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Design and Fabrication of UAV Used for Medical Emergency

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ABSTRACT: It is an unmanned aerial vehicle that would be able to deliver blood in hard to reach places in a short amount of time. The design would a great help to society as large since such an act would be life saving.

KEYWORDS: Unmanned Aerial Vehicle.

INTRODUCTION

For a fixed - wing UAV, the preliminary design phase is performed in two steps:

- 1) Estimate aircraft maximum take-off weight
- 2) Determine wing area and engine thrust (or power) simultaneously

We would be first estimating maximum take-off weight and then determine the wing area and engine thrust.

The maximum take-off weight (WTO) is broken into four elements:

- 1) Payload weight (WPL).
- 2) Autopilot weight (WA).
- 3) Fuel weight (WF) if non - electric engine, or Battery weight (WB), if electric engine.
- 4) Empty weight (WE).

$$WTO = WPL + WA + WF + WE$$

The above equation could be further modified to be In order to find WTO, one needs to determine four variables of WPL, WA, WB/WTO, and WE/WTO. The first three parameters, namely payload, autopilot, and fuel weight fraction are determined fairly accurately, but the last parameter (i.e., empty weight fraction) is estimated from statistics.

I. DESIGNING PARAMETERS

[1]The payload (useful load) is the net carrying capacity of an UAV which is originally required and designed to carry. The payload includes sensors (e.g., camera and radar), store, military equipment, and other intended loads. The payload weight is a known value, since it is selected by the designer, or given by the customer. For our mission we need to carry blood or other emergency medical supplies we choose our payload weight to be 1500 grams. We initially target to carry three pints of blood at a time. Since each pint of blood carry 525 ml of blood and weight about 500 grams, our payload weight is quite appropriate.

$$WPL = 1500\text{Grams}$$

[2] Another part of the aircraft weight is the weight of the autopilot which is responsible to conduct the flight operations and serving subsystems and payloads. If the UAV is controlled by an operator from the ground control station, a communication system including receiver, transmitter, and data link is required too. In terms of weight build up, we do not consider the communication system in the autopilot group. The weight of autopilot is a function of the UAV mission. If a commercial off - the - shelf (COTS) autopilot is selected, the weight will be a known value.



However, if the designer would like to design a custom - made autopilot, the weight will be determined when the design is done. For such conditions, statistical values for the weight of autopilot for a similar UAV Since weight for such autopilot system is usually between 20 to 70 grams, we choose to keep it 70 grams for the worst case scenario.

$$WA = 70\text{Grams}$$

[3] The first step to determine the total fuel weight is to define the flight mission segments. The mission profile is demonstrated in Figure 1.4 for our project. The flight operation is composed of taxi/take-off, climb, cruise, drop the payload, return back, descent, and landing. Each mission segment is numbered; for instance, 1 denotes the beginning of take-off and 2 is the end of take-off. The three most fuel/energy consuming are: Cruise, loiter, and cruise back. In a similar fashion, the aircraft weight at each phase of the flight mission is numbered.

$$W_{TO} - W_L = W_F$$

Thus, in a regular flight mission, the ratio between the aircraft weight at the end of flight to the aircraft weight at the beginning of flight is:

$$\frac{W_L}{W_{TO}} = \frac{W_{TO} - W_F}{W_{TO}}$$

The fuel weight fraction is obtained as follows:

$$\frac{W_F}{W_{TO}} = 1 - \frac{W_8}{W_1}$$

$$\frac{W_8}{W_1} = \frac{W_2}{W_1} \frac{W_3}{W_2} \frac{W_4}{W_3} \frac{W_5}{W_4} \frac{W_6}{W_5} \frac{W_7}{W_6} \frac{W_8}{W_7}$$

Therefore, in order to find the fuel weight fraction, one must first determine these weight fractions for all of the mission segments. There are primarily seven flight segments as take-off, climb, cruise, drop payload, cruise back, descent, and landing. These flight phases or segments can be divided into two groups:

Firstly, there are certain segments during which the fuel weight that is burnt is almost nothing and negligible compared with maximum take-off weight. These include, taxi, take-off, climb, descent, approach, and landing. The fuel weight fractions for these mission segments are estimated based on the statistics. illustrates typical average values for fuel fractions of take-off, climb, descent, and landing.

Secondly there are segments during which the fuel weight that is burnt is considerable. These segments are – 1. cruise to target area, 2. Drop Payload, and 3. cruise back to the base

(or operator) – are determined through mathematical calculations. The fuel weight fraction for cruise segment is determined by employing the Breguet range equations.

$$\frac{W_{i+1}}{W_i} = e^{\frac{-R \cdot c}{V \cdot (L/D)}}$$

Thus for calculating empty weight fraction we have written a python code which would take all inputs and output empty weight fraction. Python code is attached at last for reference.



We selected the following parameters for calculating empty weight fraction:

Sr No.	Mission Segment	Wi+1/Wi
1	Taxi and Take-off	0.99
2	Climb	0.97
3	Descent	0.99
4	Approach and landing	0.997
5	Payload drop	0.95

For cruise segment we selected the following parameters

Sr No.	Particular	Quantity	Units
1	Range	120	km
2	Specific fuel consumption (c)	0.59	lb/(hp.h)
3	Propeller Efficiency	0.8	-
4	Maximum (L/D) Ratio	13	-

Thus fuel weight fraction is

$$\frac{W_F}{W_{TO}} = 0.1174$$

The last term in determining maximum take-off weight is the empty weight fraction (WE/WTO). The empty weight mainly includes weights of structure/airframe, engine(s), landing gear, and systems. At this moment (preliminary design phase), the aircraft has been designed only conceptually, hence, there is no geometry or sizing. Therefore, the empty weight fraction cannot be calculated analytically. The only way is to employ the past history and statistics. The only known information about the aircraft is the configuration and aircraft type based on the mission. According to this data, equation from Design of Unmanned Aerial Vehicle book to determine the empty weight fraction. The equations are based on the published data taken from published sources. In general, the empty weight fraction varies from 0.5 to 0.9. The empty weight fraction is mathematically modelled by the following experimental linear equation:

$$\frac{W_E}{W_{TO}} = aW_{To} + b$$

where a and b are presented in Table . Note that the Eq. is curve fitted in the British unit system. Thus, the unit for maximum takeoff weight and empty weight is lb. Note that the unit of WTO in Table is lb.

No	Unmanned Aerial Vehicle (UAV) type	a	b
1	Small remote controlled (RC)	-0.00296	0.87
2	HALE	1.07×10^{-5}	0.126
3	MALE	5.1×10^{-6}	0.42
4	Quadcopter	-4.6×10^{-6}	0.68



$$\frac{W_E}{W_{TO}} = -0.00296W_{TO} + 0.87$$

After calculating fuel weight fraction and empty weight fraction and substituting it in equation 1.2 we can get maximum take-off value. A python code for same was written and is referenced at last

$$W_{TO} = 14.54 \text{ kg}$$

II. WING AND ENGINE SIZING

The second crucial step in the aircraft preliminary design phase is to determine wing reference area (S_{ref}) and engine thrust (T), or engine power (P). Unlike the first step in preliminary design phase at which the main reference was statistics, this phase is solely depending upon the aircraft performance requirements and employs flight mechanics theories.

Both wing and engine have significant roles in a cruising flight. Four main UAV forces (lift, L; drag, D; weight, W; and thrust, T) are in equilibrium. In a cruising flight, lift is equal to UAV weight (W), and engine thrust (T) is equal to UAV drag (D).

$$L = W, D = T$$

In these two equations, the contribution of thrust components due to angle of attack, and engine setting angle are neglected. Equations indicate that the weight is balanced by lift, while the drag is balanced by engine thrust. The weight is a function of UAV mass:

$$W = mg$$

The aircraft performance requirements that are utilized to size the aircraft in this phase are:

- (a) maximum speed (V_{max});
- (b) maximum rate of climb (ROC $_{max}$);
- (c) take-off run (STO);
- (d) ceiling (h_c).

These values should be known or inferred from competitor study.

The Matching plot technique is performed in six steps:

- 1) Derive one equation for each aircraft performance requirement (W/P or T/W as a function of W/S).
- 2) Sketch all derived equations in one plot.
- 3) Identify the acceptable region inside the regions that are produced by the axes and the graphs. Determine the design point (i.e., the optimum selection).
- 4) From the design point, obtain two numbers; corresponding wing loading; (W/S) $_d$ and corresponding power loading; (W/P) $_d$.
- 5) Calculate the wing area and engine thrust/power from these two values, since the aircraft maximum take-off weight (W $_{TO}$) has been previously determined.

The following is the list of equations for each performance requirement –

$$\left(\frac{W}{S}\right)_{V_s} = \frac{1}{2} \rho V_s^2 C_{L_{max}}$$



$$\left(\frac{W}{P}\right)_{V_{max}} = \frac{\mu_p}{\frac{1}{2}\rho_0 V_{max}^3 C_{D0} \left(\frac{W}{S}\right) + \frac{2K}{\rho\sigma V_{max}} \left(\frac{W}{S}\right)}$$

$$\left(\frac{W}{P}\right)_{S_{T0}} = \frac{1 - \exp\left(0.6\rho g C_{Dg} S_{T0} \frac{1}{W/S}\right)}{\mu - \left(\mu + \frac{C_{Dg}}{C_{LR}}\right) \left[\exp\left(0.6\rho g C_{Dg} S_{T0} \frac{1}{W/S}\right)\right]} \frac{\mu_p}{V_{T0}}$$

Where,

$$C_{Dg} = (C_{DT0} - \mu C_{LT0})$$

$$\left(\frac{W}{P}\right)_{ROC} = \frac{1}{\frac{ROC}{\mu_p} + \sqrt{\frac{2}{\rho\sqrt{\frac{3C_{D0}}{K}}}} \left(\frac{W}{S}\right) \left(\frac{1.155}{(L/D)_{max}\mu_p}\right)}$$

$$\left(\frac{W}{P}\right)_{c} = \frac{\sigma_c}{\frac{ROC_c}{\mu_p} + \sqrt{\frac{2}{\rho_c\sqrt{\frac{3C_{D0}}{K}}}} \left(\frac{W}{S}\right) \left(\frac{1.155}{(L/D)_{max}\mu_p}\right)}$$

$$\left(\frac{W}{P}\right)_{AC} = \frac{\sigma_{AC}}{\sqrt{\frac{2}{\rho_{AC}\sqrt{\frac{3C_{D0}}{K}}}} \left(\frac{W}{S}\right) \left(\frac{1.155}{(L/D)_{max}\mu_p}\right)}$$

In the above equations, the new parameters are: K, wing induced drag factor, ηP, propeller efficiency, CDG, UA V drag coefficient at the ground, CLTO, UA V drag coefficient at the ground, μ, runway friction coefficient, σ, air density ratio, and (L/D)max denotes the maximum lift to drag ratio. From the matching plot, the design point is selected, by which two numbers are extracted. The output of the plot would be (W/S)d and (W/P)d for a prop- driven UA V. Then, the wing area and engine power/thrust will be calculated as follows

:

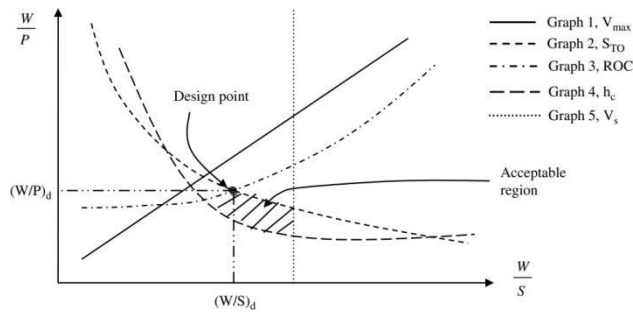
$$S = \frac{W_{T0}}{(W/S)_d}$$

$$P = \frac{W_{T0}}{(W/P)_d}$$

$$T = \left(\frac{T}{W}\right)_d . W_{T0}$$



Therefore, the wing area and engine power/thrust will be readily calculated. The matching plot for our design is presented below –



The values of wing loading and power loading we decided after studying matching plots are –

Wing Loading	18 Kg/m ²
Power Loading	0.59 Kg/KW

III. CONCLUSION

By keeping track of all this parameters we are going to get the most important and needed designing parameters such has:

Sr No.	Particular	Units
1	Maximum Takeoff Weight	14.54 kg
2	Wing Area	0.80 m ²
3	Engine Power	24.64 KW

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