DESIGN OF A 1.2Kg PAYLOAD DELIVERY UAV

BY

IDRIS HABEEB BABATUNDE

AS

ASSESSMENT FOR AIRCRAFT DESIGN

AAE 401

DEPARTMENT OF AERONAUTICAL AND ASTRONAUTICAL ENGINEERING

KWARA STATE UNIVERSITY

INSTRUCTOR

DR. YOMI OBIDI

SUBMISSION DATE: JUNE 15, 2017

Table of Contents

ABSTRACT

The design approach used (in this work) is basically the system engineering approach. The detailed description of the design processes and stages involved in this work were presented based on the steps followed.

First pass design was done based on the information acquired from the mission profile, customer design specifications and design objectives. The customer design specifications were incorporated into the conceptual design which provided guidelines through the whole design processes where various adequate equations were applied to obtain the desired parameters. Reasonable guesses were made and various tables were consulted in some instances where there were no available befitting mathematical equations to get the actual required values.

In some instances, the weight of the aircraft is assumed to be constant because it will be using a battery and the payload weight (after dropping it) has no significant change in the weight. Center of gravity (CG) position is also assumed to be constant because the store (for the payload) is situated directly under the CG position which led apparently, to no change in the CG position after releasing the payload.

During the design processes, some parameters of already designed components were reviewed to satisfy the requirements of other components. Federal Aviation Regulations (FAR) were followed and some rules of thumb were applied. Lots of materials resources were also considered to complete this work as all the required information can't be obtained from a single materials.

CHAPTER 1

INTRODUCTION

This design is a RC (remote controlled) UAV whose range depends on both battery and transmitter frequency (to maintain transmitter-receiver communication for control) and the endurance depends solely on the battery. The design order is based on the extent of the required parameters which is the reason why the wing was designed immediately after the initial sizing and the tail plane was designed after the wing. The control surfaces were deemed to be designed after all other aircraft components have been designed because of the complexity of the required information.

Aircraft design is an iterative process which requires various parameters to be altered, over and over again, to satisfy various requirements. To make the iteration easy, accurate and to save time, the equations are transformed into MATLAB codes. In calculating the actual lift generated by lifting surfaces, lifting line theory was employed and all the steps involved in it were embedded in a MATLAB program.

Generally, in design, computer aided design (CAD) software are used. The CAD software used for this design is Solidworks. After designing each component, 3D designs of each of the components was produced and were assembled at the end of the whole design process.

CHAPTER 2

MISSION PROFILE

The aircraft will takeoff within a ground roll of about 8m, climb to an altitude of 15m, within 30 seconds, cruise for some time to cover a distance around 100m, then, loiter around for about 10 seconds to drop a light payload (of about 10g of mass). The aircraft will then return and land with a maximum ground roll of 12m. The flight time will be around 5 minutes and the takeoff mass shouldn't be more than 1.2kg. The stall speed should not exceed 8m/s and maximum speed should not be less than 15m/s.

From above, the following were deduced:

1. S_{TO} = 8m
2. $S_{landing}$ = 12m
3. Cruise altitude = 15m
4. V_{stall} = 8m/s
5. V_{max} = 15m/s
6. ROC = 15m/30s = 0.5m/s
7. Endurance = 10s
8. Range = about 150m
9. Flight time = 5min
10. M_{TO} = 1.2kg

DESIGN OBJECTIVES

Listed below, are the design objectives commonly considered in designing an aircraft. They were used as guidelines during the design process. Because it is impossible to satisfy all these objectives at once, they were arranged based on their importance to this design.

1. Low cost
2. Control and stability
3. Light weight
4. Ease of manufacture
5. Aerodynamic performance
6. Structural Integrity
7. Heavy payload

CHAPTER 3

CONCEPTUAL DESIGN

Proposed aircraft: 1.2kg, single engine, prop-driven light-weight UAV

Intended use: object dropping (alternative use is surveillance)

Numbers of seats: no seat

Range: about 150m

Rate of climb: 0.5m/s

Fuel capacity: NIL

Engine selection: Brushless electric motor.

Velocities

1. Max speed: 15m/s
2. Cruise speed: 11.538m/s (i.e. $V_{max}/1.3$)
3. Stall speed: 8m/s
4. Landing speed: 10.4m/s (i.e. $1.3V_{stall}$)

Takeoff and Landing Requirement

1. Takeoff ground roll: 8m
2. Landing ground roll: 12m

Wing Configuration

1. Number of wings: monoplane
2. Wing location: High wing
3. Wing type: Rectangular (because of weight, cost and ease of manufacture)
4. High lift device: Plain flap
5. Shape: Fixed shape
6. Structural configuration: Cantilever

Tail Configuration

1. Tail location: Aft tail
2. Horizontal and vertical tails: conventional tail
3. Attachment: Fixed

Landing Gear Configuration

1. Landing gear mechanism: Fixed

2. Landing gear type: Tricycle
3. Runway: Land based

Manufacturing-Related Item Configuration

1. Material for manufacturing: Foam (Styrofoam, foam board or any other)
2. Assembly technique: use of glue, screws and bolts & nuts

Subsystems Configuration

1. Primary control surfaces: conventional (i.e., elevator, aileron and rudder)
2. Secondary control surface: High lift device (flap)
3. Power transmission: fly-by-wire
4. Fuel tank: inside fuselage (battery)
5. Storage: light object (payload of about 10g)

CHAPTER 4

FIRST PASS DESIGN

1. Payload mass: 10g
2. Gross weight: 11.772N (i.e. 1.2Kg)
3. Wing loading

$$\frac{W_g}{S_e} = \frac{\rho_o V_{stall}^2}{2} C_{L_{max}} \tag{4.1}$$

Where ρ_o = 1.225kg/m^3, V_{stall} = 8m/s and C_{Lmax} was selected to be 1.6

$$\frac{W_g}{S_e} = \frac{1.225 \times 8^2}{2} 1.6 = 62.72N/m^2 \tag{4.2}$$

$$V_{landing} = 1.3V_{stall} = 1.3 \times 8 = 10.4m/s$$

4. Effective wing area,

$$S_e = \frac{W_g}{\left(\frac{W_g}{S_e}\right)} = \frac{11.772}{62.72} = 0.1878m^2 \tag{4.3}$$

5. Dynamic pressure

$$q = \frac{\rho \sigma V_{cr}^2}{2} \tag{4.4}$$

Where air density and relative density (at cruising altitude (15m)), are 1.2232kg/m^3 and 0.9986 respectively. Cruising speed is 11.538m/s

$$q = \frac{1.2232 * 0.9986 * 11.538^2}{2} = 81.31N/m^2$$

6. Coefficient of lift at cruise

$$C_L = \frac{W_g/S_e}{q} = \frac{62.72}{81.31} = 0.7714 \tag{4.5}$$

$$C_L = \frac{1}{F} \therefore F = \frac{1}{C_L} = \frac{1}{0.7714} = 1.2964 \tag{4.6}$$

7. Coefficient of drag

$$C_D = C_{D_P} + \frac{C_L^2}{\pi e AR} \tag{4.7}$$

Where CL = 0.7714, CD_P was selected to be 0.025, e to be 0.825 and AR to be 7

$$C_D = 0.025 + \frac{0.7714}{\pi \times 0.825 \times 7} = 0.0578$$

8. $\frac{L}{D} = \frac{C_L}{C_D} = \frac{0.7714}{0.0578} = 13.346$ (4.8)
9. $D = W_g \cdot \frac{C_D}{C_L} = 11.772 \times \frac{1}{13.346} = 0.8821N$ (4.9)
10. Thrust horsepower required at cruise altitude

$$THP = \frac{DV_{cr}}{745.7} = \frac{0.8821 * 11.538}{745.7} = 0.01365hp\ (10.177W) \tag{4.10}$$

11. Maximum brake horsepower (BHP)

$$BHP_{cr} = \frac{THP}{\eta_{prop}}/0.7 \tag{4.11}$$

η_{prop} was selected to be 0.85

$$BHP_{cr} = \frac{0.01365}{0.85}/0.7 = 0.0229hp\ (17.1W) \tag{4.12}$$

$$\text{BHP at sea level } = BHP_{cr}\left(\frac{\rho_o}{\rho}\right)^{0.96} = 0.0229\left(\frac{1.225}{1.2232}\right)^{0.96} = 0.023hp \tag{4.13}$$

12. Optimum cruise lift coefficient

$$C_L = \sqrt{\pi \cdot e \cdot AR \cdot C_{D_P}} = \sqrt{\pi \cdot 0.825 \cdot 7 \cdot 0.025} \tag{4.14}$$

$$= 0.6735$$

$$\text{Optimum } V_{cr} = \sqrt{\frac{W_g}{\frac{\rho}{2}\sigma \cdot C_{L_{cr_{opt}}} \cdot S_e \cdot F}} = \sqrt{\frac{11.772}{\frac{1.2232}{2} \times 0.6735 \times 0.1878 \times 1.2964}} = 10.834m/s \tag{4.15}$$

13. Power required to satisfy the required ROC:

$$\left(\frac{W}{P}\right) = \frac{1}{\frac{ROC}{\eta_p} + \sqrt{\frac{2}{\rho\sqrt{\frac{3C_{D_O}}{K}}}\left(\frac{W}{S}\right) \cdot \left(\frac{1.155}{(L/D)\eta_p}\right)}} \tag{4.16}$$

Where W/P is the power loading, ROC is the rate of climb, η_p is the propeller efficiency, ρ is the air density at sea level, CDo is the zero lift drag coefficient, K is the induced drag factor, (W/S) is the wing loading, and (L/D) is the lift to drag ratio.

$$K = \frac{1}{\pi \cdot e \cdot AR} \tag{4.17}$$

$$K = \frac{1}{3.142 \times 0.825 \times 7} = 0.0551$$

ROC has been stated earlier to 0.5m/s and CDo was selected to be 0.035.

$$\left(\frac{W}{P}\right) = \frac{1}{\frac{0.5}{0.85} + \sqrt{\frac{2}{1.225 \times \sqrt{\frac{3 \times 0.035}{0.0551}}} \times 62.72 \times \left(\frac{1.155}{13.346 \times 0.85}\right)}} = 0.6636N/W$$

$$P = \frac{W}{\left(\frac{W}{P}\right)} \tag{4.18}$$

$$P = \frac{11.772}{0.6636} = 17.7W$$

Therefore, the power required is 17.7W

FIRST PASS FINAL RESULT

1. Gross weight, Wg = 11.772N
2. Wing loading = 62.7N/m^2
3. Stall velocity = 8m/s
4. Wing area = 0.1878m^2

5. Dynamic pressure, q = 81.3N/m^2
6. Total drag coefficient, C_D = 0.0578
7. Lift to drag ratio = 13.346
8. Drag, D = 0.8821N
9. Thrust horsepower (THP) = 0.0136hp (10.177W)
10. Brake horsepower at cruise altitude $(BHP)_{cr}$ = 0.0229 (17.1W)
11. BHP at sea level = 0.023hp
12. Optimum cruise lift coefficient = 0.6735
13. Optimum cruise speed = 10.834m/s
14. ROC = 0.5m/s

SECOND PASS DESIGN

Second pass design wasn't considered (because it involves obtaining a more accurate gross weight); the gross weight used during first pass design was stated as a requirement. Fuel and oil weights aren't needed because the aircraft will be using a RC electric motor and battery. Second pass design also optimizes the wing area, drag and brake horsepower at sea level by multiplying them by a conversion factor (calculated using a new gross weight). Since the gross weight won't be altered, these and other parameters obtained from first pass design will remain the same.

CHAPTER 5

WING AIRFOIL SELECTION

PRACTICAL STEPS IN SELECTING AN AIRFOIL

1. Aircraft weight: 11.772N
2. Aircraft ideal cruise lift coefficient (C_{LC}):

$$C_{L_C} = \frac{2W_{avg}}{\rho V_C^2 S} \tag{5.1}$$

$$C_{L_C} = \frac{2 \times 11.772}{1.2232 \times 10.834^2 \times 0.1878} = 0.8732$$

3. Wing cruise lift coefficient ($C_{L_{C_W}}$).

$$C_{L_{C_W}} = \frac{C_{L_C}}{0.95} \tag{5.2}$$

$$C_{L_{C_W}} = \frac{0.8732}{0.95} = 0.9191$$

4. Wing airfoil ideal lift coefficient (C_{li}):

$$C_{l_i} = \frac{C_{L_{C_W}}}{0.9} \tag{5.3}$$

$$C_{l_i} = \frac{0.9191}{0.95} = 1.0212$$

5. Aircraft maximum lift coefficient (C_{Lmax}):

$$C_{L_{max}} = \frac{2W_{TO}}{\rho_0 V_s^2 S} \tag{5.4}$$

Where V_s is the aircraft stall speed, ρ_o is the air density at sea level, and W_{TO} is the aircraft maximum take-off weight.

$$C_{L_{max}} = \frac{2 \times 11.772}{1.225 \times 8^2 \times 0.1878} = 1.6$$

6. Wing maximum lift coefficient ($C_{L_{\max\_w}}$).

$$C_{L_{\max\_w}} = \frac{C_{L_{max}}}{0.95} \tag{5.5}$$

$$C_{L_{\max\_w}} = \frac{1.6}{0.95} = 1.6832$$

7. Airfoil gross maximum lift coefficient (Clmax_gross):

$$C_{l_{\max\_gross}} = \frac{C_{L_{\max\_w}}}{0.9} \tag{5.6}$$

$$C_{l_{\max\_gross}} = \frac{1.6832}{0.9} = 1.8703$$

8. High lift device (HLD): Plain Flap.
9. HLD contribution to the wing maximum lift coefficient (ΔC_{lHLD}): 0.55.

10. Wing airfoil net maximum lift coefficient (C_{lmax}):

$$C_{l_{\max}} = C_{l_{\max\_gross}} - \Delta C_{l_{\mathrm{HLD}}} = 1.3203 \quad (5.7)$$

11. Reynolds number: 100,000 (using cruise speed and chord length of 10.834m/s and 0.16m respectively)

The following airfoils were considered

1. SD7037
2. E193
3. S9000
4. Clark Y

At Reynolds number of 100000, the values for various parameters (as shown in the table below) were obtained for the airfoils. These values were obtained with the aid of system software (XFOIL) and a website (www.airfoiltools.com)

AIRFOIL PARAMETERS	AIRFOILS			
	SD7037	E193	S9000	CLARK Y
Cl_{max}	1.3364	1.283	1.2702	1.3663
$Alfa_s$ (deg)	12	12	11	12
$Alfa_0$ (deg)	-1.7679	-3.2868	-1.2702	-2.2492
C_{l0}	0.3046	0.4068	0.2071	0.3713
C_{dmin}	0.01519	0.01452	0.01311	0.01722
C_{li}	0.04	0.24	0.08	0.8
$(C_l/C_d)_{max}$	55.2	58.7	53.3	53
Cl_{alfa}	6.0628	6.1183	6.0628	6.1852
C_{mmin}	-0.084	-0.096	-0.073	-0.092
S.Q.	Moderate	Abrupt	Moderate	Docile
Max. thickness (unit)	0.092	0.102	0.09	0.1171

Table 5.1: Airfoil data

From these, Clark Y airfoil was selected.

WING DESIGN

STEP 1: Wing incidence

The selected airfoil ideal lift coefficient is 0.8. Since at ideal lift coefficient, minimum drag is experienced, the angle of attack corresponding to the ideal lift coefficient was set as the wing incidence (or wing setting angle). To obtain this value, a reference was made to the Cl-α curve (through the use of XFOIL).

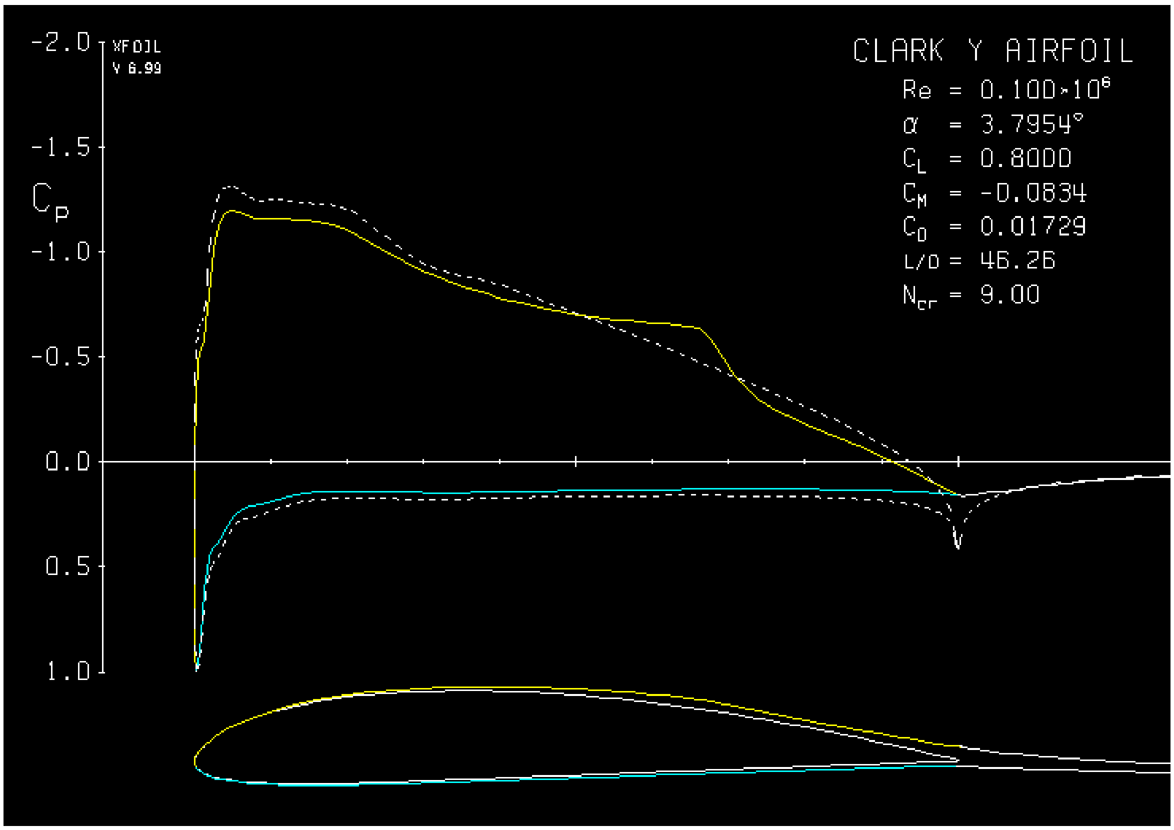


Fig 5.1: wing setting angle from XFOIL

Therefore, wing setting angle (α_{set}) or wing incidence (i_w) is equal to **3.7954 deg.**

STEP 2: ASPECT RATIO

Considering the various advantages and disadvantages of high and low values, aspect ratio was selected to be **7**.

STEP 3: TAPER RATIO

The effects of taper ratio was carefully considered but for some reasons (such as ease of manufacturing, cost and to keep the design simple) the wing was left straight **i.e. no taper**.

STEP 4: SWEEP ANGLE:

It was stated (in ref 1) that If the aircraft maximum speed is less than Mach 0.3 (the borderline to include the compressibility effect), no sweep angle is recommended for the wing, since its disadvantages will negate all the improvement produced. For this reason and some others, **no sweep angle** was considered.

STEP 5: TWIST ANGLE

This design will use **no twist angle**. Though, twist angle have the advantages of avoiding tip stall before root stall and modification of the lift distribution to an elliptical one but it give rise to reduction in lift. In addition, the complexity of manufacturing, increase in weight as well cost required prevent its use.

STEP 6: DIHEDRAL ANGLE

No dihedral.

STEP 7: REQUIRED LIFT COEFFICIENT DURING CRUISE:

$$C_{L_C} = \frac{2W_{avg}}{\rho V_C^2 S} \tag{5.8}$$

$$C_{L_C} = \frac{2 \times 11.7720}{1.2232 \times 10.834^2 \times 0.1878} = 0.8732$$

STEP 8: REQUIRED AIRCRAFT TAKEOFF LIFT COEFFICIENT:

$$V_{TO} = kV_{stall} \tag{5.9}$$

Where k is 1.1 for fighter aircraft and 1.2 for general aviation (GA) aircraft

$$V_{TO} = 1.2 \times 8 = 9.6m/s$$

$$C_{L_{TO}} = \frac{2W_{TO}}{\rho V_{TO}^2 S} \tag{5.10}$$

$$C_{L_{TO}} = \frac{2 \times 11.7720}{1.225 \times 9.6^2 \times 0.1878} = 1.1105$$

STEP 9: WING INCIDENCE FROM LIFTING LINE THEORY TO SATISFY CRUISE C_L

A MATLAB code has been written (in Ref 1) for lifting line theory which divides the wing into sections to calculate the lifts generated per section. They were then summed up to obtain the total lift produced by the wing at that incidence. This MATLAB program has lift distribution curve incorporated in it.

The following were used in calculating lift coefficient at cruise

1. Wing area = 0.1878m^2
2. Aspect ratio = 7
3. Taper ratio = 1
4. Twist angle = 0
5. Wing setting angle = 3.7954deg
6. Airfoil lift curve slope = 6.1852rad^{-1}
7. Zero-lift angle of attack = -2.2492deg

MATLAB CODE

```
clc
clear
N = 9; % (number of segments - 1)
S = 0.1878; % m^2
AR = 7; % Aspect ratio
lambda = 1; % Taper ratio
alpha_twist = -1e-8; % Twist angle (deg)
i_w = 3.7954; % wing setting angle (deg)
a_2d = 6.1852; % Airfoil lift curve slope (1/rad)
alpha_0 = -2.2492; % zero-lift angle of attack (deg)
b = sqrt(AR*S) % wing span (m)
MAC = S/b % Mean Aerodynamic Chord (m)
Croot = (1.5*(1+lambda)*MAC)/(1+lambda+lambda^2); % root chord (m)
theta = pi/(2*N):pi/(2*N):pi/2;
alpha = i_w+alpha_twist:-alpha_twist/(N-1):i_w;
% segment's angle of attack
z = (b/2)*cos(theta);
c = Croot * (1 - (1-lambda)*cos(theta)); % Mean Aerodynamics Chord at each
segment (m)
mu = c * a_2d / (4 * b);
LHS = mu .* (alpha-alpha_0)/57.3; % Left Hand Side
% Solving N equations to find coefficients A(i):
for i=1:N
for j=1:N
B(i,j) = sin((2*j-1) * theta(i)) * (1 + (mu(i) * (2*j-1)) /sin(theta(i)));
end
end
A=B\transpose(LHS);
for i = 1:N
sum1(i) = 0;
sum2(i) = 0;
for j=1:N
sum1(i) = sum1(i) + (2*j-1) * A(j)*sin((2*j-1)*theta(i));
sum2(i) = sum2(i) + A(j)*sin((2*j-1)*theta(i));
end
end
CL = 4*b*sum2 ./ c;
CL1=[0 CL(1) CL(2) CL(3) CL(4) CL(5) CL(6) CL(7) CL(8) CL(9)];
y_s=[b/2 z(1) z(2) z(3) z(4) z(5) z(6) z(7) z(8) z(9)];
plot(y_s,CL1,'-o')
grid
```

```
title('Lift distribution')
xlabel('Semi-span location (m)')
ylabel ('Lift coefficient')
CL_wing = pi * AR * A(1)
```

With these, the lift coefficient produced by the wing is 0.4898. The wing incidence was then varied via trial and error to obtain the required lift coefficient at cruise. At a wing incidence of 8.53deg, the generated wing lift coefficient is 0.8735. Although, this satisfies cruise requirement (based of lift coefficient) but the incidence is too high and the design has to be reviewed.

The major aim of reviewing the design is to reduce the required cruise lift coefficient which can easily be done by increasing the cruise speed. Cruise speed is a function of maximum speed ($V_{cr} = V_{max}/1.3$). Therefore, maximum speed was adjusted to 19m/s. first pass design was then reviewed and the following are the new data:

1. Gross weight, Wg = 11.772N
2. Wing loading = 62.7N/m^2
3. Stall velocity = 8m/s
4. Cruise speed = 14.6154m/s
5. Maximum speed = 19m/s
6. Landing speed = 10.4m/s
7. Wing area = 0.1878m^2
8. Dynamic pressure, q = 130.4605N/m^2
9. Cruise lift coefficient = 0.4808
10. Total drag coefficient, C_D = 0.0377
11. Lift to drag ratio = 12.7394
12. Drag, D = 0.9241N
13. Thrust horsepower (THP) = 0.0181hp (13.497W)
14. Brake horsepower at cruise altitude $(BHP)_{cr}$ = 0.0304 (22.67W)
15. BHP at sea level = 0.0305hp (22.74W)
16. ROC = 0.5m/s

While reviewing airfoil selection criteria, the new required airfoil ideal lift coefficient is 0.5612. Reynolds number also changes to 150,000. At this Reynolds number, Clark Y airfoil has the following parameters:

1. Zero-lift angle of attack = -3.1264deg
2. Lift at zero AOA = 0.4345
3. Maximum Cl = 1.3769

4. Stall AOA = 13deg
5. Cd_{min} = 0.01244
6. Cl_i = 0.49
7. $(L/D)_{max}$ = 64.68
8. Cm_{min} = -0.099
9. Cl_α = 6.1852rad^{-1}

While reviewing wing design the following steps has these new results:

Step1: wing incidence equals 0.5327deg

Step 7: required lift coefficient during cruise is 0.48

With the use of the same MATLAB program, at a wing incidence of 2.8deg, the lift generated is 0.4808 and the lift distribution is as in the figure below. Therefore, the wing incidence was set to be **2.8deg**.

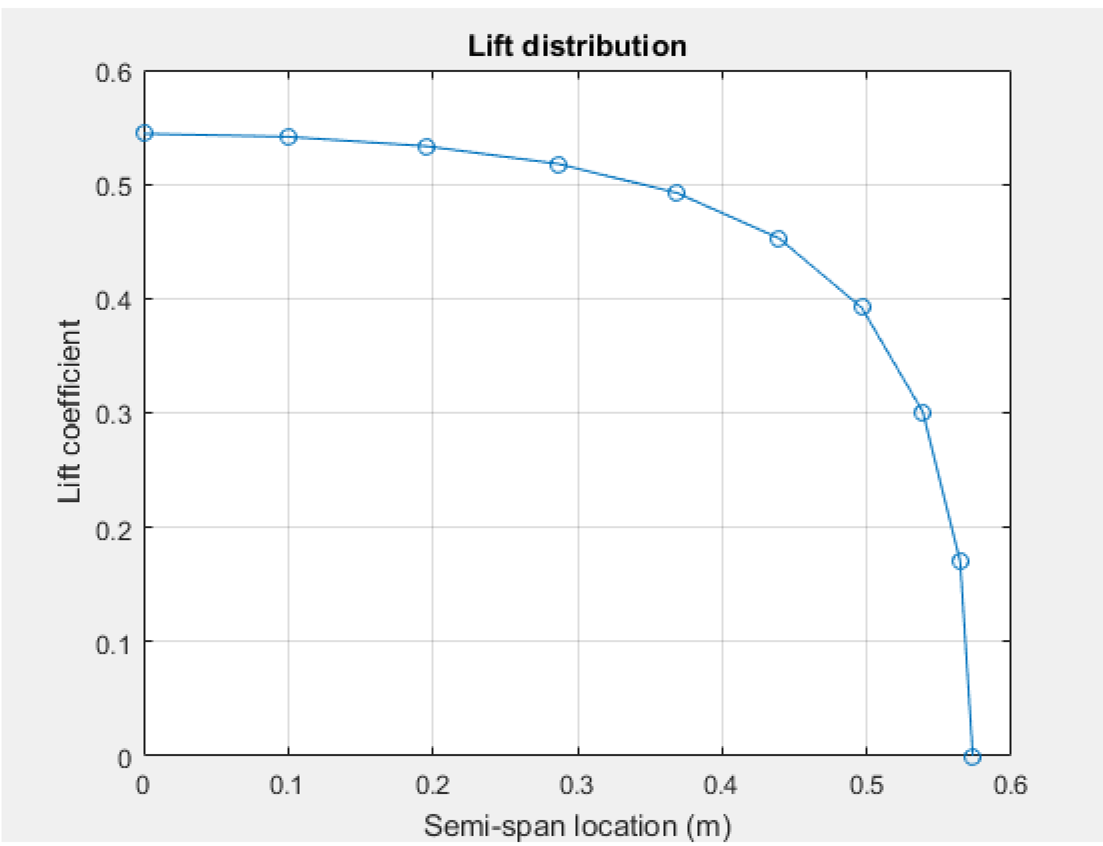


Fig 5.2: Lift distribution

Although, the lift distribution is not elliptical but can be worked with as it is close to elliptical lift distribution. To achieve an elliptical lift distribution, the twist angle will have to be altered (to around -1.5deg). Considering ease of manufacturing, light weight and cost; the twist angle was left unaltered.

STEP 10: FLAP DESIGN

In designing the flap, flap; span to wing span ratio (b_f/b); chord to wing chord ratio (C_f/C) and; maximum deflection (δ_{fmax}) were used. The latter two were used to determine the approximate increment in the wing zero-lift angle of attack (due to flap deflection) through the use of the following equation:

$$\Delta\alpha_o \approx -1.5 \cdot \frac{C_f}{C}\delta_f \tag{5.11}$$

Flap span to wing span ratio was used to determine the flap region across the span. These were incorporated into the MATLAB code for lifting line theory to determine the total lift generated by the wing (with flap down).

As an initial value, the following were used: C_f/C = 20%, b_f/b = 60% and δ_f = 13deg and α_{TO} = 9deg (wing angle of attack at takeoff)

With these, the total lift generated is 1.1863. When the wing angle of attack was set to 8.07deg, the lift generated by the wing is 1.1109. The wing angle of attack was reduced because of stall; the wing stalls earlier when flap is deflected.

MATLAB CODE WITH FLAP DEFLECTED

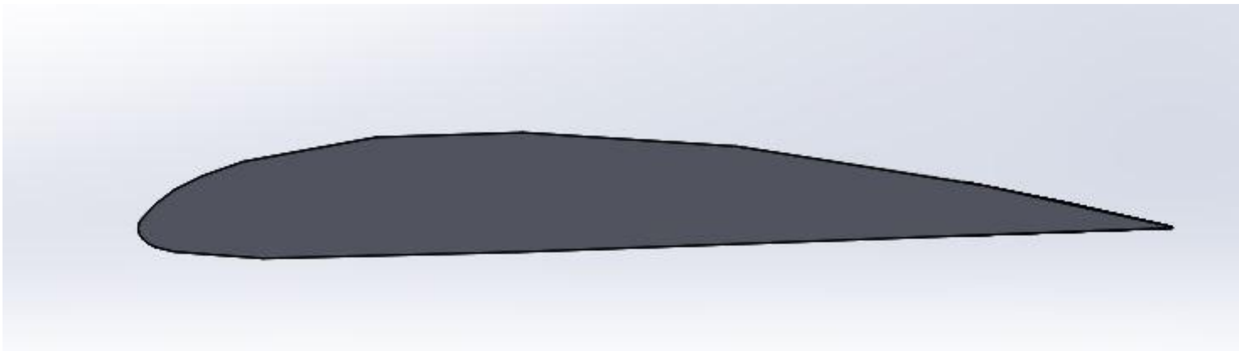
```
clc
clear
N = 9; % (number of segments-1)
S = 0.1878; % m^2
AR = 7; % Aspect ratio
lambda = 1; % Taper ratio
alpha_twist = -1e-10; % Twist angle (deg)
i_w = 8.07; % wing setting angle (deg)
a_2d = 6.1852; % lift curve slope (1/rad)
a_0 = -3.1264; % flap up zero-lift angle of attack (deg)
cf_c=0.2; %flap to wing chord ratio
df=13; %flap deflection(deg)
a_0_fd = 2*(-1.15*cf_c*df); % flap down zero-lift angle of attack (deg)
b = sqrt(AR*S); % wing span
bf_b=0.6; %flap-to-wing span ratio
MAC = S/b; % Mean Aerodynamic Chord
Croot = (1.5*(1+lambda)*MAC)/(1+lambda+lambda^2); % root chord
theta = pi/(2*N):pi/(2*N):pi/2;
alpha=i_w+alpha_twist:-alpha_twist/(N-1):i_w;
% segment’s angle of attack
for i=1:N
```

```
if (i/N)>(1-bf_b)
    alpha_0(i)=a_0_fd; %flap down zero lift AOA
else
alpha_0(i)=a_0; %flap up zero lift AOA
end
end
z = (b/2)*cos(theta);
c = Croot * (1 - (1-lambda)*cos(theta)); % MAC at each segment
mu = c * a_2d / (4 * b);
LHS = mu .* (alpha-alpha_0)/57.3; % Left Hand Side
% Solving N equations to find coefficients A(i):
for i=1:N
for j=1:N
B(i,j) = sin((2*j-1) * theta(i)) * (1 + (mu(i) * (2*j-1)) / sin(theta(i)));
end
end
A=B\transpose(LHS);
for i = 1:N
sum1(i) = 0;
sum2(i) = 0;
for j=1:N
sum1(i) = sum1(i) + (2*j-1) * A(j)*sin((2*j-1)*theta(i));
sum2(i) = sum2(i) + A(j)*sin((2*j-1)*theta(i));
end
end
CL_TO = pi * AR * A(1)
```

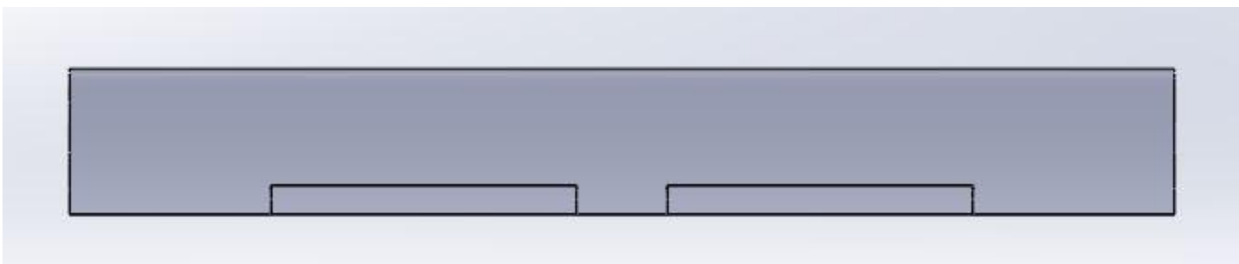
SUMMARY OF WING PARAMETER

1. Number of wings: monoplane
2. Wing vertical location: High wing
3. Airfoil section: Clark Y
4. Wing setting angle or wing incidence, i_w = 2.8 deg
5. Takeoff wing angle of attack, $\propto_{TO_w}$ = 8.07deg (i.e. the fuselage will pitch up by 8.07 – 2.8 = 5.27deg)
6. AR = 7
7. Wing span, b = √(AR* S) = √(7*0.1878) = 1.1466m
8. Wing chord, C = S/b = 0.1878/1.1466 =0.164m (16.4cm)
9. Taper ratio = 1 i.e. root chord = tip chord = 0.164m
10. Sweep angle = 0 deg
11. No twist
12. Dihedral angle = 0 deg
13. Flap chord, C_f = 0.2 * C =0.2 * 0.164 =0.0328m (3.28cm)
14. Flap span, b_f = 0.6 * b = 0.6 * 1.1466 = 0.6885m (68.8cm)
15. Flap deflection at takeoff = 13deg

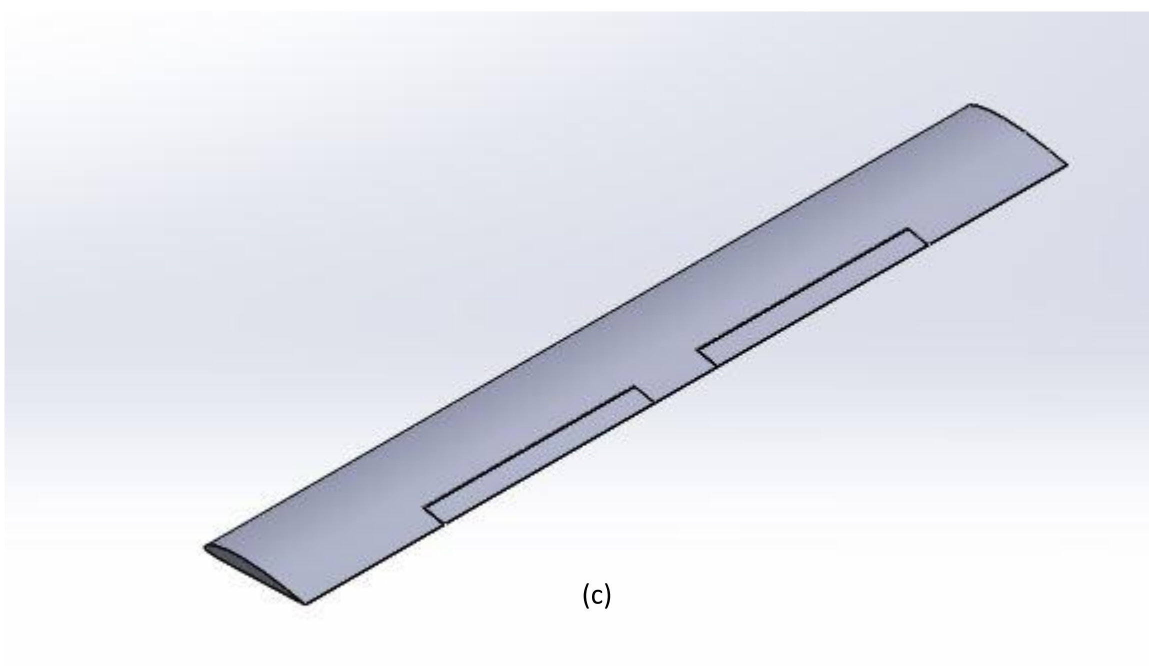
With the use of Solidworks and the information above, the figures below show the wing in views.



(a)



(b)



(c)

Figure 5.3: the designed wing in; (a) side view; (b) top view; (c) isometric

CHAPTER 6

TAIL DESIGN

PRACTICAL DESIGN STEPS

STEP 1: Tail Configuration Selection – Conventional Tail

HORIZONTAL TAIL DESIGN

STEP 2: Horizontal Tail Location (Aft or Forward (Canard)) – Aft

STEP 3: Horizontal Tail Volume Coefficient (V_H) Selection

Considering the effect of high and lower value of this parameter, V_H was selected to be **0.6** as an initial value

STEP 4: Calculation Of Optimum Tail Moment Arm (l_{opt}) To Minimize the Aircraft Drag and Weight

Optimum arm was calculated using the equation below:

$$l_{opt} = K_c\sqrt{\frac{4CSV_H}{\pi D_f}} \tag{6.1}$$

Where, Kc is the correction factor which compensates for fuselage aft section deviation from conical shape (it is between 1 and 1.4), C is the wing MAC, S is the wing area, V_H is the horizontal tail volume coefficient and D_f is the maximum fuselage diameter. Kc was selected to be 1.1 and D_f was set to be 10cm.

$$\therefore\ l_{opt} = 1.1 \times \sqrt{\frac{4 \times 0.164 \times 0.1878 \times 0.6}{\pi \times 0.1}} = 0.5335m = 53.35cm$$

From table 2 below, tail arm to total aircraft length ratio was selected to be **0.6**. Therefore, total aircraft length is 0.8892m.

No.	Aircraft configuration/type	l/L
1	An aircraft whose engine is installed at the nose and has an aft tail	0.6
2	An aircraft whose engine(s) are installed above the wing and has an aft tail	0.55
3	An aircraft whose engine is installed at the aft fuselage and has an aft tail	0.45
4	An aircraft whose engine is installed under the wing and has an aft tail	0.5
5	Glider (with an aft tail)	0.65
6	Canard aircraft	0.4
7	An aircraft whose engine is inside the fuselage (e.g., fighter) and has an aft tail	0.3

Table 6.1: Typical values of l/L for various aircraft configurations

STEP 5: Horizontal Tail Planform Area, Sh

$$S_h = \frac{V_H S C}{l_h} \tag{6.2}$$

$$\therefore \; S_h = \frac{0.6 \times 0.164 \times 0.1878}{0.5335} = 0.0346 m^2$$

STEP 6: Wing/Fuselage Aerodynamic Pitching Moment Coefficient (Cmo_wf) Calculation

This can be calculated using the following equation:

$$C_{m_{0\_wf}} = C_{m_{af}} \cdot \frac{AR cos^2(\Lambda)}{AR + 2\cos(\Lambda)} + 0.01\alpha_t \tag{6.3}$$

Where $C_{m_{af}}$ is the wing airfoil section pitching moment coefficient, AR is the wing aspect ratio, Λ is the wing sweep angle, and α_t is the wing twist angle (in degrees).

The airfoil for this design is CLARK Y and its pitching moment is -0.099. From wing design, AR is 7, sweep angle and twist angle are zero

$$C_{m_{0\_wf}} = -0.099 \cdot \frac{7 \times cos^2(0)}{7 + 2\cos(0)} + 0.01(0) = -0.077$$

STEP 7: Cruise Lift Coefficient, C_{LC}

It has been calculated earlier to be **0.48**.

STEP 8: Calculation of Horizontal Tail Desired Lift Coefficient at Cruise from Trim Equation

The trim equation was derived (in Ref 1) to be:

$$C_{m_{o\_wf}} + C_{L_C}(h - h_o) - \eta_h \mathrm{V_H} C_{L_h} = 0 \tag{6.4}$$

Where $C_{m_{o\_wf}}$ is the wing/fuselage aerodynamic pitching moment, C_{L_C} is the aircraft cruise lift coefficient, η_h is the horizontal tail efficiency (which is not always 100% because of wake and downwash and was selected to be **90%** or **0.9**), V_H is the horizontal tail volume coefficient and C_{LC} is the aircraft cruise lift coefficient.

The variable h_o denotes the non-dimensional wing/fuselage aerodynamic center $\left(\frac{X_{ac_{wf}}}{C}\right)$ position. A typical value for h_o is about 0.2–0.25 for the majority of aircraft configurations. The parameter h denotes the non-dimensional aircraft cg position$\left(\frac{X_{cg}}{C}\right)$. The typical values for the aircraft non-dimensional center of gravity limit are: h =0.1 to 0 .3. Therefore, the values for **h and h_o are selected to be 0.2 and 0.25 respectively**.

Equation (14) was rearranged as:

$$C_{L_h} = \frac{C_{m_{o\_wf}} + C_{L_C}(h - h_o)}{\eta_h \mathrm{V_H}} \tag{6.5}$$

$$C_{L_h} = \frac{-0.077 + 0.48(0.2 - 0.25)}{0.9 \times 0.6} = -0.1870$$

Therefore, the desired horizontal tail lift coefficient is **-0.1870**

STEP 9: Horizontal Tail Airfoil Section Selection

The following are considered in selecting the horizontal tail airfoil:

1. Symmetric airfoil
2. Minimum drag
3. Large lift curve slope
4. Lower maximum thickness compared to the wing airfoil

The horizontal tail airfoil section was selected to be **NACA 0008** which has the following properties at Reynolds number of 150,000:

1. Zero-lift angle of attack (α_o) is zero
2. Stall angle of attack is 8deg
3. Lift curve slope is 6.0176 1/rad
4. Ideal lift coefficient (Cli) is 0
5. Maximum Thickness to chord ($(t/C)_{max}$) is 8%
6. Cl_{max} is 0.77
7. Cd_{min} is 0.00958
8. Minimum Pitching moment is -0.009
9. Maximum (L/D) is 38.77

Lift curve slope (1/rad) was calculated using the following equation:

$$C_{l_\alpha} = \frac{dC_l}{d\alpha} = 1.8\pi\left(1 + 0.8\left({}^{t}/_{c}\right)_{max}\right) \tag{6.6}$$

STEP 10: Selection of Horizontal Tail Sweep Angle and Dihedral

As an initial value, the sweep and dihedral angles were set to be **zero degree**.

STEP 11: Horizontal Tail Aspect Ratio and Taper Ratio

Considering the effects of aspect ratio on the horizontal tail, the initial value for aspect ratio was calculated using this relationship

$$AR_h = \frac{2}{3}AR_w \tag{6.7}$$

$$AR_h = \frac{2}{3} \times 7 = 4.667$$

The horizontal tail taper ratio was selected to be **0.7**

STEP 12: Calculation of Horizontal Tail Lift Curve Slope, CL_{α_h}

The horizontal tail lift curve slope was determined with the following equation:

$$C_{L_{\propto\_h}} = \frac{dC_{L_h}}{d\alpha_h} = \frac{C_{l_{\propto\_h}}}{1+\frac{C_{l_{\propto\_h}}}{\pi \cdot AR_h}} \tag{6.8}$$

Where $C_{l_{\propto\_h}}$ is the horizontal tail airfoil lift curve slope.

$$C_{L_{\propto\_h}} = \frac{6.0176}{1 + \frac{6.0176}{\pi \cdot 4.667}} = 4.2666 \text{ rad}^{-1}$$

STEP 13: Horizontal Tail Angle Of Attack at Cruise

$$\alpha_h = \frac{C_{L_h}}{C_{L_{\propto\_h}}} \tag{6.9}$$

$$\alpha_h = \frac{-0.187}{4.2666} = -0.04383 rad \ (-2.511 deg)$$

STEP 14: Downwash Angle (ε) at the Tail

The downwash is a function of wing angle of attack (α_w) and is determined as follows:

$$\varepsilon = \varepsilon_o + \frac{\partial \varepsilon}{\partial \alpha} \alpha_w \tag{6.10}$$

Where ε_o (downwash angle at zero angle of attack (in rad)) and $\frac{\partial \varepsilon}{\partial \alpha}$ (downwash slope) are found as:

$$\varepsilon_o = \frac{2C_{L_w}}{\pi \cdot AR} \tag{6.11}$$

$$\frac{\partial \varepsilon}{\partial \alpha} = \frac{2C_{L_{\propto\_w}}}{\pi \cdot AR} \tag{6.12}$$

Where $\propto_w$ is the wing setting angle (in rad), C_{L_w} is the wing lift coefficient at cruise, $C_{L_{\propto\_w}}$ is the wing lift curve slope (rad^{-1}) and AR is the wing aspect ratio.

From wing design, the following were obtained:

1. $C_{L_{\propto\_w}} = 4.8276 \text{ rad}^{-1}$
2. AR = 7
3. $C_{L_w} = C_{L_C} = 0.48$ (since horizontal tail is to be design to satisfy cruise conditions)
4. $\propto_w = 2.8\ deg$ (0.04887rad)

$$\frac{\partial \varepsilon}{\partial \alpha} = \frac{2 \times 4.8276}{\pi \times 7} = 0.4390\ rad/rad$$

$$\varepsilon_o = \frac{2 \times 0.48}{\pi \times 7} = 0.0436\ rad$$

$$\varepsilon = 0.0436 + 0.4390 \cdot 0.04887 = 0.0651\ rad$$

STEP 15: Horizontal Tail Incidence Angle, i_h

Due to the presence of the downwash at the horizontal tail location, the tail effective angle of attack is defined as:

$$\propto_h = \propto_f + i_h - \varepsilon \tag{6.13}$$

Where $\propto_f$ is the fuselage angle of attack at cruise (which is zero), α_h is the horizontal tail angle of attack (Calculated at step 13).

$$i_h = \propto_h - \propto_f + \varepsilon \tag{6.14}$$

$$i_h = (-0.04383) - 0 + 0.0651 = 0.02127rad\ (1.2187deg)$$

STEP 16: Tail Span, Tail Root Chord, Tail Tip Chord, and Tail MAC

The following equations were used to obtain the values of horizontal tail span; root chord; tip chord; and MAC (or C)

$$AR_h = \frac{b_h}{C_h} \tag{6.15}$$

$$AR_h = \frac{{b_h}^2}{S_h} \tag{6.16}$$

$$\lambda_h = \frac{C_{h_{tip}}}{C_{h_{root}}} \tag{6.17}$$

$$C_h = \frac{2}{3} C_{h_{root}} \left(\frac{1+\lambda_h+{\lambda_h}^2}{1+\lambda_h} \right) \tag{6.18}$$

On rearranging the equations above, I have the following:

$$b_h = \sqrt{AR_h \times S_h} \tag{6.19}$$

$$C_h = \frac{b_h}{AR_h} \tag{6.20}$$

$$C_{h_{root}} = \frac{C_h}{\frac{2}{3}\left(\frac{1+\lambda_h+{\lambda_h}^2}{1+\lambda_h} \right)} \tag{6.21}$$

$$C_{h_{tip}} = \lambda_h \times C_{h_{root}} \tag{6.22}$$

Therefore;

$$b_h = \sqrt{4.667 \times 0.0346} = 0.402m$$

$$C_h = \frac{0.402}{4.667} = 0.0862m$$

$$C_{h_{root}} = \frac{0.0862}{\frac{2}{3}\left(\frac{1+0.7+0.7^2}{1+0.7}\right)} = 0.1004m$$

$$C_{h_{tip}} = 0.7 \times 0.1004 = 0.07028m$$

STEP 17: Calculation of Horizontal Tail Lift Coefficient at Cruise (Using Lifting Line Theory)

To do this, the tail was treated as a small wing and the following parameters were used:

1. Horizontal tail area ($0.0346m^2$)
2. Horizontal tail aspect ratio (4.667)
3. Horizontal tail taper ratio (0.7)
4. Tail angle of attack (-2.511 deg)
5. Tail airfoil lift curve slope ($6.0176\ rad^{-1}$)
6. Airfoil zero lift angle of attack (0 deg)
7. Tail span (0.402m)
8. MAC (0.0862m)
9. Root chord (0.1004m)

To calculate the lift generated by the horizontal tail, the same MATLAB code that was used to for the wing lift was used. With the parameters above, the lift generated by the tail is **-0.1818**

MATLAB CODE

```
clc
clear
N = 9; % (number of segments-1)
Sh = 0.0346; % m ^ 2
ARh = 4.667; % Aspect ratio
lambdah = 0.7; % Taper ratio
alpha_twist = 1e-8; % Twist angle (deg)
a_h = -2.511; % tail angle of attack (deg)
a_2d = 6.0176; % tail airfoil lift curve slope (1/rad)
alpha_0 = 0; % Airfoil zero-lift angle of attack (deg)
b = sqrt(ARh*Sh); % tail span
Chroot = 0.1004; % root chord
MAC = (2/3)*Chroot*(1+lambdah+lambdah^2)/(1+lambdah); % Mean Aerodynamic
Chord
theta = pi/(2*N):pi/(2*N):pi/2;
alpha=a_h+alpha_twist:-alpha_twist/(N-1):a_h;% segment’s angle of attack
z = (b/2)*cos(theta);
c = Chroot * (1 - (1-lambdah)*cos(theta)); % Mean Aerodynamics chord at each
segment
mu = c * a_2d / (4 * b);
LHS = mu .* (alpha-alpha_0)/57.3; % Left Hand Side
```

```
% Solving N equations to find coefficients A(i):
for i=1:N
for j=1:N
B(i,j)=sin((2*j-1) * theta(i)) * (1+(mu(i) *(2*j-1))/sin(theta(i)));
end
end
A=B\transpose(LHS);
for i = 1:N
sum1(i) = 0;
sum2(i) = 0;
for j=1:N
sum1(i) = sum1(i) + (2*j-1) * A(j)*sin((2*j-1)*theta(i));
sum2(i) = sum2(i) + A(j)*sin((2*j-1)*theta(i));
end
end
CL_htail = pi * ARh * A(1)
```

STEP 18: Adjustment of the Horizontal Tail Angle Of Attack to Meet the Required Lift Coefficient

The lift coefficient required from the horizontal tail is **-0.1870**. The result obtained at step 17 is not the required lift coefficient. Therefore, the angle of attack was adjusted through trial and error with the same MATLAB program. On doing this, the horizontal tail was found to have this lift coefficient at **-2.59 deg**.

The horizontal tail incidence was recalculated as:

$$i_h = (-2.59deg) - 0 + \left(0.0651rad \times \frac{57.3deg}{1rad}\right) = 1.14deg$$

STEP 19: Checking Of Horizontal Tail Stall

The tail airfoil was expected to stall at 8°, the tail setting angle is 1.14°, meaning that the fuselage will pitch up to 6.86° before the tail stalls. The wing setting angle is 2.8deg and the airfoil will stall at 13deg. Therefore, the fuselage will pitch up to 10.2deg meaning that horizontal tail will stall before the wing which must be prevented.

STEP 20: Design Review

To avoid this, another airfoil (Naca0010) which stalls later than the current airfoil was selected. The airfoil has the following parameters (at Re = 150,000):

1. Cl_{max} = 0.91
2. Stall angle = 10°
3. Zero-lift angle of attack = 0
4. Lift curve slope = 6.108rad^{-1}

5. Cd_{min} = 0.01037
6. Cm_{min} = -0.02
7. $(t/c)_{max}$ = 10% or 0.1
8. $(L/D)_{max}$ = 41.09

Based on these, the following are the new values:

1. Horizontal tail lift curve slope = 4.3118rad^{-1}
2. Initial tail angle of attack = -0.04337rad (-2.485deg)
3. Tail angle of attack = 2.56deg
4. Tail incidence = 1.17deg

On checking stall, the tail stalls when fuselage pitches up to 8.83deg. Since the tail still stall earlier than the wing, some parameters that will reduce the tail incidence were looked into.

To reduce the tail angle of attack, the negative value of the desired tail lift coefficient has to be reduced which is a function of tail volume coefficient. Reducing the tail volume coefficient will increase the desired tail lift coefficient's negative value.

After considering effects of some values of tail volume coefficient, it was set to be 0.3. With this the following are the new revised values:

1. Horizontal tail volume coefficient, V_H = 0.3
2. Optimum arm, l_{opt} = 0.3773m
3. Aircraft length, L = 0.6288m
4. Horizontal tail area, S_h =0.0245m^2
5. Desired horizontal tail lift coefficient, C_{Lh} = -0.374
6. Horizontal tail span, b_h = 0.3381m
7. Horizontal tail MAC, C_h = 0.0724m
8. Horizontal tail root chord, C_{h_root} = 0.0844m
9. Horizontal tail tip chord, C_{h_tip} = 0.059m
10. Horizontal tail setting angle from lifting line theory, α_h = -5.12deg
11. Horizontal tail incidence (with downwash effect), i_h = -1.39deg
12. The tail airfoil stalls when fuselage pitches up to 11.39deg and wing stalls when fuselage pitches up to 10.2deg.

STEP 21: STATIC MARGIN CHECK

Static margin which is a measure of the degree of stability or instability of an aircraft was calculated using:

$$SM = \frac{x_{np} - x_{cg}}{c} \tag{6.23}$$

Where, x_{np} is the neutral point position on the wing.

$$\frac{x_{np}}{C} \simeq \frac{1}{4} + \frac{1+2/AR}{1+2/AR_h}\left(1 - \frac{4}{AR+2}\right)V_H \tag{6.24}$$

$$\frac{x_{np}}{C} \simeq \frac{1}{4} + \frac{1+2/7}{1+2/4.667} \cdot \left(1 - \frac{4}{7+2}\right) \cdot 0.3 \simeq 0.4$$

Equation (33) can be rewritten as:

$$SM = \frac{x_{np}}{C} - \frac{x_{cg}}{C} \tag{6.25}$$

It was recalled that the distance between wing leading edge and aircraft center of gravity (CG) is hC (i.e. h multiplied by MAC). Mathematically;

$$x_{cg} = hC \tag{6.26}$$

Therefore,

$$h = \frac{x_{cg}}{C} \tag{6.27}$$

h was earlier stated as 0.2. Then, static margin is:

$$SM = 0.4 - 0.2 = 0.2$$

This means that the aircraft is reasonably stable. In Ref 3, it was stated that "*a well behaved aircraft typically has a V_H value which falls in the range of 0.3 and 0.6*". And in Ref 1, it was stated that "*the typical value for static margin is from 0.1 to 0.3*"

The position of the aircraft CG location on the wing was calculated as:

$$x_{cg} = h \times C = 0.2 \times 0.164 = 0.0328m \tag{6.28}$$

Meaning that, the CG is 3.28cm from the wing leading edge.

STEP 22: Calculation of Horizontal Tail Contribution to the Static Longitudinal Stability Derivative ($C_{m\alpha}$).

For the aircraft to be stable, Cm_α must be negative. Cm_α can be expressed as:

$$C_{m_\alpha} = C_{L_{\alpha\_wf}}(h - h_o) - C_{L_{\alpha\_h}}\eta_h \frac{S_h}{S}\left(\frac{l}{C} - h\right)\left(1 - \frac{d\varepsilon}{d\alpha}\right) \tag{6.29}$$

All these parameters have been described and obtained earlier; their values were just slotted into the equation. $C_{L\alpha\_wf}$ is the wing/fuselage lift curve slope and was set to be the same as wing lift curve slope. All the angles were considered in radians. η_h is the horizontal tail efficiency.

$$C_{m_\alpha} = 4.8276(0.2 - 0.25) - 4.2666 \cdot 0.9 \cdot \frac{0.0245}{0.1878} \cdot \left(\frac{0.3773}{0.164} - 0.2\right)(1 - 0.4390)$$
$$= -0.83 rad^{-1}$$

The table below was used as a guideline.

No.	Requirements	Stability derivatives	Symbol	Typical value (1/rad)
1a	Static longitudinal stability	Rate of change of pitching moment coefficient with respect to angle of attack	C_{m_α}	−0.3 to −1.5
1b	Static longitudinal stability	Static margin	$h_{np} - h_{cg}$	0.1–0.3
2	Dynamic longitudinal stability	Rate of change of pitching moment coefficient with respect to pitch rate	C_{m_q}	−5 to −40
3	Static directional stability	Rate of change of yawing moment coefficient with respect to sideslip angle	C_{n_β}	+0.05 to +0.4
4	Dynamic directional stability	Rate of change of yawing moment coefficient with respect to yaw rate	C_{n_r}	−0.1 to −1

Table 6.2: The static and dynamic longitudinal and directional stability requirements

STEP 23: Analysis of the Dynamic Longitudinal Stability

The dynamic longitudinal stability analysis is performed after all aircraft components are designed and the roots (λ) of the longitudinal characteristic equation are calculated. A general form of the aircraft longitudinal characteristic equation looks like the following:

$$A_1\lambda^4 + B_1\lambda^3 + C_1\lambda^2 + D_1\lambda + E_1 = 0 \tag{6.30}$$

The longitudinal stability derivatives cannot be determined unless all aircraft components, including wing and fuselage, have been designed. This is why it was resorted to a simplifying criterion that could be a base for the horizontal tail preliminary design; when the horizontal tail volume coefficient (V_H) is close to the ballpark number as stated in step 21, it is 90% confident that the longitudinal stability requirements have been satisfied.

STEP 24: Horizontal Tail Vertical Location

To avoid deep stall, the following experimental equations were recommended for the initial approximation of the horizontal tail vertical height [1]:

$$h_t > l \cdot tan(\alpha_s - i_w + 3) \tag{6.31}$$

$$h_t < l \cdot tan(\alpha_s - i_w - 3) \tag{6.32}$$

Where α_s is the wing stall angle (in degrees) and i_w is the wing setting angle (in degrees). The first equation is for horizontal tail above the wing level while the second is for tail below wing level. The second equation was used because the design is high wing and the horizontal tail is to be mounted on the fuselage.

$$h_t < 0.3773 \cdot tan(13 - 2.8 - 3)$$

$$h_t < 0.0477m$$

A MATLAB code was written for all of the processes involved in horizontal tail design which made the iterations easier.

MATLAB CODE

```
clc
clear
VH = 0.3; % horizontal tail volume coefficient
Kc = 1.1; % Fuselage aft shape correction factor
MAC = 0.164; % wing MAC (m)
S = 0.1878; % Wing area (m^2)
Df = 0.1; % Fuselage diameter (m)
lL = 0.6; % Ratio of horizontal tail arm to aircraft legngth
Cm_af = -0.099; % wing airfoil pitching moment
AR = 7; % Wing aspect ratio
swp = 0; % Wing sweep angle (deg)
alfa_twist = 0; % Wing twist angle (deg)
Wavg = 11.772; % Aircraft average weight (N)
Vc = 14.6154; hc = 15; % Cruise speed (m/s) and cruise altitude (m)
nh = 0.9; % Horizontal tail (HT) efficiency
h = 0.2; % Center of gravity limit (m/m)
ho = 0.25;% Wing/fuselage aerodynamic center (m/m)
swp_h = 0; % HT sweep angle (deg)
dihed = 0; % HT dihedral angle (deg)
lambdah = 0.7;% HT taper ratio
ARh = 2*AR/3; % HT aspect ratio
Cl_alfa = 6.108; % HT airfoil lift curve slope (1/rad)
CL_alfa_w = 4.827578; % Wing lift curve slope (1/rad)
iw = 2.8; % Wing setting angle (deg)
alfa_sw = 13; % Wing airfoil stall angle (deg)
alfa_fus = 0; % Fuselage AOA at cruise (rad)
alfa_sh = 10; % HT airfoil stall AOA (deg)
```

```
l_opt = Kc *sqrt(4*MAC*S*VH/(3.142*Df))% optimum tail arm (m)
l = l_opt;
L = l/lL
Sh = VH*MAC*S/l % Horizontal tail area (m^2)
Cmo_wf= (Cm_af*AR*(cosd(swp)^2)/(AR+(2*cosd(swp)^2)))+(0.01*alfa_twist)  %
Wing/fuselage aerodynamic pitching moment coefficient
x=hc;
x0=0; y0=1;
x1=305; y1=0.9711;
x2=610; y2=0.9428;
L0=((x-x1)*(x-x2))/((x0-x1)*(x0-x2));
L1=((x-x0)*(x-x2))/((x1-x0)*(x1-x2));
L2=((x-x0)*(x-x1))/((x2-x0)*(x2-x1));
y=(L0*y0)+(L1*y1)+(L2*y2);
rho_c=y*1.225; % Air density at cruise altitude (kg/m^3)
CLc=2*Wavg/(rho_c*S*Vc^2) % Aircraft cruise lift coefficient
CLh = (Cmo_wf+(CLc*(h-ho)))/(nh*VH) % Horizontal tail required CL at cruise
CL_alfa_h = Cl_alfa/(1+(Cl_alfa/(3.142*ARh))) % HT lift curve slope (1/rad)
set_h = CLh/CL_alfa_h % Initial HT setting angle (rad)
eo = 2*CLc/(3.142*AR) % Downwash angle at zero angle of attack (rad)
e_slope = (2*CL_alfa_w)/(3.142*AR) % Downwash slope (rad/rad)
E = eo+(e_slope*(iw/57.3)) % Downwash angle at tail (rad)
bh = sqrt(ARh*Sh) % HT span
MACh = bh/ARh % HT MAC
Ch_root =3*MACh/(2*((1+lambdah+(lambdah^2))/(1+lambdah))) % HT root chord
Ch_tip = lambdah * Ch_root % HT tip chord
ih = set_h-alfa_fus+E % HT setting angle (rad)

% Calculation of HT CL with lifting line theory
N = 9; % (number of segments-1)
alpha_twist = 0.00001; % Twist angle (deg)
a_h = -5.12; % tail angle of attack (deg)
a_2d = Cl_alfa; % tail airfoil lift curve slope (1/rad)
alpha_0 = 0; % Airfoil zero-lift angle of attack (deg)
theta = pi/(2*N):pi/(2*N):pi/2;
alpha=a_h+alpha_twist:-alpha_twist/(N-1):a_h;% segment’s angle of attack
z = (bh/2)*cos(theta);
c = Ch_root * (1 - (1-lambdah)*cos(theta)); % Mean Aerodynamics chord at each
segment
mu = c * a_2d / (4 * bh);
LHS = mu .* (alpha-alpha_0)/57.3; % Left Hand Side
% Solving N equations to find coefficients A(i):
for i=1:N
for j=1:N
B(i,j)=sin((2*j-1) * theta(i)) * (1+(mu(i) *(2*j-1))/sin(theta(i)));
end
end
A=B\transpose(LHS);
for i = 1:N
sum1(i) = 0;
sum2(i) = 0;
for j=1:N
sum1(i) = sum1(i) + (2*j-1) * A(j)*sin((2*j-1)*theta(i));
sum2(i) = sum2(i) + A(j)*sin((2*j-1)*theta(i));
end
end
CL_htail = pi * ARh * A(1)
```

```
set_h = a_h; % HT setting angle (deg) from lifting line theory
ih = set_h-alfa_fus+(E*57.3) % HT setting angle (deg)
HT_stall_AOA = alfa_sh-ih % Fuselage AOA for HT to stall (deg)
hnp = (1/4)+((1+(2/AR))/(1+(2/ARh))*(1-(4/(AR+2)))*VH); % Neutral point to
chord ratio
Xcg = h*MAC; % CG distance (m) from Wing LE towards TE
SM = hnp-h % Static Margin
HT_Wing_area = Sh/S; % HT to wing area ratio
Cm_alfa = (CL_alfa_w *(h-ho))-(CL_alfa_h*nh*(Sh/S)*((l/MAC)-h)*(1-e_slope))%
Longitudinal static stability derivative (1/rad)
ht_above = l*tand(alfa_sw-iw+3); % HT location above wing (m) for HT with
higher vertical location relative to the wing ac
ht_below = l*tand(alfa_sw-iw-3) % HT location below wing (m) for HT with
lower vertical location relative to the wing ac
```

HORIZONTAL TAIL DESIGN RESULT

1. Horizontal tail location: aft tail.
2. Horizontal tail volume coefficient: 0.3
3. Optimum arm: 0.3773m
4. Fuselage length: 0.6288m
5. Horizontal tail area: 0.0245m^2
6. Wing/fuselage aerodynamic pitching moment coefficient: -0.0770
7. Horizontal tail desired lift coefficient: -0.3740
8. Horizontal tail airfoil section: NACA 0010
9. No sweep angle and no dihedral angle
10. Aspect ratio: 4.6667
11. Taper ratio: 0.7
12. Lift curve slope: 4.3118 rad^{-1}
13. Span: 0.3381m
14. Root chord: 0.0844m
15. Tip chord: 0.0590m
16. MACh: 0.0724m
17. Horizontal tail incidence: -1.39deg
18. Static Margin, SM: 0.2
19. Horizontal tail vertical location (relative to wing), $h_t < 0.0477$m
20. Horizontal distance from wing root leading edge to tail root leading edge: 38.82cm
21. Horizontal distance from wing root leading edge to tail root trailing edge: 47.26cm

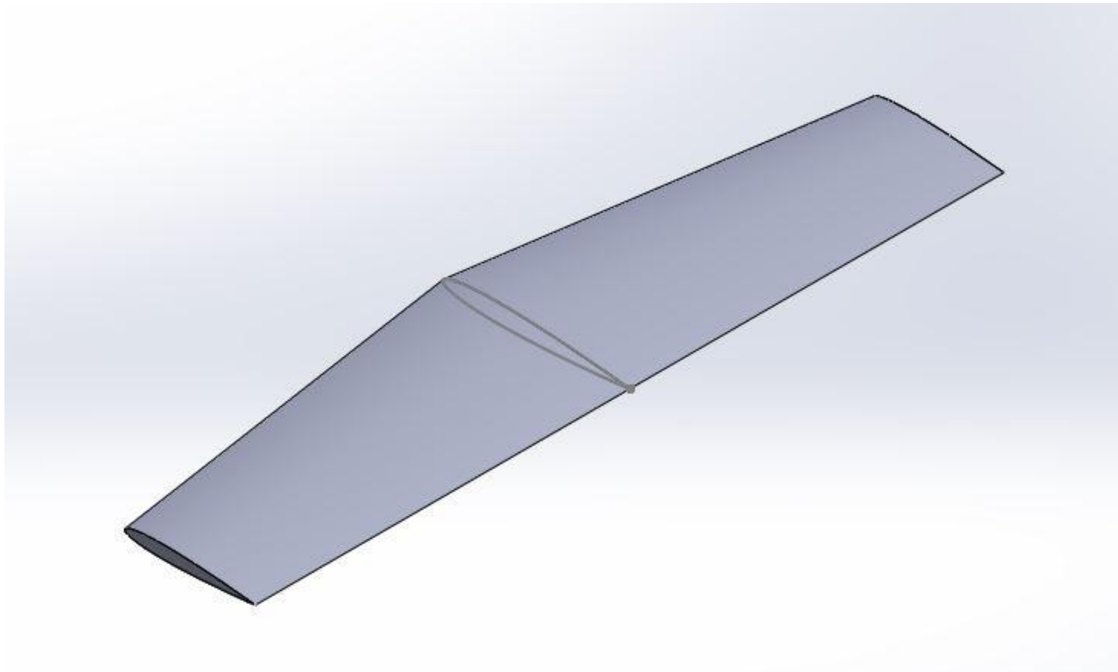


Figure 6.1: Designed horizontal tail in isometric view

VETRTICAL TAIL DESIGN

The third lifting surface in a conventional aircraft is the vertical tail, which is sometimes referred to as a vertical stabilizer or fin. The vertical tail tends to have two primary functions: (i) directional stability and (ii) directional trim. The vertical tail will be designed to meet these two requirements. The vertical tail is a major contributor in maintaining directional control, which is the primary function of the rudder.

STEP 25: Vertical Tail Configuration: Conventional

STEP 25: Vertical Tail Volume Coefficient

Considering the effects, the vertical tail volume coefficient was selected to be **0.035**.

STEP 26: Vertical Tail Moment Arm (l_v)

This was select to be the same as the horizontal tail arm which is **0.3773m**.

STEP 27: Vertical Tail Planform Area, S_V

$$S_V = \frac{S \cdot b \cdot V_V}{l_V} \tag{6.33}$$

$$S_V = \frac{0.1878 \cdot 1.1466 \cdot 0.035}{0.3773} = 0.02m^2$$

STEP 28: Vertical Tail Airfoil Section

The vertical tail airfoil section was selected to be **NACA 008** which has the following properties:

1. Zero-lift angle of attack (α_o) is zero
2. Stall angle of attack is 8deg
3. Lift curve slope is 6.0176 1/rad
4. Ideal lift coefficient (Cli) is 0
5. Maximum Thickness to chord ($(t/C)_{max}$) is 8%
6. Cl_{max} is 0.77
7. Cd_{min} is 0.00958
8. Minimum Pitching moment is -0.009
9. Maximum (L/D) is 38.77

STEP 29: Vertical Tail Aspect Ratio, AR_V

Considering the effects of vertical tail aspect ratio, it was selected to be **1.5**. As a guideline, it was suggested (in Ref 1) that vertical tail aspect ratio should be between 1 and 2

STEP 30: Vertical Tail Taper Ratio: The vertical tail taper ratio was selected to be **0.75**

STEP 31: Vertical Tail Incidence Angle: The vertical tail incidence was set to be zero.

STEP 32: Vertical Tail Sweep: no sweep angle

STEP 33: Vertical Tail Dihedral: no dihedral angle.

STEP 34: Vertical Tail Span, Root Chord, Tip Chord and MAC

To calculate the above tail parameters:

$$AR_V = \frac{b_V}{C_V} \tag{6.34}$$

$$\lambda_V = \frac{c_{V_{tip}}}{C_{V_{root}}} \tag{6.35}$$

$$C_V = \frac{2}{3} C_{V_{root}} \left(\frac{1+\lambda_V+\lambda_V^2}{1+\lambda_V}\right) \tag{6.36}$$

$$AR_V = \frac{b_V^2}{S_V} \tag{6.37}$$

Therefore:

$$1.5 = \frac{b_V^2}{0.02} \quad \therefore b_V = 0.173m$$

$$C_V = \frac{b_V}{AR_V} = \frac{0.173}{1.5} = 0.115m$$

$$0.115 = \frac{2}{3} C_{V_{root}} \left(\frac{1 + 0.75 + 0.75^2}{1 + 0.75} \right) \quad \therefore C_{V_{root}} = 0.131m$$

$$c_{V_{tip}} = \lambda_V . C_{V_{root}} = 0.75 \times 0.131 = 0.0983m$$

STEP 35: Spin Recovery

Since this aircraft is not designed to be spinnable, spin recovery will not be considered.

STEP 36: Directional Trim

Since the aircraft is designed to be right-left symmetric, the directional trim equation is:

$$\sum N_{cg} = 0 \Rightarrow T \cdot Y \; + L_v l_{vt} \; = 0 \tag{6.38}$$

Where T denotes the engine thrust, Y is the distance between the thrust line and the aircraft cg in the xy plane, l_{vt} is the distance between the vertical tail aerodynamic center and the aircraft cg. L_V is the horizontal tail lift.

The value for Y is **zero** because the center of gravity is to be at the center (since the aircraft is right-left symmetric) and the engine will be at the nose center.

The value for L_V is also **zero** because the airfoil zero lift angle of attack is zero and the tail incidence is also zero. Therefore, vertical tail generated lift coefficient is zero which also makes the lift force to be zero.

$$\sum N_{cg} = 0 \Rightarrow T \cdot 0 \; + 0 \cdot l_{vt} \; = 0$$

STEP 37: Directional Stability

$$C_{n_\beta} = K_{f_1} \cdot C_{L_{\alpha\_V}} \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \cdot \frac{l_{vt} S_v}{bS} \tag{6.39}$$

Where $C_{n\beta}$ is the directional stability, K_f is the fuselage contribution to directional stability, $C_{L\alpha\_v}$ is the vertical tail lift curve slope (in rad^{-1}), η_v is vertical tail efficiency, $\frac{d\sigma}{d\beta}$ is the vertical tail side wash gradient (in rad) and $\frac{l_{vt} S_v}{bS}$ is vertical tail volume coefficient.

The typical value for K_{f1} ranges from 0.65 to 0.85 and was selected to be **0.75**. Sidewash gradient was selected to be **-2.6**. The vertical tail efficiency was selected to be **0.9** (90%) tail volume coefficient has been selected earlier to be **0.035**. vertical tail lift curve slope was calculated as:

$$C_{L_{\propto\_v}} = \frac{dC_{L_v}}{d\alpha_v} = \frac{C_{l_{\propto\_v}}}{1+\frac{C_{l_{\propto\_v}}}{\pi \cdot AR_v}} \tag{6.40}$$

$$C_{L_{\propto_v}} = \frac{6.0176}{1+\frac{6.0176}{\pi \cdot 1.5}} = 2.66rad^{-1}$$

$$C_{n_\beta} = 0.75 \times 2.66 \times \big(1-(-2.6)\big) \times 0.9 \times 0.035 = 0.226rad^{-1}$$

VERTICAL TAIL DESIGN RESULT

1. Tail configuration: conventional
2. Vertical tail volume coefficient: 0.035
3. Vertical tail arm: 0.3773
4. Vertical tail area: 0.02m^2
5. Airfoil section: Naca 0008
6. Vertical tail aspect ratio: 1.5
7. Vertical tail taper ratio: 0.75
8. Vertical tail incidence: 0°
9. No sweep and dihedral angle
10. Vertical tail span: 0.173m
11. Vertical tail MAC: 0.115m
12. Vertical tail root chord: 0.131m
13. Vertical tail tip chord: 0.0983m
14. Horizontal distance between wing LE to vertical tail LE: 37.355cm
15. Horizontal distance between wing LE to vertical tail TE: 50.455cm

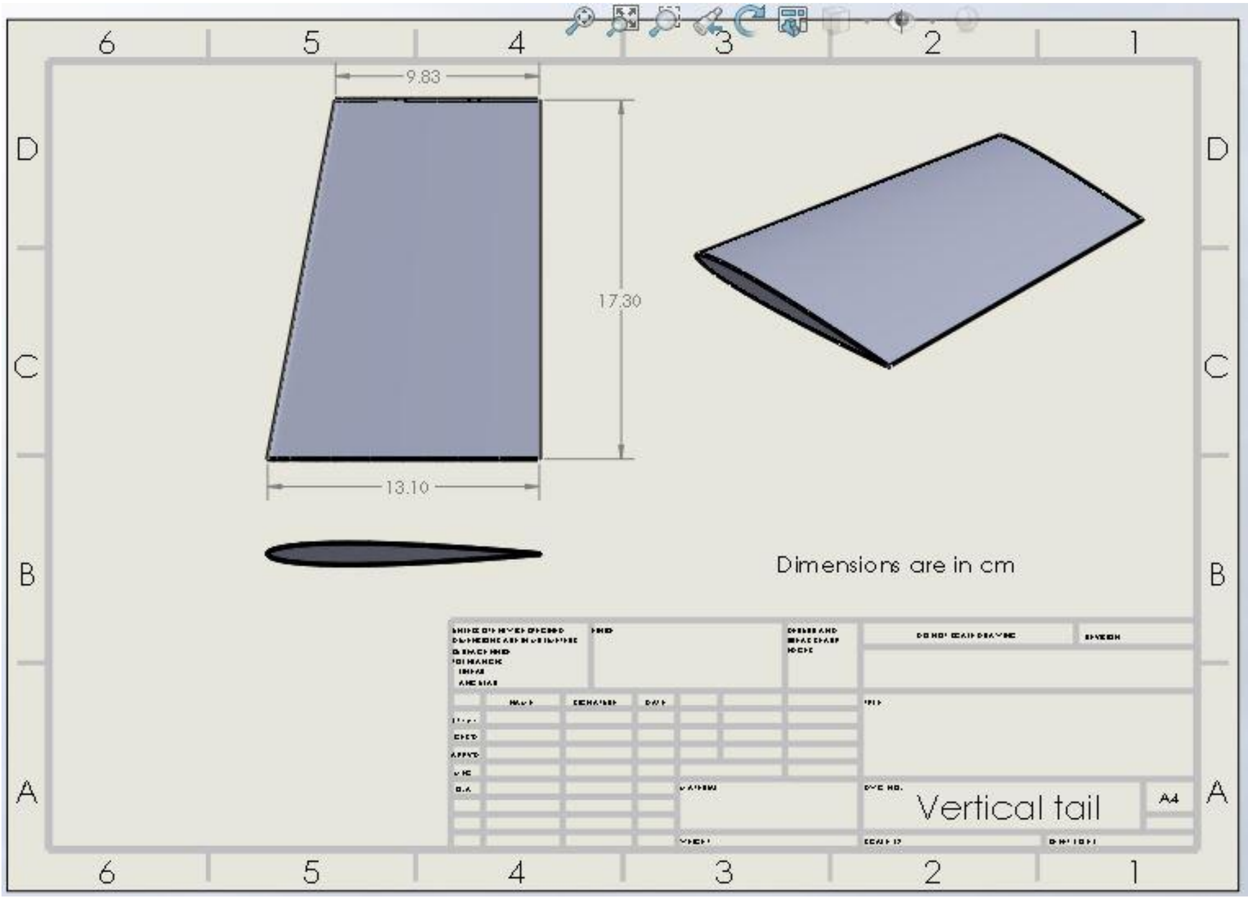


Figure 6.2: Designed vertical tail

CHAPTER 7

FUSELAGE DESIGN

This design is a UAV which will carry a light payload (as light as a sweet or a chew gum). The payload is to be contained by the fuselage. Other components which are to be accommodated by the fuselage are batteries, receiver, switch, servo, wing and engine.

The fuselage was designed to accommodate all of these and this report encompasses a step-wise description of the procedures.

STEP 1: Design Requirement

I. Light weight
II. Low drag
III. Ease of manufacture
IV. Accommodation of all internal components
V. Accommodation of the engine and the lifting surfaces
VI. Give room for required tail arm

Type: low speed, single engine, prop-driven aircraft.

Payload: 10g sweet

STEP 2: Number of Crew Members – Nil

STEP 3: Number of Flight Attendant – Nil

STEP 4: Number of Technical Personnel – Nil

STEP 5: Human Size and Target Passenger – Nil

STEP 6: Fuselage Layout (Internal) Configuration

The following are to be accommodated by the fuselage:

I. Engine (Nose)
II. Payload (Cabin)
III. Battery (Cabin)
IV. Receiver (Cabin)
V. Switch (Cabin)
VI. Servo (Cabin)
VII. Wing (Cabin top)
VIII. Tails (aft fuselage)

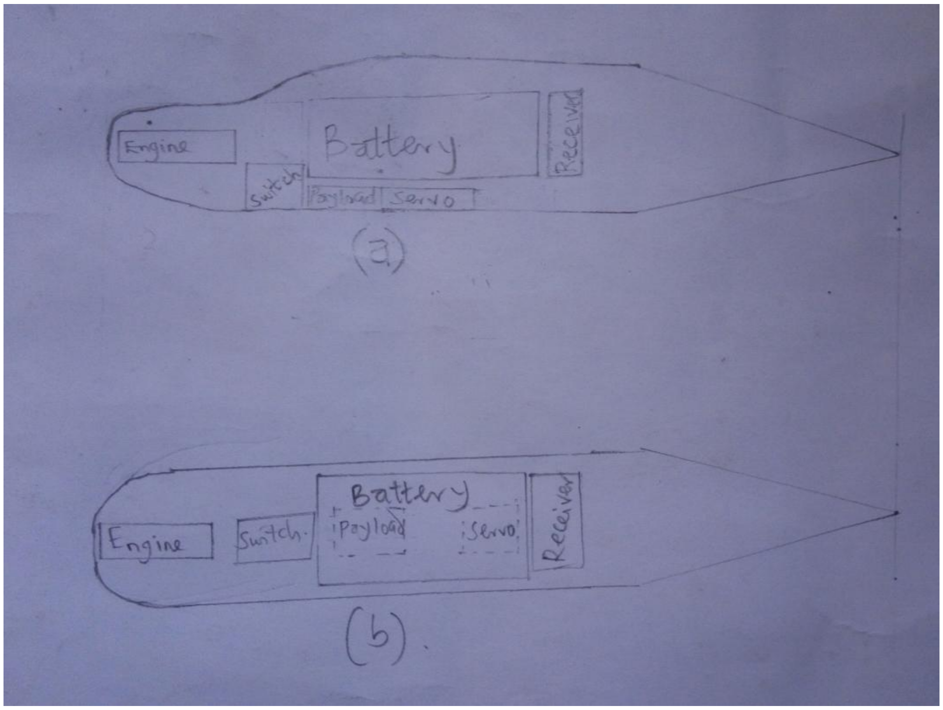


Figure 1: Sketch of fuselage configuration in (a) side view and (b) top view

STEP 7: Cockpit Instrument – Nil

STEP 8: Cockpit Design

Since this is a UAV, there is no special design for the cockpit. Rather, it was designed alongside the nose.

STEP 9: Optimum Length to Diameter Ratio $(L_F/D_F)_{opt}$

Considering the zero-lift drag coefficient, surface area and light weight, the fuselage length-to-diameter ratio was selected to be **6**.

STEP 10: Cabin Design

As stated in step 6, the components in the cabin are: battery, payload, receiver, switch and servo. The component with the highest size (dimension) is the battery and was used to define the cabin configuration.

Cabin Width (W_C)

The battery width is about 4.5cm. To hold the battery down to the fuselage, a space of 1.5cm was dedicated which makes the cabin width to be **6cm**.

Cabin Length (L_C)

Battery length (L_B) = 12cm, Receiver length (L_R) = 5cm and Switch length (L_S) = 5cm. The payload and servo will be embedded in the base wall.

$$L_C = L_B + L_R + L_S \tag{7.1}$$

$$L_C = 12 + 5 + 5 = 22cm$$

Therefore, the cabin length is **22cm**.

Cabin Height (H_C)

The component with the highest height is the battery whose height is around 7cm. A tolerance of 1cm was left for holding the battery to the fuselage and to allow few wires. This made the cabin height to be **8cm**.

STEP 11: Cargo/Load Compartment Design – Nil

STEP 12: Required Volume for Other Compartment – Nil

STEP 13: Space for Other Compartment – Nil

STEP 14: Fuselage Maximum Diameter (D_F)

The proposed material for construction is Styrofoam with a thickness of 2cm. This implies that the wall thickness (T_W) is 2cm.

Therefore, fuselage maximum width is **10cm** ($2T_W + W_C$) and fuselage maximum height is **12cm** ($2T_W + H_C$).

STEP 15: Number of Doors – two (cockpit windshield and aft fuselage)

STEP 16: Fuselage Nose Section

The fuselage nose length to maximum fuselage diameter ratio (L_N/D_F) should be between 1.5 and 2 and was selected to be 1.75. This makes the nose length to be **21cm**.

As stated earlier, the nose is meant to accommodate the engine. To do this and insure ease of manufacturing, the nose section of the fuselage was designed as shown below.

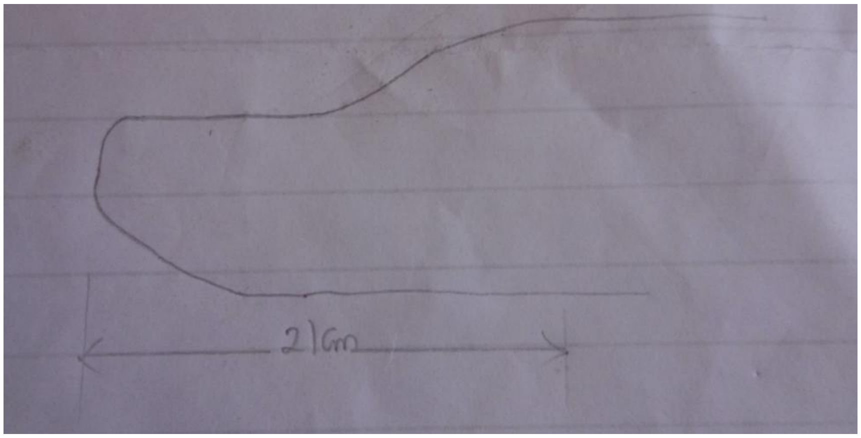


Figure 7.1: Sketch of fuselage nose section

STEP 17: FUSELAGE REAR SECTION

As recommendation, the fuselage is desired to follow an airfoil shape to reduce drag. From top view, it is desired to be symmetric airfoil shape. To achieve this, the tail section was tapered to zero diameter (from top view) to form a cone-like shape. To avoid flow separation, the cone angle was selected to be as small as **12deg**.

$$tan(\alpha_{cone}) = \frac{W_F/2}{L_R} \tag{7.2}$$

$$L_R = \frac{W_F/2}{tan(\alpha_{cone})} = \frac{10/2}{tan12} = 23.5cm$$

STEP 18: UPSWEEP ANGLE

For takeoff clearance, during takeoff rotation about the main landing gear (tricycle gear configuration), the fuselage aft section was expected not to touch the ground under normal flight operation and the upsweep angle was recommended to be less than 20deg (as seen from side view). The horizontal tail is to be mounted at the aft fuselage section and was designed to be some centimeters below the wing vertical location. To get this achieved, total upsweep wasn't employed and the same approach as step 17 was used. For the fuselage rear section (from side view) to have the same length as in step 17, the upsweep angle was calculated as:

$$tan(\alpha_{us}) = \frac{D_F/2}{L_R} \tag{7.3}$$

$$(\alpha_{us}) = tan^{-1}\left(\frac{D_F/2}{L_R}\right) = tan^{-1}\left(\frac{12/2}{23.5}\right) = 14.3deg$$

Since the upsweep angle is less than 20deg, it was left unaltered.

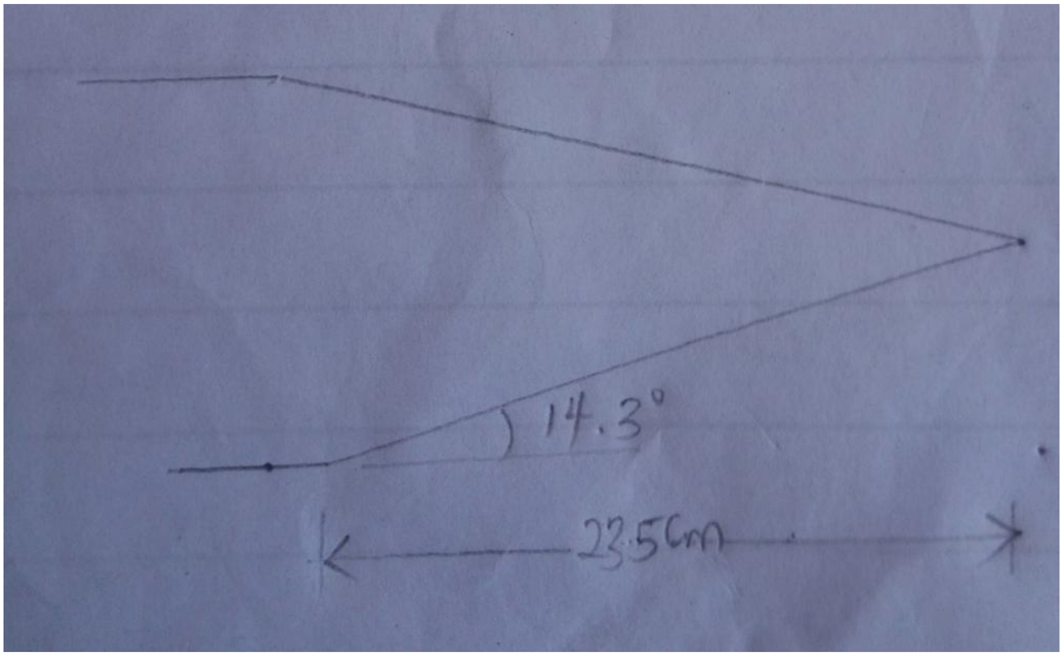


Figure 7.2: Sketch of fuselage tail section

STEP 19: Fuselage Overall Length (L_F)

$$L_F = L_N + L_C + L_R = 21 + 22 + 23.5 = 66.5cm \quad (7.4)$$

STEP 20: Check for Design Requirement Satisfaction

The fuselage length is 66.5cm and the maximum diameter is 12cm. Considering the horizontal distance between wing root leading edge and horizontal tail trailing edge (47.26cm), there is no sufficient space for the wing on the fuselage (cabin). Therefore, this design requirement is not satisfied.

To satisfy the requirement and to give room for other components positioning to put the aircraft center of gravity (CG) in the selected position, cabin length was increased by **10cm** which increases the fuselage length to **76.5cm**.

With this, the fuselage length-to-diameter ratio is 6.375 which is close to the selected value (i.e. 6).

STEP 21: FUSELAGE DRAWING

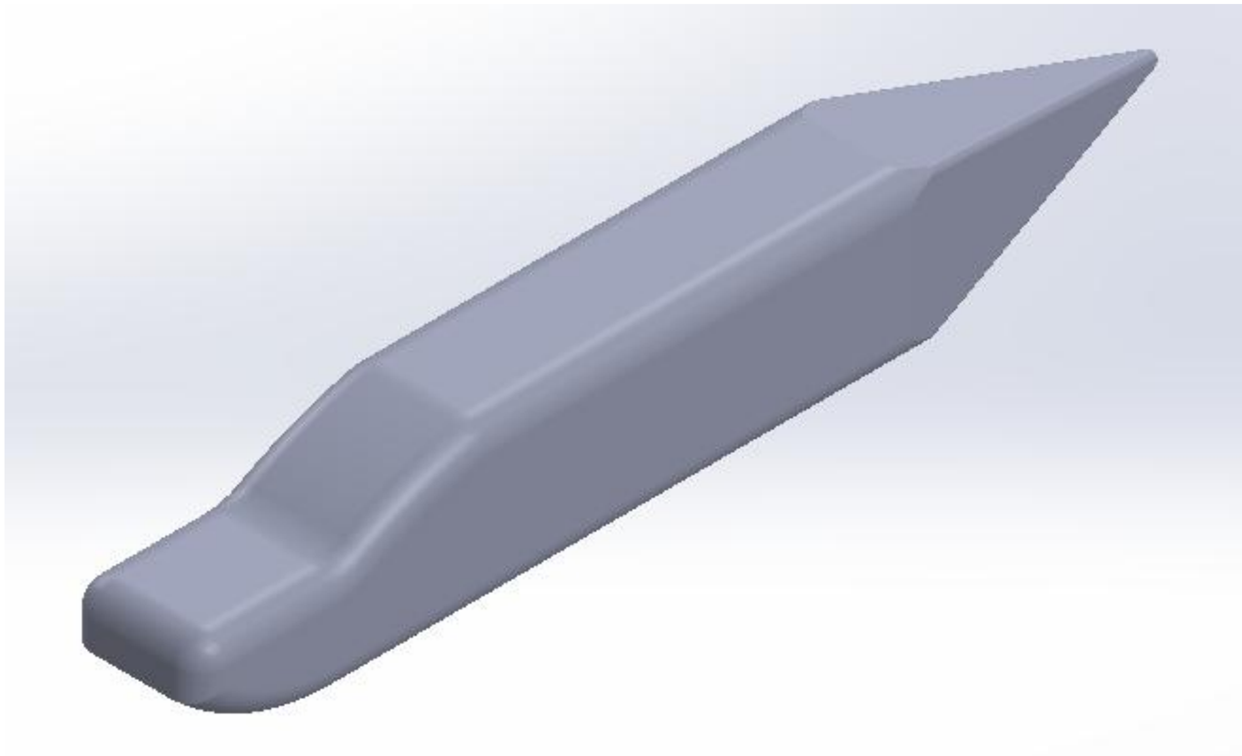


Figure 7.3: Fuselage final drawing

CHAPTER 8

PROPULSION SYSTEM DESIGN

The propulsion system was expected to produce a maximum power of 22.7W and the propeller efficiency was expected to be around 85%. Below are the design steps taken in designing the propulsion system.

STEP 1: Design Requirements

1. Aircraft performance (speeds)
2. Engine cost
3. Operating cost
4. Engine efficiency
5. Maintainability
6. Engine weight
7. Aircraft stability
8. Noise and vibration

STEP 2: Engine Type

To satisfy the requirements stated above, especially engine weight, engine cost and operating cost, the engine type was selected to be **electric motor**. The reasons being that it is light in weight, low cost and no special operating cost (since the engine will be using a rechargeable battery).

STEP 3: Numbers of Engine – One.

STEP 4: Engine Location

Since there's only a single engine, the options are for the aircraft to be a tractor or a pusher. But the decision of being a tractor was made because of longitudinal stability and reduction of weight as there will be need to compensate for weight or make weight distribution tasking/challenging in order to make the aircraft longitudinally stable. Therefore, the engine will be buried in the **fuselage nose**.

STEP 5: Engine Selection

From remote controlled (RC) airplane store, an electric motor with a power of around 30W was selected which was inherently designed to be compatible with a propeller with a diameter of 10 inches (25.4cm) and pitch 4.5inches.

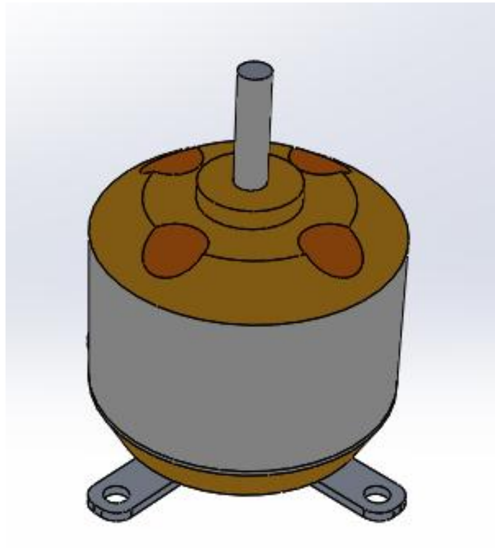


Figure 8.1: Electric motor

STEP 6: Propeller Design – There is an already design propeller for the selected electric motor.

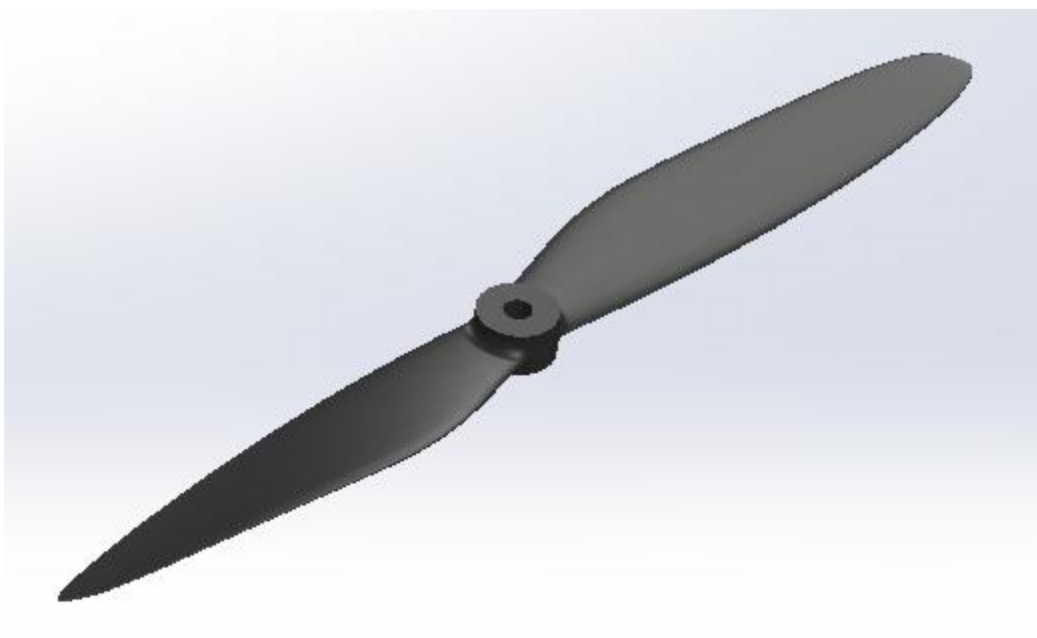


Figure 8.2: Propeller

STEP 7: Inlet Design – Nil

STEP 8: Engine Installation Design

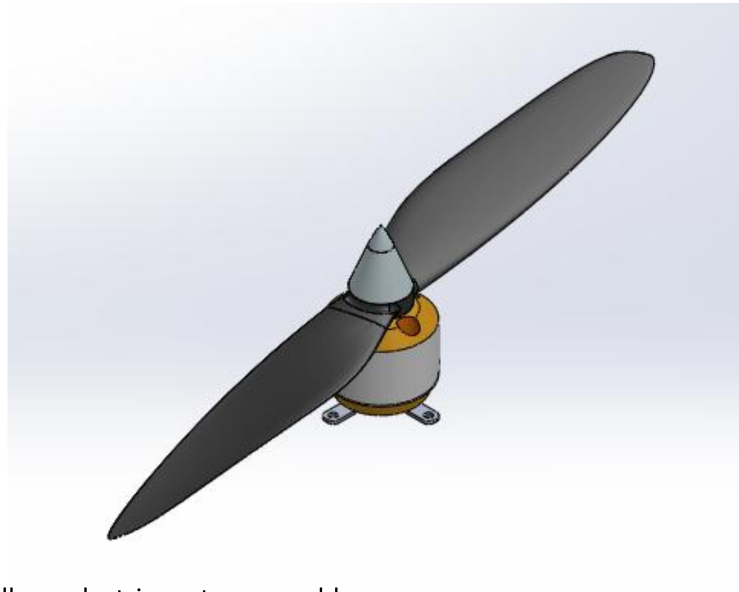


Figure 8.3: Propeller – electric motor assembly

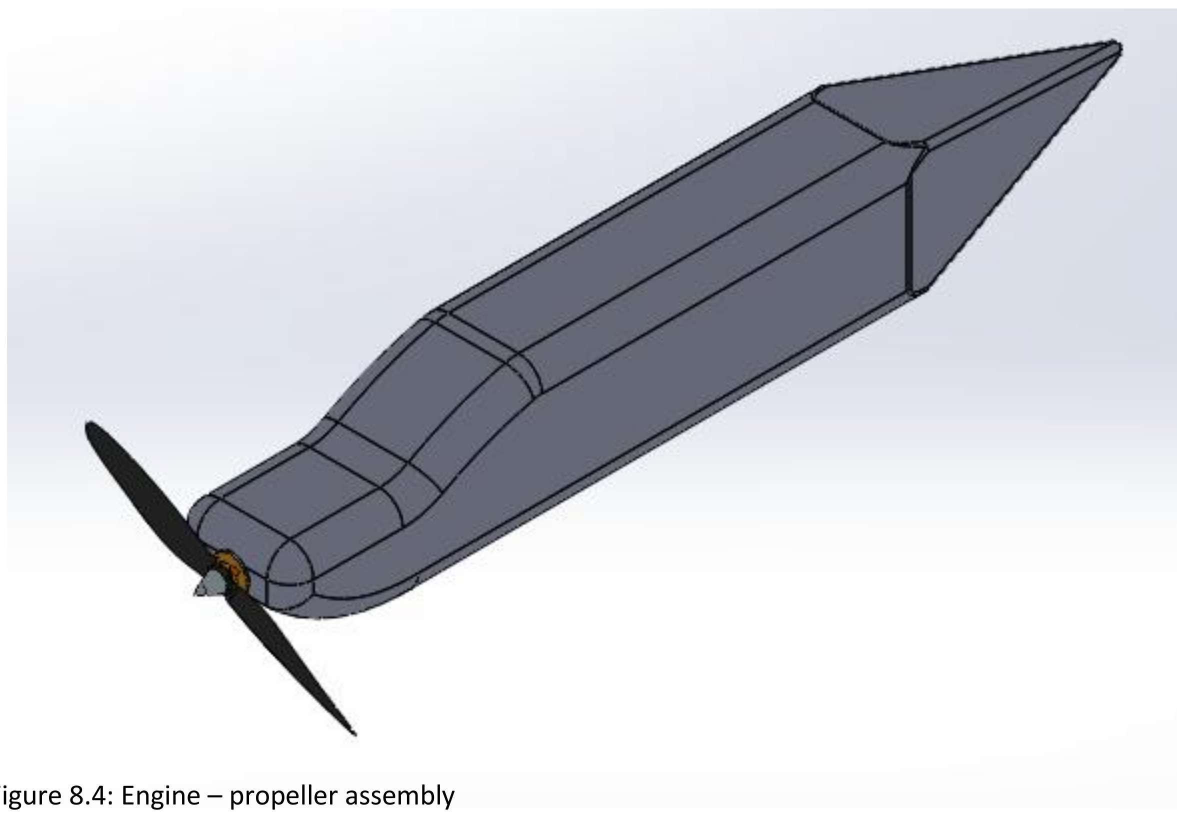


Figure 8.4: Engine – propeller assembly

CHAPTER 9

LANDING GEAR DESIGN

The landing gear was designed to enable the aircraft to take-off and land safely on a concrete floor, carry the aircraft loads (static and dynamic) and other requirements that will be stated later in the process. The following parameters have been obtained during the earlier design phases and were employed in this section:

1. m_{TO} = 1.2kg
2. Vs = 8m/s
3. K = 0.0551
4. S = 0,1878m^2
5. S_h = 0.0245m^2
6. Cm_{ac_wf} = -0.077
7. MAC = 0.164m
8. MACh: 0.0724m
9. CL_{TO} = 1.1105
10. CL_h = -0.374
11. L_h = 0.3773m
12. Xcg = 20% of MAC
13. Xac = 25%
14. CD_{o_TO} = 0.0495
15. Dprop = 25.4cm
16. Pmax = 30W
17. ηp_{TO} = 0.5
18. α_{fus_TO}=5.27deg
19. μ (concrete floor)= 0.04
20. Iyy = 0.61kgm^2
21. Takeoff pitch angular rate = 12deg/s^2

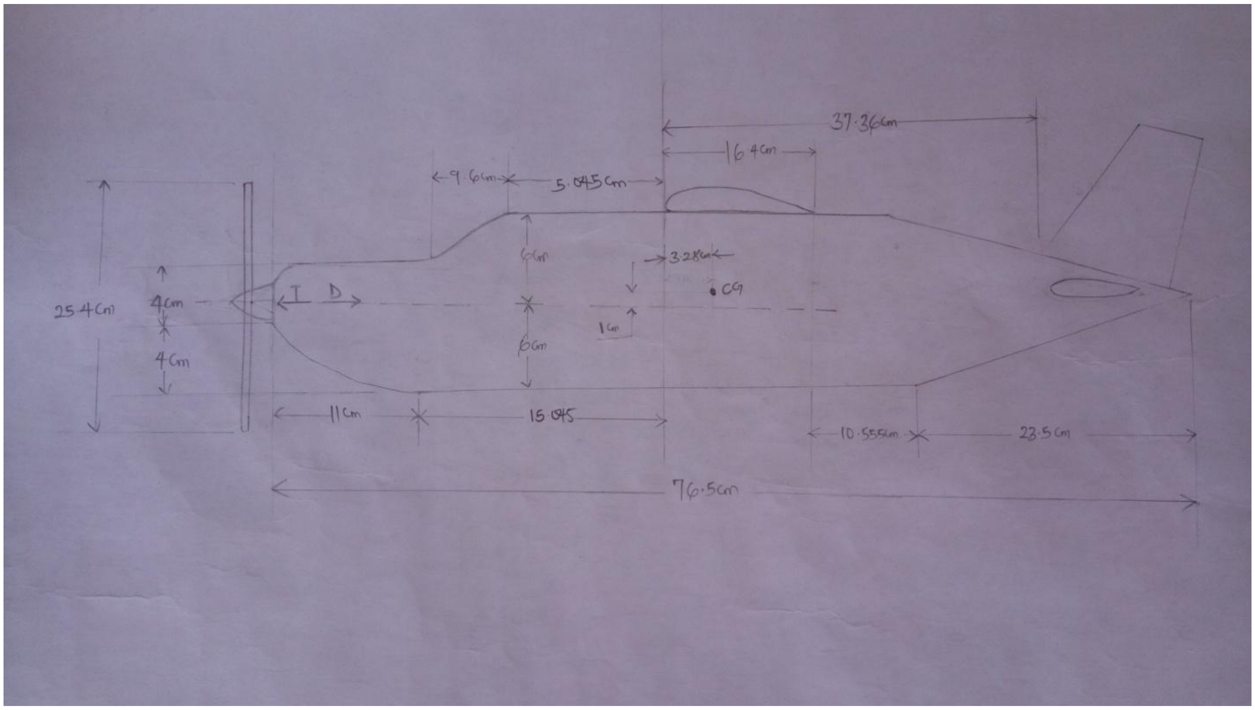


Figure 9.1: Fuselage/Engine sketch with dimension

The following are the steps taken in landing gear design:

STEP 1: Design Requirement

The following are the design requirements that will be satisfied:

1. Ground clearance
2. Tipback angle
3. Takeoff rotation
4. Overturn angle
5. Aircraft ground stability
6. Aircraft ground controllability
7. Low cost
8. Maintainability
9. Manufacturability

STEP 2: Landing Gear Configuration – Tricycle

STEP 3: Landing Gear Type – Fixed

STEP 4: Fixed Type: Unfaired

STEP 5: Aircraft Forward and Aft Center of Gravity (Cg) Location

The aircraft CG is located at 20% of the wing MAC which gives a distance of **3.28cm** from the wing leading edge (LE). The aircraft CG is constant thus; there is no difference in the forward and aft CG location.

STEP 5: Calculation of Landing Gear Height Based On Ground Clearance Requirements

As can be seen from the figure above, propeller tip is the lowest point on the aircraft. To ensure safety, a clearance (ΔH_{clear}) of 15cm was specified for the prop tip. The distance between the CG and prop center (ΔCG) is 1cm, Propeller diameter is 10in (25.4cm).

$$\therefore H_{CG} = \Delta H_{clear} + \frac{D_{prop}}{2} + \Delta CG \tag{9.1}$$

$$H_{CG} = 15 + \frac{25.4}{2} + 1 = 28.7cm$$

Therefore, the distance between the CG point and ground is 28.7cm. Then, the main gear was assumed to be attached to the fuselage where the distance between the CG and the fuselage base (h_{CG}) is 7cm.

$$\therefore H_{LG} = H_{CG} - h_{CG} \tag{9.2}$$

$$H_{LG} = 28.7 - 7 = 21.7cm$$

Therefore, landing gear height is 21.7cm. The fuselage attachment and landing gear height will be reviewed if they do not satisfy other requirements.

STEP 7: Calculation of the Distance between the Main Gear and the Most Forward CG Position

The CG position is constant as stated earlier and is at an horizontal distance of 3.28cm from the wing LE

The maximum allowable distance between the main gear and the CG was obtained by considering the tree equilibrium governing equations of an aircraft which was simplified as:

$$x_{mg} = \frac{I_{yymg}\theta - D(Z_D - Z_{mg}) + T(Z_T - Z_{mg}) - M_{ac_{wf}} - ma(Z_{cg} - Z_{mg}) - W \cdot x_{cg} + L_{wf}x_{ac_{wf}} + L_h x_{ac_h}}{L_{wf} + L_h - W} \tag{9.3}$$

Each of the parameters in the equation above was obtained one after the other and was then substituted into the equation.

$$C_{D_{TO}} = C_{D_{O_{TO}}} + K \cdot C_{L_{TO}}^2 \tag{9.4}$$

$$C_{D_{TO}} = 0.0495 + 0.0551 \cdot 1.1105^2 = 0.1174$$

$$D = \frac{1}{2}\rho V_R^2 C_{D_{TO}} S \quad (9.5)$$

$$V_R = 1.1V_S = 1.1 \times 8 = 8.8m/s$$

Where V_R is the aircraft linear forward speed at the instant of rotation.

$$D = \frac{1}{2} \cdot 1.225 \cdot 8.8^2 \cdot 0.1174 \cdot 0.1878 = 1.046N$$

$$T = \frac{\eta_{P_{TO}} \times P_{max}}{V_R} \quad (9.6)$$

$$T = \frac{0.5 \times 30}{8.8} = 1.7045N$$

$$M_{ac_{wf}} = \frac{1}{2}\rho V_R^2 S C_{m_{ac_{wf}}} C \quad (9.7)$$

$$M_{ac_{wf}} = \frac{1}{2} \times 1.225 \times 8.8^2 \times 0.1878 \times (-0.077) \times 0.164 = -0.1125N \cdot m$$

From the first equilibrium equation, this was obtained:

$$a = \frac{T - D - F_f}{m} \quad (9.8)$$

$$F_f = \mu N \quad (9.9)$$

$$N = W - L_{TO} \quad (9.10)$$

$$L_{TO} = \frac{1}{2}\rho V_R^2 C_{L_{TO}} S \quad (9.11)$$

$$L_{TO} = \frac{1}{2} \cdot 1.225 \cdot 8.8^2 \cdot 1.1105 \cdot 0.1878 = 9.892N$$

$$N = (1.2 \times 9.81) - 9.892 = 1.88N$$

$$F_f = 0.04 \times 1.88 = 0.0752N$$

$$a = \frac{1.7045 - 1.046 - 0.0752}{1.2} = 0.486m/s$$

$$L_h = \frac{1}{2}\rho V_R^2 C_{L_h} S_h \quad (9.12)$$

$$L_h = \frac{1}{2} \cdot 1.225 \cdot 8.8^2 \cdot (-0.374) \cdot 0.0245 = -0.4346N$$

$$L_{wf} = L_{TO} - L_h \tag{9.13}$$

$$L_{wf} = 9.892 - (-0.4346) = 10.3266N$$

$Z_D - Z_{mg}$ implies the distance between the Drag force point and the ground which is 27.7cm (6cm + 21.7cm).

$Z_T - Z_{mg}$ implies the distance between the Drag force point and the ground which is 27.7cm (6cm + 21.7cm).

$Z_{cg} - Z_{mg}$ implies the distance between the Drag force point and the ground which is 28.7cm (1cm + 6cm + 21.7cm). This has also been calculated earlier.

X distances are measured relative to a reference point. The reference here was chosen to be the aircraft CG for simplicity. Therefore, the distance between the wing/fuselage aerodynamic centre and cg ($x_{ac_{wf}}$) was calculated as shown below since the wing is rectangular.

$$x_{ac_{wf}} = \left(X_{ac} - X_{cg}\right) \times MAC \tag{9.14}$$

$$x_{ac_{wf}} = (0.25 - 0.2) \times 16.4 = 0.82cm$$

Horizontal tail aerodynamic center location (x_{ac_h}) is located on the tail MAC which has to be transferred to the root chord for easy reference to the wing leading edge.

$$x_{ac_h} = l_h + x_{ac_{wf}} \tag{9.15}$$

$$x_{ac_h} = 37.75 + 0.82 = 38.55cm$$

The reference point is the CG point therefore, x_{cg} is zero.

$$x_{mg} = \frac{0.61 \cdot \frac{12}{57.3} - 1.046 \cdot (0.277) + 1.7045(0.277) - (-0.1125) - 1.2 \cdot 0.486(0.287) - 0 + (10.3266 \cdot 0.0082) + (-0.4346) \cdot 0.3855}{10.3266 + (-0.4346) - (11.772)} = 0.917m$$

In getting this value, the moment of inertia used is about the aircraft CG which has to be transferred to the main gear using parallel axis theorem and the x_{mg} was recalculated with the new moment of inertia value.

$$I_{xx_{mg}} = I_{xx_{cg}} + m\left(\sqrt{x_{mg}{}^2 + H_{CG}{}^2}\right)^2 \tag{9.16}$$

$$I_{xx_{mg}} = 0.61 + 1.2(0.0917^2 + 0.287^2) = 0.72$$

$$x_{mg} = \frac{0.72 \cdot \frac{12}{57.3} - 1.045 \cdot (0.277) + 1.7045(0.277) - (-0.1125) - 1.2 \cdot 0.4869(0.287) - 0 + (10.3266 \cdot 0.0082) + (-0.4346) \cdot 0.3855}{10.3266 + (-0.4346) - (11.772)}$$
$$= 0.1181m$$

The maximum distance from the CG and the main gear is 11.81cm. This value is quite much and was reduced to 6.5cm.

STEP 8: Calculation of the Distance between the Main Gear and Most Aft CG

The aircraft CG position is constant. Therefore, $Xcg_{aft} = Xcg_{for} = 0.2MAC$

STEP 9: Tipback Check

To prevent tipback during takeoff rotation, the tipback angle must be greater than the fuselage takeoff angle of attack by a minimum of 5deg.

$$\alpha_{tb} > \alpha_{TO} + 5 \qquad (9.17)$$

$$\alpha_{tb} = tan^{-1}\left(\frac{x_{mg_{aft}}}{H_{CG}}\right) \qquad (9.18)$$

Where $Xmg_{aft} = Xmg_{aft} = 6.5$cm and H_{CG} has been calculated earlier to be 28.7cm

$$\alpha_{tb} = tan^{-1}\left(\frac{6.5}{28.7}\right) = 12.76^{o}$$

Tipback angle is greater than the aircraft takeoff rotation (5.27°)

$$\alpha_{tb} > 5.27 + 5$$

STEP 10: Takeoff Rotation Clearance Check

The take-off rotation ground clearance requirement to prevent a fuselage hit is as follows:

$$\alpha_C \geq \alpha_{TO} \qquad (9.19)$$

In order to determine the clearance angle (α_c), the two distances required are (i) the main gear distance from the fuselage upsweep point (X_{mg-us}) and (ii) the distance between the fuselage lowest point and the ground (H_f).

$$x_{mg-LE} = x_{mg} + \left(X_{cg} \times MAC\right) \qquad (9.20)$$

$$x_{mg-LE} = 6.5 + (0.2 \times 16.4) = 9.78cm$$

$$x_{mg-us} = x_{LE-us} - x_{mg-LE} \qquad (9.21)$$

$$x_{mg-us} = (16.4 + 10.555) - 9.78 = 17.175cm$$

Where x_{mg-LE} is the distance between the main gear and wing LE; x_{LE-us} is the distance between the wing LE and the upsweep point; and x_{mg-us} is the distance between the main gear and the upsweep point.

The distance between the fuselage lowest point and the floor (H_f) is the same as the landing gear height (21.7cm)

$$\alpha_C = tan^{-1}\left(\frac{H_f}{x_{mg-us}}\right) \tag{9.22}$$

$$\alpha_C = tan^{-1}\left(\frac{21.7}{17.175}\right) = 51.64^{o}$$

Since the clearance angle ($\alpha_C = 51.64^{o}$) is greater than the takeoff rotation angle, the fuselage will not touch the ground.

STEP 11: Wheel Base Calculation

Based on ground controllability requirement, the nose gear is required not to carry less than about 5% of the total load and also not to carry more than about 20% of the total load. Thus, the main gear carries about 80 – 95% of the aircraft load. To meet this requirement, it was decided that the nose gear should carry **20%** of the total load and the main gear 80% of the total load.

To calculate the wheel base, the following equilibrium equations were employed:

$$\sum F_z = 0\,;\ F_n + F_m = W \tag{9.23}$$

$$\sum M_o = 0\,;\ F_n B - W B_m = 0 \tag{9.24}$$

$$\therefore F_n = \frac{B_m}{B} W \tag{9.25}$$

$$F_m = \frac{B_n}{B} W \tag{9.26}$$

Where, F_n and F_m are the static loads carried by the nose gear and main gear respectively, B_n and B_m are the distances from the nose gear and main gear to the CG respectively, B is the wheel base (distance between the main gear and the nose gear along the x-axis) and W is the aircraft weight.

The nose gear was said to carry 20% of the load (i.e. $F_n = 0.2W$) and B_m is the same as x_{mg} (6.5cm).

Equation 24 can be rewritten as:

$$B = \frac{B_m}{F_n} W = \frac{B_m}{0.2W} W = \frac{6.5}{0.2} = 32.5cm \tag{9.27}$$

Therefore, the wheel base is 32.5cm.

STEP 12: WHEEL TRACK CALCULATION

The three main design requirements which drive the wheel track, T, (distance between the wheels of the main gear along the lateral axis) are: (i) ground lateral control, (ii) ground lateral stability, and (iii) structural integrity. The overturn angle is the angle which is critical to the aircraft overturn.

The lateral distance between each main gear to the cg must be greater than 25 deg. Hence, 30deg was considered. The minimum allowable value for wheel track was calculated as:

$$tan(30) = \frac{T_{min}/2}{H_{CG}} \tag{9.28}$$

$$T_{min} = 2tan(30) \times H_{CG} = 2tan(30) \times 28.7 = 33.14cm$$

GROUND CONTROLLABILITY

This was considered so that the aircraft will not roll over during taxi. The force that can roll the aircraft over is the centrifugal force (F_C) which is created during a turn due to centripetal acceleration:

$$F_C = \frac{mV^2}{R} \tag{9.29}$$

Where V is the speed and R is the turn radius. The speed and turn radius were selected to be 6m/s and 7m respectively.

$$F_C = \frac{1.2 \times 6^2}{7} = 6.17N$$

The restoring moment of the aircraft weight is a function of the wheel track:

$$T_{GC} > 2\frac{F_C \cdot H_{CG}}{mg} \tag{9.30}$$

$$T_{GC} > 2 \times \frac{6.17 \times 28.7}{1.2 \times 9.81} > 30.08cm$$

The corresponding overturn angle is given as:

$$\varphi_{ot} > tan^{-1}\left(\frac{F_C}{mg}\right) \quad (9.31)$$

$$\varphi_{ot} > tan^{-1}\left(\frac{6.17}{1.2 \times 9.81}\right) = 27.66^{\circ}$$

Therefore, based on ground controllability, the wheel track should be greater than 30.08cm with a corresponding overturn angle greater than 27.66°

GROUND STABILITY

The consideration of ground stability is based on generation of restoring moment to balance the forces on an aircraft at ground created by crosswind. The cross-wind force (FW) on an aircraft may be modeled as a drag force and is calculated as follows:

$$F_W = \frac{1}{2}\rho V_W^2 A_S C_{D_S} \quad (9.32)$$

Where V_W represents the wind speed, and A_S represents the aircraft side area. The parameter CD_S is called the aircraft side drag coefficient and its value varies between 0.3 and 0.8.

CD_S was selected to be 0.55, V_w was assumed to be 20m/s and with the aid of solidworks, A_S was obtained to be 0.05m^2.

$$F_W = \frac{1}{2} \times 1.225 \times 20^2 \times 0.05 \times 0.55 = 6.7375N$$

The wheel track for ground stability I given as:

$$T_{GS} > 2\frac{F_W \cdot H_C}{W} \quad (9.33)$$

Where, H_C is the centroid height from the ground. The centroid is 7cm above the fuselage base. Therefore, H_C is 28.7cm (7cm + 21.7cm).

$$T_{GS} > 2 \times \frac{6.7375 \times 28.7}{11.772} > 32.85cm$$

Therefore, the wheel track based on ground stability should be greater than 32.85cm.

STEP 13: LANDING GEAR ATTACHMENT

Landing gear attachment has been stated earlier to be under the fuselage and it satisfies all requirements. The wheel base can be achieved with this attachment. Also the wheel height will make it easy to achieve the wheel track.

STEP 14: CALCULATION OF LOADS ON EACH GEAR

The load carried by the landing gear is of two types: (i) static load and (ii) dynamic load. The calculation of landing gear load involves the two loads and was calculated as below:

$$F_{n_{max}} = \frac{B_m}{B} W \tag{9.34}$$

$$F_{n_{max}} = \frac{6.65}{32.5} \cdot 11.772 = 2.409N$$

Therefore, the maximum nose gear static load is 2.409N

$$F_{n_{dyn}} = \frac{W|a_L|H_{CG}}{gB} \tag{9.35}$$

Where, Fn_{dyn} is nose gear dynamic load, H_{CG} is the CG height from ground and a_L is the landing gear deceleration and was selected to be -0.5m/s^2.

$$F_{n_{dyn}} = \frac{11.772 \cdot 0.5 \cdot 28.7}{9.81 \cdot 32.5} = 0.5298N$$

$$F_{m_{max}} = \frac{B_n}{B} W \tag{9.36}$$

Where, Fm_{max} is the main gear maximum static load.

$$B_n = B - B_m \tag{9.37}$$

$$B_n = 32.5 - 6.5 = 26cm$$

$$F_{m_{max}} = \frac{26}{32.5} \cdot 11.772 = 9.418N$$

$$F_{m_{dyn}} = \frac{W a_T H_{CG}}{gB} \tag{9.38}$$

Where, Fn_{dyn} is the main gear dynamic load and a_T is the aircraft takeoff acceleration and was calculated in STEP 7 to be 0.486

$$F_{m_{dyn}} = \frac{11.772 \cdot 0.486 \cdot 28.7}{9.81 \cdot 32.5} = 0.515N$$

The total load on each gear is the summation of both static and dynamic loads.

$$F_N = F_{n_{max}} + F_{n_{dyn}} \tag{9.39}$$

$$F_N = 2.409 + 0.5298 = 2.9388N$$

$$F_M = F_{m_{max}} + F_{m_{dyn}} \quad (9.40)$$

$$F_M = 9.481 + 0.515 = 10N$$

Therefore, the total loads carried by the nose gear and the main gear are 2.9388N and 10N respectively.

LANDING GEAR DESIGN RESULTS

1. Landing gear configuration – Tricycle
2. Landing gear type – Fixed
3. Aircraft CG location – 3.28cm from wing LE
4. CG height from ground – 28.7cm
5. Landing gear height from ground – 21.7cm
6. Main gear distance from CG – 6.5cm
7. Tipback angle – 12.76°
8. Takeoff rotation clearance – 51.64cm
9. Wheel base – 32.5cm
10. Minimum wheel track – 33.14cm
11. Wheel track based on ground controllability > 30.08cm
12. Overturn angle – 27.66°
13. Wheel track based on ground stability > 32.85cm
14. Landing gear attachment – under fuselage
15. Nose gear load – 2.9388N
16. Main gear load – 10N

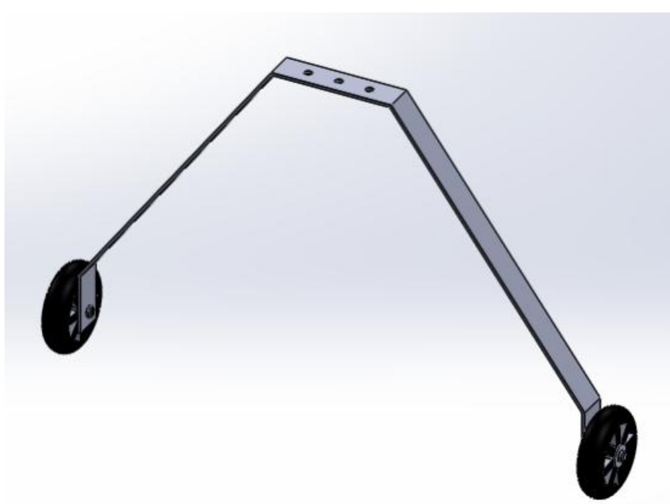


Figure 9.2: Main gear drawing

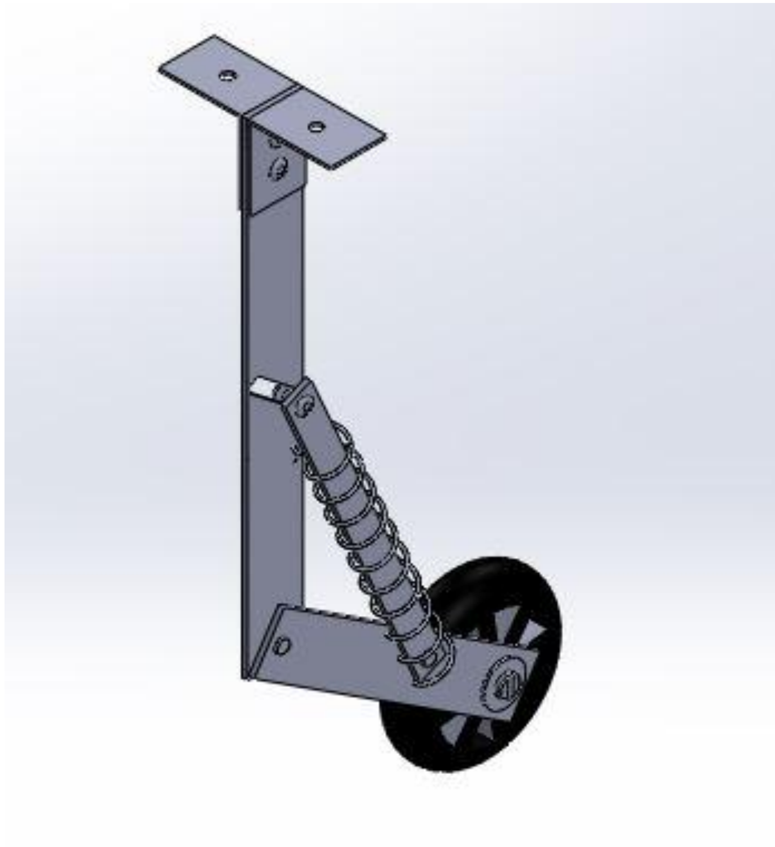


Figure 9.3: Nose gear drawing

CHAPTER 10

WEIGHT OF COMPONENTS

The weight of the components for this design was divided into:

1. Fuselage weight
2. Wing weight
3. Tail weight
4. Landing gear weight
5. Propulsion system weight
6. Payload and store weight
7. Others

The weight of the components was then analyzed using this division as guidelines. To obtain the weights, the density of the various materials to be used was obtained and volumes of each of the components were calculated. From these, the mass was obtained which was then converted to weight. The following are the steps taken.

FUSELAGE MASS

For ease in calculation of the fuselage volume, the views (side, top and bottom views) are divided into sections. This section form shapes assumed to be triangles and rectangle because they are similar to these shapes as shown in the figure below.

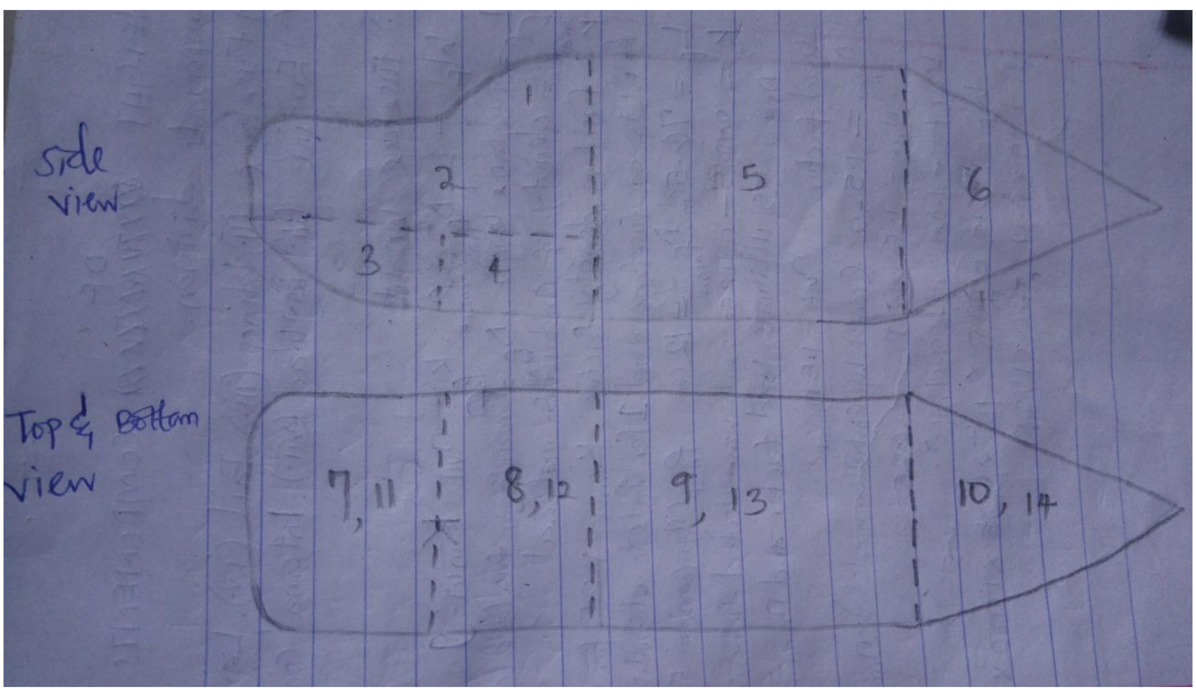


Figure 10.1: Fuselage sections sketch

The dimensions for these sections were obtained from the figure below

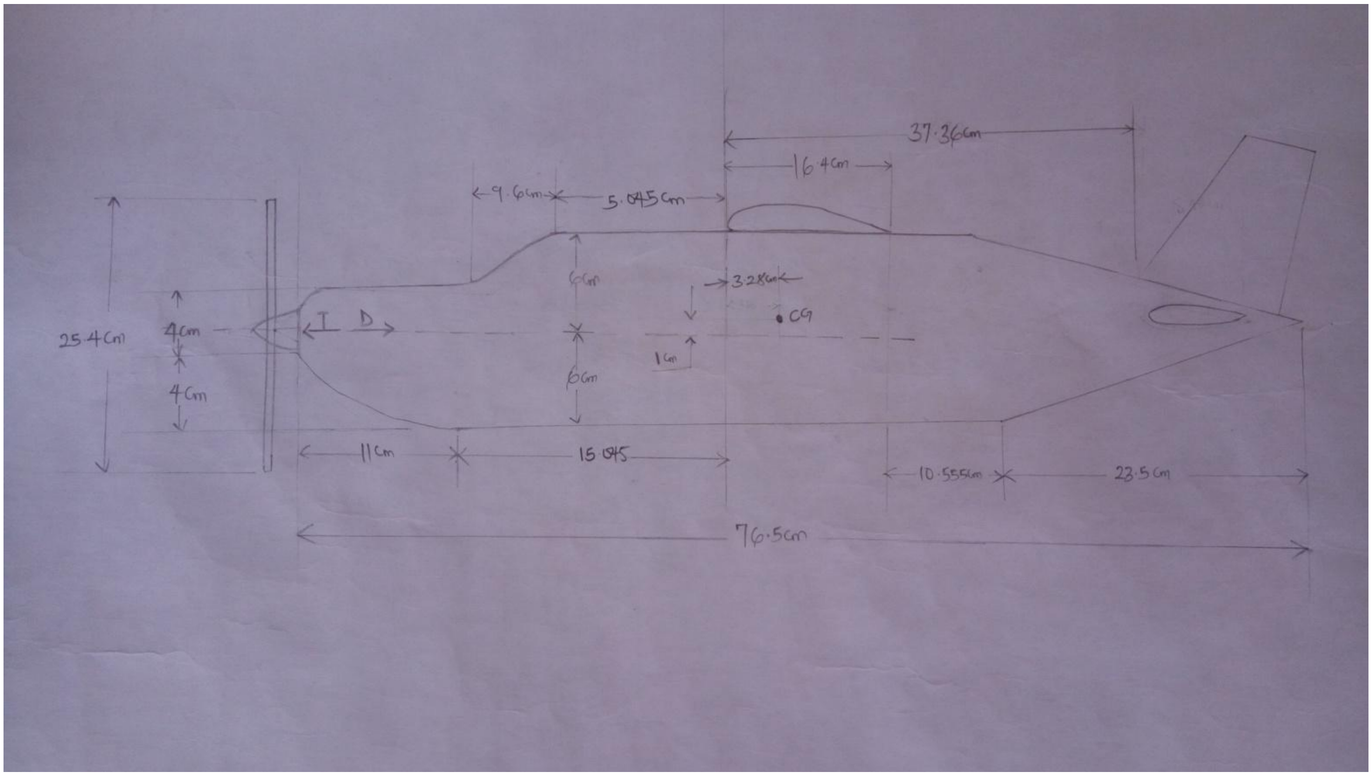


Figure 10.2: Aircraft sketch with dimension

Sections 1 to 6, 7 to 10 and 11 to 14 are for side view, top view and bottom view. The two sides of the aircraft are symmetrical. The wall thickness is 2cm. This will affect the values to be used for fuselage width in sections 7 to 14.

Section 1: $A_1 = \frac{1}{2} \times b \times h = \frac{1}{2} \times 9.6 \times 4 = 19.2cm^2$ (10.1)

Section 2: $A_2 = l \times h = 21 \times 4 = 84cm^2$ (10.2)

Section 3: $A_3 = \frac{1}{2} \times b \times h = \frac{1}{2} \times 11 \times 4 = 22cm^2$

Section 4: $A_4 = l \times h = 10 \times 4 = 40cm^2$

Section 5: $A_5 = l \times h = 32 \times 12 = 384cm^2$

Section 6: This section was treated as two triangles. The actual horizontal length was gotten from the top view which is equivalent to the slant length as seen from top view.

Therefore, from Pythagoras theorem,

$$b = \sqrt{\left(\frac{F_w}{2}\right)^2 + {L_R}^2} = \sqrt{\left(\frac{10}{2}\right)^2 + 23.5^2} = 24.03cm \quad (10.3)$$

$$A_6 = 2 \times \left(\frac{1}{2} \times b \times h\right) = 2 \times \left(\frac{1}{2} \times 24.03 \times \frac{12}{2}\right) = 144.18cm^2$$

Section 7: $A_7 = l \times w = 11 \times (10 - 2 - 2) = 66cm^2$ (10.4)

Section 8: From side view, it is seen that the length seen from top view is slant. So, Pythagoras theorem was used for the length.

$A_8 = l \times w = \sqrt{10^2 + 4^2} \times (10 - 2 - 2) = 64.62cm^2$

Section 9: $A_9 = l \times w = 32 \times (10 - 2 - 2) = 192cm^2$

Section 10: In this section, the same approach used in section 6 was used to calculate the length.

$$l = \sqrt{\left(\frac{F_h}{2}\right)^2 + {L_R}^2} = \sqrt{\left(\frac{12}{2}\right)^2 + 23.5^2} = 24.25cm \quad (10.5)$$

The sidewall thickness has taken part of the total geometry of this section as shown below.

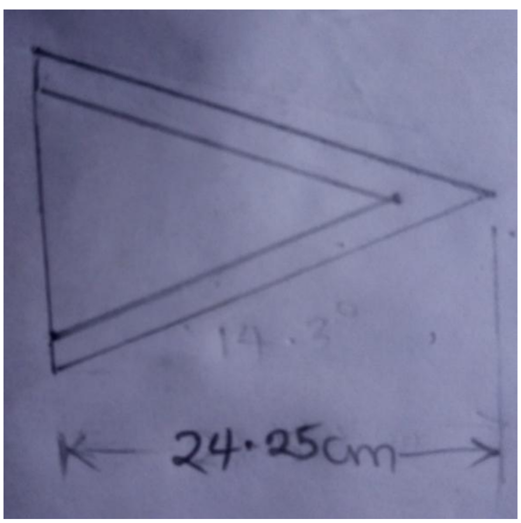


Figure 10.3: Aft fuselage sidewall thickness sketch

Therefore, $b = \left(\frac{10}{2} - 2\right) \times tan(90 - \alpha_{cone}) = 3 \times tan(90 - 12) = 14.114cm$

$$A_{10} = 2 \times \left(\frac{1}{2} \times b \times h\right) = 2 \times \left(\frac{1}{2} \times 14.114 \times \left(\frac{10}{2} - 2\right)\right) = 42.34cm^2$$

Section 11: $A_{11} = l \times w = \sqrt{11^2 + 4^2} \times (10 - 2 - 2) = 70.23cm^2$

Section 12: $A_{12} = l \times w = 10 \times (10 - 2 - 2) = 60cm^2$

Section 13: $A_{13} = A_9 = 192cm^2$

Section 14: $A_{14} = A_{10} = 42.34cm^2$

Total Fuselage Area (A_{fus})

$$A_{fus} = 2\sum_{i=1}^{6} A_i + \sum_{i=7}^{14} A_i \quad (10.6)$$

$$A_{fus} = 2(19.2 + 84 + 22 + 40 + 384 + 144.18) + (66 + 64.62 + 192 + 42.34 + 70.23 + 60 + 192 + 42.34)$$

$$A_{fus} = (2 \times 693.38) + 729.53 = 2116.29cm^2$$

Fuselage Volume (V_{fus})

$$V_{fus} = A_{fus} \times Wall\ thickness \quad (10.7)$$

$$V_{fus} = 2116.29 \times 2 = 4232.58cm^3$$

$$V_{fus} = 4232.58 \times 10^{-6} m^3 = 0.004233m^3$$

Fuselage material is Styrofoam, with density of 50Kg/m^3.

$$m_{fus} = \rho_{fus_{mat}} \times V_{fus} \quad (10.8)$$

$$m_{fus} = 50 \times 0.004233 = 0.21165Kg\ or\ 211.65g$$

<u>WING MASS</u>

Wing compositions are:

i. Six ribs
ii. One spar
iii. Covering
iv. Two ailerons
v. Two flaps
vi. Others (Servo, push-pull rod and miscellaneous)

<u>Six Ribs</u>

Parameters:

a. Airfoil section: Clark Y
b. MAC: 16.4cm

c. Airfoil Maximum thickness to chord: 11.71%
d. MAC airfoil section area, $A_{section}$: 21.78cm^2
e. Rib thickness, t_{rib}: 2cm
f. Rib material (Styrofoam) density: 50Kg/m^3

$$V_{rib} = A_{section} \times t = 21.78 \times 2 = 43.56cm^3 \tag{10.9}$$

$$m_{rib} = \rho_{rib_{mat}} \times V_{rib} \tag{10.10}$$

$$m_{rib} = 50 \times 43.56 \times 10^{-6} = 0.0022Kg$$

Six ribs mass = 6(0.0022) = 0.0132Kg (13.2g)

<u>Spar</u>

Parameters:

a. Spar thickness: 0.5cm
b. Height: 1.7cm
c. Spar length = 125cm (fuselage occupied space was considered)
d. Spar material density: 200Kg

$$V_{spar} = t_{spar} \times h_{spar} \times l_{spar} = 0.5 \times 1.7 \times 125 = 106.25cm^3 \tag{10.11}$$

$$m_{spar} = \rho_{spar_{mat}} \times V_{spar} \tag{10.12}$$

$$m_{spar} = 200 \times 106.25 \times 10^{-6} = 0.02125Kg$$

Spar mass = 0.02125Kg (21.25g)

<u>Covering</u>

Parameters:

a. Wing span: 125cm (adding fuselage width to wetted span)
b. MAC airfoil section arc length: 33.52cm
c. Covering material density: 100Kg/m^3
d. Covering thickness: 0.2cm

$$V_{cov} = S_{cov} \times l_{cov} \times t_{cov} = 125 \times 33.52 \times 0.2 = 838cm^3 \tag{10.13}$$

$$m_{cov} = \rho_{cov_{mat}} \times V_{cov} \tag{10.14}$$

$$m_{cov} = 100 \times 838 \times 10^{-6} = 0.0838Kg$$

Covering mass =0.0838Kg (83.8g)

Two ailerons

The two ailerons mass is assumed to be 10% of rib, spar and covering mass.

$$\therefore m_{ail} = m_{rib} + m_{spar} + m_{cov} \qquad (10.15)$$

$$m_{ail} = 0.1 \times (0.0132 + 0.02125 + 0.0838) = 0.1(0.11825) = 0.011825Kg$$

Ailerons mass = 0.011825Kg (11.83g)

Two flaps

The two flaps mass is assumed to be 15% of rib, spar and covering masses.

$$\therefore m_{flap} = m_{rib} + m_{spar} + m_{cov} \qquad (10.16)$$

$$m_{ail} = 0.15 \times (0.0132 + 0.02125 + 0.0838) = 0.15(0.11825) = 0.01774Kg$$

Flaps mass = 0.01774Kg (17.74g)

Others

A servo mass = 9g. Therefore, 4servos =36g

A push-pull rod mass = 4g, 4 push-pull rods = 16g

Miscellaneous 10g

Total others mass =62g (0.062Kg)

TOTAL WING MASS, m_{wing} = 0.01320 + 0.02125 + 0.0838 + 0.011825 + 0.01774 + 0.062 = 0.2098Kg

TAIL MASS

Horizontal Tail Mass

Horizontal tail compositions are:

i. Five ribs
ii. One spar
iii. Covering
iv. Elevator

v. Others (Servo, push-pull rod and miscellaneous)

Five Ribs

Parameters:

a. Airfoil section: NACA 0010
b. MAC: 7.24cm
c. Airfoil Maximum thickness to chord: 10%
d. MAC airfoil section area, $A_{section}$: 3.59cm^2
e. Rib thickness, t_{rib}: 1cm
f. Rib material (styrofoam) density: 50Kg/m^3

$$V_{rib} = A_{section} \times t = 3.59 \times 1 = 3.59cm^3 \quad (10.17)$$

$$m_{rib} = \rho_{rib_{mat}} \times V_{rib} \quad (10.18)$$

$$m_{rib} = 50 \times 3.59 \times 10^{-6} = 0.0001795Kg$$

Five ribs mass = 5(0.0001795) = 0.0008975Kg (0.9g)

Spar

Parameters:

a. Spar thickness: 0.5cm
b. Height: 0.6cm
c. Spar length = 33.8cm
d. Spar material density: 200Kg

$$V_{spar} = t_{spar} \times h_{spar} \times l_{spar} = 0.5 \times 0.6 \times 33.8 = 10.14cm^3 \quad (10.19)$$

$$m_{spar} = \rho_{spar_{mat}} \times V_{spar} \quad (10.20)$$

$$m_{spar} = 200 \times 10.14 \times 10^{-6} = 0.002028Kg$$

Spar mass = 0.002028Kg (2.03g)

Covering

Parameters:

a. Horizontal tail area: 0.0245m^2
b. MAC: 7.24cm
c. Covering span: 33.84cm (area/MAC)
d. MAC airfoil section arc length: 14.71cm

e. Covering material density: 100Kg/m^3
f. Covering thickness: 0.2cm

$$V_{cov} = S_{cov} \times l_{cov} \times t_{cov} = 33.84 \times 14.71 \times 0.2 = 99.56cm^3 \quad (10.21)$$

$$m_{cov} = \rho_{cov_{mat}} \times V_{cov} \quad (10.22)$$

$$m_{cov} = 100 \times 99.56 \times 10^{-6} = 0.009956Kg$$

Covering mass =0.009956Kg (9.96g)

<u>Elevator</u>

The elevator mass is assumed to be 20% of rib, spar and covering mass.

$$\therefore m_{ele} = m_{rib} + m_{spar} + m_{cov} \quad (10.23)$$

$$m_{ail} = 0.2 \times (0.0008975 + 0.002028 + 0.009956) = 0.2(0.0128815) = 0.0025763Kg$$

Elevator mass = 0.0026Kg (2.6g)

<u>Others</u>

A servo mass = 9g, 2servos =18g

A push-pull rod mass = 4g, 2 push-pull rods = 8g

Miscellaneous = 5g

Total others mass =31g (0.031Kg)

TOTAL HORIZONTAL TAIL MASS, m_{HT} = 0.0008975 + 0.002028 + 0.009956 + 0.0026 + 0.031 = 0.0464815Kg

<u>Vertical Tail Mass</u>

Vertical tail compositions are:

i. Four ribs
ii. One spar
iii. Covering
iv. Rudder
v. Others (Servo, push-pull rod and miscellaneous)

<u>Four Ribs</u> Parameters:

a. Airfoil section: NACA 0008

b. MAC: 11.5cm
c. Airfoil Maximum thickness to chord: 8%
d. MAC airfoil section area, $A_{section}$: 7.26cm^2
e. Rib thickness, t_{rib}: 1cm
f. Rib material (styrofoam) density: 50Kg/m^3

$$V_{rib} = A_{section} \times t = 7.26 \times 1 = 7.26cm^3 \quad (10.24)$$

$$m_{rib} = \rho_{rib_{mat}} \times V_{rib} \quad (10.25)$$

$$m_{rib} = 50 \times 7.26 \times 10^{-6} = 0.000363Kg$$

Four ribs mass = 4(0.000363) = 0.001452Kg (1.45g)

<u>Spar</u>

Parameters:

a. Spar thickness: 0.4cm
b. Height: 0.5cm
c. Spar length = 17.3cm
d. Spar material density: 200Kg

$$V_{spar} = t_{spar} \times h_{spar} \times l_{spar} = 0.4 \times 0.5 \times 17.3 = 3.46cm^3 \quad (10.26)$$

$$m_{spar} = \rho_{spar_{mat}} \times V_{spar} \quad (10.27)$$

$$m_{spar} = 200 \times 3.46 \times 10^{-6} = 0.000692Kg$$

Spar mass = 0.000692Kg (0.7g)

<u>Covering</u>

Parameters:

a. Vertical tail area: 0.02m^2
b. MAC: 11.5cm
c. Covering span: 17.4cm (area/MAC)
d. MAC airfoil section arc length: 23.23cm
e. Covering material density: 100Kg/m^3
f. Covering thickness: 0.2cm

$$V_{cov} = S_{cov} \times l_{cov} \times t_{cov} = 17.4 \times 23.23 \times 0.2 = 80.8404cm^3 \quad (10.28)$$

$$m_{cov} = \rho_{cov_{mat}} \times V_{cov} \quad (10.29)$$

$$m_{cov} = 100 \times 80.8404 \times 10^{-6} = 0.00808404Kg$$

Covering mass =0.00808404Kg (8.08g)

Rudder

The rudder mass is assumed to be 20% of rib, spar and covering mass.

$$\therefore m_{ele} = m_{rib} + m_{spar} + m_{cov} \tag{10.30}$$

$$m_{ail} = 0.2 \times (0.001452 + 0.000692 + 0.00808404) = 0.2(0.0102) = 0.002Kg$$

Elevator mass = 0.002Kg (2g)

Others

A servo mass = 9g

A push-pull rod mass = 4g

Miscellaneous = 5g

Total others mass =18g (0.018Kg)

TOTAL VERTICAL TAIL MASS, m_{VT} = 0.001452 + 0.000692 + 0.00808404 + 0.002 + 0.018 = 0.0302Kg

TOTAL TAIL MASS, m_{tail} = VERTICAL TAIL + HORIZONTAL TAIL = 0.0302 + 0.0464815 = 0.07668Kg

LANDING GEAR MASS

Landing strut material is aluminum with density of 2711Kg/m^3

Nose Gear

Nose gear strut components:

i. Vertical components
ii. Slant component
iii. End-of-travel (E.O.T) rod/spring guide rod
iv. Fuselage attachment

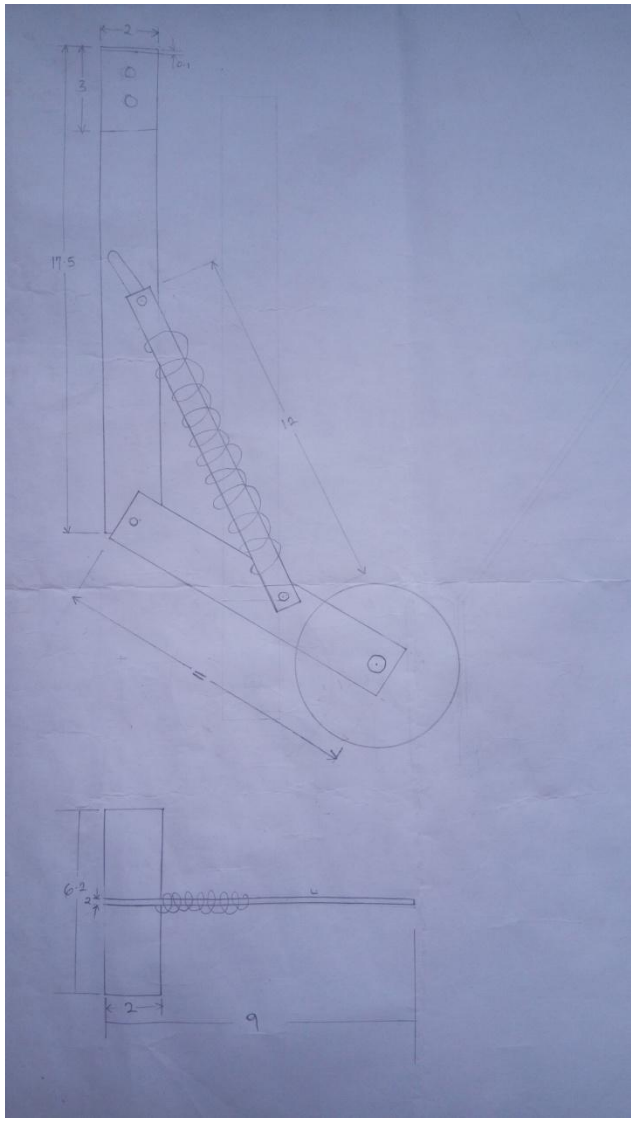


Figure 10.4: Nose gear sketch with dimension

Vertical component

Length: 17.5cm, Breadth: 2cm and thickness: 2mm

$$V_{vert} = l \times b \times t = 17.5 \times 2 \times 0.2 = 7cm^3 \quad (10.31)$$

Slant component

Length: 11cm, Breadth: 2cm and thickness: 2mm

$$V_{slant} = l \times b \times t = 11 \times 2 \times 0.2 = 4.4cm^3 \quad (10.31)$$

E.O.T rod

Length: 12cm, Breadth: 1cm and thickness: 2mm

$$V_{E.O.T.} = l \times b \times t = 12 \times 1 \times 0.2 = 2.4cm^3 \quad (10.31)$$

Fuselage attachment

Length: 6cm, Breadth: 2cm and thickness: 1mm

$$V_{fus} = l \times b \times t = 6 \times 2 \times 0.1 = 1.2cm^3 \quad (10.31)$$

Total nose gear strut volume, $V_{NGS} = V_{vert} + V_{slant} + V_{E.O.T.} + V_{fus}$ (10.32)

$$V_{NGS} = 7 + 4.4 + 2.4 + 1.2 = 15cm^3$$

$$m_{NGS} = \rho_{NGS_{mat}} \times V_{NGS} \quad (10.33)$$

$$m_{NGS} = 2711 \times 15 \times 10^{-6} = 0.040665Kg$$

<u>Others</u>

a. Spring mass = 15g
b. Tire = 25g
c. Bolts, nuts washers and rivets = 30g
d. Miscellaneous = 10g

Nose gear mass, m_{NG} = 40.7 + 15 + 25 + 30 + 10 = 120.7g (0.1207Kg)

<u>Main Gear</u>

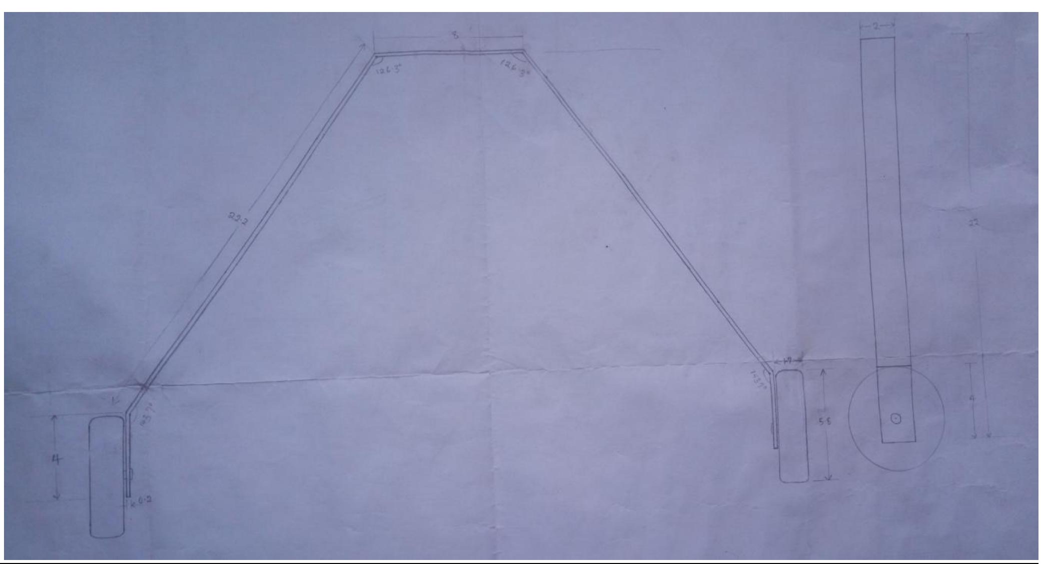


Figure 10.5: Main gear sketch with dimension

Strut parameters mass:

Thickness = 2mm, width = 2cm and Length = 2(4 + 22.2) + 8 = 60.4cm.

$$V_{strut} = t \times w \times l = 0.2 \times 2 \times 60.4 = 24.16cm^3$$

$$m_{strut} = \rho_{strut_{mat}} \times V_{strut}$$

$$m_{strut} = 2711 \times 24.16 \times 10^{-6} = 0.0655Kg$$

Others

a. Tire (2) = 50g
b. Bolts, nuts and washers = 35g
c. Miscellaneous = 15g

Main gear mass, m_{MG} = 65.5 + 50 + 35 + 15 = 165.5g (0.1655Kg)

<u>Landing gear mass, m_{LG} = Nose gear mass + Main gear mass = 0.1207 + 0.1655 = 0.2862Kg</u>

PROPULSION SYSTEM MASS

Electric motor = 47g, Propeller = 7.5g and Engine mount = 20g

Total mass, m_{PS} = 47 + 7.5 + 20 = 74.5g (0.0745Kg)

STORE AND PAYLOAD MASS

i. Payload = 10g
ii. Store casing 15g
iii. Servo = 9g
iv. push-pull rod = 4g

Total mass, m_s = 10 + 15 + 9 + 4 = 38g (0.038Kg)

INTERNAL COMPONENTS AND OTHERS

i. Receiver = 8g
ii. Battery = 240g
iii. Electronic speed controller (ESC) = 23g
iv. Lightening =30g
v. Wires = 30g
vi. Switch = 10g

Total mass, m_{int} = 8 + 240 + 23 + 50 + 30 + 10 = 341g (0.361Kg)

Total aircraft mass, $m_{aircraft} = m_{fus} + m_{wing} + m_{tail} + m_{LG} + m_{PS} + m_{S} + m_{int}$ (10.34)

$$m_{aircraft} = 211.65 + 209.8 + 76.68 + 286.2 + 74.5 + 38 + 341 = 1237.83g$$

Therefore, total mass = 1237.83g = 1.238Kg

Aircraft weight, $W_{aircraft} = m_{aircraft} \times g = 1.238 \times 9.81 = 12.14N$ (10.35)

TOTAL AIRCRAFT WEIGHT = 12.14N

CHAPTER 11

AIRCRAFT WEIGHT DISTRIBUTION

This section covers the calculation of aircraft center of gravity (CG) coordinates. To do this, the centers of gravity of each component were calculated and the distances from a reference point was considered. The details are discussed in the following steps:

STEP 1: Reference Line

The x-axis reference line is selected to be the prop spinner tip, y-axis reference is fuselage center line and z-axis reference line is the ground. The CG coordinate along the y-axis is assumed to be zero because the aircraft is designed to be right-left symmetric and the components are also distributed symmetrically. Therefore, only x and z axis are considered.

STEP 2: Coordinate Axis System – Body axis system

STEP 3: Wing Leading Edge Distance from Reference, X_{WLE-R}

The wing leading edge is at a distance of 26.045cm from the fuselage nose and the spinner tip at a distance of 4cm from fuselage nose. Therefore, X_{WLE-R} is **30.045cm**.

STEP 4: Wing CG Coordinate

The wing CG (on x-axis) is assumed to be at 35% MAC and wing MAC is 16.4cm.

$$X_W = 0.35MAC + X_{W-R} \tag{11.1}$$

$$X_W = (0.35 \times 16.4) + 30.045 = 35.785cm$$

Wing airfoil maximum thickness is 11.71% of MAC, fuselage height, H_{fus} is 12cm and landing gear height is 22cm. Therefore, the y-axis coordinate for the wing CG is calculated as:

$$Z_W = \left(0.5 \times \left({}^{t}/_{C}\right)_{max} \cdot MAC\right) + H_{fus} + H_{LG} \tag{11.2}$$

$$Z_W = (0.5 \times 0.1171 \times 16.4) + 12 + 22 = 34.96$$

STEP 5: Fuselage CG Location

The fuselage CG location was calculated to be at 34cm from the nose tip and 6cm. Therefore, X_{fus} is 38cm from the x-reference line and Z_{fus} is 28cm from z--reference line.

STEP 6: Horizontal Tail CG

Horizontal tail CG is assumed to be at 30% of its MAC. From horizontal tail design, the following results were obtained:

1. Horizontal tail root chord, C_{hr} = 8.44cm
2. Horizontal tail MAC, MAC_h = 7.24cm
3. Distance from tail root leading edge to wing root leading edge, $X_{HTLE-WLE}$ = 38.82cm

$$X_{HT} = 0.3MAC_h + \left(C_{h_r} - MAC_h\right) + X_{HTLE-WLE} + X_{WLE-R} \tag{11.3}$$

$$X_{HT} = (0.3 \times 7.24) + (8.44 - 7.24) + 38.82 + 30.045 = 72.24cm$$

The chord line of the horizontal tail coincides with the fuselage center line. Therefore, Z_{HT} is 28cm.

STEP 7: Vertical Tail CG

Vertical tail CG is assumed to be at 30% of its MAC. From vertical tail design, the following results were obtained:

1. Vertical tail root chord, C_{vr} = 13.1cm
2. Vertical tail MAC, MAC_v = 11.5cm
3. Distance from tail root leading edge to wing root leading edge, $X_{HTLE-WLE}$ = 37.355cm

$$X_{VT} = 0.3MAC_v + \left(C_{v_r} - MAC_v\right) + X_{VTLE-WLE} + X_{WLE-R} \tag{11.4}$$

$$X_{VT} = (0.3 \times 11.5) + (13.1 - 11.5) + 37.355 + 30.045 = 72.45cm$$

Using similar approach as for the wing, Z_{VT} is calculated to be 38.654cm

STEP 8: Main Gear CG Location

The main gear x-CG location coincides with its point of attachment. The aircraft CG was assumed to be at 3.28cm from wing leading edge during landing gear design and main gear was designed to be at 6.5cm from that CG position. Therefore;

$$X_{MG} = x_{mg} + 3.28 + X_{WLE-R} \tag{11.4}$$

$$X_{MG} = 6.5 + 3.28 + 30.045 = 39.825cm$$

Z_{MG} is assumed to be 10cm.

STEP 9: Nose Gear CG Location

The wheel base is 32.5cm. Therefore;

$$X_{NG} = X_{MG} - B\text{r} \tag{11.5}$$

$$X_{NG} = 39.825 - 32.5 = 7.325cm$$

Z_{NG} is assumed to be 8cm.

STEP 10: Engine CG Location

The whole propulsion system CG is at approximately 5.5cm from the x-reference line (i.e. X_{PS} = 5.5cm) and coincides with the fuselage center line (i.e. Z_{PS} = 28cm).

STEP 11: Store CG Location

The store CG is expected to be at the assumed CG location.

$$X_S = 3.28 + X_{NG} \tag{11.6}$$

$$X_S = 3.28 + 30.045 = 33.325cm$$

Z_S is 1cm above the fuselage base. Therefore, Z_S is 23cm.

STEP 12: Battery CG Location

X_{BAT} is selected to be 16cm and Z_{BAT} is calculated to be 26cm

STEP 13: Other Components CG Location

The location for this is assumed to be at the assumed CG location and 5cm from fuselage base. Therefore, X_O is 33.325 and Z_O is 26.5cm

STEP 14: Calculation OF CG Location

The masses and CG distances are summarized in the table below

S/N	Component	Mass (g)	X_{CG} (cm)	Z_{CG} (cm)
1	Fuselage	211.65	38	28
2	Wing	209.8	35.785	34.96
3	Vertical Tail	30.2	72.45	378.654
4	Horizontal Tail	46.482	72.24	28
5	Main Gear	165.5	39.826	10
6	Nose Gear	120.7	7.325	8
7	Propulsion System	74.5	5.5	28

8	Store	38	33.325	23
9	Battery	240	16	26
10	Others	101	33.325	27

Table 11.1: Components masses and CG locations

$$X_{CG} = \frac{\sum_{i=1}^{n} m_i X_{CG_i}}{\sum_{i=1}^{n} m_i} \quad (11.7)$$

$$X_{CG} = \frac{m_{fus}X_{fus} + m_{wing}X_W + m_{VT}X_{VT} + m_{HT}X_{HT} + m_{MG}X_{MG} + m_{NG}X_{NG} + m_{PS}X_{PS} + m_S X_S + m_{BAT}X_{BAT} + m_O X_O}{m_{fus} + m_{wing} + m_{VT} + m_{HT} + m_{MG} + m_{NG} + m_{PS} + m_S + m_{BAT} + m_O} \quad (11.8)$$

$$X_{CG} = \frac{(211.65 \times 38) + (209.8 \times 35.785) + (30.2 \times 72.45) + (46.482 \times 72.24) + (165.5 \times 39.826) + (120.7 \times 7.325) + (74.5 \times 5.5) + (38 \times 33.325) + (240 \times 16) + (101 \times 33.325)}{211.65 + 209.8 + 30.2 + 46.482 + 165.5 + 120.7 + 74.5 + 38 + 240 + 101}$$

$$X_{CG} = \frac{37453}{1237.8} = 30.257cm$$

Similarly,

$$Z_{CG} = \frac{\sum_{i=1}^{n} m_i Z_{CG_i}}{\sum_{i=1}^{n} m_i} \quad (11.9)$$

$$Z_{CG} = \frac{m_{fus}Z_{fus} + m_{wing}Z_W + m_{VT}Z_{VT} + m_{HT}Z_{HT} + m_{MG}Z_{MG} + m_{NG}Z_{NG} + m_{PS}Z_{PS} + m_S Z_S + m_{BAT}Z_{BAT} + m_O Z_O}{m_{fus} + m_{wing} + m_{VT} + m_{HT} + m_{MG} + m_{NG} + m_{PS} + m_S + m_{BAT} + m_O} \quad (11.10)$$

$$Z_{CG} = \frac{(211.65 \times 28) + (209.8 \times 34.96) + (30.2 \times 38.654) + (46.482 \times 28) + (165.5 \times 10) + (120.7 \times 8) + (74.5 \times 28) + (38 \times 23) + (240 \times 26) + (101 \times 26.5)}{211.65 + 209.8 + 30.2 + 46.482 + 165.5 + 120.7 + 74.5 + 38 + 240 + 101}$$

$$Z_{CG} = \frac{30227}{1237.8} = 24.42cm$$

STEP 15: CG Location on Wing MAC

The CG location on wing MAC is calculated as:

$$x_{cg} = \frac{X_{CG} - X_{WLE-R}}{MAC} \quad (11.11)$$

$$x_{cg} = \frac{30.257 - 30.045}{16.4} = 0.013$$

This implies that the aircraft CG is at 1.3% of wing MAC

STEP 16: Weight Distribution to Reposition CG Location

In other to reposition the aircraft CG location, some weights have to be repositioned. To keep this simple and easier, battery was repositioned. Via trial and error, when the battery was repositioned to 31.8cm from the reference line, the value of X_{CG} is 33.3216.

$$x_{cg} = \frac{33.3216 - 30.045}{16.4} = 0.1998 \cong 0.2$$

Therefore, the aircraft CG, when X_{BAT} is 31.8cm, is located at 20% of wing MAC.

The CG coordinates (x, y, z) using body axis system is **(33.32, 0, 24.42)** in centimeters from their reference axes.

STEP 17: Stability and Control Check

The lateral, longitudinal and directional stability and control was considered during wing and tail design. The result will remain as considered earlier, since the CG location remains the same.

STEP 18: Aircraft Moment of Inertia Check

The aircraft longitudinal, lateral and directional mass moments of inertia about the x, y and z axis are calculated to be:

Ixxcg = 0.03517Kgm^2, Iyycg = 0.042Kgm^2 and Izzcg = 0.058Kgm^2

STEP 19: Aircraft Components Redesign

The aircraft parameters that changed are the CG coordinates and moments of inertial which were used during landing gear design. The calculated change in the aircraft mass is 37g which might cause some changes in the aircraft performance but its value will be worked with.

With a CG height of 2.42cm from fuselage base and lateral mass moment of inertia about the aircraft cg (Iyycg) of 0.042Kgm^2, the landing gear was redesigned and the following are the new parameters.

LANDING GEAR REVIEWED DESIGN RESULTS

1. Landing gear configuration – Tricycle
2. Landing gear type – Fixed
3. Aircraft CG location – 3.28cm from wing LE
4. CG height from ground – 24.12cm
5. Landing gear height from ground – 21.7cm
6. Mass moment of inertial about main gear, Iyymg = 0.11140Kgm^2
7. Main gear distance from CG – 5.06cm
8. Tipback angle – 11.87°
9. Main gear distance from wing leading edge, X_{MG-LE} – 8.34cm
10. Takeoff rotation clearance – 49.4°
11. Wheel base – 25.32cm
12. Minimum wheel track – 27.85cm
13. Wheel track based on ground controllability > 25.29cm
14. Overturn angle – 27.67°
15. Wheel track based on ground stability > 32.85cm
16. Landing gear attachment – under fuselage
17. Nose gear load – 2.93N
18. Main gear load – 10N

CHAPTER 12

CONTROL SURFACES DESIGN

Considering the tables below, the aircraft is considered to be of class I and level of acceptability – 1.

Class	Aircraft characteristics
I	Small, light aircraft (maximum take-off mass less than 6000 kg) with low maneuverability
II	Aircraft of medium weight and low-to-medium maneuverability (maximum take-off mass between 6000 and 30 000 kg)
III	Large, heavy, and low-to-medium maneuverability aircraft (maximum take-off mass more than 30 000 kg)
IV	Highly maneuverable aircraft, no weight limit (e.g., acrobatic, missile, and fighter)

Table 12.1: Aircraft Classes

Level	Definition
1	Flying qualities clearly adequate for the mission flight phase.
2	Flying qualities adequate to accomplish the mission flight phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.
3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A flight phases can be terminated safely, and Category B and C flight phases can be completed.

Table 12.2: Levels of acceptability

AILERON DESIGN

The following parameters and some new others are used for aileron design

i. b_f =60% of b (68.8cm)
ii. C_f = 20% of MAC (3.28cm)
iii. b = 1.15m

iv. C or MAC $= C_r = C_t = 16.4cm$
v. $C_{L\alpha W} = 4.8276 rad^{-1}$
vi. $S = S_W = 0.1878m^2$
vii. $S_h = 0.0245m^2$
viii. $S_{Vt} = 0.02m^2$
ix. $\lambda = 1$
x. V_{app} or $V_{land} = 10.4m/s$

STEP 1: Design Requirement

The following are the requirements the roll control surface should satisfy

i. Maneuverability
ii. Roll controllability
iii. Ease of manufacturing
iv. Low cost

STEP 2: Roll Control Surface Configuration

Due to design simplicity, low cost and ease of manufacturing, Aileron is selected

STEP 3: Maneuverability and Roll Control Requirement

The table below was considered for the requirement which specifies the time to achieve a specific bank angle.

STEP 4: Aircraft Class and Critical Flight Phase for Roll Control

Aircraft class is class I and the critical flight phase for roll control is at the lowest speed (towards landing/approach). From the table below, Landing/approach falls under phase C

(a) Time to achieve a specified bank angle change for Class I			
Level	Flight phase category		
	A	B	C
	Time to achieve a bank angle of 60° (s)	Time to achieve a bank angle of 45° (s)	Time to achieve a bank angle of 30° (s)
1	1.3	1.7	1.3
2	1.7	2.5	1.8
3	2.6	3.4	2.6

Table 12.3: Roll control requirement

Category	Examples of flight operation
A	(i) Air-to-air combat (CO); (ii) ground attack (GA); (iii) weapon delivery/launch (WD); (iv) aerial recovery (AR); (v) reconnaissance (RC); (vi) in-flight refueling (receiver) (RR); (vii) terrain following (TR); (viii) anti-submarine search (AS); (xi) close formation flying (FF); and (x) low-altitude parachute extraction system (LAPES) delivery.
B	(i) Climb (CL); (ii) cruise (CR); (iii) loiter (LO); (iv) in-flight refueling in which the aircraft acts as a tanker (RT); (v) descent (D); (vi) emergency descent (ED); (vii) emergency deceleration (DE); and (viii) aerial delivery (AD).
C	(i) Take-off (TO); (ii) catapult take-off (CT); (iii) powered approach (PA); (iv) wave-off/go-around (WO); and (v) landing (L).

Table12. 4: Flight phase categories

STEP 5: Handling Quality Design Requirements

The roll control handling qualities design requirement is identified from Table 3, which states that the aircraft in Class I, flight phase C, for a level of acceptability of 1, is required to be able to achieve a bank angle of 30◦ in 1.3 seconds.

STEP 6: Aileron Inboard and Outboard Positions

Considering the flap location, the inboard and outboard aileron locations are tentatively selected to be 70% and 95% of the wing span respectively (i.e. $b_{ai}/b = 0.7$ and $b_{ao}/b = 0.95$)

STEP 7: Aileron Chord to Wing Chord Ratio

Aileron to wing chord ratio, C_a/C is tentatively selected to be 20%.

STEP 8: Aileron Effectiveness Parameter

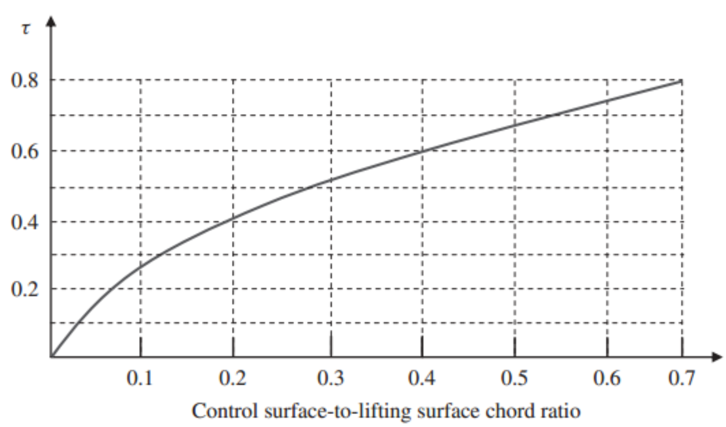


Figure 12.1: Control surface angle of attack effectiveness parameter

Knowing the aileron chord to wing chord ratio to be 0.2, the aileron effectiveness parameter, τ_a, was selected to be 0.41 using the table above.

STEP 9: Aileron Rolling Moment Coefficient Derivative

To calculate the aileron rolling moment coefficient derivative, $Cl_{\delta A}$, the following equation was used.

$$C_{l_{\delta_A}} = \frac{2C_{L_{\alpha W}}\tau C_r}{Sb}\left[\frac{y^2}{2}+\frac{2}{3}\left(\frac{\lambda-1}{b}\right)y^3\right]_{y_i}^{y_o} \tag{12.1}$$

$$C_{l_{\delta_A}} = \frac{2C_{L_{\alpha W}}\tau C_r}{Sb}\left[\left(\frac{{y_o}^2}{2}+\frac{2}{3}\left(\frac{\lambda-1}{b}\right){y_o}^3\right)-\left(\frac{{y_i}^2}{2}+\frac{2}{3}\left(\frac{\lambda-1}{b}\right){y_i}^3\right)\right] \tag{12.2}$$

Where, $C_{L\alpha W}$ is the wing lift curve slope, τ is the effectiveness parameter, C_r is the wing root chord, S is the wing area, b is the wing span, λ is the wing taper ratio, y_o and y_i are the inboard and outboard aileron positions respectively, relative to the fuselage centerline.

$$y_o = 0.95 \times \frac{b}{2} = 0.95 \times \frac{1.15}{2} = 0.5463m \tag{12.3}$$

$$y_i = 0.7 \times \frac{b}{2} = 0.7 \times \frac{1.15}{2} = 0.4025m \tag{12.4}$$

$$C_{l_{\delta_A}} = \frac{2 \times 4.8276 \times 0.41 \times 0.164}{0.1878 \times 1.15}\left[\left(\frac{0.5463^2}{2}+\frac{2}{3}\left(\frac{1-1}{1.15}\right)0.5463^3\right) - \left(\frac{0.4025^2}{2}+\frac{2}{3}\left(\frac{1-1}{1.15}\right)0.4025^3\right)\right] = 0.2050rad^{-1}$$

STEP 10: Maximum Aileron Deflection

The maximum aileron deflection, δ_{max} is selected to be ±25°.

STEP 11: Aircraft Rolling Moment Coefficient

The aircraft rolling moment coefficient, C_l was calculated using the following equation:

$$C_l = C_{l_{\delta_A}} \cdot \delta_A \tag{12.5}$$

The parameter $Cl_{\delta A}$ is referred to as the aircraft rolling moment coefficient due to aileron deflection derivative and is also called the aileron roll control power and δ_A is the aileron deflection which is selected to be the maximum deflection in this calculation.

$$C_l = 0.2050 \times \frac{25}{57.3} = 0.0894rad^{-1}$$

STEP 12: Aircraft Rolling Moment

The aerodynamic rolling moment is generally modeled as a function of the wing area (S), wing span (b), and dynamic pressure (q) as:

$$L_A = qSC_l b \tag{12.6}$$

$$q = \frac{1}{2}\rho {V_T}^2 \tag{12.7}$$

The critical flight phase is towards landing. Therefore, V_T was set to be the equal to V_{land} (10.4m/s).

$$q = \frac{1}{2} \cdot 1.225 \times 10.4^2 = 66.248 N/m^2$$

$$L_A = 66.248 \times 0.1878 \times 0.0894 \times 1.15 = 1.2796 Nm$$

STEP 13: Steady-State Roll Rate

The steady state roll rate, P_{SS} was calculated using the following equation:

$$P_{SS} = \sqrt{\frac{2 \cdot L_A}{\rho(S_W + S_h + S_{Vt}) C_{D_R} \cdot {y_D}^3}} \tag{12.8}$$

Where S_W, S_h and S_{Vt} are the wing, horizontal tail and vertical areas, y_D is the rolling drag center along the y-axis and the center of gravity and C_{DR} is the aircraft drag coefficient in rolling motion. This coefficient is about 0.7-1.2, which includes the drag contribution of the fuselage.

The areas have been specified earlier, parameter y_D is assumed to be 40% of the wing span and C_{DR} is selected to be 0.95.

$$P_{SS} = \sqrt{\frac{2 \cdot 1.2796}{1.225 \cdot (0.1878 + 0.0245 + 0.02) \times 0.95 \cdot \left(0.4 \frac{1.15}{2}\right)^3}} = 27.9 \text{rad/s}$$

STEP 14: Bank Angle for Steady State Roll Rate

$$\varphi = \frac{\mathrm{I_{xx}}}{\rho(S_W + S_h + S_{Vt}) C_{D_R} \cdot {y_D}^3} \ln({P_{SS}}^2) \tag{12.9}$$

$$\varphi_1 = \frac{0.03517}{1.225 \cdot (0.1878 + 0.0245 + 0.02) \times 0.95 \cdot \left(0.4 \frac{1.15}{2}\right)^3} ln(27.9^2) = 71.18 rad \ (4078.31°)$$

STEP 15: Aircraft Roll Rate Produced by Aileron Rolling Moment Until Aircraft Reaches Steady State Roll Rate

This parameter is expressed as:

$$\mathrm{P} = \frac{P_{SS}{}^2}{2\varphi_1} \tag{12.10}$$

$$\mathrm{P} = \frac{27.9^2}{2 \times 71.8} = 5.47\mathrm{rad/s^2}$$

STEP 16: Time to Reach Desired Bank Angle

The bank angle, ϕ_1 (4078.31°) is greater than the required bank angle, ϕ_{req} (30°) in step 5.

Time to reach the desired bank angle is expressed as:

$$t_2 = \sqrt{\frac{2\varphi_{des}}{P}} \tag{12.11}$$

$$t_2 = \sqrt{\frac{2 \times 30}{5.47 \times 57.3}} = 0.44s$$

STEP 17: Roll Time Comparison

The required roll time specified to reach a bank angle of 30° (in step 5) is 1.3s and the time to achieve this bank angle (as calculated in step 16) is 0.44s. Since the calculated roll time (0.44s) is less than the required roll time (1.3s), the aileron design satisfies the requirement.

STEP 18: Aileron Stall

With the use of a software, XFOIL, the aileron (with a $C_a/C = 0.2$) at a deflection, δ of 25° stalls at 8° angle of attack as shown in the figure below.

STEP 19: Adverse Yaw

Adverse yaw occurs as a result of the difference in the local lift and thus, induced drag on each sides of the wing. This difference in the induced drag gives rise to a corresponding opposite yawing moment which makes the bank/turn not coordinated. To avoid such undesirable yawing motion, the solution selected is to employ a simultaneous aileron/rudder deflection so as to eliminate the adverse yaw which requires an interconnection between the aileron and the rudder.

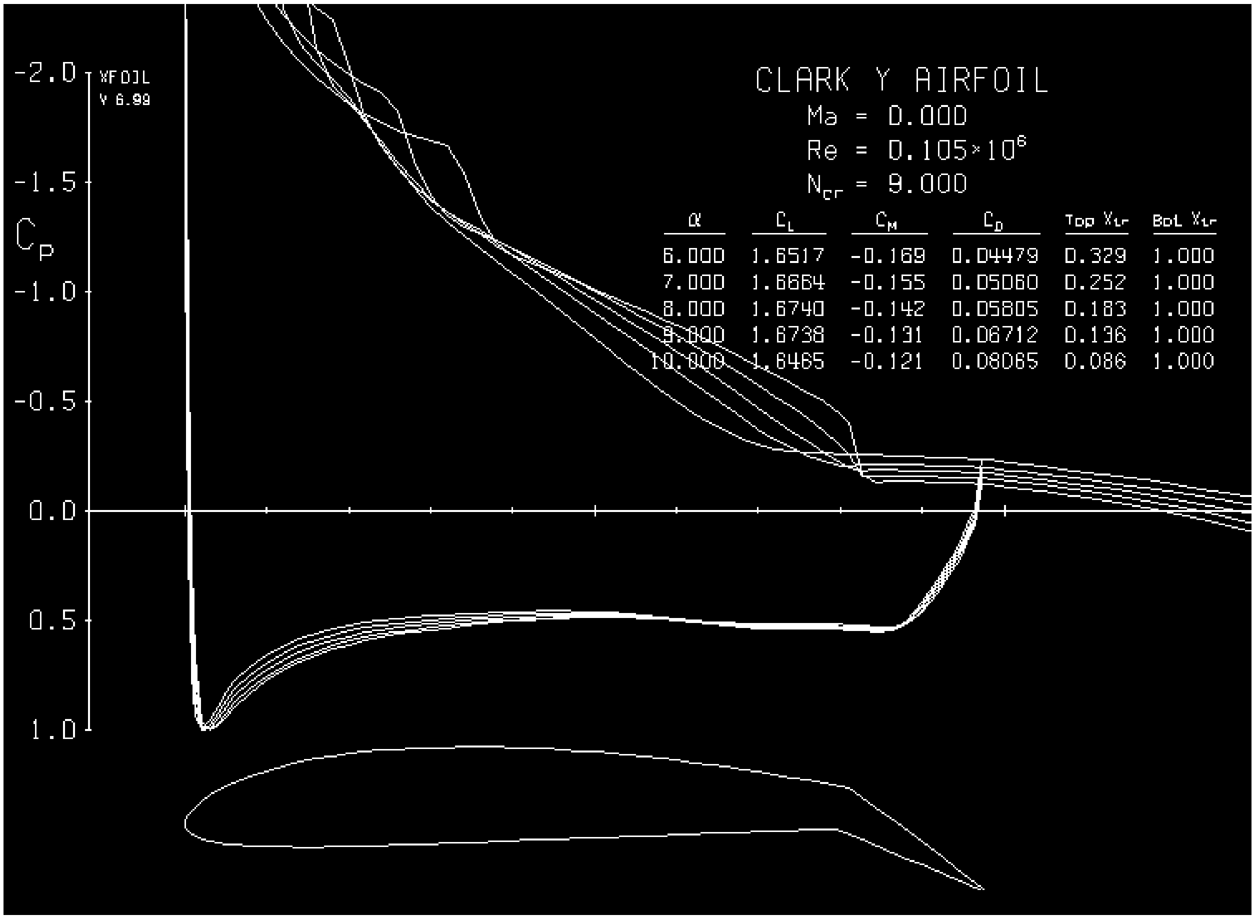


Figure 12.2: Aileron stall using XFOIL

STEP 20: Aileron Reversal

This is a phenomenon where the wing losses its lift at high speed due to aeroelastic twists and the deflected aileron produces a rolling moment in a reversed direction. The solution selected to avoid this is to make the wing stiffer and to limit aileron deflection to about 17° at high speed.

STEP 21: Aerodynamic Balance/Mass Balance

The type of aerodynamic balance selected is the overhanging balance – where the hinge line is moved aft (a bit) towards the aileron center of pressure. The leading edge of the control surface will be elliptical. To insure mass balance and avoid flutter, the control surface center of gravity will be moved towards or in front of the hinge line.

STEP 22: Aileron Span, Chord and Area

Aileron span, b_a, chord, C_a and area, A_a are calculated as follows:

$$b_a = y_o - y_i \tag{12.12}$$

$$b_a = 0.5463 - 0.4025 = 0.1438m$$

$$C_a = 0.2C \tag{12.13}$$

$$C_a = 0.2 \times 0.164 = 0.0328m$$

$$S_a = 2 \times b_a \times C_a \tag{12.14}$$

$$S_a = 2 \times 0.1438 \times 0.0328 = 0.0094m^2$$

AILERON DESIGN RESULT

1. y_o =54.63cm
2. y_i = 40.25cm
3. δ_{max} = 25°
4. ϕ_{des} = 30°
5. t_2 = 0.44s
6. adverse yaw solution – simultaneous aileron/rudder deflection
7. aileron reversal – stiffer wing and δ_{max} = 17° at high speed
8. Aerodynamic balance – overhanging balance with elliptical aileron leading edge.
9. Mass balance – aileron CG movement towards or in front of hinge line.
10. b_a = 14.38cm
11. C_a = 3.28cm
12. S_a (both sides) = 0.0094m^2

ELEVATOR DESIGN

The following parameters and some new others are used for elevator design:

i. $\ddot{\theta}$ =12deg/s^2
ii. L_{wf} = 10.327N
iii. D = 1.046N
iv. T = 1.7045N
v. M_{ac_wf} = – 0.1125Nm
vi. a = 0.486m/s^2
vii. m =1.2Kg
viii. $x_{mg\text{-}cg}$ = 5.06cm
ix. $x_{mg} - x_{ac\_wf}$ = 4.24cm
x. $x_{ach} - x_{mg}$ = 32.29cm
xi. $Z_D - Z_{mg}$ = 27.7cm
xii. $Z_T - Z_{mg}$ = 27.7cm

xiii. $Z_{cg} - Z_{mg} = 24.42cm$
xiv. $Iyy_{mg} = 0.1114Kgm^2$
xv. $V_R = 8.8m/s$
xvi. $S_h = 0.0245m^2$
xvii. $S = 0.1878m^2$
xviii. MAC = 16.4cm
xix. $\alpha_h = -5.12°$
xx. $CL_{\alpha h} = 4.3118rad^{-1}$
xxi. $\eta_h = 0.9$
xxii. $V_H = 0.3$
xxiii. $T_C = 1.7435N$
xxiv. $V_C = 14.6154m/s$
xxv. $Cm_\alpha = -0.83377rad^{-1}$
xxvi. $CL_\alpha = 4.828rad^{-1}$
xxvii. $CL_C = 0.4798$

STEP 1: Elevator Design Requirements

a. Takeoff rotation
b. Longitudinal trim requirement
c. Low cost
d. Manufacturability

STEP 2: Takeoff Rotation Acceleration Requirement – $12deg/s^2$

STEP 3: Elevator Span

Using the table below as guideline, the elevator span to tail span ratio (b_E/b_h) is 0.8.

STEP 4: Maximum Elevator Deflection (δ_E)

From the table above, maximum up deflection (negative) is selected to be -25deg and maximum down deflection (positive) is +20deg.

STEP 5: Wing/Fuselage Lift (L_{wf}), Aircraft Takeoff Rotation Drag (D) and Wing/Fuselage Pitching Moment (M_{ac_wf})

i. $L_{wf} = 10.327N$
ii. $D = 1.046N$
iii. $M_{ac\_wf} = -0.1125Nm$

Control surface	Elevator	Aileron	Rudder
Control surface area/lifting surface area	$S_E/S_h = 0.15–0.4$	$S_A/S = 0.03–0.12$	$S_R/S_V = 0.15–0.35$
Control surface span/lifting surface span	$b_E/b_h = 0.8–1$	$b_A/b = 0.2–0.40$	$b_R/b_V = 0.7–1$
Control surface chord/lifting surface chord	$C_E/C_h = 0.2–0.4$	$C_A/C = 0.15–0.3$	$C_R/C_V = 0.15–0.4$
Control surface maximum deflection (negative)	−25 deg (up)	25 deg (up)	−30 deg (right)
Control surface maximum deflection (positive)	+20 deg (down)	20 deg (down)	+30 deg (left)

Table 12.5: Typical values for geometry of control surfaces

STEP 6: Aircraft Linear Acceleration during Takeoff – a = 0.486m/s^2

STEP 7: Contributing Pitching Moments during Takeoff Rotation

The contributing pitching moments during takeoff rotation are: aircraft weight moment, M_W, aircraft drag moment, M_D, engine thrust moment, M_T, Wing/fuselage lift moment, M_{Lwf}, wing/fuselage aerodynamic pitching moment, M_{ac_wf}, and linear acceleration moment, M_a. Since these parameters are considered for takeoff rotation, the moments are about the main gear contact with the ground.

For these calculations, the most forward CG position is to be considered (but the CG position for this design is constant). These moments are calculated as follows.

$$M_w = W(x_{mg} - x_{cg}) \tag{12.15}$$

$$M_w = 1.2 \times 9.81 \times 0.0506 = 0.5957Nm$$

$$M_D = D(Z_D - Z_{mg}) \tag{12.16}$$

$$M_D = 1.046 \times 0.277 = 0.2897Nm$$

$$M_T = T(Z_T - Z_{mg}) \tag{12.17}$$

$$M_T = 1.7045 \times 0.277 = 0.4721Nm$$

$$M_{L_{wf}} = L_{wf}\left(x_{mg} - x_{ac_{wf}}\right) \tag{12.18}$$

$$M_{L_{wf}} = 10.327 \times 0.0424 = 0.4379Nm$$

$$M_a = m \cdot a\left(Z_{cg} - Z_{mg}\right) \tag{12.19}$$

$$M_a = 1.2 \times 0.486 \times 0.2442 = 0.1424Nm$$

STEP 8: Desired Horizontal Tail Lift during Takeoff Rotation

The desired horizontal tail lift is expressed as:

$$L_h = \frac{\begin{bmatrix} L_{wf}\left(x_{mg} - x_{ac_{wf}}\right) + M_{ac_{wf}} + m \cdot a\left(Z_{cg} - Z_{mg}\right) - W\left(x_{mg} - x_{cg}\right) + D\left(Z_D - Z_{mg}\right) \\ -T\left(Z_T - Z_{mg}\right) - I_{yy_{mg}}\ddot{\theta} \end{bmatrix}}{x_{ac_h} - x_{mg}}$$

(12.20)

This can be simplified as:

$$L_h = \frac{M_{L_{wf}} + M_{ac_{wf}} + M_a - M_W + M_D - M_T - I_{yy_{mg}}\ddot{\theta}}{x_{ac_h} - x_{mg}} \tag{12.21}$$

Since all the parameters of this equation have been gotten already, they were plugged into the equation.

$$L_h = \frac{0.4379 + (-0.1125) + 0.1424 - 0.5957 + 0.2897 - 0.4721 - \left(0.1114 \times \frac{12}{57.3}\right)}{32.29}$$

$$= -1.0332N$$

STEP 9: Desired Horizontal Tail Lift Coefficient

$$C_{L_h} = \frac{2L_h}{\rho {V_R}^2 S_h} \tag{12.22}$$

$$C_{L_h} = \frac{2 \times (-1.0332)}{1.225 \times 8.8^2 \times 0.0245} = -0.889$$

STEP 10: Elevator Angle of Attack Effectiveness

The elevator angle of attack effectiveness, τ_e was obtained from an equation used to model the horizontal tail lift coefficient.

$$C_{L_h} = C_{L_{\alpha_h}}\alpha_h + C_{L_{\alpha_h}}\tau_e\delta_E \tag{12.23}$$

This was rearranged as:

$$\tau_e = \frac{C_{L_h} - C_{L_{\alpha_h}} \alpha_h}{C_{L_{\alpha_h}} \delta_E} \tag{12.24}$$

$$\tau_e = \frac{-0.889 - \left(4.3118 \times \frac{-5.12}{57.3}\right)}{4.3118 \times \frac{-25}{57.3}} = 0.2678$$

STEP 11: Elevator-To-Tail Chord Ratio (C_E/C_h)

With the value of elevator angle of attack effectiveness obtained above (τ_e = 0.2678), the elevator chord-to-tail chord ratio is obtained to be 0.096 from the figure below

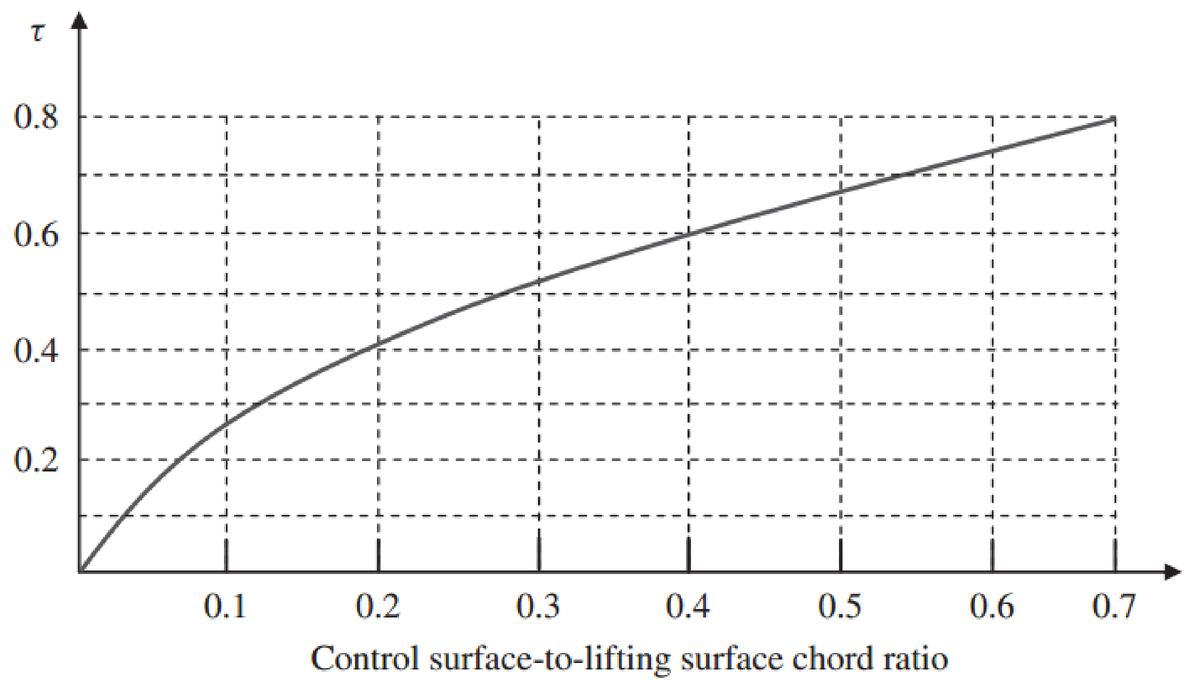


Figure 12.3: Control surface angle of attack effectiveness parameter

STEP 12.2: Elevator-To-Tail Chord Ratio (C_E/C) Check

Since the value of this ratio (C_E/C_h = 0.096) is less than 0.5, an all moving tail won't be used and the obtained value will be used.

STEP 13: Horizontal Tail Lift Distribution and Lift Coefficient from Lifting Line Theory with Maximum Negative Elevator Deflection

The MATLAB program for lifting line theory used in wing and tail design was employed for this. The contribution of the elevator deflection to the tail zero lift angle of attack is calculated as follows:

$$\Delta\alpha_{o_E} \approx -1.15\frac{C_E}{C_h}\delta_E \tag{12.25}$$

$$\Delta\alpha_{o_E} \approx -1.15 \times 0.096 \times (-25) = 2.76deg$$

The following are the parameters used for lifting line theory:

i. $\Delta\alpha_{o_E}$ = 2.76deg
ii. AR_h = 4.667
iii. λ_v = 0.7
iv. Cl_α = 6.108rad^{-1}
v. α_{twist} = 0deg
vi. zero lift angle of attack, α_o = 0deg
vii. C_E/C_h = 0.096
viii. b_e/b = 0.8
ix. i_h = - 1.39deg

With these, the horizontal tail lift coefficient is -0.5015 and the lift distribution is as shown below

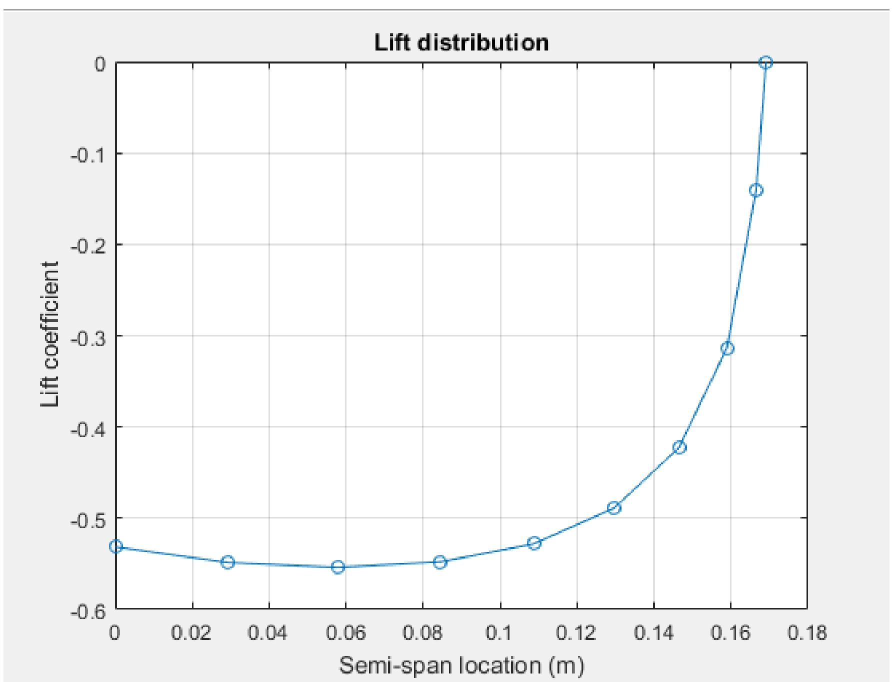


Figure 12.4: Horizontal tail lift distribution

STEP 14: Lift Coefficient Comparison

The lift coefficient produced (-0.5015) by the tail is not equal to the desired lift coefficient (-0.889). Via trial and error, at an elevator chord to wing chord ratio of 0.189, the lift coefficient is -0.889.

STEP 15: Elevator Effectiveness Derivatives

I. Rate of change of aircraft pitching moment coefficient with respect to elevator deflection.

$$C_{m_{\delta_E}} = \frac{\partial C_m}{\partial \delta_E} = -C_{L_{\alpha_h}} \eta_h V_H \frac{b_E}{b_h} \tau_e \tag{12.26}$$

With a C_E/C of 0.189, τ_e was obtained to be 0.4 (from figure 3 above). All other parameters have been obtained or stated earlier.

$$C_{m_{\delta_E}} = -4.3118 \times 0.9 \times 0.3 \times 0.8 \times 0.4 = -0.3725 rad^{-}$$

II. Rate of change of aircraft lift with elevator deflection

$$C_{L_{\delta_E}} = \frac{\partial C_L}{\partial \delta_E} = C_{L_{\alpha_h}} \eta_h \frac{S_h}{S} \frac{b_E}{b_h} \tau_e \tag{12.27}$$

$$C_{L_{\delta_E}} = 4.3118 \times 0.9 \times \frac{0.0245}{0.1878} \times 0.8 \times 0.4 = 0.162 rad^{-}$$

III. Rate of change of horizontal tail lift with elevator deflection

$$C_{L_{h_{\delta_E}}} = \frac{\partial C_{L_h}}{\partial \delta_E} = C_{L_{\alpha_h}} \tau_e \tag{12.28}$$

$$C_{L_{h_{\delta_E}}} = 4.3118 \times 0.4 = 1.7247 rad^{-}$$

STEP 16: Elevator Deflection to Maintain Longitudinal Trim at Various Flight Conditions

The equation for the elevator deflection required for longitudinal trim is modeled as:

$$\delta_E = -\frac{\left(\frac{T \cdot Z_T}{q \cdot S \cdot C} + C_{m_o}\right) C_{L_\alpha} + \left(C_{L_1} - C_{L_o}\right) C_{m_\alpha}}{C_{L_\alpha} C_{m_{\delta_E}} - C_{m_\alpha} C_{L_{\delta_E}}} \tag{12.29}$$

Z_T is the distance between the thrust line and the CG. This is negative when thrust line is above aircraft CG (based on how the equation was derived). Therefore, Z_T is -3.28cm.

$$T = \frac{\eta_p \times P}{V} \tag{12.30}$$

η_p = 0.85, P = 30W and V = 14.6154m/s

$$T = \frac{0.85 \times 30}{14.6154} = 1.7447\text{N}$$

$$q = \frac{1}{2} \rho V^2 \tag{12.31}$$

$$q = 0.5 \times 1.2232 \times 14.6154^2 = 130.64 N/m^2$$

$CL_1 = CL_C = 0.4798$, $Cm_o = -0.077$, $CL_\alpha = 4.828 rad^{-1}$ and $CL_o = 0.8$

$$\delta_E = -\frac{\left(\frac{1.7566 \times (-0.0328)}{130.64 \times 0.1878 \times 0.164} + (-0.077)\right) 4.828 + (0.4798 - 0.8)(-0.8337)}{(4.828 \cdot -0.3725) - (-0.8377 \cdot 0.162)}$$
$$= -0.0209 rad \ (-1.2 deg)$$

Therefore, there will be -1.2deg elevator deflection (up deflection) during cruising flight.

STEP 17: Elevator Deflection versus Airspeed and Flight Altitude

To plot this, the equation for elevator deflection was used with flight speeds varying from stall speed to maximum speed. The thrust and dynamic pressure equations (used in the step above) are employed and their values vary with airspeed and altitude. The lift coefficient at each case is calculated as follows

$$C_{L_1} = \frac{m \cdot g}{q \cdot S} \tag{12.32}$$

Where m is the aircraft mass, g is acceleration due to gravity, q is the dynamic pressure and S is the wing area. With this, the values of CL1 also vary with flight speed and altitude.

A MATLAB program was written to calculate and plot the graph. The result is as shown below.

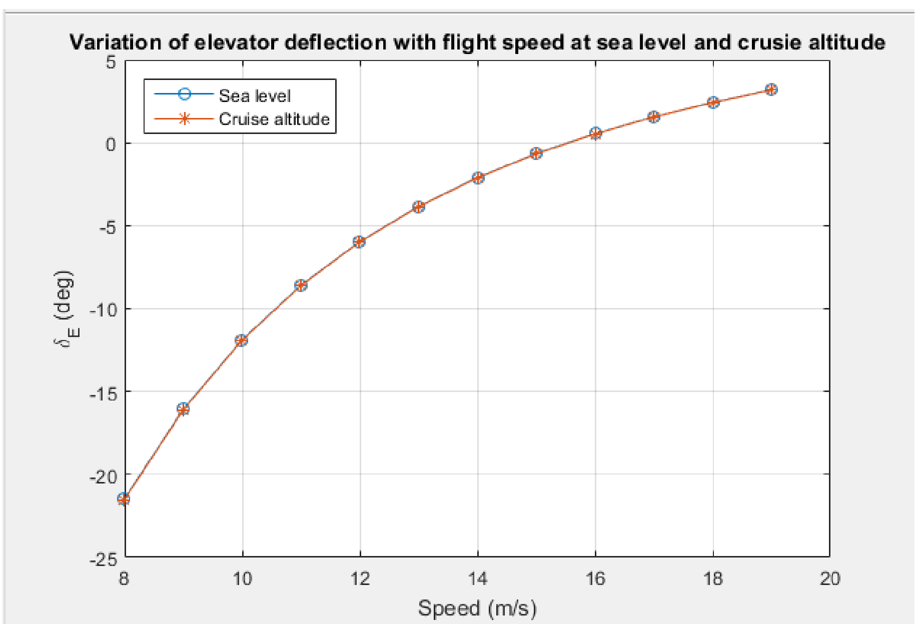


Figure 12.5: Variation of elevator deflection with flight speed at sea level and cruising altitude – trim curve

There is no obvious difference in sea level and cruise altitude since the difference in altitude is 15m

STEP 18: Maximum Deflection Check

In the graph above, the maximum deflection to maintain longitudinal trim is around -22deg (up) at stall speed and +1deg (down) at maximum speed. Since these values are lesser than the maximum deflections stated in step 4, the elevator design is OK.

STEP 19: Horizontal Tail Stall Check during Takeoff Rotation

The relationship between horizontal tail angle at takeoff and fuselage takeoff angle of attack (α_{TO}) is obtained as:

$$\alpha_{h_{TO}} = \alpha_{TO}\left(1 - \frac{d\varepsilon}{d\alpha}\right) + i_h - \varepsilon_o \tag{12.33}$$

α_{TO} = 5.27deg, $\frac{d\varepsilon}{d\alpha}$ = 0.4390rad/rad, , i_h = -1.39deg and ε_o= 0.0436rad

$$\alpha_{h_{TO}} = 5.27 \cdot (1 - 0.4390) + (-1.39) - 0.0436 = -0.93181deg$$

The tail stall angle of attack during takeoff rotation is expressed as:

$$\alpha_{h_s} = \pm\left(\alpha_{h_{s:\delta_E=0}} - \Delta\alpha_{h_E}\right) \tag{12.34}$$

$\alpha_{h_{s:\delta_E=0}}$ is the tail stall angle when elevator is not deflected and $\Delta\alpha_{h_E}$ is the reduction in tail stall angle of attack due to elevator deflection.

$\alpha_{h_{s:\delta_E=0}}$ = 10deg and $\Delta\alpha_{h_E}$ is obtained from the figure below.

δ_E (deg)	Tail-to-elevator chord ratio C_E/C_h										
	0	0.1	0.2	0.3	0.4	0.5	0.6	0.7	0.8	0.9	1
0	0	0	0	0	0	0	0	0	0	0	0
±5	0	0.3	0.5	1.1	1.6	2.2	2.7	3.3	3.9	4.4	5
±10	0	0.6	1	2.1	3.2	4.4	5.5	6.6	7.7	8.9	10
±15	0	0.9	1.5	3.2	4.9	6.5	8.2	9.9	11.6	13.3	15
±20	0	1.2	2	4.2	6.5	8.7	11	13.2	15.5	17.7	20
±25	0	1.6	2.5	5.3	8.1	11	13.7	16.5	19.4	22.2	25
±30	0	1.9	3	6.4	9.7	13.1	16.5	19.9	23.2	26.6	30

Table 12.5: Reduction in tail stall angle ($\Delta\alpha_{h_E}$, deg) when elevator is deflected

With an elevator chord to wing chord ratio of approximately 0.2 and deflection of 25deg, the reduction in stall angle of attack is 2.5deg.

$$\alpha_{h_s} = \pm(10 - 2.5) = 7.5deg \quad (12.35)$$

Since the stall angle (7.5deg) is less than the angle of attack at takeoff (-0.93181deg), the tail is safe from stall during takeoff.

STEP 20: Aerodynamic and Mass Balance

The selected aerodynamic balance is the overhanging balance – where the hinge line is moved aft (a bit) towards the elevator center of pressure. The leading edge of the control surface will be elliptical.

To insure mass balance and avoid flutter, the control surface center of gravity will be moved towards or in front of the hinge line.

STEP 21: Elevator Span, Chord and Area

$$\mathrm{b_E} = \mathrm{b_h} \times \frac{\mathrm{b_E}}{\mathrm{b_h}} \quad (12.36)$$

Horizontal tail span = 33.81cm

$$\mathrm{b_E} = 0.3381 \times 0.8 = 0.2705\mathrm{m}$$

$$\mathrm{C_E} = \mathrm{C_h} \times \frac{\mathrm{C_E}}{\mathrm{C_h}} \quad (12.37)$$

Horizontal MAC = 7.24cm

$$\mathrm{C_E} = 0.0724 \times 0.189 = 0.0137$$

$$\mathrm{S_E} = \mathrm{C_E} \times \mathrm{b_E} \quad (12.38)$$

$$\mathrm{S_E} = 0.0137 \times 0.2705 = 0.0037m^2$$

ELEVATOR DESIGN RESULT

1. δ_{Ecr} = -1.2deg
2. δ_{Emax} = -25deg and +20deg
3. Aerodynamic balance – overhanging balance with elliptical aileron leading edge.
4. Mass balance – aileron CG movement towards or in front of hinge line.
5. b_E = 27.05cm
6. C_E = 1.37cm
7. S_E = 0.0037m^2

RUDDER DESIGN

STEP 1: Most Critical Flight Condition

This design is not a spinnable aircraft and it employs a single engine. Therefore, spin recovery and asymmetric thrust are not considered as the critical flight condition. Thus, the critical flight conditions for rudder design are coordinated turn, adverse yaw and crosswind landing.

Crosswind landing is considered to be the most critical flight condition but nevertheless, adverse yaw and coordinated turn will also be looked into.

STEP 2: Most Unfavorable CG Location, Aircraft Weight and Altitude for Directional Control

The CG location is 37.73cm from vertical tail aerodynamic center. During landing, the aircraft weight will be 11.67N ((1200g-10g)*9.81m/s). And the most unfavorable altitude will be at sea level.

STEP 3: Maximum Crosswind Landing Speed (V_w) at Which the Aircraft Must Land Safely

Maximum crosswind landing speed (V_W) is selected to be 3.5 m/s.

STEP 4: Maximum Aircraft Approach Speed

The landing/approach speed, V_{land} is 10.4m/s

STEP 5: Aircraft Total Airspeed (V_T) With Crosswind

$$V_T = \sqrt{{U_1}^2 + {V_W}^2} \tag{12.39}$$

Where U_1 is the aircraft approach airspeed (10.4m/s)

$$V_T = \sqrt{10.4^2 + 3.5^2} = 10.97m/s$$

STEP 6: Aircraft Projected Side Area

From the side view of the aircraft, the engine, horizontal tail and the wing will be seen to be inside the fuselage. The aircraft projected side area is mainly the fuselage projected side area plus the vertical tail planform area plus the landing gear projected side area.

From weights of components, fuselage projected side area is 693.38cm^2, nose gear projected side area is 95.42cm^2 and main gear projected side area is 78.82cm^2. The vertical tail planform area is 0.02m^2.

$$S_S = S_{Fus} + S_{NG} + S_{MG} + S_{VT} \tag{12.40}$$

$$S_S = (693.38 \times 10^{-4}) + (95.42 \times 10^{-4}) + (78.82 \times 10^{-4}) + 0.02 = 0.097m^2$$

STEP 7: Center of the Aircraft Projected Side Area (S_S) and its Distance from CG (d_c)

The center of the aircraft project side area, Ca is expressed as:

$$x_{C_a} = \frac{\sum_{i=1}^{n} A_i x_i}{\sum_{i=1}^{n} A_i} \tag{12.41}$$

The reference line for this is selected to be the prop spinner vertex. The x-distances from the vertex are: x_{fus} = 38cm, x_{VT} = 73.95cm, x_{NG} = 7.325cm and x_{MG} = 39.82cm.

$$x_{C_a} = \frac{(0.069338 \times 0.38) + (0.009542 \times 0.07325) + (0.007882 \times 0.3982) + (0.02 \times 0.7395)}{0.069338 + 0.009542 + 0.007882 + 0.02}$$
$$= 0.4213m$$

X_{cg} is 33.32cm from the reference line. Therefore, the distance between projected side center and cg is expressed as:

$$d_c = x_{C_a} - x_{cg} \tag{12.42}$$

$$d_c = 0.4213 - 0.3332 = 0.0881m$$

STEP 8: Aircraft Side Force Produced by the Crosswind, F_W

$$F_W = \frac{1}{2}\rho {V_W}^2 S_S C_{D_y} \tag{12.43}$$

ρ is the air density at the most critical altitude – sea level (1.225Kg/m^3), V_W is the cross wing speed, S_S is the aircraft side area and C_{Dy} is the aircraft side drag which varies from 0.5 to 0.8 for conventional aircraft. C_{Dy} is selected to be 0.65.

$$F_W = \frac{1}{2} \times 1.225 \times 3.5^2 \times 0.097 \times 0.65 = 0.4731N$$

STEP 9: Rudder Span-To-Vertical Tail Span Ratio, b_R/b_V

From the table 5, b_R/b_V is selected to be 1

STEP 10: Rudder Chord-To-Vertical Tail Chord Ratio, C_R/C_V

Also from table 5, C_R/C_V was selected to be 0.3

STEP 11: Aircraft Sideslip Angle

$$\beta = tan^{-1}\left(\frac{V_W}{U_1}\right) \tag{12.44}$$

$$\beta = tan^{-1}\left(\frac{3.5}{10.4}\right) = 18.6deg$$

STEP 12: Aircraft Sideslip Derivatives C_{n_β} and C_{y_β}

From vertical tail design, C_{n_β} = 0.266rad^{-1} and C_{y_β} is expressed as:

$$C_{y_\beta} = -K_{f_2} C_{L_{\alpha_V}} \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \cdot \frac{S_v}{S} \tag{12.45}$$

Where, K_{f_2} is the contribution of the fuselage to the derivative C_{y_β}, $C_{L_{\alpha_V}}$ is the vertical tail lift curve slope, $\frac{d\sigma}{d\beta}$ is the vertical tail sidewash gradient, η_v is the vertical tail dynamic pressure ratio, S_v is the vertical tail area and S is the wing area. The typical value of K_{f2} for a conventional aircraft is about 1.3 – 1.4 and is selected to be 1.35.

From vertical tail design, the values for the following parameters were obtained

I. $C_{L_{\alpha_V}}$ = 2.66rad^{-1}
II. $\frac{d\sigma}{d\beta}$ = - 2.66
III. η_v = 0.9
IV. S_v = 0.02m^2 and
V. S = 0.1878m^2

$$C_{y_\beta} = -1.35 \times 2.66 \times \left(1 - (-2.66)\right) \times 0.9 \times \frac{0.02}{0.1878} = -1.239rad^{-1}$$

STEP 13: Rudder Angle Of Attack Effectiveness, τr

From figure 3, τr was extracted to be 0.52 (when C_R/C_V = 0.3)

STEP 14: Aircraft Control Derivatives, $C_{n_{\delta_R}}$ and $C_{y_{\delta_R}}$

$$C_{n_{\delta_R}} = -C_{L_{\alpha_V}} V_V \eta_v \tau_r \frac{b_R}{b_V} \tag{12.46}$$

$$C_{y_{\delta_R}} = C_{L_{\alpha_V}} \eta_v \tau_r \frac{b_R}{b_V} \frac{S_V}{S} \tag{12.47}$$

Where V_V is vertical tail volume coefficient and is obtained to be 0.035 from vertical tail design.

$$C_{n_{\delta_R}} = -2.66 \times 0.035 \times 0.9 \times 0.52 \times 1 = -0.0436rad^{-1}$$

$$C_{y_{\delta_R}} = 2.66 \times 0.9 \times 0.52 \times 1 \times \frac{0.02}{0.1878} = 0.1326rad^{-1}$$

STEP 15: Rudder Control Derivative

$$\frac{1}{2}\rho {V_T}^2 Sb\left(C_{n_o} + C_{n_\beta}(\beta - \sigma) + C_{n_{\delta_R}}\delta_R\right) + F_w d_c \cos\sigma = 0 \qquad (12.48)$$

$$\frac{1}{2}\rho {V_w}^2 S_S C_{D_y} - \frac{1}{2}\rho {V_T}^2 S\left(C_{y_o} + C_{y_\beta}(\beta - \sigma) + C_{y_{\delta_R}}\delta_R\right) = 0 \qquad (12.49)$$

The aircraft is assumed to be right-left symmetric. Therefore, the two unknowns, C_{n_o} and C_{y_o} are assumed to be 0. Wing span, b is 1.15m.

$$\frac{1}{2} \times 1.225 \times 10.97^2 \times 0.1878 \times 1.15 \times \left(0 + 0.266(18.6 - \sigma) + (-0.0436\delta_R)\right) + (0.4731 \times 0.0881 \times \cos\sigma = 0) \qquad (12.49 - 1)$$

$$0.4731 - \left(\frac{1}{2} \times 1.225 \times 10.97^2 \times 0.1878\right)(0 - 1.2391(18.6 - \sigma) + 0.1326\delta_R) = 0$$

(12.49 – 2)

These two equations are solved simultaneously to obtain the values of δ_R and σ to be -0.325rad (-18.62deg) and 0.387rad (22.17deg)

STEP 16: Rudder Deflection Check

Since rudder deflection (18.62deg) is less than 30deg, the rudder satisfies crosswind landing requirement.

STEP 17: Rudder Angle of Attack Effectiveness Check

Since the rudder angle of attack effectiveness parameter, τ_r (0.52) is less than 1, the rudder design is good.

STEP 18: Adverse Yaw

As stated in aileron design, to prevent adverse yaw, there will be simultaneous deflection of aileron and rudder which might require an interconnection between rudder and aileron.

STEP 19:

$$\alpha_{v_s} = \pm\left(\alpha_{v_{s:\delta_R=0}} - \Delta\alpha_{v_R}\right) \qquad (12.50)$$

$\alpha_{v_{s:\delta_R=0}}$ = 8deg and $\Delta\alpha_{v_R}$ is selected to be 5.3deg from table 6 where chord ratio is 0.3 and maximum deflection is assumed to be 25deg.

$$\alpha_{v_s} = \pm(8 - 5.3) = 2.7deg \qquad (12.51)$$

The vertical tail incidence is zero. Therefore, the tail won't stall due to rudder deflection.

STEP 20: Aerodynamic Balance/Mass Balance

The selected aerodynamic balance is the overhanging balance – where the hinge line is moved aft (a bit) towards the rudder center of pressure. The leading edge of the control surface will be elliptical.

To insure mass balance and avoid flutter, the control surface center of gravity will be moved towards or in front of the hinge line.

STEP 21: Rudder Span, Chord and Area

$$b_R = b_V \times \frac{b_R}{b_V} \tag{12.52}$$

Vertical tail span = 17.3cm

$$b_R = 17.3 \times 1 = 17.3\text{cm}$$

$$C_R = C_V \times \frac{C_R}{C_V} \tag{12.53}$$

Horizontal MAC = 11.5cm

$$C_R = 11.5 \times 0.3 = 3.45cm$$

$$S_R = C_R \times b_R \tag{12.54}$$

$$S_R = 0.0345 \times 0.173 = 0.006\text{m}^2$$

RUDDER DESIGN RESULT

1. δ_{Rmax} = ±25deg
2. Aerodynamic balance – overhanging balance with elliptical rudder leading edge.
3. Mass balance – aileron CG movement towards or in front of hinge line.
4. b_R = 17.3cm
5. C_R = 3.45cm
6. S_E = 0.006m^2

REFERENCES

1. AAE 401 – Aircraft design lecture note by Dr Yomi Obidi. Aeronautic and Astronautic Engineering Department, Kwara State University, Malete, Kwara State, Nigeria.
2. AAE 451 – Orbit and Attitude Control lecture note – mass moment of inertial by Professor Christopher Odetunde. Aeronautic and Astronautic Engineering Department, Kwara State University, Malete, Kwara State, Nigeria.
3. Mohammad H. Sadraeyauth. Aircraft Design: A Systems Engineering Approach 2012_2.
4. Raymer – Aircraft Design – A Conceptual Approach.
5. The international standard Atmosphere by Mustafa Cavar, Anadolu University, 26470 Eskisehir, Turkey
6. Design Approach for Selection of Wing Airfoil with Regard to Micro-UAVs by Vladimir Brusov and Vladimir Petruchik. Moscow Aviation Institute (MAI), Flight Dynamics and Control Department.
7. Airfoil Design For Flying Wing UAV (Unmanned Aerial Vehicle) By Prasetyo Edi, Nukman Yusoff And Aznijar Ahmad Yazid Department Of Engineering Design & Manufacture, Faculty Of Engineering, University Of Malaya, 50603 Kuala Lumpur, Malaysia.
8. Volume I Performance Phase, Chapter 11 Cruise Performance Theory. USAF Test Pilot School Edwards AFB, CA.
9. Aeronautics & Mechanics, AENG11300 - Cruise Performance, Department of Aerospace Engineering, University of Bristol.
10. Aircraft Flight Dynamics by Robert Stengel, MAE 331. 2016.
11. Lab 8 Notes – Basic Aircraft Design Rules. 6 Apr 06.
12. Roskam, J. (2007) Airplane Flight Dynamics and Automatic Flight Control Part I, DAR Corporation.
13. Rotaru Circiu Edu Ciuica Mihai. Nonlinear effects of the sidewash gradient on an airplane vertical tail.
14. Design of Quiet UAV Propellers by Alex Morgan Stoll, 2012.
15. The USAF Stability and control digital DATCOM, Volume I, Users Manual.
16. The USAF Stability and control digital DATCOM, Volume III, Plot Module.
17. G. Leng, Flight Dynamics, Stability & Control
18. A Tutorial For XFLR5 – Version 1