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DESIGN OF WING AND ENGINE SELECTION FOR A SINGLE SEATER HOME BUILT AIRCRAFT

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Abstract— **To design the wing dimensions by various calculation and historical data for the single seater home built aircraft. The engine must be selected such that the thrust required must be equal to the thrust produced by the engine for a single seater home built aircraft**

Keywords- wing design; Engine Selection; home built aircraft; C Software

1. INTRODUCTION

AIRCRAFT DESIGN

Three major types of airplane designs are

i). Conceptual design
ii). Preliminary design
iii). Detailed design

i). Conceptual design:

It depends on what are the major factors for designing the aircraft.

(a) Power plant Location :

The Power plant location is either padded (or) Buried type engines are more preferred. Rear location is preferred for low drag, reduced shock & to the whole thrust.

(b) Selection of Engine :

The engine should be selected according to the power required.

(c) Wing selection :

The selection of wing depends upon the selection of

(1) Low wing
(2) Mid wing
(3) High wing

ii). Preliminary design:

Preliminary is based on Loitering. 'U' is the mathematical method of skinning the aircraft, the aircraft look like a masked body.

Preliminary design is done with help of C SOFTWARE.

iii). Detailed design:

In the detailed design considers each & every rivets, bolts, paints etc. In this design the connection & allocations are made

2. PROCEDURE:

2.1 WING PARAMETERS:

To design the wing we have to find the length of fuselage,

So that, $\text{LENGTH} = aW_o^{\,C}$

From the historical data,

$$a = 3.68, c = 0.23$$

From the weight estimation, we have

$$W_o = 771.23\text{Kg}\left(1700.27\text{lbs}\right)$$

$$L = 3.68 \times 1700.27^{0.23}$$

$$L = 20.36\text{ft}$$

From the historical data,

Wing loading, $W/S = 70\text{Kg}/\text{m}^2$

$$S = \frac{W}{70}$$

$$S = \frac{771.23}{70} = 11.017\text{m}^2\left(118.58\text{ft}^2\right)$$

For our aircraft, aspect ratio, AR=6.

$$\frac{b^2}{S} = 6$$

$$b = 8.13m(26.67ft)$$

The root chord of the wing can be calculated from the equation,

$$C_{root} = \frac{2 \times S_w}{b(1+\lambda)}$$

$$C_{root} = \frac{2 \times 11.017}{8.13(1+0.5)}$$

$$C_{root} = 1.8068m$$

$$C_{tip} = \lambda C_{root}$$

$$C_{tip} = 0.9034m$$

To design the aerodynamic centre,

$$\overline{C} = \left(\frac{2}{3}\right) C_{root} \frac{1+\lambda+\lambda^2}{1+\lambda}$$

$$\overline{C} = \left(\frac{2}{3}\right) 1.8068 \frac{1+0.5+0.5^2}{1+0.5}$$

$$\overline{C} = 1.4052m$$

$$\overline{Y} = \left(\frac{b}{6}\right)\left(\frac{1+2\lambda}{1+\lambda}\right)$$

$$\overline{Y} = \left(\frac{8.13}{6}\right)\left(\frac{1+2\times 0.5}{1+0.5}\right)$$

$$\overline{Y} = 1.80667m$$

The vertical and horizontal surface of the tail,
The equation we have is,

$$C_{VT} = \frac{L_{VT} S_{VT}}{b_W \times S_W}$$

$$C_{HT} = \frac{L_{HT} S_{HT}}{C_W \times S_W}$$

From these equation,

$$S_{VT} = \frac{C_{VT} \times b_W \times S_W}{L_{VT}}$$

$$S_{HT} = \frac{C_{HT} \times C_W \times S_W}{L_{HT}}$$

From the historical data the value of C_{VT} and C_{HT} for the single seat home built aircraft is,

$$C_{VT} = 0.04; C_{HT} = 0.5.$$

The length of vertical tail is the 50-60% of the fuse length.

$$L_{VT} = 0.5 \times 20.1 = 10.1ft$$

$$S_{VT} = \frac{0.04 \times 26.67 \times 118.58}{10.1}$$

$$S_{VT} = 11.09ft^2$$

The span length of the horizontal tail is the 25% of the fuselage length,

$$b_W = 0.25 \times 20.1 = 4.75ft$$

$$S_{HT} = \frac{0.5 \times 4.7 \times 118.58}{10.14} = 24.44ft^2 \quad \textbf{2.2}$$

FUEL TANK:

The volume of the fuel tank is,
Volume of fuel=weight of fuel/density of fuel
The density of fuel is taken as 800Kg/m^3

$$V = \frac{209.07}{800}$$

$$V = 0.2612m^3$$

Now we assume 50% of the fuel stored in the wing,

$$0.5\times0.5\times0.2612 = \left(\left(\frac{t}{C}\times\bar{C}\times0.5\times\bar{C}\right)\times\left(0.5\times b\times0.75\times2\right)\right)$$

$$\frac{t}{C} = 8.97\times10^{-3}$$

To find root thickness,

$$\frac{t}{C_{root}} = 8.97\times10^{-3}$$

$$t = 0.01622m$$

To find tip thickness,

$$\frac{t}{C_{tip}} = 8.97\times10^{-3}$$

$$t = 0.008m$$

2.3 ENGINE CALCULATIONS:
To calculate the max thrust required for take-off,
We have,

$\frac{h_p}{W} = 0.8$ as reference for home built aircraft .

The thrust required by analytical approach,

$$\frac{T}{W} = \left(\frac{550\eta_{propeller}}{V}\right)\times\left(\frac{h_p}{W}\right)$$

$$\frac{T}{W} = \frac{550\times0.8}{83.33}\times0.08$$

$$\frac{T}{W} = 0.4224$$

$$T_{required} = 0.4224\times780.71$$

$$T_{required} = 329.77Kg\left(361.364h_p\right)$$

The engine must be selected such that the thrust required must equal thrust produced by the engine.
The following engine matches the purpose,

1) 1xLycoming produces a thrust of 396KW.
2) 1xLycoming produces a thrust of 395KW.
3) 1xVedeneyer produces a thrust of 420KW.

The wet aspect ratio can be calculated using the formula,

$$AR_{wet} = \frac{AR}{\left(\frac{S_{wet}}{S_{rep}}\right)}$$

$$AR_{wet} = \frac{6}{5}$$

$$AR_{wet} = 1.2$$

2.4 THRUST MATCHING:

For a propeller aircraft the required take-off $\left(\frac{h_p}{W}\right)$ can be found using,

$$\left(\frac{h_p}{W}\right)_{take-off} = \left(\frac{V_{cruise}}{550\eta_{propeller}}\right)\times\left(\frac{1}{\left(\frac{L}{D}\right)_{cruise}}\right)\times\left(\frac{W_{cruise}}{W_{take-off}}\right)\times\left(\frac{h_{p_{take-off}}}{h_{p_{cruise}}}\right)$$

$$\left(\frac{L}{D}\right)_{cruise} = 11$$

$$\left(\frac{W_{cruise}}{W_{take-off}}\right) = W_{climb}\times W_{cruise}$$

$$\left(\frac{W_{cruise}}{W_{take-off}}\right) = \frac{W_2}{W_1} \times \frac{W_3}{W_2} = \frac{W_3}{W_1} = 0.7998$$

Piston engines with supercharges cruise at 75% of take-off power,

$$\left(\frac{h_{p_{take-off}}}{h_{p_{cruise}}}\right) = \frac{1}{0.75} = 1.33$$

$$\left(\frac{h_p}{W}\right)_{take-off} = \frac{300}{550 \times 0.8} \times \frac{1}{11} \times 0.799 \times 1.33 = 0.0658$$

$$\left(\frac{h_p}{W}\right)_{take-off} = 0.0658$$

Since, the reference $\left(\frac{h_p}{W}\right)$ and calculated $\left(\frac{h_p}{W}\right)$ are, the assumption is correct.

3 ADVANTAGES OF Lycoming 250-B17 Engine**:**

1. Engine is in free air stream.
2. Mass flow rate is high.
3. C_g location is front portion of fuselage.

3.1 DISADVANTAGES OF Lycoming 250-B17 Engine**:**
Stability is very poor.

ENGINE CONFIGURATION:

MODEL	TYPE	MAXIMUM POWER AT SEA LEVEL (SHP)	SPECIFIC FUEL CONSUMPTION AT MAXIMUM POWER	OVERALL PRESSURE RATIO AT MAX POWER
OPVS P.I. 380	C-P	305	0.750	6
250-B17C	AC-P	420	0.66	7.2
250-B17E	AC-P	420	0.66	7.2
250-B17F	AC-P	450	0.61	7.9

ENGINE DIMENSION:

MODEL	MAXIMUM ENVELOPE DIAMETER (in)	MAXIMUM ENVELOPE LENGTH (in)	DRY WEIGHT LESS TAIL PIPE (lb)
OPVS P.I. 380	22.4	41.8	TBA
250-B17C	22.6	45	198
250-B17E	22.5	45	202
250-B17F	22.6	45	205

4 CONCLUSIONS:

Thus, by the above calculation we have select the Lycoming 250-B17 for the single seater home built aircraft. It is located in the nose of the aircraft for free air stream and also we had determined the wing dimensions and fuel tank size in wings of the single seat home built aircraft and based on the design we have selected low wing.

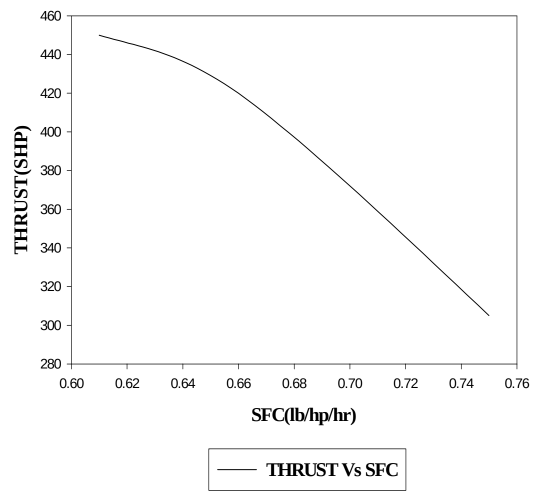

Figure – 1 Thrust Vs SFC

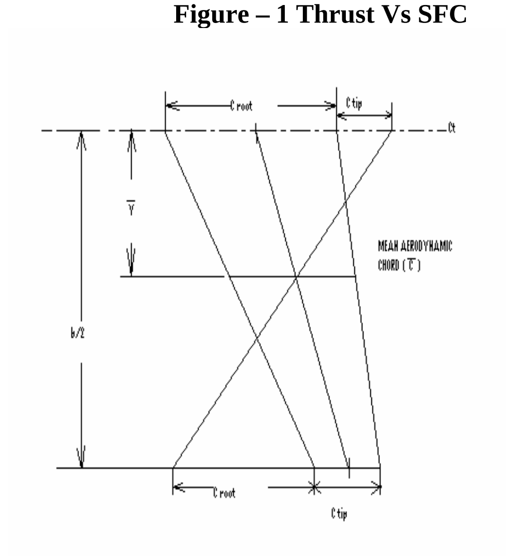

Figure – 2 WING

SYMBOLS USED

W-Weight of aircraft
W_o-Overall weight
W_f-weight of fuel
W_e-Empty weight

L_f – fuselage length
D_f – diameter of fuselage
S_w - wing area
T_w - wing thickness
b_w,b – wing span
S_{ht} – horizontal tail area
t_{ht} – horizontal tail thickness
b_{ht} - horizontal tail span
AR – aspect ratio
S – Surface area
S_{vt} – vertical tail area
t_{vt} - vertical tail thickness
b_{vt} – vertical tail span
Cd – coefficient of drag
C_L - coefficient of lift
F, T – thrust
T/W-Thrust loading
W/S-Wing loading
A.R-Aspect ratio
C_r,C_t-Chord length of root,tip
T_r,T_t-Thickness of root,tip
$V\infty$ -Free stream velocity
C-Chord
Lf-Length of fuselage
VT-Vertical tail
HT-Horizontal tail
W_o – optimum weight
C_r – root chord

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Dr. M.Venkatesan received the Ph. D Award from the International University of Contemporary Studies, Washington DC in 2009, Masters in Thermal Engineering (2001) and Bachelor Degree in Mechanical Engineering (1997) from University of Madras. He is currently the Vice Principal (Academics), Professor and Head of Aeronautical Engineering Department in PMR Engineering College, Chennai, TamilNadu, India. He has more than 13+ years of experience in Teaching, Research and Administration at National and International Level. His fields of interests are various, viz., Alternative fuels, Heat Transfer, Aeronautics, Design, and Supply chain Management. He has more than 10 publications to his credit both in National and International Journals and conferences and has authored 5 books on Engineering viz., Engineering Mechanics, Aero Engineering Thermodynamics, Fluid Mechanics and Fluid Machinery, Engineering Graphics and Workshop Practice as per Anna University Chennai regulation. He has dedicated his whole soul and life to research and education and he has been serving as Editorial Board Member, Advisory Board Member and Editor-in-Chief for International Journals.