AE461A

AIRCRAFT DESIGN

Final Report

Instructor: Prof. Mangal Kothari

Submitted by:

Name: Aditya Prakash
Roll Number: 190065
Email: adityaap@iitk.ac.in
Department of Aerospace Engineering
Indian Institute of Technology Kanpur

Other Team Members

Azhar Tanweer
Bidhan Arya
Rahul Godara

Mission Objective

The mission objectives of the aircraft design are following:

- Range of the Aircraft (R): **94.5 km**
- Endurance (E): **20 minutes**
- Total Payload (W_p): **665 kg**
- Velocity of cruise (V): **69 ms^{-1} (250 km/hr)**
- Height of cruise: **10 km**
- Density at height of cruise (ρ): **0.413509 kgm^{-1}**
- Dynamic Pressure: $q = \frac{1}{2}\rho V^2 =$ **984.358 Pa**

Power Plant Selection

- Engine Used: **RED A03**
- Power Output: **338 kW (460 HP)** Maximum continuous at **1995** Propeller RPM
- Specific fuel consumption (C_{bhp}): **210 g/KWh**
- Propulsive Efficiency (η_p): **0.813**

Mass Estimation

We know that,

$$W_t = \frac{W_p}{1 - \dfrac{W_e}{W_t} - \dfrac{W_f}{W_t}}$$

From Mission objective, $W_p = 665$ kg

Calculation of W_f/W_t

We take, $C_L/C_D = 17$ (ref. Raymer), Cruise Velocity (V) = 250 km/hr

For propeller, specific fuel consumption is given by,

$$c = C_{\text{bhp}} \frac{V}{550\eta_p} = 0.53/hr = 0.0001472/s$$

For cruise: $\dfrac{W_2}{W_1} = \exp -\dfrac{Rc}{V(L/D)}$, this gives $\dfrac{W_2}{W_1} = 0.9883$

For loitering: $\dfrac{W_3}{W_2} = \exp -\dfrac{Ec}{L/D}$, this gives $\dfrac{W_3}{W_2} = 0.9904$

Therefore, $\dfrac{W_3}{W_1} = 0.9788$, this gives, $\dfrac{W_f}{W_t} = 1 - \dfrac{W_3}{W_1} = 0.0212$

Calculation of W_e/W_t

For general aviation aircraft, $\dfrac{W_e}{W_t} = 2.36W_t^{-0.18}$ [ref. Raymer]

Calculation of W_t

Using above results, we have

$$W_t = \frac{W_p}{1 - 2.36W_t^{-0.18} - 0.0212}$$

Using, Newton Raphson Method, we get W_t = **1811 kg**

```
R = 94.5 * 1000
W_p = 665
C_c = 0.53 / 3600
V_c = 250 * 1000/3600
L_D = 17
W2_W1 = exp(-R*C_c/(V_c*L_D))

V_l = 210 * 1000/3600
C_l = 0.49/3600
E = 20 * 60
W3_W2 = exp(-E*C_l/L_D)
W3_W1 = W3_W2 * W2_W1
Wf_Wt = 1 - W3_W1
```

```
A = 2.36
C = -0.18
Wt = 1500;
Wt_old = 20000;
iter = 0
while abs(Wt_old -  Wt) > 10^-2
    Wt_old = Wt;
    temp = 1 - A*(Wt^C) - Wf_Wt
    f = (Wt - W_p/temp)
    f_ = 1 - (A*W_p*C * Wt^(C-1))/temp^2
    Wt = Wt - f/f_
    iter = iter + 1
    fprintf("Iteration: %d", iter)
end
disp(Wt)
```

Wing Design

C_L calculation

Assuming, the wing loading of 90 kgm^{-2} (ref Raymer)

Then, wing loading is $\frac{W}{S}$ = 90 gives, S = 20.12 m^2

Using, $L = C_L q S = W$, we get $C_L = 0.89$ which is not in ideal range (1 - 1.5)

Increasing wing loading to 100 kgm^{-2}, we get S = 18.11 m^2, this gives $C_L = 0.99658$

Increasing wing loading to 115 kgm^{-2}, we get S = 15.75 m^2, this gives $C_L = 1.146$ which is within the ideal range.

Therefore, **wing loading = 115 kgm^{-2}, S = 15.85 m^2, C_L = 1.146**

Max C_L calculation

Take Stall velocity (V_s) = 55 ms^{-1} = 198 km/hr

This gives, $C_{L_{\max}} = \frac{(W/S)g}{\frac{1}{2}\rho V_s^2} = 1.8$

Wing Geometry

Aspect and Taper Ratio

We choose the wing configuration to be mid-wing configuration.

Surface Area of the wing = S = 15.85 m^2

From literature survey, Aspect Ratio = AR = 7

There is no sweep in the wing. To make the wing geometry close to elliptical wing, we take Taper ratio = λ = 0.45

Using,

$$b * (\lambda * c + c)/2 = S$$

$$\frac{2 * b}{\lambda * c + c} = AR$$

we get, **b (span of the wing) = 10.5432 m** and **c (chord length at root) = 2.07748 m** and **chord length at tip = $\lambda * c$ = 0.9349** and **mean chord length = 1.51 m**

Dihedral Angle

Table 4.2 Dihedral guidelines

	Wing position		
	Low	Mid	High
Unswept (civil)	5 to 7	2 to 4	0 to 2
Subsonic swept wing	3 to 7	−2 to 2	−5 to −2
Supersonic swept wing	0 to 5	−5 to 0	−5 to 0

Figure 1: Dihedral Guidelines

Since we have chosen mid-wing configuration, based on fig. 1, we choose a dihedral angle of 3 deg.

Twist

From Raymer, the twist angle of 3 deg is used to provide adequate stall characteristics. Hence, we use a twist angle of 3 deg.

Airfoil Selection

we have, AR = 7

Using,

$$e = 1.78(1 - 0.045\mathrm{AR}^{0.68}) - 0.64$$

we get, e (Oswald Efficiency) = 0.84

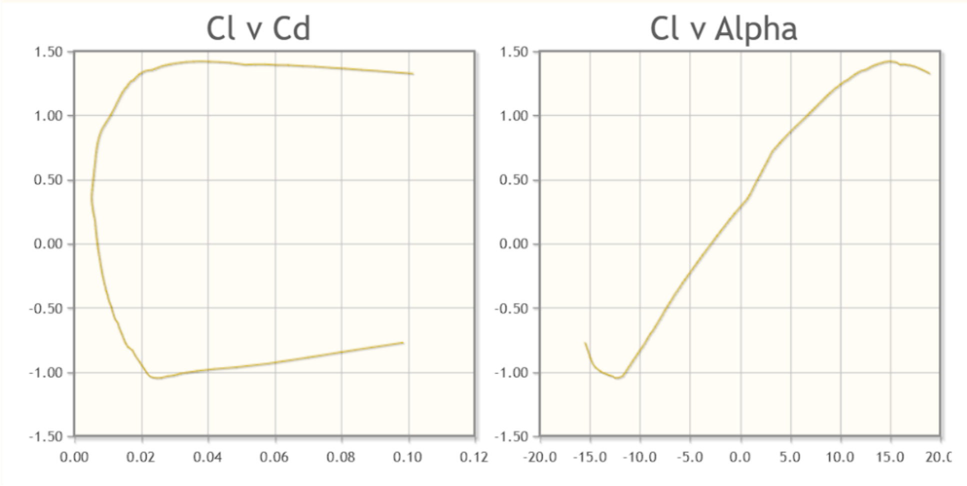
We have, $C_L = 1.146$. Using the formula,

$$C_L = \frac{C_l}{1 + \frac{C_l - C_{l_0}}{\pi(AR)e\alpha}}$$

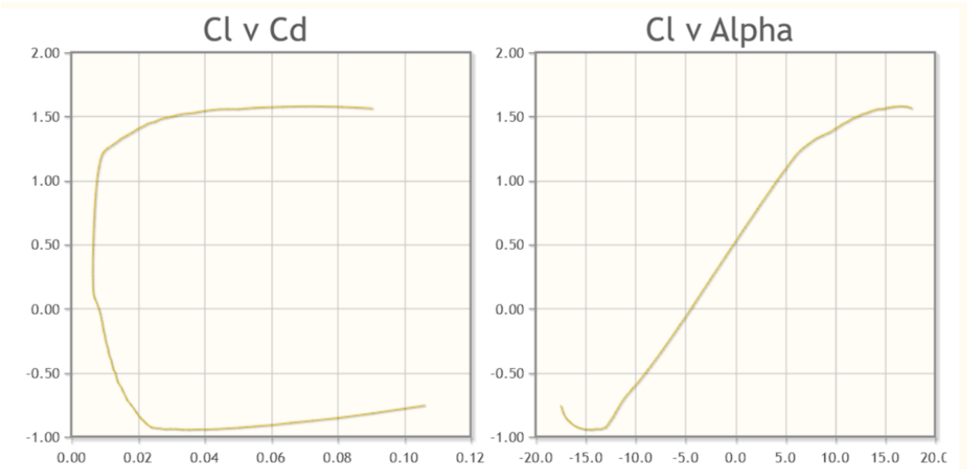
we get, $C_l = 1.1602$ when $\alpha = 2$ deg.

Based on this C_l, we choose 5 airfoils which have this range of C_l within given α range:

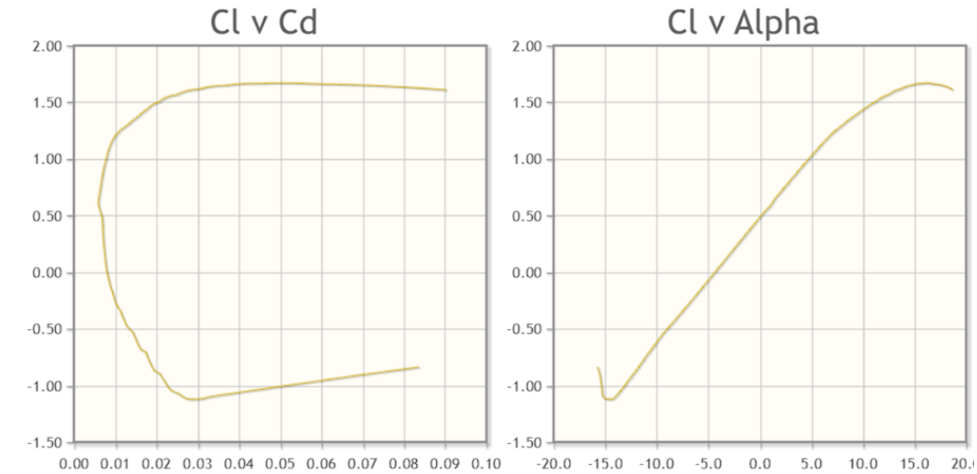
NACA2411



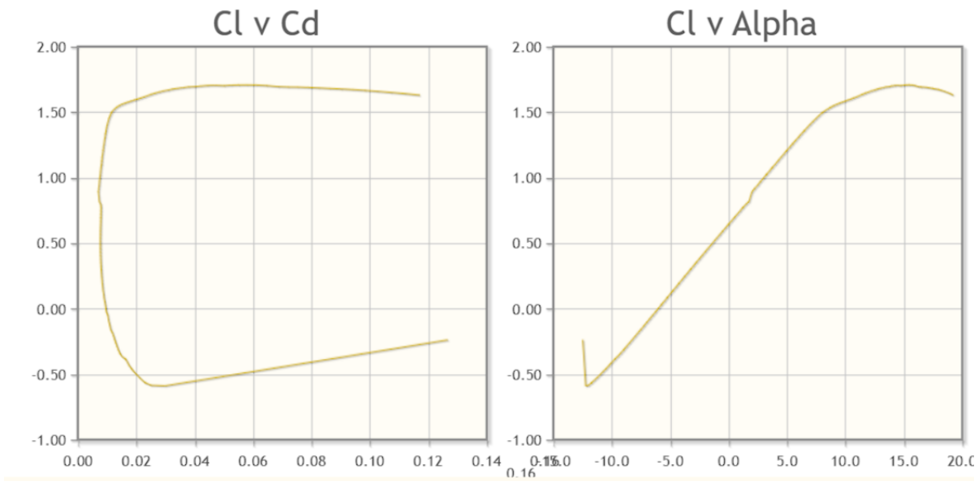
NACA63(2)-615



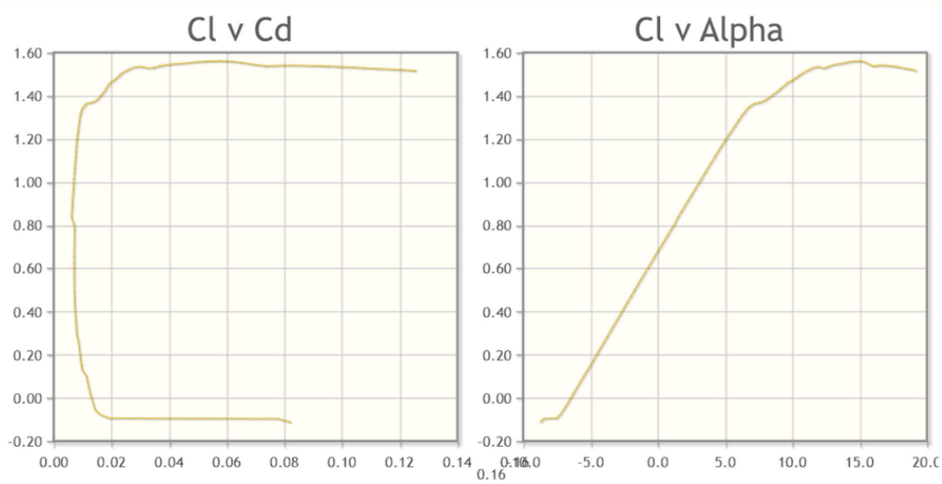
NACA4412



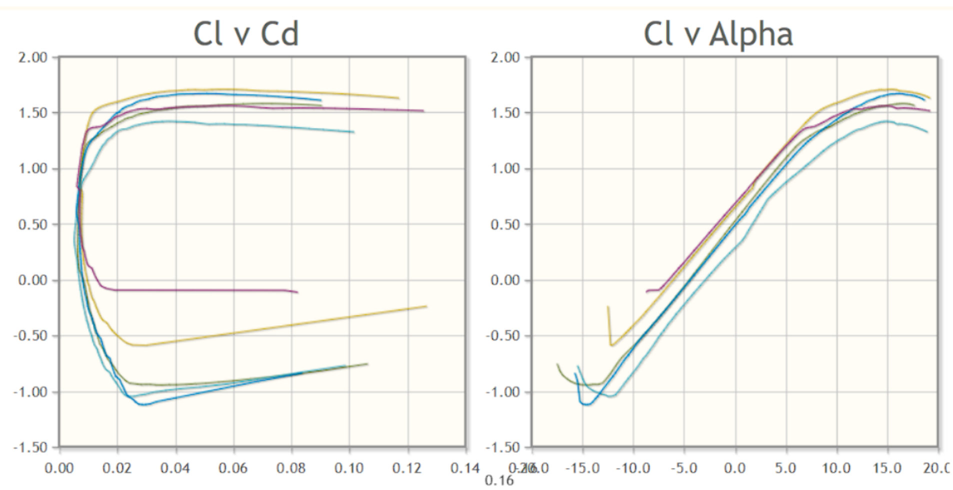
NACA6412

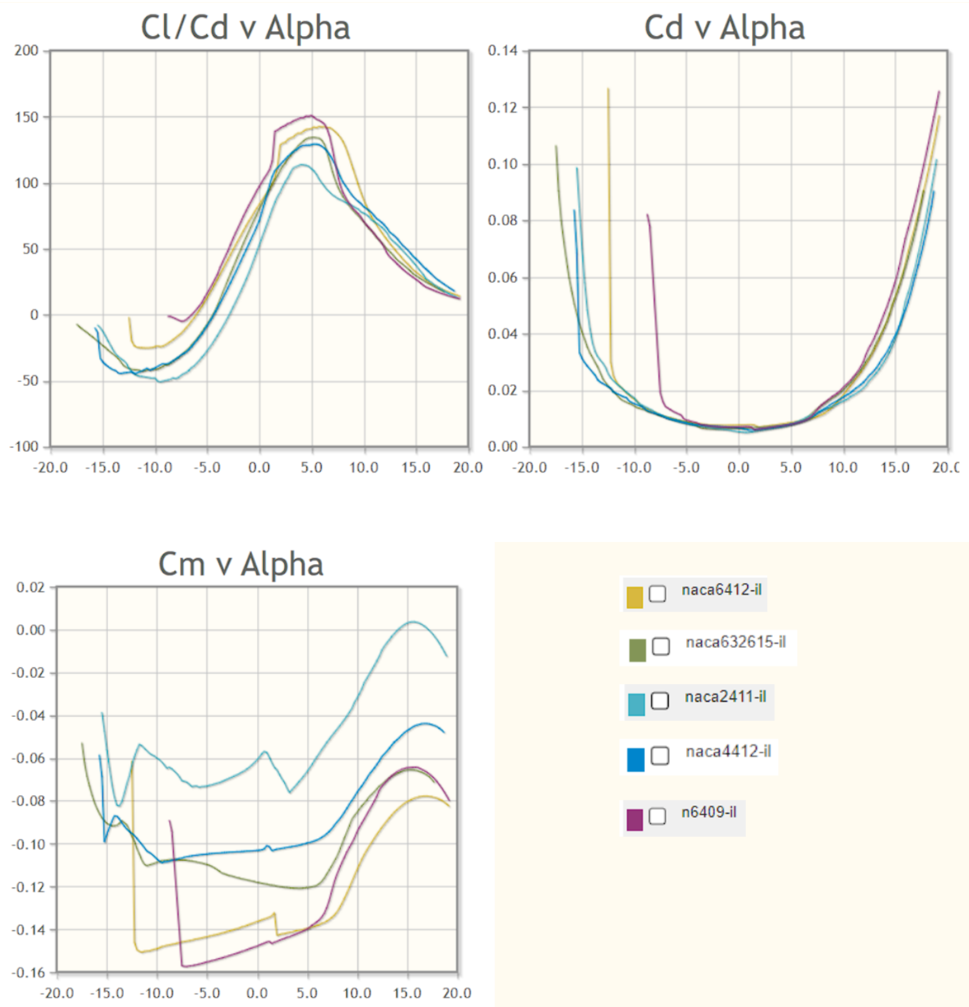


NACA6409



Comparison





Final Airfoil Selection

Based on the aircraft design, the final airfoil chosen is: **NACA6412**.

For $C_l = 1.1602$, we have $\alpha = 4.3$ deg. The corresponding $C_{d_0} = 0.0083$. Using,

$$C_D = C_{d_0} + \frac{C_L^2}{\pi(AR)e} = 0.07264$$

Therefore, $C_L/C_D = 15.776$

Tail Design

Next, we move on to the tail design. A plane's tail is made up of horizontal and vertical tails, each of which has its own control surfaces called the elevator and the rudder. Trim, stability, and control are all provided by the tail. Trim is the process of creating a lift force that balances another moment the aircraft produces by operating through a tail moment arm about its centre of gravity. Trim mostly entails balancing the moment produced by the wing for the horizontal tail. To offset the wing pitching moment, an aft horizontal tail normally has a negative incidence angle of roughly 2-3 degrees. The horizontal tail incidence can often be adjusted via a range of around 3 degrees up and down since the wing pitching moment varies depending on the flight conditions.

Choosing a tail arrangement

There are several possible tail arrangements. Some of them are given below.

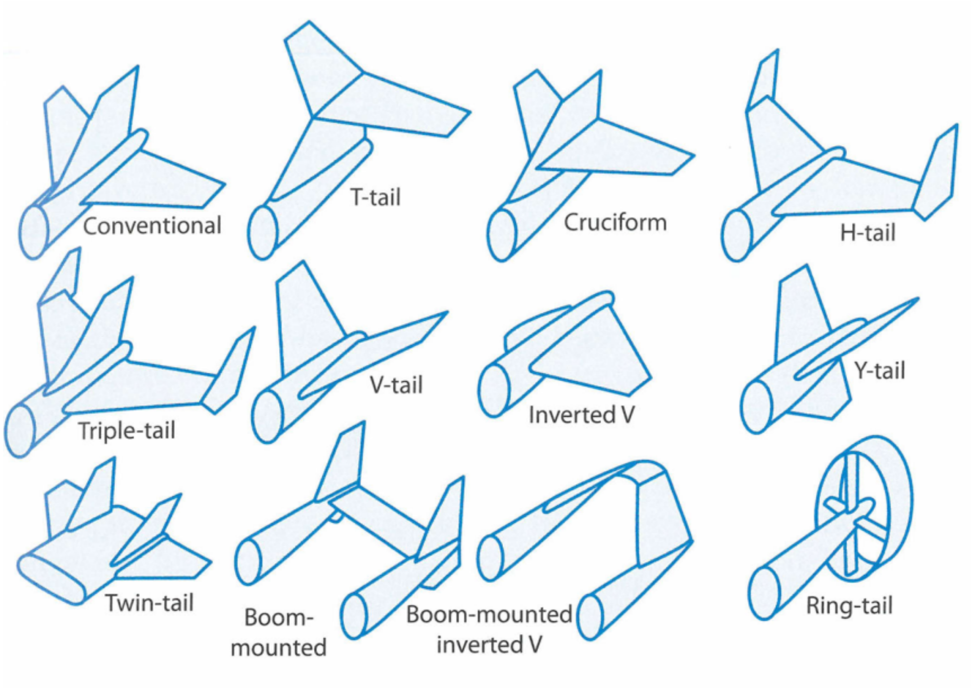


Figure 2: Possible aft-tail arrangements (source : Raymer)

For our needs, we opt for the conventional tail layout because it is effective and it works. A large percentage of aeroplanes in service—perhaps 70% or more—have this tail configuration. The conventional tail will offer sufficient stability and control at the lightest weight for the majority of aircraft designs. This configuration places the horizontal surface where it typically experiences smooth airflow, fastens it to the fuselage where there is typically sufficient structure, and makes it simple to mechanise control linkages.

Tail Positioning

Now, we have to position the tail. The location of an aft horizontal tail with respect to the wing is critical to the stall characteristics of the aircraft. If the tail enters the wing wake during the stall, control will be lost, and pitch-up can be encountered. Figure 3 shows the acceptable boundaries for the positioning of horizontal tail to avoid this problem.

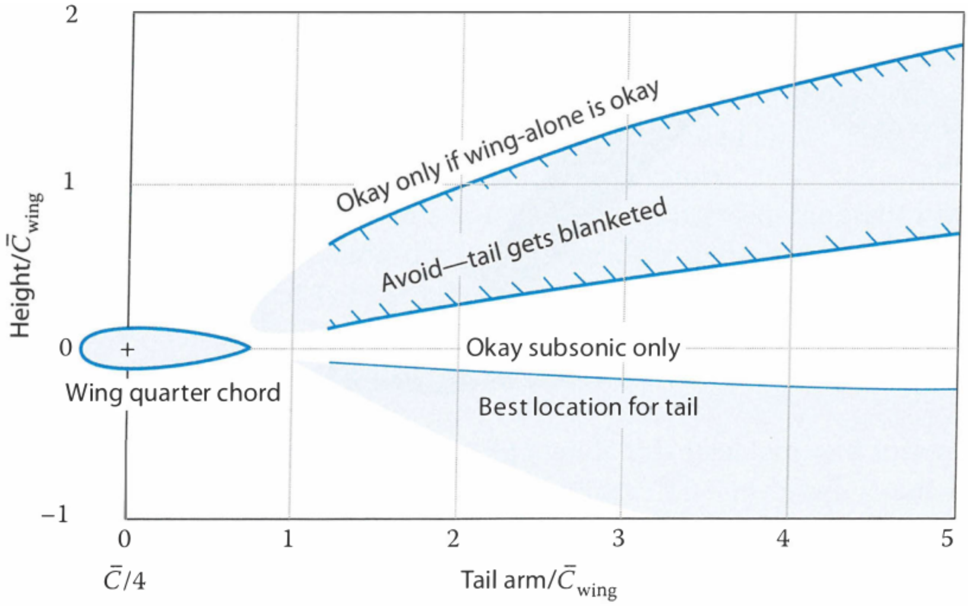


Figure 3: Aft-tail positioning (source : Raymer)

The Area of the wing (S_w) we have from previous calculations is 15.85 m^2. The Aspect Ratio AR is 7. The mean chord (c_w) and span (b_w) are calculated as:

$$AR = \frac{b_w^2}{S_w} = \frac{b_w^2}{15.85m^2} = 7$$

This gives

$$b_w = 10.533\ m$$

$$c_w = 1.51\ m$$

Refering to the graph in figure 3, we choose

$$\frac{\text{Tail Arm}}{c_w} = 5$$

$$\frac{\text{Height}}{c_w} = -0.5$$

So we have

$$\text{Tail Arm} = l_t = 7.525\ m$$
$$\text{Height} = h_t = -0.7525\ m$$

Note that l_t is measured from $c_w/4$ to $c_t/4$.

Horizontal Tail Area and Geometry

We use tail volume coefficient method for determining the horizontal tail area S_{HT}. The Horizontal tail volume coefficient is defined as

$$C_{HT} = \frac{L_{HT} S_{HT}}{c_w S_w}$$

Hence

$$S_{HT} = \frac{c_w S_w C_{HT}}{L_{HT}}$$

	Typical Values	
	Horizontal c_{HT}	Vertical c_{VT}
Sailplane	0.50	0.02
Homebuilt	0.50	0.04
General aviation—single engine	0.70	0.04
General aviation—twin engine	0.80	0.07
Agricultural	0.50	0.04
Twin turboprop	0.90	0.08
Flying boat	0.70	0.06
Jet trainer	0.70	0.06
Jet fighter	0.40	0.07–0.12*
Military cargo/bomber	1.00	0.08
Jet transport	1.00	0.09

Figure 4: Typical values for Tail Volume Coefficient (Source: Raymer)

We use the value 0.70 for the horizontal tail volume coefficient referring to the table in figure 4. Therefore,

$$S_{HT} = \frac{c_w S_w C_{HT}}{L_{HT}} = \frac{c_w S_w C_{HT}}{l_t} = \frac{1.51 \cdot 15.85 \cdot 0.70}{7.525} = 2.22\ m^2$$

Now that we have the tail area, we can determine the geometry. Tail aspect ratio and taper ratio show little variation over a wide range of aircraft types. We choose a rectangular tail, similar to the wing. For the aspect ratio, we refer to the table in figure 5 taken from Raymer. It provides a guide for selection of tail taper ratio and aspect ratio. We take AR = 6.

	Horizontal Tail		Vertical Tail	
	A	λ	A	λ
Fighter	3–4	0.2–0.4	0.6–1.4	0.2–0.4
Sailplane	6–10	0.3–0.5	1.5–2.0	0.4–0.6
Others	3–5	0.3–0.6	1.3–2.0	0.3–0.6
T-tail	–	–	0.7–1.2	0.6–1.0

Figure 5: Typical values for Tail Aspect Ratio and Taper Ratio (Source: Raymer)

$$AR = \frac{S_w}{c_w^2} = \frac{2.22m^2}{c_w^2} = 6$$

Hence chord c_t and span b_t are

$$c_t = 0.61\ m$$
$$b_t = 3.65\ m$$

Vertical Tail Area and Geometry

We use tail volume coefficient method for determining the vertical tail area S_{VT}. The vertical tail volume coefficient is defined as

$$C_{VT} = \frac{L_{VT} S_{VT}}{b_w S_w}$$

Hence

$$S_{VT} = \frac{b_w S_w C_{VT}}{L_{VT}}$$

We use the value 0.05 for the vertical tail volume coefficient referring to the table in figure 4. Therefore,

$$S_{VT} = \frac{b_w S_w C_{VT}}{L_{VT}} = \frac{b_w S_w C_{VT}}{l_t + 0.2} = \frac{10.533 \cdot 15.85 \cdot 0.05}{7.725} = 1.08\ m^2$$

Now that we have the tail area, we can determine the geometry. For the aspect ratio, we refer to the table in figure 5 once again. We take aspect ratio $AR_{VT} = 1.5$ and taper ratio $\lambda_{VT} = 0.5$. Therefore, we have

$$AR_{VT} = \frac{b_{VT}^2}{S_{VT}} = \frac{b_{VT}^2}{1.08m^2} = 1.5$$

Hence,

$$b_{VT} = 1.273\ m$$

$$c_{\text{VT,root}} = \frac{2S_{VT}}{b_{VT}(1 + \lambda_{VT})} = 0.566\ m$$
$$c_{\text{VT,tip}} = \lambda_{VT} \cdot c_{\text{root}} = 0.283\ m$$

Airfoil selection

We select the symmetric airfoil NACA0012 for the tail.

Stability considerations

We know that

$$C_{m_{0t}} = \eta C_{HT} C_{L_{\alpha t}} (\epsilon_0 + i_w - i_t)$$

we estimate ϵ_0 as

$$\epsilon_0 = \frac{2C_{L_0}}{\pi AR_w} = \frac{2 \cdot 0.7}{\pi \cdot 7} = 0.064rad$$

$$\epsilon_0 = 3.64$$

typical values for $\eta = 1$, $C_{m_{0t}} = 0.4$ and $C_{L_{\alpha t}} = 0.08/deg$. Taking $i_w = 2$, we have

$$\boxed{i_t = -1.5}$$

We also know that for vertical tail's contribution to yaw stability can be seen by

$$C_{n_\beta} = \eta_v C_{VT} C_{L_\alpha}\left(1 + \frac{d\sigma}{d\beta}\right)$$

where $\eta_v \left(1 + \frac{d\sigma}{d\beta}\right)$ can be estimated as

$$\eta_v \left(1 + \frac{d\sigma}{d\beta}\right) = 0.724 + 3.06\frac{S_{VT}/S_w}{1 + \Lambda_{c/4,w}} + 0.4\frac{z_w}{d} + 0.009AR_w$$

where $\Lambda_{c/4,w}$ = sweep of the wing quarter chord, z_w = distance from wing root quarter chord point to fuselage centre-line measured parallel to z axis and d = maximum fuselage depth.

Taking $C_{L_\alpha} = 0.23$ and estimating $\eta_v \left(1 + \frac{d\sigma}{d\beta}\right) = 10$ gives us

$$C_{n_\beta} = 0.115$$

For the Horizontal tail stability derivative $C_{m_{\delta_e}}$ is important. This can be estimated using

$$C_{m_{\delta_e}} = -C_{HT}\eta C_{L_\alpha}\tau$$

where τ is flap effectiveness parameter which can be determined from the graph in figure 6.

We take Control surface area/ lifting surface area = 0.25 giving us $\tau \approx 0.5$

Hence,

$$C_{m_{\delta_e}} = -0.0805$$

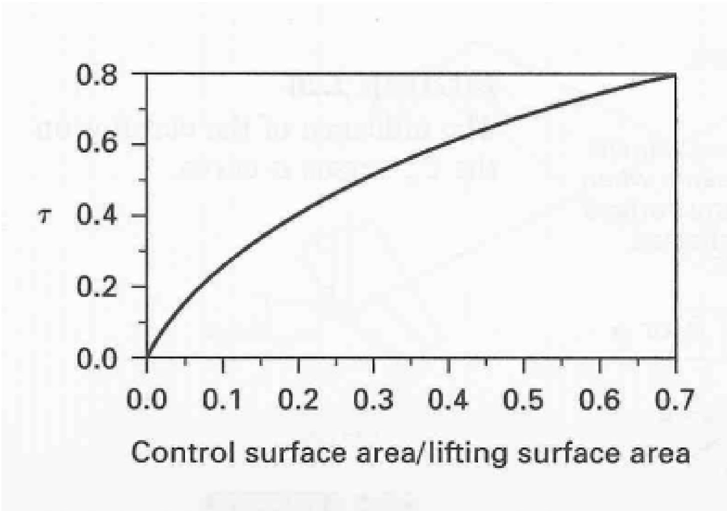


Figure 6: Flap effectiveness parameter (source: Robert Nelson. Flight Stability and Control)

V_n diagram and Gust V_n diagram

V_n diagram

$C_{L_{\max}} = 1.7$

$C_{L_{\min}} = -.5$

Max Load Factor ($n_{\max}$) $= \frac{C_{L_{\max}} \rho S V^2}{2W} = 0.0031\ V^2$

Min Load Factor ($n_{\min}$) $= \frac{C_{L_{\min}} \rho S V^2}{2W} = -0.000875\ V^2$

n_1 Structural Limit Load
- Civil transport ≈ 2.5
- Military attack ≈ 8.0

n_3 Structural Limit Load (negative)
- Civil transport ≈ -1.0
- Military attack ≈ -3.0

Figure 7: Limit Load Data

Using, the data given in fig. 9, for the considered aircraft $n_1 = 2.5$ and $n_3 =$ -1.0

Then,

$$V_A = \sqrt{\frac{2n_1(W/S)}{\rho C_{L_{\max}}}} = 28.72 \text{ m/s}$$

Cruising Speed $(V_C) = 69$ m/s and Diving Speed $(V_D) = 80$ m/s

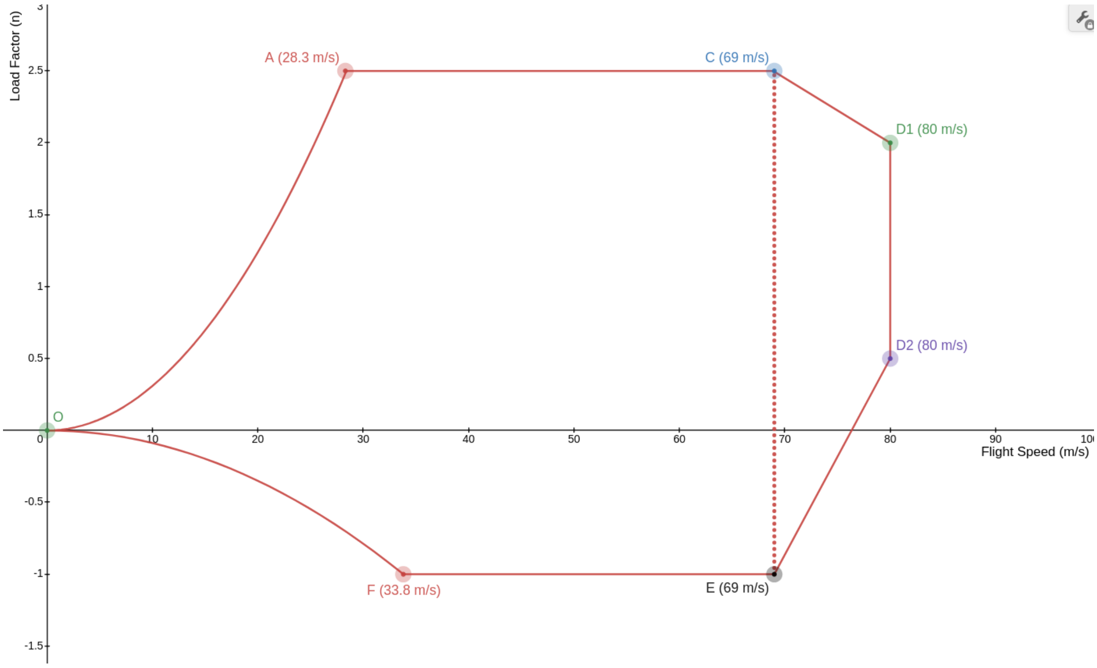


Figure 8: $V - n$ diagram

Gust V_n diagram

$$n_g = 1 \pm \frac{U * k * V}{9.81(W/S)}$$

$$k = \frac{C_{L_{\max}} - C_{L_{\min}}}{\alpha_{\max} - \alpha_{\min}} = 5.48/\text{rad}$$

Using typical gust velocity values, $U_1 = 20$ m/s, $U_2 = 15.25$ m/s and $U_3 = 7.5$ m/s

$$n_{g1} = 1 \pm 0.097V$$

$$n_{g2} = 1 \pm 0.074V$$

$$n_{g3} = 1 \pm 0.036V$$

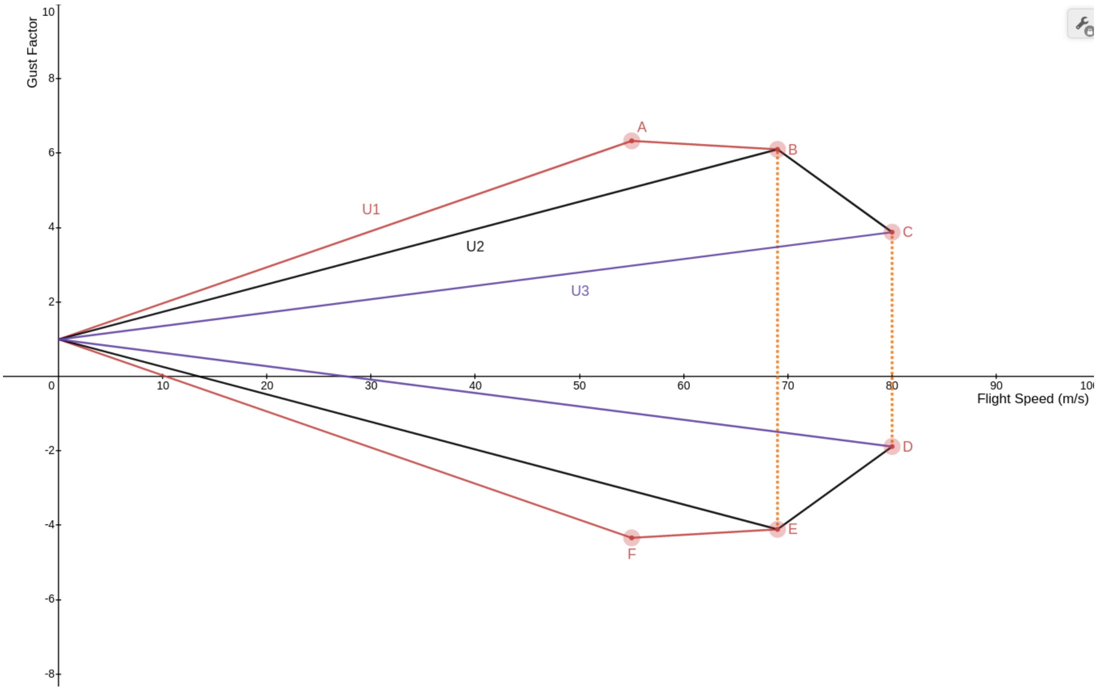


Figure 9: Gust Diagram

Airplane Model

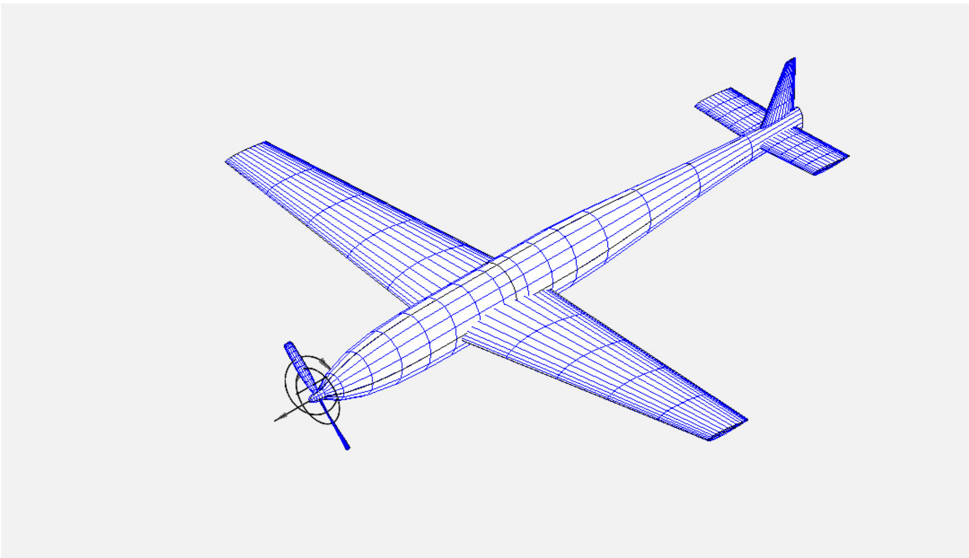


Figure 10: Isometric View

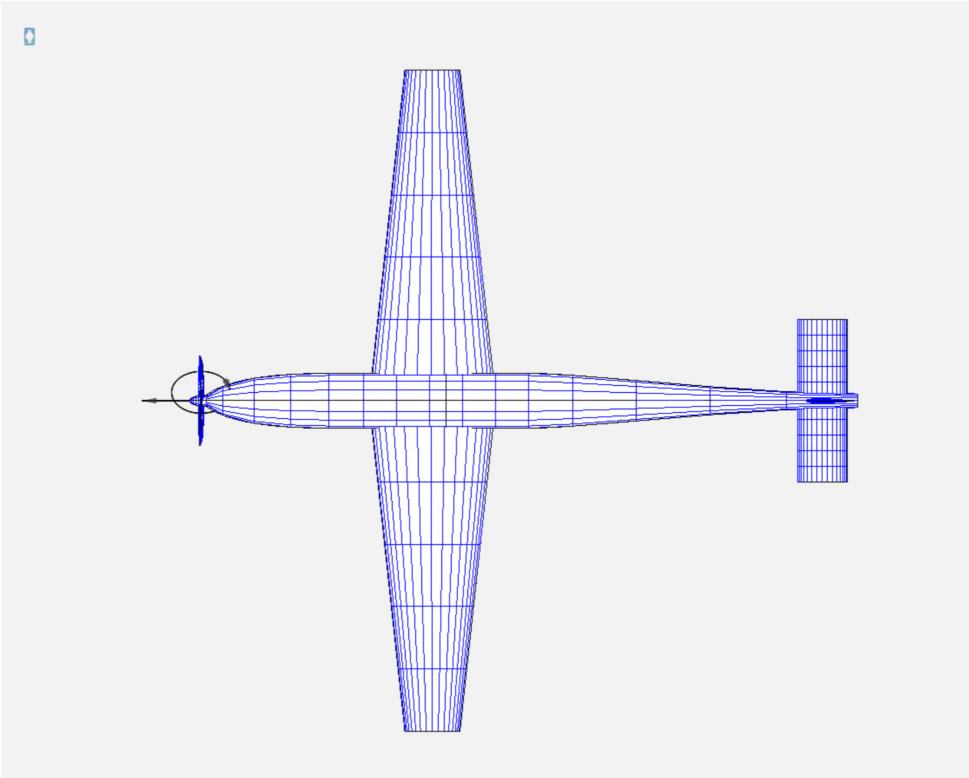


Figure 11: Top View

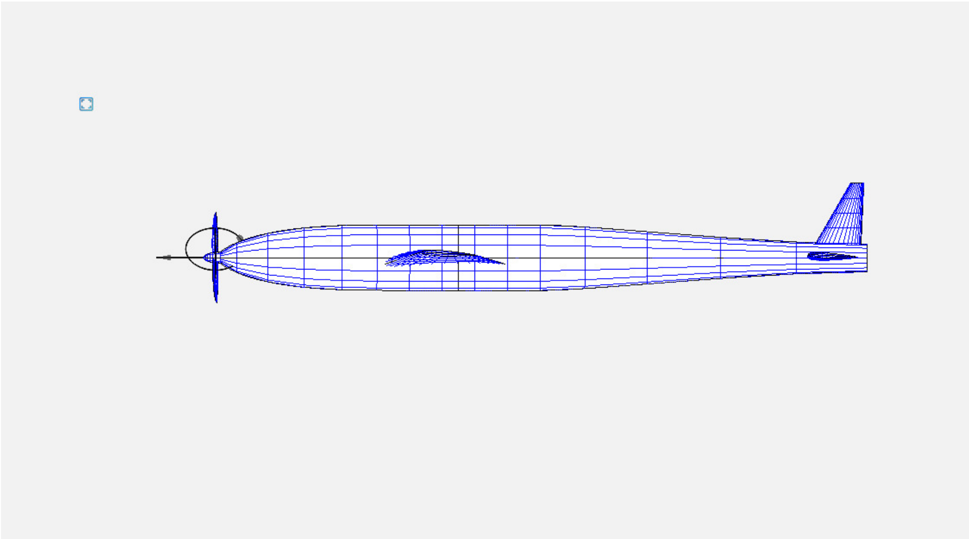


Figure 12: Side View

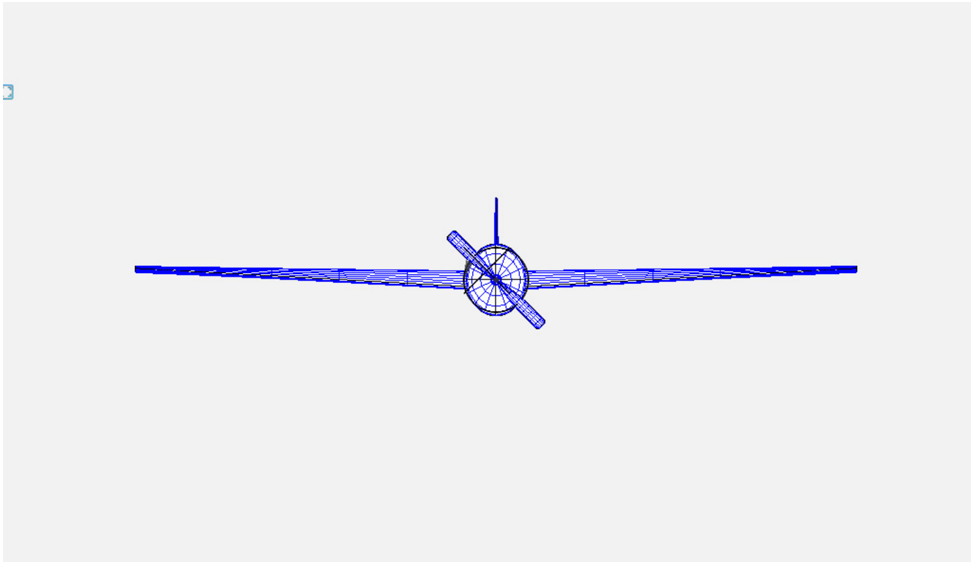


Figure 13: Front View

References

Aircraft Design, Raymer