AE461 Project Report
Design of a Premium Class Aircraft for Luxury Tourism

Group-2

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Chapter 1

Introduction

Mission: Design a fixed-wing aircraft tailored for luxury tourism, offering comfort, exclusivity, and immersive travel experiences to high-end clientele and foreign tourists.

1.1 Objectives

- Design and development of a luxury class aircraft.
- Considerations for comfort, exclusivity, and premium experiences.
- Compliance with mission requirements such as range, passenger capacity, etc.

1.2 Mission Requirements

1. **Passenger Capacity**: 10 to 15 passengers (10 Tourists + 5 Crew) = 1425 Kg (75 Kg Average Weight + 25 Kg Luggage of Tourist + 50 Kg Buffer Weight)
2. **Maximum Range**: 3000-3500 Km (To cover the entire India with the Tourist destinations, say, Andaman and Nicobar Islands)
3. **Cruise Speed**: 700-900 Km/Hr
4. **Aircraft Dimensions**: 15-18 m Length * 3-4.5 m Width * 2–3 m Height (to ensure proper space and luxury setting)
5. **Minimum Take-off and Landing Runway Length**: 1.5 – 1.8 Km
6. **Cruise Altitude**: 10 – 13 Km
7. **Service Ceiling altitude**: 12km
8. **Propulsion System**: Turbojet

Chapter 2

Weight Estimation

This chapter covers the methodologies and results of weight estimation. Include the pipeline of weight estimation code, the calculations, and the assumptions made.

$$\frac{W_f}{W_o} + \frac{W_p}{W_o} + \frac{W_e}{W_o} = 1 \tag{2.1}$$

where :

- W_f is the Fuel Weight
- W_p consists of the weight of the passenger, luggage, amenities and all the other non-flight essential cargo/goods.
- W_e consists of all the structural, avionics and essential weight of the airplane.
- W_o is the Total takeoff weight.

2.1 Payload W_p

We estimated the possible components inclusive in the weight of the payload:-

Category	**Amenities/Components**	**Weight (kg)**
Cockpit	Pilot Seats	40
Passenger Weights	Tourist(10 Tourist with avg. wt. 75)	750
	Crew Weight(5 crew members)	375
Luxury Passenger Compartment	Luxury seating (reclining seats with leather upholstery) (25 per seat)	250
	In-flight entertainment systems (large screens, premium audio systems) (5 per seat)	50
	Customized interior design (wood paneling, luxury finishes, drapes, artifacts)	100
	Meals and Drinks	40
Cargo Hold	Passenger General Luggage(50 kg per Tourist)	500
	Buffer Weight(for crew luggage)	100
	High-end goods storage	200

Category	Amenities/Components	Weight (kg)
Total	Margin of 10%	**2425**

2.2 Empty Weight W_e

We will consider the general trends (Ref. [1]) of airlines and private jets and incorporate them in our weight estimation code.

2.2.1 Nomenclature

Variable	Description
AR	Aspect Ratio
S	Wing area
v_{cruise}	Cruise speed
m_to_ft	Meters to feet conversion factor
kg_to_lb	Kilograms to pounds conversion factor
$MTOW$	Maximum Takeoff Weight
W_{zerofuel}	Zero fuel weight
t_c	Thickness to chord ratio
λ	Taper ratio
b	Wingspan
D_{fuselage}	Fuselage diameter
L_{fuselage}	Fuselage length
N_{ult}	Ultimate load factor
W_{engine}	Engine weight
$Passengers$	Number of passengers
sfc	Specific fuel consumption
L/D	Lift-to-drag ratio

Table 2.2: Index of Variables Used

2.2.2 Weight estimation formulas

Component	**Weight Formula**
Gust Load Factor	$N_{\text{gust}} = \frac{1+6.3\times AR\times S\times v_{\text{cruise}}\times(\text{m\_to\_ft})^3}{MTOW\times\text{kg\_to\_lb}\times(2+AR)}$
Manoeuvre Load Factor	$N_{\text{manu}} = \max\left(2.5, 2.1 + \frac{10900}{4530+MTOW\times\text{kg\_to\_lb}}\right)$
Ultimate Gust Load Factor	$N_{\text{gust\_ult}} = 1.5 \times N_{\text{gust}}$
Ultimate Manoeuvre Load Factor	$N_{\text{manu\_ult}} = 1.65 \times N_{\text{manu}}$
Ultimate Load Factor	$N_{\text{ult}} = \max(N_{\text{manu\_ult}}, N_{\text{gust\_ult}})$
Wing Weight	$W_{\text{wing}} = \frac{4.22\times S\times(\text{m\_to\_ft})^2+1.642\times10^{-6}\times N_{\text{ult}}\times(b\times\text{m\_to\_ft})^3\times(1+2\times\lambda)\times(MTOW\times W_{\text{zerofuel}}\times\text{kg\_to\_lb}^2)^{0.5}}{t_c\times S\times(\text{m\_to\_ft})^2\times(1+\lambda)\times\text{kg\_to\_lb}}$
Fuselage Weight	$W_{\text{fuselage}} = \frac{0.0737\times\left(2\times(D_{\text{fuselage}}\times\text{m\_to\_ft})\times(v_{\text{cruise}}\times\text{m\_to\_ft})^{0.338}\times(L_{\text{fuselage}}\times\text{m\_to\_ft})^{0.857}\times(MTOW\times\text{kg\_to\_lb}\times\right.}{\text{kg\_to\_lb}}$
Tail Weight	$W_{\text{tail}} = 0.10 \times W_{\text{wing}}$
Propulsion System Weight	$W_{\text{propulsion}} = 2.575 \times W_{\text{engine}}$
Landing Gear Weight	$W_{\text{landinggear}} = \frac{0.043\times MTOW^{0.65}}{\text{kg\_to\_lb}}$
Auxiliary Power Unit (APU)	$W_{\text{apu}} = 0.001 \times MTOW$
Oxygen System	$W_{\text{oxygen}} = \frac{30+1.2\times\text{Passengers}}{\text{kg\_to\_lb}}$
Miscellaneous Systems	$W_{\text{misc}} = 0.006 \times MTOW$
Surface Controls	$W_{\text{surfacecontrols}} = \frac{0.4915\times(MTOW\times\text{kg\_to\_lb})^{\frac{2}{3}}}{\text{kg\_to\_lb}}$
Fuel Weight	$W_{\text{fuel}} = MTOW \times \left(1 - \frac{1}{\exp\left(\frac{\text{range}\times1000\times\text{sfc}}{v_{\text{cruise}}\times L/D}\right)}\right)$

Table 2.3: Weight Formulas for Various Aircraft Components

2.2.3 Weight Estimation results

The initial parameters are given in Table 2.4.

Parameter	**Value**
Cruise Speed (v_{cruise})	700 km/h
Max Speed	900 km/h
Stall Speed	170 m/s
Range	5000 km
Cruise Altitude	10 km
Ceiling Altitude	11 km
Cruise Time	5 hours
Payload (m_{payload})	2425 kg
Number of Passengers	15
Wing Root Chord	$\approx 5m$
Wing Taper Ratio	0.4
Wing Span	31 m
Aspect Ratio (AR)	$\frac{b^2}{S}$ 12.85
Lift-to-Drag Ratio (L/D)	10
L/D max	15
Rate of Climb	10 m/s
Rate of climb @ ceiling	3 m/s
Rolling Friction Coeficient	0.6
Specific Fuel Consumption (SFC)	0.2 kg/hr/kW
Initial Guess for MTOW	20000 kg
Zero Fuel Weight	15000 kg
Fuselage Length (L_{fuselage})	16 m
Fuselage Diameter (D_{fuselage})	4 m
Horizontal Tail Area (S_{ht})	$10\% \times S$
Horizontal Tail Length (L_{ht})	8 m
Horizontal Tail Diameter (D_{ht})	5 m
Engine Weight (W_{engine})	240 kg
Tolerance for Convergence	10^{-6}

Table 2.4: Initial Parameters Used for Aircraft Weight Calculation

The general algorithm used is shown in Fig. 2.1.

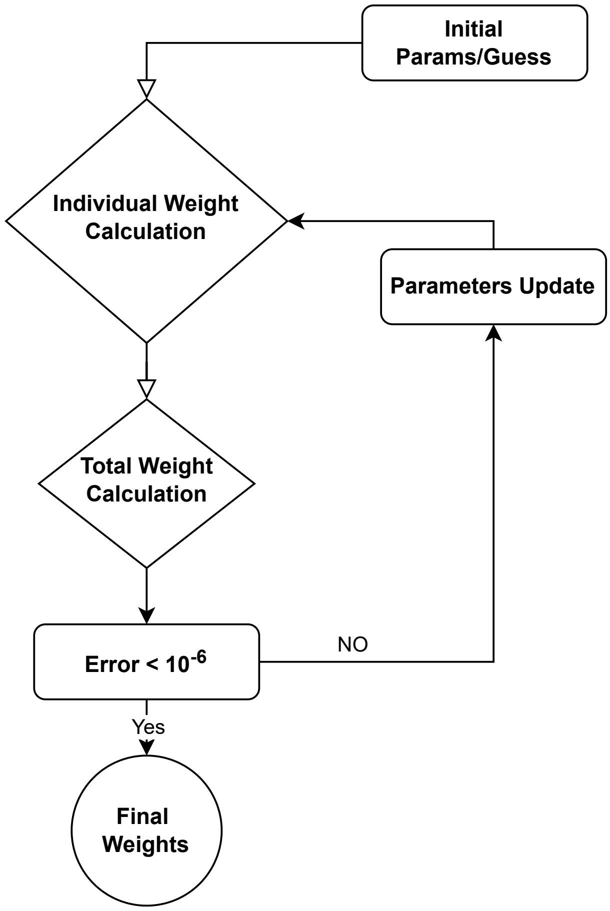


Figure 2.1: Weight estimation algorithm

The lift calculation uses very trivial formulas to ensure the weight estimation of the wings is practical. The results are given in Table 2.5.

Component	**Value (kg/kgf or kg/m²)**
Lift Force	59433.61 kgf
Converged MTOW	59094.14 kg
Empty Weight	55050.22 kg
Fuel Weight	1618.91 kg
Wing Weight	4578.10 kg
Fuselage Weight	42363.30 kg
Tail Weight	457.81 kg
Propulsion System Weight	6246 kg
Landing Gear Weight	24.64 kg
Auxiliary Power Unit Weight	59.09 kg
Oxygen System Weight	21.77 kg
Miscellaneous Systems Weight	354.56 kg
Surface Controls Weight	572.94 kg
Wing Loading	3945.01 kg/m^2

Table 2.5: Aircraft Weight Breakdown and Wing Loading

Chapter 3

Aerodynamic Analysis

We will decide the design point for our aircraft, using the following initial given parameters:

3.1 Initial Parameters

```
v_cruise = 750 * 1000 / 3600; % m/s (km/h to m/s)
range = 5000; % km
h = 8000; % cruise altitude in m
h_ceiling = 10000; % ceiling
cruise_time = 6; % hrs
mpayload = 2400; % kg (payload)
MTOW = 60000; % kg
g = 9.8; % m/s^2
Passengers = 15;
Vmax = 800 * 1000 / 3600; % m/s

ROC = 20; % rate of climb
LD_max = 15;
mu = 0.45; % Rolling friction coefficient
ROC_c = 5; % rate of climb at ceiling
V_stall = 150; % Minimum Stall velocity in m/s
```

3.2 Wing Loading Estimation and Sizing

3.2.1 Drag coefficient

Cd0 values for 5 aircrafts

1. Boeing 737-800: Cd0 = 0.0225
2. Airbus A320: Cd0 = 0.024
3. Dassault Falcon 900: Cd0 = 0.018
4. Bombardier CRJ700: Cd0 = 0.0275

5. Lockheed C-130 Hercules: Cd0 = 0.0325

MATLAB Code:

```
cd0_values = [0.0225, 0.024, 0.018, 0.0275, 0.0325];
CD0 = mean(cd0_values);
AR = 12;
e = 0.9; % oswald factor
k = 1 / (pi * e * AR);
```

- Drag coefficient : 0.0249
- k : 0.0295

3.2.2 Stall Velocity (V_s)

The formula for stall velocity can be derived using the following relation:

$$\left(\frac{W}{S}\right)_{V_s} = \frac{1}{2}\rho V_s^2 C_{L_{\max}}$$

Where:

- $\frac{W}{S}$ is the wing loading.
- ρ is the air density.
- V_s is the stall speed.
- $C_{L_{\max}}$ is the maximum lift coefficient.

3.2.3 Maximum Velocity ($V_{\max}$)

For maximum velocity, the formula is different for jet-powered and propeller-driven aircraft.

Jet-powered aircraft:

$$\left(\frac{T_{\mathrm{SL}}}{W}\right)_{V_{\max}} = \frac{\rho V_{\max}^2 C_{D_0}}{2\left(\frac{W}{S}\right)} + \frac{2K}{\rho\sigma V_{\max}^2}\left(\frac{W}{S}\right)$$

Where:

- T_{SL} is the sea-level thrust.
- C_{D_0} is the zero-lift drag coefficient.
- K is the induced drag factor.
- σ is the density ratio $\frac{\rho}{\rho_0}$.

3.2.4 Rate of Climb (ROC)

Jet-powered aircraft:

$$\left(\frac{T}{W}\right)_{\text{ROC}} = \frac{\text{ROC}}{\sqrt{2\frac{C_{D_0}}{K}\left(\frac{W}{S}\right)}} + \frac{1}{(L/D)_{\max}}$$

3.2.5 Ceiling Altitude

Jet-powered aircraft:

$$\left(\frac{T_{\text{SL}}}{W}\right)_{h_c} = \frac{\text{ROC}_c}{\sigma_c\sqrt{2\rho_c\frac{C_{D_0}}{K}\left(\frac{W}{S}\right)}} + \frac{1}{\sigma_c(L/D)_{\max}}$$

Where:

$$\sigma = (1 - 6.873 \times 10^{-6}h)^{4.26} \quad \text{for } h \leq 11\,\text{km}$$

$$\sigma = 0.2967e^{(1.7355-4.8075\times10^{-5}h)} \quad \text{for } h > 11\,\text{km}$$

3.2.6 Take-off Run Distance

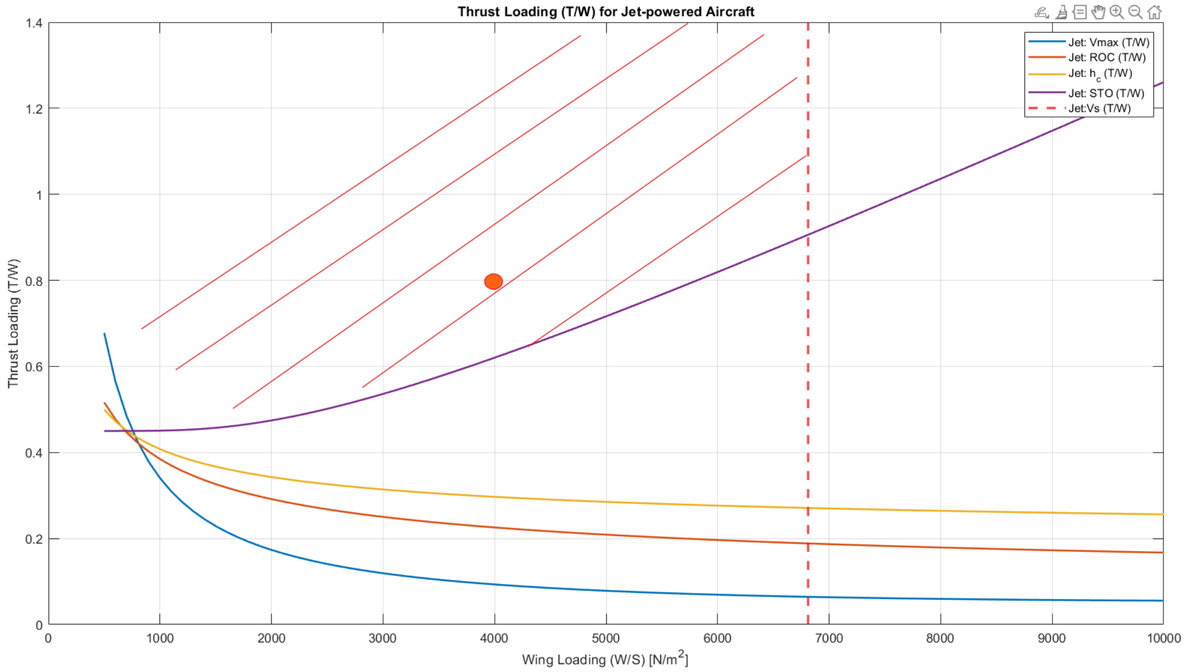
Jet-powered aircraft:

$$\left(\frac{T}{W}\right)_{S_{\text{TO}}} = \frac{\mu - \left(\mu + \frac{C_{D_G}}{C_{L_R}}\right)e^{(0.6\rho g C_{D_G} S_{\text{TO}})\frac{1}{W/S}}}{1 - e^{(0.6\rho g C_{D_G} S_{\text{TO}})\frac{1}{W/S}}}$$

Where:

- $C_{D_G} = C_{D_{\text{TO}}} - \mu C_{L_{\text{TO}}}$
- $C_{L_{\text{TO}}} = C_{L_c} + \Delta C_{L_{\text{flapTO}}}$
- $C_{L_R} = \frac{2mg}{\rho S V_R^2}$
- $C_{D_{\text{TO}}} = C_{D_0} + C_{D_{o_{\text{LG}}}} + C_{D_{o_{\text{HLDT}}}}$

From the above-desired formulas and conditions:



design point =(4000,0.8)

3.3 Engine Selection

Formula:

$$\text{Area } S = \frac{\text{MTOW}}{\left(\frac{W}{S}\right)}$$

MATLAB Code:

```
Area_final = MTOW * g / 4000;
```

MATLAB Code:

```
Thrust_final = 0.8 * MTOW * g;
fprintf("Area: %f m^2\nThrust: %f KN\n", Area_final, Thrust_final / 1000);
```

Output:

- Area: 122.625000 m^2
- Thrust: 362.400000 KN

Rolls-Royce Trent 900 can produce 360KN of thrust,upto 400KN with EFI systems.

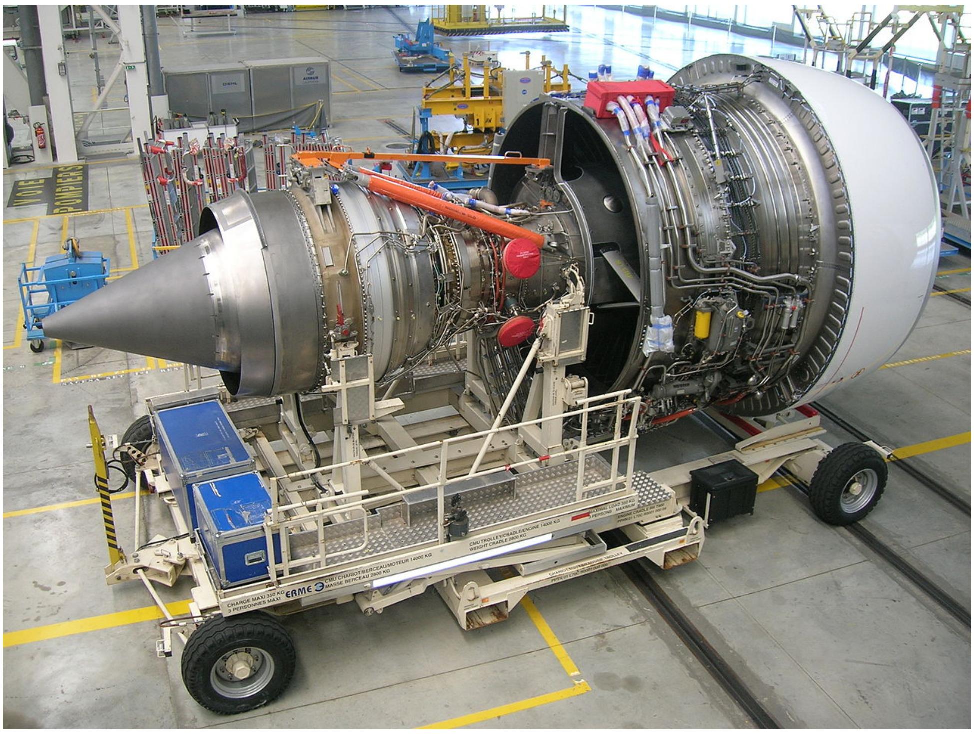


Figure 3.1: Rolls-Royce Trent 900

3.4 Wing estimation

3.4.1 Specifications

- Wing Loading: 4000 N/m^2
- Maximum Takeoff Weight (MTOW): 60,000 kg
- Zero-Lift Drag Coefficient (C_{D0}): 0.025
- Lift-to-Drag Ratio (L/D): 12, Max L/D: $1.2 \times 12 = 14.4$

3.4.2 Wing Area Calculation

Using the following code to calculate the wing area:

```
WingLoading = 4000;
g = 9.81;
MTOW = 60000 * g;
Cdo = 0.025;
L_D = 12;
L_D_max = 14.4;
Wing_area = MTOW / WingLoading;
fprintf("Wing Area: %f m^2", Wing_area);
```

Output:

$$\text{Wing Area} = 122.625 \text{ m}^2$$

3.4.3 Induced Drag Factor Calculation

Using:

$$k = \frac{1}{4 \cdot L_{D\max}^2 \cdot C_{D0}}$$

The MATLAB code:

```
k = 1 / (4 * L_D_max^2 * Cdo);
fprintf("Induced drag factor k: %f\n", k);
```

Output:

$$\text{Induced drag factor } k = 0.048225$$

3.4.4 Iterations for Optimal Aspect Ratio (AR)

Initial settings:

- Minimum AR: 8
- Maximum AR: 15
- Step size: 0.1

MATLAB code:

```
AR_min = 8;
AR_max = 15;
AR_step = 0.1;
tolerance = 1e-6;
max_iterations = 100;
L_D_max = 15;
Cdo = 0.02;

% Loop over a range of AR values
for AR = AR_min:AR_step:AR_max
    % Iterative process
    while error > tolerance && iteration < max_iterations
        % Calculate e and k_new for each AR
    end
end
```

Output:

$$\text{Optimal Aspect Ratio } AR = 8.000000$$

$$\text{Maximum Oswald Efficiency Factor } e = 0.716197$$

$$\text{Corresponding Induced Drag Factor } k = 0.055556$$

3.4.5 Wingspan and Taper Ratio Iteration

Using the calculated optimal AR and wing area:

```
b = sqrt(best_AR * Wing_area);
fprintf('Wingspan b: %f m\n', b);
```

Output:

$$\text{Wingspan } b = 31.320920 \text{ m}$$

3.4.6 Root Chord vs. Taper Ratio

MATLAB code for plotting root chord against taper ratio:

```
lambda_values = 0.2:0.01:1;
Cr_values = zeros(size(lambda_values));

for i = 1:length(lambda_values)
    lambda = lambda_values(i);
    Cr_values(i) = (2 * Wing_area) / ((1 + lambda) * b);
end

plot(lambda_values, Cr_values);
xlabel('Taper Ratio (\lambda)');
ylabel('Root Chord (Cr) [m]');
```

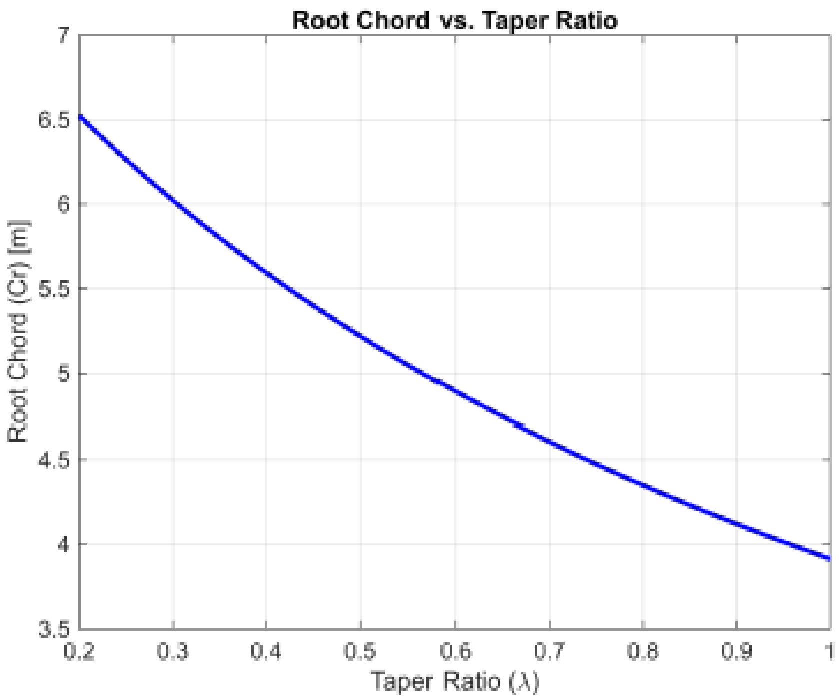


Figure 3.2: Taper ratio plot

Choosing $\lambda = 0.4$ from 3.2, results in a root chord of 5.5 m, tip chord of 2.2 m with $b = 31.32$ m.

WING PLANFORM

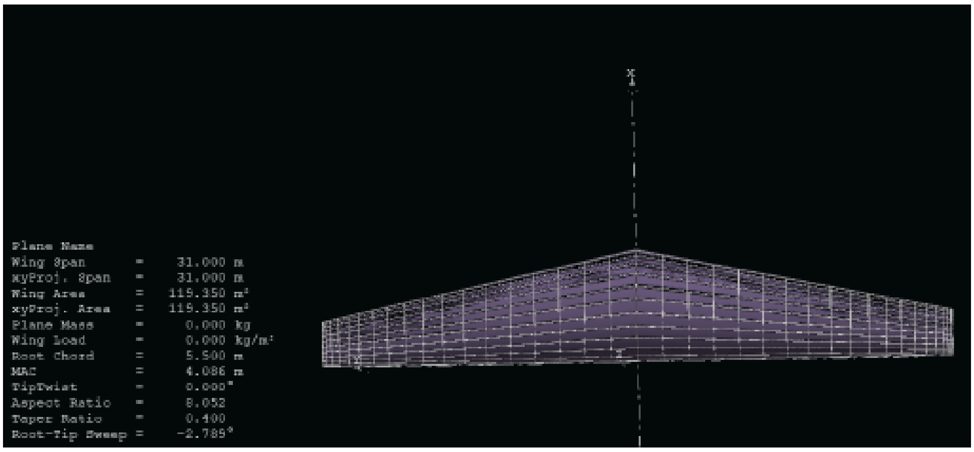


Figure 3.3: Wing planform preliminary design

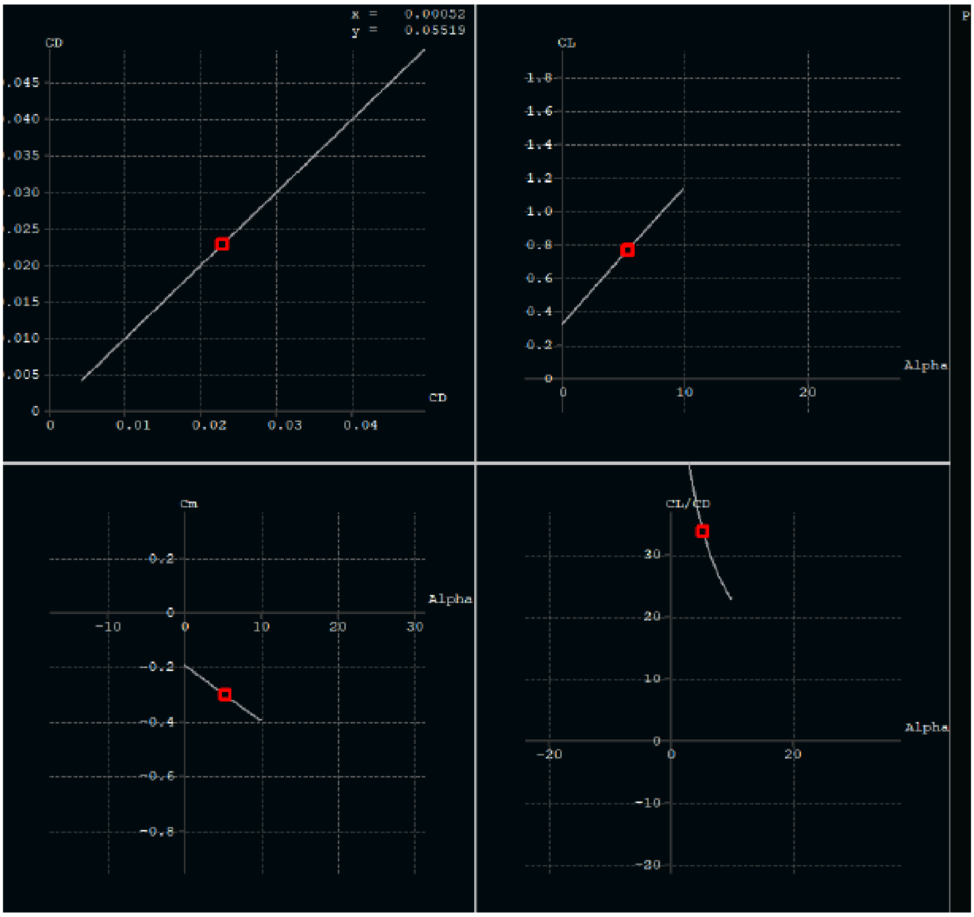


Figure 3.4: Preleminary XFLR analysis for wing

Chapter 4

Design of major components

Discuss the wing design, fuselage shape selection, tail sizing, and considerations for passenger comfort.

4.1 Wing design and Airfoil selection

4.1.1 Specifications

- Wing Area: 122 m^2
- Span : 30 m
- Cruise Speed: 190–200 m/s
- Maximum Takeoff Weight (MTOW): 60,000 kg
- Cruise Altitude: 10 km
- Air Density at Cruise Altitude: 0.41 kg/m^3
- Oswald Efficiency Factor (e): 0.71
- Aspect Ratio (AR): 8
- Tail Taper Ratio (λ): 0.7

4.1.2 Design Lift Coefficient Calculation

```
wing_area = 122; % m^2
cruise_speed = 200; % m/s
g = 9.81; % m/s^2
Weight_force = 60000 * g; % N
rho = 0.41; % kg/m^3
e_oswald = 0.71;
AR_wing = 8;
pi = 3.14;
lambda = 0.7; % tail taper ratio
```

The calculated 3D design lift coefficient (C_L) is as follows:

$$C_{L\text{design}} = \frac{2 \times \text{Weight\_force}}{\rho \times \text{cruise\_speed}^2 \times \text{wing\_area}}$$

Output:

$$\text{3-D Design lift coefficient} = 0.588365$$

4.1.3 Possible Combinations of $C_{L\alpha}$ and C_{L0}

The range for $C_{L\alpha}$ is typically between 0.08 to 0.15. Using the MATLAB code (HW8), the following table lists possible combinations of the design angle of attack (α_d) and zero-lift angle of attack (α_{CL0}).

Design AoA (α_d)	Zero-Lift AoA (α_{CL0})	$C_{L\alpha}$ (2D)
2.0	-2.5	0.1317
2.0	-2.0	0.1483
2.5	-2.5	0.1185
2.5	-2.0	0.1317
2.5	-1.5	0.1483
3.0	-2.5	0.1076
3.0	-2.0	0.1185
3.0	-1.5	0.1317
3.0	-1.0	0.1483
3.5	-2.5	0.0986
3.5	-2.0	0.1076
3.5	-1.5	0.1185
3.5	-1.0	0.1317
3.5	-0.5	0.1483
4.0	-2.5	0.0910
4.0	-2.0	0.0986
4.0	-1.5	0.1076
4.0	-1.0	0.1185
4.0	-0.5	0.1317
4.0	0.0	0.1483
4.5	-2.5	0.0845
4.5	-2.0	0.0910
4.5	-1.5	0.0986
4.5	-1.0	0.1076
4.5	-0.5	0.1185
4.5	0.0	0.1317
5.0	-2.0	0.0845
5.0	-1.5	0.0910
5.0	-1.0	0.0986
5.0	-0.5	0.1076
5.0	0.0	0.1185
5.5	-1.5	0.0845
5.5	-1.0	0.0910
5.5	-0.5	0.0986
5.5	0.0	0.1076
6.0	-1.0	0.0845
6.0	-0.5	0.0910
6.0	0.0	0.0986
6.5	-0.5	0.0845
6.5	0.0	0.0910
7.0	0.0	0.0845

Table 4.1: Possible Combinations for $C_{L\alpha}$ (2D)

	Airfoil Characteristics							
Airfoil	C_{lo}	$C_{l\alpha}$	$C_{l\,Max}$	$\alpha(C_l = 0)$(rad)	α(stall)(rad)	C_m(ac)	α(Cl= 0)(deg)	$C_{l\alpha}$(degrees)
NACA 0006	0	5.72957	0.9	0	0.1745	0	0	0.09994917
NACA 1408	0.1	5.72957	1.5	-0.01745	0.26175	-0.025	-1.00031847	0.09994917
NACA 1412	0.125	6.4457663	1.6	-0.0193695	0.270475	-0.025	-1.1103535	0.11244281
NACA 4424	0.4	5.72957	1.4	-0.0698	0.2792	-0.075	-4.00127389	0.09994917
NACA 23012	0.1	5.72957	1.8	-0.01745	0.3141	-0.0125	-1.00031847	0.09994917
NACA 23018	0.1	5.72957	1.6	-0.01745	0.2792	0	-1.00031847	0.09994917
NACA 63-009	0	5.72957	1.15	0	0.183225	0	0	0.09994917
NACA 63-209	0.1667	6.6864082	1.4	-0.02492907	0.2094	-0.025	-1.42905497	0.11664068
NACA 631-212	0.2	6.6864082	1.6	-0.02991454	0.2443	-0.05	-1.71484596	0.11664068
NACA 64-108	0	5.72957	1.1	0	0.1745	0.0125	0	0.09994917
NACA 643-218	0.1333	6.3655523	1.5	-0.026175	0.3141	-0.025	-1.50047771	0.11104352
NACA 65-209	0.15	6.4457663	1.3	-0.0232085	0.2094	-0.0375	-1.33042357	0.11244281

Figure 4.1: Airfoil database

From exercise results, configurations matching certain airfoils were identified.The possible airfoil configurations were:

- NACA 23012 is similar to configuration: 5 -1 0.0986
- NACA 63-209 is similar to configuration: 4 -1.5 0.1076
- NACA 643-218 is similar to configuration: 4 -1.5 0.1076
- NACA 65-209 is similar to configuration: 3.5 -1.5 0.1185

The chosen configuration is **NACA 65-209** with a design angle of attack of **3.5 degrees.**

The mean aerodynamic chord $\bar{c}$ is calculated as:

$$\bar{c} = \frac{2}{3} c_r \frac{1 + \lambda_v + \lambda_v^2}{1 + \lambda_v} \bar{c} = 0.408m \tag{4.1}$$

From the above data, the CAD model(Ref. 4.2) for the half-wing was designed.

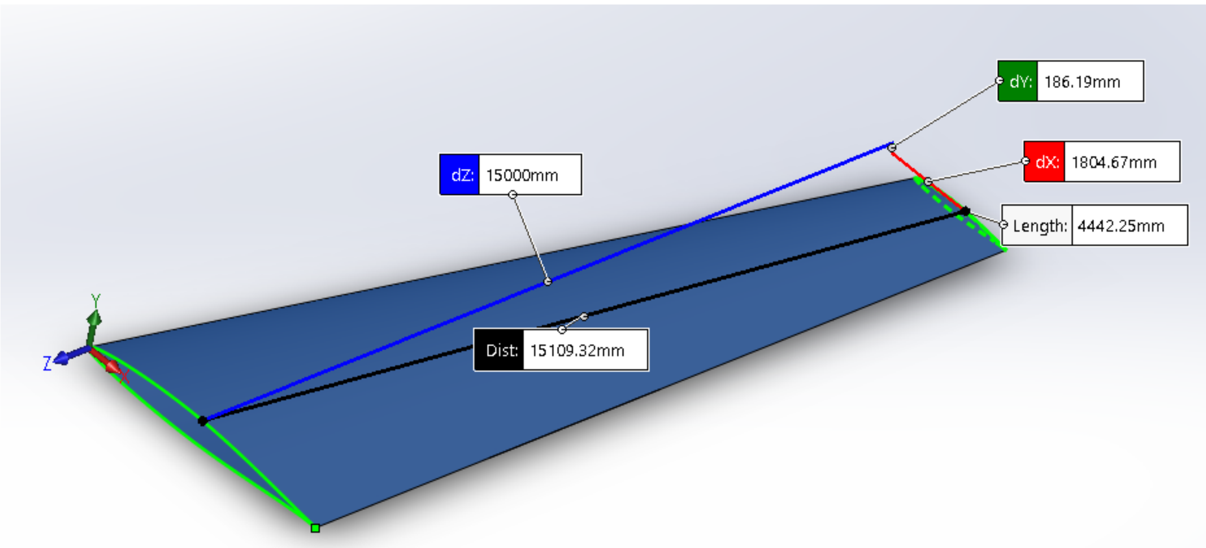


Figure 4.2: NACA 65-209 wing

4.2 Fuselage Design

The component specifications are given in table 4.2.

Component	**Dimensions (L x W x H)**	**Weight (kg)**	**Description**
Luxury Seats	0.8m x 1m x 1.1m	25	High-end passenger seat, adjustable features
Sofa	2m x 1.2m x 0.9m	50	2-seater luxury sofa for seating arrangement
Table	2m x 0.8m x 0.75m	30	Dining or working table, foldable options
TV	1.2m x 0.05m x 0.7m	10	50-inch flat-screen TV for entertainment
Kitchen/Pantry	2.2m x 2.6m x 2.2m	150	Compact kitchen setup with cooking facilities
Washroom	1.5m x 1.2m x 2.2m	200	Includes toilet, shower, and wash basin
Cockpit	2m x 2.8m x 2.2m	30	Ergonomic seat for pilot with controls access
Crew seats	1m x 0.6m x 1.1m	15	Simple seats for crew members

Table 4.2: Components Specifications

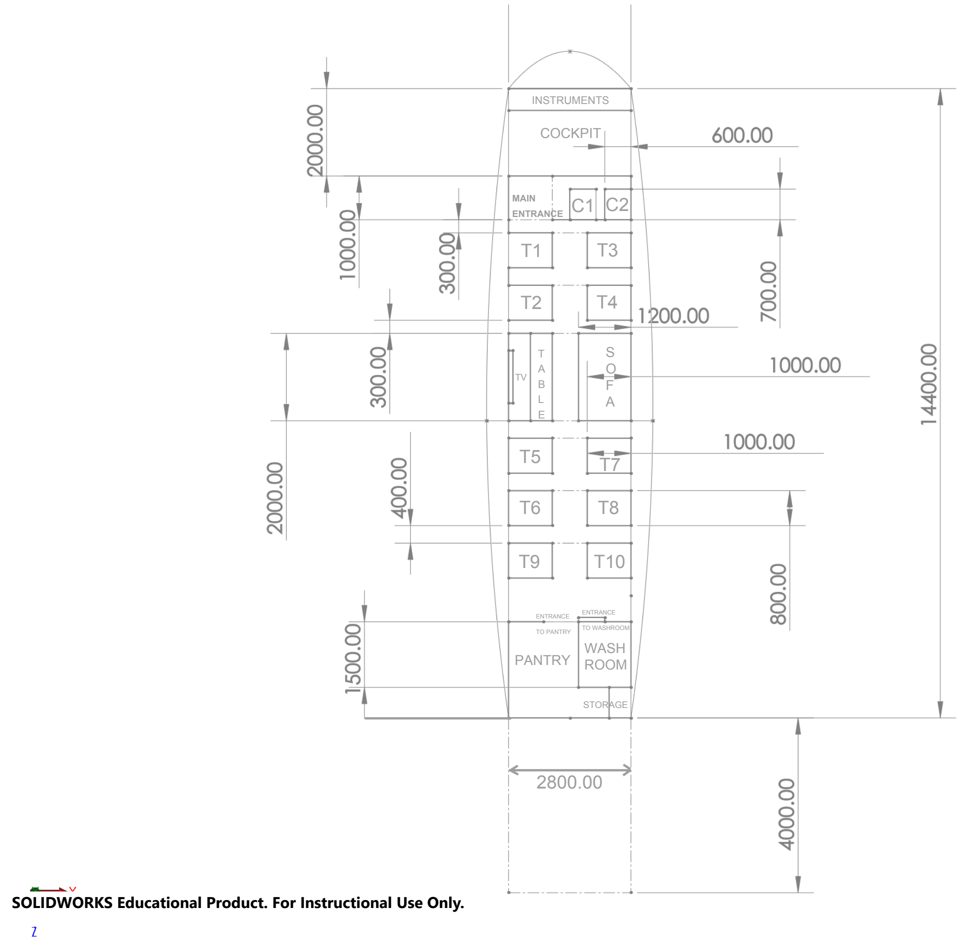


Figure 4.3: Solidworks 2-D interior design

A CAD model (Fig. 4.5) with an interior height of 2 m was made in Solidworks. It has a glass cockpit and space for an engine.

Figure 4.4: Side View

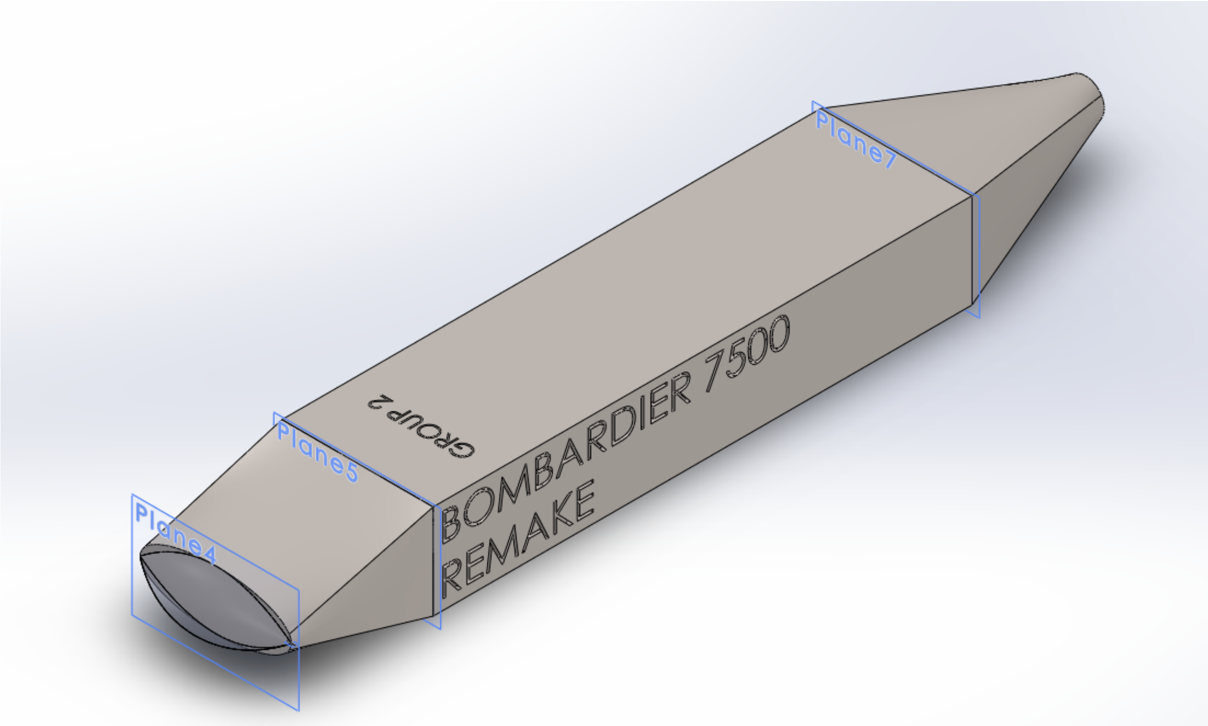


Figure 4.5: Isometric view

4.3 horizontal tail sizing

Since our aircraft is a luxury jet with a focus on maximum comfort, we chose the tail-to-wing area ratio as:

$$\frac{S_t}{S} = 0.4 \tag{4.2}$$

Given that the wing area $S = 122\,\mathrm{m}^2$, we can calculate the tail area S_t as:

$$S_t = S \times \frac{S_t}{S} = 122 \times 0.4 = 48.8\,\mathrm{m}^2 \tag{4.3}$$

The aspect ratio of the tail, AR_t, is taken as 3. From the graph, we obtained the tail span b_t:

$$b_t = 12.09\,\mathrm{m} \tag{4.4}$$

Based on a literature review, the typical tail taper ratio λ_t for general aviation aircraft is between 0.7 and 1. We chose:

$$\lambda_t = 0.7 \tag{4.5}$$

Next, we calculate the root chord of the tail C_{r_t} and the tip chord of the tail C_{e_t}:

$$\begin{aligned} C_{r_t} &= \frac{2S_t}{b_t(1+\lambda_t)} && (4.6) \\ &= \frac{2 \times 48.8}{12.09 \times (1+0.7)} && (4.7) \\ &= 4.74\,\mathrm{m} && (4.8) \end{aligned}$$

$$C_{e_t} = \lambda_t \times C_{r_t} \tag{4.9}$$
$$= 0.7 \times 4.74 = 3.31\,\mathrm{m} \tag{4.10}$$

Now, we calculate the mean aerodynamic chord of the tail $\bar{c}_t$:

$$\bar{c}_t = \frac{2}{3} C_{r_t} \frac{1 + \lambda_t + \lambda_t^2}{1 + \lambda_t} \tag{4.11}$$

Substituting the values:

$$\bar{c}_t = \frac{2}{3} \times 4.74 \times \frac{1 + 0.7 + 0.7^2}{1 + 0.7} \tag{4.12}$$
$$= 4.06\,\mathrm{m} \tag{4.13}$$

Finally, we find the 3-dimensional lift curve slope $C_{L_{\alpha_t}}$(3-D) for the tail. Given the 2-dimensional lift curve slope $c_{l_{\alpha_t}}$(2-D), we use the following relation:

$$C_{L_{\alpha_t}}(\text{3-D}) = \frac{c_{l_{\alpha_t}}(\text{2-D})}{1 + \frac{c_{l_{\alpha_t}}(\text{2-D})}{\pi e_t AR_t}} \tag{4.14}$$

where e_t is the Oswald efficiency factor for the tail.

Since we are using a symmetrical airfoil NACA 0012, the 2-dimensional lift curve slope is:

$$c_{l_{\alpha_t}}(\text{2-D}) = 7.79\,\text{per radian} \tag{4.15}$$

Given:

$$AR_t = 3 \tag{4.16}$$
$$e_t = 0.8 \tag{4.17}$$

We substitute these values into the equation for the 3-dimensional lift curve slope $C_{L_{\alpha_t}}$(3-D):

$$C_{L_{\alpha_t}}(\text{3-D}) = \frac{c_{l_{\alpha_t}}(\text{2-D})}{1 + \frac{c_{l_{\alpha_t}}(\text{2-D})}{\pi e_t AR_t}} \tag{4.18}$$

Substituting the values:

$$C_{L_{\alpha_t}}(\text{3-D}) = \frac{7.79}{1 + \frac{7.79}{\pi \times 0.8 \times 3}} \tag{4.19}$$
$$= \frac{7.79}{1 + \frac{7.79}{7.5398}} \tag{4.20}$$
$$= \frac{7.79}{2.0328} \tag{4.21}$$
$$= 3.39 \tag{4.22}$$

Therefore, the 3-dimensional lift curve slope $C_{L_{\alpha_t}}$(3-D) is:

$$C_{L_{\alpha_t}}(\text{3-D}) = 3.39 \tag{4.23}$$

With all the data given above, A CAD model(Ref. 4.6) was designed with half span.

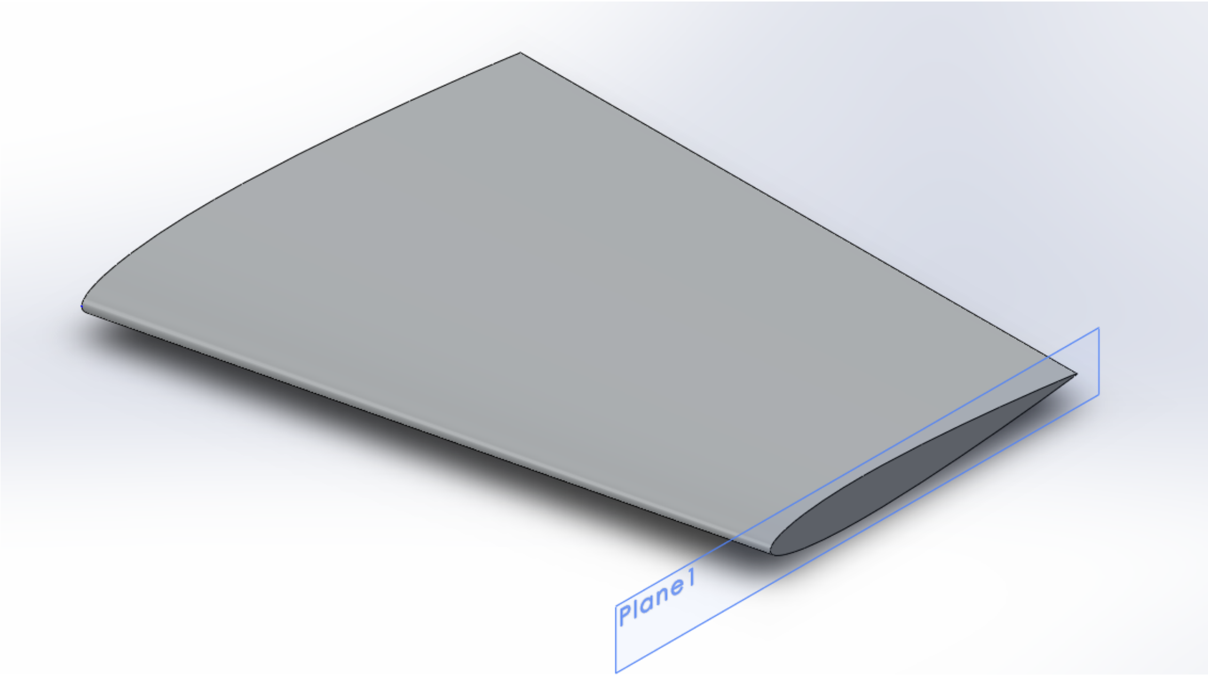


Figure 4.6: Horizontal tail isometric view

4.4 Vertical Tail Design

To design the vertical tail, we will use a symmetrical NACA airfoil. We follow these steps:

4.4.1 Determine Planform Area S_{vt} of the Vertical Tail

The planform area S_{vt} is determined by varying the vertical tail area-to-wing area ratio $\frac{S_{vt}}{S}$ over the range [0.2:0.05:0.6]. Given the wing area $S = 122\,\mathrm{m}^2$, we calculate S_{vt} for each chosen ratio as follows:

$$S_{vt} = S \times \frac{S_{vt}}{S} \tag{4.24}$$

where:

$$S = 122\,\mathrm{m}^2 \tag{4.25}$$

we take $\frac{S_{vt}}{S} = 0.2$:

$$S_{vt} = 122 \times 0.2 = 24.4\,\mathrm{m}^2 \tag{4.26}$$

4.4.2 Aspect Ratio of the Vertical Tail AR_{vt}

The aspect ratio AR_{vt} is varied within the range [2:1:6]. Assuming $AR_{vt} = 6$ as an example, we calculate the span b_{vt} based on S_{vt}:

$$b_{vt} = \sqrt{S_{vt} \times AR_{vt}} \tag{4.27}$$

If $S_{vt} = 24.4\,\text{m}^2$ and $AR_{vt} = 6$:

$$b_{vt} = \sqrt{24.4 \times 6} \tag{4.28}$$
$$= 12.09\,\text{m} \tag{4.29}$$

4.4.3 Root Chord $c_{r_{vt}}$ and Tip Chord $c_{t_{vt}}$

We assume a taper ratio $\lambda_{vt} = 0.7$ based on general aviation guidelines. Then, the root chord $c_{r_{vt}}$ and tip chord $c_{t_{vt}}$ can be calculated as follows:

$$c_{r_{vt}} = \frac{2S_{vt}}{b_{vt}(1 + \lambda_{vt})} \tag{4.30}$$

For example, if $S_{vt} = 24.4\,\text{m}^2$, $b_{vt} = 12.09\,\text{m}$, and $\lambda_{vt} = 0.7$:

$$c_{r_{vt}} = \frac{2 \times 24.4}{12.09 \times (1 + 0.7)} \tag{4.31}$$
$$= 4.74\,\text{m} \tag{4.32}$$

$$c_{t_{vt}} = \lambda_{vt} \times c_{r_{vt}} \tag{4.33}$$

So,

$$c_{t_{vt}} = 0.7 \times 4.74 \tag{4.34}$$
$$= 3.31\,\text{m} \tag{4.35}$$

4.4.4 Mean Aerodynamic Chord $\bar{c}_{vt}$

The mean aerodynamic chord $\bar{c}_{vt}$ is calculated as:

$$\bar{c}_{vt} = \frac{2}{3} c_{r_{vt}} \frac{1 + \lambda_{vt} + \lambda_{vt}^2}{1 + \lambda_{vt}} \tag{4.36}$$

Substituting the values:

$$\bar{c}_{vt} = \frac{2}{3} \times 4.74 \times \frac{1 + 0.7 + 0.7^2}{1 + 0.7} \tag{4.37}$$
$$= 4.06\,\text{m} \tag{4.38}$$

5. 3D Lift Curve Slope $C_{L_{\alpha_{vt}}}$(3-D)

Using the 2D lift curve slope $c_{l_{\alpha_{vt}}}$(2-D) $= 7.79$ per radian (as for NACA 0012), the aspect ratio $AR_{vt} = 6$, and the Oswald efficiency factor $e_{vt} = 0.8$, the 3D lift curve slope is:

$$C_{L_{\alpha_{vt}}}(\text{3-D}) = \frac{c_{l_{\alpha_{vt}}}(\text{2-D})}{1 + \frac{c_{l_{\alpha_{vt}}}(\text{2-D})}{\pi e_{vt} AR_{vt}}} \tag{4.39}$$

Substitute the values:

$$C_{L_{\alpha_{vt}}}(\text{3-D}) = \frac{7.79}{1 + \frac{7.79}{\pi \times 0.8 \times 6}} \tag{4.40}$$

$$= \frac{7.79}{1 + \frac{7.79}{15.0796}} \tag{4.41}$$

$$= \frac{7.79}{1.516} \tag{4.42}$$

$$= 5.14 \tag{4.43}$$

Therefore, the 3-dimensional lift curve slope $C_{L_{\alpha_{vt}}}(\text{3-D})$ is:

$$C_{L_{\alpha_{vt}}}(\text{3-D}) = 5.14 \tag{4.44}$$

The CAD model(Ref. 4.7) was created using the above values.

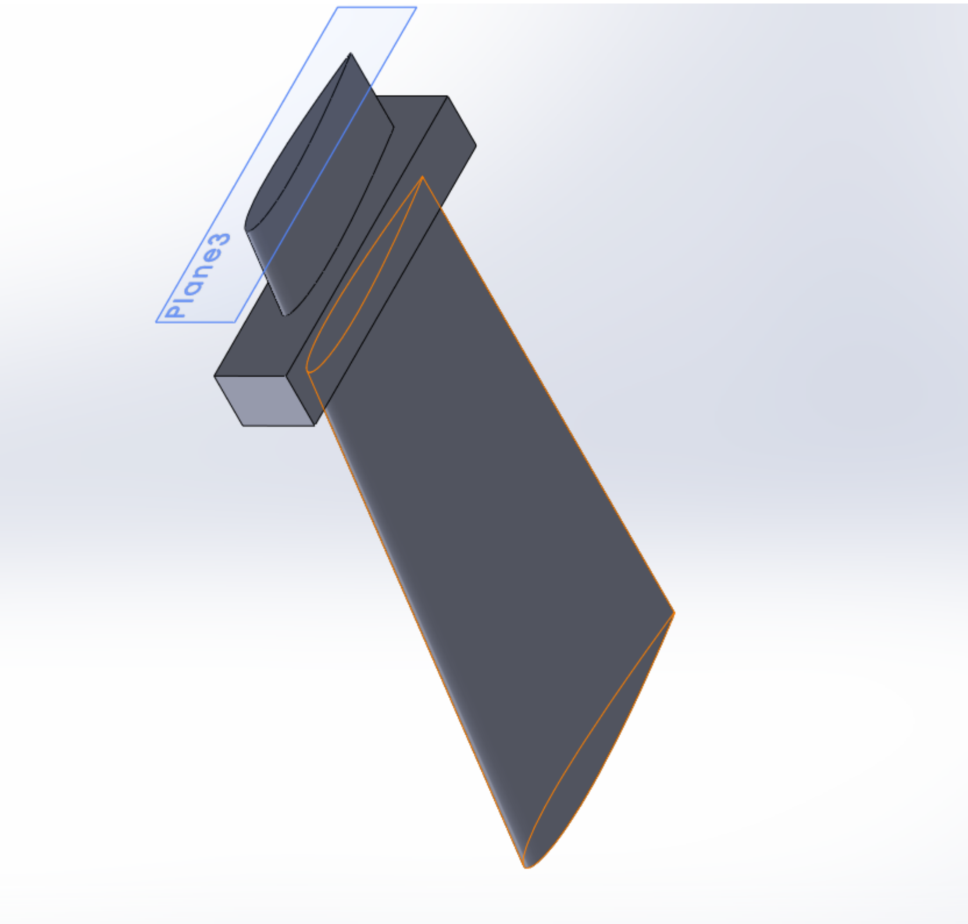


Figure 4.7: Vertical Stabilizer

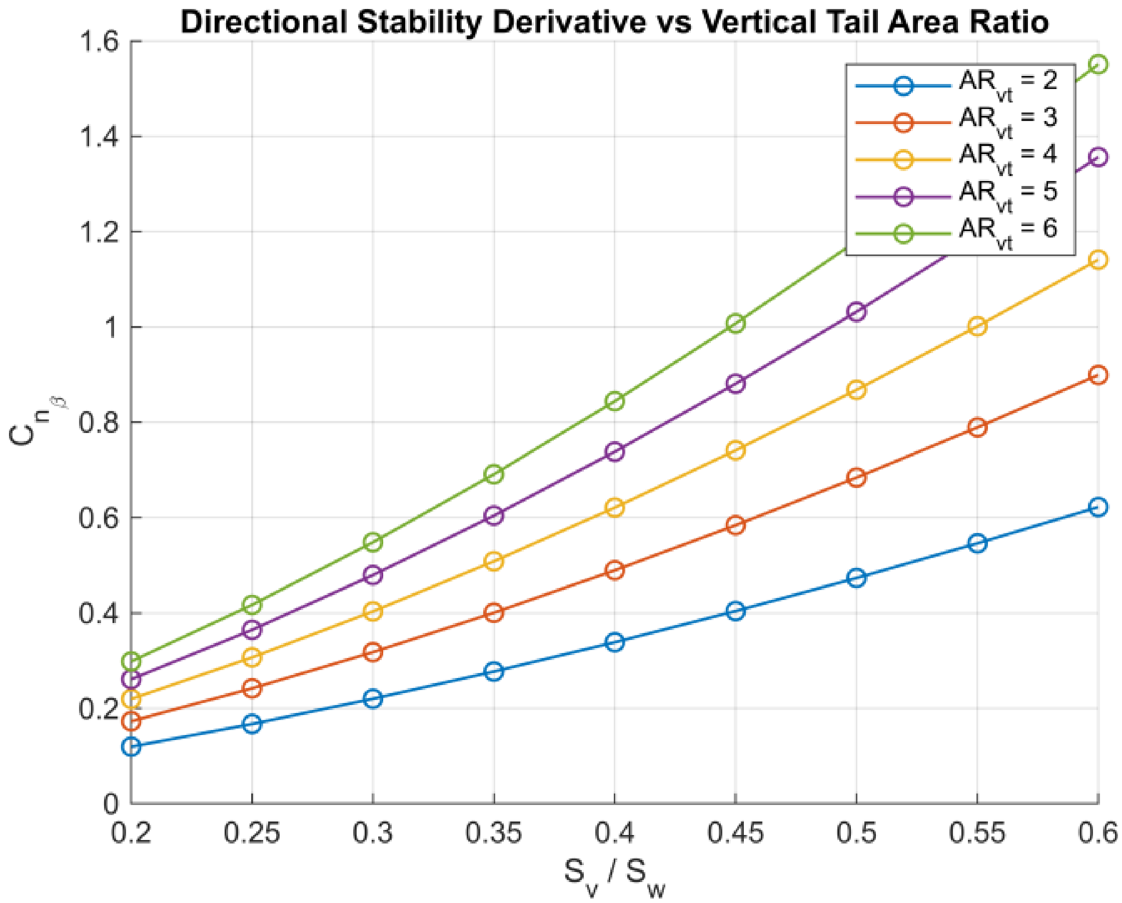


Figure 4.8: C_{n_β} vs Aspect Ratio AR

The final assembled CAD is given in Fig. 4.9

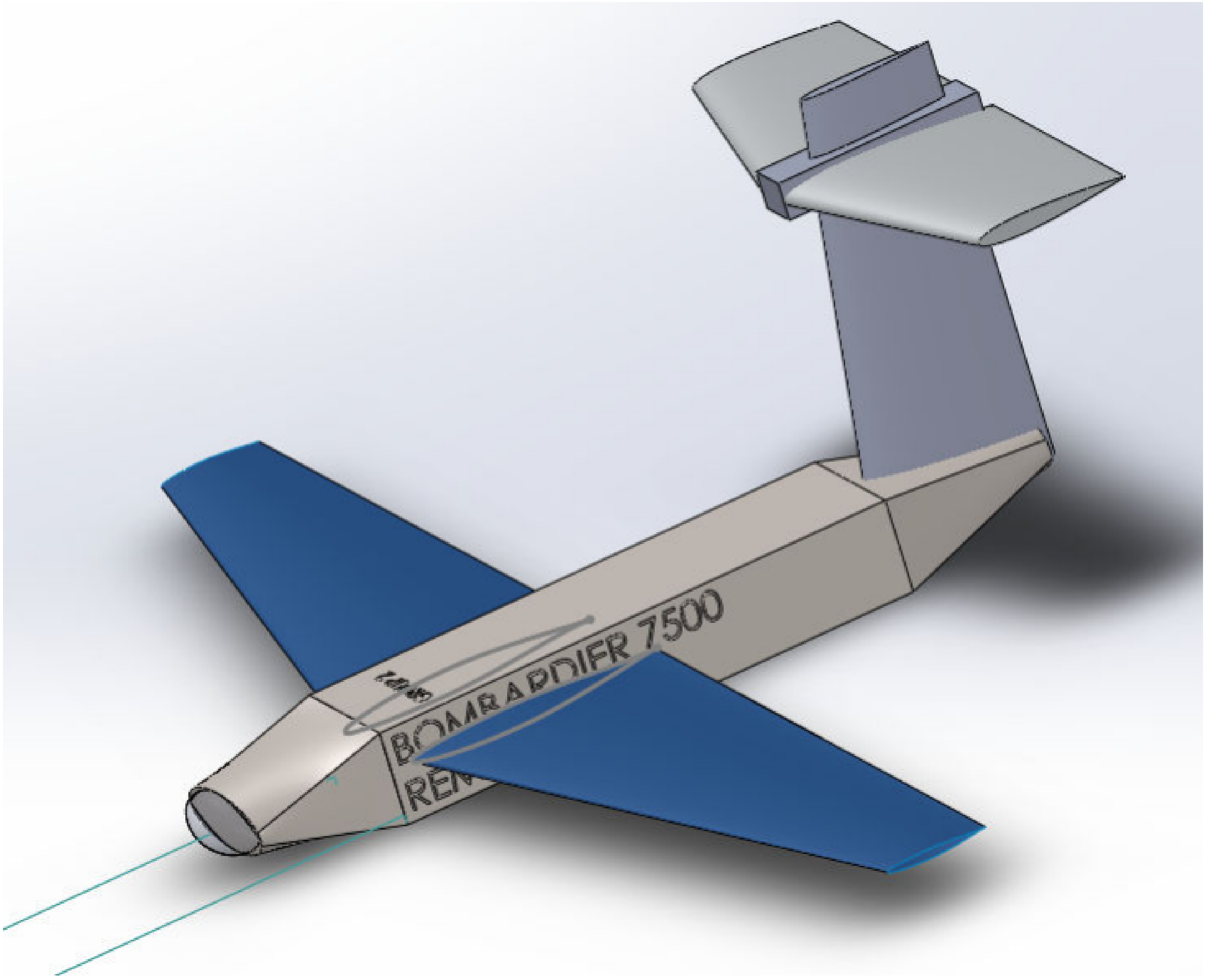


Figure 4.9: Design of the model

(PS: We know VT is kind of big :?)

Chapter 5

Stability and Control

Estimate the aircraft stability derivatives and control surface sizing. Books like [3] and [2] were used to get some insights on mathematical formulas.

5.1 Elevator design

The equations are given as follows:

$$\left(\frac{\partial \delta_e}{\partial C_L}\right)_{\text{trim}} = \frac{\delta_{e_{\max}}}{C_{L_{\max}} - C_{L_d}} \tag{5.1}$$

and

$$C_{m_{\delta_e}} = \frac{\bar{x}_{np} - \bar{x}_{cg}}{\left(\frac{\partial \delta_e}{\partial C_L}\right)_{\text{trim}}} \tag{5.2}$$

To find the relationship between $\frac{S_e}{S_t}$ for different values of δ_e:

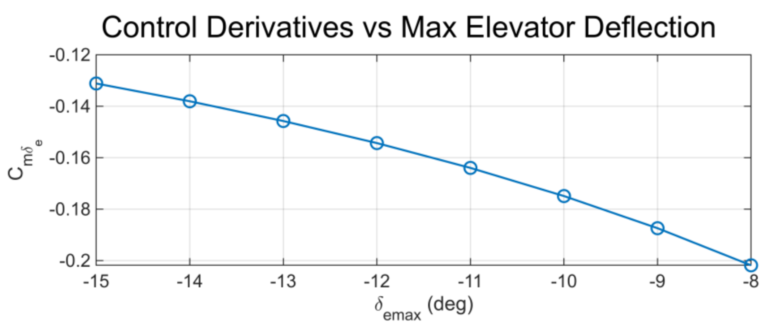
$$\frac{S_e}{S_t} = \text{expression in terms of } \delta_e \tag{5.3}$$

we take max deflection as 10 degrees for our design . so according to that we got our area ratio as 0.19

Given that $S_t = 48.8$, the value of S_e is calculated as:

$$S_e = S_t \times 0.19 = 48.8 \times 0.19 = 9.27m^2 \tag{5.4}$$

We obtained the following plots



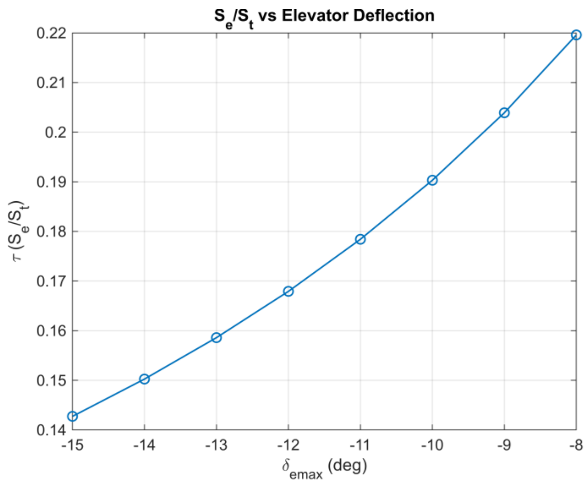
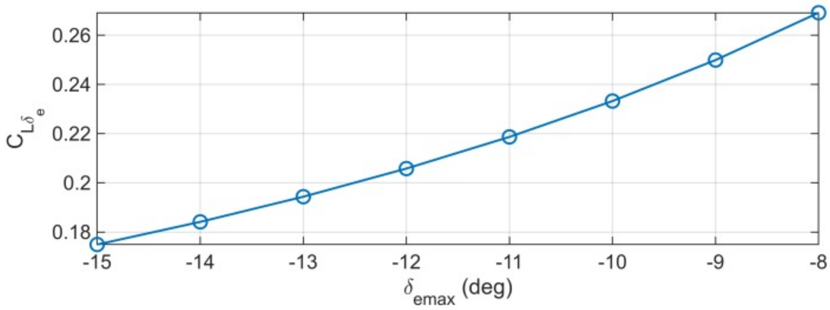
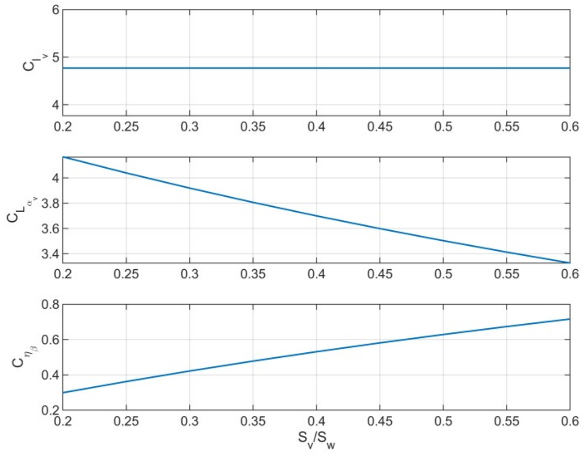


Figure 5.1: control surface area ratio vs elevator deflection

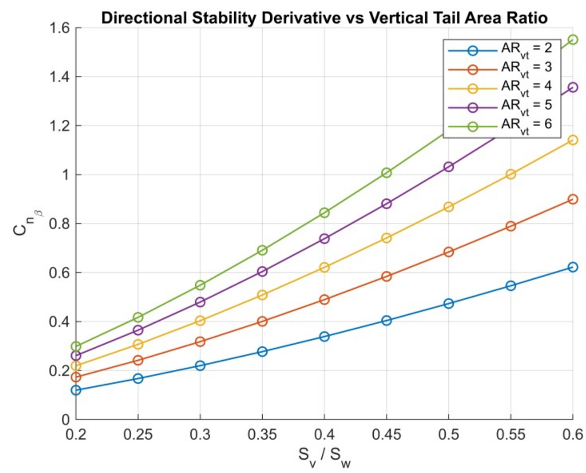


5.2 Vertical Tail

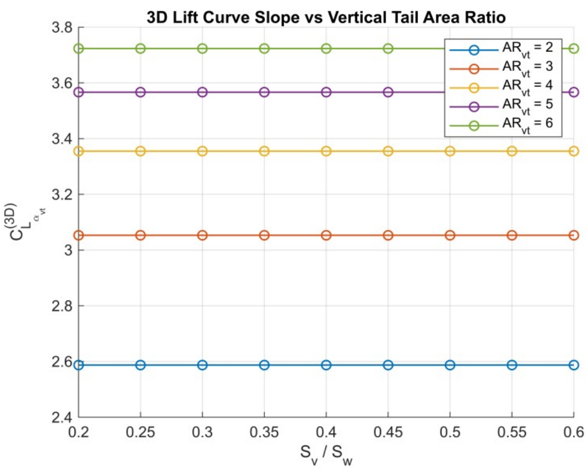
We obtained the following plots



S_v/S_w



Directional Stability Derivative vs Vertical Tail Area Ratio
$AR_{vt} = 2$
$AR_{vt} = 3$
$AR_{vt} = 4$
$AR_{vt} = 5$
$AR_{vt} = 6$
C_{n_β}
S_v / S_w



3D Lift Curve Slope vs Vertical Tail Area Ratio
$AR_{vt} = 2$
$AR_{vt} = 3$
$AR_{vt} = 4$
$AR_{vt} = 5$
$AR_{vt} = 6$
$C^{(3D)}_{L_{\alpha_{vt}}}$
S_v / S_w

Chapter 6

V-n Diagram

We drew the V-n diagram for our flight, considering the velocity limits to decide the structural limits and profile of our flight limits. We used the following code for this.

```
V_stall = 110;
V_max = 250;
V_dive = 1.25 * V_max;
n_p = [4.4, 6.6];
n_n = [-1.8, -2.7];

VAp = sqrt(n_p) * V_stall;
VAn = sqrt(-n_n) * V_stall;

V_range_upper = linspace(0, VAp(2), 100);
V_range_lower = linspace(0, VAn(2), 100);

n_upper_load = (n_p(1) / VAp(1)^2) * V_range_upper.^2;
n_lower_load = (n_n(1) / VAn(1)^2) * V_range_lower.^2;
n_upper_ultimate = (n_p(2) / VAp(2)^2) * V_range_upper.^2;
n_lower_ultimate = (n_n(2) / VAn(2)^2) * V_range_lower.^2;

figure;
hold on;
grid on;

plot(V_range_upper, n_upper_load, 'k-', 'DisplayName', 'Load Limit', 'LineWidth', 1);
plot(V_range_lower, n_lower_load, 'k-', 'HandleVisibility', 'off', 'LineWidth', 1);
plot([VAp(1), V_dive], [n_p(1), n_p(1)], 'k-', 'HandleVisibility', 'off',
'LineWidth', 1);
plot([VAn(1), V_dive], [n_n(1), n_n(1)], 'k-', 'HandleVisibility', 'off',
'LineWidth', 1);

plot(V_range_upper, n_upper_ultimate, 'r-', 'DisplayName', 'Ultimate Limit',
'LineWidth', 1);
plot(V_range_lower, n_lower_ultimate, 'r-', 'HandleVisibility', 'off',
'LineWidth', 1);
```

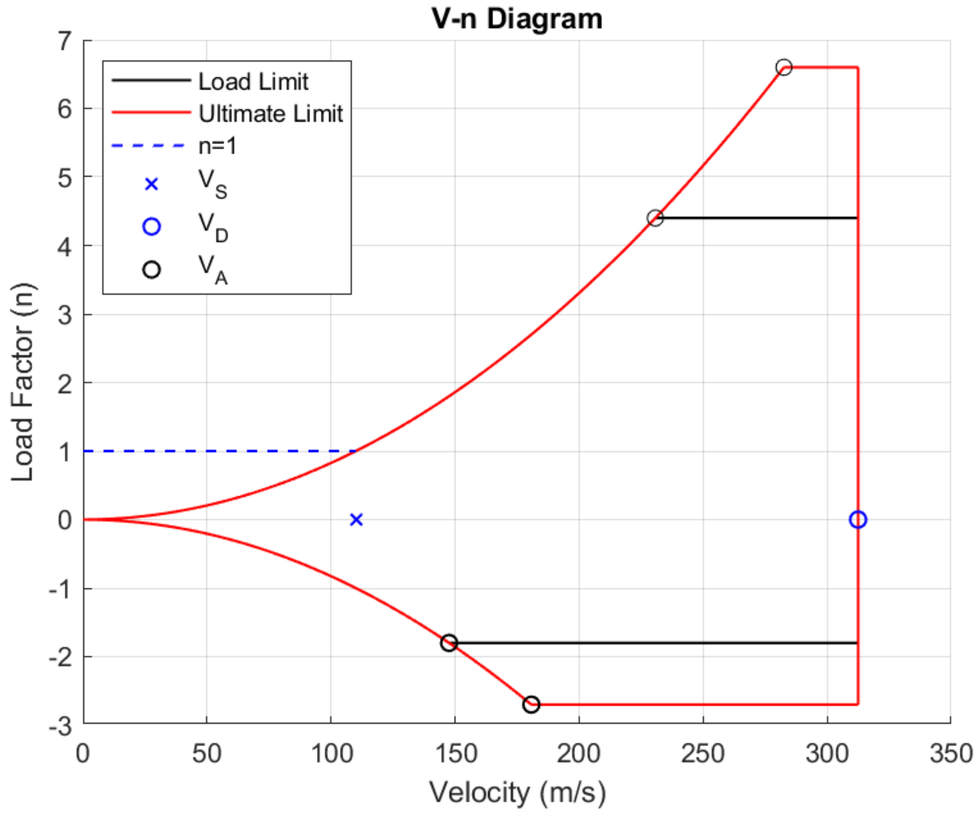


Figure 6.1: V-n Diagram

```
plot([VAp(2), V_dive], [n_p(2), n_p(2)], 'r-', 'HandleVisibility', 'off',
'LineWidth', 1); % Upper horizontal line
plot([VAn(2), V_dive], [n_n(2), n_n(2)], 'r-', 'HandleVisibility', 'off',
'LineWidth', 1); % Lower horizontal line

plot([V_dive, V_dive], [n_n(2), n_p(2)], 'r-', 'HandleVisibility', 'off',
'LineWidth', 1); % Vertical line at V_dive

plot([0, V_stall], [1, 1], 'b--', 'DisplayName', 'n=1', 'LineWidth', 1);

plot(V_stall, 0, 'xb', 'DisplayName', 'V_{S}', 'LineWidth', 1);
plot(V_dive, 0, 'bo', 'DisplayName', 'V_{D}', 'LineWidth', 1);
plot(VAn, n_n, 'ko', 'DisplayName', 'V_{A}', 'LineWidth', 1);
plot(VAp, n_p, 'ko', 'HandleVisibility', 'off');

xlabel('Velocity (m/s)');
ylabel('Load Factor (n)');
title('V-n Diagram');
legend('Location', 'NorthWest');
hold off;
```

The plot was observed as shown in the figure attached.
We obtained the variation of Lift Coefficient with Velocity too:-

```
W = 60000* 9.81;
```

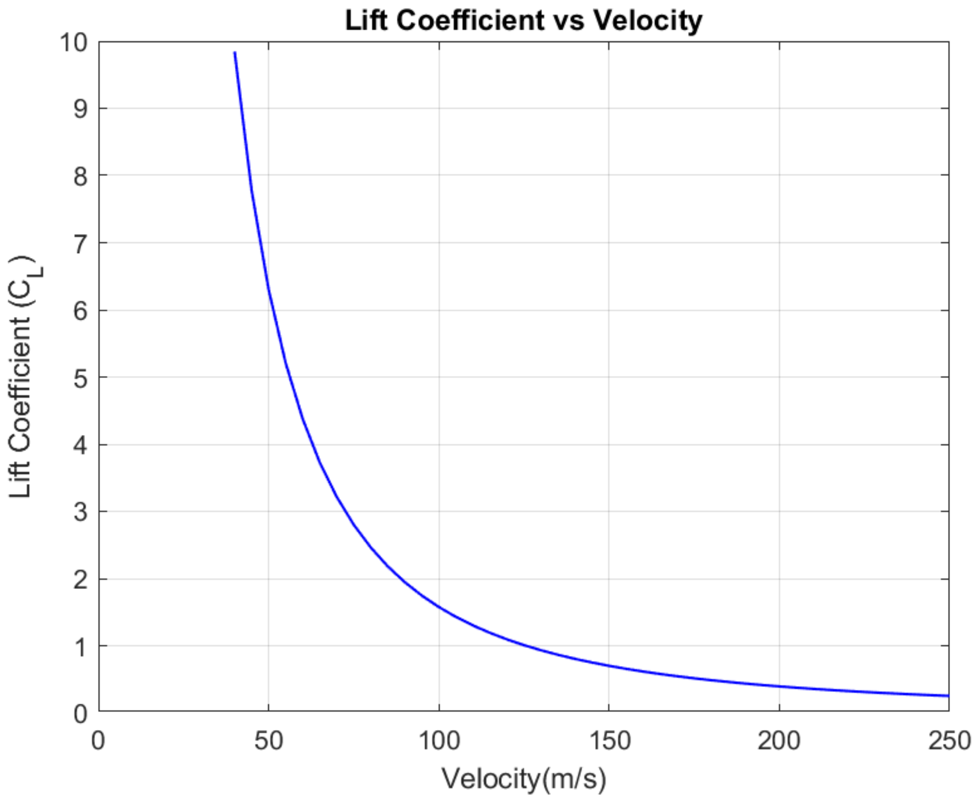


Figure 6.2: Lift Coefficient vs Velocity

```
rho = 1.225;
S = 122;
n = 2;
V_range =40:5:250;
C_L = (2* n * W) ./ (rho * S * V_range.^2);
figure;
plot(V_range,C_L, 'b-', 'LineWidth',1);
xlabel('Velocity(m/s)');
ylabel('Lift Coefficient (C_L)');
title('Lift Coefficient vs Velocity');
grid on;
```

Chapter 7

Conclusion

Concluding this work, the final values taken for the design are given below

Table 7.1: Geometrical parameters

S.no	Parameter	Value
1.	Weight (W_{TO})	60000 Kg
2.	Span (b)	31 m
3.	Tip chord (C_t)	2.2 m
4.	Taper ratio (λ)	0.4
5.	Wing planform area (S_w)	120 m^2
6.	Aspect ratio of wing (AR_w)	8
7.	Horizontal tail area (S_t)	48.8 m^2
8.	Moment of Inertia matrix I	We did not have proper mass distribution of engine, and wing
9.	x_{NP} w.r.t Wing root chord	1.82
10.	x_{CG} w.r.t Wing root chord	1.42
11.	x_{ac_w} w.r.t Wing root chord	1.02
12.	x_{act} w.r.t Wing root chord	0.128

Table 7.2: Aerodynamic parameters

Longitudinal parameters	Analytical estimation values
C_{D_0}	0.0249
k	0.0295
C_{L_0}	0.15
C_{L_α}	0.1124 (degrees)
$C_{L_{\delta_e}}$	0.23
C_{m_0}	0.0281
C_{m_α}	-0.01124
$C_{m_{\delta_e}}$	-.176

Bibliography

[1] Omran Al-Shamma and Rashid Ali. Aircraft weight estimation in interactive design process. 04 2014.

[2] R.C. Nelson. *Flight Stability and Automatic Control*. McGraw-Hill aerospace science & technology series. McGraw-Hill Education, 1998.

[3] B.N. Pamadi. *Performance, Stability, Dynamics, and Control of Airplanes*. AIAA Education Series. American Institute of Aeronautics & Astronautics, 2004.