DERIVATION AND APPLICATION OF A METHOD FOR FIRST-ORDER ESTIMATION OF PLANETARY AERIAL VEHICLE POWER REQUIREMENTS

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ABSTRACT

One of the most fascinating options for the future of unmanned planetary exploration is the use of planetary aerial vehicles. From the perspective of endurance, it is preferable for such vehicles to rely on non-chemical forms of propulsion, meaning that power requirements estimation becomes particularly important in design.

Presented here is a primarily analytical (as opposed to empirical) method for first-order estimation of power requirements for planetary airplanes and rotorcraft. Equations are derived and applied to examples which provide an approximate basis for comparing difficulties of flying on Venus, Earth, Mars, and Titan. Results indicate a two-order-of-magnitude difference in power requirement between Mars and Titan aerial vehicles and at least a factor of two difference in power requirement between rotorcraft and airplanes in general. Highlighted is the importance of the planet-dependent parameter $g^{1.5}/\rho^{0.5}$. System implications for solar, radioisotope thermoelectric generator, and battery power options are also considered.

1. INTRODUCTION

Over the past few years, successes of planetary orbiters and landers have drawn considerable attention to unmanned planetary exploration. The future holds promise for unmanned exploration, and one of the most exciting options for the future is the use of planetary aerial vehicles. Whereas planetary orbiters are limited in the detail they gather and traditional landers are limited in range from their landing sites, aerial vehicles offer the unique combination of regional-scale range and detailed scientific measurements.

One aerial vehicle proposal, NASA Langley's ARES design [1], utilizes a rocket to power its flight. However, as designs become increasingly ambitious, it is likely that rocket propulsion will be replaced by electric propulsion due to the expendable nature of rocket propellant. An airplane powered by solar or nuclear energy, for example, could have a lifetime only limited by the wear and tear of its components. As soon as electricity is used in lieu of chemical energy, power consumption becomes as large a design issue as mass and volume constraints.

This paper presents a method for the first-order estimation of power requirements for planetary airplanes and rotorcraft. One method proposed by [2] has already detailed such power estimation based on purely empirical considerations. Presented here is a more analytical approach to the problem, originally developed as part of a study on Titan aerial vehicles in the Georgia Tech Space Systems Design Lab.

2. AIRPLANE POWER ESTIMATION

2.1. Airplane Power Equation

In this method, necessary installed power for an airplane is estimated using the condition of steady and level flight. For this condition, thrust power required is given by Eq. 1. [3] Here, D is drag and V is velocity:

$$P_{thrust} = DV \qquad (1)$$

Clearly, this relationship alone is not sufficient to estimate the power requirement for a planetary aerial vehicle: Drag is entirely unknown and velocity may be any of a range of values. In addition, margin, efficiency effects, and non-propulsive terms are omitted.

Incorporating the terms P_{other} (for the non-propulsive power requirement), R_p (margin to account for power losses due to conversion and wire resistances), R_m (safety margin), and η (propulsive efficiency, including propeller, motor, and gearbox power losses):

$$P_{required} = (1 + R_m)(1 + R_p)\left[\frac{P_{thrust}}{\eta} + P_{other}\right] \qquad (2)$$

Also note that the following substitutions can be made for the terms of P_{thrust} (here, m is airplane mass, g is the gravitational constant, L/D is the steady flight lift-to-drag ratio, l_w is cruise wing loading, C_L is lift coefficient, ρ is atmospheric density, and S is wing planform area):

$$D = \frac{mg}{\frac{L}{D}}$$
$$V = \sqrt{\frac{2l_w}{C_L \rho}} = \sqrt{\frac{2mg}{SC_L \rho}} \tag{3}$$

In the end, required power for a planetary airplane can be estimated with Eq. 4:

$$P_{required} = (1+R_m)(1+R_p)\left[\frac{mg}{\eta \frac{L}{D}}\sqrt{\frac{2mg}{SC_L\rho}} + P_{other}\right] \tag{4}$$

Note that if drag or cruise velocity are known from the outset of a project, they can be substituted for their respective representations in Eq. 4. Also, it may occasionally prove more convenient to use the velocity expression involving wing loading as shown in Eq. 3.

2.2. Application of the Airplane Power Equation

Unfortunately, the equations developed here cannot be truly demonstrated since no airplanes have flown on other worlds to date. However, their results can be checked for reasonability, especially against the results predicted by [2]. To do so, power requirements will be estimated for a notional 300 kg aerial vehicle on Venus, Earth, Mars, and Titan.

2.2.1. Baseline Vehicle Parameter Estimation

The first step to sizing the airplane for this application is to determine wing parameters. If low-speed flight is assumed, educated guesses for certain wing parameters for this generic 300 kg airplane will lead to definitions of S, L/D, and C_L. If it is assumed that this airplane has a wing planform area (S) of 20 m^2,[1] a reasonable wing aspect ratio (AR) of 7,[2] and a reasonable cruise lift coefficient (C_L) of 0.8, then the induced drag coefficient C_{Di} can be calculated (as in Eq. 5 from [3]) and lift-to-drag ratio (L/D) can be roughly estimated if it is assumed that parasitic drag is equal to induced drag[3] plus a modest 25% margin. The calculation of induced drag also depends on the spanwise efficiency factor e, which will be taken as 0.98.[4]

$$C_{Di} = \frac{C_L^2}{\pi \cdot AR \cdot e} \tag{5}$$

Making the assumptions above, C_{Di} is 0.0297, C_{Dp} is found to be 0.0371, and thus L/D is found to be 12.0.

The final assumptions that need to be made are those of the efficiency and margin factors in Eq. 4 and the non-propulsive power requirement. Here, a propulsive efficiency factor of 0.70 is assumed (that is, if 0.90 is taken as motor efficiency [5], if 0.90 is taken as gearbox efficiency [5], and if 0.87 is taken as propeller efficiency [6]), power loss margin is taken as 25% (which includes losses due to power converters and wiring [7]), power safety margin is taken at a realistic 50%, and the non-propulsive power requirement (for flight computers, sensors, payloads, etc.) is assumed to be 150 W for this class of vehicle[5].

A summary of these assumptions is shown in Table 1. Recall that most of these parameters use notional values for a vehicle that is hardly defined (in fact the only thing defined at the outset was its mass). If further information is known about a given design problem, the appropriate parameters can certainly be replaced. Here, the information will be used to make comparisons between flight power requirements on different worlds and, later, to determine the difference between rotorcraft and airplane power requirements.

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[1] This is the same as the wing area of an early design for a 300 kg Mars airplane in [4].

[2] While a large aspect ratio would indeed be desirable, this estimate is somewhat conservative to account for anticipated aeroshell packaging constraints which will likely be present for any missions in the near future.

[3] Essentially, this assumes that the airplane is cruising at the speed for minimum drag; in other applications, it may be desirable to estimate parasitic drag independently based on the airplane geometry as described in [3]. For applications requiring high-speed flight (such as with Mars), drag due to compressibility would also need to be accounted for. In this paper, this rough drag assumption is made for the purposes of demonstration and rough comparison.

[4] Reference [3] indicates that this factor is typically between 0.98 and 1.00 for unswept wings of most practical planforms.

[5] This 150 W estimate is a variant of a non-propulsive power budget for a 300 kg Titan helicopter designed by the Georgia Tech Space Systems Design Lab. Here, 50 W is notionally allotted for payload, 60 W for attitude determination and control, and 40 W for communications and command and data handling.

Table 1. Example Airplane Parameter Assumptions.

m	300 kg	R_m	0.50
L/D	12.0	R_p	0.25
S	20 m²	η	0.70
C_L	0.8	P_{other}	150 W

2.2.2. Power Requirements on Different Worlds

Now the only parameters missing from the equation are density and gravitational constant, which vary with the planet or moon in question. This data is shown in Table 2. For purely the purposes of comparison, flight is assumed to occur at an altitude of 500 m on all worlds.

Table 2. Conditions at 500 m Altitude on Different Worlds. *Superscripts indicate source of given data.*

	Venus	Earth	Mars	Titan
Density (kg/m³)	63.2[5]	1.17[8]	0.015[9]	5.34[10]
Gravity (m/s²)	8.93[5]	9.81	3.73[5]	1.35[5]

When these final values, plus the values in Table 1, are plugged into Eq. 4, final power requirements are generated. These data are shown by the dark bars in Fig. 1. The light bars represent the purely empirical results of applying the appropriate equation from [2].[6]

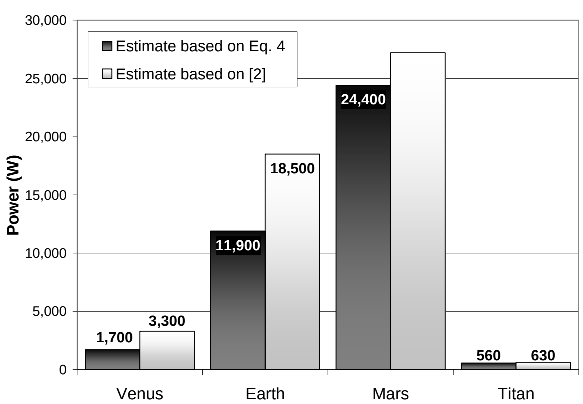

Figure 1. Approximate Power Requirements for 300 kg Airplanes.

[6] To apply the proper equation from [2], velocity must be known and is given by Eq. 3. The variable n in the Ref. [2] equation is set to 0 to represent no changes in propeller efficiency due to density (i.e. propeller disk area is scaled to compensate for density changes).

While the values differ somewhat between the two estimates (recall again that there are no real data points to verify either), it is clear that the trends are the same. The important fact to recall is that the estimate from [2] is purely empirical based on Earth aircraft (plus adjustments for gravity and atmospheric density) and the estimate from Eq. 4 is based on an analytical equation. Thus, with Eq. 4, the engineer has complete insight into the factors controlling the power estimate, whereas with [2], many factors are "black-boxed" by the method's empirical nature.

3. ROTORCRAFT POWER ESTIMATION

3.1. Rotorcraft Power Equation

With this methodology, necessary installed power for rotorcraft is estimated based on ideal actuator disk theory [3][11] for hover conditions. According to ideal actuator disk theory, the power required by a rotor is:

$$P_{thrust} = T\sqrt{\frac{T}{2\rho A}} \tag{6}$$

In Eq. 6, T is thrust, ρ is density, and A is disk area. Substituting $\frac{1}{4}\pi d^2$ for A (where d is rotor diameter) and fulfilling the hover condition with $T = mg$ (where m is mass and g is gravitational acceleration):

$$P_{thrust} = \frac{mg}{d}\sqrt{\frac{2mg}{\rho\pi}} \tag{7}$$

While this is a reasonable estimate for required hover power, it is recognized that the desired installed power estimate will also consist of non-propulsive power requirements, propulsive cruise power, and margin. Since the factor of propulsive cruise power is not as simple to estimate for rotorcraft as it is for airplanes, the final assumption in this rotorcraft methodology is an empirical one: According to [2], "Most conventional helicopters appear to have installed powers ~100% larger than predicted [by actuator disk theory applied to hover], the difference being due to rotor drag, fuselage blockage, and tail rotor power requirements, and that practical aircraft must do more than merely hover." This observation is accurate based on the data in [2] on 64 rotorcraft, shown in Fig. 2. To be more precise, this data yields a mean ratio of installed power to ideal hover power of 2.36.

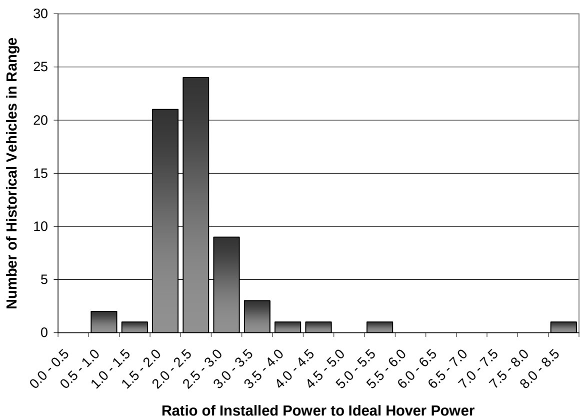

Figure 2. Historical Distribution of Rotorcraft Installed Power to Ideal Hover Power Ratios. [2]

Thus, multiplying the estimate of Eq. 7 by the empirical factor *B*, taken here as 2.36:

$$P_{required} = B\frac{mg}{d}\sqrt{\frac{2mg}{\rho\pi}} = 2.36\frac{mg}{d}\sqrt{\frac{2mg}{\rho\pi}} \quad (8)$$

3.2. Application of the Rotorcraft Power Equation

As before, the equations developed here cannot be truly verified since no rotorcraft have flown on other worlds to date. However, they can be checked for reasonability, especially against the results predicted by [2]. To do so, power requirements will be estimated for a 300 kg rotorcraft on Venus, Earth, Mars, and Titan.

3.2.1. Baseline Vehicle Parameter Estimation

Compared to Eq. 4, Eq. 8 involves few parameters. The mass of the rotorcraft is defined up front in this application, and g and ρ are properties of the world on which the craft flies. Thus, the only parameter that requires discussion is d, or rotor diameter.

Clearly there are structurally-driven upper bounds to the diameter of a rotor; however, Eq. 8 lends no insight into those bounds. If Eq. 8 is taken alone, required power can be brought to zero by increasing rotor diameter to infinity. What is desired is some relationship between rotor diameter and vehicle mass. This is provided in part by [2]: "Regression indicates that the rotor diameter scales as $m^{0.4}$". In symbols (in which K is a proportionality constant):

$$d = Km^{0.4} \quad (9)$$

Using the 64 data points provided in [2], the best-fit constant K is found to be 0.449 m/kg$^{0.4}$. About 72% of the historical data points fall within a K-value of 0.40-0.55 m/kg$^{0.4}$ (see Fig. 3). Based on the best-fit K, the diameter d is chosen as 4.4 m for this vehicle. Note that based on the typical range of K mentioned above, rotor diameters of 3.9 - 5.4 m may also be reasonable. Of course, if rotor diameter is known up-front for a particular design, it may be entered into Eq. 8 and this empirical rotor diameter estimation can be bypassed.

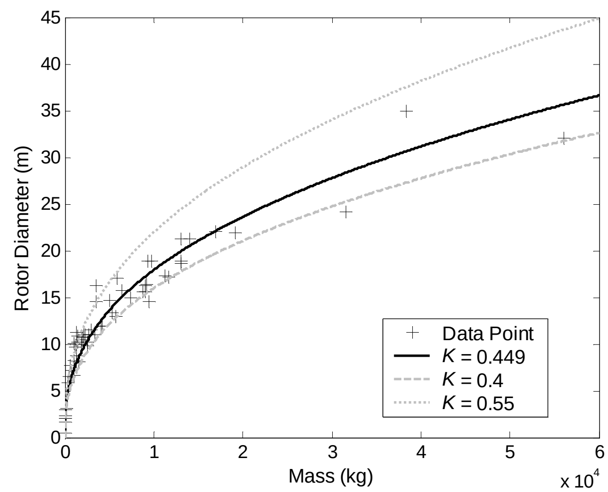

Figure 3. Historical Helicopter Rotor Diameter vs. Vehicle Mass. [2]

3.2.2. Power Requirements on Different Worlds

Using the 300 kg mass and 4.4 m rotor diameter assumption, the only parameters missing are density and gravitational constant, which are given in Table 2. In Fig. 4, dark bars indicate the results of applying Eq. 8, and light bars indicate the results of the same vehicle evaluated using the appropriate equation from [2].

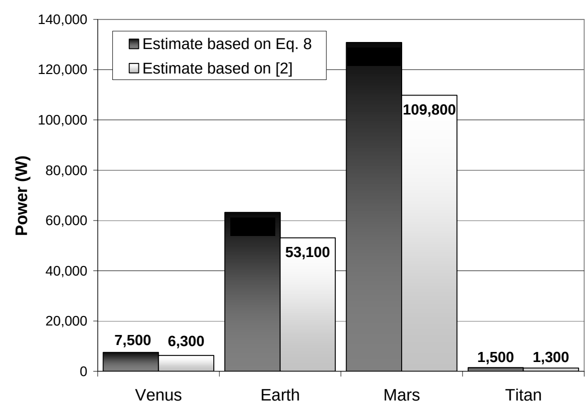

Figure 4. Approximate Power Requirements for 300 kg Rotorcraft.

While the values differ between the two estimates, it is clear that the trends are the same. Again, the estimate from [2] is purely empirical based on Earth aircraft and the estimate from Eq. 8 is based on a more analytical equation. Here, Eq. 8 has "un-black-boxed" the parameter of rotor diameter.

4. IMPLICATIONS

4.1. Comparison of Flight on Different Worlds: The Importance of $g^{1.5}/\rho^{0.5}$

In Figs. 1 and 4, clear trends exist in regards to power requirements on different worlds. Whether the vehicle is an airplane or rotorcraft or whether the estimation is analytical or empirical, the trends are remarkably similar: At 500 m on Venus, required power is about 14% of that required on Earth. On Mars, required power is about 190% of that on Earth, and on Titan the number is 3.2%. From a power standpoint, Mars is the worst choice for flight and Titan the best (to the extent that Titan is nearly 60 times easier than Mars).

The reason for this consistency in power differences among worlds is evident upon inspection of Eqs. 4 and 8. As P_{thrust} grows much larger than P_{other} in Eq. 4, airplane power varies as $g^{1.5}/\rho^{0.5}$. From inspection of Eq. 8, it is clear that rotorcraft power also varies as $g^{1.5}/\rho^{0.5}$. Very similar relationships are present in [2]. On Mars, atmospheric density is about 1% that of Earth's, but gravity is still 38% of Earth's. This results in a very large "ratio" of g to ρ. However, on Titan, density is 360% greater than on Earth and gravity is only 14% that of Earth. Fig. 5 shows the value of $g^{1.5}/\rho^{0.5}$ at 500 m altitude on Venus, Earth, Mars, and Titan. As can be seen, the trends clearly reflect those in Figs. 1 and 4.

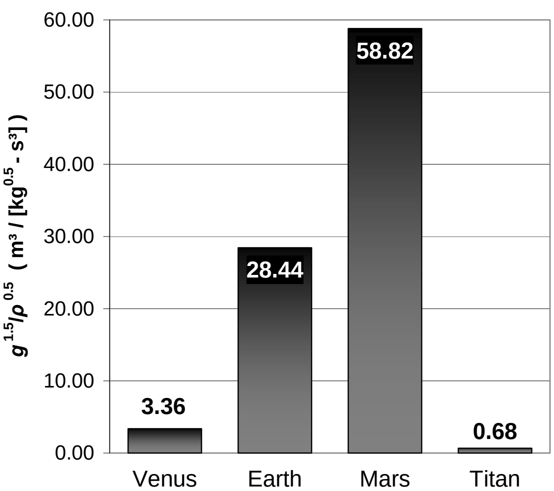

Figure 5. Magnitude of $g^{1.5}/\rho^{0.5}$ at 500 m Altitude

Note, of course, that certain practical factors are not taken into consideration in the estimates in Fig. 1 and Fig. 4. For example, the temperature at 500 m altitude on Venus is about 460 degrees Celsius[5] and the pressure is about 88 times that at sea level on Earth. This is not to say that flight on Venus is impossible; in fact, more recent Venus airplane designs have sought to fly at altitudes of 71-76 km [12]. At such altitudes, the quotient of g and ρ would be much higher due to the decreased density and virtually unchanged gravitational constant. The flight conditions would effectively become more Earth-like.

To illustrate the fact that a range of $g^{1.5}/\rho^{0.5}$ values exist on any world with an atmosphere, Fig. 6 shows the relationship between $g^{1.5}/\rho^{0.5}$ and altitude on Venus, Earth, Mars, and Titan.[7] As can be seen, to reach an Earth-sea-level-like power requirement on Venus, an aerial vehicle must fly at about 55 km altitude. Interestingly, on Titan, such a power requirement is not reached until 185 km altitude (not shown in Fig. 6). Mars-like power requirements could be simulated on Earth at an altitude of approximately 13 km.

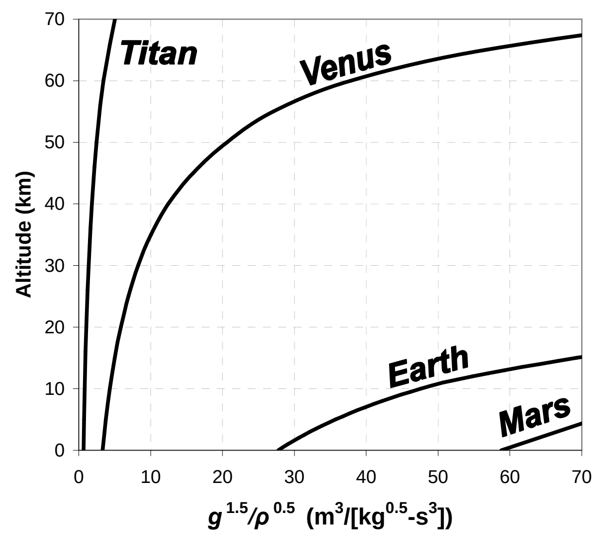

Figure 6. Relationship between Altitude and $g^{1.5}/\rho^{0.5}$

4.2. Wings or Rotors?

Common sense suggests (correctly) that it generally takes less power to fly an airplane than a helicopter. The thrust-to-weight ratio of a helicopter is necessarily greater than one, while, in contrast, the installed thrust

[7] Density profiles for Venus, Earth, Mars, and Titan are from [13], [3], [9], and [10], respectively. Though small, gravity variation with altitude is also included.

to weight ratio of a Boeing 747-400 is approximately 0.27 [14]. Figs. 1 and 4 allow benefits of planetary airplanes versus rotorcraft to be quantified.

As is clear from Fig. 7, differences exist between this paper's method and [2] in the calculation of the relative advantage of airplanes over rotorcraft. However, in both cases it is clear a significant advantage exists: In almost every instance, the rotorcraft requires at least twice as much power as the airplane. In some scenarios this may mean the difference between using a battery for limited-duration flights and using solar power for continuous flight. In others it may mean an airplane flying twice as long as a rotorcraft on the same battery.

As before, however, a design must be based on more than power requirements. An airplane will raise operational problems involved with landings and takeoffs that a rotorcraft could easily solve. The designer must weigh the alluring power advantage of an airplane against its operational disadvantages.

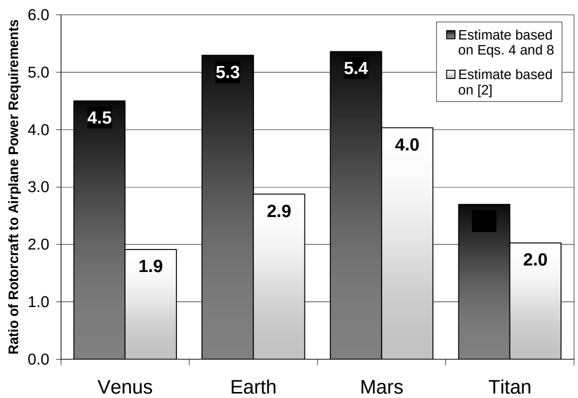

Figure 7. Relative Advantages of using Airplanes over Rotorcraft

4.3. System Implications

A valuable perspective to this study is that of what the power requirements estimated here mean in terms of the requirements and capabilities associated with the planetary aerial vehicle system. To put the example 300 kg aerial vehicle requirements into perspective, solar, radioisotope thermoelectric generator (RTG), and battery power subsystem options are sized using mass relationships from [7] and [15].

4.3.1. Solar Power Option

The sizing of the solar power option involves accounting for solar arrays, batteries, a power control unit, regulators and converters, and wiring. Solar arrays are sized taking into account solar intensity differences due to distance from the Sun and atmospheric opacity.[8] An optimistic 15% solar cell efficiency is assumed as well as an optimistic end-of-life specific performance of 47 W/kg [7]. In the cases of Earth and Mars, airplanes and rotorcraft both require regulator and converter systems which are much more massive than the vehicle itself. In the case of Titan, power requirements are very small, but the required solar array areas are on the order of a football field (see Fig. 8). Only in the case of Venus is a solar power solution nearly feasible; however, the prohibitive solar array size due to the very low 500 m altitude selected prevents this case from proving truly feasible. It should be noted, however, that in the case of Venus, higher altitudes may be selected which may increase the solar intensity by a factor more than the increase in $g^{1.5}/\rho^{0.5}$. For example, flight at 20 km altitude increases $g^{1.5}/\rho^{0.5}$ by about 77% but increases solar flux by approximately 226% relative to the 500 m values. Higher altitudes on Venus also afford a more benign thermal environment, which was not taken into account in the sizing of this system. Thus, a solar-powered aerial vehicle on Venus is not outside the realm of possibilities. Because of the very long periods of sunlight on Venus (a solar day lasts 116.8 Earth days [20]), an aerial vehicle may even be designed for a mission lasting continuously for months.

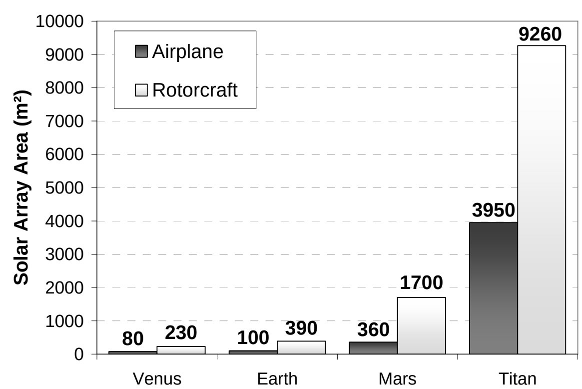

Figure 8. Required Solar Array Area for Continuous Solar Operation.

4.3.2. RTG Option

The sizing of the RTG power system option accounts for RTGs, radiators, regulators and converters, and wiring. The end-of-life output of the high-performance

[8] Solar intensity data is derived from [13], [16], [17], [18], and [19].

second-generation SRG (Stirling RTG) is assumed, and mass and power data is taken from [15]. As with the solar power option, airplanes and rotorcraft on Earth and Mars require regulator and converter systems more massive than the vehicle itself. For Venus, the number of RTGs required for continuous operation exceeds the vehicle mass for both airplane and rotorcraft. For Titan, however, it is found that an airplane is feasible and a rotorcraft is nearly feasible on RTG power alone. The power system mass breakdown for the airplane is shown in Table 3. Batteries are not included in this first-order mass budget since the vehicle can fulfill its power needs via RTGs alone. Additionally, a 50% safety margin is included in the original power estimate, and the seven SRGs actually provide 70 W more than the estimated requirement. The rotorcraft is considered nearly feasible on RTG power because its total power system mass is found to be 280.4 kg. It will be shown that this mass can be reduced via the use of batteries.

Table 3. RTG-Powered Titan Airplane Power System Mass Breakdown.

Component	Mass (kg)
RTGs (7 SRGs)	98.0
Radiator (20.8 m² area)	13.9
Regulators and Converters	14.8
Wiring	6.0
Total Power System Mass	132.7

4.3.3. Battery Options

In this final analysis, batteries are considered in combination with RTG and solar options to determine whether mission capabilities may be enhanced. Power system mass is limited to 150 kg (half the vehicle mass) and battery life is assessed. Nickel-hydgrogen batteries are assumed with a 100% depth of discharge and 45 W-hr/kg ratio of battery capacity to mass. [7]

As with the solar-only and RTG-only power options, airplanes and rotorcraft on Earth and Mars require regulator and converter systems more massive than the vehicle itself. Thus, regardless of battery mass, no 300 kg Earth or Mars aerial vehicle with the specifications given earlier will result in a closed design.

In the case of Venus, it is found that a battery-powered rotorcraft is infeasible. However, an airplane is found to be feasible, and a batteries-only power system outperforms solar-plus-battery and RTG-plus-battery systems. The resulting maximum duration for this system is found to be 90 minutes with a 12.5 km range. The mass breakdown of this power system is in Table 4.

Table 4. Battery-Powered Venus Airplane Power System Mass Breakdown.

Component	Mass (kg)
Batteries (3710 W-hr cap.)	82.4
Regulators and Converters	61.6
Wiring	6.0
Total Power System Mass	150.0

In the case of Titan, it is found that a battery-only power system is feasible for both airplane and rotorcraft. This results in a 3.5 hour rotorcraft duration and 9.8 hour airplane duration (with an 81.4 km range). These vehicles' power system mass breakdowns are shown in Tables 5 and 6. In both cases, it is found that the addition of solar arrays has a positive but negligible (less than 0.4%) effect on mission duration due to the low solar intensity at Titan. The addition of RTGs drives the airplane to the RTG-only solution but drives the rotorcraft to the battery-only solution shown below.

Table 5. Battery-Powered Titan Airplane Power System Mass Breakdown.

Component	Mass (kg)
Batteries (5810 W-hr cap.)	129.2
Regulators and Converters	14.8
Wiring	6.0
Total Power System Mass	150.0

Table 6. Battery-Powered Titan Rotorcraft Power System Mass Breakdown.

Component	Mass (kg)
Batteries (4920 W-hr cap.)	109.3
Regulators and Converters	34.7
Wiring	6.0
Total Power System Mass	150.0

5. CONCLUSIONS

The prime goal of this paper has been the presentation of a method for prediction of planetary aerial vehicle power requirements. While at least one comprehensive method has been published [2], it leaves many design parameters hidden because of its empirical roots. This paper attempts to "un-black-box" those parameters so the engineer may more fully understand the relationships involved in power estimation. The reduction of empirical considerations may also help the engineer avoid potential bias introduced from necessarily Earth-based empirical data.

In the case of airplane power estimation, this method has succeeded in removing all empirical variables: Eq. 4 is based solely on efficiencies, margins, and variables with physical meaning. For rotorcraft, empirical data is still used, but the design engineer can now observe the effect of rotor diameter in addition to that of mass, gravity, and density variations. Ideally, the rotorcraft equation would also avoid such empirical parameters.

The power estimation equations derived here represent a simple but useful method for conducting top-level studies and comparisons of planetary aerial vehicles. Two of the most important results that can be drawn from these equations involve the quantification of the power advantages to using airplanes instead of rotorcraft and to flying on Titan instead of Mars, Earth, or Venus. Unlike purely empirical equations, these equations are flexible in the sense that the designer has the freedom to change vehicle parameters to essentially adjust his vehicle design to modify his power requirement. From this power requirement, power sources can be selected, battery life can be determined, and mission capabilities can be estimated. This adds notably to the tools available to the engineer involved in preliminary planetary aerial vehicle design.

6. ACKNOWLEDGMENTS

The author would like to thank Dr. Robert Braun and the Georgia Tech Space Systems Design Lab Planetary Aerial Vehicles Team under the lead of Ravi Prakash for their continued support of this work.

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