Design and Fabrication of Fixed-Wing UAV for Commercial Monitoring

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Abstract – *The fixed-wing UAV's present huge potentiality as far as concern both precision and efficiency, allowing not only to bring down operating costs with respect to traditional planes or helicopters, but also to intervene in environments where it couldn't be possible to operate by an aircraft with pilot on board. This report highlights the entire design cycle for the production of fixed-wing UAV using flight dynamics approach at minimal cost. The mission requirement specified is commercial monitoring using a fixed camera attached to the body. The fixed-wing designed UAV possess various advantages like longer flight capability, larger areas covered in less time, better quality of photographic result, better control of flight parameters, higher flight safety and high range and endurance.*

Introduction

In today's world, the UAVs, or Remotely Piloted Vehicles (RPV's) plays a very important role in completing various tasks like goods distribution, commercial monitoring, surveillance for military applications, object tracking, quality aerial imaging etc., that are not possible by the manned flight. The UAVs are controlled remotely which only requires a trained operator. Moreover, these minimizes the danger to the person controlling it and also provides flexibility for quick inspection in the critical regions. The most important advantage for which these flying aircrafts are used are the safety of the pilot who does not accompany the vehicle and the low cost of the production.

The purpose of this project is to design and fabricate the fixed-wing UAV for the commercial monitoring purpose. The fixed-wing UAV is the type of UAV, wherein the wings are the main lifting component. A critical aerodynamic study is required for the selection of the wings which can provide an efficient flight. The fixed wing designs have various advantages over the quadcopters or drones. They provide high thrust, more range and endurance, high aerodynamic efficiency and stability, flight safety, etc. This design of fixed wing UAV is based on flight dynamics approach. This approach mainly focuses on the study and the calculations on the longitudinal stability and control, C.G determination, weight estimation and the performance estimation. The entire design process is divided into three segments: conceptual design, preliminary design and detail design. The conceptual design discusses about the initial design of the fixed-wing UAV, body sizing, and wing sizing. In the preliminary design, all the analytical and the computational work is performed. The detail design consists of the generation of the CAD model, drafting sheet and the structural analyses. The 3-D printing technology is also incorporated in order to produce a 3-D prototype of the model. The printed model works for highlighting the critical components of the actual fabricated part. Finally, the fixed-wing UAV is manufactured after the proper selection of the materials, and components. The flight testing will verify the scope of the project i.e., commercial monitoring (surveillance) and also verifies the calculations performed in the preliminary design stage.

Hence, the report provides the detailed description on the design and fabrication of the fixed-wing UAV for commercial monitoring and proves its advantages over the rotary wing (drones). The report also highlights the use of the flight dynamics approach to ensure that the fixed-wing UAV is stable during the flight and verification is done by flight testing of the model.

1. Unmanned-Aerial Vehicle

The wings are the main lifting component in the fixed wing design. A critical aerodynamics study is done to study the various airfoil shapes which can give appropriate lift and thrust as per the design and mission requirements. The design changes in the airfoil design can result in better L/D. Hence, the UAV can generate more lift with a minimal drag. The fixed wing design require less power and thrust loading when compared with the multi copter.

The fixed-wing UAVs are very advanced machines which are exclusively significant in terms of aerodynamics performance, maneuvering, structure load carrying capabilities and performing stealth, military and commercial operations. Various flight tests and experimental analysis have shown that the UAV's offer remarkable performance with higher range and endurance when compared to the manned systems.

2. Design Cycle for Fixed-Wing UAV

The following is the design methodology usually adopted when designing any sort of aerial machine.

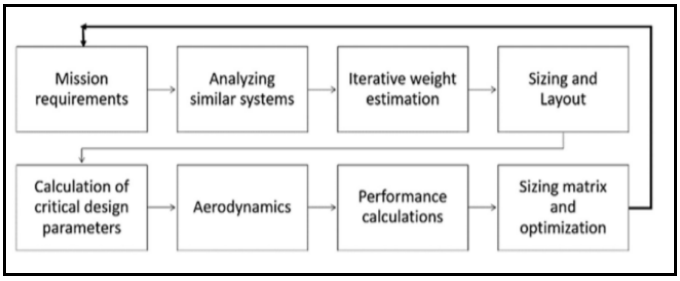


Fig 1 : Design Methodology

The aircraft design process is also similar to the general design process. However, there are some additions to the design guidelines because the aircraft design has to pass through various tests and quality certification so that the final product meets the project requirement. The complexity of an aircraft and the tight safety regulations required make it a very expensive and long process.

In order to obtain an accurate product or a part, it is essential to thoroughly understand each and every design stage and frame the design idea by highlighting the key design requirements and mission specifications.

Problem Definition: At the initial design stage, the market survey is done, and the requirements are listed down. Here, the main focus is done on the available concepts and products and study is done on the design improvement to meet the mission requirements. Discussion is done on the key requirements of the project mission and to determine the workflow pattern.

Conceptual Design: It is the first stage of the aircraft design. Here, the rough sketches and the design geometries are sketch on a paper. Different layout is worked on so that the best design can be chosen for further calculation. The designers work to obtain a suitable aircraft configuration that meets the market demand and also fulfil the mission requirement. Here, consideration is laid on the aerodynamics, stability, structural configuration, performance, weight, and the control systems. However, the initial design begins with the selection of wings, fuselage design, and the aircraft's engine size.

Preliminary Design: The second stage of the design process is the preliminary design. This design process involves the calculation parts which proves that the selected design meets the requirements. The calculations are performed on aerodynamics, flight mechanics, flight dynamics, structural analysis. Further, the use of the software's like CATIA for CAD design, ANSYS for structural analysis, MATLAB for weight estimation is also used to verify the analytical calculations. The calculations and the iterations are performed so that the concept can be transformed to reality and at a reasonable cost.

Detail Design: The detail design is the final design stage where the concept is designed using a CAD software. Since the calculations in the preliminary design stage is fixed, so the next stage is to fix the final design. The drafting is done based on the 3rd angle projection and the 1st angle projection. This is a highly complex and critical stage of the process. After the CAD design and the drafting, the component is prepared for the fabrication of the parts. The fabrication can be done in two ways:

• Directly using the drafting sheets for fabrication and assembly of parts

• Using the prototype model in order to study the critical design areas and then proceed with fabrication accordingly.

The critical components that are designed are airfoil, wing, fuselage, empennage, propulsion system, landing gear, control surfaces, etc.

Flight Testing: In this stage, the fabricated component is tested whether it is meeting the mission requirements or not. The flight testing is a lengthy and expensive process because it must be ensured that the fabricated structure satisfies all the tests and produces good results.

Critical Design Review: In this, stage, certain changes can be done in the fabricated model to ensure that everything works properly. For e.g., setting location of the centre of gravity in the fabricated component. After this stage, the design is competed and can be prepared so that it can perform its mission requirements.

3. Design Methodology: Conceptual Design

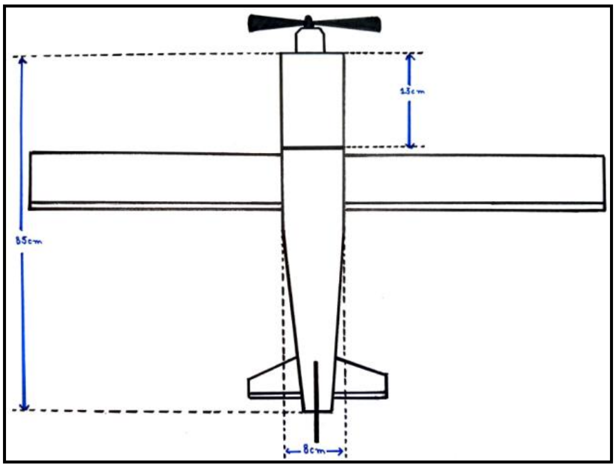


Fig 2 : Top View

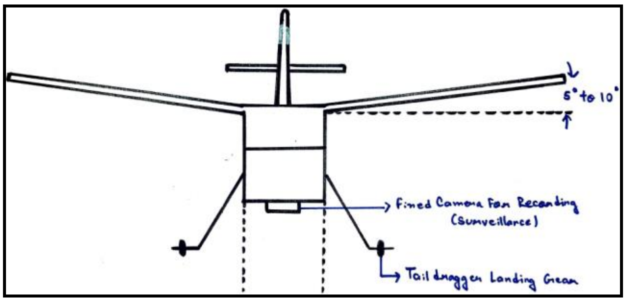


Fig 3 : Front View

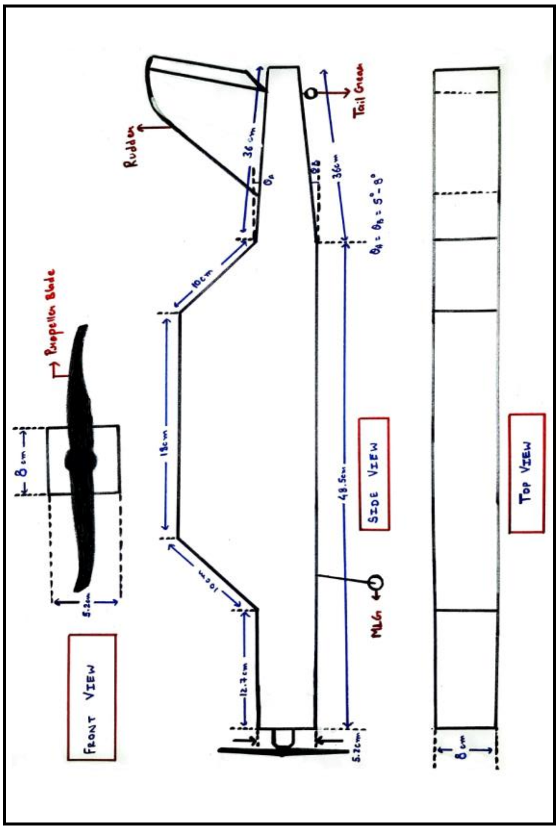


Fig 4 : Side View

3.1 Wing Selection and Geometry Consideration

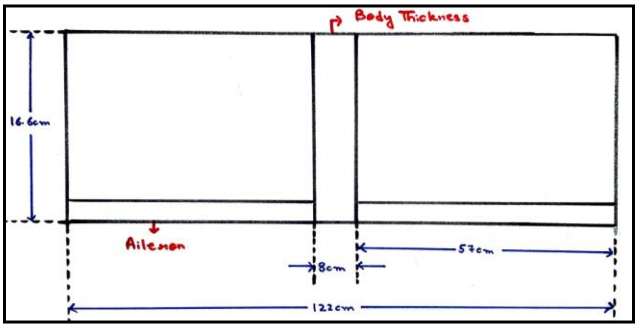


Fig 5 : Rectangular Wing Planform-Conceptual Design

The wing selected for the design calculations and the fabrication is the rectangular wing. The rectangular wing is specially selected since the mission requirement is not for high altitude flying and the rectangular section wing will be easy to fabricate. Rectangular wings better tolerate minor lapses in quality control/airfoil smoothness. The rectangular wings are the root chord stallers.

If a situation occurs where stalling takes place, then the root chord will first stall followed by effects on the control surfaces like ailerons and flaps. This root chord stall will help the aircraft to regain stability again.

Airfoil Selection: For the fixed wing UAV, the **AG38** is selected as the reference airfoil having a flat pressure surface. The dimensions and the wing sizing ae specified below.

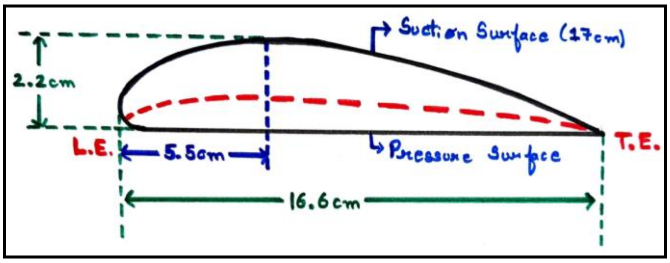


Fig 6 : Airfoil Configuration

Flight Dynamics Design Calculations (Preliminary Design): -

For a Rectangular wing with Flat Bottom Airfoil, the design considerations are: -

Wing Span (b): 1.22 m (120 cm to 130 cm)

Chord Length (c): 0.166 m (14 cm to 18 cm)

Width of the Wing (w): 0.166 m (Same as chord length due to flat bottom)

Width of the Fuselage (wf): 0.08 m (7 cm to 9 cm)

Maximum Thickness of Airfoil: 0.022 m (2 cm to 3 cm)

Maximum Thickness Distance from LE: 0.055 m (5 cm to 6 cm)

Suction Surface Camber Length: 0.17 m (16 cm to 18 cm)

Maximum Weight of the Fixed Wing UAV = 0.7 kg (600 g to 700 g)

Area of Rectangular Wing (s) = span x width = 1.22 x 0.166 = 0.20252 m^2

Maximum Flight Velocity (V_∞) = 5.55 m/s to 10.55 m/s (19.98 ft/s to 37.98 ft/s)

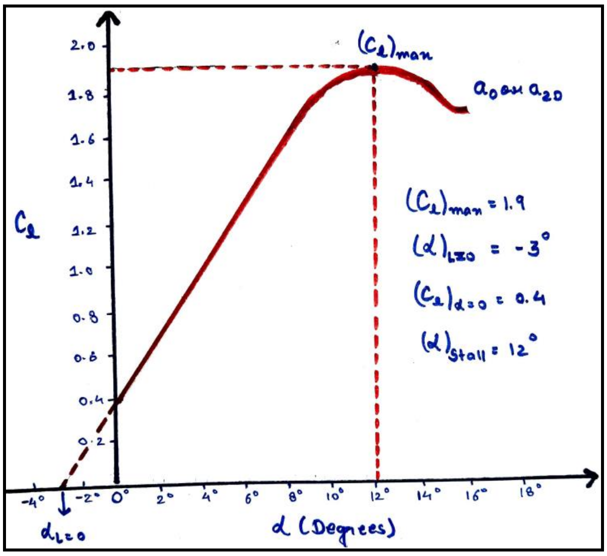


Fig 7 : 'Cl' vs 'α' Curve for Selected Airfoil Configuration

e (Oswald's Efficiency) = 0.8

$C_{d,o}$ = 0.03 (for complete aircraft)

The efficiencies mentioned below are determined based on the market survey and by the study of configuration of the electronic equipment's available.

Efficiencies: i. Electrical Efficiency ($\eta_{electrical}$) = 0.98

ii. Motor Efficiency (η_{motor}) = 0.9

iii. Propeller Efficiency ($\eta_{propeller}$) = 0.95

Thickness/chord (t/c) ratio: 0.022/0.166 = 0.1325

The **Aspect Ratio (A.R)** is defined as the ratio of square of span and the area of the wing.

$$\mathbf{A.R} = \frac{b^2}{s} = \frac{1.22*1.22}{1.22*0.166}$$

A.R = 7.3493

The **Taper Ratio (λ)** is defined as the ratio of Tip Chord (C_t) to Root Chord (C_r).

$$\lambda = \frac{Ct}{Cr} = \frac{0.166}{0.166}$$

$$\boldsymbol{\lambda} = \mathbf{1}$$

Since the wing planform is symmetrical so the tip chord and the root chord have the same value and the corresponding taper ratio is 1.

The **Mean Aerodynamic Chord (MAC)** is also defined as the average length of the chord.

$$\mathbf{MAC} = \frac{2}{3}\ .\ Cr\frac{\lambda^2+\lambda+1}{\lambda+1}$$

$$\mathbf{MAC} = \frac{2}{3}\ .\ 0.166\ (\frac{3}{2})$$

$$\mathbf{MAC} = 0.166$$

The MAC = 0.166 depicts that the rectangular planform is unform from the tip chord to the root chord of the airfoil.

The **Sweep Back Angle** is zero due to rectangular wing planform.

NOTE:

In the mission requirements, the **Maximum Cruise Speed** of the UAV selected is : 5.55 m/s.

Maximum Time of Flight : 10 min to 15 mins

Maximum Altitude : 20 m (65.6167 ft)

The **Design Lift Coefficient** is given by:-

$$\mathbf{C_{l,max}} = \frac{(2.(W))}{\rho.\ V^2.s} = \frac{2\ x\ 0.7\ x\ 9.8}{1.225\ x\ 5.55^2\ x\ 0.20252}\ \text{(From Maximum Design Weight)}$$

$$\mathbf{C_{l,max}} = 1.7954$$

From the **2D Lift Curve Slope**, the value of a_{2D} is calculated as: -

$$\mathbf{a_{2D}} = \frac{0.4-0}{(0-(-3))}$$

$$\mathbf{a_{2D}} = 0.1333 \text{ per degree} = 7.6375 \text{ per radian}$$

The **3D Lift Curve Slope (a_{3D})** is given by: -

$$a_{3D} = \frac{a2d}{1+(\frac{a2d}{\pi.\, e.AR})}$$

On Substituting the values, 3D lift curve slope value is determined to be a_{3D} = 5.4036 per radian = 0.09431 per degree.

Now, **k** = 1/(π.e.AR) = 1/ (3.14 x 0.8 x 7.3493)

k = 0.05413

The **Induced Drag** is given by,

$$C_{d,i} = C_l^2 / (\pi.\, e.\, AR) = 1.7954^2 \times 0.05413$$

$$C_{d,i} = 0.17451$$

The **Coefficient of Drag (Cd)** is given by,

$$C_d = C_{d,o} + k\,.\, C_l^2 = 0.03 + (0.05413 \times 1.7954^2)$$

$$C_d = 0.20451$$

Since the airfoil used for the design of the **fixed-wing UAV** resembles the symmetrical airfoil, hence **the thin airfoil theory** can be applied, and the values of symmetrical airfoil can be considered for the UAV design.

The **Mean Aerodynamic Centre (X̄ac)** of the airfoil is located at the quarter chord point from the thin airfoil theory.

$$\bar{X}_{ac} = c/4 = .166/4$$

$$\bar{X}_{ac} = 0.0415 \text{ m}$$

Since, for a thin airfoil, the **moment coefficient about the quarter chord point** i.e. $C_{M,1/4.c}$ is '0' and at that point, angle of attack has no effect on the airfoil.

So, the point at which the pitching moment is zero at thin airfoil, is called the **quarter chord point**.

Aerodynamic Center is defined as the point on the chord line where the pitching moment remains constant with the change in the angle of attack. **Hence, it can be concluded that the quarter chord point of the airfoil selected for UAV design is also the aerodynamic center.**

Hence, from the conceptual design, the following parameters have been determined: Aspect Ratio, Taper Ratio, Mean Aerodynamic Chord, Wingspan, Design Lift Coefficient, 2-D Lift Curve Slope, 3D Lift Curve Slope, Induced Drag, Coefficient of Drag from selected flight velocity and weight.

3.2 Tail Selection and Geometry Consideration

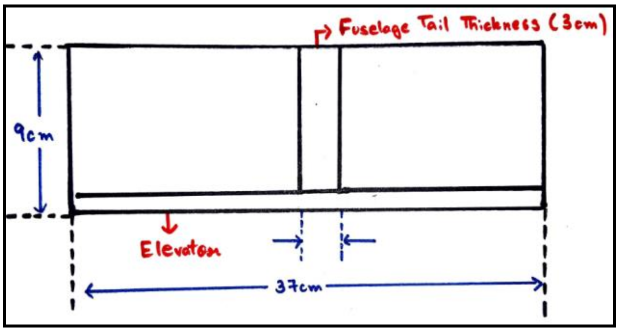


Fig 8 : Rectangular Wing Planform-Conceptual Design

Airfoil Selection: Generally, the airfoil selected for the elevator design is the symmetrical airfoil. For the design of the fixed-wing UAV, **NACA 0006 symmetrical airfoil** is selected. This airfoil also lies in the thin airfoils category, so the **Thin Airfoil Theory** is applicable for this airfoil.

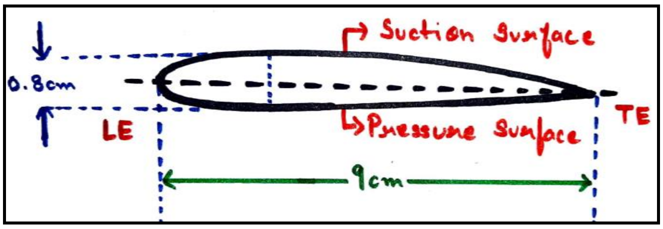


Fig 9 : Airfoil Configuration

Flight Dynamics Design Calculations (Preliminary Design): -

For a Rectangular wing with Symmetrical Airfoil, the design considerations are: -

Wing Span (b): 0.37 m (35 cm to 40 cm)

Chord Length (c): 0.09 m (5 cm to 10 cm)

Width of the Wing (w): 0.09 m (Same as chord length due to flat bottom)

Width of the Fuselage at tail end (wf): 0.03 m (2 cm to 5 cm)

Maximum Thickness of Airfoil: 0.008 m (0.5 cm to 1 cm)

Area of Rectangular Wing (s) = span x width = 0.37 x 0.09 = 0.0333 m^2

Maximum Flight Velocity (V_∞) = 5.55 m/s to 10.55 m/s (19.98 ft/s to 37.98 ft/s)

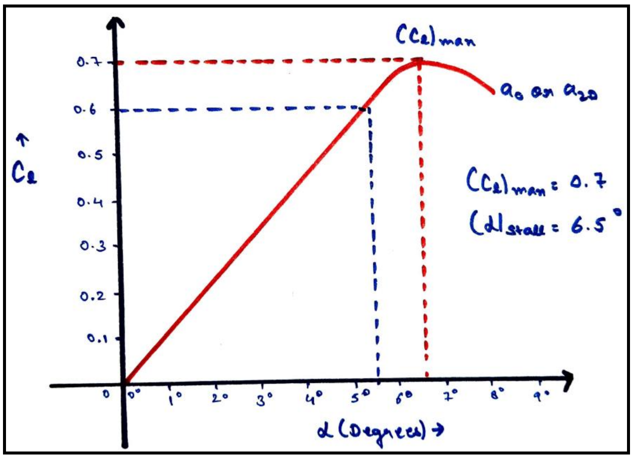


Fig 10 : Lift Curve slope for NACA 0006 Airfoil

e (Oswald's Efficiency) = 0.8

$C_{d,o}$ **=** 0.03 (for complete aircraft)

The thickness/chord (t/c) ratio is given by : 0.008/0.09 = 0.0888

The **Aspect Ratio (A.R)** is defined as the ratio of square of span and the area of the wing.

$$\mathbf{A.R} = \frac{b^2}{s} = \frac{0.37*0.37}{0.37*0.09}$$

$$\mathbf{A.R} = 4.1111$$

The **Taper Ratio (λ)** is defined as the ratio of Tip Chord (C_t) to Root Chord (C_r).

$$\lambda = \frac{Ct}{Cr} = \frac{0.09}{0.09}$$

$$\lambda = \mathbf{1}$$

Since the wing planform is symmetrical so the tip chord and the root chord have the same value and the corresponding taper ratio is 1.

The **Mean Aerodynamic Chord (MAC)** is also defined as the average length of the chord.

$$\mathbf{MAC} = \frac{2}{3} \, . \, Cr \frac{\lambda^2+\lambda+1}{\lambda+1}$$

$$\mathbf{MAC} = \frac{2}{3} \, . \, 0.09 \, (\frac{3}{2})$$

$$\mathbf{MAC} = 0.09$$

From the **2D Lift Curve Slope**, the value of a_{2D} is calculated as: -

$$\mathbf{a_{2D}} = \frac{0.6-0}{(5.5-0)}$$

$$\mathbf{a_{2D}} = 0.10909 \text{ per degree} = 6.2503 \text{ per radian}$$

The **3D Lift Curve Slope (a_{3D})** is given by: -

$$\mathbf{a_{3D}} = \frac{a2d}{1+(\frac{a2d}{\pi.\, e.AR})}$$

On Substituting the values, 3D lift curve slope value is determined to be $\mathbf{a_{3D}}$ **=** 3.8948 per radian = 0.06797 per degree.

The **Mean Aerodynamic Centre ($\bar{X}$ac)** of the airfoil is located at the quarter chord point from the thin airfoil theory.

$$\mathbf{\bar{X}_{ac} = c/4 = 0.09/4}$$

$$\bar{X}_{ac} = 0.0225 \text{ m (Quarter Chord Point)}$$

Since, for a thin airfoil, the **moment coefficient about the quarter chord point** i.e. $\mathbf{C_{M,1/4.c}}$ is '0' and at that point, angle of attack has no effect on the airfoil.

4. Longitudinal Stability and Control

The longitudinal static stability is basically the **stability during the pitching motion of the aircraft**. Pitching motion is the nose up and down condition which is the **motion about the y-axis or the pitch axis or the lateral axis. Elevator** is the control surface which is used to maintain the longitudinal stability of the aircraft.

Trim (Equilibrium) Condition: The trim condition for the longitudinal static stability is: $\Sigma M_{C.G} = 0$ i.e., pitching moment about the C.G location must be zero.

In terms of the pitching moment coefficient, $\mathbf{C_{m,C,G}} = 0$

The moment expression is given by **M** = ½ . ρ . V² . s. c . C_m

Condition for Longitudinal Static Stability: -

1. The pitching moment coefficient curve slope i.e., $\mathbf{C_{m,\alpha}}$ **must negative (less than zero).**

2. The value of the pitching moment coefficient at zero angle of attack must be positive i.e., $\mathbf{C_{m,0} > 0}$.

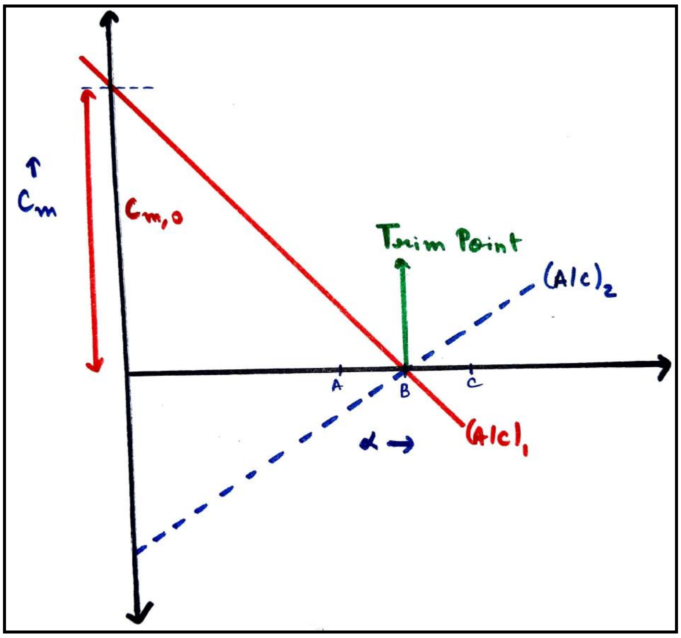


Fig 11 : General 'C_m' vs 'α' Plot

In the above plot, the aircraft 1 is longitudinally stable while the aircraft 2 is longitudinally unstable. The aircraft 1 satisfies both stability criteria but for the aircraft 2, $C_{m,0}$ is negative hence it is unstable and will require more stick force application and thereby more efforts of the pilot.

In the above plot**, increase in 'α' should lead to the nose down moment for stability due to the negative restoring moment of the aircraft (Point C).**

With the low angle of attack, the restoring moment is positive, and the nose will go up for maintaining the longitudinal stability (Point A).

4.1 Contribution of Wing towards Longitudinal Static Stability

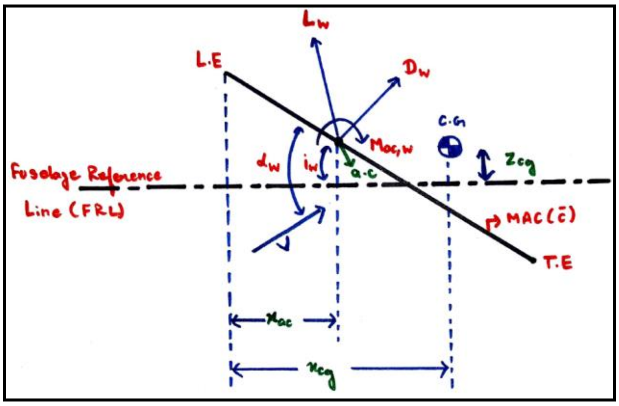


Fig 12 : Wing Contribution

$\mathbf{i_w}$ = Wing Setting Angle

$\boldsymbol{\alpha_w}$ = Angle of attack of Wing

$\mathbf{x_{c.g}}$ = Distance between 'C.G' and 'L.E'

a.c = Aerodynamic Center

$\mathbf{z_{c.g}}$ = Distance between 'C.G' and 'F.R.L'

$\mathbf{x_{a.c}}$ = Distance between 'L.E' and 'Aerodynamic Center'

Assumption: 1. Angle of attack is very small.

2. $C_L >> C_D$

3. Vertical Shift of 'C.G' i.e., $z_{c.g}$

$$C_{mcg,w} = C_{mac,w} + C_{L,w}\left(\frac{x_{cg} - x_{ac}}{\bar{c}}\right)$$

$$\therefore C_{mcg} = C_{m0,w} + C_{m\alpha,w} \cdot \alpha_w$$

Now, $C_{L,w} = C_{L\alpha,w} \cdot \alpha_w$

$$\boxed{C_{m,w} = C_{L\alpha,w}\left(\frac{x_{cg} - x_{ac}}{\bar{c}}\right)}$$

Hence, $$\boxed{C_{m\alpha,w} = \underbrace{C_{L\alpha,w}}_{1}\underbrace{\left(\frac{x_{cg} - x_{ac}}{\bar{c}}\right)}_{2}}$$

The first term $\mathbf{C_{L\,\alpha,w}}$ is always positive, and the second term is also positive because $\mathbf{x_{c.g} > x_{ac}}$.

Hence, The $\mathbf{C_{m\alpha,w}}$ **is also positive**. It can be concluded that the wing contributes to the aircraft by **destabilizing it.**

The **Fuselage Contribution** is not calculated because its contribution to the longitudinal static stability is very less or insignificant as the lift force is negligible in the fuselage.

4.2 Contribution of Tail towards Longitudinal Static Stability

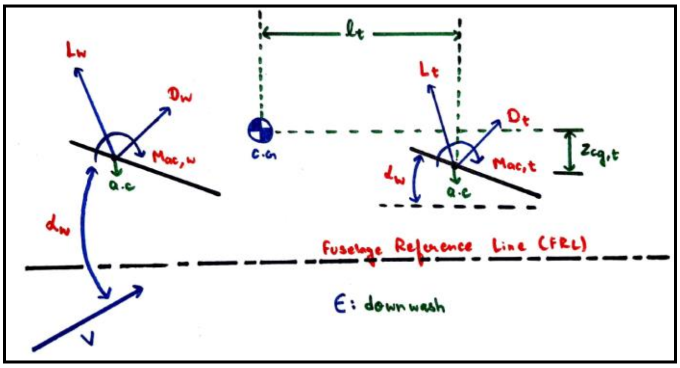


Fig 13 : Tail Contribution

$\mathbf{i_w}$ = Wing Setting Angle

$\mathbf{i_t}$ = Tail Setting Angle

$\boldsymbol{\alpha_w}$ = Angle of attack of Wing

$\mathbf{x_{c.g}}$ = Distance between 'C.G' and 'L.E'

a.c = Aerodynamic Center

$\mathbf{z_{c.g}}$ = Distance between 'C.G' and 'F.R.L'

$\mathbf{x_{a.c}}$ = Distance between 'L.E' and 'Aerodynamic Center'

$\mathbf{C_{L,t}}$ = Tail Lift Curve Slope

$\mathbf{i_w}$ = 3.2º

$\mathbf{i_t}$ = 2.5º

η is the tail efficiency and generally lies between **0.8 to 1.2.** It depends upon the location from C.G. It is preferred to establish the tail farther from the wing.

V_H is the Horizontal Tail Volume Ratio = $l_t \cdot s_t / (s \cdot c)_w$

$$C_{m,t} = -V_H \cdot \eta \cdot C_{L,\alpha,t}$$

Pitching Moment Coefficient of the tail i.e., $C_{m,t}$ is negative, so the tail gives the stabilizing effect to the complete aircraft.

$$C_{m,c.g,t} = C_{m,0,t} + C_{m,\alpha} \cdot \alpha$$

$$C_{m,0,t} = \eta \cdot V_H \cdot C_{L,\alpha,t} \cdot (i_w - i_t)$$

$C_{m,0,t}$ can be made positive by increasing tail efficiency(η) and the horizontal tail volume ratio(V_H).

$$C_{m,\alpha,t} = -V_H \cdot \eta \cdot C_{L,\alpha,t} (1 - d\varepsilon/d\alpha)$$

Hence, the tail portion nullify the effect of the wing destability and provides longitudinal static stability by providing more negative **'$C_{m,\alpha,t}$'**. So, the resultant is the **'stability of the aircraft'**.

The complete wing-body-tail effect on the longitudinal static stability for the fixed-wing UAV is studied and calculated in the further section of the project.

4.3 Longitudinal Stability for Complete Wing-Body-Tail Configuration

Wing Setting Angle, i_w = 3.2º
Tail Setting Angle, i_t = 2.5º
η is the tail efficiency and generally lies between **0.8 to 1.2**. It depends upon the location from C.G. It is preferred to establish the tail farther from the wing.

η = 0.9 (For UAV-from historical data)

V_H is the Horizontal Tail Volume Ratio = $l_t \cdot s_t / (s \cdot c_{mac})_w$
l_t = 0.52 m
s_t = b x c = 0.0333 m²
s = 0.20252 m²
c = 0.166 m (Mean Aerodynamic Centre)

V_H = 0.51507

'ε' is the downwash angle.

$$d\varepsilon / d\alpha = 2. C_{L\alpha,w} / \pi .e. AR_w$$

a_{3D} (wing) = 0.09431 per degree
e = 0.8 (historical data)

A.R of Wing = 7.3493

$$d\varepsilon / d\alpha = 0.0081694$$

a_{3D} (tail) = 0.06797 per degree
a_{3D} (wing) can also be described as **a_{wb}** (Wing Body Configuration Lift Curve Slope)

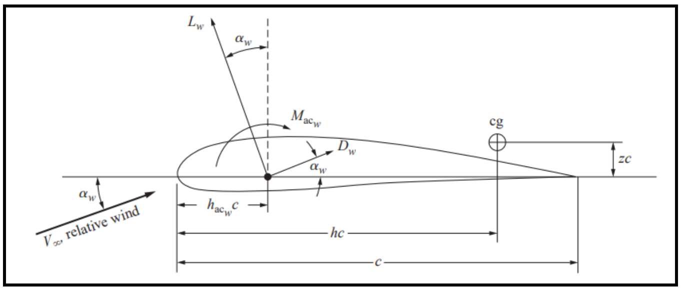


Fig 14 : Airfoil Nomenclature and Geometry

$$C_{M,cg,wb} = C_{M,ac,wb} + a_{wb} \cdot \alpha \cdot (h - h_{ac,wb})$$

$C_{M, cg, wb}$ = moment coefficient about the centre of gravity

$C_{M, ac, wb}$ = moment coefficient about the aerodynamic center

h (center of gravity in fractions of chord length) = 0.35 (From Historical Data) (Non-Dimensionalized)

$h_{ac,wb}$ = Aerodynamic Centre in fraction (Non-Dimensionalized)
α = Angle of Attack

For the similar configuration aircrafts, from the historical data, it is observed that at **1° angle of attack, $C_{M, cg, wb}$ = -0.01** and **at 8° angle of attack, $C_{M, cg, wb}$ = 0.05**
At, **α =** 1°, **C_l =** 0.5292 (From the Lift Curve Slope)
At, **α =** 8°, **C_l =** 1.4553 (From the Lift Curve Slope)

Solving for 1o angle of attack, we get

$$\mathbf{-0.01} = C_{M, ac, wb} + 0.37724. (h - h_{ac, wb})$$

Now, solving for 8o angle of attack, we get

$$\mathbf{0.05} = C_{M, ac, wb} + 1.03471. (h - h_{ac, wb})$$

The above two equations have two unknowns '$C_{M, ac, wb}$' and '$(h - h_{ac, wb})$', They can be solved simultaneously. Subtracting the second equation from the first, we get

$$-0.06 = -0.66017 (h - h_{ac, wb})$$

$$\mathbf{h - h_{ac, wb}} = 0.09088$$

Since **h** = 0.35 (Non-Dimensionalized)

$$\mathbf{0.35 - h_{ac, wb}} = 0.06451$$

$\mathbf{h_{ac, wb}}$ = 0.25912 = **Aerodynamic Centre in fraction (Non-Dimensionalized)**

Now, back substituting in the equation: -0.01 = CM, ac, wb + 0.37724. $(h - h_{ac, wb})$,

$$\mathbf{-0.01} = C_{M, ac, wb} + 0.52. (0.35 - 0.25912),$$

Hence, **$C_{M, ac, wb}$** (moment coefficient about the aerodynamic center) = -0.04428.

Total Pitching Moment About the Center of Gravity: -

Since **$M_{cg, t}$** denotes the contribution to moments about the airplane's center of gravity due to the horizontal tail.

$$C_{M,\mathrm{cg}_t} = -a_t V_H \alpha_{\mathrm{wb}} \left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right) + a_t V_H (\varepsilon_0 + i_t)$$

$C_{M, cg, t}$ = Pitching moment coefficient about the airplane's center of gravity due to the horizontal tail.

$$C_{M,\mathrm{cg}} = C_{M,\mathrm{ac_{wb}}} + C_{L_{\mathrm{wb}}}(h - h_{\mathrm{ac_{wb}}}) - V_H C_{L,t}$$

$C_{M, C.G}$ is the total moment coefficient about the center of gravity for the complete airplane.

$$C_{M,\mathrm{cg}} = C_{M,\mathrm{ac_{wb}}} + a_{\mathrm{wb}}\alpha_{\mathrm{wb}}\left[h - h_{\mathrm{ac_{wb}}} - V_H \frac{a_t}{a_{\mathrm{wb}}}\left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)\right] + V_H a_t (i_t + \varepsilon_0)$$

Equations for Longitudinal Static Stability (Wing+Body+Tail): -

$$dC_{M,c,g} / d\alpha_a = a\ [\ (h - h_{ac,wb}) - \{V_H \cdot \eta \cdot (a_t/a) \cdot (1 - (d\varepsilon / d\alpha))\}\ ]$$

a = 0.09431 per degree (3D Curve Slope for Wing)

$h - h_{ac, wb}$ = 0.09088

η = 0.9 (For UAV-from historical data)

V_H = 0.51507

a_{3D} (tail) = 0.06797 per degree = **a_t**
dε / dα = 0.0081694

On substituting the values, we obtain,

$C_{M,\alpha}$ (for complete aircraft) = $dC_{M, c.g} / d\alpha_a$
$C_{M,\alpha}$ = 0.09431 [(0.09088)–{0.51507.0.9. (0.06797/0.09431). (1-0.0081694)}]

On solving the equation, we obtain,

$$\mathbf{C_{M,\alpha}} = dC_{M, c.g} / d\alpha_a = -0.02267$$

Hence, the slope of the moment coefficient curve for the entire aircraft is negative, which denotes the aircraft is longitudinally stable. The major stability is given by the tail region and due to which the pitching moment coefficient curve slope is found out to be negative. This has satisfied the first criteria of the longitudinal static stability.

However, is the model longitudinally balanced?

To answer this, it is required to determine **$C_{M,0}$**, which in combination with the preceding result for **$dC_{M, c.g} / d_{\alpha a}$** will yield the equilibrium angle of attack **α_{trim}** or **α_e**.

$$C_{M,0} = C_{M,ac,wb} + V_H . \eta . a_t . (\varepsilon_o + i_t)$$

ε_o = 0 (From Data Source)

$C_{M,\ ac,\ wb}$ (moment coefficient about the aerodynamic center) = -0.04428

Tail Setting Angle, i_t = 2.5°
η = 0.9 (For UAV-from historical data)
V_H = 0.51507
a_{3D} (tail) = 0.06797 per degree = **a_t**

After substituting the values,
$C_{M,0}$ = -0.04428 + (0.51507. 0.9. 0.06797). (0 + 2.5°)

$C_{M,0}$ = 0.0344909

Hence, the value of $C_{M,0}$ is greater than '0' and falls in the permissible range. Therefore, the above calculation prove that the UAV is longitudinally stable.
This has satisfied the second criteria of the longitudinal static stability.

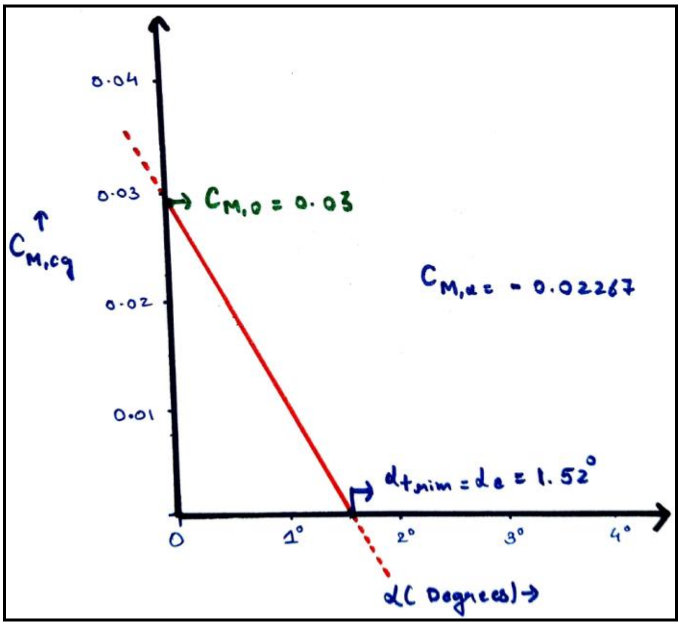


Fig 15 : Pitching Moment Curve Slope for the Entire UAV

Now, to determine trim angle of attack, **α_{trim} or α_e, $C_{M,\ c.g}$ = 0** in the Pitching moment coefficient curve slope.

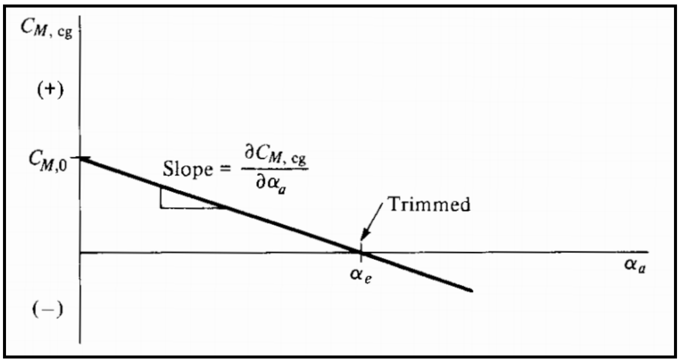


Fig 16 : Pitching Moment Coefficient Curve Slop for Entire Aircraft

For calculating the trim angle of attack, the following equation will be used: -

$$C_{M,\ c.g} = C_{M,0} + C_{M,\alpha} . \alpha$$

If, $\alpha = \alpha_{trim}$, then **$C_{M,\ c.g}$ = 0**
Hence, we obtain,

$$0 = 0.03449 + (-0.02267).\ \alpha_{trim}$$

$$\alpha_{trim} = 1.52130$$

The equation,

$$dC_{M,c,g} / d\alpha_a = a\ [\ (h - h_{ac,wb}) - \{V_H . \eta . (a_t/a) . (1- (d\varepsilon / d\alpha))\}\ ]$$

indicates that static stability is a strong function of **h**.

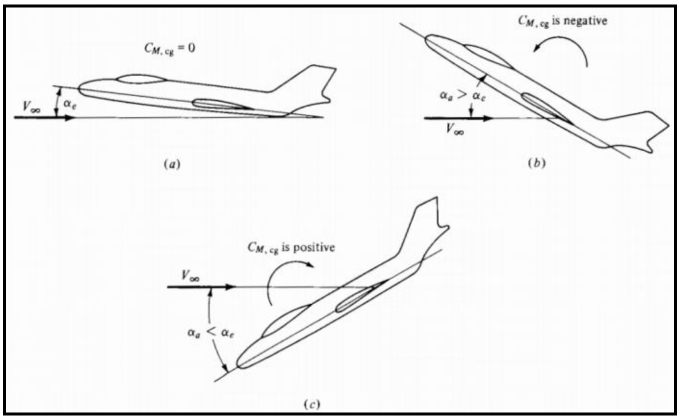


Fig 17 : Illustration of Static Stability (a) Equilibrium position (trimmed), (b) Pitched upward by disturbance, (c) Pitched downward by disturbance. In both b and c, the airplane has the initial tendency to return to its equilibrium position.

There are two possibilities: If the aircraft is pitched upwards i.e., $\alpha > \alpha_e$, the moment about the centre of gravity will be negative. The restoring moment is negative due to the change in angle of attack and this moment generated will lead to the nose down moment for the longitudinal stability. Hence, the airplane will initially tend to move back toward its equilibrium position after being disturbed. On the other hand, if the plane is pitched downward by the gust i.e., $\alpha < \alpha_e$, the resulting moment about the center of gravity will be positive (clockwise) and will tend to pitch the nose upward. Thus, again we have the situation where the airplane will initially tend to move back toward its equilibrium position after being disturbed due the positive restoring moment. This precisely gives the definition of static stability.

Neutral Point (h_n): The C.G location at which $C_{M,\alpha} = 0$ **i.e., the slope of pitching moment coefficient vs alpha is zero** then the aircraft is said to be neutrally stable and at $C_{M,\alpha} = 0$, stick fixed neutral point can be determined.
The location of the neutral point is readily obtained from equation,

$$dC_{M,c.g} / d\alpha_a = a\ [\ (h - h_{ac,wb}) - \{V_H \cdot \eta \cdot (a_t/a) \cdot (1 - (d\varepsilon / d\alpha))\}\ ]$$

by substituting $h = h_n$ and $dC_{M,\,c.g} / d\alpha_a = 0$,

$$0 = a\left[h_n - h_{ac_{wb}} - V_H \frac{a_t}{a}\left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)\right]$$

$$h_n = [\ h_{ac,wb} + \{V_H \cdot \eta \cdot (a_t/a) \cdot (1 - (d\varepsilon / d\alpha))\}]$$

On substituting the values in the above equation, we get,

$$h_n = 0.25912 + \{0.51507.\ 0.9.\ (0.7207).\ (1 - 0.0081694\}]$$

$$h_n = 0.5904 \approx 0.6 \text{ (Non-Dimensionalized)}$$

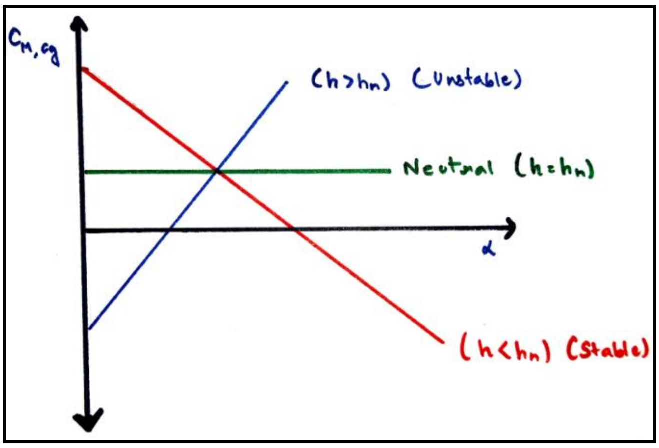


Fig 18 : Effect of the location of the center of gravity, relative to the neutral point, on static stability

From the above plot, it can be concluded that if the C.G is shifting forwards, i.e., towards the leading edge, the aircraft becomes more stable. However, if the C.G is shifted backwards, the aircraft becomes unstable and when it crosses the line of $C_{M,\alpha} = 0$ **(Neutrally Stable)**, then $C_{M,\alpha} > 0$. Hence,

For Static Stability, C.G should be ahead of Neutral Point (N.P).
For Unstability, C.G should be behind of Neutral Point (N.P)
For Neutral Static Stability, $h = h_n$

Static Margin: Static Margin is the distance between the neutral point and the actual C.G position. The static margins talk about the stability. Higher is the static margin, the aircraft becomes more stable.

$$\textbf{Static Margin} = (x_{NP} / MAC) - (x_{C.G}/MAC)$$

$$\textbf{Static Margin} = h_n - h = 0.6 - 0.35$$

$$\textbf{Static Margin} = 0.25$$

Also, the equation,

$$C_{M,\,c.g} / d\alpha a = -a\ (h_n - h) = -a.\ \text{(Static Margin)}$$

On substituting the values, we get,

$$C_{M,\,c.g} / d\alpha_a = -0.02357$$

This shows that the static margin is a direct measure of longitudinal static stability. For static stability, the static margin must be positive.
Moreover, the larger the static margin, the more stable the airplane.

Hence, from the Aircraft Stability and Control concept, it is proved that the UAV designed is longitudinally stable and the values of the parameters lies within the permissible range.

4.4 Locations of Aerodynamic Centre, C.G, Neutral Point and Static margin

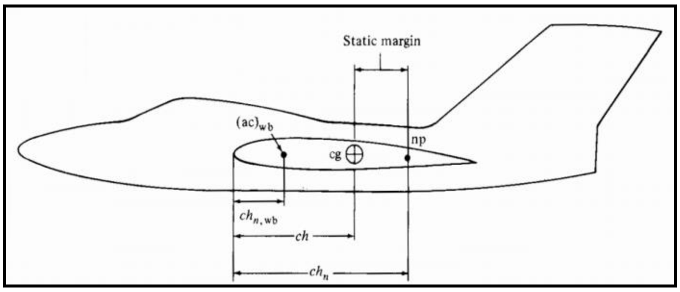


Fig 19 : Illustration of the Locations: A.C, C.G, N.P, Static Margin

S.No.	LOCATIONS	VALUE
1.	Centre of Gravity (C.G) from Leading Edge (L.E)	C.G = c x h = 0.166 x 0.35 **C.G = 0.0581 m = 5.81 cm**
2.	Neutral Point (N.P) from Leading Edge (L.E)	N.P = c x h_n = 0.166 x 0.6 **N.P = 0.0996 m = 9.96 cm**
3.	Static Margin	**Static Margin =** 0.166 x 0.25 **Static Margin = 0.0415 m = 4.15 cm**
4.	Aerodynamic Centre (Wing-Body)	$(a.c)_{w.b}$ = c x $h_{ac,wb}$ = 0.166 x 0.25912 $(a.c)_{w.b}$ **= 0.04301 m = 4.3013 cm**

4.5 Elevator Trim Angle

The longitudinal control during the aircraft flight is done by the elevator located on the horizontal stabilizer. The elevator functions for the pitch control during the flight and is deflected upwards or downwards accordingly to maintain an equilibrium flight. This process allows the aircraft to fly at trim angle of attack and is said to be longitudinally stable.

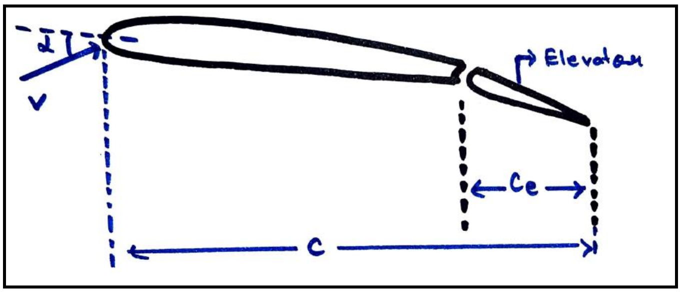


Fig 20 : Elevator Control for Pitch

δ_e is the elevator deflection. If δ_e is positive then the elevator is deflected downwards, generating more lift on the tail section. On the other hand, if δ_e is negative then the elevator is deflected upwards, generating more drag, and thereby reduced lift on the tail section.

c_e**:** Elevator chord length

Elevator Control Effectiveness (τ): The control effectiveness is the measure of how effective the control deflection is in producing the desired control moment.

Control Effectiveness = f_n (Size of flap, V_H))

V_H **= Horizontal Tail Volume Ratio =** $l_t \cdot s_t / (s \cdot c_{mac})_w$

If the elevator size is increased, then the control effectiveness will get increased, since the area of tail will get increased.
The deflection of the elevator will not change the slope of $C_{M,\alpha}$, it only changes the trim angle. With the change in the elevator angle, trim point changes and hence the trim angle also changes.

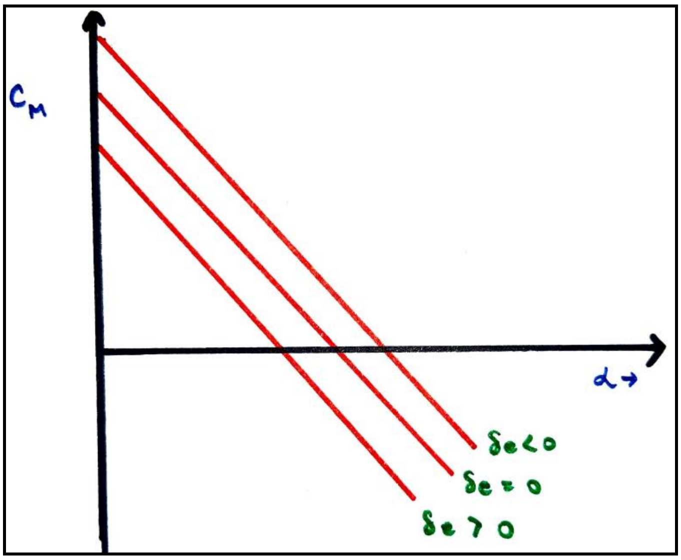


Fig 21 : Change in Trim Angle with Elevator Deflection

Elevator Control Power ($C_{M,\delta,e}$**):** The elevator control power is defined as the ratio of the coefficient of the pitching moment to the elevator deflection.
$C_{M,\delta,e}$ is also known as the **Stability Derivative.** It tells that how much the elevator has the controlling power. It tells that with the elevator deflection, how much pitching moment changes

$$C_{M,\delta,e} = dC_M / d\delta_e$$

It is also given by the relation,

$$C_{M,\delta,e} = -V_H \cdot \eta \cdot C_{L,\alpha,t} \cdot \tau$$

$\mathbf{V_H}$ = Horizontal Tail Volume Ratio

$\boldsymbol{\eta}$ = Tail Efficiency (0.8 to 1.2)

$\mathbf{C_{L,\alpha,t}}$ = 3D lift curve slope of tail section

$\boldsymbol{\tau}$ = Elevator effectiveness parameter

$$\boldsymbol{\tau} = d\alpha_t / d\delta_e$$

The designer can control the magnitude of the elevator control effectives by proper selection of the horizontal tail volume ration and the elevator size.

For the design of fixed wing UAV, the following values have been calculated: -

$\mathbf{C_{M,0}}$ = 0.0344

$\mathbf{C_{L,\alpha}}$ = 0.09431 per degree

$\mathbf{C_{L,\alpha,t}}$ = 0.06797

$\mathbf{C_{M,\alpha}}$ = -0.02267

$\boldsymbol{\alpha_{trim}}$ = 1.52°

$\boldsymbol{\alpha_{cruise}}$ = 4°

$\mathbf{C_{L,cruise}}$ = 0.66017

$\mathbf{C_{L,\delta,e}}$ = 0.04 (From Free Flight Database)

$\mathbf{V_H}$ = 0.51507

$\boldsymbol{\eta}$ = 0.9

Now,

$$\mathbf{C_{L,\delta,e}} = dC_{L,t} / d\delta_e = 0.04$$
$$\mathbf{d\alpha_t / d\delta_e} = 0.04 / a_t$$

After re-substituting, we obtain, **$\boldsymbol{\tau}$ (Elevator Effectiveness Parameter) =** $d\alpha_t / d\delta_e$

$$\mathbf{d\alpha_t / d\delta_e} = 0.04/ 0.0679$$
$$\mathbf{d\alpha_t / d\delta_e} = 0.5891$$

Elevator Control Power ($\mathbf{C_{M,\delta,e}}$) = $-V_H .\ \eta\ .\ CL_{,\alpha,t}\ .\ \tau$

$$\mathbf{C_{M,\delta,e}} = -0.51507.\ 0.9.\ 0.06797.\ 0.5891$$

$$\mathbf{C_{M,\delta,e}} = -0.01856$$

Calculation of Elevator Angle of Trim ($\boldsymbol{\delta_{trim}}$): -

Since the pitching moment of the centre of gravity is given by: -

$$C_{M,\text{cg}} = C_{M,\text{ac}_{\text{wb}}} + C_{L,\text{wb}}(h - h_{\text{ac}}) - V_H\left(a_t\alpha_t + \frac{\partial C_{L,t}}{\partial \delta_e}\delta e\right)$$

The above equation explicitly gives the effect of elevator deflection on moments about the center of gravity of the airplane.

The increment in $\mathbf{C_{M,\ c.g}}$ due only to a given elevator deflection δ_e is given by,

$$\Delta C_{M,\text{cg}} = -V_H \frac{\partial C_{L,t}}{\partial \delta_e}\delta_e$$

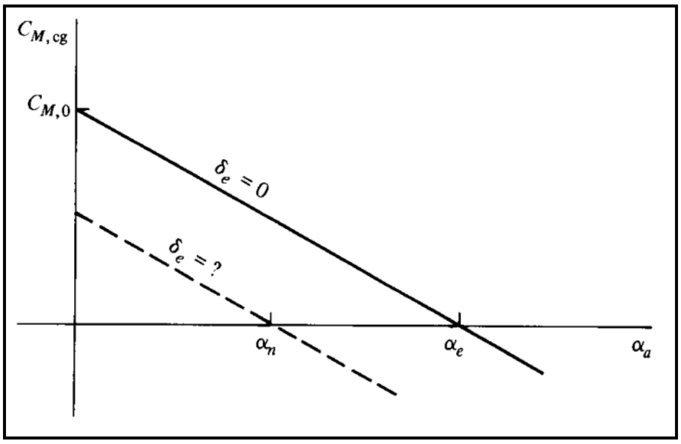


Fig 22 : Elevator Deflection to Maintain the Equilibrium

The geometric equation of the new slope is given by: -

$$C_{M,\text{cg}} = C_{M,0} + \frac{\partial C_{M,\text{cg}}}{\partial \alpha}\alpha$$

Now, if the elevator is deflected by an angle δ_e, the above equation will have an additional term as $\mathbf{\Delta C_{M,\ c.g}}$

$$C_{M,\text{cg}} = C_{M,0} + \frac{\partial C_{M,\text{cg}}}{\partial \alpha}\alpha + \Delta C_{M,\text{cg}}$$

Also,

$$\Delta C_{M,\text{cg}} = -V_H \frac{\partial C_{L,t}}{\partial \delta_e}\delta_e$$

On substitung $\Delta C_{M,\ c.g}$ in the above equation, we get,

$$C_{M,cg} = C_{M,0} + \frac{\partial C_{M,cg}}{\partial \alpha}\alpha - V_H \frac{\partial C_{L,t}}{\partial \delta_e}\delta_e$$

In order to bring the aircraft to the equilibrium position, the elevator deflection must be at its trim location i.e., δ_{trim}. Also, at the trim condition, $C_{M,\,c.g} = 0$.

Hence, on substituting in the above equation, we get,

$$0 = C_{M,0} + \frac{\partial C_{M,cg}}{\partial \alpha}\alpha - V_H \frac{\partial C_{L,t}}{\partial \delta_e}\delta_{trim}$$

$$\delta_{trim} = \frac{C_{M,0} + (\partial C_{M,cg}/\partial \alpha)\alpha}{V_H(\partial C_{L,t}/\partial \delta_e)}$$

On substituting the values in the above equation, we get,

$$\delta_{trim} = [(0.034 + (-0.02267)) \times 4] / (0.51507 \times 0.04)$$

$$\delta_{trim} = -2.7452^o$$

Hence, to trim the aircraft to its equilibrium position, the elevator deflection required is **-2.7452°** i.e., the elevator must be deflected upwards by an angle of 2.7452°.

5. Iterative Battery Weight Estimation

The mission requirement and the conceptual design dimensions are required to be defined before proceeding with the Iterative Weight Estimation.

Since, the project contains the fabrication part also, so the initial sizing and weight were fixed based on the material available in the market and the purpose of fabrication of the UAV.

During the conceptual design study, following parameters were kept for the initial layout of the fixed wing UAV: -

The initial design weights were considered for the UAV for commercial monitoring.

1. Gross Weight of the UAV (W) = 700g (Design Weight)
2. Weight of the Propulsion System (Wp) = 60g to 80g (Based on Market Research)
3. Weight of Payload (Camera for Surveillance) (Wpl) = 10g to 20g (Based on Market Research)
4. Weight of the Structure (Balsa Wood + Foam Board or Corrugated Sheets) (Wstr) = 463 g (400g to 500g) (Design Weight Goal for Structure)

The efficiencies mentioned below are determined based on the market survey and by the study of configuration of the electronic equipment's available.

5. Efficiencies: i. Electrical Efficiency ($\eta_{electrical}$) = 0.98
ii. Motor Efficiency (η_{motor}) = 0.9
iii. Propulsion Efficiency ($\eta_{propulsion}$) = 0.95
6. Maximum Flight Velocity(V_∞) = 5.55 m/s to 10.55 m/s (19.98 ft/s to 37.98 ft/s)
7. Maximum Time of Flight = 15 min to 25 mins
8. Maximum Altitude = 20 m (65.6167 ft)

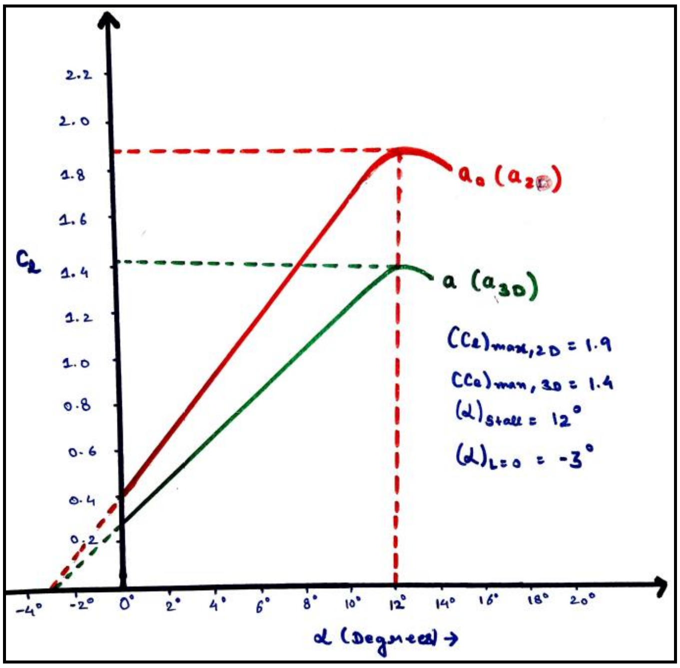


Fig 23 : Lift Curve Slope for Airfoil and Wing

From the flight data, it has been observed that cruise angle of attack lies between 4° to 6°.

9. Cruise Angle of Attack = 4°
10. Coefficient of Lift (C_L) at 4° angle of attack = 0.09431 = $(C_L - 0) / (4-(-3))$

$$(C_L)_{\alpha=4^o} = 0.66017 \text{ (Cruise)}$$

11. Coefficient of Drag (C_D) = $C_{D,0} + k \,.\, CL2 = 0.03 + 0.054 \,.\, 0.662$

$$C_D = 0.05352$$

12. (C_L/C_D)cruise = 0.6601/0.0535

$$(C_L/C_D)\text{cruise} = 12.4528$$

Also, the maximum design time and the cruising velocity is also determined during the conceptual design phase. All these design parameters will be useful in the iterative weight estimation using MATLAB. This software will be used to predict whether the calculated values of the total weight, structure, battery and payload are similar to those as specified during the conceptual design study or not.

13. Weight Ratios = i. Wstr/W = 0.6614 (From the Historical Database)

ii. Wp/W = 0.1028 (From the Historical Database)

14. Specific Energy Density (S.E.D) for **2200mAh LiPo Battery (11.1V)**

S.E.D = (2200 x 11.1)/ (1000 x 07)

S.E.D = 34.8857 Wthr/Kg

15. Output from Brushless Motor: P_s (Shaft Power)

16. Power Available to move the UAV forward: P_A

17. Input Power for the brushless motor form electric source: $P_{electric}$

18. Efficiencies: i. Electrical Efficiency ($\eta_{electrical}$) = $P_{electric}/P_{battery}$ = 0.98

ii. Motor Efficiency (η_{motor}) = $P_s/P_{electric}$ = 0.9

iii. Propulsion Efficiency ($\eta_{propulsion}$) = P_A/P_s = 0.95

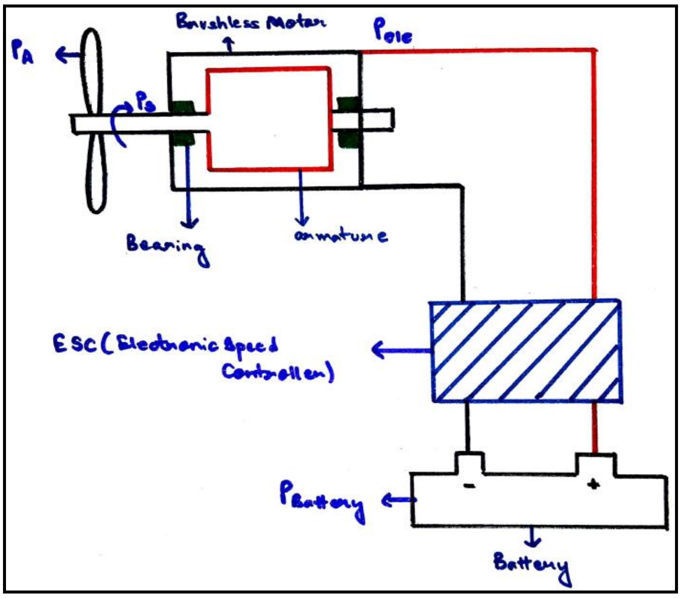


Fig 24 : Typical Architecture of Brushless Motor Propulsion

Approach of Writing the MATLAB Code for Weight Estimation: -

1. Consider the weight and ratios of the baseline UAV.
2. Specify the iteration for convergence: $(W_{i+1} - W_i) < 10^{-10}$ (Criteria for Convergence)
3. New Weight Calculation for UAV: $W_i \rightarrow W_{battery} \rightarrow W_N$ (New Weight of UAV)

$$W_N = W_{str} + W_{propulsion} + W_{battery} + W_{pl}$$

Hence, the new weight of the UAV must be equal to the weight of baseline UAV selected during the initial phase for iterations. Iterations will also determine the weight of the battery which will be required to fly the UAV and maintain longitudinal static stability.

MATLAB Code for Weight Estimation: -

```
clear all
close all
clc

V=input('Enter the Flight Velocity in m/s \n');
t=input('Enter the Total Time of Flight in Hrs \n');
SED=input('Enter the Specific Energy Density of Battery in Wh.Hr/Kg \n');

W_b=0.7;           %Base Line Aircraft Weight in Kg
Wst_W=0.6614;      %Structural Weight Ratio Wstr/W
Wpro_W=0.1028;     %Propulsion System Weight Ratio Wpro/W
W_PL=0.015;        %Payload of EO/IR Sensor in Kg

eff_propulsion=0.95;
eff_electrical=0.98;
eff_motor=0.9;

k=0;
for LbyD = 3:1:14
    k = k+1;
 L_D(k,1)= LbyD;
  W_STR = Wst_W*W_b;
  W_PRO = Wpro_W*W_b;
     i = 0;
    Wi = W_b;
    Wf = 0;
while abs (Wi-Wf)>10^(-10)
   i = i+1;
  if i==1
    Wi = W_b;
  else
    Wi = W;
  end
   Pr = (Wi*9.81*V)/(LbyD);
 P_bat = (1/(eff_propulsion*eff_electrical*eff_motor))*Pr;
 W_Bat = (P_bat*t)/(SED);
     W = W_STR + W_PRO + W_Bat + W_PL;
    Wf = W;
```

```
end

  W_STR = Wst_W*W;
  W_PRO = Wpro_W*W;
        i = 0;
     Wi = W;
     Wf = 0;
 while abs (Wi-Wf)>10^(-10)
         i = i+1;
      Wi = W;
       Pr = (Wi*9.81*V)/(LbyD);
  P_bat = (1/(eff_propulsion*eff_electrical*eff_motor))*Pr;
 W_Bat = (P_bat*t)/(SED);
    W = W_STR + W_PRO + W_Bat + W_PL;
    Wf = W;
 end

          W_Final(k,1) = W;
    W_BATTERY(k,1) = W_Bat;
W_PROPULSION(k,1)= W_PRO;
 W_STRUCTURE(k,1)= W_STR;
end

figure(1)
subplot(4,1,1),plot(L_D,W_Final,'*k')
                      ylabel('W_t_o_t (kg)')
subplot(4,1,2),plot(L_D,W_BATTERY,'*k')
                      ylabel('W_b_a_t (kg)')
subplot(4,1,3),plot(L_D,W_PROPULSION,'*k')
                      ylabel('W_P_r_o (kg)')
subplot(4,1,4),plot(L_D,W_STRUCTURE,'*k')
                      ylabel('W_s_t_r (kg)')
                      xlabel('L/D')
```

Fig 25 : MATLAB Command Window

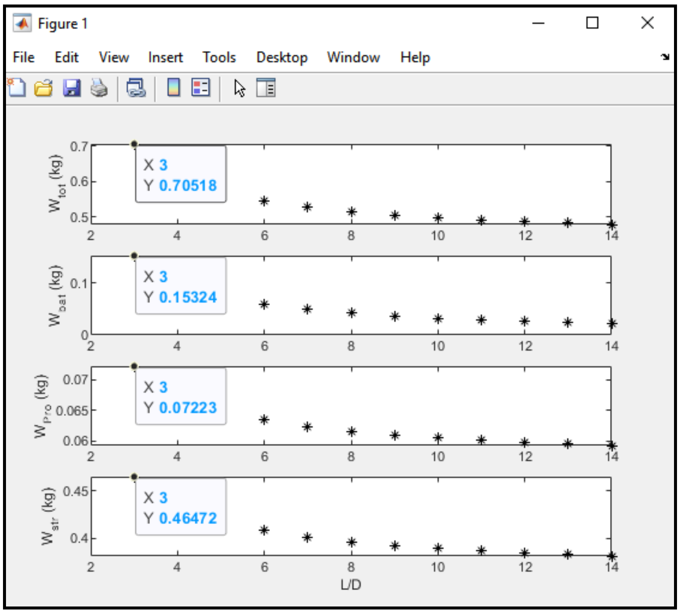


Fig 26 : Battery Weight Determination from MATLAB Iterations

Results and Discussions: From the MATLAB Weight Estimation Program, the following weights were determined for the fixed wing UAV: -

1. Weight of Battery : 153.24 g
2. Weight of Propulsion System: 72.23 g
3. Weight of the Structure: 464.72 g

Hence, it was observed that the weight of the propulsion system and the structure is found in the desirable range. However, the main objective of performing the iterations was to obtain the battery weight for the designed UAV. The weight was found to be 153.24 g which is 21.8914% of the design weight of UAV.

It can be concluded that, from the MATLAB iterations on weight estimation, the gross weight of the UAV is 700 g and for maintaining the longitudinal static stability, the battery required for the design is to be around 153 g.

6. Performance Estimation

The forces acting on the aircraft are: -

- Lift L, which is perpendicular to the flight path direction.
- Drag D, which is parallel to the flight path direction.
- Weight W, which acts vertically toward the center of the earth (and hence is inclined at angle θ with respect to the lift direction).

- Thrust T, which in general is inclined at the angle αT with respect to the flight path direction.

Thrust required for level flight,

T/W=D/L (Level Flight Condition)

Minimum thrust required,

$$T_{R,\,min} = W / (C_L/C_D)_{max},$$

Here, thrust required is minimum when L/D is maximum

Stalling velocity is given by,

$$V_{min} = V_{stall} = (L / (1/2) \,.\, \rho \,.\, s \,.\, C_{L,max})^{(1/2)}$$

Power required is given by,

$$P_R = (2 \,.\, W^3 / (\rho_\infty \,.\, s))^{(1/2)} \,.\, (1 / (C_L^{(3/2)} / C_D)),$$

Here, the power required is minimum when CL(3/2)/ CD is maximum.

For fixed-wing UAV,
V_{stall} = 5.5528 m/s
V_{cruise} = 9.1573 m/s

Power Available for Flight: -

The rate of climb (R/C) is also given by **R/C =** $(P_A - P_R)/ W$,
R/C = 1.12,

$P_A = (R/C) \,.\, W + P_R$

$P_A = (R/C) \,.\, W + (D \,.\, V_\infty)$

$P_A = (1.12 \times 0.7 \times 9.81) + (0.5 \times 1.225 \times 5.553 \times 0.20252\ (0.03 + (0.0541 \times 0.17942))$

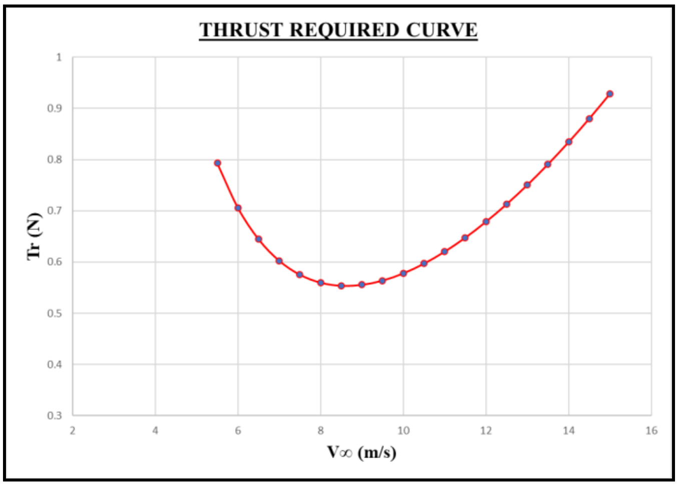
P_A (Power Available) = 8.36413 W

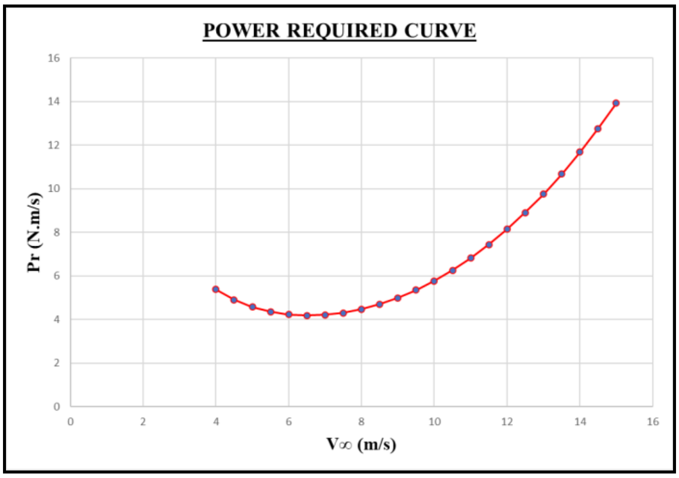
The maximum flight velocity for UAV can be determined when $P_A = P_R$. Hence, the power required plots can be utilized to measure the value of the maximum velocity.

For the fixed wing UAV, the performance calculations are performed for the determination of C_L/C_D, $C_L^{(3/2)}/CD$, thrust required and power required.

V∞ (m/s)	Cl	Cd	Cl/Cd	Cl^(1/2)/Cd	Cl^(3/2)/Cd	Tr (N)	Pr (N.m/s)
4	3.4599757	0.67801362	5.103107	2.743456695	9.492293592	1.3456509	5.38260367
4.5	2.733808	0.43455171	6.291099	3.804893795	10.40184905	1.0915421	4.9119396
5	2.2143845	0.29542638	7.495554	5.037061382	11.1539905	0.916143	4.58071524
5.5	1.8300698	0.21128979	8.661421	6.402584539	11.71717667	0.792826	4.36054314
6	1.537767	0.15800269	9.732537	7.848393805	12.06900094	0.7055714	4.23342864
6.5	1.3102867	0.12293315	10.65853	9.31138193	12.20057961	0.6442727	4.18777271
7	1.129788	0.09909266	11.40133	10.72647373	12.11864125	0.6022982	4.2160877
7.5	0.9841709	0.0824299	11.93949	12.0351213	11.84461586	0.5751503	4.31362696
8	0.8649939	0.07050085	12.26927	13.19204641	11.41104012	0.559691	4.47752823
8.5	0.766223	0.0617796	12.40252	14.16878192	10.85644659	0.5536776	4.70625944
9	0.683452	0.05528448	12.36246	14.95377231	10.22018555	0.5554721	4.99924919
9.5	0.6134029	0.05036712	12.17864	15.54983452	9.538313595	0.5638562	5.35663394
10	0.5535961	0.04658915	11.88251	15.97025106	8.841068983	0.5779081	5.77908106
10.5	0.502128	0.04364793	11.50405	16.23467244	8.151883565	0.5969202	6.26766242
11	0.4575175	0.04133061	11.0697	16.36560253	7.487548779	0.620342	6.82376246
11.5	0.4185982	0.0394849	10.60148	16.38579962	6.859066203	0.64774	7.44901023
12	0.3844417	0.03800017	10.11684	16.31660326	6.272783481	0.678769	8.14522842
12.5	0.3543015	0.03679492	9.629089	16.17702652	5.731545008	0.7131516	8.91439468
13	0.3275717	0.03580832	9.14792	15.98340097	5.235709303	0.7506624	9.75861175
13.5	0.3037564	0.03499447	8.680128	15.74937717	4.783974791	0.7911174	10.6800844
14	0.282447	0.03431829	8.230217	15.48612724	4.374010161	0.8343644	11.681101
14.5	0.2633037	0.03375277	7.800952	15.20264239	4.00291263	0.8802772	12.7640194
15	0.2460427	0.03327687	7.393806	14.90605789	3.66752701	0.9287503	13.9312551

Fig 27 : Calculation of the Performance Parameters





7. Computer Aided Designing (CAD): Detail Design

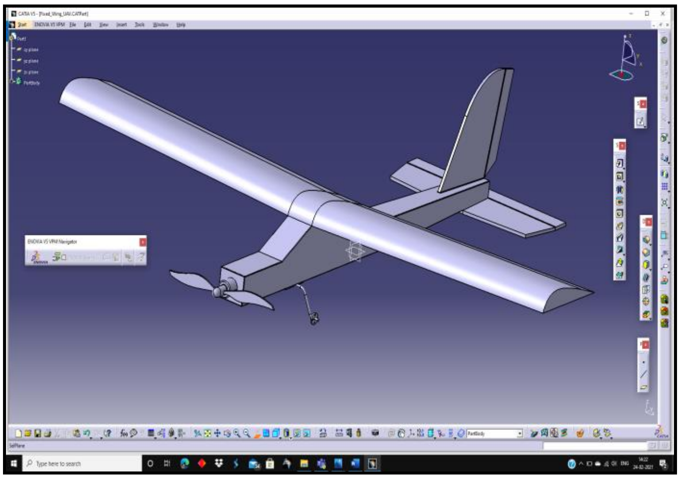


Fig 28 : Fixed-Wing UAV Design using CATIA V5

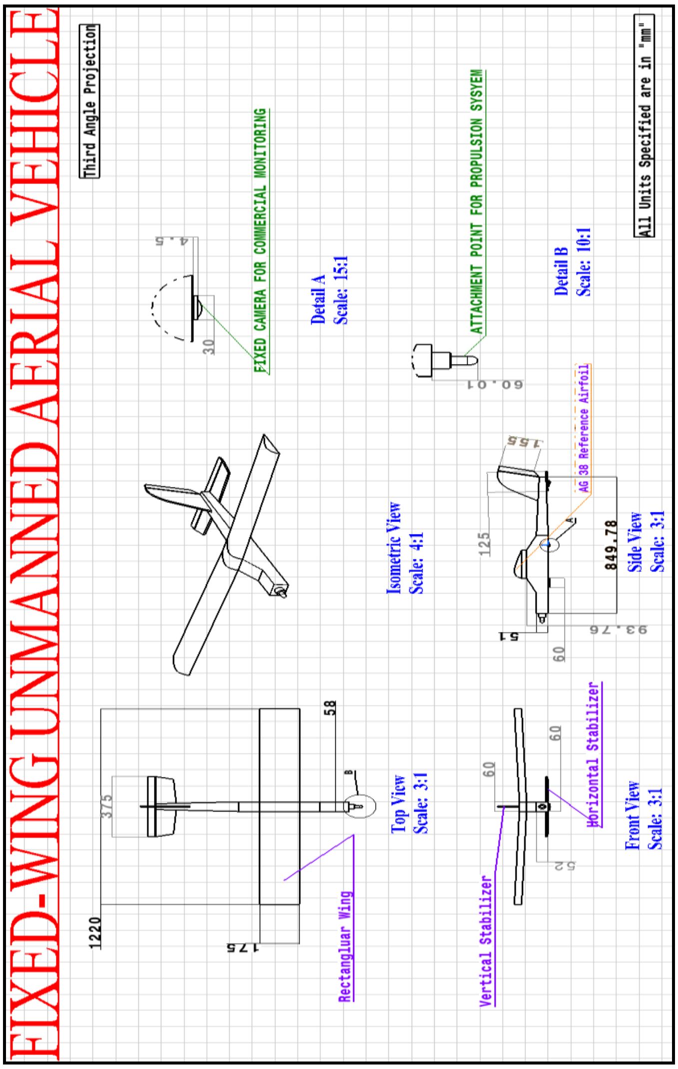


Fig 29 : Drafting Sheet for Fixed-Wing UAV Design

8. Explicit Dynamics Analysis using ANSYS

ANSYS explicit dynamics uses the Lagrangian Frame of Reference. This frame of reference is the most commonly used type and also is the fastest method to represent the analysis on the structural components. It effectively captures the material location of the meshed domain and plots the deformation values as the forces are applied.

8.1 Bird Strike Explicit Dynamics

The bird modelling is done using CATIA V5, where the cylindrical geometry is designed with the diameter D and the length of the geometry as twice of the diameter i.e., L=2 x D. This cylindrical geometry is considered equivalent to the bird geometry and can be taken into consideration while doing the explicit dynamics.
The cylindrical geometry is designed close to the wing leading edge to capture the accurate deformation on the leading-edge surface.

Bird Geometry Details: -
D = 114.0999 mm (From Data Sources)
L = 2xD = 228.1999 mm

Material Selection: -
A. For Fixed Wing UAV,
E = 11.4 GPa (Considering Soft Woods - Balsa Wood or Spruce Wood), **ρ =** 400 kg/m^3, **ν =** 0.3

B. For Cylindrical Body (Equivalent to Bird)
E = 3.2 GPa, **ρ =** 150 kg/m3 (For Indian Bird 'Corvus splendens'), **ν =** 0.3

Velocity Selection: -

The flight velocity of the UAV ranges from 5.55 m/s to 10.55 m/s based on the design guidelines. However, the maximum flight velocity can reach up to a value of 15 m/s if the UAV flies at full throttle setting.

Velocity of the UAV: 13 m/s, **Velocity of the Bird:** 5 m/s, **Relative Velocity of the Bird:** 18 m/s

Boundary Condition: -
Fixed Wing UAV: Fixed in Space, **Cylindrical Body:** Y-Velocity of 18 m/s

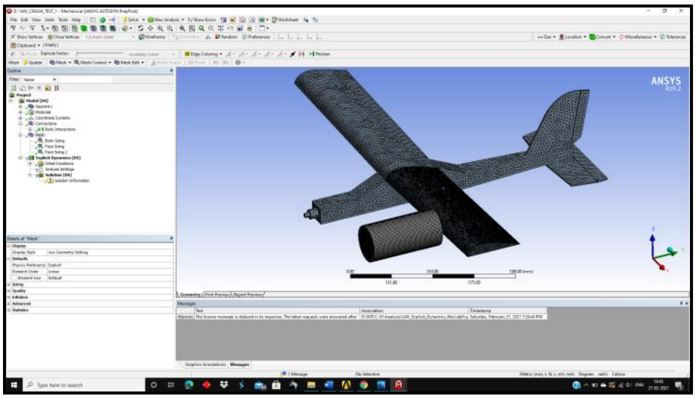


Fig 30 : Fine Mesh-Cylindrical Body, Unstructured Fine Mesh-Fixed Wing UAV

Total Nodes: 154089, **Total Elements:** 387494

Total Deformation Result,

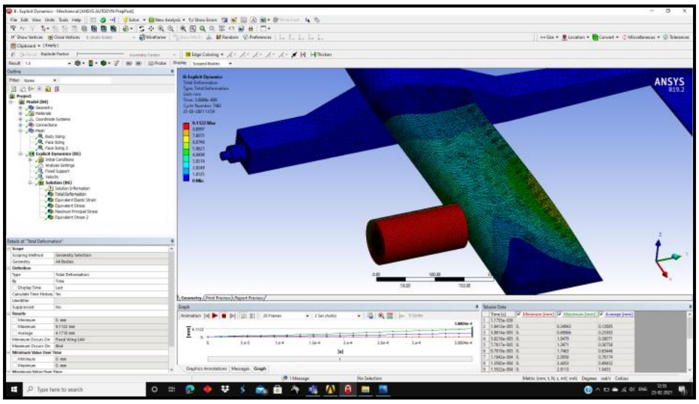


Fig 31 : Detailed Meshed Deformation

Total Deformation: 9.1122 mm

It can be observed that the fixed wing UAV being a stronger material is able to withstand the load. The deformation at the leading edge varies from 4mm to 6mm.

8.2 Underbody Crash Explicit Dynamics

The underbody crash test denotes the belly-landing situation of the UAV with no power or during the stall conditions.

The fixed wing UAV geometry is kept same without any change in the dimension. However, it is assumed that the UAV will undergo belly landing on a concrete base.
Since, ANSYS Fluent does not have the provision to select the entire surface as the ground or base, so during the modelling, a plate is designed at 2mm below the undercarriage of the fixed-wing UAV.

Concrete Base Geometry Details: -
Length (l) = 1300 mm, **Breadth (b) =** 1150 mm, **Thickness (t) =** 50 mm

Material Selection: -
A. For Fixed Wing UAV,
E = 11.4 GPa (Considering Soft Woods - Balsa Wood or Spruce Wood), ρ = 400 kg/m3, ν = 0.3

B. For Concrete Base
E = 30 GPa, ρ = 2300 kg/m^3, ν = 0.18

Drop Height of the UAV: -
h = 50 m

Impact Velocity of fixed-wing UAV: -
The impact velocity is calculated by the relation,
V = $\sqrt{(2.g.h)}$,
g = 9.8 m/s^2, **h =** 50 m

$$V = 31.3049 \text{ m/s}$$

This is the impact velocity specified to the fixed wing UAV, which will impact the concrete base.

Boundary Condition: -
Fixed Wing UAV: Z- velocity of 31.3049 m/s, **Concrete Base:** All four edges fixed in space

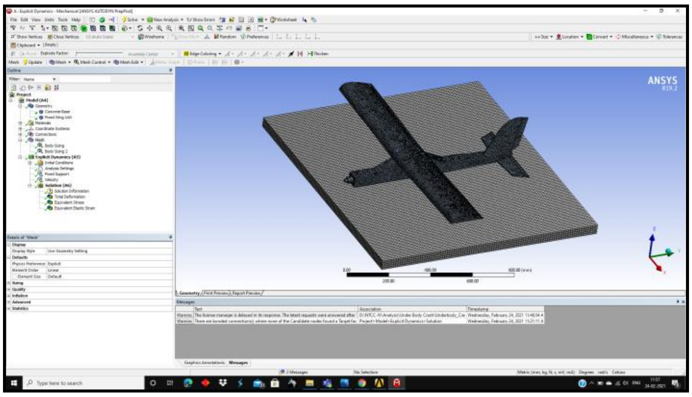


Fig 32 : Fine Mesh-Concrete Base, Unstructured Mesh-Fixed Wing UAV

Total Nodes: 92581, **Total Elements:** 180196

Total Deformation Result,

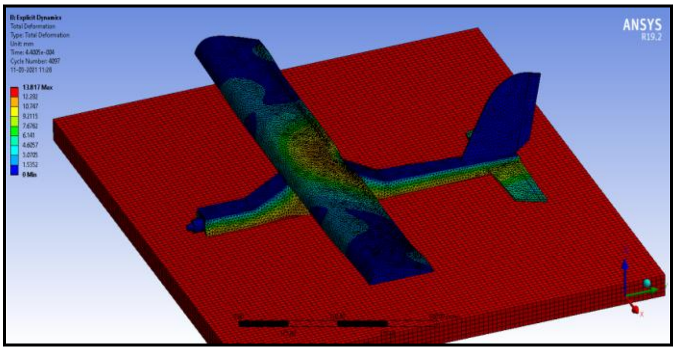


Fig 33 : Detailed Meshed Deformation

Total Deformation: 13.817 mm

It can be observed that the maximum deformation occurs on the UAV because, there is sudden interaction of the UAV with the concrete base, which will damage the underbody balsa wood. Hence, the corrugated sheet thickness must be increased by around **15 mm** at the underbody so that it can absorb the loads and protect electrical components.

9. Additive Manufacturing

The additive manufacturing is a process in which a part or a product is formed by adding the material layer by layer. It is ideal for engineering applications like prototyping, manufacturing tooling, etc. The printing methodology adopted for fixed-wing UAV is **Fused Deposition Modeling (FDM)**

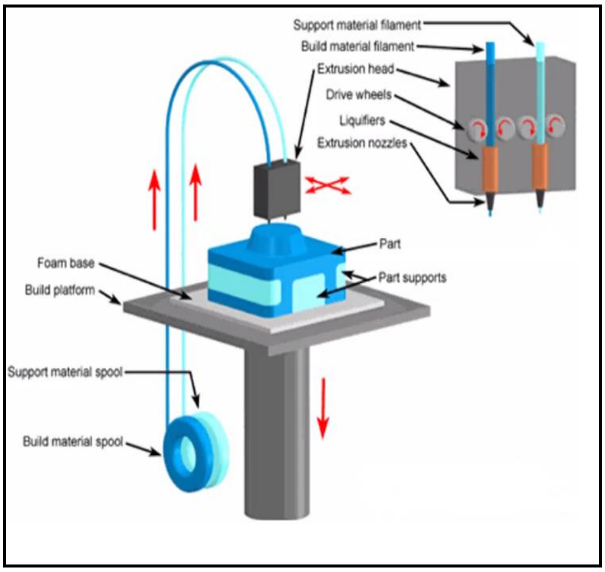


Fig 34 : Fused Deposition Modelling

9.1 3-D Printing Configuration Setup

1. Technique Used: FDM
2. 3-D Printer: ORIGINAL PRUSA I3 MK3
3. STL File Importing Software: FlashPrint
4. Slicing and gx File Generation Software: Ultimaker Cura
5. Material Used: PLA

Ultimaker Cura Print Settings: -
Quality: 1. Layer Height: 0.25 mm
2. Initial Layer Height: 0.25 mm
Material: 1. Printing Temperature: 215°C
2. Initial Printing Temperature: 215°C
3. Final Printing Temperature: 215°C
4. Build Plate Initial and Final Temperature: 60°C
Speed: 1. Print Speed: 30 mm/s
2. Travel Speed: 120 mm/s
3. Initial Layer Print Speed: 15 mm/s
4. Initial Layer Travel Speed: 60 mm/s
5. Retraction Speed: 35 mm/s
Cooling: 1. Maximum Fan Speed: 100 mm/s
Support: 1. Support Structure: Normal
2. Support Placement: Touching Build Plate
3. Support Pattern: Zigzag
Dimensions of the UAV: 183.0 x 146.6 x 35.1 mm (Scale reduced to 15%)
Estimated Weight of the 3-D Printed UAV: 42 g
Estimated Time of 3-D Printing: 4 hours 2 minutes
PLA Spool Parameters: 1.75 mm, 1044 g, white color
Nozzle Diameter: 0.4 mm

Fig 35 : ORIGINAL PRUSA I3 MK3

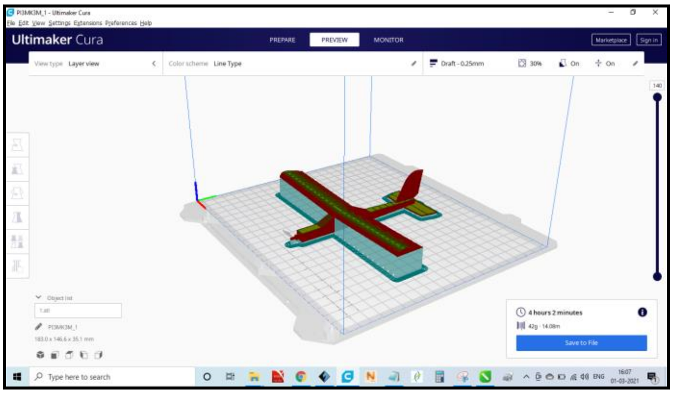


Fig 36 : Slicing Using Ultimaker Cura

9.2 3-D Printed Fixed-Wing UAV

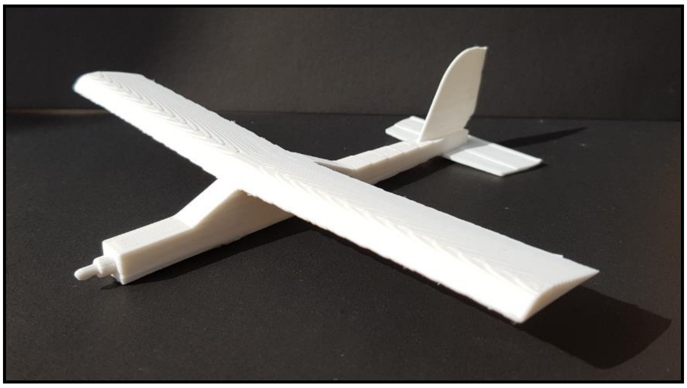


Fig 37 : Isometric View

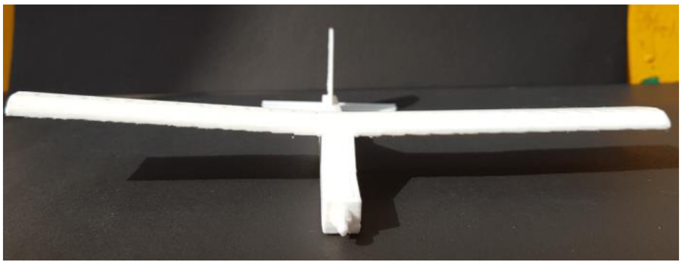


Fig 38 : Front View

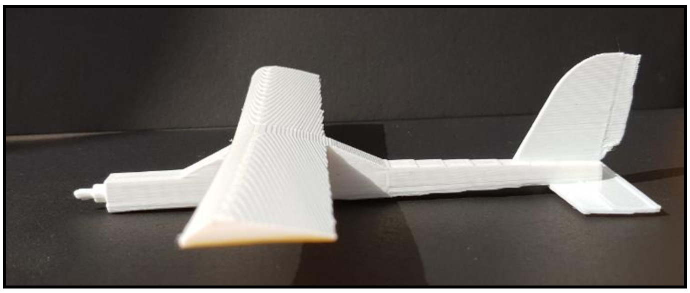


Fig 39 : Side View

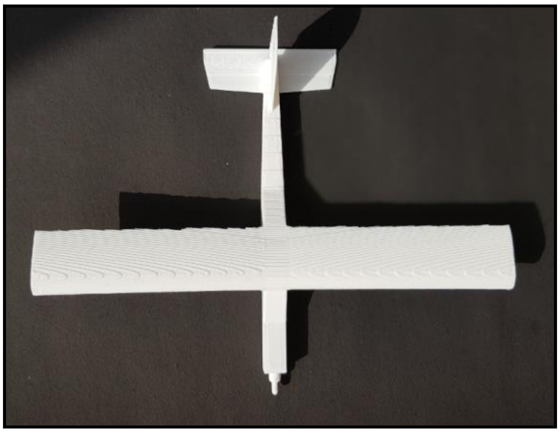


Fig 40 : Top View

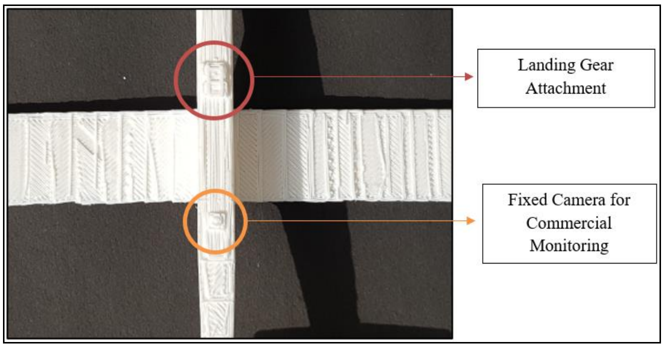


Fig 41 : Underbody View

10. Fabrication of Fixed-Wing UAV

The final design stage of the entire design guidelines is the fabrication of the product. The 3-D printed prototype using FDM technique was done for the representation purpose and to highlight the features of the final fabricated part. In the current stage, the actual fixed wing UAV will be fabricated by the selection of appropriate lightweight materials, components, etc. The main objective lies to manufacture the Fixed-Wing UAV which meets the mission requirements i.e., commercial monitoring.

10.1 Material Selection

1. Wood: The selection of the appropriate wood is the main requirement for the construction fixed-wing UAV. Since, the design weight of the UAV is around 0.7kg, so it is required that the lightest material be selected. The selected material should also have a greater strength to meet the structural strength aspects and safety during flight.

The wood selected for the fabrication of the fixed-wing UAV is Balsa Wood.
Balsa Wood Properties: -

- Lightest wood available for aeromodelling

- Wood Type: Hardwood
- Density: 120-220 kg/m^3
- Color: White, yellow, pale straw, light brown
- Elastic Modulus: 3.71 GPa

Advantages of balsa wood: -

- It is a high strength timber and extremely light in weight.
- It is easy to cut (CNC or laser), shape and carve.
- Easily available in the form of sheets and blocks.
- It retains a lot of water, which increases its strength.

2. Covering Material: The covering material used for the fixed wing UAV are the corrugated sheets. They are also known as coroplast/ fluted polypropylene sheets. They have parallel hollow flutes which make them very light material and strong.
These materials usually crush upon heavy impact instead of breaking in the case of the thermo-foam or foam board. These sheets can easily be formed into any shape or profile by certain amount of bending. These are available in various weights and colors.

10.2 Electrical Components

1. Brushless Motor

The brushless motor is the powerplant of the fixed-wing UAV. The brushless motor used is of the G-Power series and runs at 1500 KV.

- Shaft Diameter: 4 mm
- Shaft Length: 49 mm
- Brushless DC motor
- Maximum Pulling Capacity: 1200 g

Fig 42 : 1500KV Brushless Motor

2. LiPo Battery

The Lithium Polymer battery is the power hub for the propulsion system and other electrical components.

- Model No: ORANGE 2200/3S-30C
- Weight: 175.0g
- Voltage: 11.1V
- Dimensions: 23x34x106(mm)
- Charge Rate: 1-3C Recommended, 5C Max

Fig 43 : 2200 mAh LiPo Battery

3. Electronic Speed Controller (ESC)

It regulates the speed of the electric motor and can be controlled by the transmitter by the user. It also converts the battery voltage down to 5V, which is required by the transmitter to work. This property is known as the Battery Elimination Circuit (BEC).

- Model: SIMONK 30A.
- Constant Current: 30A (Max 40A < 10 sec).
- BEC: 5V 2A.
- Suitable Batteries: 2-3S LiPo

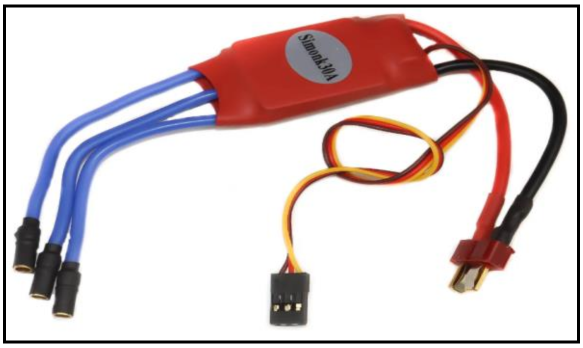


Fig 44 : Electronic Speed Controller

4. Transmitter

The transmitter is the remote control of the fixed-wing UAV. The selected configuration is of 2.4GHz frequency (band-range) with 6-channel arrangement. The transmitter sends the signal to the receiver which controls the servomechanisms to move the control surfaces and turn on and off the camera.
Transmitter Model: Fly-Sky CT6B 2.4GHz 6CH Transmitter

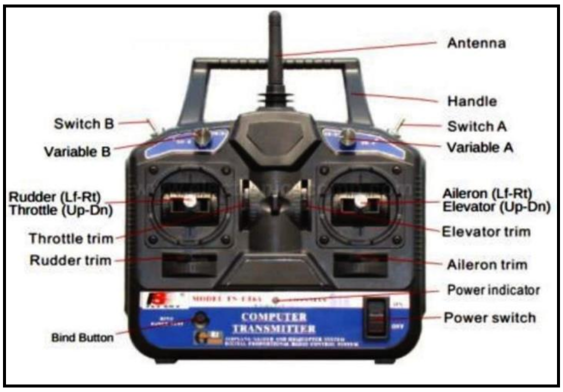


Fig 45 : Fly-Sky Transmitter

5. Receiver

It is an electronic device which uses the radio waves and extract the useful information such as the movement of the control surfaces, propulsion system and the camera function. It receives the signals from the transmitters and send signals to the servo motors for control surfaces and the ESC for the motor control.
Receiver Model: Fly-Sky-R6B Receiver

Fig 46 : Fly-Sky Receiver

6. Servo Motors

The servo motors are used as the rotary actuator in the fixed-wing UAV to move the control surfaces i.e., controlling the movement of ailerons, elevator and rudder.

- Weight: 9 gm
- Operating voltage: 3.0V~ 7.2V

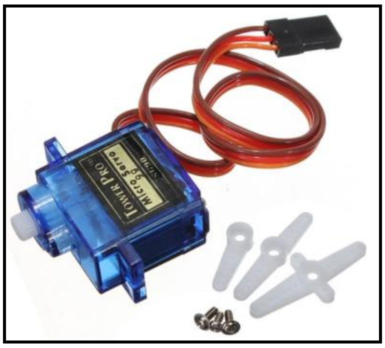
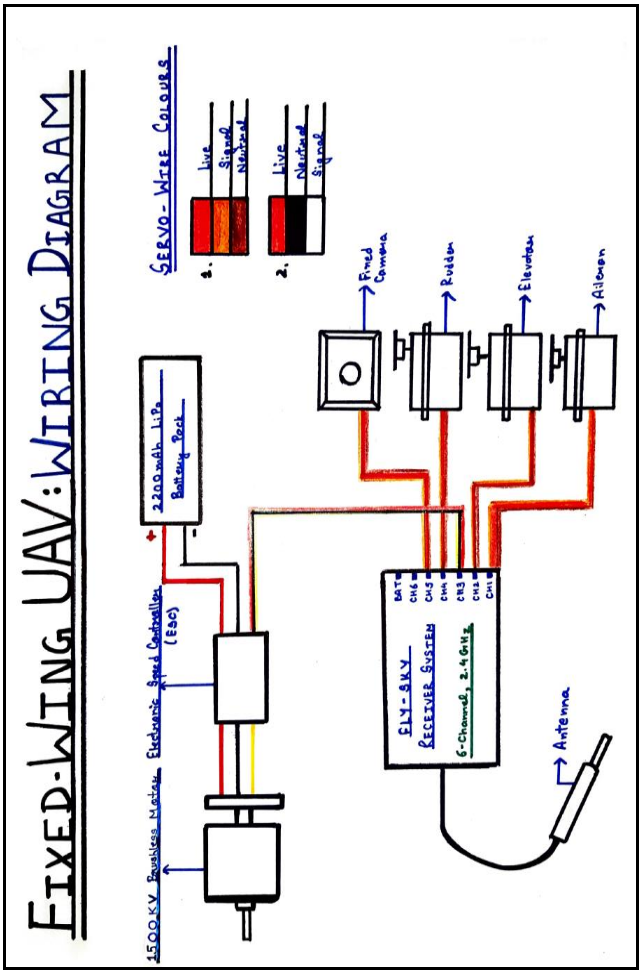


Fig 47 : SG-90 Servo Motor

10.3 Component List for Fabrication

COMPONENT LIST FOR FIXED WING UAV FABRICATION		
S.No	Product Name	Quantity
1	Li-Po Battery (2200 mAH) : Model : 22650	1
2	Simonk 30 ESC : 5V 2A	1
3	G-Power Series Brushless Motor : 1500 KV	1
4	Fly Sky 6-Channel Tranmitter and Receiver	1
5	Propeller (2-Blade)	1
6	SG-90 Servo Motors	4
7	Camera	1
8	Z Pull/ Push Control Steel Rods	4
9	Corpac Corrugated Sheets	10
10	Balsa Wood	8
11	Generic 40 Watt Glue Gun	2
12	Araldite 180g	4
13	Wires	14
14	Hard Cardboard	2
15	Landing Gear	2

10.4 Fixed-Wing UAV Wiring Diagram



10.5 Final Fabricated Fixed-Wing UAV

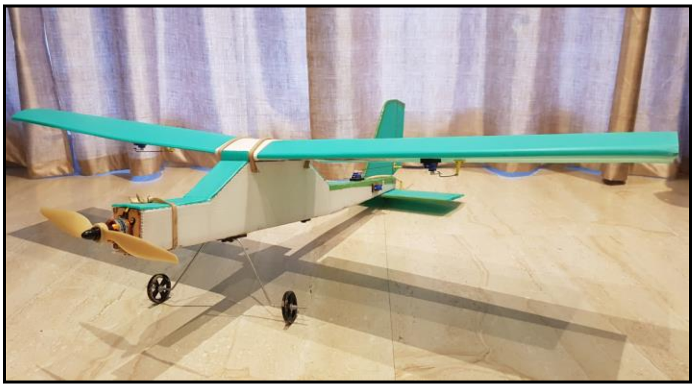


Fig 48 : Isometric View

Fig 49 : Front View

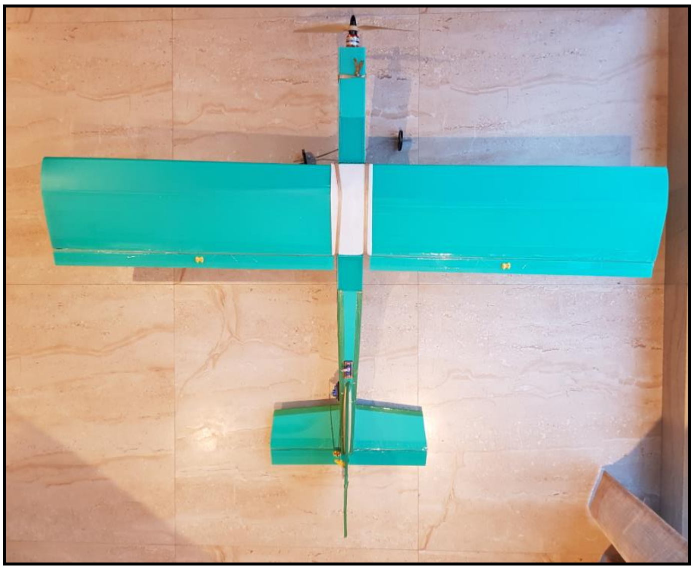


Fig 50 : Top View

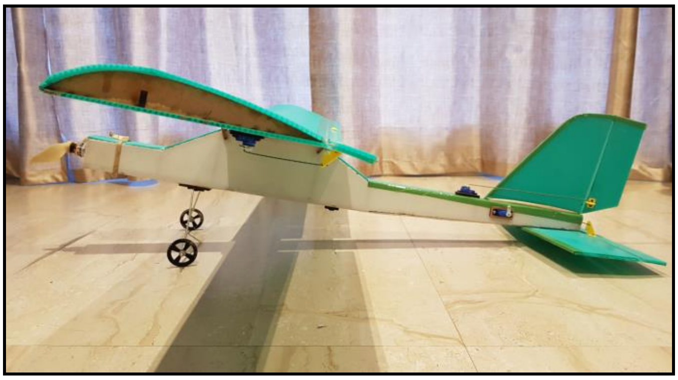


Fig 51 : Side View

11. Fixed-Wing UAV Flight Testing

Fig 52 : Flight Testing-Cruising Flight

Fig 53 : Image Captured by Fixed Camera

12. Conclusions

The report discusses about the detailed procedure for the development of the fixed wing UAV to meet its mission requirement i.e., commercial monitoring. It is well-known that most of the UAV's flying today are multi-rotor aircrafts, commonly known as quadcopter or drones. However, this project highlights the advantages of the fixed wing UAV over the conventional multi-rotor systems.

The project begins with the discussions on the existing fixed-wing design available in the global market. The mission requirement is then specified, which is the commercial monitoring (surveillance), and the project approach is selected, flight dynamics approach. The details are then highlighted on the design guidelines i.e., the design methodology. The entire design and the fabrication of the fixed-wing UAV is divided into three major design processes: conceptual design, preliminary design, and the detail design. In the conceptual design, sketch for the fixed-wing UAV was developed, wing sizing and body sizing was performed. In the preliminary design, all the analytical and the computational work was performed, beginning with the aerodynamics calculations for the wing and the tail configuration, calculations for the longitudinal stability and control, iterative battery weight estimation and performance estimation. After all the design calculations, final stage is the detail design where the CAD model is generated, drafting sheets is prepared, structural analyses are performed. 3-D printing is performed to generate the porotype model. Finally, the fixed-wing UAV is fabricated after the proper selection of material and proper dimensioning, which justifies the scope of the project. The flight testing verifies that the mission requirement has been met and it also provides a proper justification to the report.

The highlights of the fixed-wing fabricated UAV are specified below: -

- Low cost
- Better aerodynamic stability
- High range and endurance
- Better photographic results
- Better control of flight parameters
- Higher flight safety

13. References

[1] Gaurav Singhal, Babankumar Shyam Bansod, "Unmanned Aerial Vehicle Classification, Applications and Challenges: A Review", Indian Space Research Organization.

[2] Ram Gopal Lakshmi Narayanan, Oliver C., "Joint Network for Disaster Relief and Search and Rescue Network Operations", Wireless Public Safety Networks 1,2015.

[3] Pei-Hsiang Chung1, Der-Ming Ma2 and Jaw-Kuen Shiau2, Design, Manufacturing, and Flight Testing of an Experimental Flying Wing UAV, 1Department of Mechanical and Electro-Mechanical Engineering, Tamkang University, New Taipei City 25137, Taiwan, 2Department of Aerospace Engineering, Tamkang University, New Taipei City 25137, Taiwan.

[4] Seyyed Khandani, "ENGINEERING DESIGN PROCESS", Ph.D., Education Transfer Plan, August 2005.

[5] Özgür Dündar, Mesut Bilici, TarıkÜnler, "Design and performance analyses of a fixed wing battery VTOL UAV", Necmettin Erbakan University, Faculty of Aeronautics and Aerospace, Aerospace Engineering Department, 42090 Meram, Konya, Turkey.

[6] Snorri Gudmundsson, "The Anatomy of the Wing", BScAE, MScAE, FAA DER(ret.), in General Aviation Aircraft, Design, 2014.

[7] İbrahim Halil Güzelbey1, Yüksel Eraslan2, Mehmet Hanifi Doğru3, "Effects of Taper Ratio on Aircraft Wing Aerodynamic Parameters: A Comperative Study", 1,2Aircraft and Aerospace Engineering Department/Gaziantep University, Turkey, 3Pilotage Department/Gaziantep University, Turkey , İ.H. Güzelbey (0000-0002-4235-9746), Y. Eraslan (0000-0002-5158-5171), M.H.Doğru (0000-0001-6038-8308)

[8] John D. Anderson, Jr., "Introduction to Flight", Curator for Aerodynamics, National Air and Space Museum, Smithsonian Institution and Professor Emeritus, University of Maryland.

[9] John D. Anderson, Jr., "Fundamentals of Aerodynamics", Curator of Aerodynamics National Air and Space Museum, Smithsonian Institution and Professor Emeritus, University of Maryland.

[10] Dr. Subrahamanyam Saderla, "noc18-ae07 Lec 16 – Lecture 16: Iterative Weight Estimation and Wing Sizing", Aerospace Engineering, IIT Kanpur.

[11] Dr. Subrahamanyam Saderla, "noc20-ae04-lec23 - Lecture 23: Subroutine for Weight Estimation", Aerospace Engineering, IIT Kanpur.

[12] NPTEL Lectures, "Static Stability and Control - Elevator Control power, Elevator Angle to trim and Estimation of Stick Fixed Neutral Point".