itself can take bending loads. But spars should be present to provide mount points and support torsion loads.

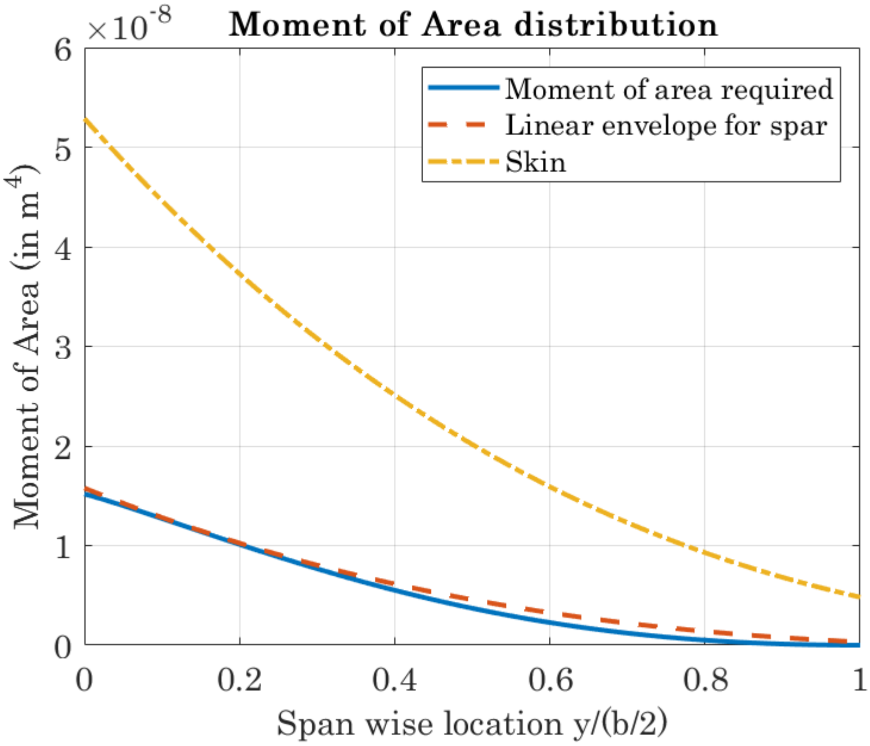

Figure 13: Comparision of Spanwise Moment of Inertia distribution of spars and skin

References

[1] Oskar Schrenk. “A simple approximation method for obtaining the spanwise lift distribution”. In: *The Aeronautical Journal* 45.370 (1941), pp. 331–336.

[2] Nur Nabillah Mohd Kamal et al. “Comparison Study Between Schrenk’s Approximation Method and Computational Fluid Dynamics of Aerodynamic Loading on UAV NACA 4415 Wing”. In: *Journal of Advanced Research in Fluid Mechanics and Thermal Sciences* 64.2 (2019), pp. 283–292.