IJEAST

INTERNATIONAL JOURNAL
OF ENGINEERING APPLIED SCIENCE
AND TECHNOLOGY

VOLUME : 2 ISSUE : 4 Print / Issue Publication Date: 31-Mar-2017

ISSN : 2455-2143

Indexed In

Google SCHOLAR OPEN ACCESS doi Crossref

WWW.IJEAST.COM

editor@ijeast.com

International Journal of Engineering Applied Sciences and Technology, 2017
Vol. 2, Issue 4, ISSN No. 2455-2143, Pages 313-315
Published Online February-March 2017 in IJEAST (http://www.ijeast.com

STABILITY OF RC AIRCRAFT

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***Abstract*— In this paper my main emphasis is on stability of a given RC AIRCRAFT. I have calculated coefficient of moment, coefficient of lift, volume ratio of tail and some dimensions of given RC AIRCRAFT to check its stability.**

Keywords— **RC AIRCRAFT, STABILITY**

I. INTRODUCTION

RC AIRCRAFT are small model radio-controlled airplanes that fly using electric motor, gas powered IC engines or small model jet engines. The RC airplanes are flown remotely with the help of a transmitter with joysticks that can be used to fly the aircraft and perform different maneuvers. The transmitter communicate with receiver and sends signals to servo which moves the control surface based on the position of joysticks on the transmitter. The servo are small motors which are mechanically linked to the control surface e.g aileron for roll control, elevator for pitch control and rudder for yaw control. Most popular rechargeable batteries for RC AIRCRAFT use includes Ni-Cad and Li-Po. Lithium polymer lasts longer and more powerful than there Ni-Cad counterparts but a bit more expensive. The throttle of electric motor is controlled using a speed controller which comes with the motor. The speed controller lead is connected to the receiver.

RC AIRCRAFT are used for gathering weather Readings, experiments, aerodynamics modeling, testing and even used as drones or spy planes.

II. METHODS AND MATERIAL

In this project, I check the stability of given RC AIRCRAFT. Given Aircraft is electric powered Aircraft, having symmetric Airfoil.

Materials used in this RC AIRCRAFT :-

- COROPLAST
- THERMOCHOLE
- MOTOR
- Li-Po BATTERY
- ESC (ELECTRONIC SPEED CONTROLLER)
- SERVOS
- LINKAGES
- TRANSMITTER
- RECEIVER
- PROPELLER

TRANSMITTER

ESC

PROPELLER

MOTOR

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SERVO

RECEIVER

Fig . 2 (ELECTRONIC USED IN ELECTRIC POWERED RC AIRCRAFT)

III. RESULT AND DISCUSSION

To check the stability of an aircraft We have to calculate coefficient of moment, which is given by :-

$$C_{m\alpha} = C_{L\alpha 3D}\ (X_{cg} - X_{ac}) / C_w - V_n \times C_{L\alpha t} + (1 - d\mathcal{E}/d\alpha)$$

Where,

$C_{m\alpha}$ = Coefficient of moment at α angle of attack.

X_{cg} = Distance of center of gravity from leading edge of wing.

X_{ac} = Distance of aerodynamic center from leading edge of wing.

C_w = Wing chord.

V_h = Volume ratio of tail.

NOW, for a given aircraft,

- Mass of an aircraft (M) = 750 gm
- Wing loading = 0.53gm/cm^2
- Aspect ratio (ARw) of wing = 6
- $d\mathcal{E}/d\alpha$ = 0.5
- α = 3
- Density of air (ρ) = 1.225 × 10^{-3} gm/cm^3
- Acceleration due to gravity (g) = 981cm/sec^2
- Cruise speed (V) = 1400 cm/sec
- Tail area (S_t) = 15% of wing area
- Aspect ratio of tail (AR_t) = 4
- Tail setting angle (i_t) = 1.8 deg
- Distance of center of gravity from the aerodynamic center of wing (X) = 2 cm

Wing area (S_{wing}) = Mass of aircraft / Wing loading

= 750 / 0.53

$\boxed{= 1415.09\ cm^2}$

Wing Span (bw) = $\sqrt{ARw \times S_{wing}}$

= $\sqrt{6 \times 1415.09}$

$\boxed{= 92.14\ cm}$

- Wing chord (C_w) = S_{wing} / bw (for rectangular wing)

  = 1415.09 / 92.04

  = 15.358 cm

- $X_{ac} = C_w \times 25 / 100$

  = 15.358 × 25/ 100

  = 3.8395 cm

- $X_{cg} = X_{ac} + X$

  = 3.8395 + 2

  =5.8395 cm

- The lift coefficient is determined from equation the lift obtained from the wing to the aircraft wing i.g,

  L = W

SO, $0.5 \times \rho \times v^2 \times C_{L3D} \times S_{wing} = M \times g$

$0.5 \times 1.225 \times 10^{-3} \times (1400)^2 \times C_{L3D} \times 1415.09$

$= 750 \times 981$

$C_{L3D} = 0.433$

- For symmetric airfoil,

$C_{L3D} = C_{L\alpha\ 3D} \times \alpha$

$C_{L\alpha\ 3D} = C_{L3D} / \alpha = 0.433 / 3$

$= 0.144\ deg^{-1}$

- $C_{L\alpha\ 3D} = 8.2512\ rad^{-1}$ $(1\ deg^{-1} = 57.3\ rad^{-1})$

- $S_t = S_{wing} \times 15/100$

$= 1415.09 \times 15/100$

$= 212.26\ cm^2$

Now the tail moment arm (L_t) is calculated from the longitudinal moment equation.

- $0.5 \times \rho \times v^2 \times C_{L3D} \times S_{wing} \times X =$

$0.5 \times \rho \times V^2 \times C_{L\alpha\ t} \times (\alpha - i_t) \times St \times L_t$

- $C_{L3D} \times S_{wing} \times X = C_{L\alpha\ t} \times (\alpha - i_t) \times S_t \times L_t$

- $0.433 \times 1415.09 \times 2 = 2\pi \times (3\text{-}1.8) \times 212.26 \times L_t$

$L_t = 43.88$ cm

- Volume ratio (Vn) of tail is defined as,

$Vn = (L_t \times S_t) / (Swing \times Cw)$

$= (43.88 \times 212.26) / (1415.09 \times 15.3858)$

$= 0.43$

- Coefficient of moment at zero angle of attack

$= (Cm_0) = C_{L\alpha t} \times i_t \times V_n$

$= 2\pi \times 1.8 \times 0.43$

$= 0.085$

Hence,

$C_{m\alpha} = [\ C_{\alpha 3D}\ (X_{cg} - X_{ac}) / C_w\ ] - [V_n \times C_{L\alpha\ t} \times (1 - d\varepsilon/d\alpha\ )]$

$= [8.2512\ (5.8395 - 3.8395) / 15.358\ ] - [0.43 \times 2\pi \times (1 - 0.5)]$

$= -0.28$

- <u>CALCULATION TABLE</u>

Angle of attack (α)deg	Coefficient of moment (Cm)
2.8	-0.46
2.9	-0.35
3.0	-0.28
3.1	-0.2
3.2	-0.15
3.3	-0.1
3.4	0

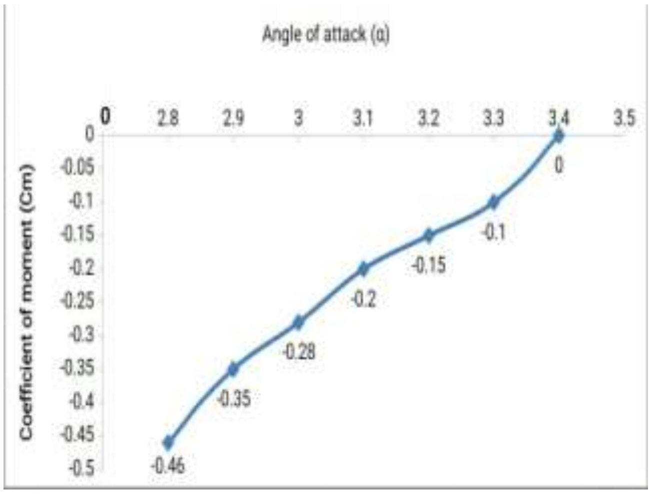

Fig . 3 (Graphical illustration of coefficient of moment vs angle of attack .)

IV. CONCULSION

For stability of an RC AIRCRAFT slope of Cm vs α , Cmα should be negative. For given aircraft Cmα = -0.28, which is negative and slop of Cm vs α is also negative as shown in fig 3. The coefficient of moment at zero angle of attack should be positive. For given aircraft Cm_0 = 0.085, which is positive. Hence Given aircraft is stable.

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