

A New Performance Based Preliminary Design Method for Electrical Powered MAV and UAV Aircrafts

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NOMENCLATURE

C_{D0} : Aircraft Drag at Zero Lift

C_L : Lift Coefficient of Aircraft

K : Constant

n : Load Factor

q : Dynamic Pressure

R : Pull-up (pull-down) Radius

r : Turn Radius

S : Wing Area

T : Take-off Thrust

V : Linear Velocity

W : Take-off Weight

λ : Thrust Reduction Coefficient

θ : Climb Angle

θ_i : Pull-up or Pull-down Angle

ρ : Density

δ : Coefficient

ϕ : Roll Angle

$\frac{dV}{dt}$: Linear Acceleration

$\frac{dh}{dt}$: Rate of Climb

ABSTRACT

A new approach, using aircraft flight performance parameters, is presented for design point selection of electrical micro and small unmanned air vehicles (MAV & SUAV). Performing maneuvers during the aircraft's mission, have been considered as the required design criteria. The aircraft's motion equations have been derived for some general maneuvers. Applying the performance parameters

in these equations results in the aircraft's matching diagram. Therefore, one may determine the allowable design space to find the aircraft's design point. This method has been employed to design a micro air vehicle. The results are used to compare the effects of various performance parameters and different maneuvers on the near optimum selection of design point for aircraft.

1-THE AIRCRAFT (W-T-S) COMBINATION

The two dimensional orthogonal space, $(T/W)_{TO}$ [or $(W/P)_{TO}$] as the vertical axis versus $(W/S)_{TO}$ as the horizontal one, is known as the aircraft design space [1]. Determining the combination of the take-off weight, engine thrust and wing area of an aircraft (or selection of the aircraft design point) in the design space, is the cornerstone for a new aircraft design cycle process. The aircraft design point selection has an important role in the analysis of the various aircraft's characteristics, the overall configuration and the aircraft sizing process in the aircraft preliminary design cycle [2]. Also it has its special importance to design some new generation of aircrafts like unmanned air vehicles, whose design standards have not quite been determined and classified yet[2].

Due to special disciplines which govern the design of the aforementioned aircrafts, it is very hard and sometimes impossible to design them with the standards which are used in conventional and classical aircraft designs [2, 3].

Reference [4] has presented a method to determine the aircraft's design point selection on the basis of the aircraft's energy.

As a new method and from a different point of view, in the present work, it is shown that how one may determine the electrical unmanned aircraft's preliminary design point, concerning the various required maneuvers at its operation. This is accomplished with the substitution of some predetermined performance parameters in the aircraft's motion equations. Doing so, ensures that the aircraft will be able to accomplish its required operational maneuvers during its mission.

2-THE PERFORMANCE BASED APPROACH TO THE UNMANNED ELECTRICAL AIRCRAFTS DESIGN

A brief introduction

According to the aircraft mission, different flight conditions have been considered to determine the design point of an unmanned electrical aircraft. The variety of situations contains the cruise, climb, turn, pull-up and pull-down maneuvers.

For every flight state, the motion equation of the aircraft has been written. Then the required aircraft's predetermined performance parameters are substituted in the motion equations.

Then by expanding the motion equations, (T/W) versus (W/S) for every situation in the aircraft's flight phases are determined.

Concerning the substitution of the required performance parameters, in the various equations of flight situations, in fact; the equations become the criteria for aircraft's design point selection. Regarding this matter automatically satisfies the design requirements to fulfill the desired operation for the aircraft at its mission.

A simultaneous use of the equations in the aircraft matching diagram determines the allowable space for the design process. Then it may be possible to select the proper aircraft's design point to empower it to accomplish its mission.

The design point is a point where the aircraft's (T/W) and (W/S) are selected in such a way that, near optimum values for the engine thrust and wing area are selected and at the same time, all the aircraft's mission requirements are satisfied.

Having the aircraft's design point and take-off weight [3], leads to the calculation of the aircraft engine thrust and wing area which, in turn, are used to select the engine (engines) and manufacture the aircraft's structure.

It should be mentioned that, the considered aircraft flight maneuvers, based on the presented method may be expanded. In this study, only some simple maneuvers are considered, however, more complex situations could be taken into account to improve the aircraft ability to accomplish more complex missions.

3- THE AIRCRAFT MOTION EQUATIONS

In this study, it is assumed that the mass of the aircraft is concentrated at its center of gravity and all lift, drag and thrust forces are acting on this point. Aircraft rigid body motion during the flight has not been considered. It has also been assumed that the thrust force is always in the direction of the drag force, and both of them are in the flight path direction. The engine installation angle is assumed to be zero and a no cross wind situation is considered. It should be mentioned that all the aircraft maneuvers are assumed to stay in the vertical or horizontal planes. Also there is no thrust vectoring, morphing or flexible configuration.

The motion equation for an aircraft obeys the Newton's second law:

$$\vec{F} = m \vec{a} \dots \dots \dots (1)$$

The above equation is a vector equation and could be applied in any directions. We now present the aircraft's motion equation and a schematic of its

situation for different flight phases or maneuvers. All variables are defined in the nomenclature. Also one may consult [5, 1] for more details.

1- Constant Speed Cruise in a Horizontal Plane

Figure 1 shows an aircraft during a constant speed cruise flight. Considering the figure, the motion equation of the aircraft may be written as:

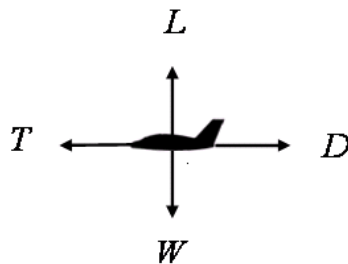


Figure 1- Cruise in a Horizontal Plane

$$T - D = \frac{W}{g} \frac{dV}{dt} \dots\dots\dots(2)$$

Since there is no acceleration at this flight state, the equation is simplified to:

$$T - D = 0 \dots\dots\dots(3)$$

$$T = D \dots\dots\dots(4)$$

The lift of the aircraft may be obtained from:

$$L = nW = C_L q S \dots\dots\dots(5)$$

Where n is the load factor which in this flight situation is 1. The lift coefficient of the aircraft may then be determined by:

$$C_L = \frac{L}{qS} = \frac{nW}{qS} \dots\dots\dots(6)$$

And the drag force of the aircraft is:

$$D = C_D q S \dots\dots\dots(7)$$

If the drag polar equation of the aircraft is assumed to be:

$$C_D = C_{D_0} + K C_L^2 \dots\dots\dots(8)$$

Combining (6) and (8) leads to:

$$C_D = C_{D_0} + K \left(\frac{nW}{qS} \right)^2 \dots\dots\dots(9)$$

So the aircraft's drag equation becomes:

$$D = C_D q S = \left(C_{D_0} + K \left(\frac{nW}{qS} \right)^2 \right) q S \dots\dots\dots(10)$$

Combining (4) and (10) leads to:

$$T = C_{D_0} q S + \frac{K n^2 S}{q} \left(\frac{W}{S} \right)^2 \dots\dots\dots(11)$$

And hence;

$$\frac{T}{W} = \frac{C_{D_0} q}{\left(\frac{W}{S} \right)} + \frac{K n^2}{q} \left(\frac{W}{S} \right) \dots\dots\dots(12)$$

It is also assumed that the aircraft's installed thrust decreases with an increase of the aircraft's altitude and speed as follows:

$$T = \lambda T_{SL} \dots\dots\dots(13)$$

Where, λ is a reduction coefficient which has a quantity smaller than 1 based on the altitude and speed of the aircraft. Using Equation (13) in (12), one may obtain the motion equation for this flight state of the aircraft as follows:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S} \right)} + \frac{K n^2}{q} \left(\frac{W}{S} \right) \right] \dots\dots\dots(14)$$

2- Constant Linear Accelerated Cruise in a Horizontal Plane

The schematic figure of the aircraft's flight in this condition is similar to the previous one, except that in this case the aircraft has a linear acceleration in the flight path direction. Because of the acceleration term, the motion equation of the aircraft becomes:

$$T - D = \frac{W}{g} \frac{dV}{dt} \dots\dots\dots(15)$$

In an exactly similar way to the previous state, the motion equation for the current situation may be derived as:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K n^2}{q} \left(\frac{W}{S}\right) + \frac{1}{g} \frac{dV}{dt} \right] \dots (16)$$

3- Constant Speed Climb in a Vertical Plane

Figure 2 shows the schematic situation of the aircraft in this flight condition:

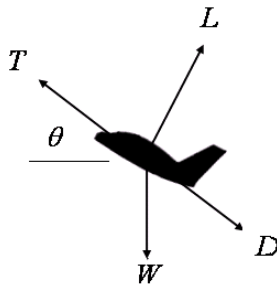


Figure 2- Climb in a Vertical Plane

Writing the motion equation of the aircraft in the flight direction leads to:

$$T - D - W \sin \theta = \frac{W}{g} \frac{dV}{dt} \dots (17)$$

Since the climb is accompanied at constant speed, we have $\frac{dV}{dt} = 0$. In this way the motion equation is simplified to:

$$T - D - W \sin \theta = 0 \dots (18)$$

$$T = D + W \sin \theta \dots (19)$$

Using Equation (10) in this equation leads to:

$$T = C_{D_0} q S + K q S \left(\frac{nW}{qS}\right)^2 + W \sin \theta \dots (20)$$

$$T = C_{D_0} q S + \frac{K n^2 S}{q} \left(\frac{W}{S}\right)^2 + W \sin \theta \dots (21)$$

$$\frac{T}{W} = \frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K n^2}{q} \left(\frac{W}{S}\right) + \sin \theta \dots (22)$$

Employing Equation (13) we reach to the following equation:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K n^2}{q} \left(\frac{W}{S}\right) + \sin \theta \right] \dots (23)$$

4-Constant Linear acceleration Climb in a Vertical Plane

The schematic figure of the flight condition in this situation is similar to that shown in Figure 2, except that in this case, the aircraft has a linear acceleration in the flight path direction. So the aircraft's motion equation has the following form:

$$T - D - W \sin \theta = \frac{W}{g} \frac{dV}{dt} \dots (24)$$

$$T = D + W \sin \theta + \frac{W}{g} \frac{dV}{dt} \dots (25)$$

Using Equation (10) and some algebraic manipulations, lead to the motion equation of the aircraft for this condition as follows:

$$T = C_{D_0} q S + \frac{K n^2 S}{q} \left(\frac{W}{S}\right)^2 + W \sin \theta + \frac{W}{g} \frac{dV}{dt} \dots (26)$$

$$\frac{T}{W} = \frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K n^2}{q} \left(\frac{W}{S}\right) + \sin \theta + \frac{1}{g} \frac{dV}{dt} \dots (27)$$

After using Equation (13), one may obtain the following relation for the aircraft's motion equation:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K n^2}{q} \left(\frac{W}{S}\right) + \sin \theta + \frac{1}{g} \frac{dV}{dt} \right] \dots (28)$$

5- Constant Speed Turn in a Horizontal Circle

Figure 3 shows the schematic of the aircraft in this flight situation:

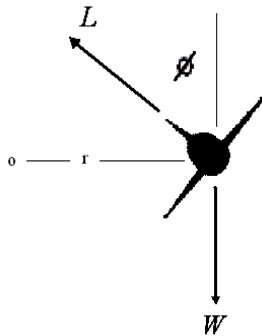


Figure 3- Turn in a Horizontal Circle

In this case, the motion equations for different directions have the following forms;

In the flight path:

$$T - D = \frac{W}{g} \frac{dV}{dt} = 0 \dots \dots \dots (29)$$

$$T = D \dots \dots \dots (30)$$

In the vertical direction:

$$L \cos \phi = W = C_L q S \dots \dots \dots (31)$$

And in the horizontal direction:

$$L \sin \phi = \frac{m V^2}{r} \dots \dots \dots (32)$$

Here the load factor which is the consequence of the maneuver is found as follows:

$$L^2 = L^2 (\sin^2 \phi + \cos^2 \phi) = L^2 \sin^2 \phi + L^2 \cos^2 \phi \dots \dots \dots (33)$$

$$L = \sqrt{L^2 \sin^2 \phi + L^2 \cos^2 \phi} \dots \dots \dots (34)$$

Finding $\sin \phi$ and $\cos \phi$ from Equation (32) and (31) and substituting them into Equation (34) and also considering $m = \frac{W}{g}$ leads to:

$$L = \sqrt{L^2 \sin^2 \phi + L^2 \cos^2 \phi} = \sqrt{W^2 \left(\frac{V^2}{gR}\right)^2 + W^2} \dots \dots (35)$$

$$L = W \sqrt{1 + \left(\frac{V^2}{gR}\right)^2} \dots \dots \dots (36)$$

And eventually we have:

$$n = \frac{L}{W} = \sqrt{1 + \left(\frac{V^2}{gR}\right)^2} \dots \dots \dots (37)$$

Substituting Equation (37) into Equation (10) leads to:

$$D = C_D q S = \left(C_{D_0} + K \left(1 + \left(\frac{V^2}{gR} \right)^2 \right) \left(\frac{W}{qS} \right)^2 \right) q S \dots \dots (38)$$

By Substituting this equation into the motion equation or Equation (30) and dividing the result by W one may obtain that:

$$\frac{T}{W} = \frac{C_{D_0} q}{\left(\frac{W}{S} \right)} + \frac{K}{q} \left(1 + \left(\frac{V^2}{gR} \right)^2 \right) \left(\frac{W}{S} \right) \dots \dots \dots (39)$$

or;

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S} \right)} + \frac{K}{q} \left(1 + \left(\frac{V^2}{gR} \right)^2 \right) \left(\frac{W}{S} \right) \right] \dots \dots (40)$$

6- Constant Acceleration Turn in a Horizontal Circle

The schematic situation in this condition is similar to that shown in Figure 3, except that in the present case, the aircraft has a linear acceleration in its circular flight path direction. So the aircraft's motion equation has the following form:

$$T - D = \frac{W}{g} \frac{dV}{dt} \dots\dots\dots(41)$$

$$T = D + \frac{W}{g} \frac{dV}{dt} \dots\dots\dots(42)$$

Similarly, one may find the motion equation as follows:

$$\frac{T}{W} = \frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K}{q} \left(1 + \left(\frac{V^2}{gR}\right)^2\right) \left(\frac{W}{S}\right) + \frac{1}{g} \frac{dV}{dt} \dots\dots\dots(43)$$

And again by using Equation (13), we get:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K}{q} \left(1 + \left(\frac{V^2}{gR}\right)^2\right) \left(\frac{W}{S}\right) + \frac{1}{g} \frac{dV}{dt} \right] \dots\dots\dots(44)$$

7- Constant Speed Pull-up in a Vertical Circle

Figure 4 shows the schematic aircraft's situation in this condition:

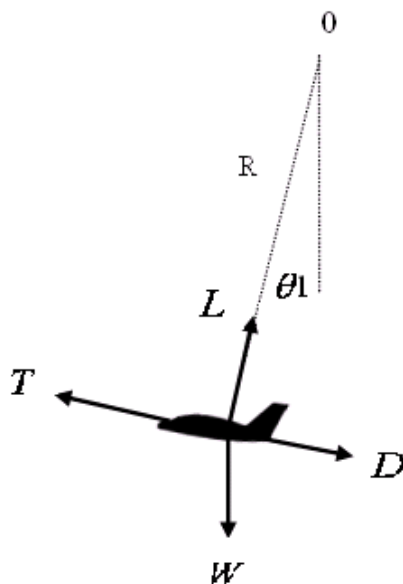


Figure 4- Pull-up in a vertical Circle

At the present situation, the aircraft's motion equations in the flight path and towards the center of the Pull-up circle are:

$$T - D - W \sin \theta_1 = 0 \dots\dots\dots(45)$$

$$T = D + W \sin \theta_1 \dots\dots\dots(46)$$

And

$$L - W \cos \theta_1 = \frac{mV^2}{R} \dots\dots\dots(47)$$

Therefore, the lift is equal to:

$$L = \frac{mV^2}{R} + W \cos \theta_1 \dots\dots\dots(48)$$

$$L = W \left[\frac{V^2}{gR} + \cos \theta_1 \right] \dots\dots\dots(49)$$

In this way the load factor is equal to:

$$n = \frac{L}{W} = \left[\frac{V^2}{gR} + \cos \theta_1 \right] \dots\dots\dots(50)$$

And the lift coefficient is equal to:

$$C_L = \frac{W \left[\frac{V^2}{gR} + \cos \theta_1 \right]}{q S} = \frac{\left[\frac{V^2}{gR} + \cos \theta_1 \right]}{q} \left(\frac{W}{S} \right) \dots\dots\dots(51)$$

By substituting the above equation in the drag polar relation or Equation (10), one may obtain:

$$D = C_{D_0} q S + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2 S}{q} \left(\frac{W}{S} \right)^2 \dots\dots\dots(52)$$

After substitution in Equation (46) and dividing by W we get:

$$\frac{T}{W} = \frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2}{q} \left(\frac{W}{S} \right) + \sin \theta_1 \dots (53)$$

And consequently we have:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2}{q} \right] \left(\frac{W}{S} \right) + \sin \theta_1 \dots (54)$$

8- Constant Acceleration Pull-up in a Vertical Circle

The schematic aircraft's situation is similar to that of Figure 4, except that in the present case, the aircraft has a linear acceleration in the Pull-up circular flight path:

$$T - D - W \sin \theta_1 = \frac{W}{g} \frac{dV}{dt} \dots (55)$$

$$T = \frac{W}{g} \frac{dV}{dt} + D + W \sin \theta_1 \dots (56)$$

Concerning the present case differences, again one may drive the motion equation relations similar to that of the previous case as:

$$\frac{T}{W} = \frac{1}{g} \frac{dV}{dt} + \frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2}{q} \left(\frac{W}{S} \right) + \sin \theta_1 \dots (57)$$

And therefore:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{1}{g} \frac{dV}{dt} + \frac{C_{D_0} q}{\left(\frac{W}{S}\right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2}{q} \right] \left(\frac{W}{S} \right) + \sin \theta_1 \dots (58)$$

9- Constant Speed Pull-down in a Vertical Circle

Figure 5 shows the schematic aircraft's state in this condition.

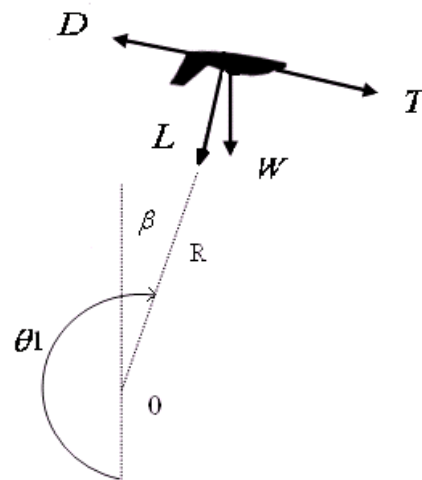


Figure 5- Pull-down in a vertical Circle

Similar to previous parts, for this case and

In the flight path we have;

$$T - D + W \sin \beta = 0 \dots (59)$$

$$\text{and} \quad \pi + \beta = \theta_1 \dots (60)$$

SO;

$$T - D + W \sin(\theta_1 - \pi) = 0 \dots\dots\dots(61)$$

$$T - D + W \sin(-(\pi - \theta_1)) = 0 \dots\dots\dots(62)$$

$$T - D - W \sin(\pi - \theta_1) = 0 \dots\dots\dots(63)$$

$$T - D - W \sin \theta_1 = 0 \dots\dots\dots(64)$$

$$T = D + W \sin \theta_1 \dots\dots\dots(65)$$

And

Toward the Pull-down center:

$$\frac{mV^2}{R} = L + W \cos \beta \dots\dots\dots(66)$$

$$\frac{mV^2}{R} = L + W \cos(\theta_1 - \pi) \dots\dots\dots(67)$$

$$\frac{mV^2}{R} = L + W \cos(-(\pi - \theta_1)) \dots\dots\dots(68)$$

$$\frac{mV^2}{R} = L + W \cos(\pi - \theta_1) \dots\dots\dots(69)$$

$$\frac{mV^2}{R} = L - W \cos \theta_1 \dots\dots\dots(70)$$

Again lift force may be found from:

$$L = \frac{mV^2}{R} + W \cos \theta_1 \dots\dots\dots(71)$$

$$L = W \left[\frac{V^2}{gR} + \cos \theta_1 \right] \dots\dots\dots(72)$$

And the load factor becomes:

$$n = \frac{L}{W} = \left[\frac{V^2}{gR} + \cos \theta_1 \right] \dots\dots\dots(73)$$

Therefore the lift coefficient is:

$$C_L = \frac{W \left[\frac{V^2}{gR} + \cos \theta_1 \right]}{qS} = \frac{\left[\frac{V^2}{gR} + \cos \theta_1 \right]}{q} \left(\frac{W}{S} \right) \dots\dots\dots(74)$$

And so the drag force from Equation (10) is found in the following form:

$$D = C_{D_0} qS + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2 S}{q} \left(\frac{W}{S} \right)^2 \dots\dots\dots(75)$$

In this way the thrust over weight ratio may be obtain as:

$$\frac{T}{W} = \frac{C_{D_0} q}{\left(\frac{W}{S} \right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2}{q} \left(\frac{W}{S} \right) + \sin \theta_1 \dots\dots\dots(76)$$

After applying the installed thrust correction factor, we get:

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{C_{D_0} q}{\left(\frac{W}{S} \right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta_1 \right]^2}{q} \left(\frac{W}{S} \right) + \sin \theta_1 \right] \dots\dots\dots(77)$$

10- Constant Acceleration Pull-down in a Vertical Circle

Except for the presence of a constant acceleration, the aircraft situation is similar to that of the previous condition. Hence:

$$\frac{W}{g} \frac{dV}{dt} = T - D - W \sin \theta_1 \dots\dots\dots(78)$$

$$T = \frac{W}{g} \frac{dV}{dt} + D + W \sin \theta_1 \dots\dots\dots(79)$$

with similar processing and substitutions, we get:

$$\frac{T}{W} = \frac{1}{g} \frac{dV}{dt} + \frac{C_{D0} q}{\left(\frac{W}{S}\right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta \right]}{q} \left(\frac{W}{S} \right) + \sin \theta \dots (80)$$

$$\frac{T_{SL}}{W} = \frac{1}{\lambda} \left[\frac{1}{g} \frac{dV}{dt} + \frac{C_{D0} q}{\left(\frac{W}{S}\right)} + \frac{K \left[\frac{V^2}{gR} + \cos \theta \right]}{q} \left(\frac{W}{S} \right) + \sin \theta \right] \dots (81)$$

Hand launching

Another important matter in the design of small and micro air vehicles is the hand launch ability at the beginning of the mission. After launching, the unmanned aircraft's velocity must be about or more than the aircraft's stall speed to prevent unwished stall. This matter restrains carrying the additional equipments and weights with aircraft's operators, so they are more agile and comfort during their operations.

The governing equation in this situation is that of the stall speed of the aircraft, hence:

$$L = W = \frac{1}{2} \rho V_s^2 C_{L_{max}} S \dots (82)$$

$$W = \frac{1}{2} \rho V_s^2 C_{L_{max}} S \dots (83)$$

$$\frac{W}{S} = \frac{1}{2} \rho V_s^2 C_{L_{max}} \dots (84)$$

4-PERFORMANCE BASED DESIGN OF UNMANNED AIRCRAFTS

Having the motion equation of the aircraft on the basis of its required performance parameters, it is possible to draw them on the aircraft's matching diagram simultaneously (Figures 8 to 11).

Then after considering the satisfaction of all requirements, it is possible to select the aircraft's design point.

The area in which these conditions are satisfied is the upper hand of the motion equation curves and the left hand of the vertical stall line on the matching diagram.

It is possible to derive a general motion equation for all considered conditions as follows:

$$\frac{T}{W} = \frac{1}{\lambda} \left[\frac{\delta n^2}{q} \left(\frac{W}{S} \right) + \frac{q C_{D0}}{\left(\frac{W}{S} \right)} + \sin \theta + \sin \theta_1 + \frac{1}{g} \frac{dV}{dt} \right] \dots (85)$$

Different parameters in this equation for different flight conditions are given in Table (1).

Parameter	n	ϕ	θ	θ_1	$\frac{dV}{dt}$	δ
Unit	-	deg	deg	deg	$\frac{m}{s^2}$	-
Constant Speed Cruise	L/W	0	0	0	0	K
Accelerated Cruise	L/W	0	0	0	$\frac{dV}{dt}$	K
Constant Speed Climb	L/W	0	θ	0	0	K
Accelerated Climb	L/W	0	θ	0	$\frac{dV}{dt}$	K
Constant Speed Turn	$\sqrt{1 + \left(\frac{V^2}{gR}\right)^2}$	ϕ	0	0	0	K
Accelerated Turn	$\sqrt{1 + \left(\frac{V^2}{gR}\right)^2}$	ϕ	0	0	$\frac{dV}{dt}$	K
Constant Speed Pull up	$\frac{V^2}{gR} + \cos \theta_1$	0	0	θ_1	0	K
Accelerated Pull-up	$\frac{V^2}{gR} + \cos \theta_1$	0	0	θ_1	$\frac{dV}{dt}$	K
Constant Speed Pull-down	$\frac{V^2}{gR} + \cos \theta_1$	0	0	θ_1	0	K
Accelerated Pull-down	$\frac{V^2}{gR} + \cos \theta_1$	0	0	θ_1	$\frac{dV}{dt}$	K

Table 1- introducing different parameters in the Equation (85) for various flight conditions

As an illustrative example, the first five rows of the Table (2) present the values of the parameters for the Ava micro air vehicle which was designed for the EMAV 2008 competition in Braunschweig, Germany. The design of the Ava MAV proved to be a successful design during the competition. This matter is some evidence that, the applied design criteria were an acceptable one.

Parameter	n	λ	θ	θ_1	$\frac{dV}{dt}$	V	q
Unit	-	-	deg	deg	$\frac{m}{s}$	$\frac{km}{h}$	$\frac{kg}{m.s^2}$
Constant Speed Cruise	1	0.7	0	0	0	90	330 at 1500 m
Accelerated Cruise	1	0.7	0	0	2	70	200 at 1500 m
Constant Speed Climb	1	0.8	8.6	0	0	60	154 at 1000 m
Accelerated Climb	1	0.8	10.3	0	1	60	154 at 1000 m
Constant Speed Turn	3 at $r=10$ m	0.7	0	0	0	60	147 at 1500 m
Accelerated Turn	3 at $r=10$ m	0.7	0	0	2	60	147 at 1500 m
Constant Speed Pull up	3.106	0.7	0	73.3	0	60	147 at 1500 m
Accelerated Pull up	3.106	0.7	0	73.3	2	60	147 at 1500 m
Constant Speed Pull down	3.83	0.7	0	360	0	60	147 at 1500 m
Accelerated Pull down	3.83	0.7	0	360	2	60	147 at 1500 m

Table 2 – the values of the parameters for the Ava micro air vehicle

5-RESULTS OF PERFORMANCE BASED DESIGN METHOD

After substitution the parameters in Equation (85), its variety is plotted in Figure (8) for various flight conditions and maneuvers used for the design of the Ava MAV. These conditions were contained Constant Speed Cruise, Accelerated Cruise, Constant Speed Climb, Accelerated Climb and Constant Speed Turn.

This diagram determines the acceptable region for the selection of aircraft's (W-T-S) combination in such a way that the designed micro air vehicle could be able to accomplish the desired flight path during its mission.

Now, suppose that the designed micro air vehicle is expected to perform more complex maneuvers.

For example, it is needed that the aircraft presents some especial performance parameters (like velocity and/or acceleration) or an especial maneuver during its flight path. To take these into account, more criteria could be considered from Table (2), for instance, which represents the parameters for the extra maneuvers (rows 6 to 10 in Table (2)). The corresponding curves for the new conditions have been plotted and have been titled to recognize, again at the Figure (8) for comparison the proper design point for every case. As one may observe, the aircraft's thrust to weight ratio increases several times as much as its initial value which was related to first simpler maneuvers case (first five rows of Table (2)).

As an example, according to the Figure (8) and at the fixed value of 0.72 (stall line) for (W/S), one may be observed that the (T/W) is about 0.85 for the first simpler maneuvers which related to constant speed turn, but it is about 2.65 for the second more complex maneuvers which related to

accelerated pull-up. It means that the growth factor of (T/W) is more than 3.

It must be mentioned that, in pull-up and pull-down maneuvers it is possible to find the angle θ_1 which maximum (T/W) occurs. This can be obtained from drawing (T/W) curve in terms of θ_1 or using its derivative. Sample of the curves were presented as followed:

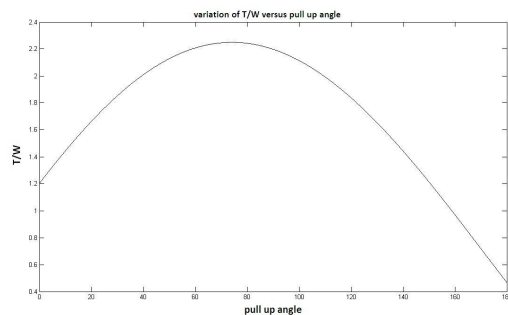


Figure 6- T/W in terms of θ_1 for pull-up

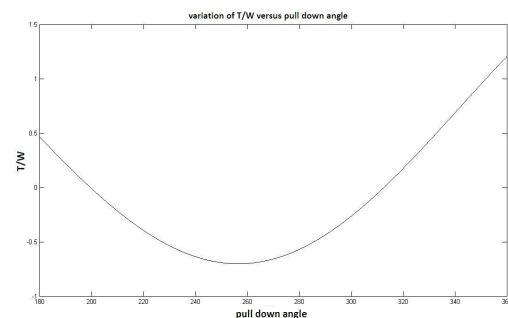


Figure 7- T/W in terms of θ_1 for pull-down

Then, it is possible to substitute this angle into the equations of motion and simultaneously plot them in the matching diagram. So, because the plotted curves are related to the maximum quantities of (T/W) , more critical conditions would be used to find the design point.

Example of the matching diagram is presented in figure 8 for mentioned conditions:

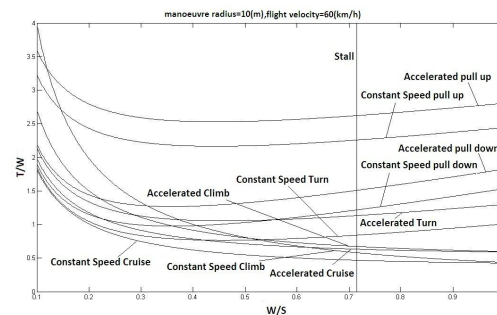


Figure 8-Ava matching diagram for maneuver radius equals to 10m and flight velocity equal to 60km/h

Also on the basis on selection of various performance parameters of aircraft, it is possible to get to different matching diagrams.

In the following, sample of such diagrams for various performance parameters were presented. It must be mentioned that, changing the matching diagram has the consequence of changing proper design point for the aircraft.

As one may observe, for a special wing area, the aircraft's design point moves up-ward with selecting more complex maneuvers or allocating some higher performance parameters to the aircraft.

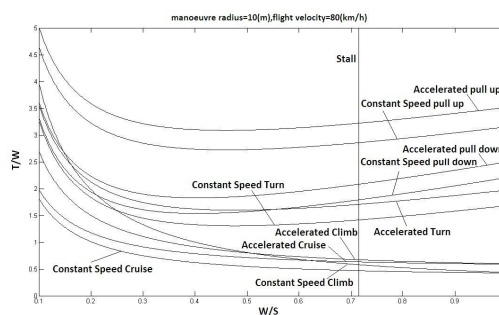


Figure 9-Ava matching diagram for maneuver radius equals to 10m and flight velocity equal to 80km/h

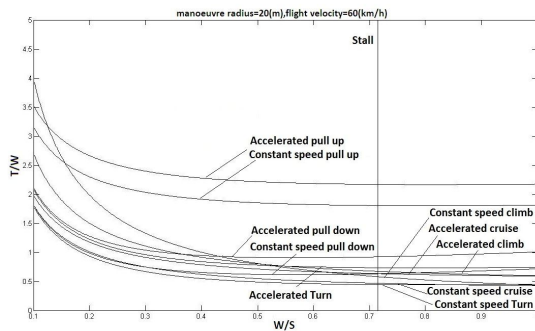


Figure 10-Ava matching diagram for maneuver radius equals to 20m and flight velocity equal to 60km/h

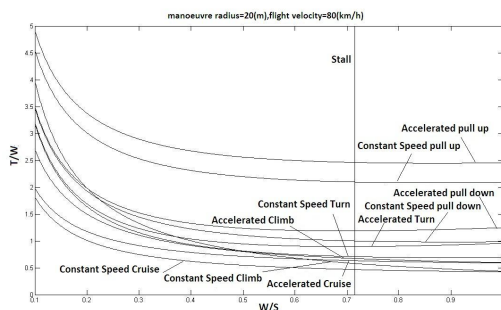


Figure 11-Ava matching diagram for maneuver radius equals to 20m and flight velocity equal to 80km/h

As a simple example one may be seen that, at the fixed value of 0.72 for (W/S), the maximum quantity of (T/W) is about 2.2 for maneuver radius equals to 20m and flight velocity equal to 60km/h (Figure (10)), but it is about 3.2 for maneuver radius equals to 10m and flight velocity equal to 80km/h (Figure (9)). It means a growth factor for the maximum (T/W) more than 1.45 with changing the maneuvers characteristics.

These matters (which explained at pages 10 to 12) cause the aircraft's thrust increasing to prevent unwished stall. It means that the aircraft need to larger and heavier engine which has a direct effect on the aircraft's design cycle and its continuation [2]. If the designer team do not respect to the above mentioned matters at the aircraft design cycle, the aircraft will not powerful enough to engage with its

mission. Many problems such as inefficiency to accomplish the desired expectations, inability to flight in some especial paths, unwanted aircraft stall problem and mission cancelation may be the consequences of thrust insufficiency and incorrect selection of design point.

The available thrust of an aircraft has a plentiful effect on its performance. Also performance has an important role in successfully accomplishment of an unmanned aircraft's mission.

So, since the performance basis design method has a comprehensive attention to the various flight maneuvers and flight parameters, it seems that, this method is a suitable one to design unmanned aircrafts.

Also it must be mentioned that, engine integration has an important effect on design cycle of unmanned air vehicles [2]. This matter finally influences on the configuration, structure, flight dynamics, flight tests and evaluation of the aircraft. Also, it must be mentioned that selection of different stall speeds may change the aircraft's wing area, which has its especial effect on the aircraft's design cycle [1]. This matter affects the allowable area of matching diagram and selection of aircraft's design point, which is also one of the design criteria at the presented design method.

It must be mentioned that, the engine thrust, the battery capacity and its weight, aircrafts endurance and range, aircraft weight and many other parameters have inter connection in aircraft design cycle [1,2,3]. Therefore it is not permitted to select an over designed engine and/or wing area in aircraft design cycle, on the contrary, these parameters must be selected optimum in a design.

Another proper characteristic of the presented method but for larger size aircrafts, is application of only performance and motion equations of aircraft to select their design point, and human standards

and factors do not apply in the this process. So, since unmanned aircrafts need some especial design method which must be harmonious with their disciplines [2], it seems that the presented method may be considered as an auxiliary one accompanying to other requirements to design unmanned aircrafts in larger sizes.

In the following for more illustration, the outcomes of the applied design method for two series of parameters have been presented for comparing the results.

The first series shows the variation of (T/W) verses radius of rotation of maneuvers, for different linear flight velocities, which have been manifested themselves by the down curves:

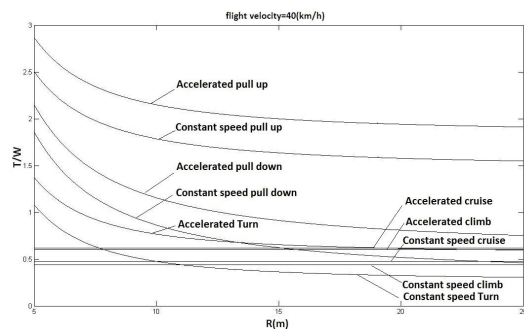


Figure 12- variation of T/W verses radius of rotation of maneuvers, for linear flight velocity=40km/h

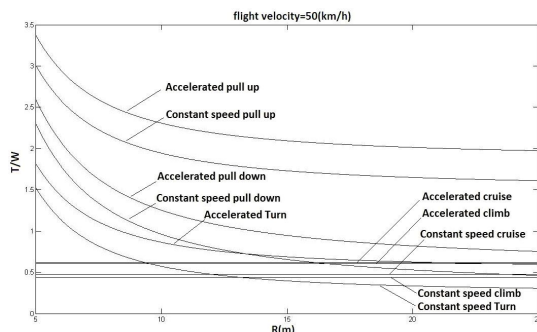


Figure 13- variation of T/W verses radius of rotation of maneuvers, for linear flight velocity=50km/h

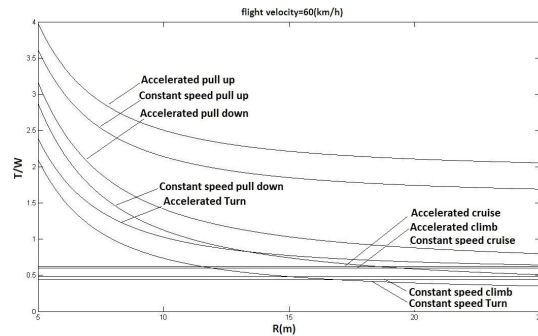


Figure 14—variation of T/W verses radius of rotation of maneuvers, for linear flight velocity=60km/h

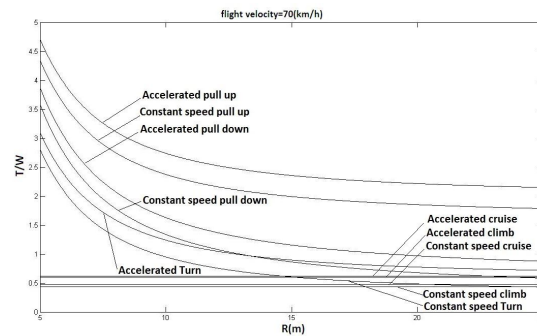


Figure 15—variation of T/W verses radius of rotation of maneuvers, for linear flight velocity=70km/h

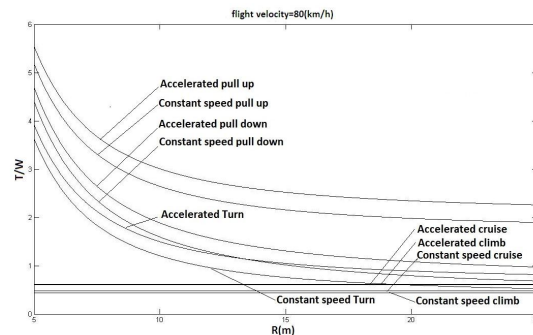


Figure 16—variation of T/W verses radius of rotation of maneuvers, for linear flight velocity=80km/h

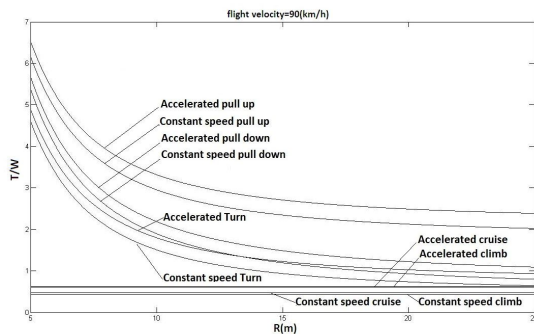


Figure 17- variation of T/W verses radius of rotation of maneuvers, for linear flight velocity=90km/h

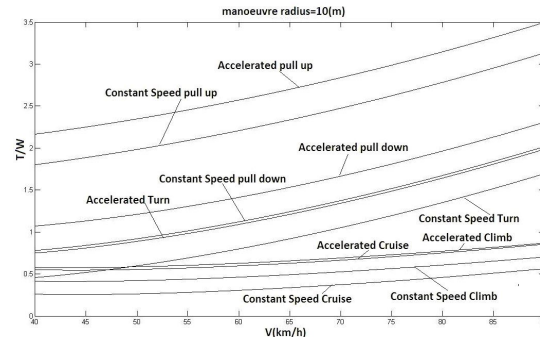


Figure 19- variation of T/W in terms of the linear flight velocity, for maneuvers radius of rotation=10m

One may observe that, (T/W) is almost constant for more simple flight phases and maneuvers with increasing the flight velocity for all cases. In the more complex maneuvers, containing rotational motions, because of increasing the load factor, (T/W) will growth with increasing the flight velocity for all cases at constant radius of rotation. This matter has a more intensity for smaller radius of rotations because of effect of load factors. The load factor relations were presented in Table (1).

The second series shows the variation of (T/W) in terms of the linear flight velocity for various maneuvers radius of rotation, which one may be observed them at the plotted curves:

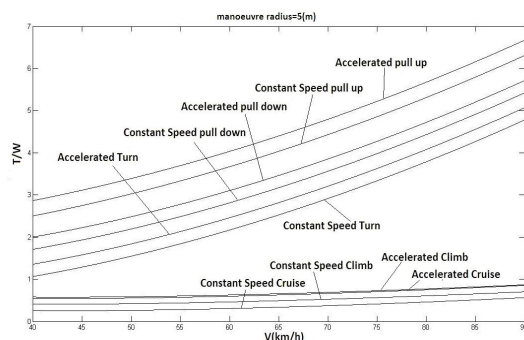


Figure 18- variation of T/W in terms of the linear flight velocity, for maneuvers radius of rotation=5m

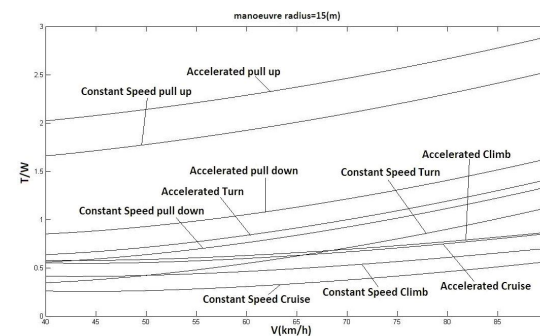


Figure 20- variation of T/W in terms of the linear flight velocity, for maneuvers radius of rotation=15m

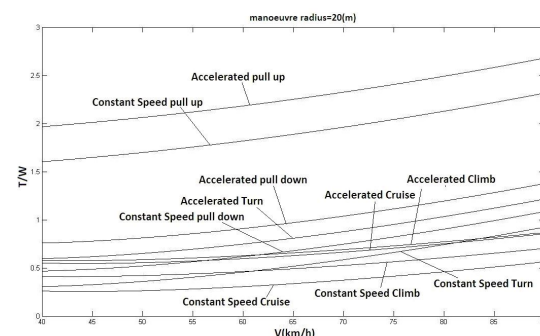


Figure 21- variation of T/W in terms of the linear flight velocity, for maneuvers radius of rotation=20m

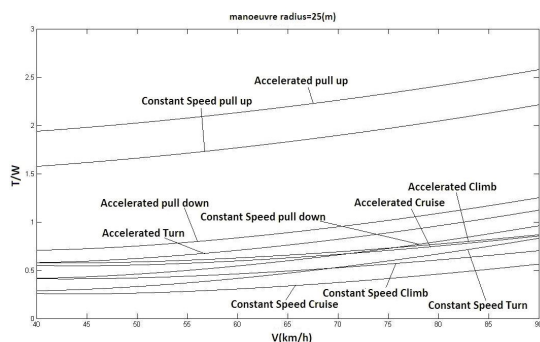


Figure 22- variation of T/W in terms of the linear flight velocity, for maneuvers radius of rotation=25m

Concerning Table (1) information, Again one may observe the effect of load factors (or maneuvers), on the plotted curves.

It should be reminded that these plots are related to a specific (W/S) of the aircraft which may be chosen in different values. So, these curves may be varied relative to the aircraft's stall speed and airfoil section of the wing.

Finally it must be mentioned that, MAVs and SUAVs are some multidisciplinary systems. So in their design cycle, all the essential design features in the form of a holistic strategy must be considered. The designer team should have an eye on all design features of the aircraft for decision making in the design process. For example the effects of low Re . number and small dimensions must be considered in the design lops at the various design cycle tasks [2].

6-CONCLUSION

A performance basis design method is presented for the micro air vehicles and small unmanned air vehicles.

The considered flight states are only some limited possible maneuvers of aircraft. They only have been used for introducing the method and its abilities. An aircraft may accomplish more complex maneuvers

in its flight path, which must be considered in its mission as the flight requirements.

Nowadays, most of the unmanned aircraft missions are performed in the autonomous phase. It seems that applying this method is suitable for autonomous MAV and SUAV flights in which the path and the required maneuvers during the flight are predetermined. This makes the aircraft able to engage with its required mission.

Also it seems that, use of this method is specially useful for designing and enabling some new generation of autonomous MAVs and SUAVs, with intercepting and attack missions to prevention of the same kind of aircrafts operation in the military fields or reducing required time and energy in the civil fields in the future.

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