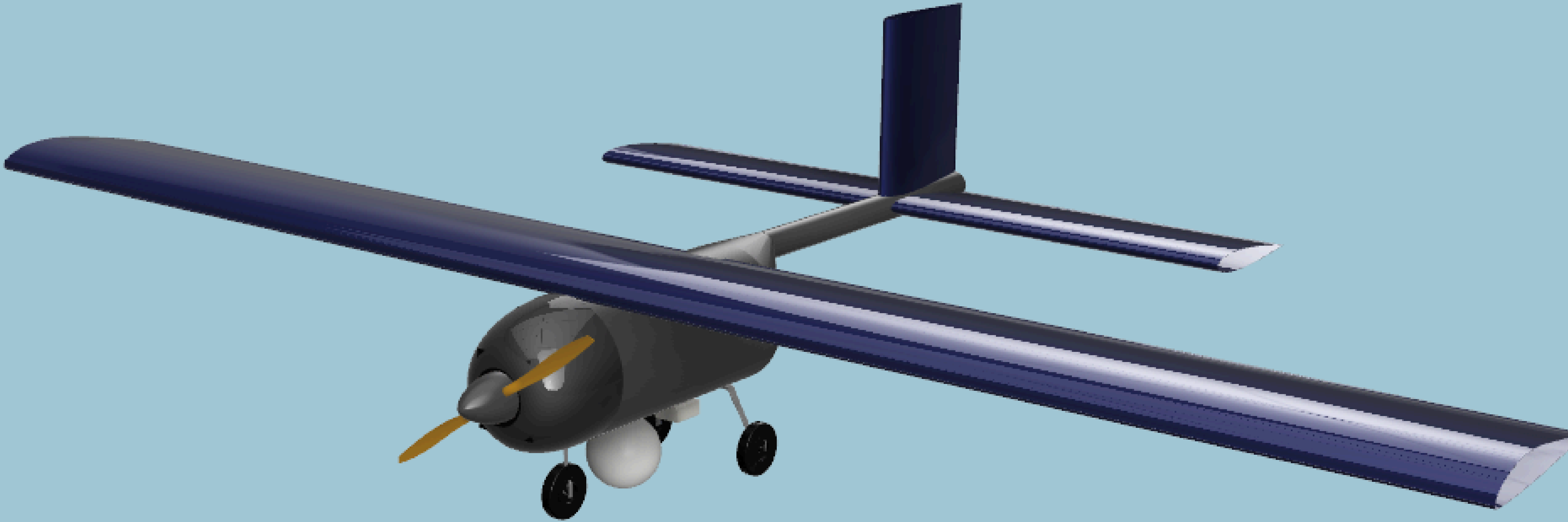
DESIGN OF UAV

Group 5

- AE21B002 : Abhigyan Roy
- AE23M004 : Vinu Mathew
- AE23M008 : Anish Konar
- AE23M014 : Gautham Anil
- AE23M006 : Aditya Sai Deepak Rachagiri

Design Overview



Parameters	Value
MTOW	8.80 Kg
Max Payload Weight	2.00 Kg
Design Payload Weight	1.50 Kg
Powerplant Weight	2.30 Kg
CG location (from nose)	0.50 m
Wing Area	0.96 m^2
Wing Span	2.80 m
Wing Taper Ratio	1
Wing Root Chord	0.34 m
Wing Tip Chord	0.34 m
Wing Aspect Ratio	8.34
Wing Twist Angle	0 Deg
Wing Sweep Angle	0 Deg
Wing Dihedral Angle	2 Deg
Wing Setting Angle	1 Deg
Wing Aerofoil	GOE 553
Aileron Area	0.05 m^2
Aileron Chord	0.09 m
Aileron Span	0.56 m
Fuselage Length	1.20 m
Fuselage Diameter	0.25 m
Fuselage Width	0.20m

Horizontal Tail Area	0.42 m^2
Horizontal Tail Span	1.5 m
Horizontal Tail Taper Ratio	1
Horizontal Tail Root Chord	0.28 m
Horizontal Tail Tip Chord	0.28 m
Horizontal Tail Aspect Ratio	5.0
Horizontal Tail Twist Angle	0 Deg
Horizontal Tail Sweep Angle	0 Deg
Horizontal Tail Dihedral Angle	0 Deg
Horizontal Tail Setting Angle	2 Deg
Horizontal Tail Aerofoil	NACA 0014
Elevator Area	0.105 m^2
Elevator Chord	0.07 m
Elevator Span	1.5 m

Vertical Tail Area	0.14 m^2
Vertical Tail Span	0.4 m
Vertical Tail Taper Ratio	1
Vertical Tail Root Chord	0.35 m
Vertical Tail Tip Chord	0.35 m
Vertical Tail Aspect Ratio	1.14
Vertical Tail Twist Angle	0 Deg
Vertical Tail Sweep Angle	0 Deg
Vertical Tail Dihedral Angle	0 Deg
Vertical Tail Setting Angle	0 Deg
Aerofoil	NACA 0014
Rudder Area	0.042 m^2
Rudder Chord	0.105 m
Rudder Span	0.4 m

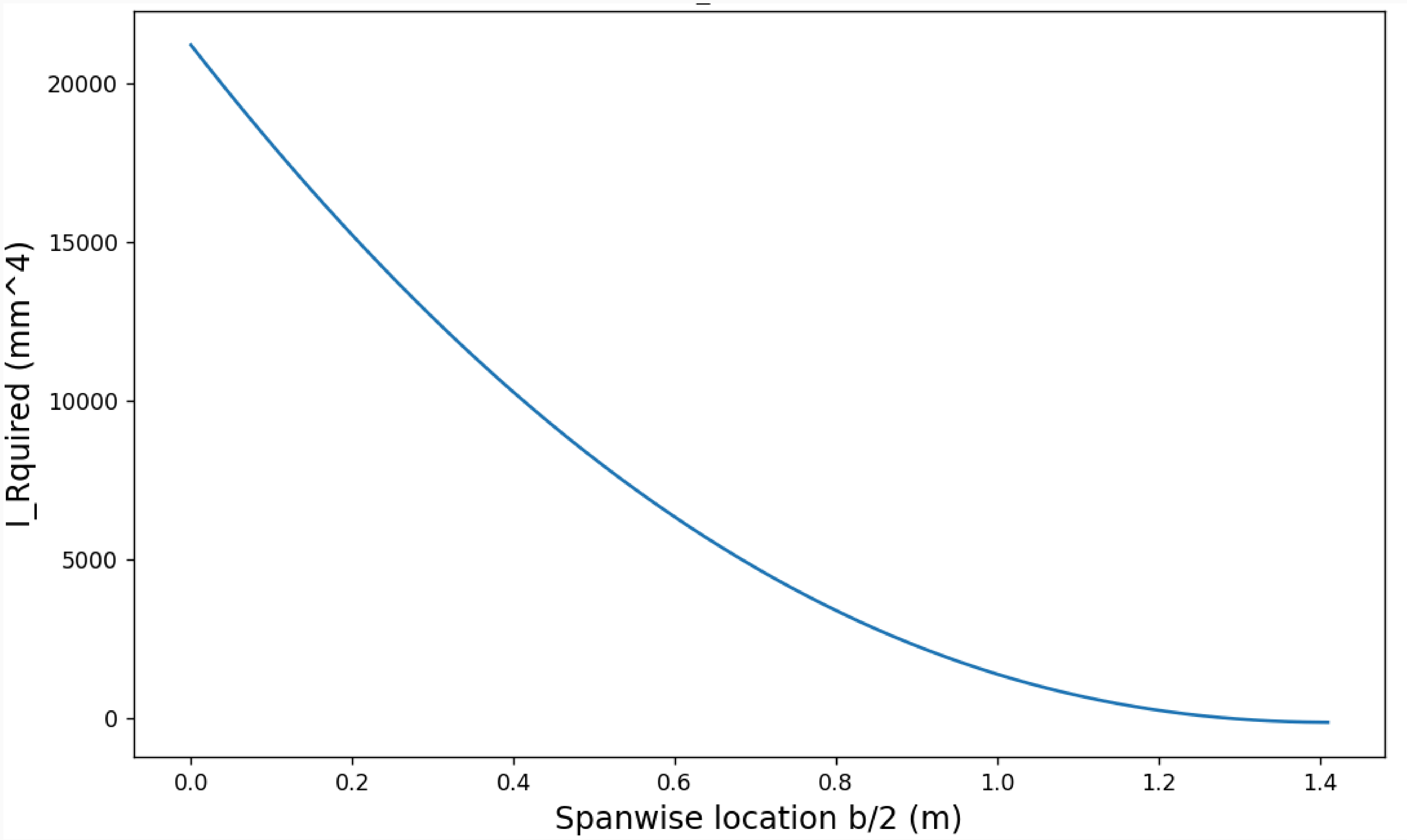
Spar Design

- Stress Concentration Factor (k = 3)
- Fatigue Factor (f = 1.5)
- Factor of Safety (m = 1.15)
- Maximum Load Factor (n = 3)
- σ_yield = 290 MPa

$$\sigma_{\text{allowable}} = \frac{\sigma_Y}{n \times f \times k \times m}$$

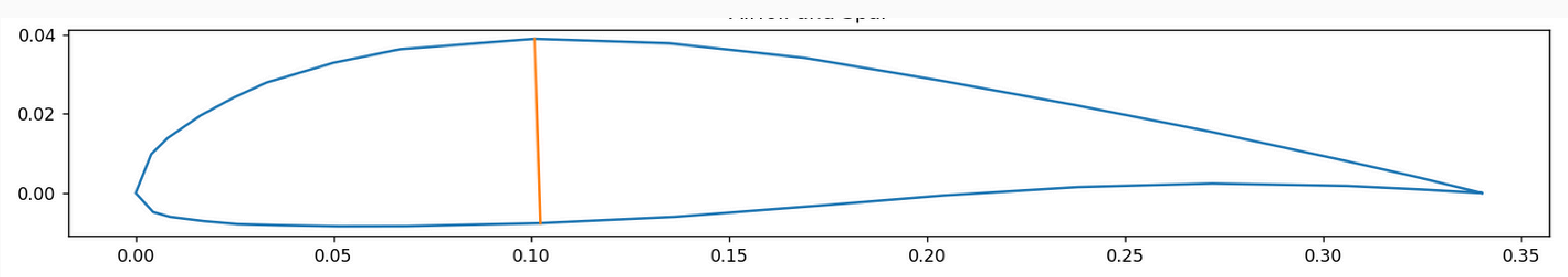
The centroid of the skin panel was calculated, and the moment of inertia calculation was performed with the centroid as the origin.

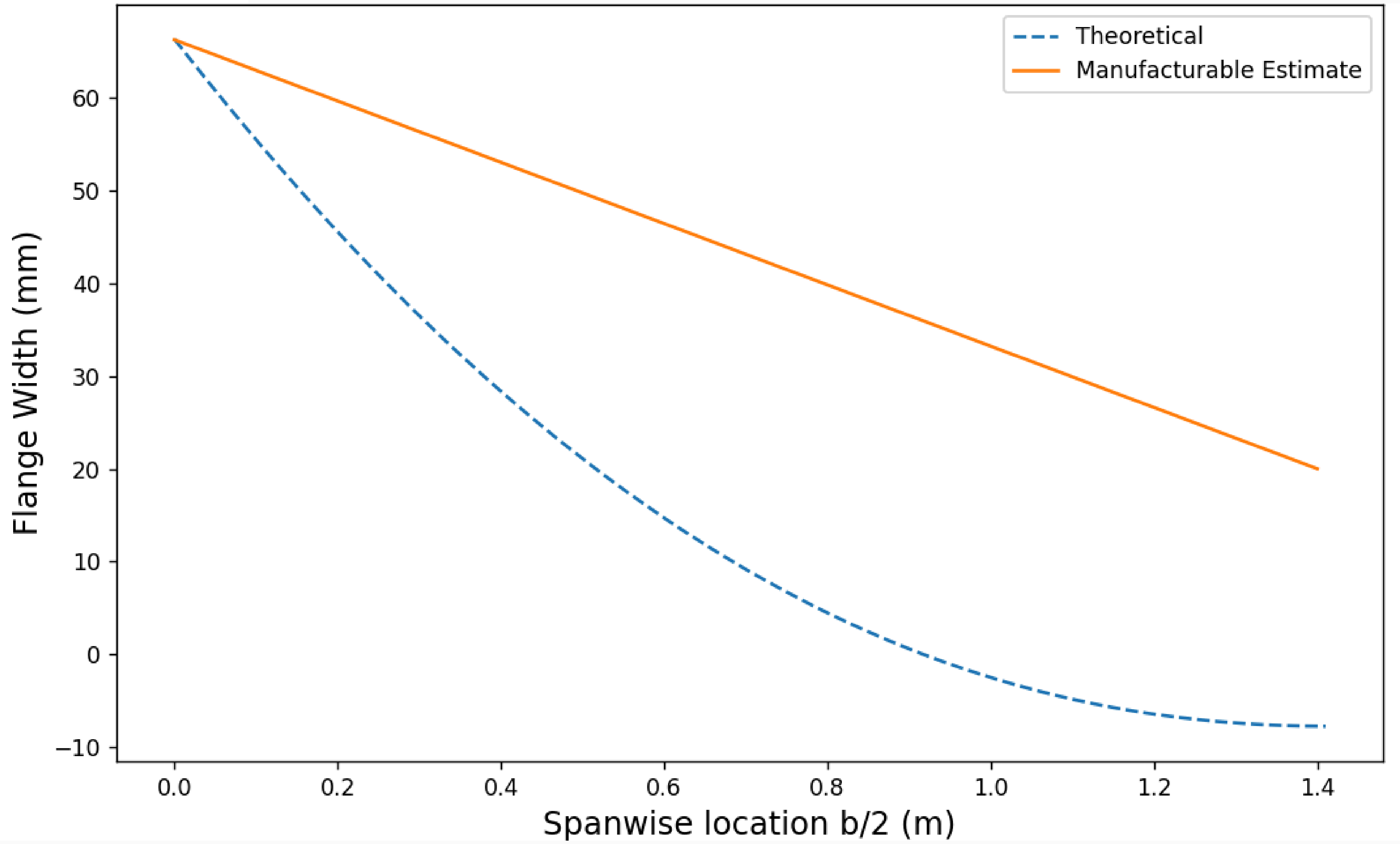
$$I_{skin} = \int_A y^2 dA = 56300 mm^4$$



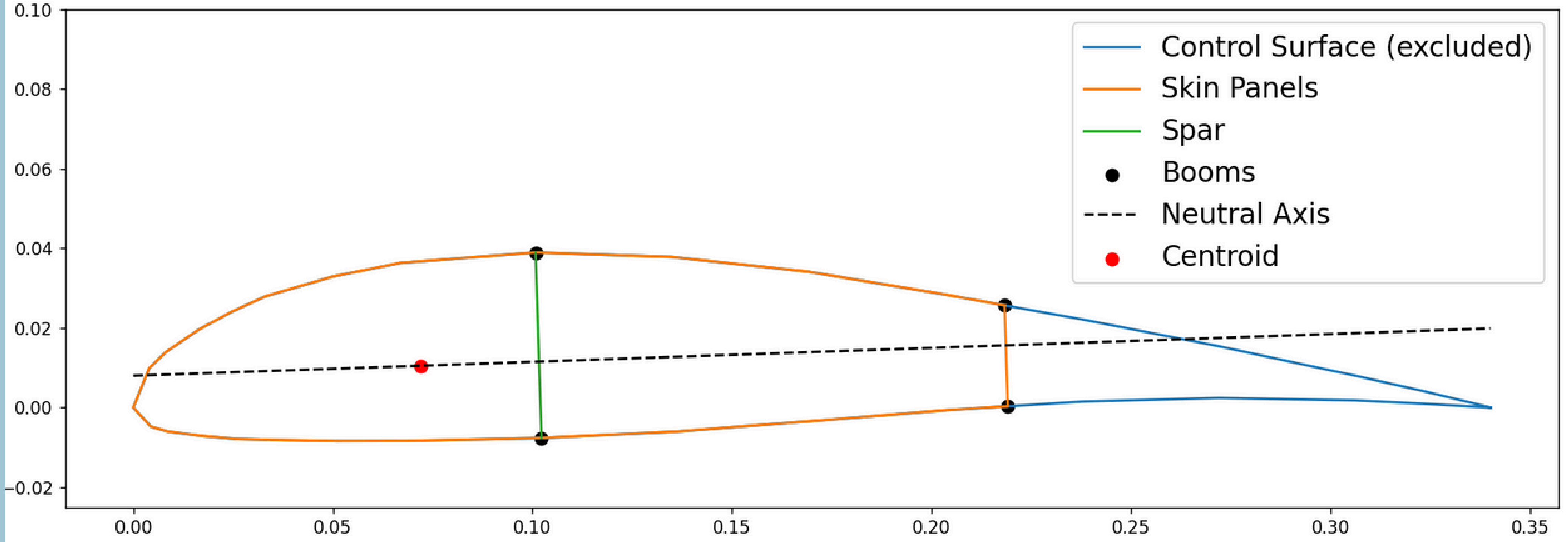
$$I_{spar} = \frac{th^3}{12} + w \times t \times \left(\frac{h}{2}\right) \times 2 = \frac{M_{\text{max}} \cdot y}{\sigma_{\text{allowable}}} - I_{skin}$$

Maximum bending moment = 27 Nm


0.04
0.02
0.00
0.00 0.05 0.10 0.15 0.20 0.25 0.30 0.35


Theoretical
Manufacturable Estimate
Flange Width (mm)
60
50
40
30
20
10
0
−10
0.0 0.2 0.4 0.6 0.8 1.0 1.2 1.4
Spanwise location b/2 (m)

Wing Shear Design



Idealised Boom Structure of Airfoil

$$F_{cr} = K_{ss}\frac{\pi^2 E}{12(1-\nu^2)}\left(\frac{t}{b}\right)^2\left[R_a + \left(\frac{R_a - R_b}{2}\right)\left(\frac{b}{a}\right)^3\right]$$

Critical Load per unit Area

$$q_{total} = \frac{-S_y}{I_{xx}}\sum_{r=1}^{n} B_r y_r + q_{s,0}$$

```
q0_1, q0_2 = symbols('q0_1 q0_2')
lhs_dtdz = 1/(2*A1*G*thickness)*(total_qb_s_1 + total_s_1*q0_1 - spar_length*q0_2)
rhs_dtdz = 1/(2*A2*G*thickness)*(total_qb_s_2 + total_s_2*q0_2 - spar_length*q0_1)
dtdz_eq = Eq(lhs_dtdz, rhs_dtdz)

moment_contrib = q0_1 * A1 + q0_2 * A2 - np.sum(qb_skin * p0_panel_averages * s)
moment_eq = Eq(moment_contrib, 0)

solution = solve((moment_eq, dtdz_eq), (q0_1, q0_2))
print("Solution:", solution)
```

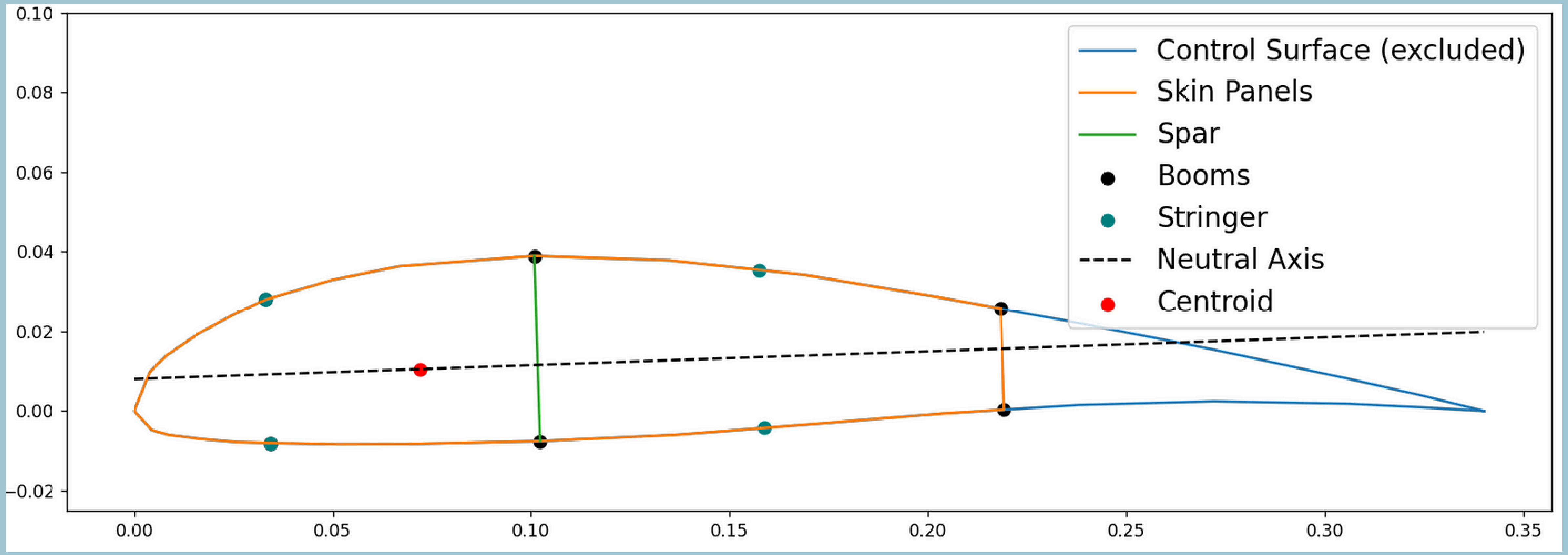
$$\sum_{R=1}^{N} M_{q,R} = \sum_{R=1}^{N}\oint_R q_b p_0 \,\mathrm{d}s + \sum_{R=1}^{N} 2A_R q_{s,0,R}$$

$$\frac{\mathrm{d}\theta}{\mathrm{d}z} = \frac{1}{2A_R}\oint_R q\frac{\mathrm{d}s}{Gt}$$

Initial q = 1200-1600 N/m

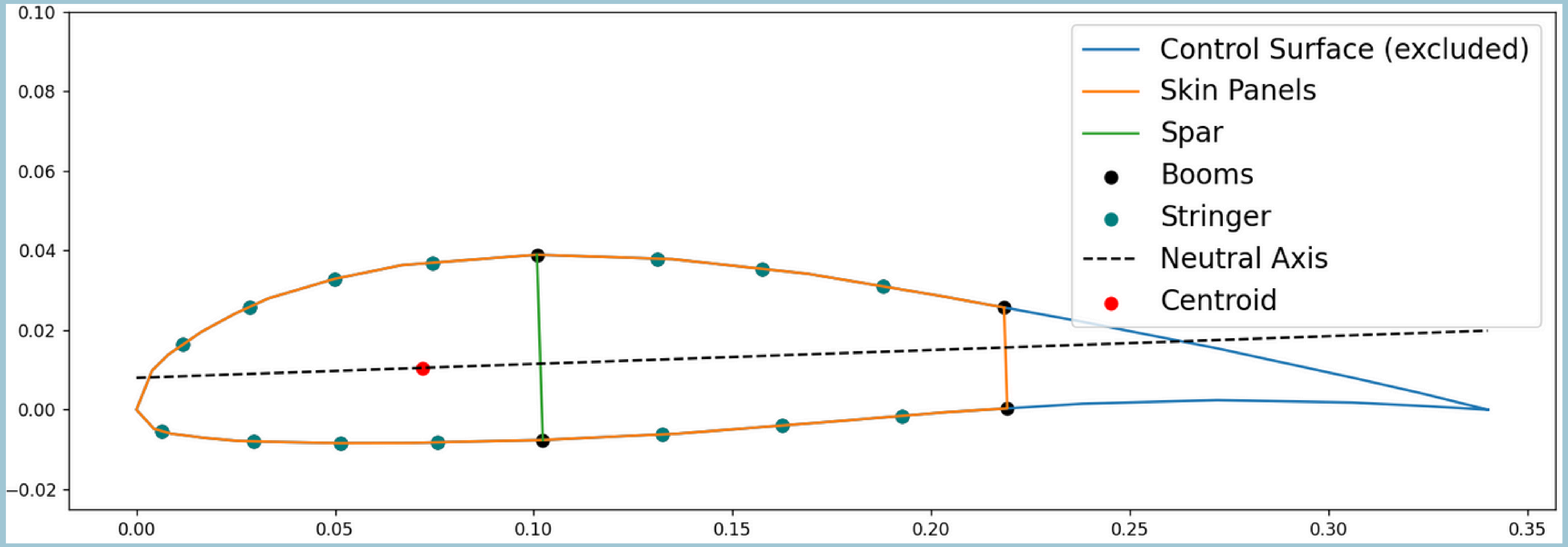
Fcr ≈ 1600 N/m

Stringer Design



q = 1200-1600 N/m
Fcr ≈ 2400 - 3600 N/m

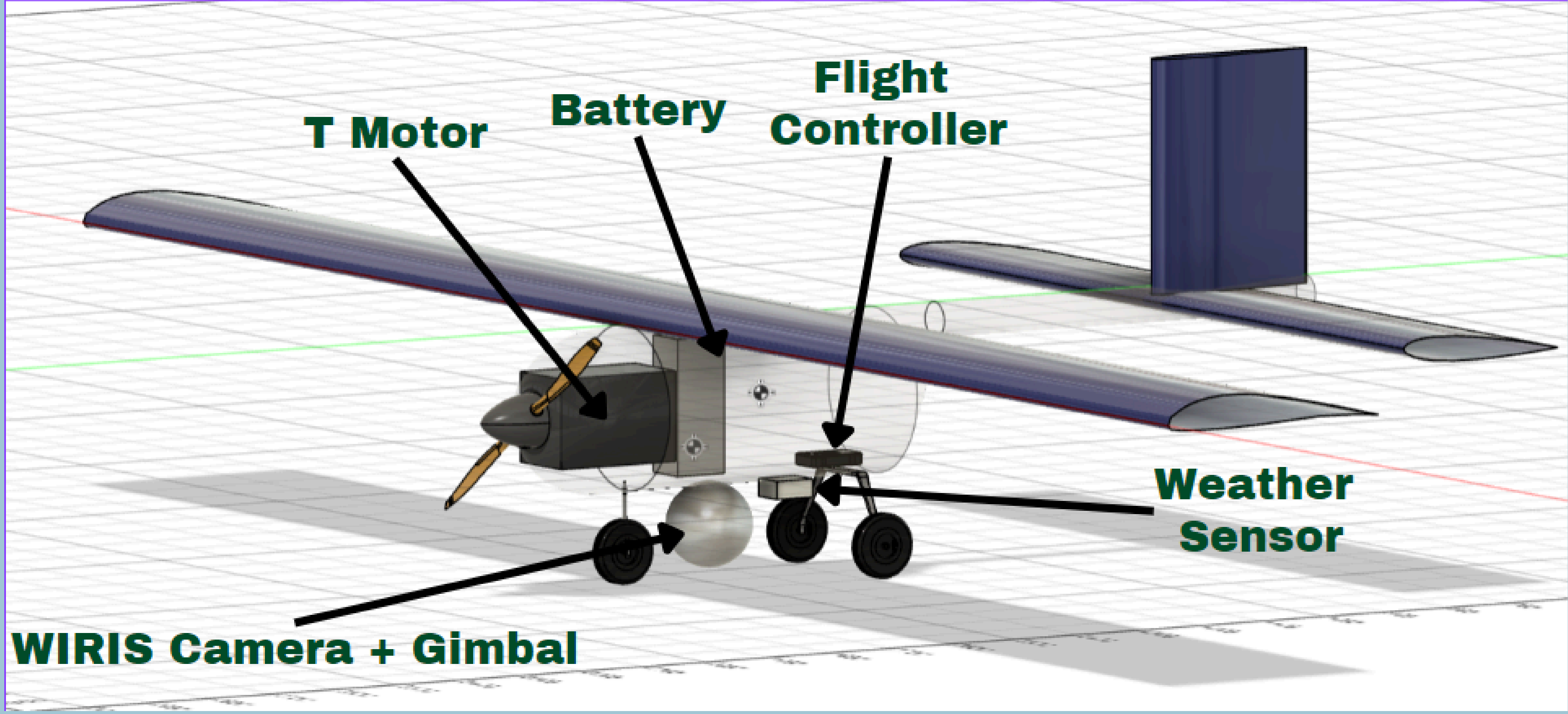
FOS ≈ 2



q = 1200-1600 N/m
Fcr ≈ 12000 - 20000 N/m

FOS ≈ 10

Fuselage Design

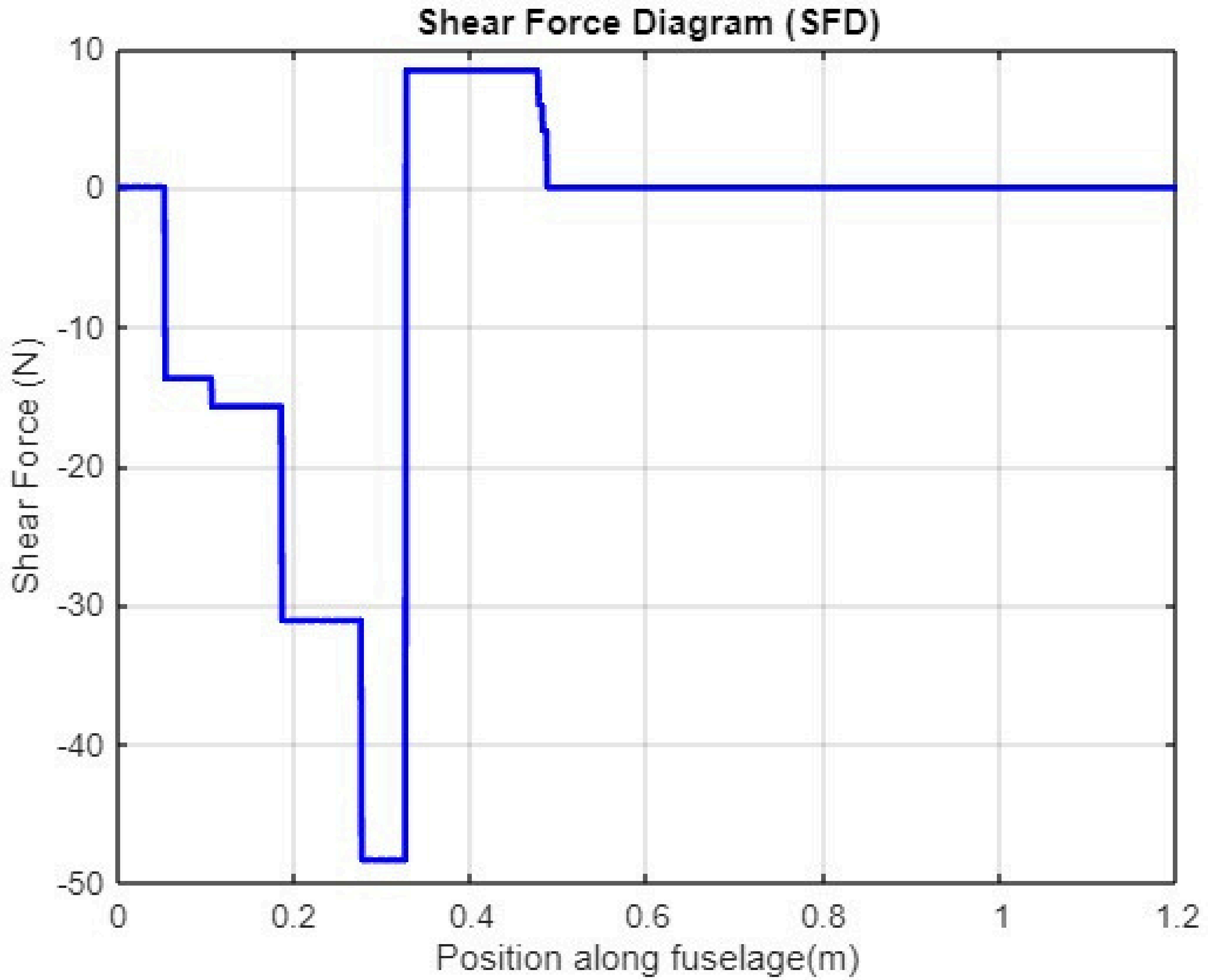


Fuselage Length = 1.2 m
Fuselage Diameter = 0.230 m

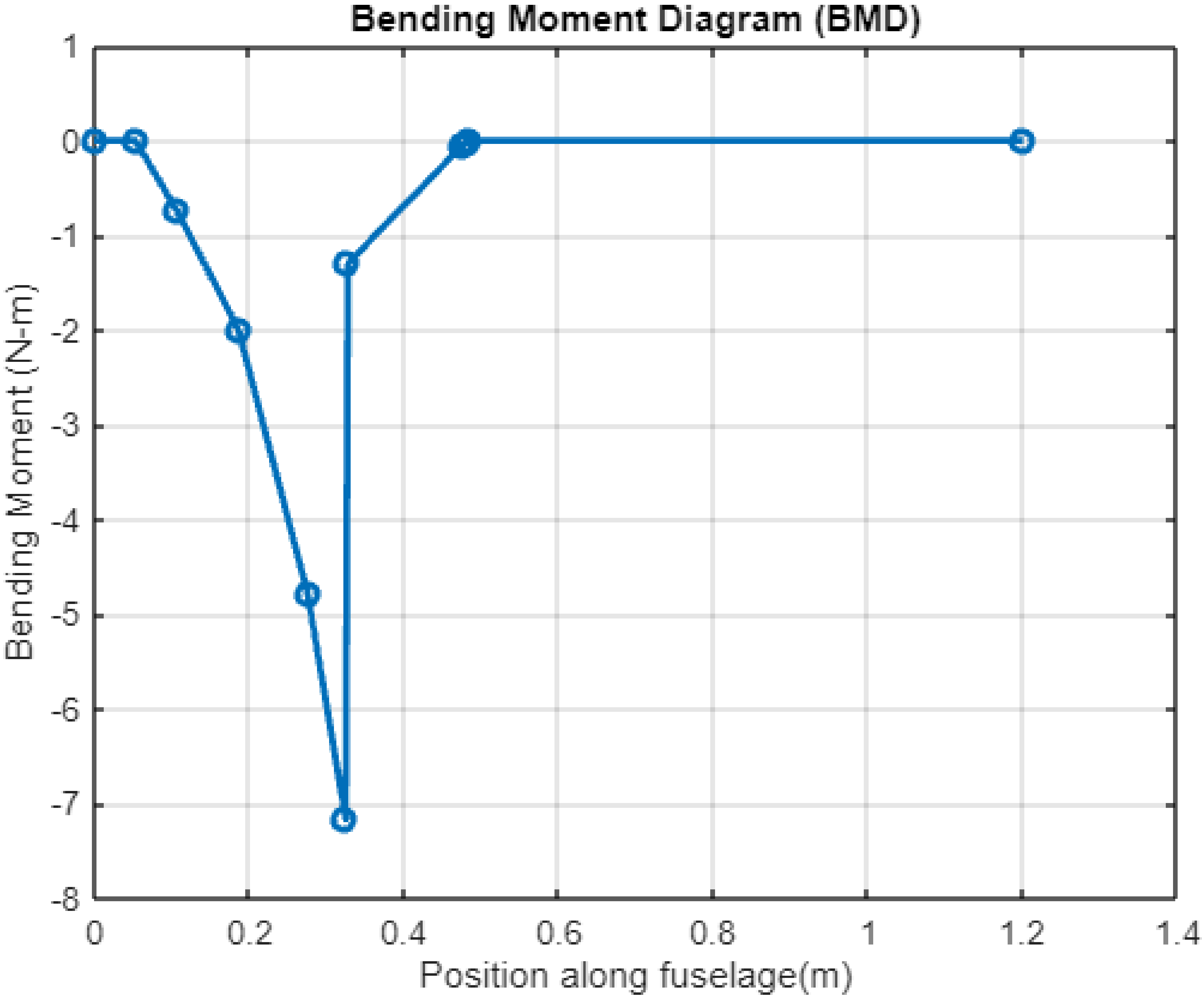
PLA
σ_yield = 50 MPa
E = 3.5 GPa
G = 4 GPa

Component	Weight (N)	Position from the Nose (m)
Motor	13.72	0.053
Battery	15.39	0.186
Wiris	17.25	0.276
Environmental Sensor	2.45	0.476
Avionics	1.96	0.481
Landing Gear@Nose	2.01	0.106
Landing Gear@Rear	4.02	0.486

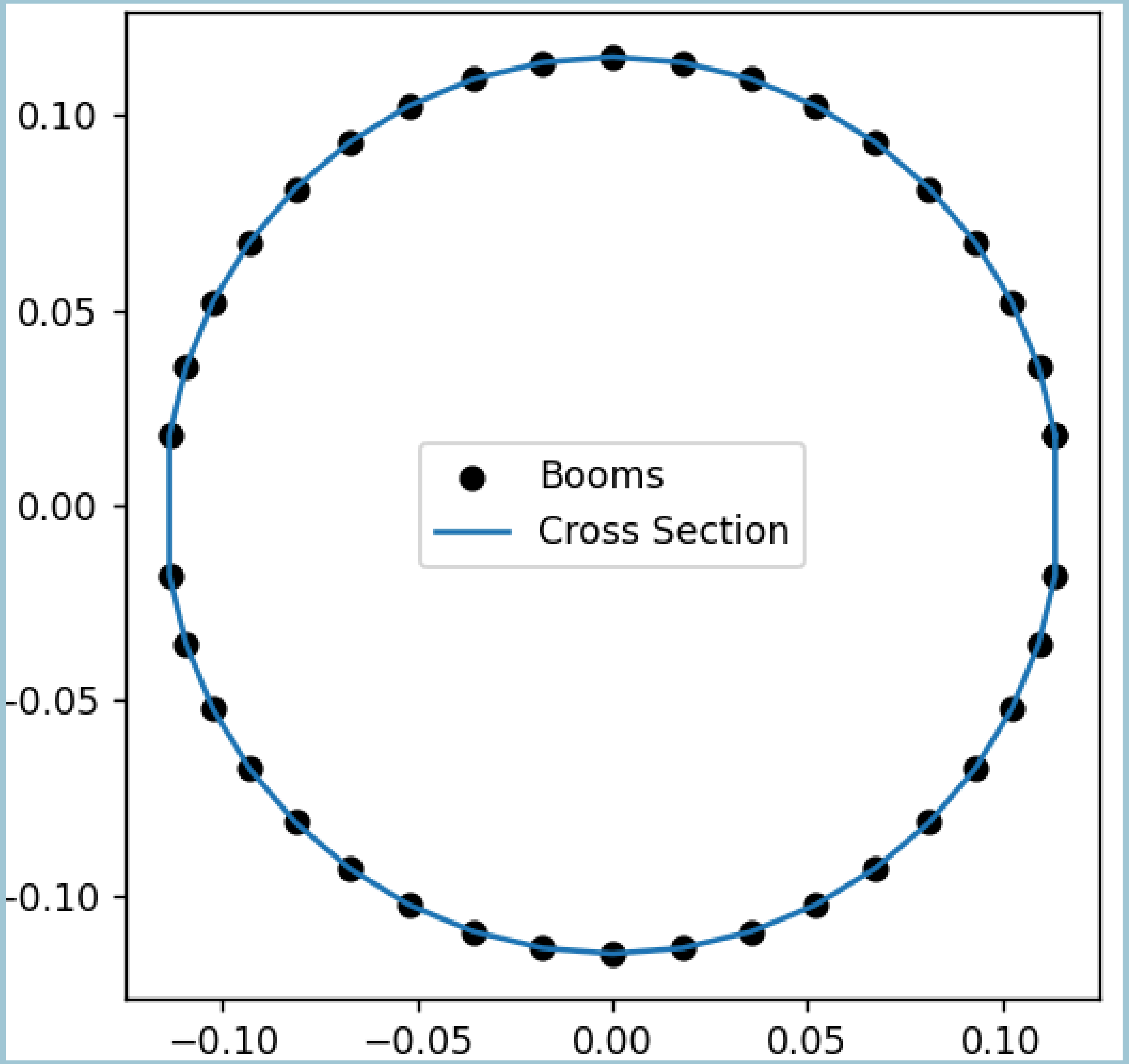
Fuselage - Shear Force Diagram



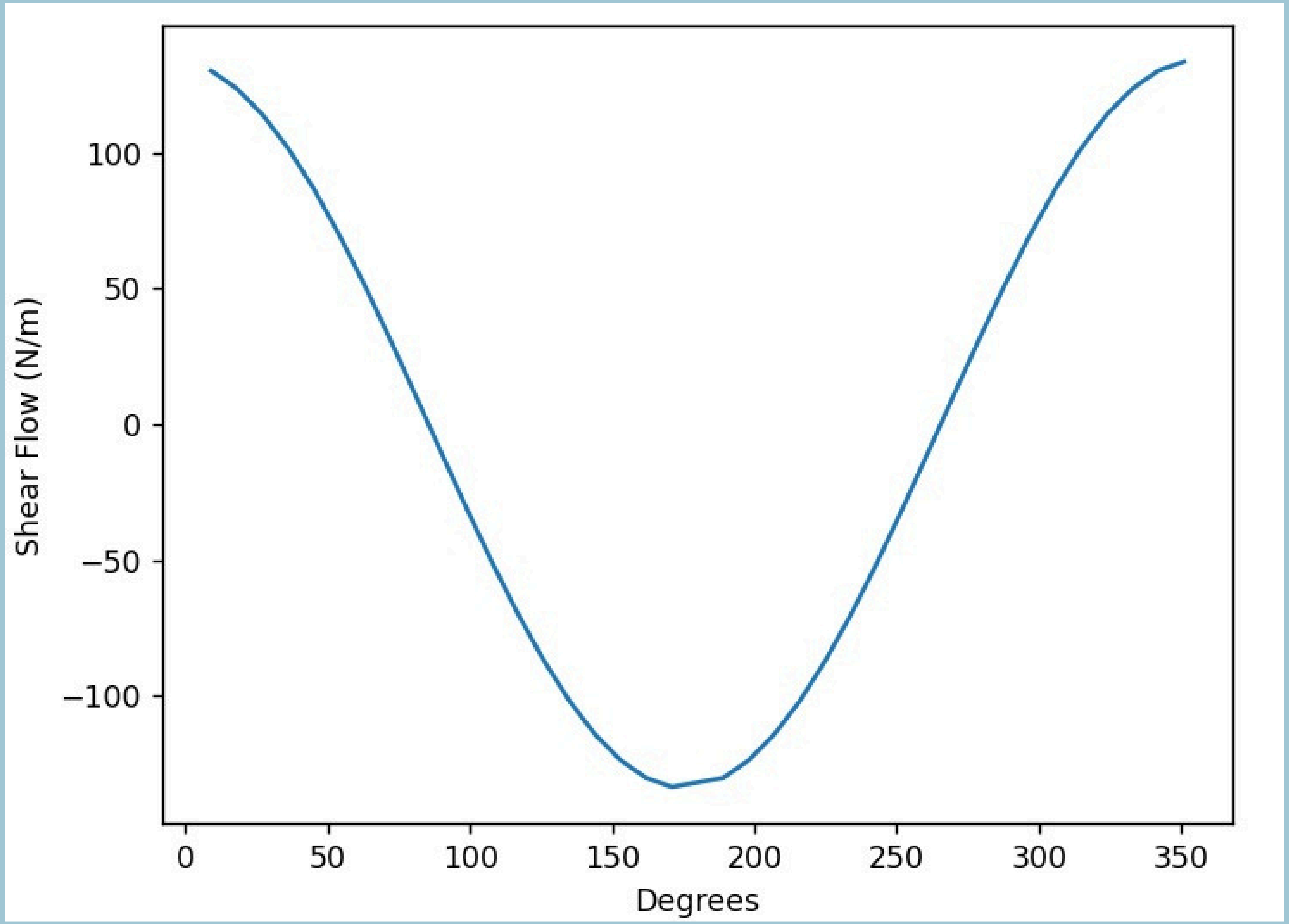
Fuselage - Bending Moment Diagram



Shear Flow in Fuselage CS

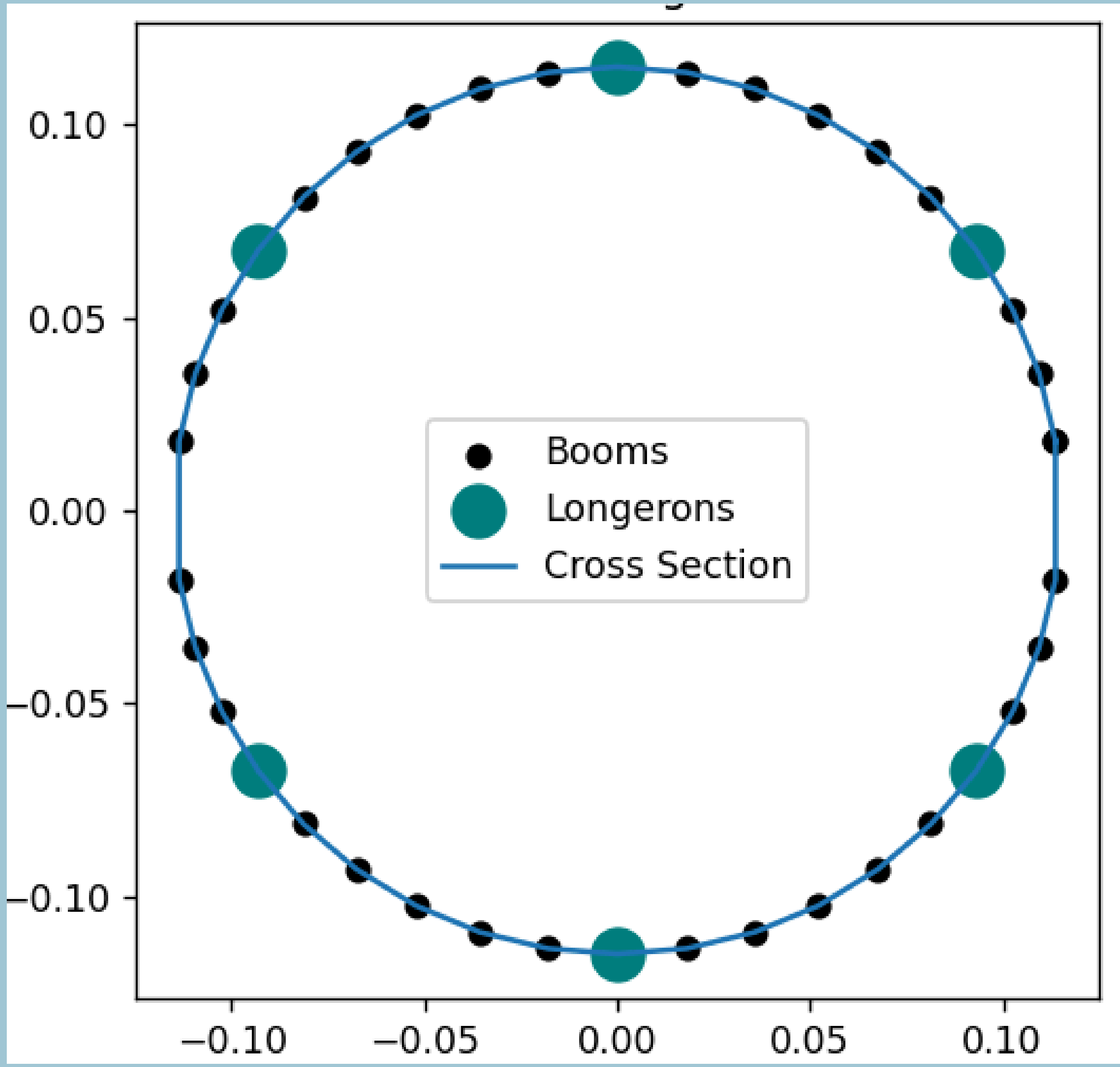


Al 6061 of 0.3 mm thickness used

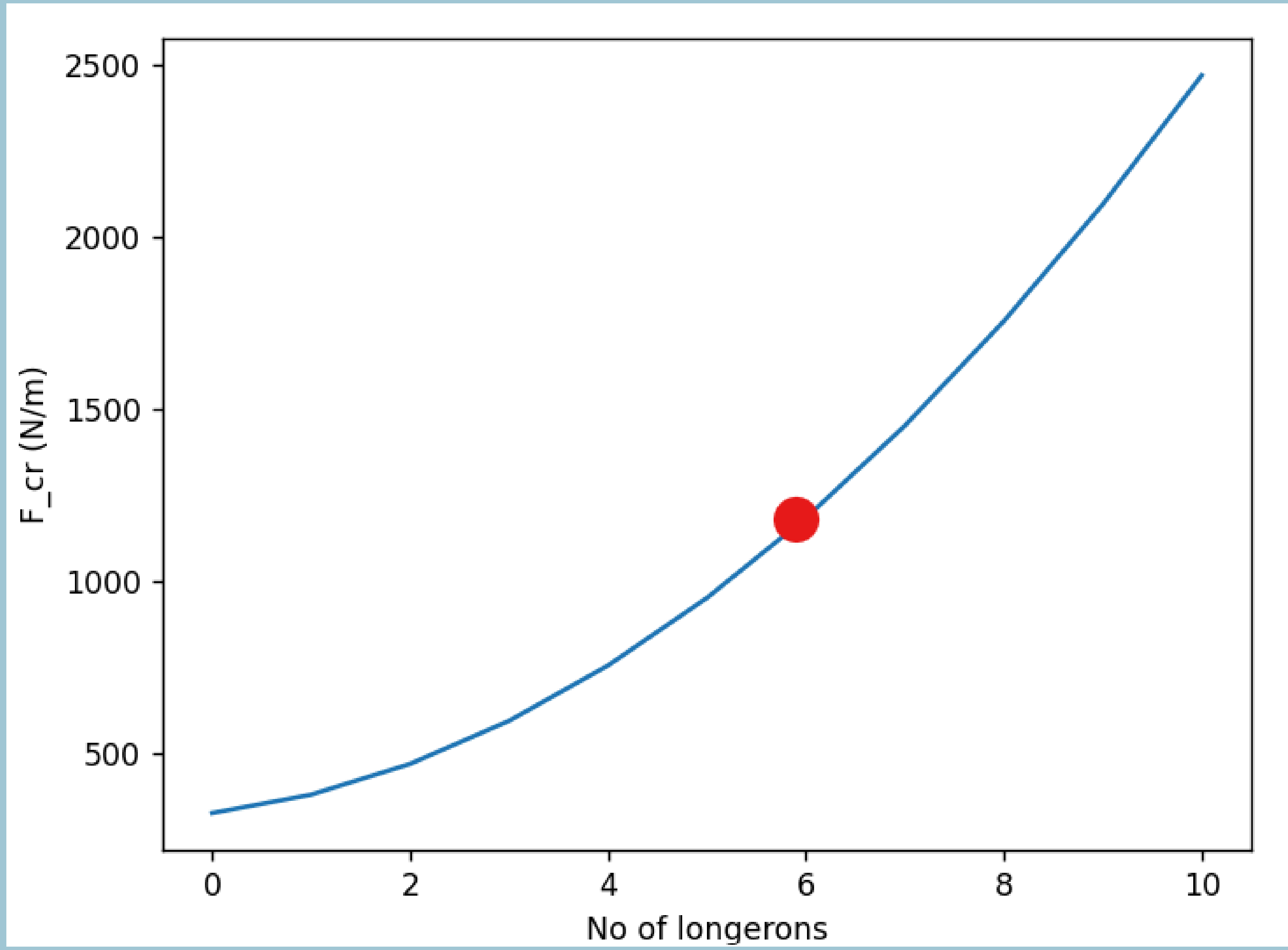


Radius = 0.115 m
Max Shear Flow = 120 N/m

Longeron Design



8 Frames along Fuselage
Al 6061 of 0.3 mm thickness used



F_cr = 1200 N/m **FOS = 10**

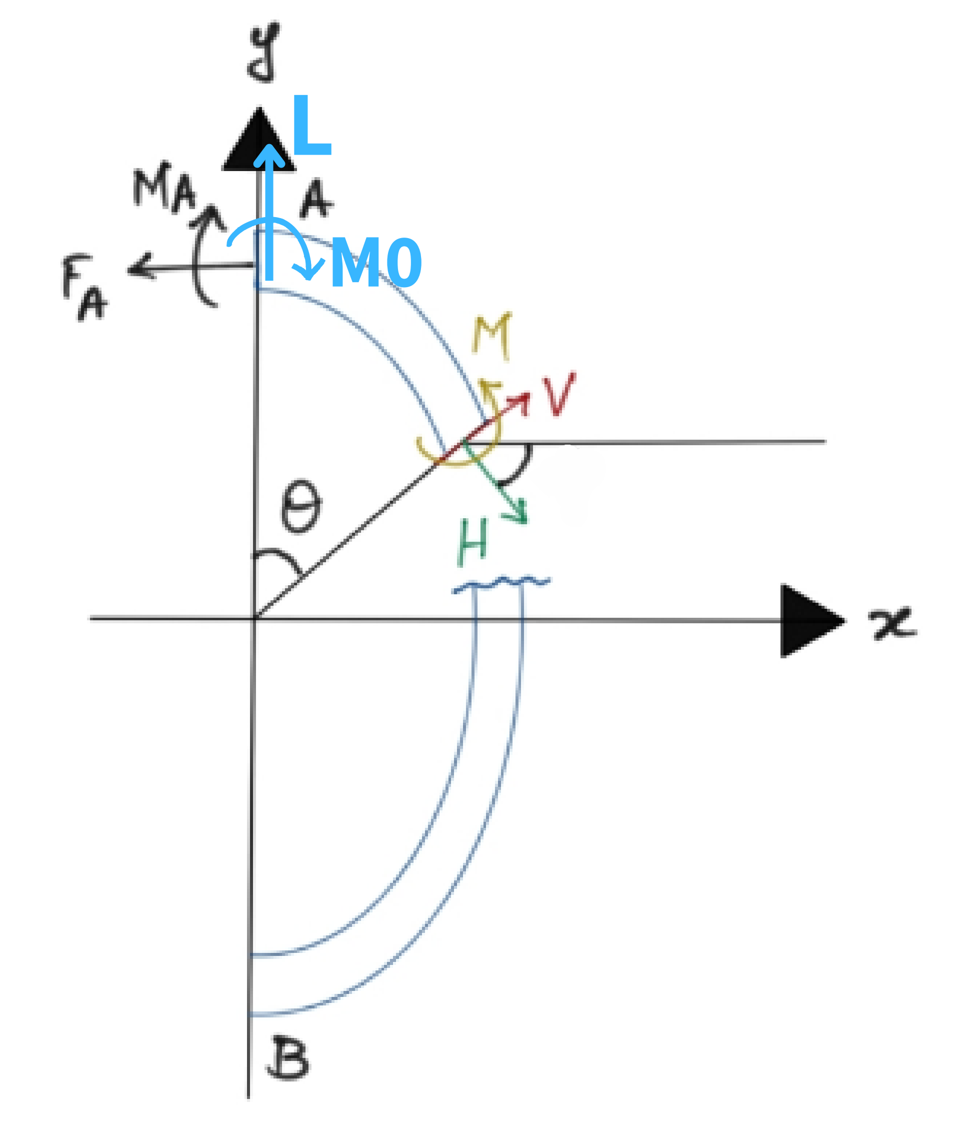
Ixx Required = 85670 mm^4
Ixx of Skin = 142780 mm^4
Ixx of each Longeron (wo skin) = 14280 mm^4

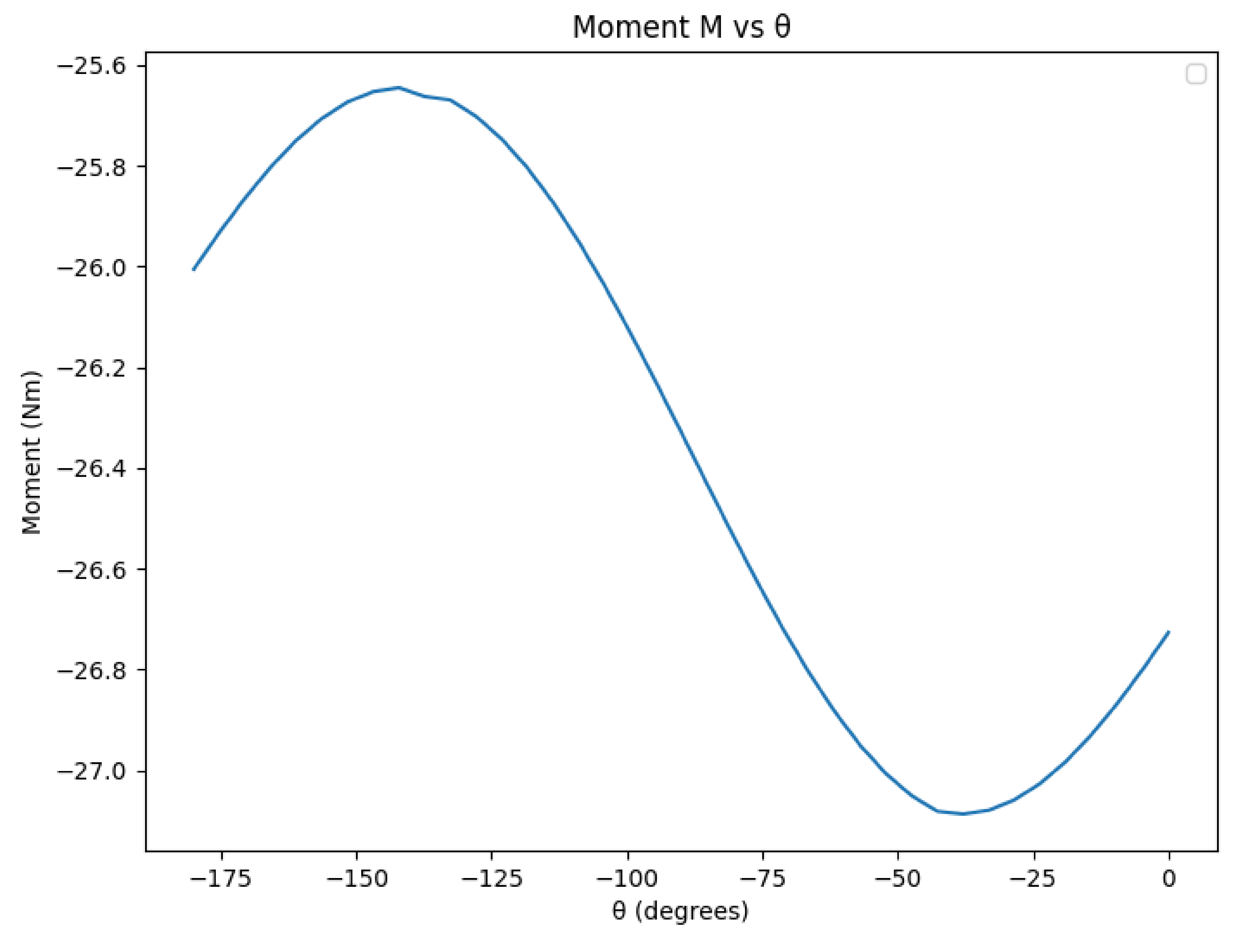
Bulkhead Sizing

$$M = M_A - F_A \cdot r \cdot (1 - \cos(\theta)) + M_q + M_0$$

$$H = \frac{F_A - F_{qx} - V \cdot \sin(\theta)}{\cos(\theta)}$$

$$V = \frac{F_{qy} - H \cdot \sin(\theta) - L}{\cos(\theta)}$$

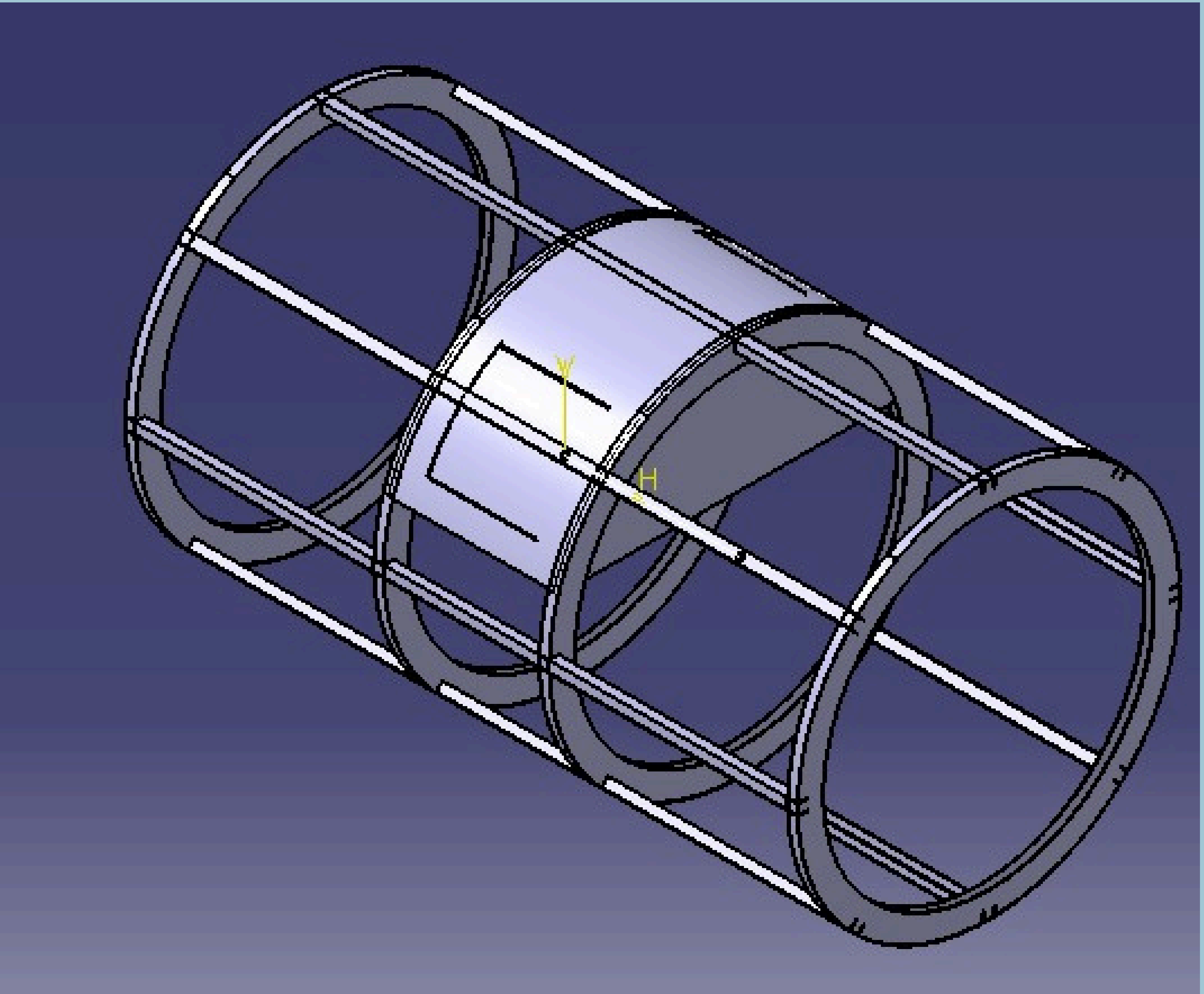


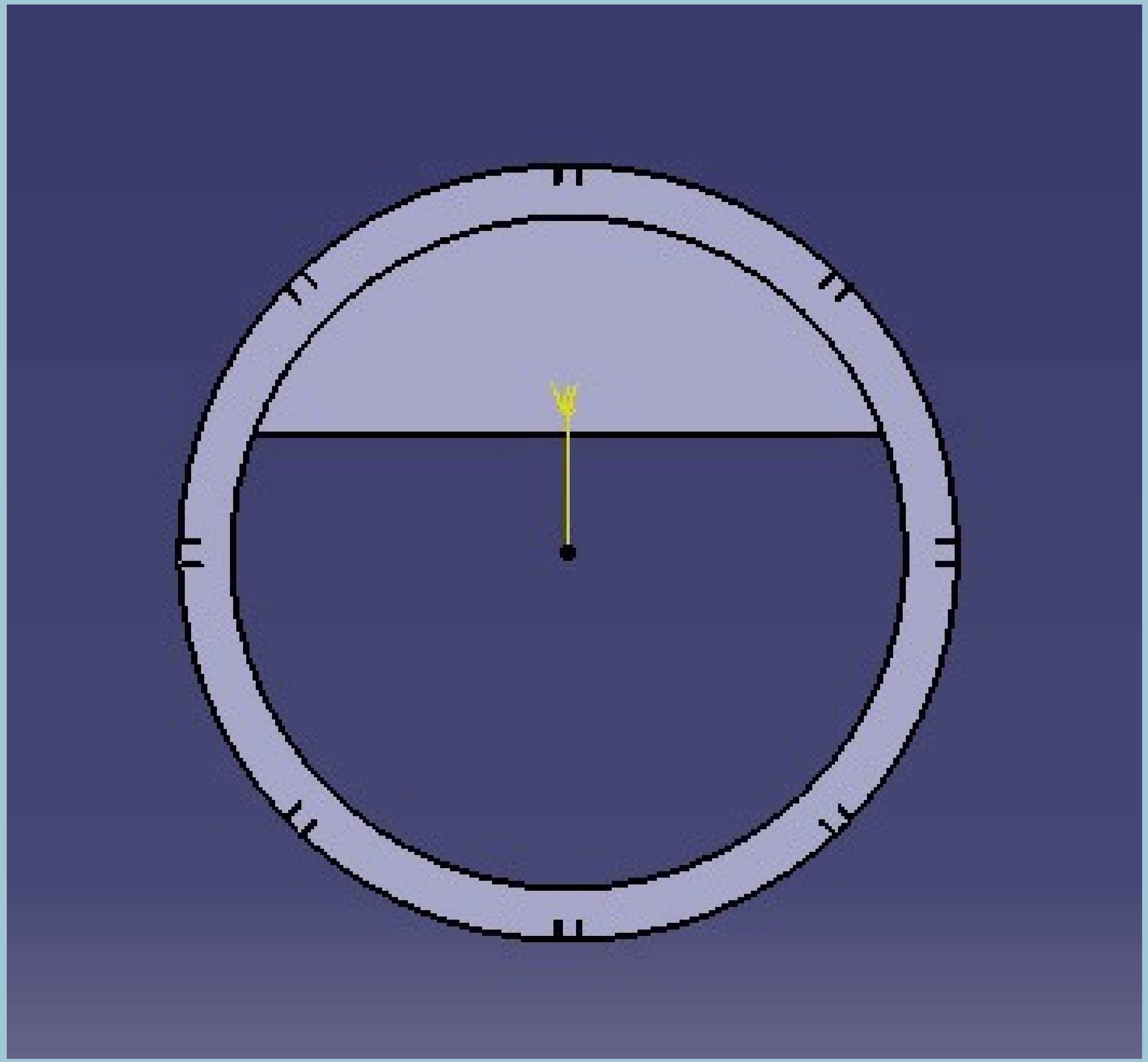


Maximum bending moment = 27 Nm

$$U_{\text{total}} = \frac{1}{2}\int \frac{M^2}{E \cdot I}\, d\theta + \frac{1}{2}\int \frac{H^2}{E \cdot A}\, d\theta + \frac{1}{2}\int \frac{V^2}{G \cdot A}\, d\theta$$

Bulkhead Design





Fuselage Wing Attachmentt

Thank You