# *AE461 presentation*

**Members :**
Ashish Shakyawar (170165)
Krishna Roy( 170349)
Naveen Balaji (170420)
Sreenivas Rapole (170721)
Vasu Bansal (160776)

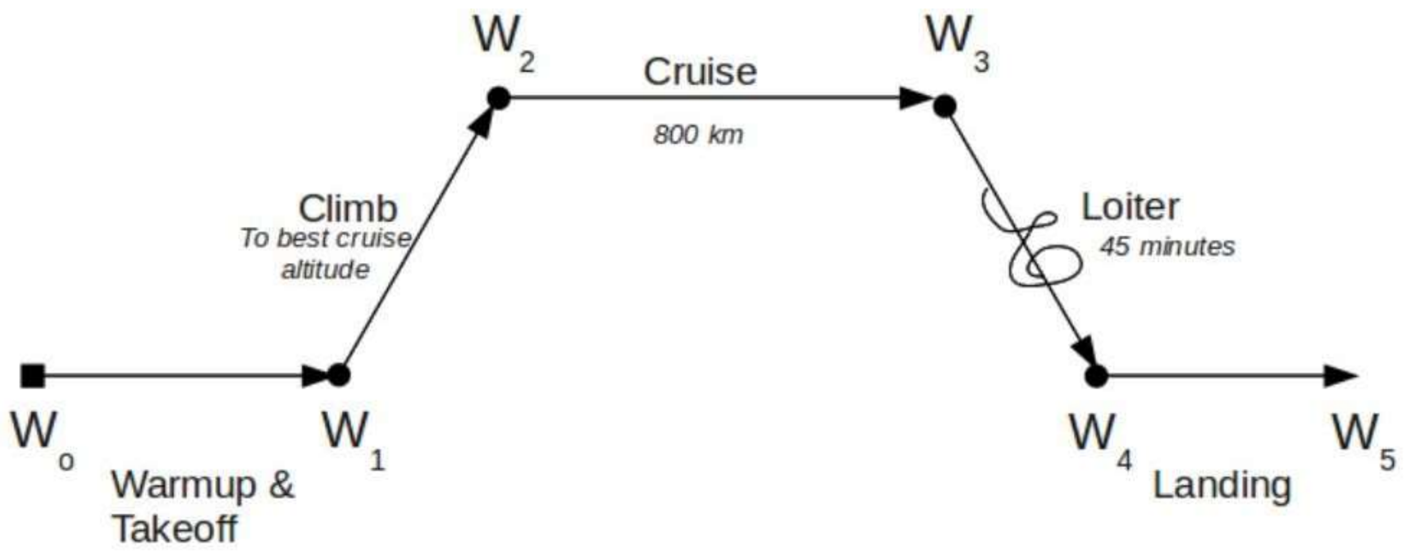
Instructed by- Mr. A.K Ghosh
Mr. D.K Giri

# Mission Requirements

1. Weight of aircraft < 1500kg
2. Number of passengers = 4. Assume 1 crew + luggage(10kg bag per each)
3. Design cruise speed – 50-70m/s
4. Stall speed(steady straight level flight) – 30-40m/s
5. Absolute ceiling(height from mean sea level) – 10,000ft
6. Rate of climb at mean sea level = 4-8m/s
7. Able to fly at minimum drag as well as minimum power. Therefore $V_{cruise_{P_{min}}} < V_{cruise_{D_{min}}}$
8. Endurance – Greater than 4 hours
9. Range – More than 1000 km (for round trip, 2000km)
10. Landing distance – 400-600m
11. Take-off distance – 300-500m
12. Service ceiling – 7km

Also we assume that we are dealing with a low speed air-craft and neutral position of control surfaces at design cruise speed.

# Step 1. Mission profile




- Take-off : 0-1
- Climb – 1-2
- Cruise – 3-4
- Loiter – 3-4
- Landing – 4-5

# Step – 2 Weight Estimation

- Types of weight
  a) Weight of crew $W_{crew} = W_c$
  b) Weight of payload $W_{payload} = W_p$
  c) Weight of fuel $W_{fuel} = W_f$
  d) Empty weight of aircraft $W_{empty} = W_e$
- Design take-off gross weight $W_0 = W_c + W_p + W_f + W_e$
- We define gross weight because actual weight changes due to fuel consumption
- First estimate of $W_0$ is made by transforming the gross-weight equation as follows

$$W_0 = \frac{W_c + W_p}{1 - \frac{W_f}{W_0} - \frac{W_e}{W_0}}$$

- This expression is useful as we can readily obtain ratios $\frac{W_f}{W_0}$ $and \frac{W_e}{W_0}$

# Step 2.1 Estimation of $\frac{W_e}{W_0}$

- Aircraft design is evolutionary. So statistical/historical data provides a starting point for conceptual design of a new aircraft.
- From Fig. 8.1 Ch-8 Airplane design in provided notes, $\frac{We}{W_0}$ with gross weight for reciprocating propeller driven airplane is given.
- For 2000-3000kg aircraft - $\frac{We}{W_0} = 0.62$
- According to sample calculations provided, ours is an aircraft < 1500kg weight, we can assume $W_0$ to be in the range 2000-3000 for now. So $\frac{We}{W_0} = 0.62$

# Step 2.2 Estimation of $\frac{W_f}{W_0}$

- The factors on which amount of fuel depends are mainly efficiency of propulsion device and aerodynamic efficiency. These are included in the *Brequet Range equation :-*

$$R = \frac{\eta_{pr}}{c}\left(\frac{L}{D}\right)\ln\left(\frac{W_0}{W_f}\right)$$

- For $i^{th}$ mission segment, weight fraction = $\frac{Wi}{W_{i-1}}$ .
- $\frac{Weight\ of\ airplance\ at\ end\ of\ \ mission}{Initial\ gross\ weight} = \frac{W_5}{W_0} = \frac{W_1}{W_0} \times \frac{W_2}{W_1} \times \frac{W_3}{W_2} \times \frac{W_4}{W_3} \times \frac{W_5}{W_4}$

Also $W_f = W_0 - W_5 \rightarrow \frac{W_f}{W_0} = 1 - \frac{W_5}{W_0}$. Allowing 6% allowance for reserve fuel and trapped fuel,

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_5}{W_0}\right)$$

- For 0-1 segment, historical data shows $\frac{W1}{W_0} \approx 0.97.$ For 1-2 segment, also historical data shows $\frac{0}{W_1} = 0.985$

- For 2-3 segment, we'll use Brequet range equation. For this $\frac{L}{D}$ estimated is needed. At this stage of design, we haven't laid out shape of airplane. So making an assumption based on existing airplanes –

| Airplane | $\left(\frac{L}{D}\right)_{max}$ |
|---|---|
| Cessna 310 | 13.0 |
| Beach Bananza | 13.8 |
| Cessna Cardinal | 14.2 |

So a reasonable estimate $\left(\frac{L}{D}\right)_{max} \approx 13.5$

- Typical value of $\eta_{pr} \approx 0.85$. Now $R = \frac{\eta_{pr}}{c}\left(\frac{L}{D}\right)\ln\left(\frac{W_2}{W_3}\right)$ where $c = 6.27 \times 10^{-7} m^{-1}$
- We are given R = 1000km. So, performing the calculations, $\frac{W2}{W_3} = 1.059$
- For segment 3-4, ignoring details of fuel consumption, $\frac{W4}{W_3} = 1$. For segment 4-5, during landing according to historical data $\frac{W5}{W_4} = 0.995$
- Now, $\frac{W5}{W_0} = \frac{W_1}{W_0} \times \frac{W_2}{W_1} \times \frac{W_3}{W_2} \times \frac{W_4}{W_3} \times \frac{W_5}{W_4} = 0.97 \times 0.985 \times \cdots \approx 0.897$
- Therefore, $\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_5}{W_0}\right) = 0.1092$

# Step 2.3 - Calculation of $W_0$

- Assumed average weight of a person is 75kg
- $W_p = 75 + 10 \times 4 = 340kg$
- $W_0 = \frac{W_c + W_p}{1 - \frac{W_f}{W_0} - \frac{W_e}{W_0}} = 1427kg$. $W_f = 155.83kg$

# Step 3 – Sizing of the wing

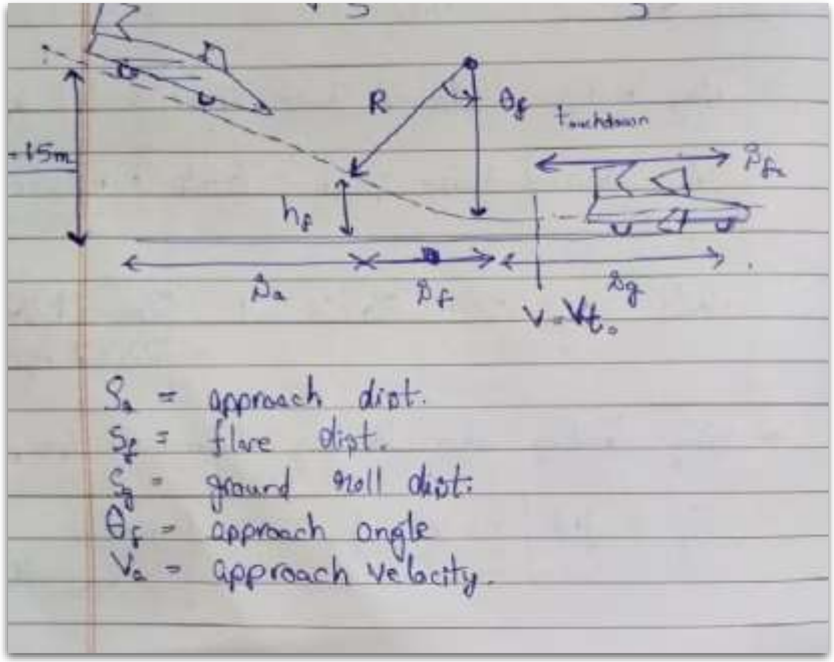
- For steady straight level flight $W = L$ (weight = lift)
- $W = \frac{1}{2}\rho V^2 S C_l \rightarrow \frac{W}{S} = \frac{1}{2}\rho V^2 C_l$
- $\frac{W}{S} = \frac{1}{2}\rho V_{stall}^2 C_{l_{max}}$ is called wing loading and $C_l$ = lift coefficient
- Taking $V_{stall} = 30m/s$ from mission requirements and $C_{l_{max}}$ = 2.3 at takeoff. $\frac{W}{S} = \frac{1267.87N}{m^2} = 129.24kg/m^2$

# Wing loading at desired cruise speed and lift coefficient

- V = desired cruise speed = 70m/s and $= 0.4$ from mission requirements.
- $\frac{W}{S} = \frac{1}{2}(1.225)(70)^2 \times 0.4 = \frac{1200.5N}{m^2} = 122.37kg/m^3$

# Wing loading when landing distance is specified

- $S = jN\sqrt{\frac{2}{\rho_\infty} \times \frac{W}{S} \times \frac{1}{C_{l_{max}}}} + \frac{\frac{j^2 W}{S}}{g\rho_\infty C_{l_{max}}\mu_r}$
- j = 1.15 for commercial airplanes, N = 3s (time increment for free roll immediately after touchdown) and $\mu = 0.4$
- $S = 2.9\sqrt{\frac{W}{S}} + \frac{0.1196\text{W}}{\text{S}}$



- Now, $S_g = S_l - S_a - S_f$
- $\theta_a = \theta_f$, obstacle length = 15m and $S_f = Rsin\theta_f$, $h_f = R \quad (1 - cos\theta_a)$
- $S_g = S_l - \frac{15 - h_f}{\tan\theta_f} - Rsin\theta_f$ . Taking $R = \frac{V_f^2}{2g}$ and $v_f = 1.23 \times V_{sta}$, $R = 693.99m$
- $S_g = 195.61m$
- Now, $195.61 = 2.9\sqrt{\frac{W}{S}} + \frac{0.1196W}{S}$. Taking $\frac{W}{S} = t^2$ and solving the resultant quadratic equation, $\frac{W}{S} = 902.75N/m^2 = 92kg/m^2$

- So,
  - $\left(\frac{W}{S}\right)_{stall\ conditions} = 129.24kg/m^2$
  - $\left(\frac{W}{S}\right)_{cruise\ velocity\ approximation} = 122.37kg/m^2$
  - $\left(\frac{W}{S}\right)_{landing\ approach} = \frac{92kg}{m^2}$
- $\left(\frac{W}{S}\right)_{landing\ approach}$ is off from other two values. If we choose $S_l = 550m$, then repeating the calculations, we get $\frac{W}{S} = 1209.82N/m^2 = \frac{123.32kg}{m^2}$
- Now we can see that all the three values are close to each other.
- To find the area of wing, $S = \frac{W}{\left(\frac{W}{S}\right)} = \frac{1427}{122.37} = 11.66m^2$
- Let aspect ratio AR of wing be 10. Then $AR = 10 = \frac{b^2}{S}$
- Wing span, $b = \sqrt{10 \times 11.66} = 10.798m$
- Wing chord, $c = \frac{b}{AR} = \frac{10.798}{10} \approx 1.08$

# Step 4 – Power Requirements

Calculating cruise power required at different altitudes

$$\left(\frac{R}{C}\right)_{\max} = \left(\frac{n_{p_r} p}{w}\right) - \left(\frac{2}{\rho_\infty}\sqrt{\frac{K}{3C_{D_0}}}\frac{w}{s}\right)^{1/2} \frac{1.155}{\left(\frac{L}{D}\right)_{\max}}$$

- $\rho$ = 1.225
- $(R/C)_{max}$ = 2m/s, 4m/s, 6m/s, 8m/s, 10m/s
- $C_{Do}$ = 0.035
- W/S = 129.37×9.81
- K = 0.075
- $W=W_o$ = 1404.96×9.8N

$$\eta_{pr}P/W = 0.8\rho/(1404.96\times9.81)$$

$$P = 1404.96\times9.81/0.8\ [(R/C)_{max} + 3.399]$$

- $(R/C)_{max} = 2 m/s$ , $P = 124.68$ hp
- $(R/C)_{max} = 4 m/s$ , $P = 170.87$ hp
- $(R/C)_{max} = 6 m/s$ , $P = 217.06$ hp
- $(R/C)_{max} = 10 m/s$ , $P = 309.44$ hp

# Step 5 – Tail Design

- Step 5.1:

$(C_L)_{design} = 0.5$

Static margin ,sm = 15% = 0.15

as, sm = - $\frac{dC_m}{dC_L}$

$C_{Mo}$ = 0.075

$C_{L_\omega} = C_{L_0} + L_{L_{\alpha_\omega}}\alpha$

Airfoil –

$C_{L_0 w}$ = 0.2084 , $C_{L_\alpha}$ = 5.9/rad , $c_{ma_c}$ = -0.08

$C_{m_0 w}$ = 0.03126

Aspect ratio(AR) = 10(approx) and e = 0.8

$$\left(C_{L_\alpha}\right)_{\omega ing} = \frac{C_{L_\alpha}}{1 + \frac{C_{L_\alpha}}{\pi e A R}}$$

$$C_{L_{a_w}} = \frac{\frac{5.9}{\pi e A R + 5.9}}{\pi e A R}$$

$$= 4.778 \times 2.5 \times \pi / 100 = 0.2084$$

$$\left(C_{m_{\mathrm{ac}}}\right)\mathrm{wing} = \left(C_{m_{ac}}\right) airfoil * \frac{AR}{AR+2}$$

= -0.08×10/12

= -0.066

- Main Formula

$$C_{m_{cg}} = C_{M_0} + C_{m_\alpha}\alpha$$

1. $$c_{m_0} = c_{m_{ac_w}} + c_{L_0 w}\left(x_{cg} - x_{ac}\right) + V_H \eta c_{L_{o_t}}\{\varepsilon_0 - it\}$$

Type equation here.

2. $$C_{m_\alpha} = C_{L_{\alpha_w}}\left(x_{cg} - x_{ac}\right) - V_H \eta c_{L_{\alpha_t}}\left[1 - \frac{d\varepsilon}{d\alpha}\right]$$

➢ Eqn. 1 And 2 should satisfy

$c_{m_0} = 0.031126$ and static margin = 0.15

$X_{ac} = C/4$ (C= 1.2 our design)

$X_{ac} = 0.3$ and $\eta = 0.9$, $V_H = 0.6$, $\varepsilon_0 = 0.76^o$

NACA 0009, -- $C_{L_{\alpha_t}}$ = 3.8/rad and $c_{L_0}$ = 0

- Iteration 1:

Assume $X_{cg}$ = 0.35 C

Find it in equation 1

0.03126= -0.066 + 0.2084(0.35-0.25) + 0.6×0.9×3,8[0.76-$i_t$]×π/180

$i_t^o$ = -1.734$^o$

➢ Equation 2

$X_{cg} = X_{NP}$ as $C_{m_\alpha} = 0$

$$X_{NP} = X_{ac} + \eta V_H \frac{C_{L_{\alpha_t}}}{C_{L_{\alpha_w}}}\left(1 - \frac{d\varepsilon}{d\alpha}\right)$$

$$\frac{d\varepsilon}{d\alpha} = 0.3041$$

$X_{NP} = 0.25 + 0.2988671$

$X_{NP} = 0.5488$

- ✓ Static margin designed = $X_{cg} – X_{NP}$ = 0.20
- ✓ Needed ( static margin) = 0.15

- Iteration 2:

  Assume $X_{cg}$ = 0.40

  $i_t^o = -1.0828^o$

➢ Equation 2

$C_{m_\alpha} = 0$

$X_{NP} = 0.549$

✓ Static margin = $X_{cg} – X_{NP}$ = 0.15(approx)

So finally,

Tail setting $i_t = -1.0828^o$

SM = 0.15 and $X_{cg} = 0.4$

# Step 6 – The stability derivatives

- (Area) S = 10.86 $m^2$
- (Span) B = 10.4 m
- (Aspect ratio) AR = 10
- (Chord) C = 1.05 m
- K = 0.044
- $C_{D_o} = 0.0881$
- $\frac{C_L}{C_D} = 15.5$ (approx)
- $\frac{\text{Thickness}}{\text{Chord}} = 12\%$
- e = 0.77

We choose NACA – 4412 airfoil

- $(C_L)_{max} = 1.25$
- $\alpha_{stall} = 13$ degree

- $C_{M_o} = -0.098$
- $C_l = 0.4$

➢ $C_{L_\alpha} = \frac{dC_L}{d\alpha} = 1.8\pi(1+0.8max/c)$

- $C_{L_{\alpha_{2d}}} = 6.198$

➢ $C_{L_\alpha} = \frac{C_{L_{\alpha_{2d}}}}{1+\frac{C_{L_{\alpha_{2d}}}}{\pi.AR}} = \frac{6.18}{1+\frac{6.18}{\pi\cdot 10}}$

- $C_{L_\alpha} = 5.164$/rad

Now,

- $\sum F_x$ = W – L

- L = $C_{L_o} + C_{L_\alpha}\alpha + C_{L_{\delta e}}\delta e$

Mean value

$$\left(\frac{dC_L}{d\alpha}\right)_{mean} = \frac{s_f}{s}\left(\frac{dc_L}{d\alpha}\right)_f + \left(1 - \frac{s_f}{s}\right) \frac{dc_L}{d\alpha}$$

(f – devotes the flap)

- $C_{D_\alpha}$

$$C_L = C_{L_\alpha}\alpha$$

$$C_D = C_{D_o} + K\, C_L^2$$

$$C_{D_\alpha} = \left(\frac{dC_D}{dC_L}\right) . \left(\frac{dc_L}{d\alpha}\right)$$

$$C_{D_\alpha} = 2K\ C_L\ .\ C_{L_\alpha}$$

$$C_{D_\alpha} = 2\times0.042\times0.45\times5.164$$

$$C_{D_\alpha} = 0.1952$$

- $C_{M_\alpha}$

$$\sum M = C_{M_o} + C_{M_\alpha}\alpha + C_{M_{\delta e}}\delta e$$

$$C_{M_\alpha} = C_{L_\alpha}\ (X_{cg} - X_N)$$

$$C_{M_\alpha} = -5.1\times0.15$$

Static margin = 0.15

$$C_{M_\alpha} = -0.765$$

# Pitch Rate

- $C_{L_q}$ – Represents change in airplane lift with varying pitching velocity, with α const.

Wing contribution to $C_{L_q}$

$$\Delta\alpha = \frac{q\left(X_{cg} - X_{\mathrm{ac}}\right)}{U}$$

($X_{cg}$ means dist. to C.G)

($X_{ac}$ means dist. to A.C)

- If a.c is very close to c.g the contribution from wing is negligible
- For light airplanes fuselage contribution to $C_{L_q}$ is smaller than wing, so ignored.

- Wing contribution

$$\frac{\partial C_L}{\partial \frac{c\alpha}{\partial U}} = \frac{\partial x'}{c} c_{L_\alpha}$$

( x' – distance from c.g to wing chord)

($c_{L_\alpha}$- wing lift slope )

➢ $c_{D_q}$ – Represents change in drag with varying pitch velocity at constant α.

- For subsonic flight, $c_{D_q}$ is very small and ingnored.

➢ $c_{M_q}$ – Represents change in pitching moment coefficient due to change in pitching velocity.

- Wing contribution to $c_{M_q}$ either opposes or increases the pitching motion
- Fuselage contribution is ignored.

$$\left(\frac{\partial C_m}{\partial C_L}\right)\left(\frac{\partial C_L}{\partial q}\right)$$

$$\rightarrow \left. C_{m_q} \right|_{tail} = \left. \frac{\partial C_m}{\partial\left(\frac{cq}{2U}\right)} \right|_{tail} = -\frac{l_t}{c} \left. \frac{\partial C_L}{\partial\left(\frac{cq}{2U}\right)} \right|_{tail}$$

* $C_{m_q}$ is always -ve, $C_{L_q}$ is +ve.

$$\rightarrow \left. C_{m_q} \right|_{wing} = \frac{\partial C_m}{\partial\left(\frac{cq}{2U}\right)} = -\frac{|x'|}{c} \left. \frac{\partial C_L}{\partial\left(\frac{cq}{2U}\right)} \right|_{wing}$$

$$\therefore C_{m_q} = -\frac{2x'}{c^2}|x'| C_{L_\alpha} - \frac{2 l_t^2}{c^2} C_{L_{\alpha_t}} \frac{S_t}{S_w} \eta_t$$

- Typical value of $c_{L_\alpha}$ for light airplane falls in range 4.0 – 7.0 depending on the type of wing.
- The value of $c_{L_q}$ for 'Cesna 182' is 3.9
- $c_{D_q}$ is chosen to be 0
- $c_{M_q}$ for Cessna 182 = -12.43

# -Drawing Layout of Aircraft-

- AR (Aspect ratio) = 10
- S ( Area) = 10.86 $m^2$
- b ( wing span) = 10.4m
- C ( wing chord) = b/AR = 1.05m
- $L_f$ ( Fuselage length) = 70% of wing span = 7.28m

Now,

Data from similar airplane

➢ Horizontal tail

  - $S_h/S = 0.03$
  - $S_{horiz\ tail} = 3.36\ m^2$
  - $A_h = 5$
  - $b_n = 4.098m$

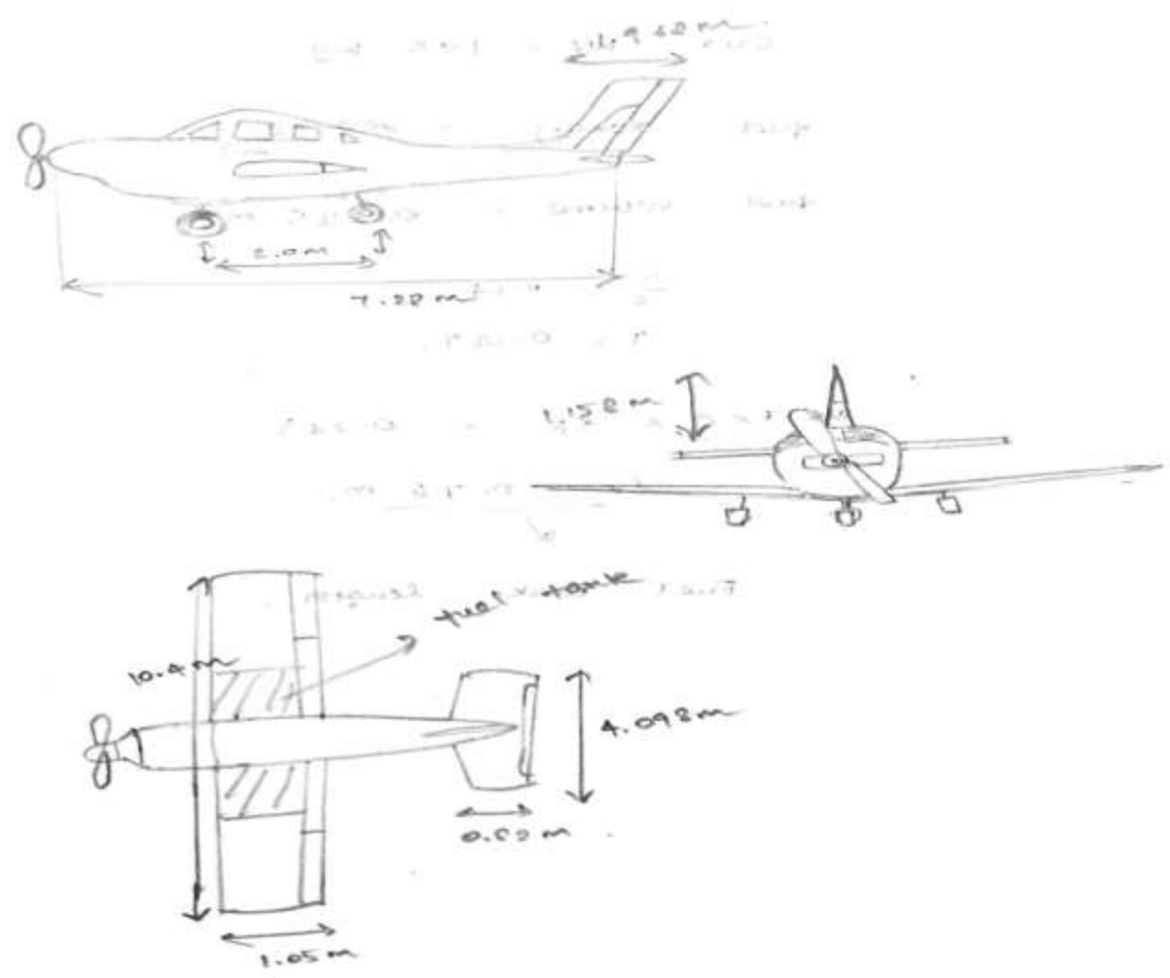
- $C_h = S/b = 0.82m$

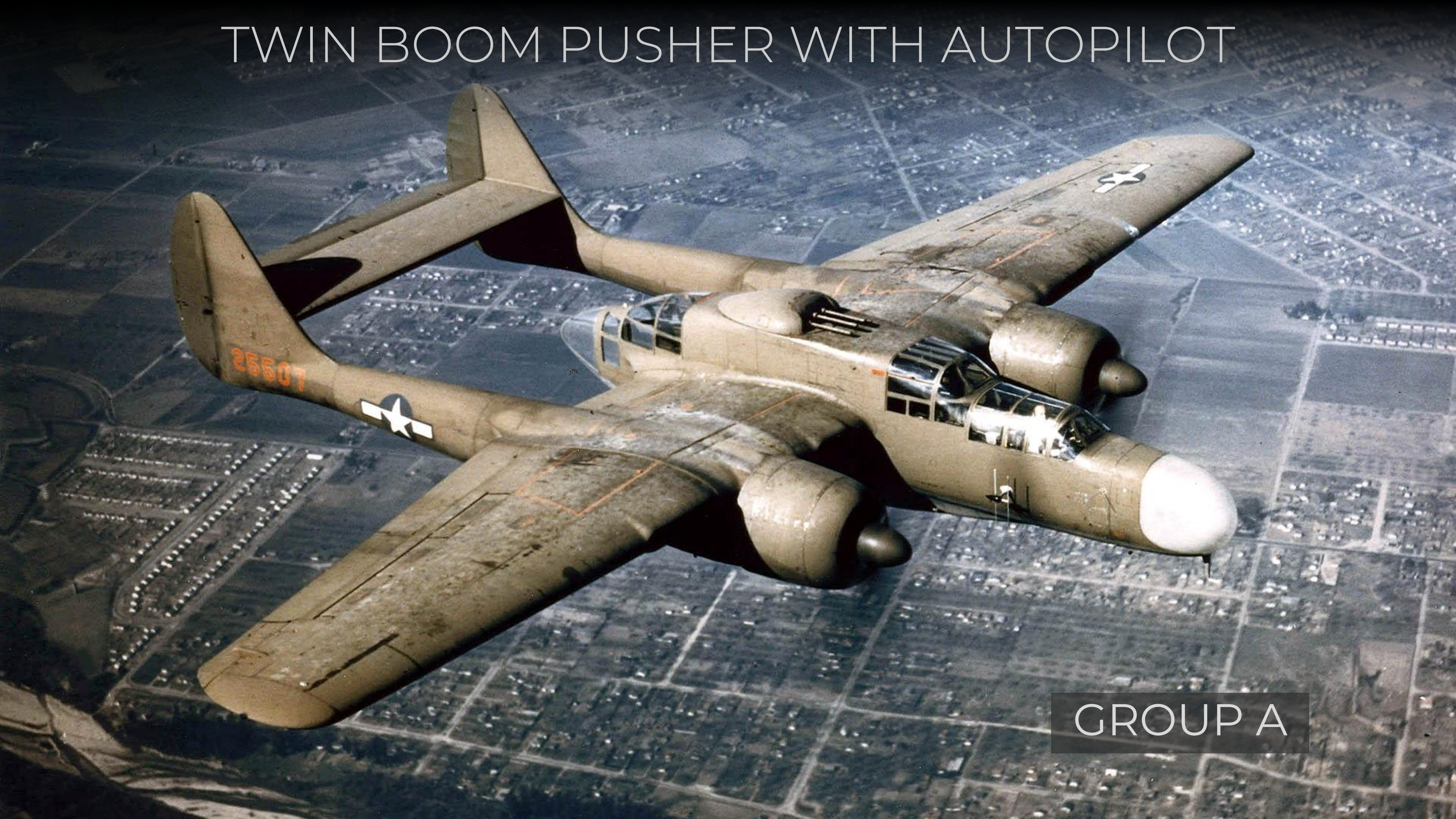
- Vertical tail
  - $S_v/S = 0.21$
  - $S_{vertical\ tail} = 2.28\ m^2$
  - $A_v = 1.7$
  - $b_v = 1.968m$
  - $C_v = 1.158m$

- Fuel tank
  - fuel weight = 195kg
  - fuel density = 800 $kg/m^3$
  - fuel volume = 0.245 $m^3$
  - $T/c = 0.14$
  - $T = 0.147$

$$T \times C \times l_f \times 2 = 0.245$$

- $l_f = 0.76m$ ( fuel tank length)




TWIN BOOM PUSHER WITH AUTOPILOT
25507
GROUP A

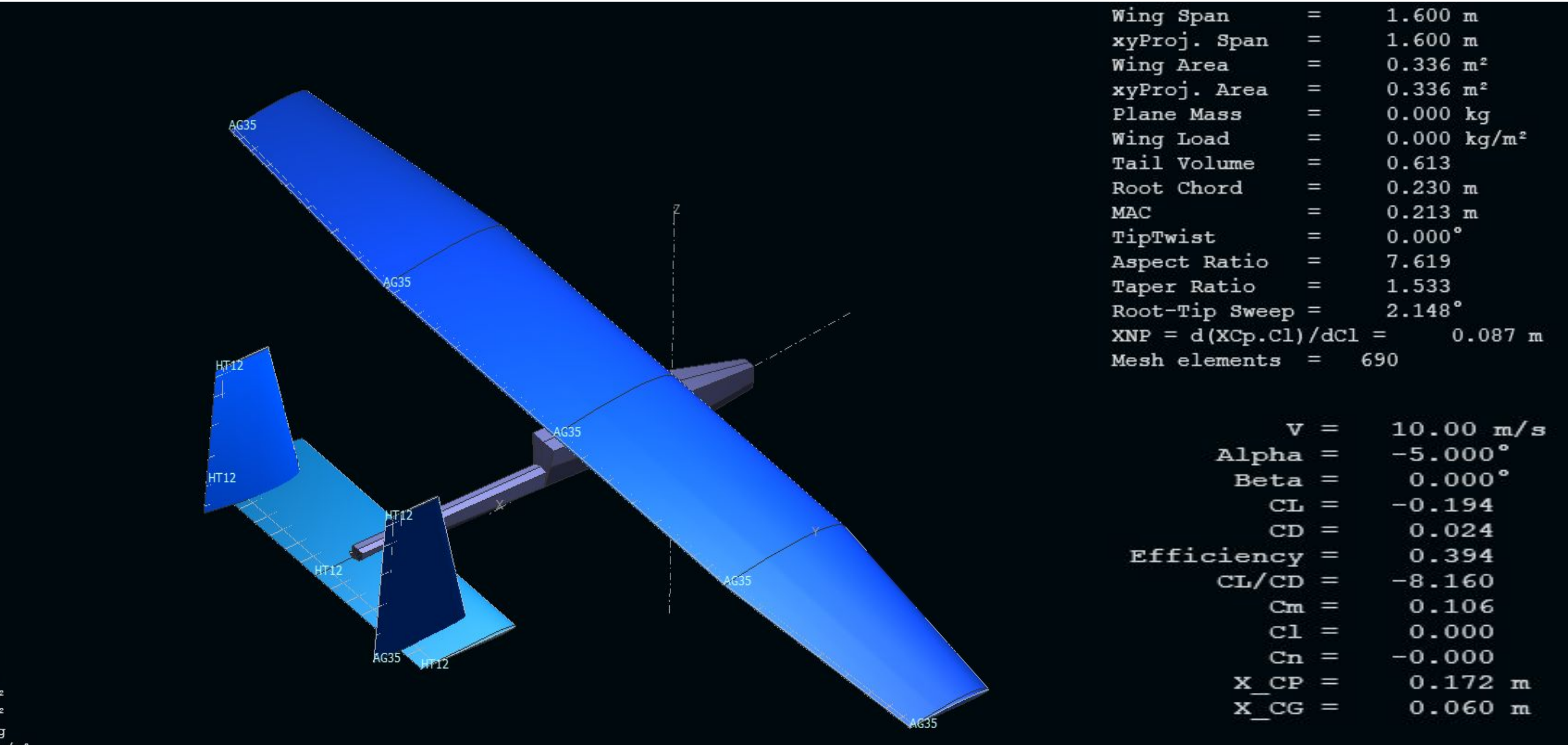
## MOTIVATION

We wanted to do something different, something challenging and something unique. History had the answer. Commemorating the WW1 warbirds!

## ABOUT TWIN BOOM AIRCRAFTS

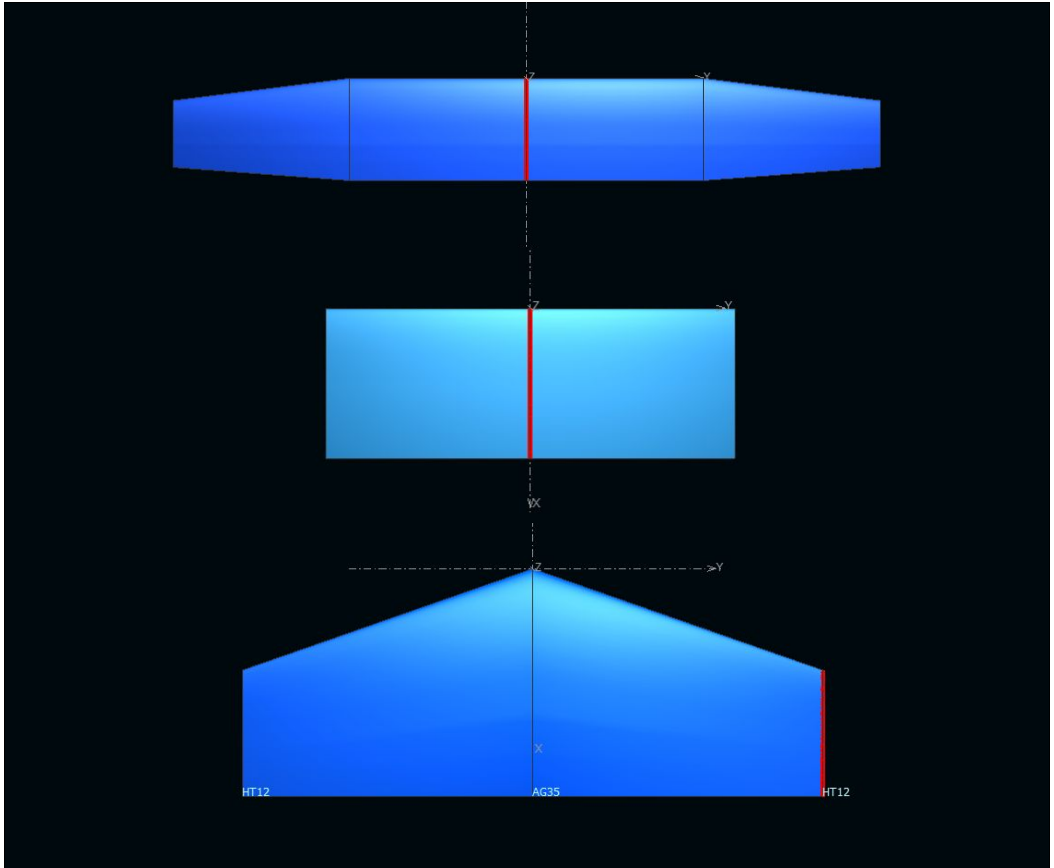
This is an aircraft with two nacelles or longitudinal booms. These booms can be used to carry fuel or can be used for supporting structural mounting. They are generally lighter and smaller, have more structural support to wings, easy in loading/unloading goods.

# DESIGN



AG35
AG35
AG35
AG35
AG35
AG35
HT12
HT12
HT12
HT12
HT12
Z
X
Y
Wing Span = 1.600 m
xyProj. Span = 1.600 m
Wing Area = 0.336 m²
xyProj. Area = 0.336 m²
Plane Mass = 0.000 kg
Wing Load = 0.000 kg/m²
Tail Volume = 0.613
Root Chord = 0.230 m
MAC = 0.213 m
TipTwist = 0.000°
Aspect Ratio = 7.619
Taper Ratio = 1.533
Root-Tip Sweep = 2.148°
XNP = d(XCp.Cl)/dCl = 0.087 m
Mesh elements = 690
V = 10.00 m/s
Alpha = -5.000°
Beta = 0.000°
CL = -0.194
CD = 0.024
Efficiency = 0.394
CL/CD = -8.160
Cm = 0.106
Cl = 0.000
Cn = -0.000
X_CP = 0.172 m
X_CG = 0.060 m

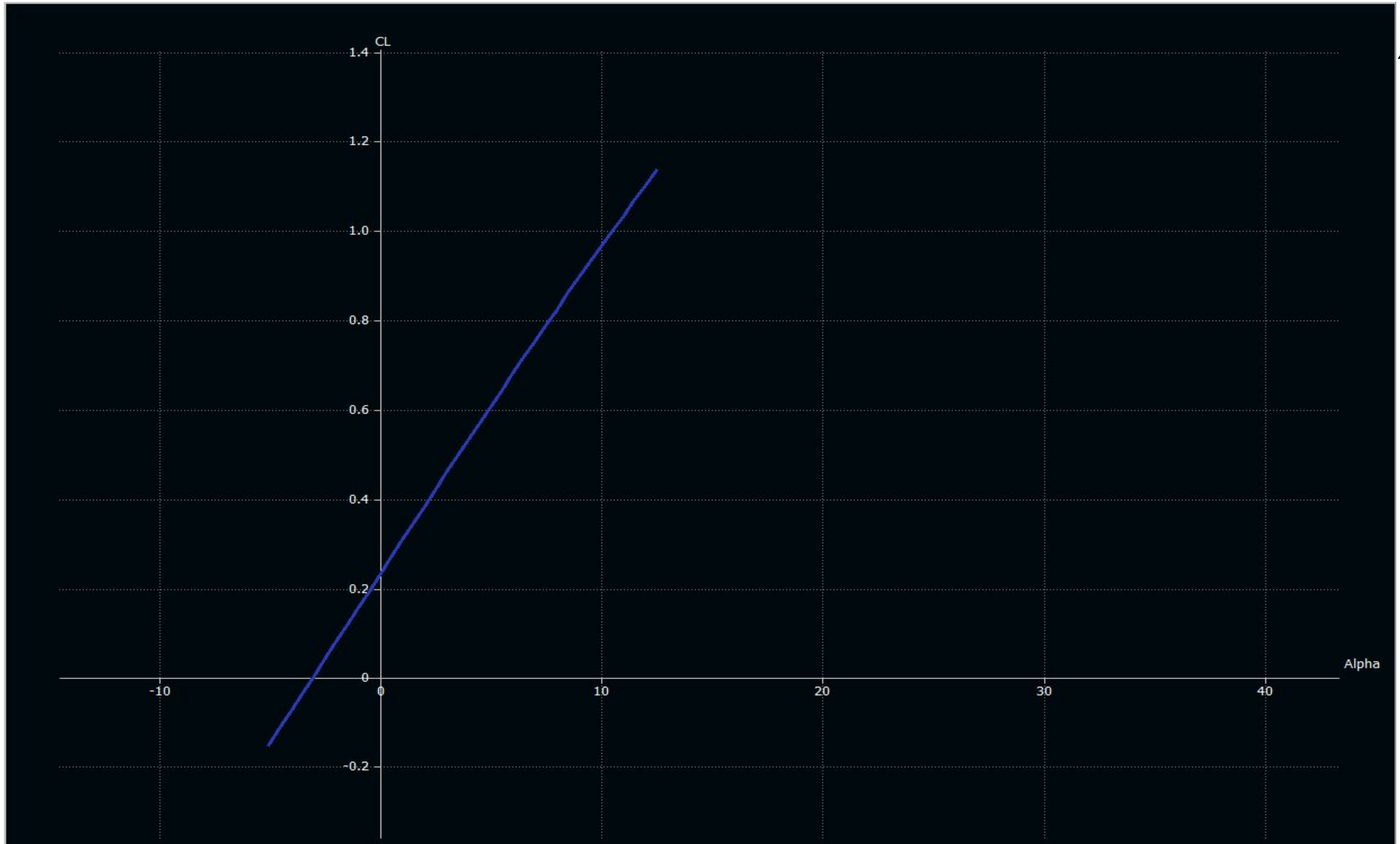
# DESIGN




AG35 Airfoil
Span = 1.6 m

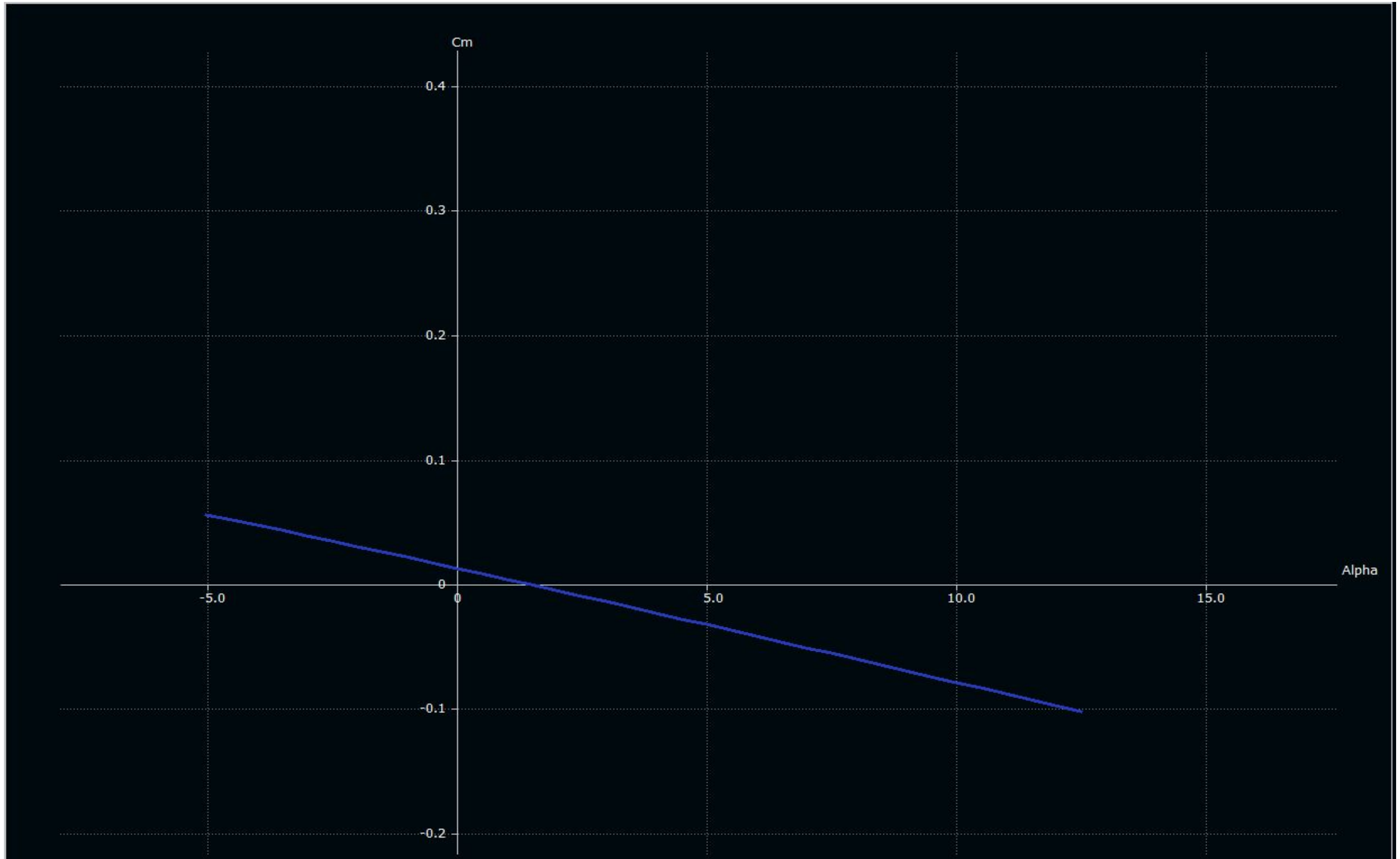
HT12 Airfoil
Span = 0.5 m

HT12 Airfoil
Span = 0.23 m



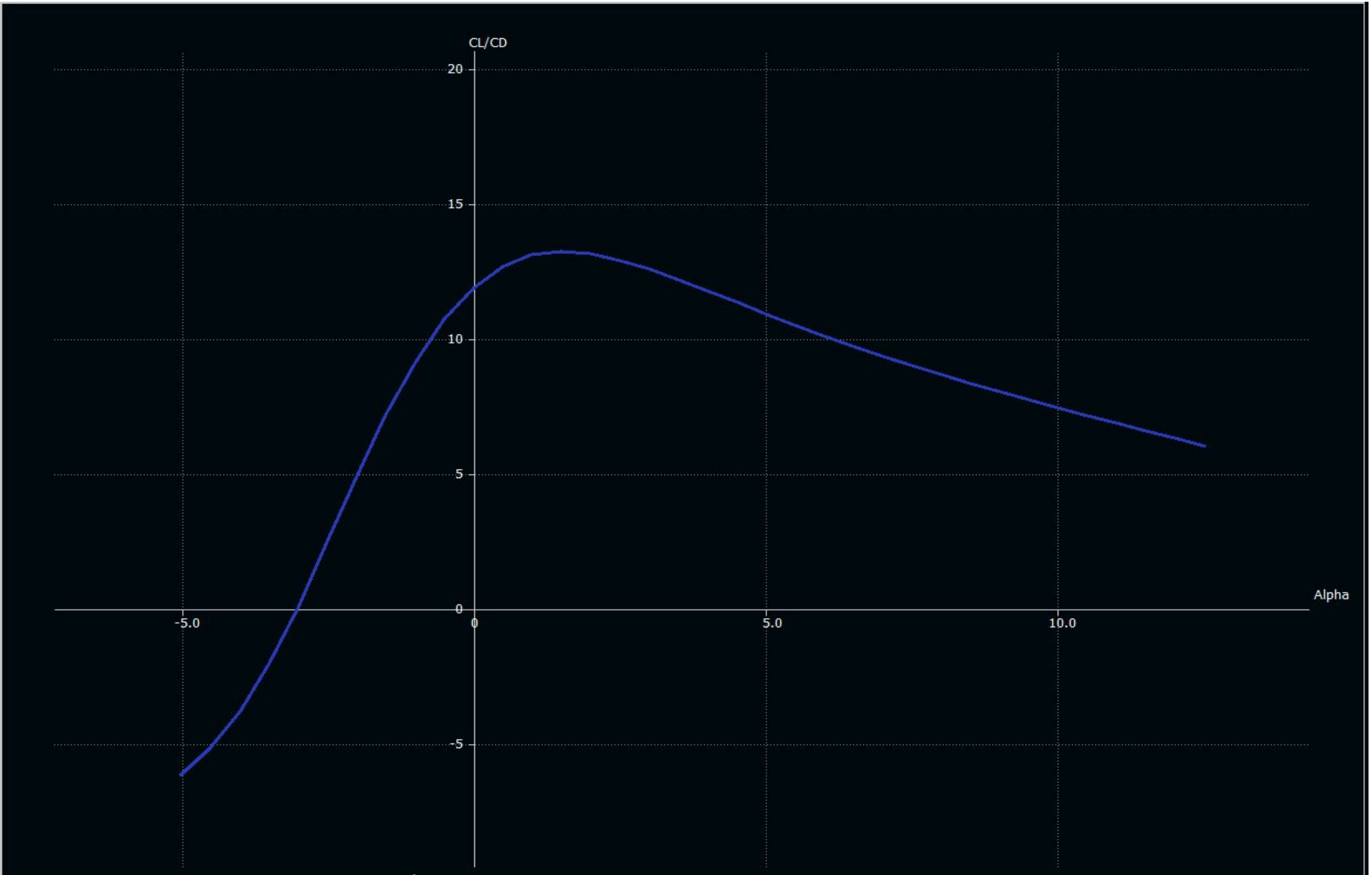
Coefficient of Lift (Cl) vs Angle of attack (alpha)

*At Alpha= Zero,*
*Cl =0.22*




Coefficient of moment (Cm) vs Angle of attack (alpha)

*Trim angle (the angle at which moment is zero)= $2.3^0$*




Cl/Cd vs Angle of attack (alpha)

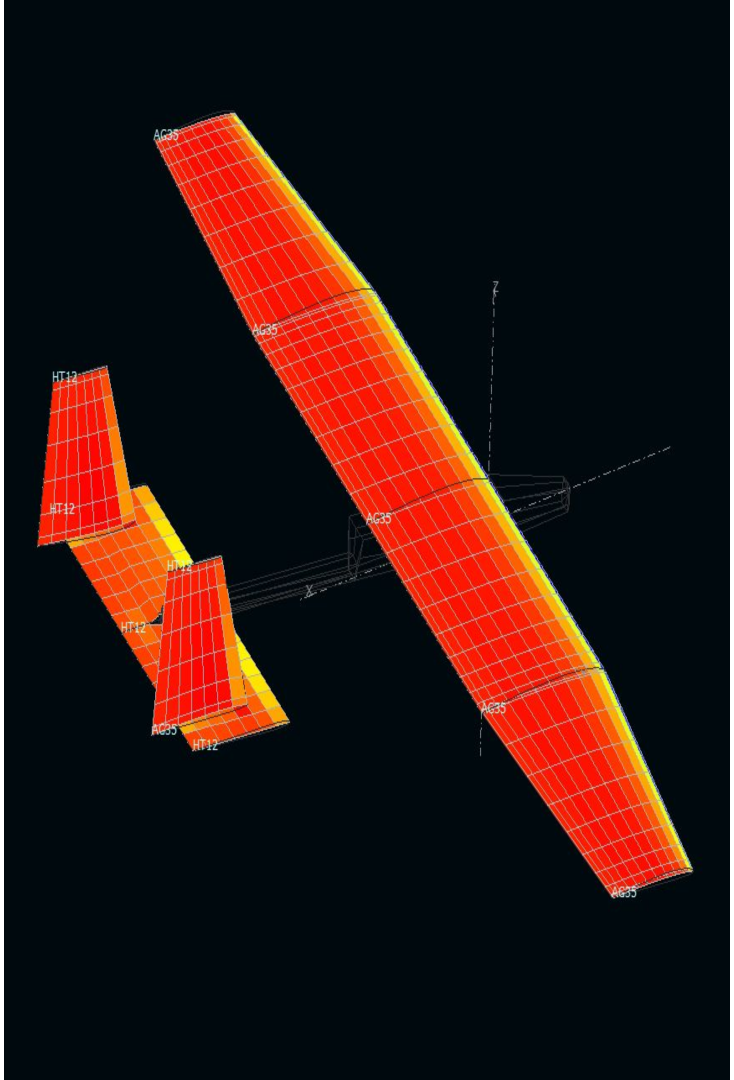
*Cl/Cd value is maximum at alpha=$2.2^0$*

# AERODYNAMIC ANALYSIS

The above-mentioned graphs and plane model were designed and analysed in XFLR5 Software.

Aerodynamic points needed to be considered while analysis ar as follows:-

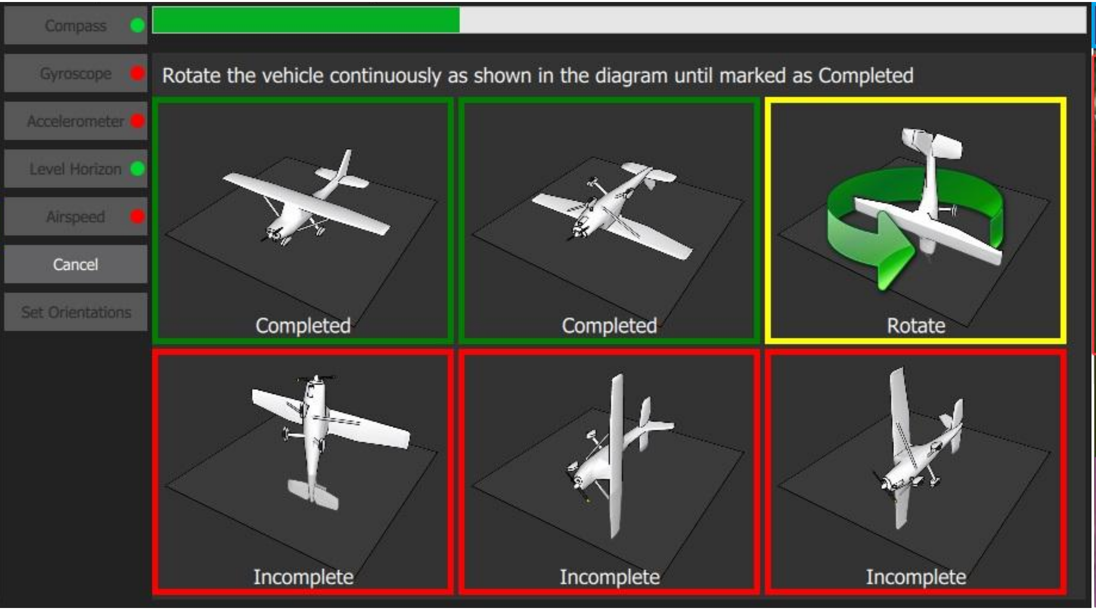
1. CG of the model must be ahead (nearer to the nose) of NP, for longitudinal stability that is stability in pitching plane
2. The slope of **Cm vs Alpha** graph should be **negative** for aerodynamically stable aircraft
3. The **trim angle**(here is $2.3^0$)  shouldn't be of much higher value.
4. The value of alpha in the graph of **Cl/Cd vs alpha** and **Cm vs alpha** should be nearly the same, for better stability and efficiency
5. The value of Cl must be **positive** at alpha=$0^0$



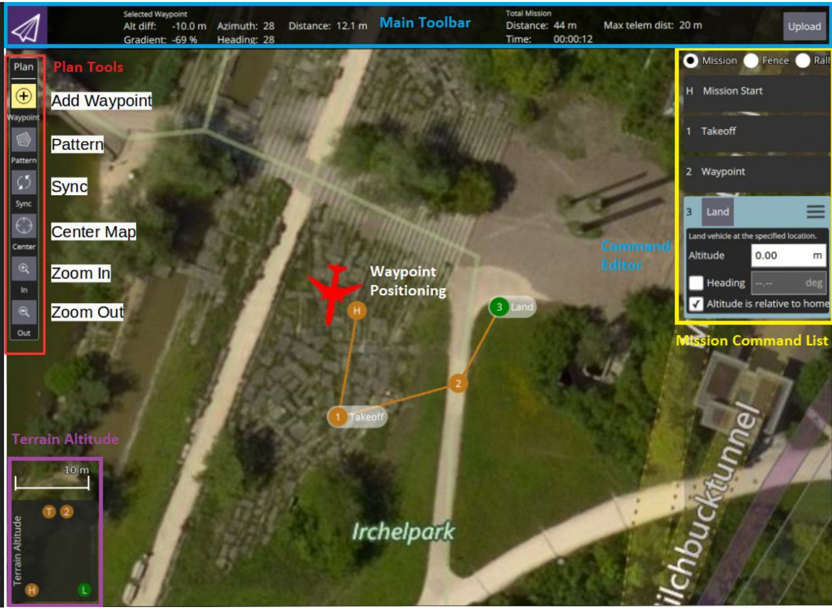
# AVIONICS

- 2200 KV BLDC Motor
- 30A ESC
- 7 inch Propeller
- Control Surface Servo
- 3S 2200mAh Battery
- Pixhawk FC
- GPS Sensor
- AirSpeed Sensor
- Telemetry Pair
- Radio Tx/Rx

# AUTONOMY




Frame Setup
Calibration
Parameter Tweaking




Mission Planning

# COST ANALYSIS

| | |
|---|---|
| Motor | INR 900 |
| ESC | INR 500 |
| Propellers | INR 200 |
| Servos | INR 400 |
| LiPo Battery | INR 1550 |
| Pixhawk with power module | INR 6000 |
| GPS | INR 1800 |
| Carbon Rod | INR 2598 |
| Airspeed Sensor | INR 2000 |
| Telemetry Pair 915 MHz | INR 1700 |
| Transmitter & Receiver | INR 7100 |
| Styrofoam and Monokote | INR 800 |
| **TOTAL** | **INR 25,548** |

# WORK ANALYSIS

- Work Done

  We cut the parts of the aircraft, wing (in sections), Horizontal and Vertical Tail, with Green Styrofoam.

  Proper sanding and cutting have been done.
- Work Planned
  - Lab-1: Fuselage and starting basic assembly.
  - Lab-2: Placing all the electronics, building control surfaces and final assembly. Pixhawk configuration.
  - Lab-3: Covering all the parts with a Monokote sheet.
  - Lab-4: Testing.

# GLIDER

A glider model for getting hands-on training in fabrication was made by our group within two and a half labs. This was made of styrofoam where we learned skills like basic fabrication techniques, handling hot wires, sanding, taping and assembly of the model.  We learned adjustment of CG and did multiple tests. However the model was damaged and we repaired it before its final flights.

# THANK YOU!

| | |
|---|---|
| Abhijeet | 170015 |
| Aditya Raghuvanshi | 170052 |
| Amrendra Pratap Singh | 170097 |
| Mataria Pence Jagatkumar | 170382 |
| Naveen Balaji N | 170420 |