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Wing and Engine Sizing by Using the Matching Plot Technique

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***Abstract* – The research focuses on the development of an Unmanned Aircraft System. For design purposes, a rather new design method called Systems Engineering Approach is used. Development of the whole system takes much time and effort. This paper contains a concise description of the research on the preliminary development phase of Unmanned Aircraft System air vehicle. The method was first introduced by NASA and later developed by authors of books on aircraft design used for information purposes for design and are mentioned in references. The obtained results are rather realistic and promising for further design process. The method is simple and understandable, and it should be used more often to make it more steady and reliable.**

***Keywords* – Aircraft design, engine, matching plot, unmanned aircraft system, wing.**

I. Introduction

The air vehicle of the Unmanned Aircraft System is designed by using a Systems Engineering Approach [1], [2]. The design method consists of four phases:

1) Conceptual Design;
2) Preliminary Design;
3) Detail Design;
4) Test and Evaluation.

During the conceptual design phase, the air vehicle is designed with inaccurate results. All parameters are determined based on a decision-making process and a selection technique.

The preliminary design phase uses the results of calculations. However, the determined parameters are not final and will be altered later. The parameters determined at this phase are essential and will directly influence the entire phase of part design. Therefore, a great deal of attention must be paid to ensuring the accuracy of the results of the preliminary design phase. During the preliminary design phase, three aircraft fundamental parameters are determined: maximum take-off weight (MTOW), wing reference area (S_{ref}), and engine power (P) for a prop-driven engine [3].

During the phase of air vehicle part design [4], the technical parameters of all components (wing, fuselage, tail, and engine), including geometry, will be calculated and finalized.

II. Conceptual Design Phase

At the conceptual design phase, the designer determines air vehicle configuration. Although there are no legal regulations for the Unmanned Aircraft System belonging to the category of micro air vehicles, the designer is firstly restricted by the customer's requirements. The designer should follow the requirements imposed by the customer unless he/she can prove to the customer that a specific requirement is not feasible. Other constraints a designer may face are imposed by the applicable certification specification. As it was mentioned before, there are no regulations for micro class UAS. Therefore, in this case European Aviation Safety Agency Certification Specification CS-VLA shall be applied [5], [6].

Problem statement: designing an Unmanned Aircraft System with Intelligence, Surveillance, Target Accusation and Reconnaissance functional capabilities and determined requirements (see Table I).

TABLE I
PARAMETERS OF THE UNMANNED AIRCRAFT SYSTEM AIR VEHICLE

Requirement parameter		**Value**
Absolute ceiling	h_{ac}	up to 5 000 m
Cruise ceiling	h_{cc}	300 m to 400 m
Operational range		50 km
Operational endurance		up to 300 min
Payload weight	M_{PL}	up to 0.4 kg
Take-off run	S_{TO}	hand launch (alternatively – folding launching unit)
Operational readiness		air vehicle assembling ≤ 10 min
Cruise speed	V_c	≤26 m/s (at cruising altitude 350 m ASL)

III. PRELIMINARY DESIGN PHASE

The preliminary design phase will be completed in two stages:
1) determining the air vehicle's maximum take-off weight (MTOW);
2) the air vehicle's wing and engine sizing (simultaneously).

A. The Air Vehicle's Maximum Take-Off Weight

To approximate the air vehicle's MTOW, in this phase the weight is divided into the following four elements:
1) payload weight: M_{PL} = 0.385 kg;
2) avionics weight: M_A = 0.681 kg;
3) fuel/battery weight: M_B = 2.0 kg;
4) empty weight: M_E = 4.0 kg.

The weights of these elements were taken from the statistical data average values [3], [7], [8].

The air vehicle's take-off weight is determined from the following equation:

$$W_{MTOW} = \left(M_{PL} + M_A + M_B + M_E\right) g = 69.32 \text{ N}. \tag{1}$$

B. Wing and engine sizing

Unlike in the first step, where statistical data were used, in this step the results depend on the air vehicle's performance, and the calculations involve the use of the flight mechanics theory. The results of this method are with high accuracy, and the requirements for air vehicle performance, which are used for determining sizing, are the following:
- stall speed (V_s);
- maximum speed (V_{max});
- rate of clime (ROC_{max});
- take-off run (S_{TO});
- ceiling (h_c).

In this step, two new parameters will be used in almost every equation:

1) **wing loading**. The air vehicle's weight and wing area ratio is referred to as wing loading and marked as *W*/*S*. This parameter indicates load on the unit of wing area;

2) **power loading**. The air vehicle's weight and engine power ratio is referred to as power loading or weight and power ratio and marked as *W*/*P*. This parameter indicates how heavy air vehicle in comparison to its engine power is.

The wing area and engine power are calculated in the following six steps:

1. Calculating an equation for each air vehicle's performance requirement (V_s, V_{max}, *ROC*, S_{TO}, h_c).

1.1. **Stall speed**. One of the main air vehicle's performance requirements is stall speed – V_s. For most aircraft, the stall speed shall not exceed some minimum defined value. Besides, this parameter is limited by the certification specification (EASA CS). This parameter is also important because landing is anticipated to be in deep stall. Alternatively, the landing will be designed with a parachute mechanism.

From the statistics, the following parameters were determined for the UAS air vehicle [7]–[9]: $C_{lmax} = 1.6$, $V_s = 8.5$ m/s.

Using the statistically determined parameters C_{lmax} and V_s the following equation is used for determining the wing loading parameter:

$$\left(\frac{W}{S}\right)_{V_s} = \frac{1}{2}\rho V_s^2 C_{lmax} = 70.805\,\frac{\mathrm{N}}{\mathrm{m}^2}, \tag{2}$$

where ρ is air density at sea level, 1.225 kg/m^3.

In general, lower stall speed is required as it results in a safer flight. Lower stall speed results in a safer take-off and landing. The take-off and landing speed are usually slightly higher than the stall speed ($1.1V_s$ to $1.3V_s$). Thus, considering the above mentioned facts, any stall speed lower than the one defined in the design requirements is acceptable.

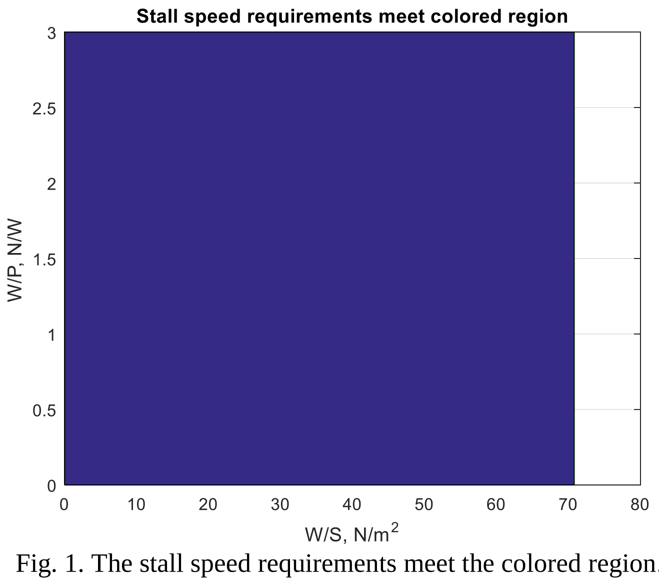

Fig. 1. The stall speed requirements meet the colored region.

1.2. **Maximum speed**. Another important air vehicle performance parameter is maximum speed. The main parameters affecting this performance parameter are air vehicle weight, wing area, and engine power. If in design requirements cruise speed requirements are defined instead of maximum speed requirements, there shall be taken a 20 % to 30 % greater maximum speed. This is because the cruise speed for propeller driven aircraft is calculated for 75 % to 80 % of engine power. So the air vehicle's maximum speed is as follows:

$$V_{max} = 1.3V_C = 33.8\,\frac{\mathrm{m}}{\mathrm{s}}. \tag{3}$$

The following equation is used for the matching plot construction:

$$\left(\frac{W}{P_{\mathrm{SL}}}\right)_{V_{\max}} = \frac{\sigma\eta_{\mathrm{P}}}{\frac{1}{2}\rho_{\mathrm{SL}} V_{\max}^3 C_{\mathrm{D0}} \frac{1}{\left(\frac{W}{S}\right)} \frac{2K}{\rho V_{\max}} \left(\frac{W}{S}\right)}. \tag{4}$$

where η_{P} is propeller efficiency coefficient equal to 0.55

The parameters for (4) are calculated as follows:

$$K = \frac{1}{\pi e \cdot AR} = 0.0331741, \tag{5}$$

where

K – induced drag factor;

e – Oswald efficiency factor (0.70 to 0.95) [10], [11];

AR – wing aspect ratio; statistically determined as – 12 [9], [12].

The zero lift-drag coefficient C_{D0} for propeller driven aircrafts is determined as follows:

$$C_{\mathrm{D0}} = \frac{2\frac{P_{\mathrm{SLmax}}\eta_{\mathrm{P}}}{V_{\max}} - \frac{4KW^2}{\rho\sigma V_{\max}^2 S}}{\rho_{\mathrm{SL}} V_{\max}^2 S}, \tag{6}$$

where

P_{SLmax} – engine maximum power at sea level, W;

ρ – air density at flight level, kg/m^3;

σ – relative air density (ρ/ρ_{SL}).

For an aircraft in the preliminary design phase, the coefficient C_{D0} can also be determined as an average value from aircraft with similar performance and configuration:

$$C_{\mathrm{D0}} = \frac{C_{\mathrm{D01}} + C_{\mathrm{D02}} + C_{\mathrm{D03}} + C_{\mathrm{D04}} + C_{\mathrm{D05}}}{5}, \tag{7}$$

where $C_{\mathrm{D0}i}$ – i-th aircraft C_{D0} coefficient [8].

At this stage, the coefficient C_{D0} is calculated as an average value:

$$C_{\mathrm{D0}} = \frac{\left(\frac{C_{\mathrm{D01min}} + C_{\mathrm{D02min}} + C_{\mathrm{D03min}}}{3}\right) + \left(\frac{C_{\mathrm{D01max}} + C_{\mathrm{D02max}} + C_{\mathrm{D03max}}}{3}\right)}{2} = 0.0245. \tag{8}$$

In further calculations, C_{D0} coefficient will be recalculated by using (7).

Based on the obtained parameters, matching plot (4) is calculated as follows:

$$\left(\frac{W}{P_{\mathrm{SL}}}\right)_{V_{\max}} = \frac{0.77322}{\frac{560.064}{\left(\frac{W}{S}\right)} + 0.001658\left(\frac{W}{S}\right)} \cdot \frac{N}{W}. \tag{9}$$

The UAS design requirements determine an operational altitude of 300 m to 400 m. Therefore, an average value of 350 m is accepted for the calculations. The design calculations at this stage are made for the air vehicle’s flight altitude of 350 m above sea level. Also, the design requirements define launching up to an absolute altitude of 5000 m above sea level, and this limit will be used in further calculations [13], [14].

To determine the acceptable region of the matching plot, which satisfies the maximum speed requirements, it is enough to analyze the equation. As V_{max} is in denominator, in case when it is increasing, the value of power loading (W/P) is decreasing. Consequently, any value of V_{max} greater than the one specified in the requirements satisfies the maximum speed requirements, and the region below the graph is acceptable.

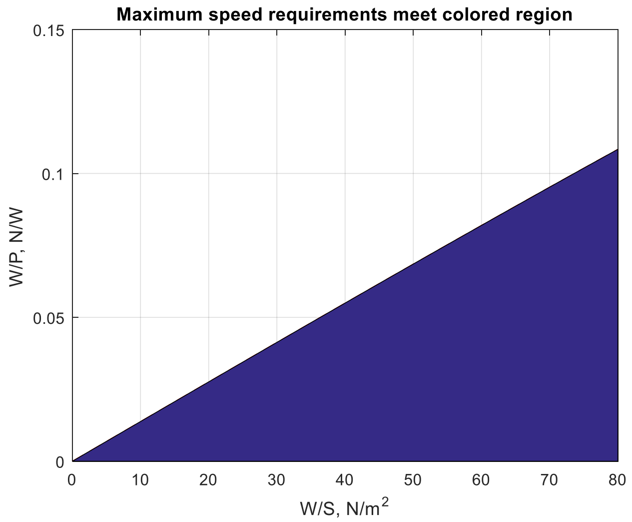

Fig. 2. The maximum speed requirements meet the colored region.

1.3. **Take-off run**. The requirements for take-off run length are usually determined as a minimum land run length as all airports have limited runways. The take-off run length is determined as a distance from the beginning of aircraft take-off to the place where a standard imaginary obstacle is placed, and the aircraft must clear it. The aircraft must clear the imaginary obstacle at the end of the air section so that the take-off run includes the land section and air section, the obstacle (EASA CS 25, CS 23, CS VLA) [4], [5].

The take-off speed is a little bit greater than the stall speed ($V_{TO} = 1.1V_s$ to $1.3V_s$):

$$V_{TO} = 1.3 \cdot 8.5 = 11.05 \, \frac{\mathrm{m}}{\mathrm{s}}. \tag{10}$$

In EASA CS VLA 51, it is defined that the range should not exceed 500 m to clear an up to 15 m tall obstacle. Thus, S_{TO} = 500 m is accepted [15].

The matching plot in this case is calculated with the equation:

$$\left(\frac{W}{P}\right)_{S_{TO}} = \frac{1 - \exp\left(0.6\rho g C_{DG} S_{TO} \frac{1}{W/S}\right)}{\mu - \left(\mu + \frac{C_{DG}}{C_{LR}}\right)\left[\exp\left(0.6\rho g C_{DG} S_{TO} \frac{1}{W/S}\right)\right]} \cdot \frac{\eta_P}{V_{TO}}. \tag{11}$$

The parameters for (11) are calculated as follows:

$$C_{D0TO} = C_{D0} + C_{D0LG} + C_{D0HLD\_TO} = 0.0245 + 0.004 + 0.055 = 0.0835, \tag{12}$$

where

C_{D0TO} – zero lift-drag coefficient during take-off;

C_{D0LG} – landing gear drag coefficient; accepted as 0.004, which is lower than the one for conventional aircraft (C_{D0LG} = 0.006 to 0.012) [9];

C_{D0HLD_TO} – high lift device drag coefficient, accepted as 0.055 (C_{D0HLD_TO} = 0.003 to 0.008) [10], [11].

The aircraft's take-off lift coefficient is as follows:

$$C_{\mathrm{LTO}} = C_{\mathrm{LC}} + \Delta C_{\mathrm{LflapTO}} = 0.3 + 0.55 = 0.85, \tag{13}$$

where C_{LC} is the aircraft's cruise lift coefficient, which is usually about 0.3 for subsonic aircraft, and $\Delta C_{\mathrm{LflapTO}}$ is high lift devices' lift coefficient in take-off configuration ($\Delta C_{\mathrm{LflapTO}}$ = 0.3 to 0.8) [10], [11].

The air vehicle's drag coefficient in take-off configuration is:

$$C_{\mathrm{DTO}} = C_{\mathrm{D0TO}} + KC_{\mathrm{LTO}}^2 = 0.0835 + 0.0331741 \cdot 0.85^2 = 0.10747. \tag{14}$$

The coefficient C_{DG} is:

$$C_{\mathrm{DG}} = C_{\mathrm{DTO}} - \mu C_{\mathrm{LTO}} = 0.10747 - 0.08 \cdot 0.85 = 0.03947, \tag{15}$$

where μ is a runway friction coefficient, and the take-off rotation lift coefficient is taken equal to C_{LTO}, that is, 0.85, and the take-off rotation speed V_{R} equal to V_{TO}, that is, 11.05 m/s.

The calculation for matching plot construction according to (11) is:

$$\left(\frac{W}{P}\right)_{S_{\mathrm{TO}}} = \frac{1 - \exp\left(\frac{142.2963}{W/S}\right)}{0.08 - 0.1264 \times \left[\exp\left(\frac{142.2963}{W/S}\right)\right]} \cdot 0.18868 \frac{N}{W}. \tag{16}$$

To determine the acceptable region of the matching plot, which satisfies the take-off run requirements, we need to analyze the equation. The equation numerator and denominator contain an exponential value of S_{TO} parameter. If the take-off runway is increasing, the exponential parameter is increasing as well, so the power loading (*W*/*P*) value is also increasing. Consequently, any value of S_{TO} greater than the one specified does not satisfy the take-off run requirements, and the region above the graph is not acceptable.

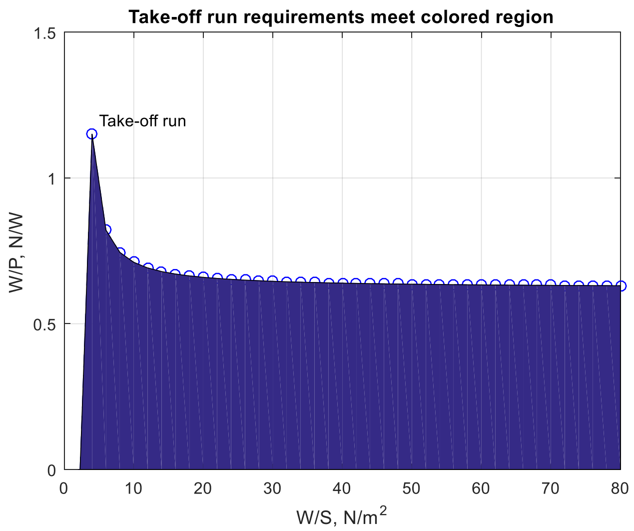

Fig. 3. The take-off run requirements meet the colored region.

1.4. **Rate of climb**. All aircraft configurations should be in conformity with the defined rate of climb (*ROC*) requirements. The rate of climb requirements are defined in the certification specification. In EASA CS VLA 65, it is defined that *ROC* should not be less than 2 m/s [4], [5]. As the maximum *ROC* value can be reached at sea level, the value "air density ρ" at sea level (1.225 kg/m^3) is used for the calculations.

The rate of climb is a function of engine power and propeller efficiency; for altitude gain, the determined propeller efficiency coefficient (η_P) is equal to 0.55 and the determined lift-to-drag ratio $(L/D)_{max}$ is equal to 11.5 [11]. Thus, the matching plot equation is the following:

$$\left(\frac{W}{P}\right)_{ROC} = \frac{1}{\dfrac{ROC}{\eta_P} + \sqrt{\dfrac{2}{\rho \times \sqrt{\dfrac{3 \times C_{D_0}}{K}}} \times \left(\dfrac{W}{S}\right)} \times \left(\dfrac{1.155}{(L/D)_{max} \times \eta_P}\right)} =$$

$$= \frac{1}{3.6363 + \sqrt{1.0969 \times \left(\dfrac{W}{S}\right)} \times 0.1826} \frac{N}{W} \tag{17}$$

The acceptable region, which satisfies the take-off run requirements, is determined by analyzing the equation. The *ROC* value is a denominator in the equations, so when the rate of climb is increasing, the value of power loading (*W*/*P*) is decreasing. Consequently, any value of *ROC* greater than the one specified complies with the rate of clime requirements, and the region below the graph is acceptable.

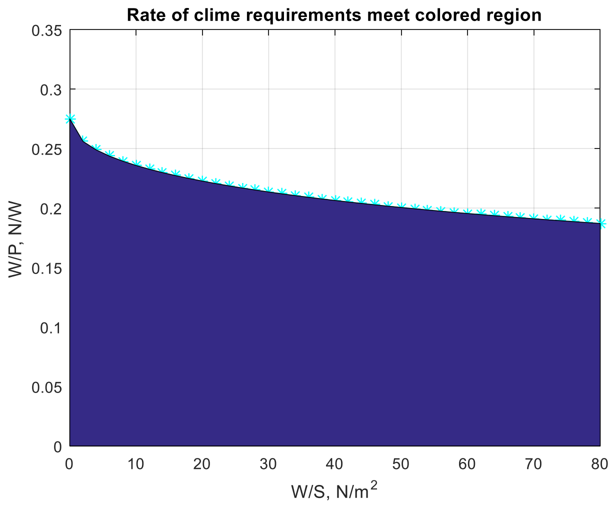

Fig. 4. The rate of climb requirements meet the colored region.

1.5. **Ceiling**. It is generally defined for several types of ceiling:

1) absolute ceiling – h_{ac}. The absolute ceiling is an altitude where the aircraft flight *ROC* is zero;
2) service ceiling – h_{sc}. The service ceiling is an altitude where the aircraft flight *ROC* is 0.5 m/s;
3) cruise ceiling – h_{cc}. The cruise ceiling is an altitude where the aircraft flight *ROC* is 1.5 m/s;
4) combat ceiling – h_{cc}. The combat ceiling is an altitude where the fighter can take altitude with a speed of 5 m/s. This altitude is defined only for combat aircraft.

The above mentioned can be summarized as follows:

ROC_{AC} = 0.0 m/s;
ROC_{SC} = 0.5 m/s;
ROC_{CrC} = 1.5 m/s;
ROC_{CoC} = 5.0 m/s.

The matching plot can be created by using the same equation (16) as for the calculation of *ROC*. In this design phase, when the air vehicle has not been completely designed, the following approximation should be used:

$$P_C = P_{SL}\left(\frac{\rho_C}{\rho_0}\right) = P_{SL}\sigma_C. \tag{18}$$

Then (16) can be rewritten in the following form:

$$\left(\frac{W}{P_{\mathrm{SL}}}\right)_{\mathrm{C}} = \frac{\sigma_{\mathrm{C}}}{\dfrac{ROC_{\mathrm{C}}}{\eta_{\mathrm{P}}} + \sqrt{\dfrac{2}{\rho_{\mathrm{C}}\sqrt{\dfrac{3C_{D0}}{K}}} \cdot \left(\dfrac{W}{S}\right)} \left(\dfrac{1.155}{(L/D)_{\max}\,\eta_{\mathrm{P}}}\right)}. \tag{19}$$

And for the absolute ceiling, it can be rewritten as follows:

$$\left(\frac{W}{P_{\mathrm{SL}}}\right)_{\mathrm{AC}} = \frac{\sigma_{\mathrm{AC}}}{\sqrt{\dfrac{2}{\rho_{\mathrm{AC}}\sqrt{\dfrac{3C_{D0}}{K}}} \cdot \left(\dfrac{W}{S}\right)} \left(\dfrac{1.155}{(L/D)_{\max}\,\eta_{\mathrm{P}}}\right)}. \tag{20}$$

According to the design requirements, the air vehicle should be operational at 350 m above ground. Therefore, for the preliminary design phase calculations, a cruise ceiling of 350 m above sea level is accepted with ROC_{CrC} = 1.5 m/s, and equation (18) is used for the calculation of the matching plot:

$$\left(\frac{W}{P_{\mathrm{SL}}}\right)_{\mathrm{C}} = \frac{0.60082}{1.875 + \sqrt{1.8256\left(\dfrac{W}{S}\right)}\,0.1255} \frac{N}{W}. \tag{21}$$

The acceptable region, which satisfies the take-off run requirements, is determined by analyzing the equation. The ROC and ρ_{C} values represent a denominator in the equations, so when the altitude is increasing, ρ_{C} is decreasing, and the relative air density is decreasing as well. Whereas ρ_{C} value is in denominator of denominator, in case of altitude increasing, power loading (W/P) is decreasing. Consequently, any altitude greater then defined h_{c} altitude satisfies the ceiling requirements, and the region below the graph is acceptable.

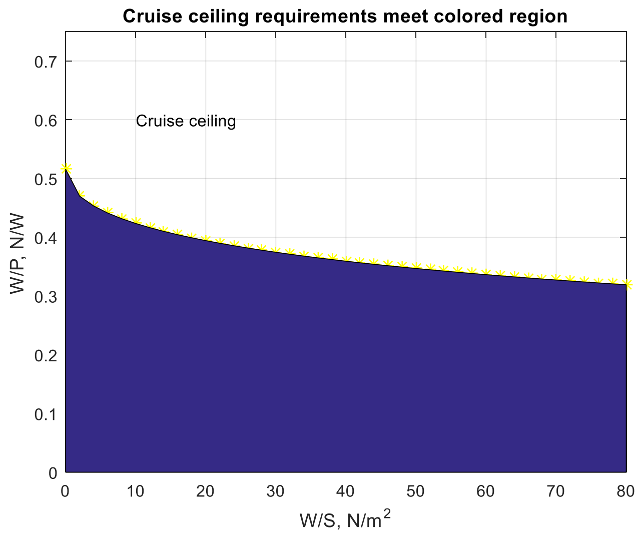

Fig. 5. The cruise ceiling requirements meet the colored region.

2. The matching plot is made on the basis of the obtained results.
The MATLAB code for the matching plot construction is as follows:

```
% The following data are obtained from the design requirements:
Vs = 8.5;          % The stall speed according to the requirements in
                   % certification specification EASA CS VLA, m/s
Vc = 26.0;         % The cruising speed according to the design requirements,
m/s
Vmax = 33.8;       % Calculated maximum speed, m/s
Vto = 2.915;       % Calculated take-off speed, m/s
Vto = 11.05;
Vr = Vto;          % Take-off rotation speed, m/s
hc = 350;          % Normal service altitude/ceiling above sea level, m
hac = 5000;        % Absolute ceiling altitude, m
Clmax = 1.6;       % Maximum lift coefficient for the preliminary design phase
e = 0.8;           % Oswald efficiency factor
AR = 12;           % Wing aspect ratio for the preliminary design phase
K = 0.0331741;     % Calculated induced drag coefficient
g = 9.81;          % Gravitational acceleration, m/s^2
Cd0 = 0.0245;      % Zero lift-drag coefficient
Cd0to = 0.0835;    % Zero lift-drag coefficient at take-off
Clto = 0.85;       % Aircraft lift coefficient at take-off
Cdto = 0.10747;    % Aircraft drag coefficient at take-off
Cdg = 0.03947;     % Coefficient
Clr = Clto;        % Lift coefficient at take-off rotation
nu = 0.08;         % Drag coefficient for the launch unit
Sto = 2;           % Launch unit length
rhosl = 1.225;     % Air density at sea level
rhoc = 1.184;      % Air density at a cruising altitude of 350 m above sea level
rhoac = 0.736;     % Air density at absolute ceiling altitude
mupto = 0.55;      % Propeller efficiency coefficient at take-off
mupac = 0.8;       % Propeller efficiency coefficient at cruising altitude
LDmax = 11.5;      % Lift drag value for the preliminary design faze
ROCAC = 0;         % Rate of climb at absolute ceiling, m/s
ROCSC = 0.5;       % Rate of climb at service ceiling, m/s
ROCCrC = 1.5;      % Rate of climb at cruise ceiling, m/s
ROCCoC = 5;        % Rate of climb at combat ceiling, m/s
% Stall speed.
WS = 1/2*rhosl*Vs^2*Clmax;
x1 = WS;
x2 = WS;
y1 = 0;
y2 = 1.5;
plot([x1,x2],[y1,y2],'-g')
text(55,1.2,'Stall speed')
axis([0 80 -0.5 1.5])
xlabel('W/S, N/m^2')
ylabel('W/P, N/W')
grid on
hold on
% Maximum speed.
WSms = 0:2:80;
WPvmax =
mupac./((0.5*rhosl*Vmax^3*Cd0./WSms)+(((2*K)./(rhoc*(rhoc/rhosl)*Vmax)).*WSms)
);
plot(WSms,WPvmax,'--r')
text(10,-0.05,'Maximum speed')
% Take-off run.
WPsto = (((1-exp(0.6*rhosl*g*Cdg*Sto)./WSms))./(nu-
(nu+Cdg/Clr).*(exp(0.6*rhosl*g*Cdg*Sto)./WSms))).*(mupto/Vto);
disp(WPsto)
plot(WSms,WPsto,'b--o')
```

```
text(5,1.2,'Take-off run')
% Rate of Climb.
WProc = 1./(3.6363+(sqrt(1.0969.*WSms)*0.1826));
plot(WSms,WProc,'*-c')
text(5,0.3,'Rate of clime')
% Cruise ceiling.
WPslc =
(rhoc/rhosl)./((ROCCrC/mupac)+sqrt((2/(rhoc*sqrt(3*Cd0/K)))*WSms)*(1.115/(LDma
x*mupac)));
plot(WSms,WPslc,'*-y')
text(10,0.5,'Cruise ceiling')
```

The matching plot is as follows:

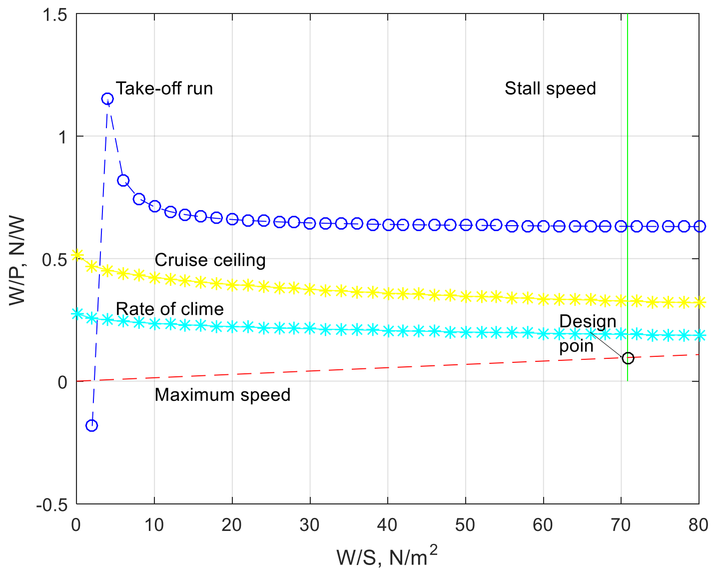

Fig. 6. Matching plot with a design point.

3. Inside the matching plot graph, an acceptable region conforming to all air vehicle performance requirements is identified. The acceptable region can be recognized by V_{max} value, which changes inside the acceptable region. As mentioned before, the acceptable regions for all the plots are situated below the graph and on the left side of the stall speed graph. Whereas V_{max} graph is the lowest, all air vehicle requirements will satisfy the region between V_{max} graph and V_s, *ROC*, S_{TO}, h_c graphs that shows lowest engine power point "Design point".
4. To define a design point. As it was mentioned above, there is only one design point, and it shows the lowest engine power. Consequently, the design point is in the intersection of V_{max} and V_s graphs. The MATLAB code for the intersection point and coordinate determination is as follows:

```
% Find design point coordinates
[xint,yint] = polyxpoly([x1,x2],[y1,y2],WSms,WPvmax);
plot(xint,yint,'ok')
text(65,-0.2,'Design point')
disp([xint,yint])
```

5. The design point makes it possible to obtain two parameters: corresponding wing loading $(W/S)_d$ and power loading $(W/P)_d$.

$$(W/S)_d = 70.805;$$
$$(W/P)_d = 0.0963.$$

6. The wing reference area and engine power are calculated from the obtained values by the following equations:

$$S = \frac{W_{\mathrm{TO}}}{(W/S)_{\mathrm{d}}} =\sim 0.98\ \mathrm{m}^2; \tag{22}$$

$$P = \frac{W_{\mathrm{TO}}}{(W/P)_{\mathrm{d}}} = 719.834\ \mathrm{W} =\sim 0.952\ \mathrm{hp}. \tag{23}$$

IV. Conclusion

The obtained results are feasible for the class of the system and prove that the Systems Engineering Approach method can be used for designing the air vehicles of micro systems as well as for other aircraft [12], [15]–[18]. The system should be regularly applied to obtain more stable results.

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