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STABILIZING THE TILT ROTOR UNMANNED AERIAL VEHICLE BY TILT H-TAIL MECHANISM WITH THE PERFORMANCE PARAMETERS IN DIFFERENT FLIGHT PHASES

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Abstract

This research work focus on the stability part of a tilt rotor unmanned aerial vehicle and the different parameter requires operating the tilt rotor unmanned aerial vehicle during its different phases of flight. The work starts from the definition of tilt rotor mechanism and proceeding to the design of the same. Combination of a brushless DC motor with appropriate variable pitch propeller is used as a thruster which will produce necessary thrust in required direction to lift the whole body with electronic payload that is required to generate thrust. The thrusters being situated at the tips of the wing eases the process of tilting through servo motors. The combination of the rectangular wing and the tail consisting of vertical stabilizers as well as horizontal stabilizer will make sure the aircraft is stable during different phases of tilt rotor unmanned aerial vehicle's flight. The 'H-tail' configured tail being tilted up and down with the servo motor helps the aircraft to arrest all the unnecessary disturbances by generating the moments about center of gravity in required angle. After defining the different flight conditions, i.e. hovering, climbing and forward motion, the research work proceeds to calculating the different parameters required in rotary aircrafts. These parameters will help us to understand the heat of the problems that will be associated by the tilt rotor mechanics. The results after the calculations are more satisfying. From the results, this research work will conclude that the design of tilt rotor unmanned aerial vehicle will operate in different phases of flight comfortably. The major step taken for the stability of the aircraft in helicopter mode, airplane mode as well as in the transition period is the design of the tail. The tail is designed in such a way that it can counteract any disturbances occurring in horizontal plain or vertical plane. The design of the tail makes this research paper work unique. Servos are used to tilt the whole tail which consists of horizontal stabilizer as well as vertical stabilizers. To define the stability this research work considered the contribution of the wing, fuselage and tail which together will define the longitudinal stability of the aircraft. The results after the stability calculations are more satisfying. From the results this research work concluding that the designed tilt rotor unmanned aerial vehicle will be stable in its flight.

Keyword: Tilt Rotor, Stability, UAV, Different flight phases

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1. INTRODUCTION

Tilt rotor Unmanned Aerial Vehicles are the best version of vertical take-off and landing as it takes off like a helicopter and cruises like an airplane.

One of the major advantages of this kind of configuration is that while it is being operated in helicopter mode, thrust in vertical direction helps the vehicle to project in the air with least land space requirements for takeoff and in aero plane mode, the horizontal thrust cruise the aircraft in desired direction where the wing produced to keep the aircraft in the desired altitude.

Unmanned aerial vehicles are becoming popular not only in the aviation sector but also in other sectors as well. Unmanned aerial vehicles are generally controlled by remotes through radio waves or programmed to fly automatically. If the UAVs are remotely operated than stability of the vehicle has to be maintained manually with great human efforts. If the UAVs are controlled and operated automatically, the stability of the vehicle is taken care by preprogrammed micro controller.

The main purpose of the research work is to design a tilt rotor UAV so that it is stability during helicopter and aero plane mode.

2. LITERATURE REVIEW

1) Presents a fresh model of big-size tilt-rotor aircraft that can be operated as a helicopter as well as fixed-wing airplane. Aerodynamics of the tilt-rotors are based on the blade element theory method. This method was used in transition phase majorly.
2) Performed calculations on XV-15 tilt rotor aircraft performance along with load and stability calculations. Later these are compared with the wind tunnel data along with the flight measurements. This comparison is performed to test the tilting capability for the given tilt prop rotor aircraft. Also, this helps in defining the requirements for any additional experimental data and also for further analysis development.
3) Constructed the conceptual design and aero dynamical model of a small-sized tilt-rotor unmanned aerial vehicle. The linearized state-space models were obtained around the trim points for airplane mode, helicopter mode and also the conversion modes. The controllers were designed using the Linear Quadratic Regulator (LQR) method. Also, the gain-scheduling was employed in order to attain the recreation for the complete flight regime.
4) Presents an enhanced proportional–integral–derivative (PID) controller for all the six-degrees-of-freedom (DOF) flight stability control for an unmanned aerial vehicle (UAV). Because of the unique stable structure of rotors, Quad configured tiltrotor hybrid UAV exhibits certain application value. Dissimilarity in the model

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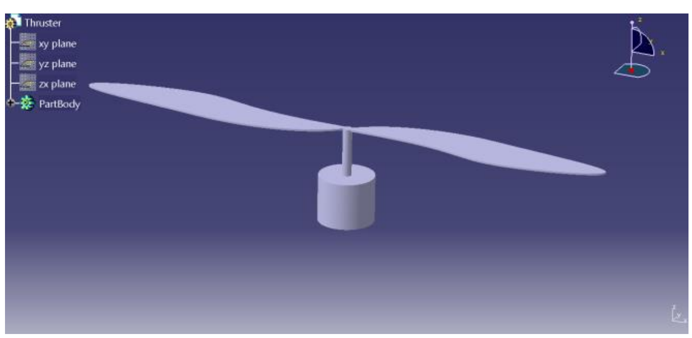
dynamics and aerodynamics as the result of tiltrotor mechanism was a great technical challenge, which has been addressed in this paper.

3. DESIGN

Thrusters:

The tilt rotor UAV which is designed must contain two BLDC motors with propellers which will take care of the thrust during both phase of the operation. We can call the combination of the BLDC motors with propeller as thruster so the thrusters are generally used to produce the vertical thrust in helicopter mode and horizontal thrust in aero plane mode. These thrusters also help the UAV to maintain the altitude during transition phase. In the transition phase thrust vector is neither perpendicular nor parallel to the earth surface. So, when the thrust vector is resolved in two components, these two components give the vertical and horizontal trust which helps the UAV to maintain altitude as well as horizontal speed. Though there will be little loss of lift during this transition piloting and preprogramming losses can be minimized but it can't be nullified because gravity will affect the response time. These thrusters are regulated through BLDC electronic speed controller which regulates the voltage supplies by the battery and hence the speed of the motor. If the speed of the motor is changed i.e., RPM is changed, then the velocity contributing the thrust from the thrusters will change, this in turn affects the position of the UAV in the air. The BLDC electronic speed controller has three terminals namely positive, negative, and pulse signal. The pulse signal terminal being situated at the middle sends PWM signals to the BLDC motor which is responsible to change the voltage supplied to the motor for a particular time. The other two terminals positive and negative used to change the polarity of the motor. Polarity of the terminals plays a major role in the tilt rotor UAV as the direction of rotation need to be altered accordingly. This is done to counter the moment produced by the thrusters.

Fig 1: Thruster designed in CATIA



Position of the thrusters:

In case of tilt rotor design the thruster is not going to remain fixed but it is going to be tilted so that the thrust direction also gets tilted as it is aligned in the direction of the axis of motor and propeller. So here the position of the thrusters means the position of the axis about which the thrusters are needed to be pivoted. The axis about which the

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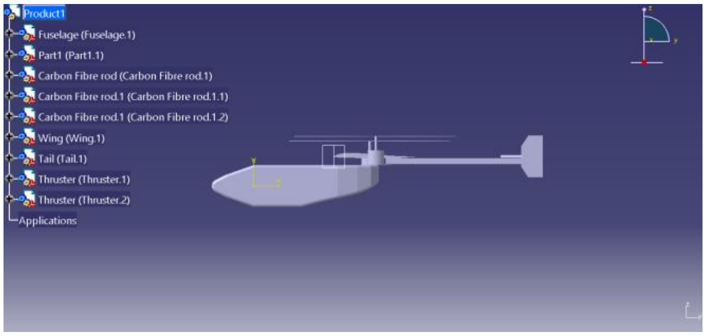
wings are needed to be pivoted is kept near to the centre of mass. In this research work the first point is kept above the centre of mass at the end of the tip of the wings through this position it is easy for us to pull the weight, which is acting through the centre of mass in desired direction by tilting the thruster in that direction by aligning the thrust rotor in that direction.

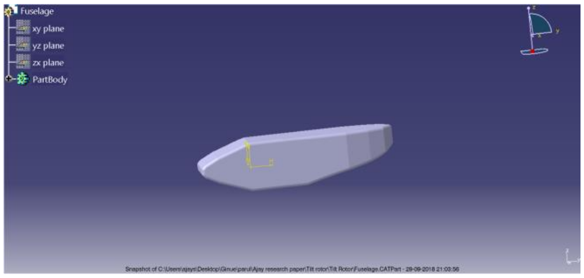
Fig 2: Positioning of Thrusters



Fuselage Design:

The fuselage is made up of chloroplast material which will be carrying the electronic equipments like batteries, receiver, ESC's and other connectors. The electronic are placed at the aft end of the fuselage so that the thrusters can act from the aft end of the fuselage as centre of gravity is located there. The fuselage front part is given aerodynamics shape which help to cruise in aero plane mode. The fuselage will having the flat bottom and with no landing gear as the aircraft will take off and land vertically like helicopter.

Fig 3: Fuselage Design



Wings:

The wings will be rectangular wing with bell A821201 (23%) airfoil which is most suitable for tilt rotor aircrafts. The aircraft has flat bottom and stream lined top which helps the wing to float even when the trust direction is tilted at different angles. The wing will strengthen through main wing span which is made up of carbon fibre having square section. The pivots of the thrusters will be mounted at the end of the main wing spare through servo motors. This servo motors being situated at the end of the wingtips will tilt the thrusters in desire direction. The material used in the wing is acrylic foam. The skin of the wing is coated with taper to reduce the skin friction drag.

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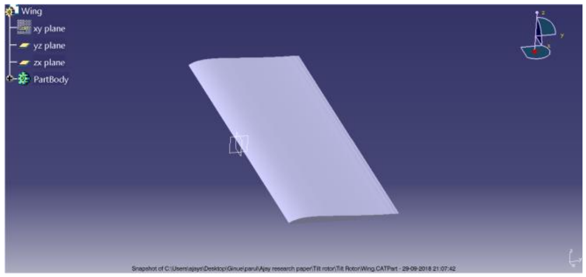
Fig 4: Wing made in CATIA

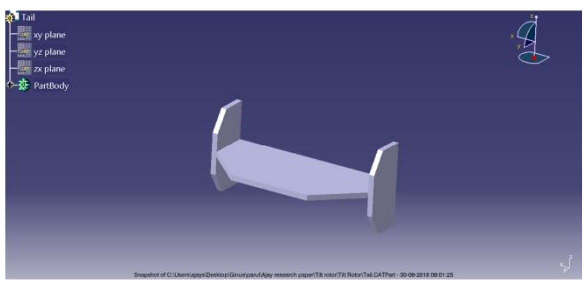


Tail:

This tilt rotor UAV will be having a horizontal stabilizer with no elevators on it and at the vertical stabilizers being situated at the ends of the horizontal stabilizers. The vertical stabilizers also will not be rudders. The tail part will be placed on the end of the tail boom of the T-section.

The uniqueness of the tail is that the whole tail can tilted up and down like an elevator. The horizontal stabilizer and vertical stabilizer itself will tilt up and down. The mechanism is done by servo motors mounted at the end of tail boom. The tail is made to tilt up and down because to provide stability at desired angle as thrust is also oscillating in different angle.

Fig 5: H- Tail Design



4. METHODOLOGY

Stability

If the forces and moments after the disturbances bring the aircraft back to the decided initial position we can call that the aircraft is stable. Depending upon the three axis system we can classify the stability as longitudinal stability, lateral stability and directional stability.

In this tilt rotor UAV, we can notice there is no aileron as well as rudder. Even the elevator is absent, but the whole tail can tilt up and down. These up and down motion of the tail will create the change and pitching motion above the lateral axis. So it

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contributes to longitudinal stability. This paper mainly focuses on the longitudinal stability of the unmanned aerial vehicle.

The definition of Longitudinal stability says that the plot between the coefficient of pitching moment about center of gravity with angle of attack should be linear as well as the slop must be negative. So after considering the stability of all the contributions of parts of aircrafts, if we achieve $\frac{dC_m}{d\alpha}$ as negative, the aircraft is said to have longitudinal stability. Also the value of Cm0 should be positive.

Wing contribution

If we consider the wing contribution alone in the aircraft, then we need to place the center of gravity before the aerodynamic center, but since we need to consider tail contribution as well as the fuselage contribution, we may get the solution other way. To obtain wing contribution, we keep the thrusters in airplane mode.

$$C_{m_{ow}} = C_{m_{acw}} + C_{L_{ow}} \left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) \quad \text{(equa-4.1)[5]}$$

$$C_{m_{\alpha w}} = C_{L_{\alpha w}} + C_{L_{ow}} \left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) \quad \text{(equa-4.2)[5]}$$

Where

$C_{m_{ow}}$ is the coefficient of zero angle of attack pitching moment of wing

$C_{m_{acw}}$ is the coefficient of pitching moment about aerodynamic center of plain

$C_{L_{ow}}$ is the coefficient of lift at zero angle of attack of wing

x_{cg} is the distance between the center of gravity to wing's leading edge

x_{ac} is the distance between aerodynamic centers to leading edge

$\bar{c}$ is the chord length of the wing

$C_{m_{\alpha w}}$ is the slop of the curve plot between coefficients of pitching moments with angles of attack of UAV's wing

$C_{L_{\alpha w}}$ is the slop of the curve plot between coefficients of lift with angles of attack of the UAV's wing

Tail contribution

In this tilt rotor UAV, tail contribution is very important as it will affect the C_{m_α} value much more than other contributions. For obtaining the tail contribution, we keep the thrusters in airplane mode; also the tail is not moved. Then in this case, the tail would get affect by downwash created by wing.

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$$C_{m_{0_t}} = \eta V_H C_{L_{\alpha_t}} (\varepsilon_0 + i_w - i_t) \qquad \text{(equa-4.3)[5]}$$

$$C_{m_{\alpha_t}} = \eta V_H C_{L_{\alpha_t}} \left(1 - \frac{d\varepsilon}{d\alpha}\right) \qquad \text{(equa-4.4)[5]}$$

Where

$C_{m_{0_t}}$ is the coefficient of pitching moment at the zero angle of attack of UAV's tail

η is ration of dynamic pressures of tail to wing, also called the tail efficiency

V_H is horizontal tail volume ratio

$C_{L_{\alpha_t}}$ is the slope of the curve plot between coefficient of lift with angle of attack of UAV's tail

ε_0 is downwash angle at zero angle of attack

i_w is the angle made by the wing chord to the fuselage reference line

i_t is the angle made by the tail chord to the fuselage reference line

$C_{m_{\alpha_t}}$ is the slope of the curve plot between coefficients of pitching moment with angle of attack of UAV's tail

$\frac{d\varepsilon}{d\alpha}$ is the rate of change of downwash angle with angle of attack

Fuselage contribution

Purpose of the fuselage area would be to provide space for the electronic devices and to hold the wing and tail parts firmly. The perfect shape where the internal volume is maximum with least drag is a body which has larger length than the width or height. From Multhopp's method for C_{m_0} and C_{m_α} contribution of fuselage, we get

$$C_{m_{0_f}} = \frac{k_2 - k_1}{36.55\bar{c}} \int_0^{l_f} w_f^{\,2}(\alpha_{0_w} + i_f)dx \qquad \text{(equa-4.5)[5]}$$

$$C_{m_{0_f}} = \frac{x_2 - x_1}{36.55\bar{c}} \int_0^{x=l_f} w_f^{\,2}(\alpha_{0_w} + i_f)\Delta x \qquad \text{(equa-4.6)[5]}$$

Where

$k_2 - k_1$ is the correction factor for the body fineness ratio

S is the wing reference area

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$\bar{c}$ is the wing mean aerodynamic chord

w_f is the average width of the fuselage sections

α_{0_w} is the wing zero lift angle relative to the fuselage reference line.

i_f is the angle of incidence of the fuselage camber line relative to the fuselage reference line at the center of each fuselage increments.

Note-The incidence angle is defined as negative for nosed droop and aft upsweep

Δx is the length of the fuselage increments.

$$C_{m_{\alpha_f}} = \frac{1}{36.55\bar{c}} \int_0^{l_f} {w_f}^2 \frac{d\varepsilon}{d\alpha} x dx$$ (equa-4.7)[5]

$$C_{m_{\alpha_f}} = \frac{1}{36.55\bar{c}} \sum_{x=0}^{x=l_f} {w_f}^2 \frac{d\varepsilon}{d\alpha} x \Delta x$$ (equa-4.7-a)[5]

Total contribution

$$C_{m_0} = C_{m_{0w}} + C_{m_{0_f}} + C_{m_{0_t}}$$ (equa-4.8)[5]

$$C_{m_\alpha} = C_{m_{\alpha w}} + C_{m_{\alpha_f}} + C_{m_{\alpha_t}}$$ (equa-4.9)[5]

If C_{m_0} is positive also C_{m_α} is negative the aircraft is said to be having longitudinal stability.

5. DIFFERENT PHASES OF FLIGHT

Hovering:

In hovering condition, the aircraft will be remaining in a same altitude. For this to be possible, the thrust generated by thrusters need to overcome the weight of the whole aircraft itself. In hovering condition, the aircraft is in one single position, so there is no any climbing velocity or forward velocity; hence the free stream velocity becomes zero. Also here drag is zero.

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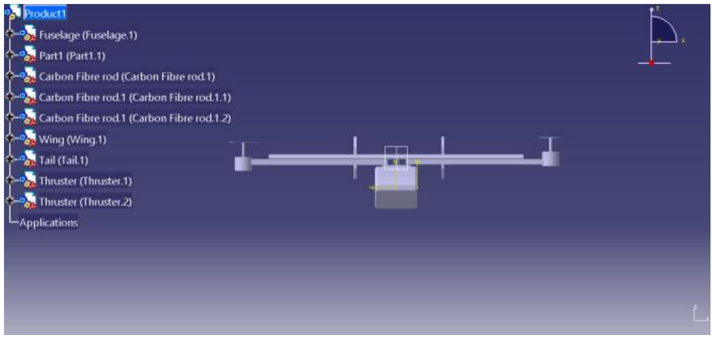
Fig 6: Hovering condition front view



$$T = 2\rho A\,(V + V_i)V_i \qquad \text{(equa-5.1)[6]}$$

$$V_h = \sqrt{\frac{W}{2\rho A}} \qquad \text{(equa-5.2)[6]}$$

$$\Omega = \frac{2\,\pi\,N}{60} \qquad \text{(equa-5.3)[6]}$$

$$V_h = \Omega\, R\sqrt{\frac{C_T}{2}} \qquad \text{(equa-5.4)[6]}$$

Thrust of a single motor and the propeller combination is given by

$$T = C_T\rho(\Omega R)^2 A \qquad \text{(equa-5.5)[6]}$$

Torque of a single motor and the propeller combination is given by

$$Q = C_Q\rho(\Omega R)^2 AR \qquad \text{(equa-5.6)[6]}$$

$$F.O.M = \frac{{C_T}^{3/2}}{\sqrt{2}\,C_Q} \qquad \text{(equa-5.7)[6]}$$

$$P = Q\Omega \qquad \text{(equa-5.8)[6]}$$

Where

T is the thrust generated by the thrusters

W is the weight of the aircraft

V_H is the hovering velocity

A is disc area of the thruster

ρ is density of air

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Ω is the angler velocity of the propellers

R is the radius of the propellers

C_t is the coefficient of thrust

C_q is the coefficient of torque

q is the torque produced due to the thrusters

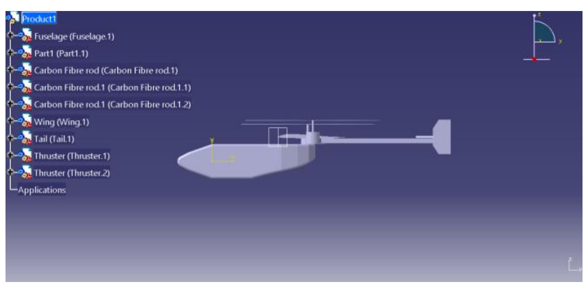
P is the total power required by the thrusters

N is the speed of the thrusters in RPM

Climbing

In climbing condition, the aircraft lifts of vertically from the ground or in the position it was before. In this condition the velocity at which the aircraft climbs called as climbing velocity. The thrusters here generate the required lift to move the aircraft in vertical direction by overcoming the drag produced due to the thrusters and aircraft as well as the weight of the whole aircraft itself/ in climbing condition, the aircraft will ascend vertically on the same axis about which the thrusters are rotating, so there will be no forward velocity but there will be climbing velocity, hence here free stream velocity is nothing but the climbing velocity. So there will be drag component as well.

Fig 7: Climbing condition side view



$$T = D+W \qquad \text{(equa-5.9)}^{[6]}$$

$$D = 0.5\,\rho\,(V + V_i)^2 A_B C_{DB} \qquad \text{(equa-5.10)}^{[6]}$$

$$\rho A(V + V_i)2\,V_i = 0.5\,\rho\,(V + V_i)^2 A_B C_{DB} + W \qquad \text{(equa-5.11)}^{[6]}$$

$$Q = 4A\,(V + V_i)\,\omega R^2 \qquad \text{(equa-5.12)}^{[6]}$$

$$Q\,(\Omega-\omega) = T(V + V_i) \qquad \text{(equa-5.13)}^{[6]}$$

$$P = Q\Omega \qquad \text{(equa-5.14)}^{[6]}$$

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Where

V_c is climbing velocity

D Drag generated

AB is the area of the propeller

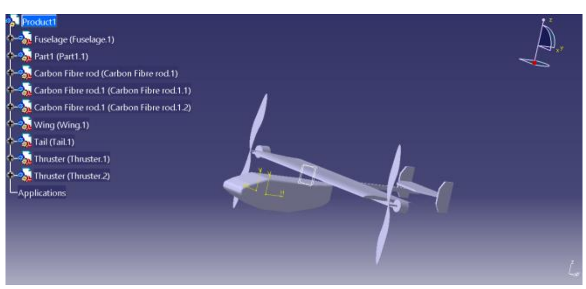
C_{D_B} is the drag coefficient due to the propeller

Forward

In forward condition, the thrusters are tilted slightly forward to make the thruster almost parallel to that of ground. In this condition, the thrust produces by the thruster is divided into two components by taking the angle of tilt as "δ" from the vertical axis. So the thrust can be resolved as $T\cos\delta$ and$T\sin\delta$, where $T\cos\delta$ is a component which gives extra lift apart from that of the wing and $T\sin\delta$ is the component which moves the aircraft forward. In this condition, the aircraft is not moving purely in forward motion but forward upward direction. But this extra lift is negligible compared to that of the one produced by the wing. In this condition, the free stream velocity is the forward velocity, but the velocity that is contributing to thrust and drag components is resultant velocity. This resultant velocity generally needs to be perpendicular to that of plain of rotation of propellers, but since the aircraft if moving forward, the forward velocity component tilts the resultant velocity component. So the drag acting which is aligned in the direction of the resultant velocity also gets tilted. The angle formed by the drag from the horizontal axis is "ε ". The drag can also be resolved into two components as $T\cos\varepsilon$ and $T\sin\varepsilon$. The Dcosepslon is the component that is acting in the direction of free stream velocity or forward velocity and $D\sin\varepsilon$ is the component that is acting along with the weight W component.

So the horizontal thrust component $T\sin\delta$ has to overcome the horizontal drag component$T\cos\varepsilon$, the vertical thrust component $T\cos\delta$ has to overcome the vertical component $T\sin\varepsilon$ as well as the weight component.

Fig 8: Forward condition



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D = 0.5 $\rho\, V_R{}^2 A_B C_{DB}$ (equa-5.15)[6]

$\tan\delta = \frac{D\cos c}{D\sin c + W}$ (equa-5.16)[6]

$T^2 = D^2 + W^2 + D\,W\sin\epsilon$ (equa-5.17)[6]

T = 2A $\rho V_i V_R$ (equa-5.18)[6]

Q = A $\rho V_R R^2\, \omega$ (equa-5.19)[6]

6. CALCULATIONS

Weight estimation:

Table 1: Weight Estimation of the UAV

2 × BLDC Motor = 320gm	Elevator = 100gm
2 × ESC = 80gm	Fuselage = 200gm
2 × Propellers = 40gm	3 × Servo motor = 60gm
1 Battery = 250gm	Receiver = 20gm
Carbon fiber tail boom = 300gm	Extras = 130gm
Total weight = 1.5 kg	

Airfoil Data:

The airfoil used in this UAV is Bell A821201 (23%) FX-66-H-60, because most of the thrust vector is going to be away from the chord line. This airfoil has flat bottom surface and streamlined upper surface which helps UAV to float stably irrespective of thrust direction.

Propeller Data:

Propeller: Radius 12.7 cm

Area = πr^2 = 0.0507 m^2
(equa-6.1)[6]

Different Contribution for stability analysis:

Wing contribution:

$C_{m_{ow}} = C_{m_{acw}} + C_{L_{ow}}\left(\frac{X_{cg}}{C} - \frac{X_{ac}}{C}\right) = 0.00575$ (equa-6.2) [5]

$C_{m_{\alpha w}} = C_{L_{\alpha w}}\left(\frac{X_{cg}}{C} - \frac{X_{ac}}{C}\right) = 0.000133$ (equa-6.3) [5]

Tail configuration:

$AR_w = \frac{b^2}{s} = 4$ (equa-6.4) [5]

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$\epsilon = 2\frac{C_{LW}}{\pi AR_W}$ = 0.1592 C_{L_W} (equa-6.5) [5]

Table 2: Downwash angle and C_L for corresponding AOA

α	C_L	ϵ
-5	-0.4	-0.0637
-3	-0.32	-0.0509
-1	-0.15	-0.0238
0	0.05	0.00796

$\eta = \frac{1}{cos\,\epsilon}$ = 1 (equa-6.6) [5]

$V_H = \frac{L_t \times S_t}{s \times \bar{c}}$ = 0.5 (equa-6.7) [5]

$\frac{d\,\epsilon}{d\alpha}$ = 0.006366 (equa-6.8) [5]

$C_{m_{0_t}} = \eta V_H C_{L_{\alpha_t}}(\varepsilon_0 + i_w - i_t)$ = 0.0001592 (equa-6.9) [5]

$C_{m_{\alpha_t}} = \eta V_H C_{L_{\alpha_t}}\left(1 - \frac{d\varepsilon}{d\alpha}\right)$ = -0.0019872 (equa-6.10) [5]

Fuselage Contribution:

$C_{m_{0_f}} = \frac{k_2 - k_1}{36.5\, s\bar{c}}\int_0^{l_f} w_f^{\,2}\left(\alpha_{0_w} + i_f\right)dx$ = 0 (equa-6.11) [5]

$C_{m_{\alpha_f}} = \frac{1}{36.5\, SC}\int_0^{L_t} w_f^{\,2}\frac{d\varepsilon_u}{d\alpha}$ dx = 0.000177 (equa-6.12)[5]

Different phases of flight:

Hovering:

Generally, the thrust produced by the motor and the propeller combination has to lift the total weight of the UAV, also need to overcome the drag produced due to rotor as well as the UAV itself.

Thrust (T) = Drag (D) + weight (W) (equa-6.13) [6]

But in hovering condition drag equals zero.

Therefore, T = W

Thrust required in hovering condition is given by

T= Weight (W) = 1.5kg = 14.7 N (equa-6.14) [6]

In hovering condition free stream velocity is zero.

Therefore, put V = 0 in the thrust equation obtained from actuator disc theory

From actuator disc theory, thrust is given by

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Vol: 41 Issue: 10-2022
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T = $2\rho A\ (V\ + V_i)V_i$ (equa-6.15)[6]

Since V becomes zero

T = $2\rho A\ (V_i)V_i$

$2\rho A{V_i}^2$ = T = W

Here V_i is nothing but the hovering velocityV_h

And it is given by

V_h = $\sqrt{\frac{W}{2\rho A}}$ (equa-6.16)[6]

For 1 motor + propeller, the hovering velocity is given by

V_h = $\sqrt{\frac{W}{2\rho A}}$ = 10.88 m/s

Where

ρ is the density of air

A is the propeller disc area

For 2 motors and 2 propellers, the hovering velocity is given by

V_h = $\sqrt{\frac{W}{2\rho A}}$ = 7.69 m/s

Angular velocity of the blade is given by

Ω = $\frac{2\ \pi\ N}{60}$ = 1278.62 rad/sec (equa-6.17) [6]

Where N is the speed of the propeller in R.P.M

V_h =Ω R$\sqrt{\frac{C_T}{2}}$ (equa-6.18) [6]

where R is the radius of the propeller disc

C_T is the Coefficient of thrust which is found out to be

$C_T = 0.004485$

Thrust of a single motor and the propeller combination is given by

T = $C_T\rho(\Omega R)^2A$ (equa-6.19)[6]

For 1 motor and propeller combination

T = 7.35 N

For 2 motors and propeller combination

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T = 14.7 N

Which comes out to be the total design weight.

Torque of a single motor and the propeller combination is given by

Q = $C_Q \rho (\Omega R)^2 AR$ (equa-6.20)[6]

Q = 0.062 Nm

F.O.M = $\frac{C_T^{3/2}}{\sqrt{2} C_Q}$ (equa-6.21)[6]

$C_Q = 0.0002985$

P = QΩ = 79.4 Nm/sec (equa-6.22) [6]

Climbing:

T = D + W (equa-6.23) [6]

D = 0.5 $\rho \, (V + V_i)^2 A_B C_{DB}$ (equa-6.24) [6]

A_B = 0.05502 m^2

C_{DB} = 0.8

Design velocity V = 20 m/s

T = D +W

$\rho A (V + V_i) 2 \, V_i = 0.5 \, \rho \, (V + V_i)^2 A_B C_{DB}$ + W

$V_i = 3.1455 \frac{m}{s}$ (equa-6.25)[6]

D = 0.5 $\rho \, (V + V_i)^2 A_B C_{DB}$ = 1.4443 N (equa-6.26) [6]

T = 2A $\rho \, (V + V_i)^2 V_i$ = 9.0434 N (equa-6.27) [6]

T = D + W

Therefore, the thrust required to climb is more than the weight.

Q = 4A $(V + V_i) \, \omega R^2$ (equa-6.28) [6]

Q (Ω-ω) = T$(V + V_i)$ (equa-6.29) [6]

Q = 0.0232 ω

Therefore

Q = 0.1646 Nm

P = QΩ = 210.461 Nm (equa-6.30) [6]

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Forward:

D = 0.5 $\rho V_R{}^2 A_B C_{DB}$ (equa-6.31) [6]

tan $\delta = \frac{D\cos\varepsilon}{D\sin\varepsilon + W}$ (equa-6.32)[6]

$\delta = 8.32$ (equa-6.33)[6]

$T^2 = D^2 + W^2$ + D W sin ϵ (equa-6.34)[6]

$T^2 = 55.4059$

T = 7.444 N

T = 2A $\rho V_i V_R$ (equa-6.35) [6]

Therefore $V_i = 2.996\frac{m}{s}$ (equa-6.36) [6]

Q = A $\rho V_R R^2$ ω = 0.1442 Nm (equa-6.37)[6]

P = 181.819 Nm (equa-6.38)[6]

7. RESULTS

As we defined earlier the aircraft is said to have longitudinal stability, if the value of C_{m_0} is positive as well as the value C_{m_α} is negative. We can see that wing alone contribution has positive cmalfa. Also the fuselage contribution has positiveC_{m_α}. But the tail contribution has larger negative C_{m_α} which pushes the overall C_{m_α} of the aircraft to negative. From that we can say that for an aircraft to be stable tail is very important. We even got the overall C_{m_0} of the aircraft to be positive. The values of C_{m_0} and C_{m_α} are found out to be

$$\boxed{C_{m_0} = 0.005909}$$

$$\boxed{C_{m_\alpha} = -0.0016772}$$

We have obtained necessary parameters in hovering conditions and are found out to be

$$\boxed{T = 14.7\ N}$$

$$\boxed{V_h = 10.88\ m/s}$$

$$\boxed{Q = 0.062\ Nm}$$

$$\boxed{P = 79.4\ Nm/s}$$

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We have obtained necessary parameters in climbing conditions and are found out to be

$$\boxed{V_i = 3.1455\frac{m}{s}}$$

$$\boxed{T = 18.086\ N}$$

$$\boxed{D = 2.8886\ N}$$

$$\boxed{Q = 0.1646\ Nm}$$

$$\boxed{P = 210.461\ Nm/s}$$

We have obtained necessary parameters in forward conditions and found out to be

$$\boxed{\delta = 8.32}$$

$$\boxed{T = 7.444\ N}$$

$$\boxed{V_i = 2.996\frac{m}{s}}$$

$$\boxed{Q = 0.1442\ Nm}$$

$$\boxed{P = 181.819\ Nm}$$

8. CONCLUSION

Since the overall C_{m_0} value of the aircraft is positive as well as the overall C_{m_α} value of the aircraft is negative we can conclude that the designed tilt rotor unmanned aerial vehicle will have a stable configuration during its flight.

From the necessary parameters obtained, we can get the thrust required and power required. Since the thrust required is less than the thrust generated, the designed tilt rotor unmanned aerial vehicle expected to perform the different phases of flight as defined. Also, the power required is less than the power generated, the aircraft will have a decent range and endurance which will help us to precede the research work further.

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