Indian Institute of Engineering Science and Technology, Shibpur

Aircraft Design and Manufacturing Techniques AE4172

FIGHTER AIRCRAFT

Department of Aerospace Engineering and Applied Mechanics

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1 Introduction

Fighter aircrafts are fixed-wing aircrafts, designed primarily to secure control of essential airspace by destroying enemy aircraft in combat. The opposition may consist of fighters of equal capability or of bombers carrying protective armament. For such purposes fighters must be capable of the highest possible performance in order to be able to out-fly and outmanoeuvre opposing fighters. Above all, they must be armed with specialized weapons capable of hitting and destroying enemy aircraft.

In this project we are designing propeller driven fighter aircraft by first studying all the similar air-crafts previously designed. Using the collected data we will first assume some initial properties of the aircraft which are required for designing and select some properties according to our requirements. Then using the selected values and assumed values, we will design and calculate the results of the aircraft and check if it is according to our requirements, or else we have to change the selected/assumed values. This is how our final design is created.

2 Literature survey

A study on aircrafts with similar characteristics is done first. Here we have gathered the data of various aircrafts with similar features and compared the performances of these aircrafts. Calculation of various parameters for these aircrafts are carried out, to determine the way the performance varies with these parameters. Thus this study gives us a vague idea on the size, shape, characteristics, performance, etc. of the aircraft we are going to design. It is seen that these information help us taking many crucial decisions while designing our aircraft.

The different data of aircrafts obtained from various sources are listed below-

2.1 Comparative study on specifications and performance

2.1.1 Crew

A group of people who works during the flight mission and operate an aircraft.

2.1.2 Empty Weight

The empty weight of an aircraft is the weight of the aircraft without including passengers, baggage, or fuel.

2.1.3 Take-off Weight

It is the maximum weight at which the pilot is allowed to attempt to take off due to structural or other limits.

2.1.4 Wing Loading

It is the total weight of an aircraft divided by the area of its wing.

2.1.5 Wing Area

It is the projected area of the wing planform and is bounded by the leading trailing edges and the wing tips.

2.1.6 Wing Span

The maximum distance between the two wing tips is wing span and is denoted by b.

2.1.7 Cruise Span

The speed at which combustion engines have an optimum efficiency level for fuel consumption and power output.

2.1.8 Service Ceiling

It is the altitude where the maximum rate of climb is 100 ft/min and it's represented the practical upper limit for steady, level flight.

2.1.9 Range

It is the maximum distance an aircraft can fly between take-off and landing, as limited by fuel capacity in powered aircraft.

2.2 Comparative Data sheet

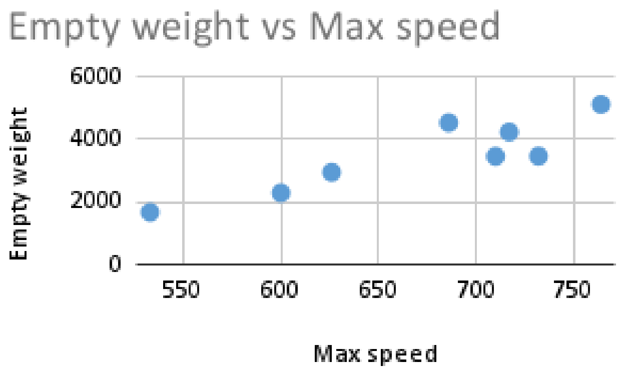
It is the collection of data of various airplanes related to the concept taken.

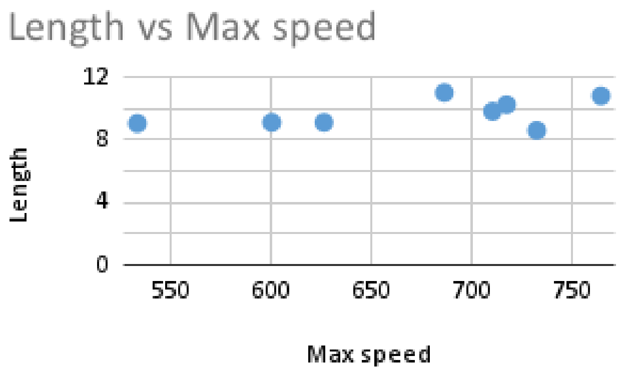
	L(m)	S(m^2)	b (m)	W_{TO}(Kg)	W_E (Kg)
Bell P-39 Airacobra	9.19	19.8	10.36	3810	2956
Mitsubishi A6M zero	9.06	22.44	12	2796	1680
North American P-51 Mustang	9.83	21.8	11.28	5488	3463
Vought F4U Corsair	10.26	29.17	12.5	6592	4238
Republic P-47 Thunderbolt	11.02	27.87	12.43	7938	4536
Supermarine spitfire	9.12	22.49	11.23	3039	2297
De Havilland Hornet	10.82	33.5	13.716	8278	5122
Grumman F8F Bearcraft	8.61	22.7	10.92	6105	3470

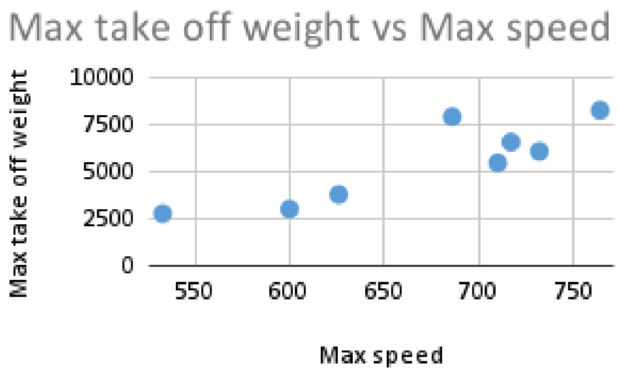
	V_{max} (Km/h)	Service Ceiling (m)	R (Km)	WL (Kg/m^2)
Bell P-39 Airacobra	626	11000	854	169
Mitsubishi A6M zero	533	10000	1870	107.4
North American P-51 Mustang	710	12800	2660	190
Vought F4U Corsair	717	12600	1617	
Republic P-47 Thunderbolt	686	13000	1660	210
Supermarine spitfire	600	11100	771	
De Havilland Hornet	764	12600	2380	213.9
Grumman F8F Bearcraft	732	12400	1778	210

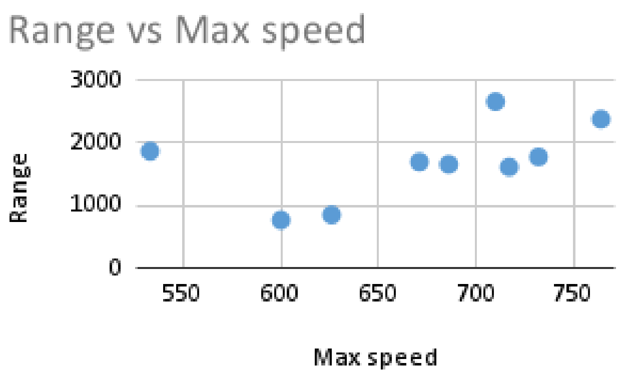
To proceed further, we need to assume some properties of the aircraft. For which we use this collection of data and assume certain values (Mostly taking average of all aircrafts). We will use these assumed values to design the parts of the aircraft according to our chosen topic.

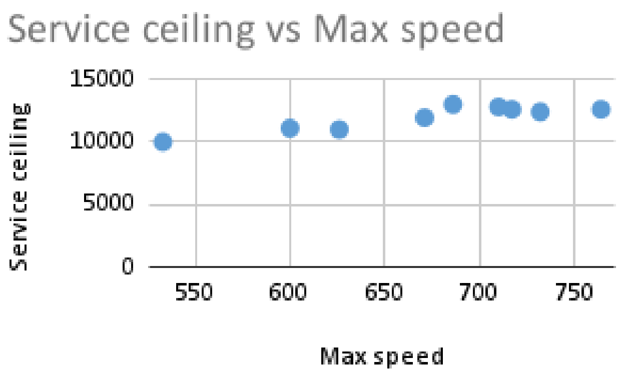
We also made graphs for all the data, so that it is easy to take an estimated average of that property -

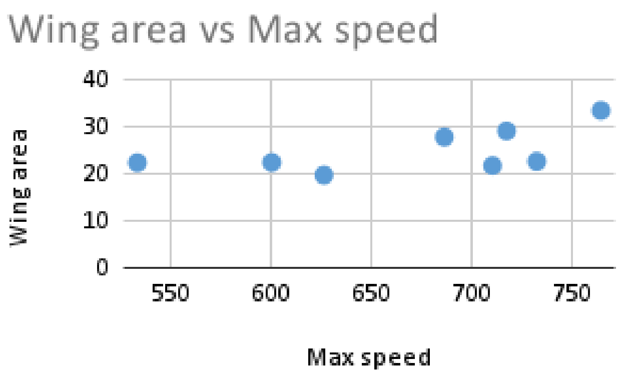


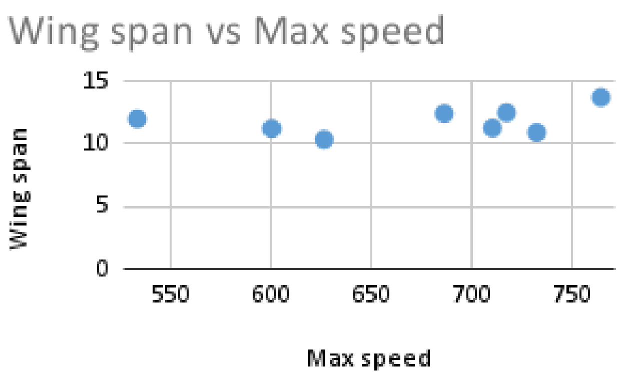





Range vs Max speed
3000
2000
1000
0
550
600
650
700
750
Range
Max speed


Service ceiling vs Max speed
15000
10000
5000
0
550
600
650
700
750
Service ceiling
Max speed


Wing area vs Max speed
40
30
20
10
0
550
600
650
700
750
Wing area
Max speed


Wing span vs Max speed
15
10
5
0
550
600
650
700
750
Wing span
Max speed

3 Airfoil Selection

Selection of airfoil is an important step which is very crucial and has a huge impact on the overall performance of the aircraft. The important features that we will need to look for are low zero lift drag (C_{d0}) , high lift curve slope ($C_{l\alpha}$) , high aerodynamic efficiency(C_l/C_d), high max Lift coefficient(C_{lmax}) and angle of stall (α_{stall}). Apart from these parameters certain other features like drag bucket, low zero lift

3.1 For wing

Properties	GOE 546	GOE 490	NACA 64(1)-212 MOD A	CLARK X	S2027
CL (α=0)	0.45	0.3857	0.2761	0.3735	0.3039
α (CL=0)	-4.5	-3.8	-2.5	-3.4	-2.75
CL max	1.5	1.3371	1.5225	1.4944	1.4151
α stall	15	13.5	15.75	15.25	15.5
Cd min	0.005	0.00689	0.0091	0.00526	0.00632
CL at Cd min	0.5	0.2372	0.4682	0.4689	0.1892
CL/CD max	125	114.12	83.45	122.42	122.89
CL at CL/CD max	0.74	0.8468	1.0239	0.8508	1.0556
LE Radius (%)	2.6	2.5	3	3	2.6

Properties	Score for Airfoil				
	GOE 546	GOE 490	NACA 64(1)-212 MOD A	CLARK X	S2027
CL (α=0)	1.0000	0.8571	0.6136	0.8300	0.6753
α (CL=0)	1.0000	0.8444	0.5556	0.7556	0.6111
CL max	0.9852	0.8782	1.0000	0.9815	0.9295
α stall	0.9000	1.0000	0.8571	0.8852	0.8710
Cd min	1.0000	0.7257	0.5495	0.9506	0.7911
CL at Cd min	1.0000	0.4744	0.9364	0.9378	0.3784
CL/CD max	1.0000	0.9130	0.6676	0.9794	0.9831
CL at CL/CD max	0.7227	0.8270	1.0000	0.8309	1.0310
Total	7.6079	6.5199	6.1797	7.1510	6.2705

The choice of airfoils from the root of the wing to the tip of the wing is dictated by both aerodynamic and structural considerations. In the root region thicker airfoils are necessary to accommodate the high wing root bending moment. It is best to add the extra thickness to the lower half of the airfoil to minimize the adverse affects of thickness. Toward the tip airfoil thickness can be reduced for better airfoil performance due to much lower bending moments. The thinner the airfoil the lower the drag coefficient for a given lift coefficient.

Using the above data, we chose the the airfoils for tip and root as:

		CL (α=0)	CL max	α stall	Cd min	CL/CD max
Root Airfoil	Clark X Airfoil	0.3735	1.4944	15.25	0.00526	122.42
Tip Airfoil	GOE 546 Airfoil	0.45	1.5	15	0.005	125

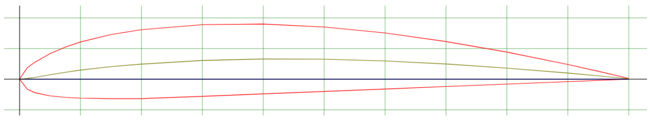
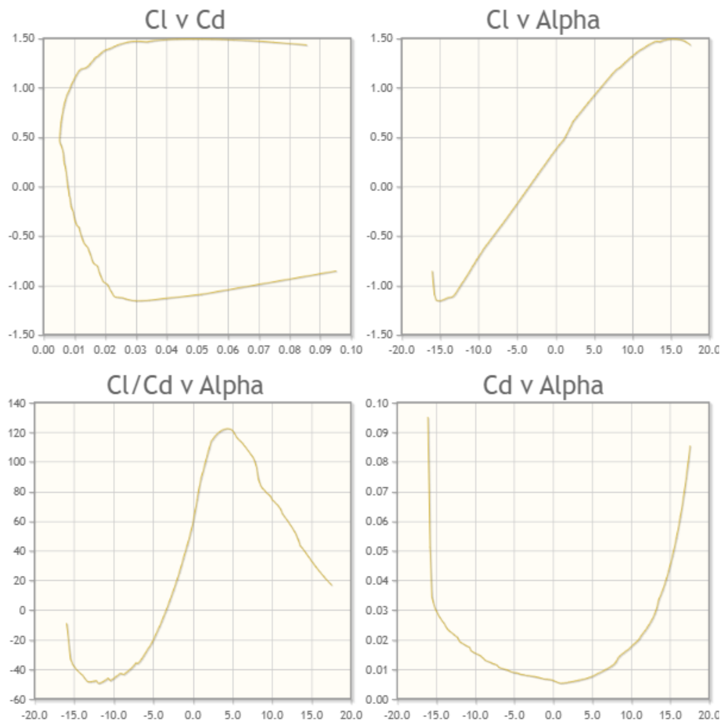
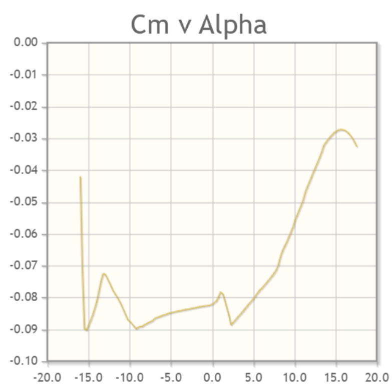


Figure 1: Clark X Airfoil





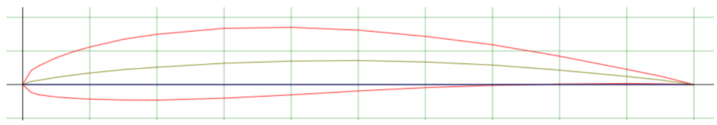
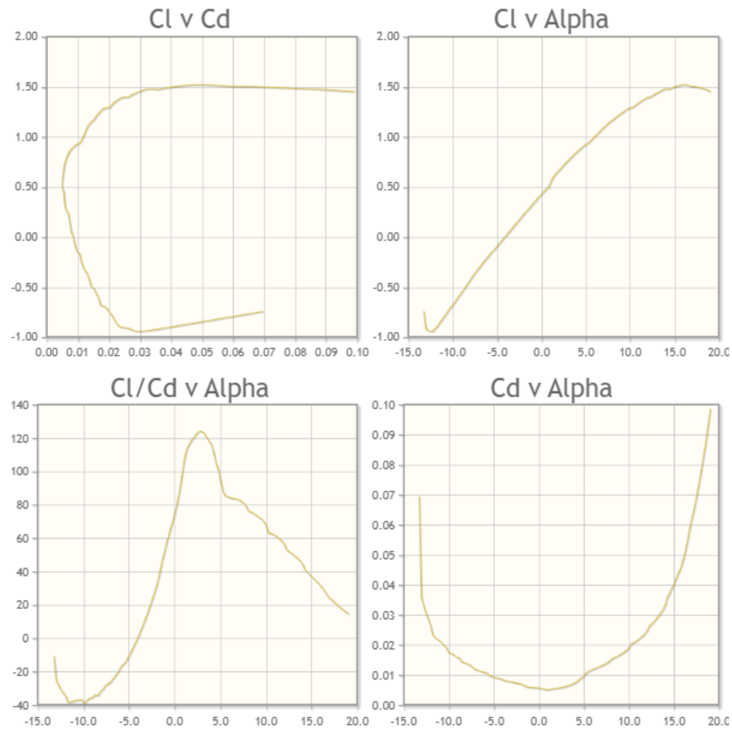
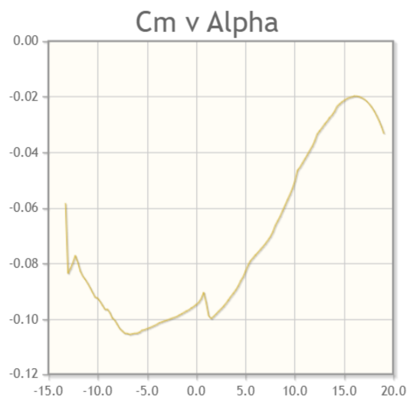


Figure 2: GOE 546 Airfoil





4 Wing Sizing

4.1 Wing vertical location

We have chosen the wing vertical location as low wing, since a low wing aircraft gives better roll stability. It is elaborated below in this report.

4.2 Wing incidence angle (i_w)

The wing incidence must satisfy the following design requirements:

1. The wing must be able to generate the desired lift coefficient during cruising flight.
2. The wing must produce minimum drag during cruising flight.
3. The wing setting angle must be such that the wing angle of attack could be safely varied(in fact increased) during take-off operation.
4. The wing setting angle must be such that the fuselage generates minimum drag during cruising flight (i.e. the fuselage angle of attack must be zero in cruise).

Satisfying the above requirements, the wing incidence angle is chosen as, i_w = 2deg.

4.3 Taper ratio (λ)

Taper ratio (λ) is defined as the ratio between the tip chord (C_t) to the root chord (C_r). By adding a taper ratio, it will lower the bending moment at the wing root and will also reduce some wing weight.

$$\lambda = \frac{C_t}{C_r} \tag{1}$$

Taper ratio is estimated by using historical data as, $\lambda = 0.5$

4.4 Sweep angle (Λ)

By adding a sweep angle it improves wing aerodynamic features by delaying the compressibility effects. Sweep angle is also chosen by the use of historical data, that is:

$$\Lambda_{c/2} = 30\,\text{deg} \tag{2}$$

4.5 Twist angle

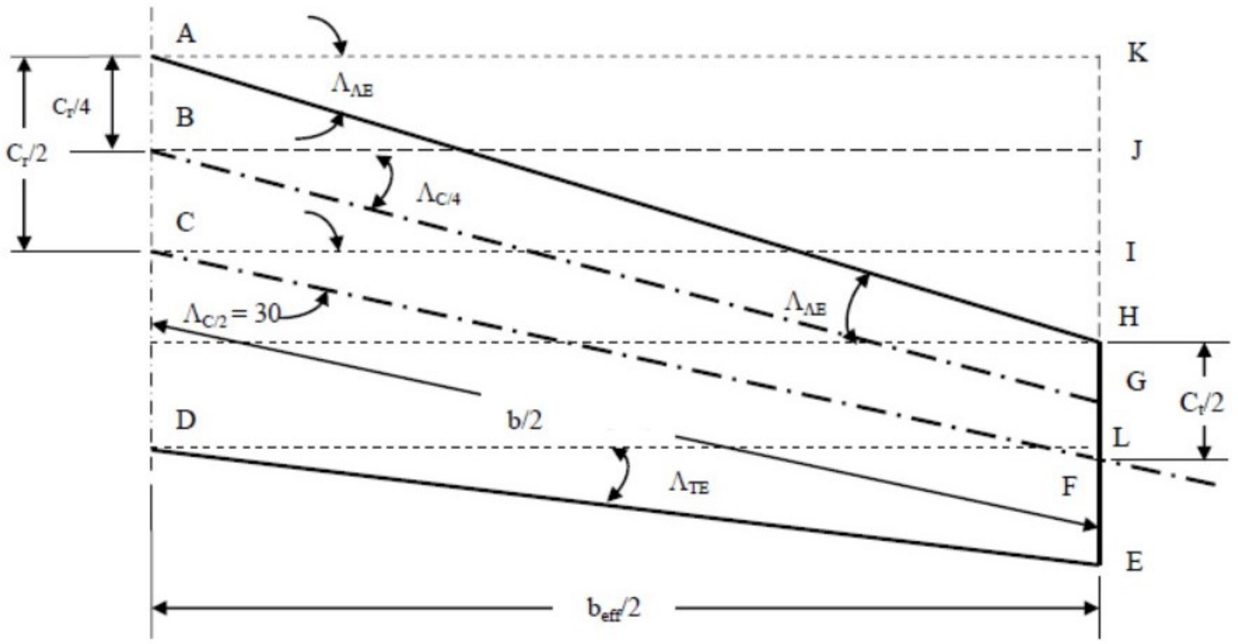
Twist angle is introduced already since we are using 2 different airfoils for tip and root. It avoids root stall before tip stall. It is taken as:

$$\alpha_t = 1\,\text{deg} \tag{3}$$

Finally, the final design of our wing is as follows:

Wing span, b (m)	11.8045
Wing Area, S (m2̂)	24.97125
Taper Ratio, λ	0.5
Sweep Angle, (deg) at c/2	30
AR	5.580266116
Density at 9000m (kg/m3̂)	0.467
Viscosity at 9000m (m2̂/s)	0.00003196
Twist angle (deg)	1
Wing incidence angle (deg)	2
Cr	2.820534542
Ct	1.410267271
C_mac	2.193749088
b_eff	10.22299688
AR_eff	4.185199587

Table 1: Wing Design



5 Weight estimation

Takeoff gross weight, W_0 is estimated in this section. It is the total weight of aircraft as it begins the mission. Design takeoff weight includes crew weight, payload (or passenger) weight, fuel weight, and the empty (the structure, engines, landing gear, fixed equipment, avionics) weight.

$$W_0 = W_{crew} + W_{payload} + W_{fuel} + W_{empty} \tag{4}$$

The crew includes 1 pilot 80 kgs. the weight of the payload is approximated 1500 kgs. The only unknowns are the fuel weight and empty weight. However, they are both dependent on the total aircraft weight. Thus, an iterative process must be used for aircraft sizing. To simplify the calculation, both fuel and empty weights can be expressed as fractions of the total takeoff weight, that is, $\frac{W_f}{W_0}$ and $\frac{W_e}{W_0}$.

The total aircraft fuel includes the mission fuel as well as an allowance for reserve and trapped fuel. This reserve fuel allowance is usually 5% and an additional allowance of 1 % for trapped or unusable fuel.

$$W_0 = W_{crew} + W_{payload} + \frac{W_f}{W_0} + \frac{W_e}{W_0} \tag{5}$$

$$W_0 - (\frac{W_f}{W_0})W_0 - (\frac{W_e}{W_0})W_0 = W_{crew} + W_{payload} \tag{6}$$

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - \frac{W_f}{W_0} - \frac{W_e}{W_0}} \tag{7}$$

Now, $\frac{W_f}{W_0}$ and $\frac{W_e}{W_0}$ are estimated to determine W_0.

$$\frac{W_f}{W_0} = 1.06(1 - \frac{W_x}{W_0}) \tag{8}$$

5.1 Mission Profile

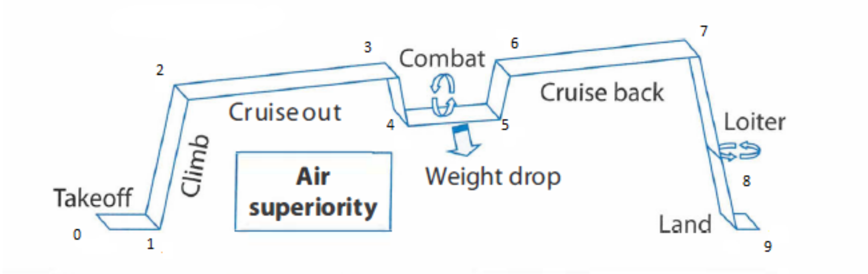


Figure 3: Mission profile

Maximum range estimated as 1698.75.
For propeller aircraft
Cruise, $(\frac{L}{D})_{max} = 14$ and loiter, $(\frac{L}{D})_{max} = 0.866X14 = 12.124$

5.2 Mission-Segment Weight Fractions

1. Warm-up and takeoff, $\frac{W_1}{W_0} = 0.97$
2. Climb, $\frac{W_2}{W_1} = 0.985$
3. Cruise out, $R = 1698.75/2 = 849.375$
   $\eta_{prop} = 0.85$
   c = 0.5 lb/hph = 0.0000002525
   $\frac{W_2}{W_3} = e^{(\frac{Rc}{\eta(\frac{L}{D})})}$ (From Breguet range equation)

   $\frac{W_3}{W_2} = 0.9821$
4. Descend, $\frac{W_4}{W_3} = 0.99$
5. Combat, $\frac{W_5}{W_4} = 0.954$
6. Ascend, $\frac{W_6}{W_5} = 0.99$
7. Cruise back (same as 3), $\frac{W_7}{W_6} = 0.9821$
8. Loiter, Endurance = 3600 sec
   c = 0.6 lb/hph = 0.0000003030
   $(\frac{L}{D})_{max} = 12.124$

   $\frac{W_7}{W_8} = e^{(\frac{Ec}{\frac{L}{D}})}$

   $\frac{W_8}{W_7} = 0.9139$
9. Land, $\frac{W_9}{W_8} = 0.995$

From the above estimated weight fractions, the following are calculated.

$$\frac{W_9}{W_0} = (0.97)(0.985)(0.9821)(0.99)(0.954)(0.99)(0.9821)(0.9139)(0.995) = 0.783640052 \tag{9}$$

$$\frac{W_f}{W_0} = 1.06(1 - 0.783640052) = 0.2293415449 \tag{10}$$

$$\frac{W_e}{W_0} = 0.92W_0^{-0.05} \tag{11}$$

$$W_0 = \frac{80 + 1500}{1 - 0.2293415449 - \frac{W_e}{W_0}} \tag{12}$$

W_0 (guess)	W_e/W_0	W_0 calculated
10000	0.5804807569	8308.019369
8000	0.5869935484	8602.62327
8500	0.5852169294	8520.205999
8520	0.5851481652	8517.047766
8517	0.585158469	8517.520856

Gross takeoff weight, $W_0 = 8517$ kgs
Empty weight, $W_e = 1953.421392$
Fuel weight, $W_f = 4984.453205$

6 Power-plant Selection

6.1 Thrust Required calculation

Thrust required is calculated as:

$$T_R = \frac{W}{\frac{C_L}{C_D}} \tag{13}$$

where W is taken as the take-off weight, and C_L and C_D are calculated as:

$$C_L = \frac{W}{\frac{1}{2}\rho V^2 S}, \tag{14}$$

$$C_D = C_D 0 + KCL^2 \tag{15}$$

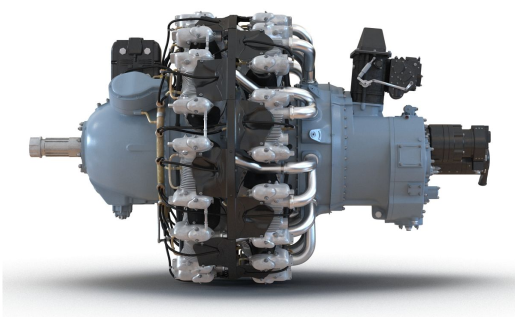
Now calculating the thrust required using above equations when the aircraft is cruising at a height of 9000m above sea level,
e = 0.7,
ρ = 0.467 kg/m^3,
$M_c ruise$ = 0.35,
$V_c ruise$ = 115.5 m/s

At these conditions, we get:
C_L = 0.1095020046,
C_D = 0.001182541433,

And finally, T_R = 91.98 N

6.2 Engine selection

According to the requirements and historical data, we selected pratt and whitney r2800 radial engine as our powerplant.



7 Tail sizing

7.1 Wing Configuration and Tail Selection

7.1.1 Introduction

Here we study the selection of wings, and calculation of wing design parameter.

7.1.2 Wing Selection

After the final weight estimation of the aircraft and the airfoil selection,, the primary components of the aircraft to be designed is the wing configuration and stable tail design .The first step towards designing the wing is the thickness estimation. The Position of the Wing(low Wing Dihedral The location of the wing in the fuselage (along with the vertical axis) is very important. Each configuration (ow, high, and mid) has its own advantages, but in this design, the low wing with dihedral offers significant advantages.

7.2 Advantages of choosing low wing over mid and high in our configuration

1. Center of Gravity and Stability: Low-wing configurations tend to position the wings closer to the aircraft's center of gravity, contributing to improved lateral stability, especially during maneuvers and changes in pitch.

2. Ease of Access and Maintenance: Low-wing designs often provide more accessible and convenient maintenance access to engines and other critical components, simplifying inspection and repair procedures for propeller-driven aircraft.

3. Ground Effect: Low-wing aircraft benefit from ground effect during takeoff and landing, enhancing lift and stability close to the ground. This can contribute to shorter takeoff and landing distances.

4. Visibility: The low-wing design typically offers better visibility over the nose and a clearer upward view for pilots, facilitating better situational awareness during flight, especially in the landing phase.

5. Fuselage Integration: Placing the wings lower on the fuselage allows for streamlined integration of the wings and the fuselage, reducing drag and potentially improving overall aerodynamic efficiency.

6. Structural Rigidity: Low-wing configurations can contribute to increased structural rigidity, providing a more stable platform for mounting engines and minimizing wing flex during flight.

7. Fuel Transfer: In some low-wing designs, the aircraft's fuel system can take advantage of gravity to facilitate fuel transfer, potentially simplifying fuel management.

8. Ground Handling: Low-wing aircraft often exhibit more stable ground handling characteristics, making them easier to taxi and maneuver on the ground.

9. Aesthetic Considerations: The low-wing design can offer a sleek and modern appearance, which may be preferred in certain aircraft categories, such as high-performance or executive propeller-driven planes.

10. Common Design Choice: Many popular propeller-driven aircraft models, including training and general aviation aircraft, adopt the low-wing configuration, highlighting its widespread acceptance and success in various applications.

7.3 Dihedral effect

The dihedral effect is the aerodynamic phenomenon resulting from the upward angle of an aircraft's wings from the horizontal axis.

1. Lateral Stability: Dihedral enhances lateral stability by creating a restoring force when the aircraft experiences a disturbance, such as a lateral gust or roll.

2. Rolling Motion: When subjected to a lateral force, the dihedral effect causes the aircraft to roll in the opposite direction of the disturbance.

3. Aerodynamic Center Shift: Dihedral influences the position of the aerodynamic center, contributing to the stability of the aircraft by affecting the distribution of lift along the wingspan.

4. Increased Stability in Turns: Aircraft with dihedral tend to exhibit more stable turns, as the rolling motion induced by banking helps maintain the aircraft's equilibrium.

5. Effect on Bank Angle: Dihedral reduces the tendency of an aircraft to roll excessively during turns, promoting smoother and more controlled maneuvers.

6. Common Design Feature: Many fixed-wing aircraft incorporate dihedral in their design to optimize lateral stability and overall flight characteristics. 7. Applicability Across Aircraft Types: Dihedral is utilized in various aircraft, from small general aviation planes to larger commercial and military aircraft, showcasing its versatility in enhancing flight stability.

7.4 Vertical Tail and Horizontal Tail:-

$$C_{mcg} = C_{macw} + C_{lw}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) - v_h \eta C_{l\alpha t}\left(i_t - \varepsilon_o + \alpha\left(1 - \frac{d\varepsilon}{d\alpha}\right) + \tau\delta_e\right)$$

Stick-fixed neutral point: $\frac{x_{NP}}{\bar{c}} = \frac{x_{ac}}{\bar{c}} + v_h \eta C_{l\alpha t}(1 - \frac{d\varepsilon}{d\alpha})$

Static Margin, SM = $\frac{x_{NP}}{\bar{c}} - \frac{x_{cg}}{\bar{c}}$

Downwash Variation, $\frac{d\varepsilon}{d\alpha} = \frac{C_{l\alpha w}}{\pi(AR)_w}$

$$\frac{dC_m}{dC_l} = -SM$$

$$C_{m\alpha} = -SM\, C_{l\alpha}$$

$$C_{mcg} = C_{mo} + C_{m\alpha}\alpha$$

$$C_{mo} = C_{macw} + C_{low}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) - v_h \eta C_{l\alpha t}(i_t - \varepsilon_o) \qquad \text{----------1}$$

$$(C_{m\alpha})_{stick-fixed} = C_{l\alpha w}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) - v_h \eta C_{l\alpha t}(1 - \frac{d\varepsilon}{d\alpha}) \qquad \text{----------2}$$

From equation 1, $v_h = \frac{C_{low}}{\eta C_{l\alpha t}(i_t - \varepsilon_o)}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) - (\frac{C_{mo} - C_{macw}}{\eta C_{l\alpha t}(i_t - \varepsilon_o)})$

From equation 2, $v_h = \frac{C_{l\alpha w}}{\eta C_{l\alpha t}(1 - \frac{d\varepsilon}{d\alpha})}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) - (\frac{(C_{m\alpha})_{stick-fixed}}{\eta C_{l\alpha t}\left(1 - \frac{d\varepsilon}{d\alpha}\right)})$

7.5 ii) For Vertical Tail: -

$$C_{n\beta\ desired} = 0.005\left(\frac{W}{b^2}\right)^{\frac{1}{2}}$$

Factors affecting $C_{n\beta}$ are –

- Wing Sweep
- Wing fuselage interaction

- Vertical tail size

$$C_{n\beta} = C_{n\beta\ wing} + C_{n\beta wf} + C_{n\beta vt}$$

$C_{n\beta\ wing}$ depends on wing sweep angle

Empirical relation between $C_{n\beta\ wf}$ and fuselage size and wing locations are used here

$C_{n\beta wf} = -k_n k_{Rl}\frac{S_{fs}l_f}{S_w b}$ (per degree)

Vertical tail contribution $C_{n\beta} = V_v\eta_v C_{L\alpha}(1+\frac{d\sigma}{d\beta})$, where

$$\eta_v\left(1+\frac{d\sigma}{d\beta}\right) = 0.724 + 3.06\frac{S_v/S}{1+cos\Lambda_{c/4w}} + 0.4\frac{z_w}{d} + 0.009(AR)_w$$

$$V_v = \frac{l_v S_v}{(Sb)}$$

$$C_{n\beta v} = C_{n\beta\ desired} - C_{n\beta w} - C_{n\beta wf}$$

So, the following quadratic equation is solved to obtain vertical tail area ratio:

$$\left(3.06\frac{l_v/b}{1+cos\Lambda_{c/4w}}\right)\left(\frac{S_v}{S}\right)^2 + \left(0.724 + 0.4\frac{z_w}{d} + 0.009AR_w\right)\left(\frac{l_v}{b}\right)\left(\frac{S_v}{S}\right) - \left(C_{n\beta\ desired} - C_{n\beta w} - C_{n\beta wf}\right) = 0$$

7.6 TAIL ARM OPTIMIZATION:

The designer should always try to minimize drag. The largest part of drag during cruise is caused by skin friction. The method shown here minimizes the wetted area of the tail boom, HT,and VT in a manner that will result in a stable aircraft. Assumptions compatible with fighter aircraft design:

- Aspect ratio of horizontal tail: 3.0
- Aspect ratio of vertical tail: 1.0
- Tail volume ratio of vertical tail: 0.06
- Tail volume ratio of horizontal tail: 0.4
- Taper ratio of horizontal tail:

$$Taperofratiowing:$$

Sailplane	0.5	0.02
Homebuilt	0.5	0.04
General aviation single engine	0.7	0.04
General aviation twin engine	0.8	0.07
Agriculture aircraft	0.5	0.04
Twin turboprop	0.9	0.08
Flying boat	0.7	0.06
Jet trainer	0.7	0.06
Jet fighter	0.4	0.07
Military cargo/bomber	1	0.08
Jet transport	1	0.09

Table 2: Tail Volume Coefficient for Different Aircraft Types

$$V_{ht} = \frac{L_{ht} \cdot S_{ht}}{S_w \cdot C_w} \tag{16}$$

$$V_{vt} = \frac{L_{vt} \cdot S_{vt}}{S_w \cdot b_w} \tag{17}$$

$$S_{ht} = B_{ht} \cdot C_{ht} \tag{18}$$

$$S_{vt} = B_{vt} \cdot C_{vt} \tag{19}$$

Parameter	Value
Cmacw	2.57m
Cmacht	2.46m
Sht	9.05 m^2
Bht	3.75m
Bw	11.8m
Lvt	3.956m
Sw	27.317 m^2
Lf	9.2m
Lht	4.259m

Table 3: optimised parameters obtained

Lt=4.259m
length of fuselage=9.2m
aspect ratio of horizontal tail=1.552
aspect ratio of wing=5.097
estimating Volume ratio of horizontal tail Vht=0.4, Sht=9.05
estimating Volume ratio of vertical tail Vvt=0.06, Svt=3.72

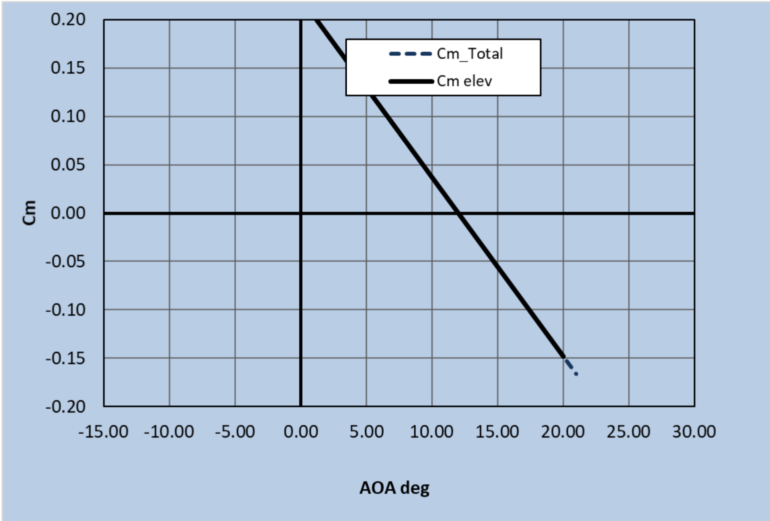


Figure 4: Cm vs AOA

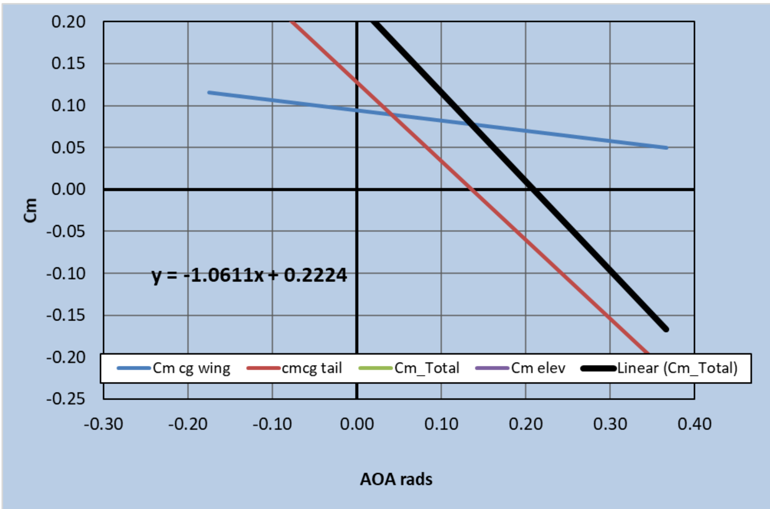


Figure 5: cm vs AOA

7.7 TAIL DESIGN REQUIREMENT FOR SATISFACTORY SPIN RECOVERY

Parameter	Value
Density	1.2
I_x	0.015
I_y	0.001
Mass (m)	1.5
Wingspan (b)	1
Wing Area (S)	0.148
$\frac{I_x - I_y}{mb^2}$	93.33
$u = \frac{m}{\text{density} \cdot S \cdot b}$	8.445946
L_1	0.49
L	0.51
L_2	0.51
R_1	0
R_2	0.001
F	0.01
L_1	0.5
L	0.51
R_1	0.001
F	0.02

Table 4: TAIL DESIGN REQUIREMENT FOR STISFACTORY SPIN RECOVERY Bvt=4.259m

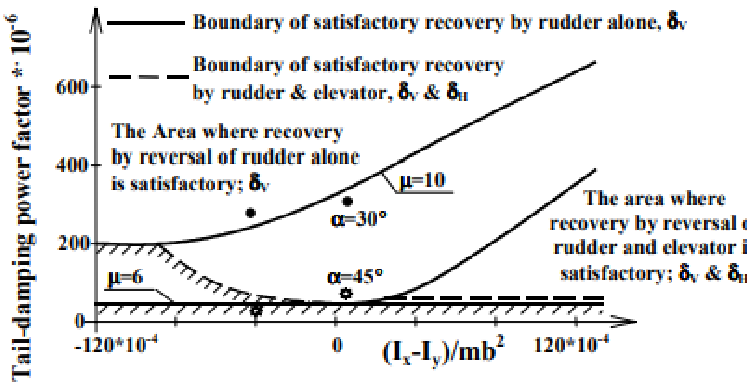
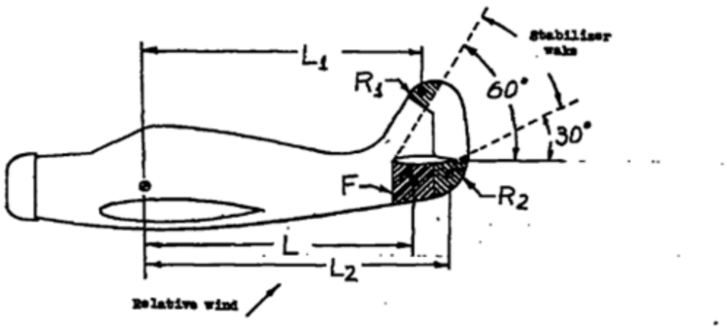
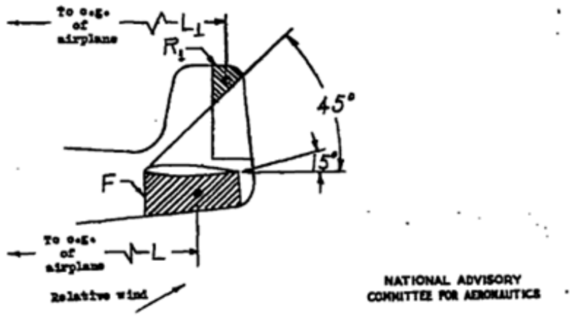


*Fig.8. Areas of satisfactory and unsatisfactory spin recoveries for light military airplanes [1] (for I-23 airplane the relative density μ= 10÷12. A symbolic location of I-23 airplane at the above figure is marked for two different angles of attack: • - for angle of attack α_h = 30° and * - for angle of attack α_h = 45°)*

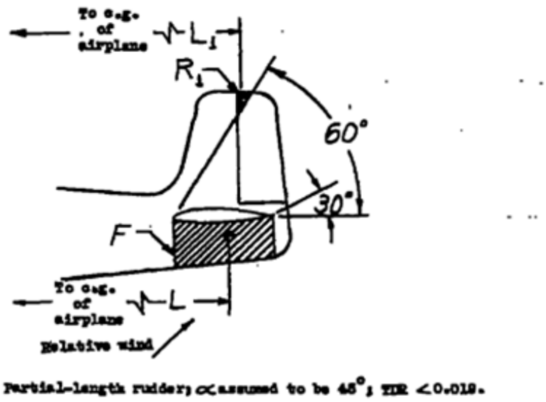
$$TDPF = \frac{FL^2}{s\left(\frac{b}{2}\right)^2} \cdot \frac{(R1L1 + R2L2)}{s\left(\frac{b}{2}\right)} \tag{20}$$



(a) Full-length rudder; α assumed to be 45°.



(c) Partial-length rudder; α assumed to be 30°; TDR >0.019.



(b) Partial-length rudder; α assumed to be 45°; TDR <0.019.

8 Fuselage and Landing Gear Design

8.1 Fuselage design

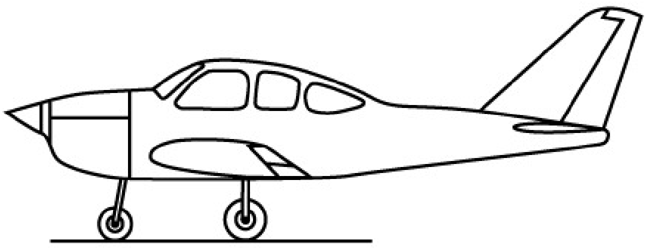
The following data is selected using the historical data of fighter aircrafts which use propellers:

Fuselage length = 9.21 m

Fuselage diameter = 4.5 m

8.2 Landing gear

The tricycle arrangement has a single nose wheel in the front, and two or more main wheels slightly aft of the center of gravity. Tricycle gear aircraft are the easiest for takeoff, landing and taxiing, and consequently the configuration is the most widely used on aircraft.



tricycle landing gear

9 Drag Estimation

Drag estimation is done as following:

$$C_L = \frac{W}{\frac{1}{2}\rho V^2 S}, \tag{21}$$

$$D_{min} = C_{Dmin}\frac{1}{2}\rho V^2 S, \tag{22}$$

Induced drag,

$$Di = KC_L^2\frac{1}{2}\rho V^2 S, \tag{23}$$

Total drag,

$$D = Di + D_{min}, \tag{24}$$

Propulsive power (PP),

$$P = DV, \tag{25}$$

Lift to Drag ratio (L/D),

$$L/D = W_{TO}/D, \tag{26}$$

V (m/s)	C_L	$D_{\min}$	D_i	D (N)	Propulsive Power (W)	L/D
115.5	0.4281	1000.16	3185.79	4185.95	483477.34	2.0348
10	57.1153	7.4973	424992.90	425000.40	4250003.98	0.0200
20	14.2788	29.9891	106248.23	106278.21	2125564.28	0.0801
30	6.3461	67.4756	47221.43	47288.91	1418667.27	0.1801
40	3.5697	119.9566	26562.06	26682.01	1067280.51	0.3192
50	2.2846	187.4322	16999.72	17187.15	859357.41	0.4956
60	1.5865	269.9023	11805.36	12075.26	724515.64	0.7054
70	1.1656	367.3671	8673.32	9040.69	632848.41	0.9421
80	0.8924	479.8264	6640.51	7120.34	569627.24	1.1962
90	0.7051	607.2803	5246.83	5854.11	526869.56	1.4550
100	0.5712	749.7287	4249.93	4999.66	499965.77	1.7036
110	0.4720	907.1718	3512.34	4419.51	486146.08	1.9273
120	0.3966	1079.6094	2951.34	4030.95	483713.87	2.1130
130	0.3380	1267.0415	2514.75	3781.79	491633.02	2.2522
140	0.2914	1469.4683	2168.33	3637.80	509291.92	2.3414
150	0.2538	1686.8896	1888.86	3575.75	536362.04	2.3820
160	0.2231	1919.3055	1660.13	3579.43	572709.45	2.3796
170	0.1976	2166.7160	1470.56	3637.28	618337.55	2.3417
180	0.1763	2429.1211	1311.71	3740.83	673348.96	2.2769
190	0.1582	2706.5207	1177.27	3883.79	737919.41	2.1931
200	0.1428	2998.9149	1062.48	4061.40	812279.43	2.0972
210	0.1295	3306.3037	963.70	4270.01	896701.34	1.9947
220	0.1180	3628.6870	878.08	4506.77	991489.74	1.8899
230	0.1080	3966.0650	803.39	4769.45	1096974.46	1.7858
240	0.0992	4318.4374	737.83	5056.27	1213505.36	1.6845
250	0.0914	4685.8045	679.99	5365.79	1341448.29	1.5874

Table 5: Performance Data

Parameter	Value
$C_{D_{\min}}$	0.00513
ρ	1.07
e	0.7
K	0.0891435442

Table 6: Aerodynamic Coefficients and Values

Parameter	Value
$C_{D_{\min}}$	0.00513
C_D	0.02147056065
T_R	427.1378552

Table 7: Aerodynamic Constants

10 Propeller design

Required power, P = 11000 W
Cruise Velocity = 115.5 m/s, $\rho = 0.467$ kg/m^3, A three bladed propeller with an rpm of 6000 is considered.
RPS,

$$n = 6000/60 = 100 \tag{27}$$

Cs is calculated as,

$$C_s = V\sqrt[5]{(\frac{\rho}{Pn^2})} = 115.5\sqrt[5]{(\frac{0.467}{11000X100^2})} = 2.44 \tag{28}$$

Source: NACA Report 640

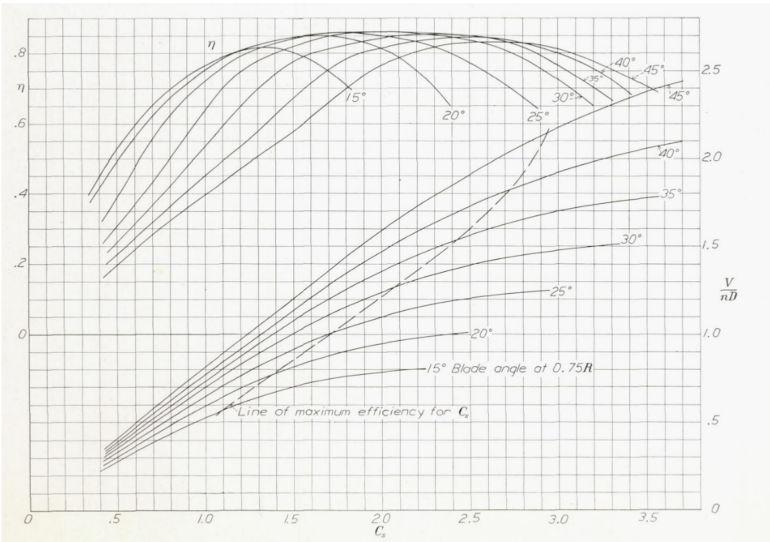


FIGURE 10.—Design chart for propeller 5868-9, Clark Y section, 3 blades.

At the calculated C_S value it is observed that the efficiency of blade angle 35° is coinciding with the line of maximum efficiency for C_S. Advance ratio, J is about 1.55 .

$$J = \frac{V}{nD} = 1.55 \tag{29}$$

$$D = \frac{V}{nJ} = \frac{115.5}{100X1.55} = 0.745m \tag{30}$$

11 FINAL DESIGN

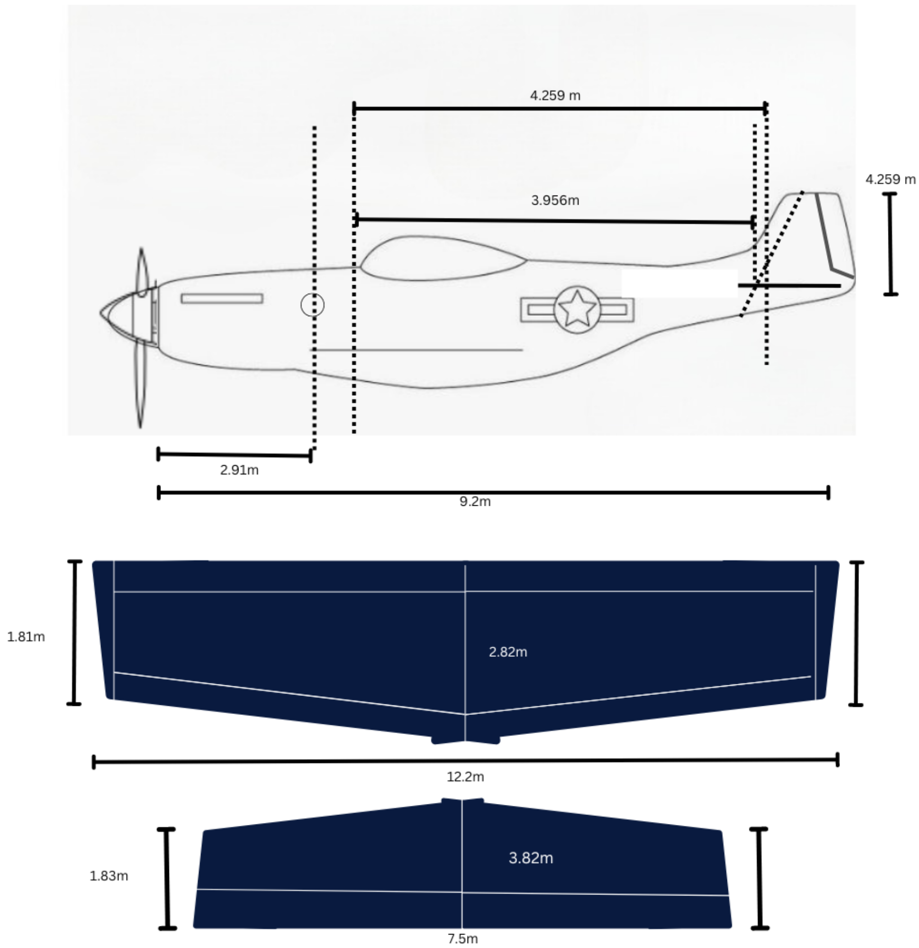


Figure 6: Dimensions