



Preliminary Engine Sizing for a Tactical Unmanned Aerial Vehicle

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ABSTRACT

An Aircraft engine represents a concentration of weight, inertia, and gyroscopic forces that require accurate estimation, selection and integration for a safe and efficient flight operation. This is more critical in tactical unmanned aerial vehicles and very small aircraft due to the severe effects of aerodynamic forces on their weight class. Therefore, this article describes engine sizing for a Medium Altitude, Long Endurance (MALE) Unmanned Aerial Vehicle (UAV) using the matching plot technique. A conceptual framework of a conceptual UAV was modeled in an open Vehicle Sketch Pad (VSP) and the wetted area of major airframe components was measured. The consideration for drag estimation based on the wetted area includes the fuselage, wing, vertical/horizontal tail and landing gear. Each of these components was separately assessed for its contribution to the overall drag estimate and consequently aircraft performance. Thus, the Matching Plot Technique based on the aircraft performance was used in sizing the engine. This technique employs the use of flight mechanics theories to evaluate aircraft performance requirements which includes stall speed (V_s), maximum speed (V_{max}), takeoff run distance (STO), maximum rate of climb (ROC_{max}), and service ceiling (h_c). The variation of power loading (W/P) with respect to wing loading (W/S), for all the parameters, was plotted on a single grid. The acceptable regions and design points were determined based on the desired performance criteria. This solution helps in ensuring that large power output is obtained per unit engine size and weight. It also ensures a balanced distribution of aerodynamic forces.

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INTRODUCTION

An aircraft requires power to fly and that power comes from the engine. The size and type of engine is determined by the aircraft type, size, characteristics, intended mission, method of launching and recovery, and the type of fuel it will use. Furthermore, engine weight, inertia, and gyroscopic effect of its operation on the flight require additional structural weight in an aircraft to balance their sum effects. The objective of engine sizing therefore is to ensure large power output per unit of engine size and weight as well as

balanced distribution of aerodynamic and gyroscopic forces during flight [1],[2].

There are basically three methods of engine sizing. These are the Computational Fluids Dynamics (CFD), statistical and the empirical correlation methods. It is a known fact that CFD codes and wind tunnel testing could be quite expensive in terms of resources, skills required and time due to their complexities. On the other extreme, the statistical method is mostly used for conceptual design and home built models. This work therefore adopts the empirical

correlation method using matching plot technique [2],[3],[4]. This is a medium fidelity method based on drag polar and flight mechanics theories for preliminary engine sizing for medium size UAVs and very light aircraft. It therefore serves the intended purpose of sizing an appropriate engine for tactical UAVs.

CLASSIFICATION OF UAVS

UAVs are classified based on mass, vehicle configuration, mission, autonomy, service ceiling, endurance, etc. [1], [3]. In UAV systems, the parameters that present risk to the general public are the mass, size, energy and operational capabilities of the vehicle [3]. Hence, the most predominant classification method is that which

categorizes them as Micro, Mini, Tactical, MALE and HALE classes. Weibel and Hansman, Jr [6] distinctively classified UAVs according to their specifications, performance and mission capabilities. This was developed further with the inclusion of more recent tactical UAVs often used for policing and surveillance missions as well as military operations. UAVs classification strategy is an evolving process. Thus, a good understanding of the vehicle characteristics, configuration, mission and operating environment is an important requirement for sizing of any UAV system [2]. Further to [6], current trend in UAVs based on maximum takeoff weight and maximum operating altitude is shown in Fig. 1.

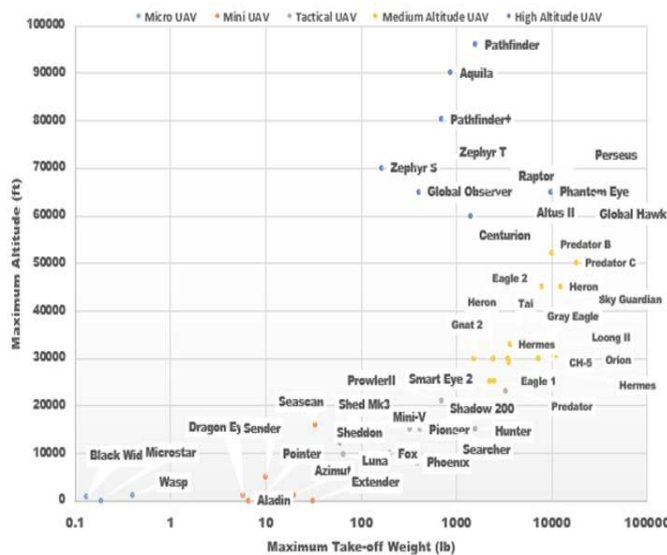


Fig 1. Current Trend in UAVs based on MTOW vs Maximum Altitude (modified from [6]).

POWER GENERATION METHODS USED ON CURRENT UAVS

UAVs can be powered by a piston or turbine engine, electric, glow or solar [3],[4]. An analysis of data obtained on 219 current UAVs, suggests that use of piston internal combustion engine is currently the most predominant power generation method among UAVs [5]. About 60% of the UAVs studied utilize piston engine for thrust

generation as shown in Fig. 2.

Piston Engines

The piston engines used on UAVs fall under four main categories. These are the 2-stroke, 4-stroke, stepped and rotary engines. Most of the UAVs using piston engines use either 2-stroke or 4-stroke type. The advantages of 2 stroke engines are that they have excellent power to weight ratio and are relatively inexpensive.

However, they are not fuel efficient and exhibit high levels of vibration and noise. On the other hand, the four-stroke engine has the advantage of higher torque over the 2 stroke engine. It also has the advantage of low noise and low vibration. Its downside is that it is complicated and difficult to maintain [4],[8].

Stepped Engines

Stepped Piston is a new development of the 2 and 4 stroke engines with an arrangement that gives better power loading, torque smoothness and fuel efficiency. Models have been built and tested but not yet in operation. The rotary (also known as Wankel) engine is that which converts pressure into rotating motion instead of reciprocating pistons. Rotary engines do not have reciprocating parts therefore deliver power very smoothly. The main disadvantages of the rotary concept are high sfc, and operating cost. Another important issue is that the range of power rating available are limited to about 28 kW or above 60 kW for aircraft application. It is therefore necessary to understand the advantages and

disadvantages of the different types of engines for good integration with the airframe and other aircraft systems. [9], [10]

ENGINE SIZING METHODOLOGY

The three most fundamental parameters in aircraft design are Maximum Takeoff Weight (MTOW), wing reference area (Sref) and engine thrust or power [2],[8]. The MTOW is essentially determined statistically whereas the engine power requirement depends on aircraft performance. The reference area could be any area including tail area, wing area and fuselage cross sectional area or fuselage surface area [11].

4.1 Theoretical Principles

Aircraft performance is a major requirement in aircraft/systems design process. This is because it helps in finding the minimum thrust, power requirements and the wing area – all of which are important preliminary design requirements. Consequently, the engine power was carefully analyzed based on the performance requirements using established methods [2], [9], [10],[16].

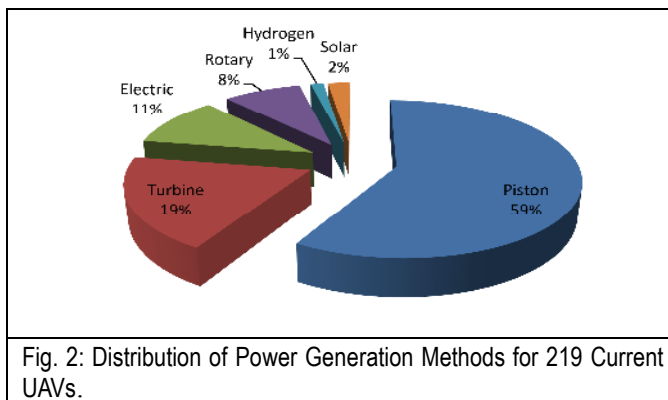


Fig. 2: Distribution of Power Generation Methods for 219 Current UAVs.

Drag Coefficients Calculations

For simplicity of calculation, it is assumed that the drag is a function of the reference area as recommended in [2]. In this case, the top-view area of the wing represents the reference area. Theoretically, the relationship between the reference and the total drag coefficient (D) is as shown below:

$$D = \frac{1}{2} \rho V^2 S C_D \quad (1)$$

The UAV was modeled in VSP and the wetted area of the major airframe components was measured. The consideration for drag estimation based on the wetted area includes the fuselage, wing, vertical/horizontal tail and landing gear. The drag coefficient has two components: zero-lift drag coefficient (C_{D0}) and lift-related drag coefficient or



induced drag coefficient (C_{Di}). The C_{D0} is the sum of the individual components drag coefficients as represented in Equation (2) whereas, the drag coefficient of each component was individually evaluated for its contribution to the overall drag estimate as shown in Equations (3-9).

$$C_{D0} = C_{D0f} + C_{D0w} + C_{D0LG} + C_{D0vt} + C_{D0ht} \quad (2)$$

and C_{Di} is given by:

$$C_{Di} = KC_L^2 \quad (3)$$

$$C_{D0f} = C_f f_{LD} f_M \frac{S_{wet}}{S} \quad (4)$$

$$C_{D0w} = C_{fw} f_{tcw} f_M \left(\frac{S_{wet}}{S}\right) \left(\frac{C_{dminw}}{0.004}\right)^{0.4} \quad (5)$$

$$C_{D0LG} = \sum C_{DLG} \left(\frac{S_{LG}}{S}\right) \quad (6)$$

$$C_{D0LG} = 0.0007 \quad (7)$$

The frontal area of each wheel of the landing gear is calculated from $S_{LG} = d_g \times w_g$ [6].

$$C_{D0vt} = C_{fvt} f_{tcvt} f_M \left(\frac{S_{wet}}{S}\right) \left(\frac{C_{dminvt}}{0.004}\right)^{0.4} \quad (8)$$

$$C_{D0ht} = C_{fht} f_{tcht} f_M \left(\frac{S_{wet}}{S}\right) \left(\frac{C_{dminht}}{0.004}\right)^{0.4} \quad (9)$$

But, the drag polar for an aircraft is given by:

$$C_D = C_{D0} + KC_L^2 \quad (10)$$

where the form factor (K) is a function of the aspect ratio (AR).

$$K = \frac{1}{\pi e AR} \quad (11)$$

The Oswald's efficiency factor (e) for a rectangular wing with no sweep is given by:

$$e = 1.78(1 - 0.045AR^{0.68}) - 0.64 \quad (12)$$

Sadraey [2] recommends that factors such as interference drag, surface roughness and antennae be accounted for using a factor Kc. For aircraft in GULMA's category, the Kc equals 1.2. Based on the above method, the derived drag polar is:

$$C_D = 0.0242 + 0.0364C_L^2 \quad (13)$$

But the required thrust (T_{req}) for level flight is equal to the total drag. Thus,

$$T_{req} = D \quad (14)$$

Therefore, total estimated (P_{req}) is calculated from the following relationship [10].

$$P_{req} = T_{req} \cdot V \cdot (\eta_p) \cdot (\eta_e) \quad (15)$$

The efficiency of a piston engine depends directly on its compression ratio (usually 10:1), whereas the combustion product (γ) of the fuel/air mixture is usually taken as 1.3. These values were used in the Equation 16 to arrive at a reasonable engine efficiency factor (η_e) [2].

$$\eta_e = 1 - \left(\frac{1}{r^{(\gamma)}}\right) \quad (16)$$

Thus, the required power is given by:

$$P_{req} = T_{req} V_{max} \cdot (\eta_p) \cdot (\eta_e) \quad (17)$$

However, engine size is usually determined by either the take-off performance or climb to the initial cruise altitude. Therefore, a detailed aircraft performance analysis was carried out to ensure that initially sized engine can meet the required performance at top of climb and other critical conditions. This analysis was carried out using matching plot technique introduced earlier. In the matching plot technique, C_{D0} for the aircraft under preliminary design can also be determined by taking average C_{D0} of several aircraft with similar performance characteristics and similar configuration.

Estimation of Wetted Area using Vehicle Sketch Pad

Estimating the surface area of an aircraft can be accomplished using traditional CAD models. However, such tools do not give the actual wetted area of an aircraft without interfacing with some CFD codes. It is for this purpose that the VSP was employed in this case. The VSP is an open source parametric geometry modeling tool developed by NASA [12]. The tool is especially designed to be used during the conceptual stage of aircraft design. The VSP can be used to model different aircraft configurations without the needs for the high expertise required for traditional CAD packages. The software

utilises parameters such as the wing span, area, sweep, taper ratio and thickness to cord ratio.

Hence, the GULMA UAV was modeled on VSP using the aircraft geometric and performance data. Since the wetted area of the landing gear components is not required for the drag polar analysis, the landing gear was not modeled on the VSP. The profile of the fuselage was modeled as closely to the actual model using multiple sections of varying thicknesses though as shown in Fig. 3. This method allows the computation of geometric parameters such as theoretical and wetted area/volume of both the individual surfaces and the entire aircraft.

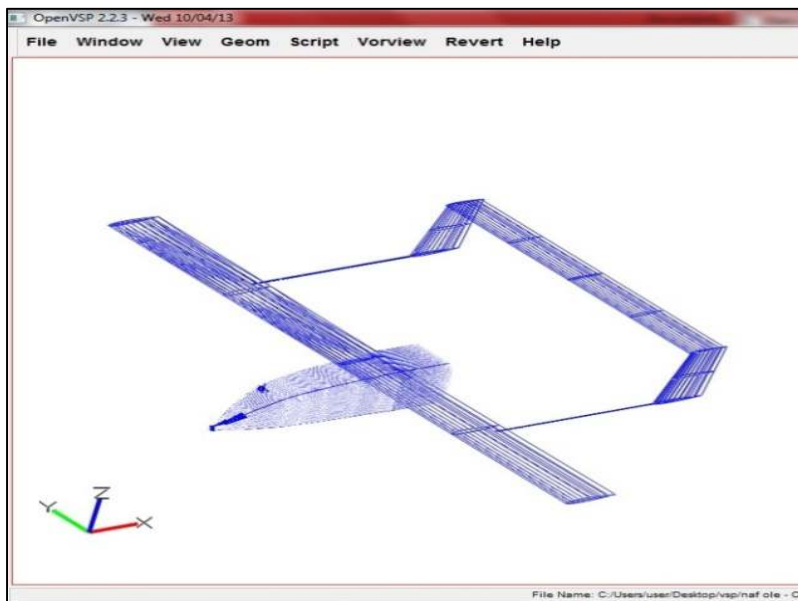


Fig. 3: Screenshot of GULMA UAV Modeled on VSP

Thrust Required and Thrust Available Curves

The maximum speed (V_{max}) is the speed at which the available thrust (T_A) equals the drag. Thus,

$$T_A = D = D_i + D_p \quad (18)$$

But the engine thrust is a function of the engine power, propeller efficiency (η_p), and aircraft speed. This implies that,

$$T_{max} = \frac{\eta_p P_{max}}{V_{max}} \quad (19)$$

where

$$\frac{\eta_p P_A}{V_{max}} = \left(\frac{W}{S}\right)^2 \frac{S}{\pi e A R q} + S D_0 q \quad (20)$$

$$q = \frac{1}{2} \rho V^2$$

Fig. 4 refers, the intersections of the total thrust and thrust required with the total drag lines give the maximum speed and design cruise

speed, respectively [10]. The above result was checked using CS-VLA 335 [13] design speed requirement which stipulates that the design cruise speed may not be less than:

$$V_c = 2.4 \sqrt{\frac{Mg}{S}} \quad (21)$$

and the design dive speed (VD) may not be less than $1.25V_c$ (CS-VLA335.(b).(2)). [13]

$$V_{max} = 1.25V_c \quad (22)$$

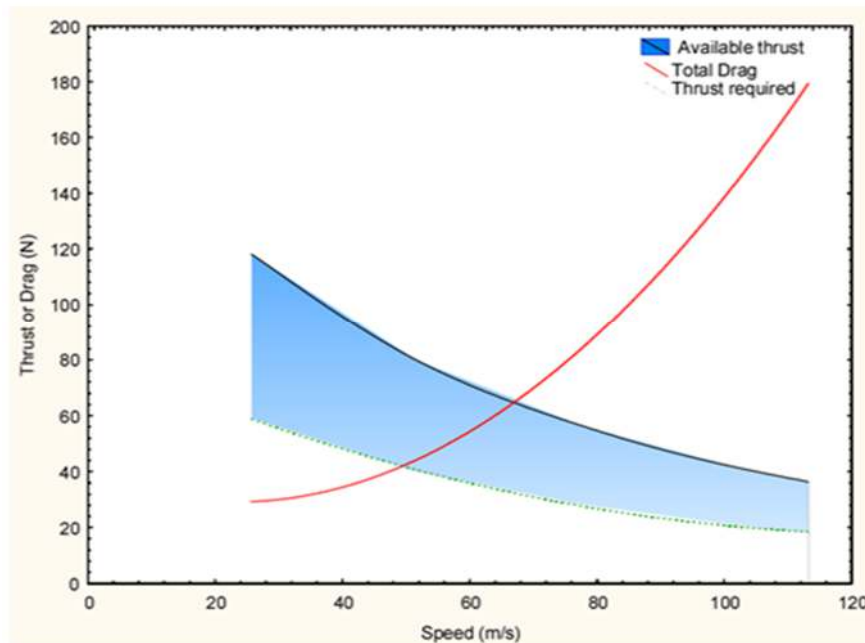


Fig. 4. Estimated Design Cruise and Maximum Speeds

Construction of Matching Plot

The matching plot technique is an analytical process developed by NASA that employs the use of flight mechanics theories to evaluate aircraft performance requirements [1],[2]. The matching criteria for all the parameters are plotted on one graph and the acceptable regions and design points are determined based on set criteria. The performance indicators used in this technique are the wing loading (W/S) and power loading (W/P). In all of the equations, W/P is defined as a function of W/P. W/S is the ratio between aircraft weight and the wing area. It

indicates how much load or weight is held by each unit area of the wing. Power loading is the ratio between aircraft weight and engine power. It gives a measure of how heavy the aircraft is with respect to engine power. Details of the matching plot technique can be obtained in references [6,9,10]. The performance parameters used in this method includes stall speed (V_s), maximum speed (V_{max}), takeoff run distance (STO), maximum rate of climb (ROC_{max}), and service ceiling (h_c). The theoretical principles based on which each parameter requirement is met are discussed in the following sub-sections [2].



Stall Speed Requirement

The stall speed (V_s) with no flaps is calculated by balancing lift (L) and weight (W) at the aircraft maximum lift coefficient (C_{Lmax}) [2][11]. Thus:

$$W = L = \frac{1}{2\rho V_s^2 S C_{Lmax}} \quad (23)$$

Solving for V_s gives

$$V_s = \sqrt{\frac{2W/S}{\rho C_{Lmax}}} \quad (24)$$

Maximum Speed Requirement

The engine sizing based on maximum speed requirements for a propeller driven aircraft is given by [2]:

$$\left[\frac{W}{P}\right]_{V_{max}} = \frac{\eta_p}{\frac{1}{2}\rho_0 V_{max}^3 C_{D0} \left(\frac{W}{S}\right) + \frac{2K}{\rho\sigma V_{max}} \left(\frac{W}{S}\right)} \quad (25)$$

Take-off Run Distance Requirement

Aircraft take-off segment includes the ground run, rotation, and climb to at least 15m [10, 13]. Analysis of this phase of flight gives an estimate of the engine size required to take-off within a desired distance. The engine sizing based on take-off run distance requirement for a propeller driven aircraft is given by [2]:

$$\left[\frac{W}{P}\right]_{S_{TO}} = \frac{1 - \exp\left[0.6\rho g C_{D_G} S_{TO} \frac{1}{W}\right]}{\mu - \left(\mu + \frac{C_{D_G}}{C_{L_R}}\right) \left[\exp\left(0.6\rho g C_{D_G} S\right)\right]} \quad (26)$$

Where the take-off speed (V_{TO}) is assumed to be $1.1V_s$ and the runway coefficient of friction μ is 0.04[6]. The takeoff lift and drag coefficients for take-off configuration are given by:

$$C_{L_{TO}} = C_{L_c} + \Delta C_{L_{flapTO}} \quad (27)$$

Where the aircraft cruise lift

coefficient C_{L_c} and the additional lift coefficient generated by flap at take-off configuration $\Delta C_{L_{flapTO}}$ are assumed to be 0.37 and 0.6, respectively [6]. Therefore, the aircraft drag coefficient at take-off configuration ($C_{D_{TO}}$) is given by:

$$C_{D_{TO}} = C_{D_{0TO}} + K C_{L_{TO}}^2 \quad (28)$$

The zero-lift drag coefficient at take-off configuration ($C_{D_{0TO}}$) is

$$C_{D_{0TO}} = C_{D_0} + C_{D_{LG}} + C_{D_{0HLDTO}} \quad (29)$$

In the absence of high lift devices drag coefficient ($C_{D_{0HLDTO}} = 0$). The take-off rotation lift coefficient (C_{L_R}) is given by:

$$C_{L_R} = \frac{C_{L_{max}}}{(1.1)^2} = \frac{1.47}{1.21} = 1.21 \quad (30)$$

and the variable C_{D_G} is given by:

$$C_{D_G} = (C_{D_{TO}} - \mu C_{L_{TO}}) \quad (31)$$

Substituting all the solved variables in Equation (4) gives the power requirement at take-off configuration.

Rate of Climb Requirement

Every aircraft small or large, military or civil must meet the certain rate of climb requirements [13]. The engine sizing based on the rate of climb (ROC) requirements is given by Equation (8) [2].

$$\left[\frac{W}{P}\right]_{S_{TO}} = \frac{1}{\frac{ROC}{\eta_p} + \sqrt{\frac{2}{\rho\sqrt{\frac{3C_{D_0}}{K}}}} \left(\frac{W}{S}\right) \left(\frac{1.155}{\left(\frac{L}{D}\right)_{max}}\right)} \quad (32)$$

Service Ceiling

By definition, service ceiling is the highest altitude an aircraft can reach by its own engine and have sustained flight. [8, 10] It is not



as critical as take-off or ROC requirement but it is part of the aircraft design considerations. The

engine sizing based on the service ceiling of a propeller driven aircraft is given by [2]:

Table 1: Result Comparison With Published Data Of Same Class UAVs.

UAV	MTOW (kg)	Wingspan (m)	Engine type	Power (hp)	Ref.
Aero-light	40	4	Piston	10	[14]
Canard	40	2.4	Piston	10	[15]
Sheddon MK3	40	4.2	Piston	10	[15]
Model	38	3.3	Piston	10.9	This work

$$\left[\frac{W}{P_{SL}} \right]_c = \frac{\sigma_c}{\frac{ROC}{\eta_P} + \sqrt{\frac{2}{\rho_c \sqrt{\frac{3C_{D0}}{K}}}} \left(\frac{W}{S} \right) \left(\frac{1.155}{\left(\frac{L}{D} \right)_{max} \eta_P} \right)} \quad (33)$$

Further reading on the above method can be obtained from [2].

RESULTS AND ANALYSIS

The equations described in the above section were solved in Excel spreadsheet and the results plotted in Statistical II software for greater effect. The plots show the variations of the power loading with rest to the wing loading as shown in Fig. 5. The area below each graph satisfies one performance criterion or another. In the case of Vs line, the left side satisfies the stall speed requirement. The intersection of Vs line with the ROCmax line gives the lowest engine power and therefore makes the design point. Based on this

method, the corresponding value on the y-axis gives the design power loading (W/P)d. From the result obtained, (W/P)d is 8.2lb/hp. Therefore the estimated engine power with the lowest operating cost is given by:

$$P = \frac{W}{(W/P)_d} = \frac{88.2lb}{\frac{8.1lb}{hp}} = 10.9hp \quad (34)$$

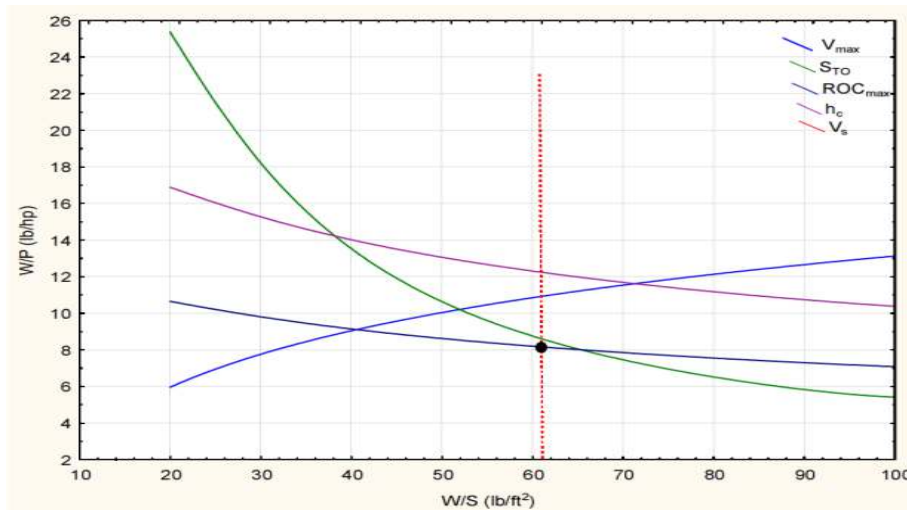


Fig. 5: Matching Plot for Engine Sizing.

Validation of Result

The performance requirements plots in Fig. 4 are completely consistent with typical matching plots in [6], [9], and [10]. The solution was validated against published data [14],[15] of three similar UAVs. The calculated values were close to the published data by over 90% as shown in Table 1. Because of the small deviation, it can be concluded that this methodology is accurate for preliminary sizing of UAV engines. Furthermore, the standard established here is comparable low altitude operating UAVs [17].

CONCLUSION

This is an on-going project, at this level

therefore the required engine power for the GULMA UAV has been sized based on the drag and performance requirements. Matching plot technique was used to determine the design point which gives indication of the lowest engine power requirement that meets the performance requirements. The initial result is in line with the published data of currently flying UAVs.

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NOMENCLATURE

C_D	Drag Coefficient
C_{D0}	Zero-lift Drag Coefficient
C_L	Lift Coefficient
C_{D0f}	Fuselage Zero-lift Drag Coefficient
C_{D0w}	Wing Zero-lift Drag Coefficient
C_{D0vt}	Vertical Tail Plane Zero-lift Drag Coefficient
C_{D0ht}	Horizontal Tail Plane Zero-lift Drag Coefficient
C_{D0LG}	Landing Gear Zero-lift Drag Coefficient
C_{D0S}	Landing Gear Strut Zero-lift Drag Coefficient
C_{D0HLD}	High Lift Device Zero-lift Drag Coefficient
C_f	Fuselage Skin Friction Coefficient



fLD	Fuselage Length-Diameter Ratio Function
fM	Mach Number Function
S _{wet}	Wetted Surface Area of an Aircraft Part
C _{Dmin}	Minimum drag coefficient (v,ht-vertical, horizontal tailplane)
ρ	Density
M	Viscosity
C _{L,R}	Lift Coefficient for Range
C _f	Fuselage Skin Friction Coefficient

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