Development of a Preliminary Design Tool for Unmanned Aerial Systems

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This project develops an unmanned aerial system (UAS) design process. A review of literature in this field identified significant gaps in knowledge on sizing relationships for rotary and electrically powered aircraft. The method incorporates empirical analysis of present platforms in order to generate statistical fits for performance and sizing characteristics in future designs. A UAS database is created, with a distinction between fixed wing and rotary configurations. Within each configuration aircraft are classified by propulsion system. A Design of Experiment model is developed to analyze the collected database and determine input parameters that significantly contribute to output performance. Results gathered are used in preliminary weight sizing of fixed wing and conventional rotary UAS. The validation of the tool compared to current platforms yields a variance of 11.38% across a range of parameters. This is well below the 20% threshold that is deemed acceptable in comparison to works such as Khromov\(^1\).

Nomenclature

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>(A)</td>
<td>disk area ([\text{m}^2])</td>
</tr>
<tr>
<td>(C_L)</td>
<td>Lift coefficient</td>
</tr>
<tr>
<td>(C_T)</td>
<td>Thrust coefficient</td>
</tr>
<tr>
<td>(E)</td>
<td>endurance ([\text{hr}])</td>
</tr>
<tr>
<td>DBV</td>
<td>database value</td>
</tr>
<tr>
<td>(EV)</td>
<td>estimated trend-line value</td>
</tr>
<tr>
<td>FOM</td>
<td>figure of merit</td>
</tr>
<tr>
<td>MTOM</td>
<td>maximum take-off mass ([\text{kg}])</td>
</tr>
<tr>
<td>(P_i)</td>
<td>induced power ([\text{W}])</td>
</tr>
<tr>
<td>(P_o)</td>
<td>profile power ([\text{W}])</td>
</tr>
<tr>
<td>(P_p)</td>
<td>parasitic power ([\text{W}])</td>
</tr>
<tr>
<td>(S)</td>
<td>wing reference area ([\text{m}^2])</td>
</tr>
<tr>
<td>SFC</td>
<td>specific fuel consumption ([1/\text{s}])</td>
</tr>
<tr>
<td>(V_i)</td>
<td>induced velocity ([\text{ms}^{-2}])</td>
</tr>
<tr>
<td>(W_B)</td>
<td>battery weight ([\text{kg}])</td>
</tr>
<tr>
<td>(W_E)</td>
<td>Empty weight of platform ([\text{kg}])</td>
</tr>
<tr>
<td>(W_f)</td>
<td>fuel weight ([\text{kg}])</td>
</tr>
<tr>
<td>(W_{pl})</td>
<td>payload weight ([\text{kg}])</td>
</tr>
<tr>
<td>(W_{TO})</td>
<td>take-off weight ([\text{kg}])</td>
</tr>
<tr>
<td>(\rho)</td>
<td>density ([\text{kgm}^{-3}])</td>
</tr>
<tr>
<td>(\omega)</td>
<td>disk loading ([\text{Nm}^{-2}])</td>
</tr>
<tr>
<td>(\sigma)</td>
<td>Blade Solidity</td>
</tr>
<tr>
<td>(\mu)</td>
<td>advance ratio</td>
</tr>
<tr>
<td>(\Lambda_{0.25c})</td>
<td>Wing Sweep at quarter chord ([\text{deg}])</td>
</tr>
</tbody>
</table>

I. Introduction

Unmanned Aerial Systems (UAS) provide maneuverability in dynamic environments, mitigating the risk and cost associated with human operators\(^1\). Understandably, this has led to increased UAS reliance and development. To sustain this, models are required to optimize and predict the performance of potential designs in relation to the specified mission. Limited resources are available at present that translate consumer requirements to a preliminary design framework. Proposed methods have included forming discretized conditions to generate an understanding of fixed wing airflow using Computational Fluid Dynamics (CFD) by Masood\(^2\). Rotary design tools, alternatively, adopt a Multidisciplinary Optimization (MDO) model for UAS parameters with mission goals such as climb and loiter\(^3\). These techniques are merited, however are complex and time consuming. The development of a simplified empirical tool will allow for rapid, accurate approximations in the preliminary design phase. Such a design aid will also account for configurations of fixed wing and rotary, each type being governed by unique statistical sizing equations.

A. Aim

The project focuses on generating trends from available data of UAS to assist in the rapid preliminary sizing of such designs. Design of Experiment (DOE) screening analysis is then implemented in order to identify the cause and effect relationships that exist between the input factors (independent variables) compared to output
responses\(^4\) (performance dependent variables). The generation of key statistical fits across parameters allows for estimations in aerodynamic, weight, power and performance outputs of a designed system. This empirical analysis is then incorporated in a user-friendly Microsoft Excel interface. The UAS design method developed is validated to ensure correspondence with design tools used in the undergraduate Aircraft Design Course I & II at the University of New South Wales.

II. Literature Review

A. Fixed Wing UAS

Palmer\(^5\) highlighted the scarcity of empirical corrections in the preliminary design phase of UAS, and developed scaling laws, through interactions of key characteristics across 856 fixed wing UAS. This method of empirical regression analysis\(^6\) is a recognized basis for initial estimation across scientific disciplines. Derivation from power laws allows the dataset to be transformed logarithmically in order observe relationships in a linear manner\(^4\), as shown in Equation 1.

\[
Y = A \cdot X^B
\]  

(1)

In this equation X and Y denote the independent and dependent variable respectively, and A and B represent regression coefficients that are derived using a least squares statistical approach. Optimal trend lines are formed in a scatter plot of data through this method\(^6\), where identified trend lines have respective \(R^2\) values. This is a percentage value of how close a model is to the data set, with higher values indicating a greater degree of consistency against available data, as shown in equation 2.

\[
R^2 = 1 - \frac{\sum r^2}{\sum (y_n - \bar{y})^2} = 1 - \frac{(\text{Sum of Residual})^2}{\text{Total Variation}}
\]  

(2)

Results obtained by Palmer\(^5\) focused on driving mission parameters including interactions between payload mass, endurance, power required and geometric sizing as a function of MTOM. Figure 1 displays a representation of these relationships, with distinct linear trend lines observed specific to the UAS propulsion system power and corresponding UAS mass. Similar trends across mission parameters were outlined. It was concluded that in order to use these sizing relations, the type of propulsion system must be specified at the beginning of the conceptual design process. The next challenge was to determine the suitability of a propulsion system to a UAS system requirements.

Veerman\(^1\) through market research of 400 platforms formed a streamlined classing system that accommodated for the NATO classification model in Figure 2 and categories prescribed by UAS design texts such as Austin\(^7\). In line with Palmer\(^5\) it was determined that a proposed design tool would delineate propulsion classification by MTOM. Figure 3 depicts the propulsion system suitable for each MTOM category, and the power output for each class is identified as a differentiating factor.

Table 8. Coefficients for power-law fits of installed engine/motor power as a function of MTOM, \(P = A \cdot \text{MTOW}^B\), where \(P\) and \(\text{MTOW}\) are in units of watts and kilograms, respectively.

<table>
<thead>
<tr>
<th>Type</th>
<th>(a)</th>
<th>(A)</th>
<th>(B)</th>
<th>(\text{MTOW}^{\text{Max}})</th>
<th>(\text{Power}^{\text{Max}})</th>
</tr>
</thead>
<tbody>
<tr>
<td>Air</td>
<td>4/8</td>
<td>98.51</td>
<td>1.999</td>
<td>0.922</td>
<td>140</td>
</tr>
<tr>
<td>Battery</td>
<td>69</td>
<td>77.98</td>
<td>1.996</td>
<td>0.800</td>
<td>120</td>
</tr>
<tr>
<td>Fuel Cell</td>
<td>22</td>
<td>56.22</td>
<td>0.599</td>
<td>0.999</td>
<td>90</td>
</tr>
<tr>
<td>Solar</td>
<td>20</td>
<td>36.21</td>
<td>0.599</td>
<td>0.999</td>
<td>90</td>
</tr>
<tr>
<td>Rocket</td>
<td>264</td>
<td>284.4</td>
<td>0.544</td>
<td>0.848</td>
<td>150</td>
</tr>
<tr>
<td>Turbine</td>
<td>11</td>
<td>130</td>
<td>0.348</td>
<td>0.656</td>
<td>220</td>
</tr>
</tbody>
</table>

Figure 1. Installed engine/motor power as a function of MTOW.
Rotary UAS fulfill mission requirements that fixed wing cannot deliver including vertical take-off and landing (VTOL), lateral flight and hovering. In contrast to methods employed for attaining fixed wing relations, Keith and Hall focused on data collection of manned VTOL systems and extrapolating the data for unmanned platforms. Empirical relations are outlined in accordance with the American Helicopter society wheel of propulsion concepts. Rapid sizing methods were determined for MTOM and empty weight. Limitations arose with the insufficient availability of data and the incorporation of short-takeoff and landing, fixed wing aircraft. This raised considerable bias in the resultant regression. The error due to this were statistically significant ranging from 36 to 41 percent dependent on propulsion approach. Additionally the authors confirmed the validity of their fits for systems greater than 2500 pounds which only account for MALE and HALE classified UAS from Figure 3. It becomes clear that rotorcraft sizing relations according to propulsion are not well classified at present, notably for the Small and Tactical classes.

Latorre examined the operation of quadrotors, where momentum and blade element theory were combined in order to optimize the electric propulsion system. This method required rotor geometry parameters to be defined initially, values that are readily available. Initial inputs such as chord, twist distribution and airfoil data, were also required, specifications that are not readily available for RUAS. Bershadsky examined empirical relations for the mass of electrical propulsion system and rotor mass, obtaining results for non-dimensional chord ratio and twist angle as a function of radial segment. These studies can be used in conjunction for an electric powered design tool with a multirotor UAS.

There still remained the disparity in development for Small-HALE classified RUAS, with piston or turbine propulsion. In order to achieve this, exploring sizing manned rotorcraft systems was necessary. Leishman echoed Latorre’s methods in identifying the sizing of the main-rotor (MR) as a critical first step as hover capability is essential for RUAS design. Trade-offs in blade planform and airfoil selection must be made in order to achieve the lowest disk loading for given MTOM. Khromov and Rand supplemented this through additions of aerodynamic properties such as blade tip speed and solidity. Tail rotor (TR) dynamics were also documented.

Analysis of current literature on rotorcraft design, indicate a complex interplay of aerodynamic characteristics. Advancing blade compressibility effects must be balanced against retreating blade stall limit. The torque interactions between the MR and TR, and the type of propulsion system used, cause difficulties in identifying driving design variables. Therefore, a statistical analysis on current RUAS, becomes necessary to determine key design parameters.

III. Method

A. Database Creation

Existing proven UAS designs are used to construct a database of key sizing and propulsion parameters. This database will be used to construct the statistical sizing framework for potential design concepts. The combination of known aerodynamic performance equations with the fidelity of a statistical model allow for outputs generated by the mission requirements and the unique selected features of a proposed UAS. The focus of this database is to identify trends for the identified gap in

<table>
<thead>
<tr>
<th>Fixed Wing</th>
<th>RUAS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Category</td>
<td>Number (n)</td>
</tr>
<tr>
<td>Piston/Wankel</td>
<td>120</td>
</tr>
<tr>
<td>Electrical</td>
<td>30</td>
</tr>
<tr>
<td>Turbopfan/jet</td>
<td>8</td>
</tr>
<tr>
<td></td>
<td>Multi-Rotor</td>
</tr>
</tbody>
</table>

Table 1. Denomination of UAS Database
RUAS. However, fixed wing UAS data is also collated to validate the design tool using the Palmer\textsuperscript{4} sizing relations. The total number of UAS studied is 242, with a detailed breakdown of the database displayed in Table 1. The data was collected from Jane’s\textsuperscript{16} Aircraft Unmanned collection and supplemented by secondary manufacturer specifications when required.

### B. Rotary Database

Statistical trends for conventional and multi-rotor configurations are collected given the availability of current RUAS data. Tandem and coaxial variants are discounted as available data is insufficient in satisfying the degree of freedom (DOF) required for the examination of interactions\textsuperscript{17}. This value indicates the number of independent and fair comparisons a data sample can provide. For example, in the tandem category, if the main effects are compared against three outputs or more, the DOF would be negative and a fair test is unviable. This is relevant as a range of input/output analysis is conducted, a process that requires large sample sizes. An excerpt of collected data is displayed in Table 2.

<table>
<thead>
<tr>
<th>Aircraft Name</th>
<th>Altitude (m)</th>
<th>Propulsion Type</th>
<th>Power (kW)</th>
<th>Endurance (hr)</th>
<th>Range (km)</th>
<th>Cruise (km/hr)</th>
<th>MTOW (kg)</th>
<th>Payload (kg)</th>
<th>Rotor Diameter (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Skeldar R350</td>
<td>2000</td>
<td>Turbine</td>
<td>25</td>
<td>3</td>
<td>80</td>
<td>120</td>
<td>150</td>
<td>38</td>
<td>3.5</td>
</tr>
<tr>
<td>Northrop RQ-8A</td>
<td>6095</td>
<td>Turbine</td>
<td>313</td>
<td>3</td>
<td>209</td>
<td>232</td>
<td>1428</td>
<td>272</td>
<td>8.398</td>
</tr>
<tr>
<td>Campocopter S-100</td>
<td>5485</td>
<td>Piston</td>
<td>36</td>
<td>6</td>
<td>180</td>
<td>102</td>
<td>200</td>
<td>100</td>
<td>3.4</td>
</tr>
</tbody>
</table>

**Table 2.** Excerpt of collected RUAS parameters within overall database

Rotorcraft momentum theory\textsuperscript{13} assumes a hovering flight condition, with a stream-tube of flow is observed above and below the rotor disk. In this state the thrust produced is equal to the weight of the RUAS. The change in momentum per unit of time is equated with Bernoulli’s change in pressure across the rotor disk. The relations for induced velocity, power and hover efficiency are given in Equations 3, 4 and 5. This theory forms the basis of derived values in Table 3.

\[
V_i = V_h = \frac{T}{2 \rho A} \quad (3) \quad P_i = T V_i = \frac{T^3}{2 \rho A} \quad (4) \quad \text{Hover Efficiency} = \frac{T}{P_i} \quad (5)
\]

<table>
<thead>
<tr>
<th>Aircraft Name</th>
<th>Disk Loading (N/m²)</th>
<th>Induced Velocity (m/s)</th>
<th>Induced Power per MR (kW)</th>
<th>Hover Efficiency (N/kW)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Skeldar R350</td>
<td>152.7932516</td>
<td>9.873978341</td>
<td>11.60877461</td>
<td>126.6283522</td>
</tr>
<tr>
<td>Northrop RQ-8A</td>
<td>252.8305161</td>
<td>12.4442232</td>
<td>142.2623596</td>
<td>98.43925093</td>
</tr>
<tr>
<td>Campocopter S-100</td>
<td>213.8845212</td>
<td>11.499639412</td>
<td>18.3985506</td>
<td>106.3301307</td>
</tr>
</tbody>
</table>

**Table 3.** Excerpt of derived values for RUAS

### C. Design of Experiment – Results

A screening test is performed with the DOE model shown in Figure 4. This essentially creates the statistical design to determine the significant factors that influence output responses. There is a combination of qualitative variables and quantitative measurements. The method adopted is a 2 –level design testing for linear effects between two points, this is justified by the sample size and factors not meeting the requirement for a 3 – level design which are more relevant for optimization problems by testing for quadratic effects\textsuperscript{4}.

The screening model was first applied to 35 conventional RUAS, the remaining 22 are set aside for validation purposes. Regression analysis highlights any outliers, and the suitability of the dataset for a linear model. A random distribution of residuals also supports the presence of a linear model\textsuperscript{4}. Such a distribution is depicted in Figure 5, with similar trends observed amongst other variables. Interaction graphs, form a visual representation of cause and effect interactions between parameters. Parallel lines indicate null interaction between process variables, the strength of a relation is proportional to the degree of departure from the parallel.

![Figure 4. Historical DOE Model of RUAS](image-url)
Interaction graphs are generated for input/output relations as per the DOE model. Figure 6 is an example of such a graph, with MTOM limits associated for each propulsion category identifiable, in conjunction with the presence of linear interaction. This statistical significance is also supported by Pearson Correlation coefficients, used to determine the strength and direction of continuous quantitative data, taking a value between -1 to 1. The statistical p-value is required to be below 0.05, implying strong evidence against the null hypotheses. In this DOE model this hypothesis is when two compared variables do not have any influence on each other. Results of significant p-values and Pearson coefficients are collated in Table 4.

<table>
<thead>
<tr>
<th>MTOM (kg)</th>
<th>Rotor Diameter (m)</th>
<th>Payload (kg)</th>
<th>Endurance (hr)</th>
<th>Power (kW)</th>
<th>Cruise (km/hr)</th>
</tr>
</thead>
<tbody>
<tr>
<td>p-Value</td>
<td>0.0</td>
<td>0.0</td>
<td>0.93659</td>
<td>0.0</td>
<td>0.0</td>
</tr>
<tr>
<td>Pearson Coefficient</td>
<td>0.89451</td>
<td>0.75164</td>
<td>0.72811</td>
<td>0.98155</td>
<td>0.56785</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Rotor Diameter</th>
<th>p-Value</th>
<th>0.0</th>
<th>0.0</th>
<th>0.0</th>
<th>0.0</th>
<th>0.0</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pearson Coefficient</td>
<td>0.89451</td>
<td>0.75164</td>
<td>0.7982</td>
<td>0.87943</td>
<td>0.71273</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Payload</th>
<th>p-Value</th>
<th>0.0</th>
<th>0.0</th>
<th>0.0</th>
<th>0.0</th>
<th>0.0004</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pearson Coefficient</td>
<td>0.93659</td>
<td>0.75164</td>
<td>0.67711</td>
<td>0.89299</td>
<td>0.44006</td>
<td></td>
</tr>
</tbody>
</table>

Table 4. Pearson correlation and p values for Parameters of Significance.

Whilst these tools allow for the identification of linear effects, Pareto plots enable highlighting the presence of the driving input factor for each output response. This enables elimination of cases with multi-collinearity, allowing for independent analysis of relationships. In this study Pareto responses indicated minimal effect of propulsion types across outputs, unlike results determined by Palmer.

The analysis of electric multi-rotor UAS provided regression results that are non-orthogonal within output parameters. This occurs when relationships exist however cannot be tested independently due to co-dependency of parameters. As seen in Figure 7, interactions between MTOW and Power do not display meaningful trends for differing rotor assemblies. These findings are supported by further statistical analysis, with no p-values of significance found between any of the variables. The source for this could be attributed to several factors. Lack of available data for electric RUAS is a major limitation, often specifications of JANE’s had to be supplemented by manufacture specifications. Additionally, aspects such as the rotor diameter had to be approximated from overall product dimensions. Distributed electric propulsion (DEP) also allows for incredible design freedom, with the ability to integrate multiple rotors without a notable weight penalty, a concept highlighted by designs such as NASA Grease Lightning. Patterson also indicated that the combination of a reduced volume and decreased mass of electric motors, in comparison to traditional combustion assemblies, allows for increased payload capacity and a reduction in wetted area. This essentially
violates traditional scaling laws\textsuperscript{15} that are used for piston/turbine aircraft designs. It also explains the lack of linearity present between inputs of payload weight and outputs such as battery power. Retests of such a DOE model can be improved by increasing the sample size and by attaining more accurate product information from manufacturers. In the proposed design tool, approximations have been made from preliminary sizing using performance equations from Gundlach\textsuperscript{20}, Gur\textsuperscript{1} and Bershadsky\textsuperscript{12}. However, the validity of these performance equations cannot be determined due to limited data and hence are confined to the preliminary sizing stage as an initial estimate.

IV. Design Tool – Fixed Wing UAS

A. Preliminary Sizing

This stage involved determining a methodology to attain an initial take-off weight estimate, \( W_{TO} \), the breakdown of which is highlighted in equation 6. Benchmark analysis was conducted to identify aircraft most suitable to mission requirements. A propulsion system is designated according to the design weight category from Figure 2 and 3. As UAS exhibit a higher degree of sensitivity to payload changes\textsuperscript{30}, compared to manned aircraft, primarily due to lower \( W_{TO} \). It seemed pertinent that payload inputs are used in conjunction with the Palmer\textsuperscript{4} power law coefficients in order to generate a first estimate of \( W_{TO} \). The fuel weight fraction\textsuperscript{11} is then calculated by separating the mission sequence into flight segments (climb, cruise, loiter, etc.). The aircraft weight at each segment is determined proportional to the weight at the preceding segment. This is done by rearranging the Breguet range (equation 7) for cruise segments and endurance (equation 8) for loiter. Lift to drag ratio, \( L/D \) approximations are made according to aspect ratio relations from Gundlach\textsuperscript{20}.

\[
W_{TO} = \frac{W_{FL}}{1 - \frac{W_{Empty}}{W_{TO}} - \frac{W_{Fuel}}{W_{TO}} - \frac{W_{Battery}}{W_{TO}} - \frac{W_{Motors}}{W_{TO}}}
\]

\( \text{Cruise : } \frac{W_i}{W_{i-1}} = \exp\left(\frac{-R(SFC)}{V(L/D)}\right) \)

\( \text{Loiter: } \frac{W_i}{W_{i-1}} = \exp\left(\frac{-E(SFC)}{L/D}\right) \)

\[
\frac{W_i}{W_{TO}} = 1.06(1 - W_x/W_{TO})
\]

Specific fuel consumption values are adjusted according to propeller or jet engines and converted to the equivalent metric specific impulse\textsuperscript{21}. Multiplying these incremental fractions provides an overall mission weight fraction \( W_x/W_{TO} \), and the resultant fuel fraction taking into account a six percent fuel reserve is determined by equation 9. The empty weight fraction is determined using the Palmer\textsuperscript{4} fits, and hence the optimum \( W_{TO} \) is calculated iteratively from an initial benchmark value, forming the second \( W_{TO} \) estimate. An average value between the two methods provides a final breakdown of take-off weight for piston/turbine variants.

Preliminary sizing for electric variants is also conducted, with initial characteristics of batteries taken from Gundlach\textsuperscript{20}. It is identified that an electrical propulsion assemble consists of a motor, battery, electronic speed controller (ESC) and propeller. Initial estimates of motor rotational speed and driver input current allow for the determining of motor mass\textsuperscript{3} and coefficients taken from Bershadsky\textsuperscript{12} allow for a secondary estimate. Following this the mass of the battery is calculated as a function of battery specific density and energy. ESC mass is determined as a function of maximum current input and propeller mass is evaluated based on material of use\textsuperscript{12}. Given that UAS operation must be optimized for loiter condition where from Gur\textsuperscript{1}; \( V_{loiter} = 1.2V_{stall} \). Using a relation for \( V_{stall} \) relative to electrical assembly mass and wing area provides the ability to check if this condition is satisfied for determined total propulsion mass.

B. Power Loading

Statistical approximations of power loading, \( P/W \) are made with the MTOM input from the Preliminary sizing phase. Thrust matching techniques from Raymer\textsuperscript{15} are then used in order to determine the Thrust to weight, \( T/W \) ratio, \( L/D \) corrections for cruise condition for propeller or jet operation are also taken into account. \( T/W \) for the climb condition is also evaluated, by including the thrust vector required for the climb gradient. Equations 10 & 11 depict this respectively. Initial \( T/W \) estimates must be the maximum value that is obtained from statistical or
thrust matching relations, which enables greater acceleration, climb rates and sustain higher turn rates. This also allows for the determination of the thrust required per engine.

\[
\frac{T}{W}_{\text{cruise}} = \frac{1}{(L/D)_{\text{cruise}}} \quad (10) \quad \frac{T}{W}_{\text{climb}} = \frac{1}{(L/D)_{\text{climb}}} + \frac{V_{\text{vertical}}}{V} \quad (11)
\]

C. Wing Loading

The Wing loading \( W/S \), is the ratio of the aircraft weight to the area of the reference wing. This parameter influences stall speed and take-off, landing and turn performance. Calculations must account for the specific weight of the aircraft at a given flight condition, and hence \( W/S \) must be multiplied by the mission segment fraction in order to be compared against an equivalent take-off condition. These require \( T/W \) as a preliminary step and to ensure that the wing provides lift in all circumstances the designer must select the lowest of the estimated wing loadings\(^5\).

Statistical fits for Wing area are determined from Palmer\(^5\). Subsequently, wing loading for required stall speed is evaluated using the assumption of lift being equivalent to weight shown in equation 12. The maximum lift coefficient is also adjusted for the wing sweep angle at the quarter chord (equation 13).

\[
W = L = q_{\text{stall}} SC_{l_{\text{max}}} = \frac{1}{2} \rho V_{\text{stall}}^2 SC_{l_{\text{max}}}
\]

\[
\frac{W}{S} = \frac{1}{2} \rho V_{\text{stall}}^2 C_{l_{\text{max}}} \quad \text{where} \quad C_{l_{\text{max}}} = 0.9 C_{l_{\text{max}}} \cos \Lambda_{0.25c}
\]

Wing Loading for best cruise performance usually provides larger \( W/S \) values than determined for stall. This results smaller wing sizes, inappropriate for the complete UAS flight sequence. \( W/S \) values for maximum propeller and jet range are derived from parasite and induced drag relations and are outlined as follows in equations 14 & 15. Parasitic drag arises due to non-lift dependent parameters and for the subsonic condition where most UAS operate this equates to skin friction drag. Induced drag on the other hand is lift dependent and due to wingtip vortices that form at finite wings. A propeller aircraft obtains maximum range at a speed for best \( L/D \), occurring when parasite and induced drag are equivalent. For jet propulsion this maximum range condition is obtained when parasite drag is triple the induced drag\(^2\).

\[
(W/S)_{prop(cruise)} = q \sqrt{\pi AR_e C_{\text{D0}}} \quad (14) \quad (W/S)_{jet(cruise)} = q \sqrt{\pi AR_e C_{\text{D0}}/3}
\]

The inverse relationship for induced and parasite drag hold true for the loiter condition and hence the following relations are used.

\[
(W/S)_{prop(loiter)} = q \sqrt{3 \pi AR_e C_{\text{D0}}} \quad (16) \quad (W/S)_{jet(loiter)} = q \sqrt{\pi AR_e C_{\text{D0}}}
\]

UAS will often require high degrees of maneuverability in response to an external event. This can be measured by the turn rates achieved during flight. Sustained turn rates\(^{15}\) occur when the aircraft has enough thrust to maintain velocity and altitude in the turn. Instantaneous turn is the highest possible rate limited only by the UAS stall or structural limit. Equations 18 & 19 depict the W/S at the given turn conditions.

\[
(W/S)_{\text{instantaneous}} = \rho V^2 C_{l_{\text{max}}}/2n \quad (18) \quad (W/S)_{\text{sustain}} = \rho V^2 / 2n \sqrt{\pi AR_e C_{\text{D0}}} \quad (19)
\]

V. Design Tool – Conventional RUAS

A. Preliminary Sizing

The breakdown of \( W_{TO} \) of a RUAS design is analogous to the fixed wing method described in equation 6. The key difference being in the calculation of fuel fraction segment shown in equation 20. In the RUAS case, these fractions are taken proportional to the power loading ratio (P/W) for each flight profile. Derived DOE statistical models form initial MTOM estimate based on input payload, displayed in Figure 8.

\[
\frac{W_i}{W_{i-1}} = \frac{SFC \times t}{W_p} \quad (20)
\]

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B. Disk Loading

This aspect of the design tool focuses on the rotor disk characteristics. The first of these is disk area, calculated by two methods; the statistical regression model and the autorotation method. The final value is chosen by the user manually. Figure 9 shows how given a mass input from the preliminary sizing phase, the appropriate rotor diameter and hence disk area can be attained. The disk loading then becomes the aircraft mass divided by disk area $W/A$.

In the autorotation method the figure of merit must be estimated, and are typically assumed to be between 0.7 and 0.8. This value represents the ratio between the ideal and actual power required to hover. Autorotation values are chosen from Taamallah\textsuperscript{23} stating typical values ranging from 4 – 12m/s. The chosen autorotation value is applied to equation 21 and the hover induced velocity is calculated. The rotor area is then evaluated from equation 3, which also provides a secondary estimate for rotor radius and disk loading.

$$\frac{V_{auto}}{V_h} = \frac{FOM^{-1} - \kappa}{1 + 3 \kappa} - \frac{7 \kappa}{1 + 3 \kappa} (\kappa = 1.15)$$

(21)

Tip speed and rotational velocity values are obtained from the statistical performance equations as a function of rotor diameter from Khromov\textsuperscript{14}. These are shown in equations 22 & 23;

$$V^{tip} = 140D^{0.171}$$

(22)

$$\Omega [\text{rad/s}] = \frac{280}{D^{0.85}}$$

(23)

The optimum number of blades can be assessed by determining the tip loss factor, which is dependent upon the thrust coefficient value. These respective values are determined by equations 24 and 25. For the number of
blades ranging from 2 to 5 consistent with current RUAS, the radius for each configuration is re-calculated, and it becomes the user choice to input an appropriate design value.

\[
C_T = \frac{T}{\rho A(\Omega R)^2} \quad (24) \quad \text{Tip loss factor} = 1 - \frac{\sqrt{C_T}}{\text{Number of blades}} \quad (25)
\]

Solidity, the ratio of blade area to rotor disk area is also estimated using statistical fits from Khromov\(^{14}\), hover efficiency at sea level, maximum load factor and retreating blade stall.

C. Power Loading

Statistical models; Figure 10, are used in tandem with performance equations for each mission segment in order to determine the appropriate RUAS installed power. Closed Form solutions\(^{13}\) for rotorcraft power are used in order to evaluate the induced, profile and parasitic drag power components for a range of flight conditions such as maximum speed, hover, climb and descent. Typical values of Tail and sub-system power are also incorporated with an appropriate multiplier to enhance the accuracy of total power predictions.

Parasitic drag estimation is given by equation 26.

\[
C_{P_{\text{parasitic}}} = \mu \frac{D_p W}{C_T} \quad (26)
\]

Using a profile drag coefficient of 0.01, from Lieshman\(^{13}\). The profile power is as follows, with empirical correction K = 4.6 in the forward flight condition and negligible in hover.

\[
C_{P_0} = \sigma \frac{C_{d_0} 8}{8} (1 + K\mu^2) \quad (27)
\]

Climb Power is according to the following relation\(^{13}\)

\[
P_{\text{climb}} = W_{TO} \times V_{\text{climb}} + (P_i + P_o + P_p)_{\text{cruise}} \quad (28)
\]

The chosen RUAS power is the maximum value that is obtained through each of these mission segments, with the statistical model as a guide. This is to ensure that designated power is sufficient to satisfy every flight operation of the RUAS.

![Figure 10. RUAS MTOM vs Power Regression Model](image)

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VI. Testing and Evaluation of Design Tool

The purpose of this phase is to compare the outputs of the design tool to UAS designs that were not incorporated in the DOE analysis. The error analysis technique employed in this paper is recognized in corresponding works\(^\text{14}\) and outlined in the following equation 29. \(EV\) and \(DBV\) represent the estimated design tool value and recorded database value respectively.

\[
e(\%) = 100 \left( \frac{EV - DBV}{DBV} \right)
\]  

(29)

In order to validate the results obtained from our design tool, it is relevant to compare the input/output behavior of the design to the database of aircraft collected. This is achieved by providing incremental mass inputs to develop a design tool trend line which can then readily be compared against trends exhibited by actual UAS platforms. A summary of the results is provided in Table 5.

Results for fixed wing aircraft against database trend lines provides error ranges well below 14 percent in MTOM, Power and Wing loading determinations. Values for W/S for turbine variants was inconclusive due to the limitations of current available data. This was a similar issue encountered by Palmer\(^3\) and hence using Raymer approximations is recommended, as most turbine UAS operate in the weight category of smaller manned platforms.

![Regression analysis - Power fit](image)

**Figure 11.** Representative comparison of Fixed Wing UAS Database values vs Design Tool for MTOW

<table>
<thead>
<tr>
<th>Fixed Wing</th>
<th>RUAS</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Category</strong></td>
<td><strong>Average Error (%)</strong></td>
</tr>
<tr>
<td>Payload vs MTOM</td>
<td></td>
</tr>
<tr>
<td>- Piston</td>
<td>10.06</td>
</tr>
<tr>
<td>- Turbine</td>
<td>11.35</td>
</tr>
<tr>
<td>MTOM vs Power</td>
<td></td>
</tr>
<tr>
<td>- Piston</td>
<td>5.52</td>
</tr>
<tr>
<td>- Turbine</td>
<td>11.38</td>
</tr>
<tr>
<td>MTOM vs Thrust</td>
<td></td>
</tr>
<tr>
<td>- Piston</td>
<td>11.38</td>
</tr>
<tr>
<td>MTOM vs Wing Loading</td>
<td></td>
</tr>
<tr>
<td>- Piston</td>
<td>13.28</td>
</tr>
<tr>
<td>MTOM vs Tail Rotor Diameter</td>
<td></td>
</tr>
</tbody>
</table>

**Table 5.** A summary of Validation Testing for Fixed wing and Rotary UAS.
For the rotary DOE model which used a randomized assortment of 38 aircraft to generate trend lines, the remaining 22 are used to validate the robustness of the proposed design tool. Figures 11 and 12 form a representative sample of the RUAS validation, and display error ranges of less than 20 percent for mass categories below 1000 kg, well within the operating MTOM of current RUAS. These values are presented in Table 5. These deviations are echoed across parameters such as power loading and rotor dimensions and were deemed acceptable as they correspond to the preliminary design works of Latorre and Rand.

Figure 11. Representative comparison of RUAS Validation Aircraft vs Design Tool MTOM

Figure 12. Representative comparison of RUAS Validation Aircraft vs Design Tool for MR Diameter

VII. Conclusions

Design Tools have been developed to enable the rapid sizing of fixed wing and conventional RUAS. This is significant in the projects contribution towards wider research, with findings of statistical significance for piston and turbine variants. Performance equations combined with statistical fits as a guide provide a working program and teaching aid that allows for the conduct of preliminary design for fixed wing and conventional rotary UAS.

VIII. Recommendations

The proposed design tools could be enhanced through greater exploration of electric propulsion concepts and their effects on parameters such as wing and power loading. At present they are beyond the scope of the ASD.
course this tool was developed for, however could prove useful to entities such as the UNSW UAS Design Team. As developments continue greater available data-sets could also assist in this area.

IX. Acknowledgements

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8 Junfong Qi, Dalei Song, Lei Dai, Jianda Han, and Yuechao Wang, “The New Evolution for SIA Rotorcraft UAV Project,” Journal of Robotics, vol. 2010,