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Design Analysis Methodology for Solar-Powered Aircraft

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A methodology for conceptual design of solar-powered aircraft is described. The method is based on traditional design methodologies adapted to address the peculiar characteristics of solar-powered aircraft. The solar propulsion notation is applied to the analysis to create constraint diagrams that have takeoff wing loading $\Omega = W_{to}/S$ as their independent variable, and the ratio of solar collector area to reference planform area $R = S_{cell}/S$ as their dependent variable. Constraints determined by the propulsion requirements for the design mission of the aircraft define a solution space on the constraint diagram. A design point is selected from within the solution space. Parametric estimates for component weights are then represented as wing loading portions. When these component wing loading portions are summed along with an expression for the required payload wing loading portion, they are equated to the design point wing loading. The resulting equation is solved for the required reference planform area, thus sizing the conceptual design. Three typical conceptual designs are described and analyzed, demonstrating the utility of the methodology.

Nomenclature

C_D = drag coefficient
C_e = equivalent skin friction coefficient
C_L = lift coefficient
cf = ratio of average solar intensity to maximum
D = total aircraft drag
h_e = energy height
I_{max} = maximum solar intensity at altitude
K = induced drag factor
L = total aircraft lift
n = aircraft load factor, L/W
q = dynamic pressure
R = ratio of solar collector area to reference area
Re = Reynolds number
S = reference area, usually wing planform area
SP = shaft power
TP = thrust power
t = time
V = true airspeed
W = weight
β = ratio of current weight to takeoff weight
η = efficiency factor
Ω = wing loading, W/S

Introduction

SOLAR-POWERED aircraft have been the subject of several design studies over the past decade. The pioneering work by MacCready et al.[1] in human and solar-powered aviation developed and proved many of the enabling technologies. Hall et al.[2,3] studied the problem in depth, developing analysis methodologies and several conceptual designs. Recent studies by the Naval Research Laboratory,[4] Wright Research Development Center,[5] and Boeing[6] have emphasized aerodynamics and propulsion technologies. Parametric conceptual design analysis methodologies have been used very successfully by Roskam[7] and by Mattingly et al.[8] This article presents a simple methodology for quickly defining and analyzing both heavier-than-air and lighter-than-air solar-powered aircraft. The methodology combines a Roskam-style parametric formulation with solar power notation described by Morgan.[9] The result is powerful, easy to learn, and lends itself to rapid sensitivity analyses.

Formulation

The methodology is developed from conventional aircraft design analysis techniques as used by Mattingly et al. and Roskam, but specially adapted for the unique characteristics of solar-powered aircraft using relations and notation described by Morgan. The process of defining and analyzing a candidate aircraft configuration has the following steps:

1) Draw or otherwise describe the significant geometric features of the candidate configuration. The configuration must be defined in sufficient detail to establish the relative sizes of aerodynamic surfaces, and to ensure adequate internal volume exists for the required systems and payload.

2) Predict the configuration's aerodynamic characteristics. This process first requires estimating the candidate configuration's total wetted area and lifting surface characteristics such as reference planform area, aspect ratio, and taper ratio. Then, lift curves and drag polars are estimated using curve fits of aerodynamic data for existing aircraft with similar geometric and mission characteristics. The drag predictions are based on the equivalent skin friction coefficient method described by Raymer,[10] but with data for similar aircraft taken from References 1, 2, 4, and 9.

3) Construct a constraint diagram for the candidate system. The variables that are plotted on this diagram are the aircraft wing loading, $\Omega = \Omega_{TO}/S$, and the ratio of total solar collector area to reference platform area, denoted by R. Boundary lines are plotted on the constraint diagram to show combinations of wing loading and minimum values of R needed to achieve each performance capability (maximum speed, rate of climb, ceiling, etc.) required for the design mission. The master equation that is used to define these curves will be developed and described in detail in the next section. The curves are boundaries to a solution space containing combinations of R and Ω that meet all the mission requirements. A boundary is

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also plotted showing the maximum value of R achievable for the geometry of the candidate configuration. This maximum R line is an upper bound to the solution space.

4) Select a design point; values of R and Ω that fall within the solution space on the constraint diagram. In general, smaller, more efficient and less expensive designs result from choosing the higher values of R and Ω, and so the best design points are usually in the upper right corner of the solution space.

5) Solve the sizing equation for S. This equation is based on a summation of wing loading portions for structural and propulsion system components required to fly the mission. Component wing loading portions are estimated using parameters representing the state-of-the-art in structural materials, battery, motor and solar cell technologies, and propeller design. A detailed development of this equation is included below.

6) Repeat steps 1–5 as needed to converge on a final value of S. The solution for S obtained in step 5 is final and does not require iteration if Reynolds number effects are ignored. If, however, Reynolds number variations in skin friction coefficient are included, resizing will cause the configuration's drag polar to change. This in turn will require iteration to get a final converged value of S.

7) Once the sized S is determined, design takeoff weight is calculated by multiplying S by the design point wing loading. The aircraft can then be redrawn with S the correct size to ensure proper fit of required components, to verify weights and aerodynamics, and to develop the concept into a preliminary design.

Aerodynamic Analysis

Before the capabilities of a configuration can be studied, estimates must be made of the aircraft's aerodynamics. Estimates are based on parameters that were developed from empirical data for similar aircraft. For the case of lift characteristics, the equations presented by Raymer for lift curve slope, zero lift angle of attack, and drag due to lift are used as given. For zero-lift drag estimates, the equivalent skin friction coefficient method described by Raymer is well-suited to this level of analysis, however, the method requires some modification to represent the effects of the low Reynolds numbers that would be encountered by high-flying low-speed aircraft. Based on data from existing low-speed solar- and human-powered aircraft taken from Refs. 1, 2, 4, and 9, an equivalent skin friction coefficient of 0.005 is assumed for a Reynolds number of 5×10^5. Assuming that the minimum drag coefficient varies with Reynolds number approximately as the turbulent skin friction coefficient, then equivalent skin friction coefficient C_e for a configuration of interest is given by

$$C_e = 0.005(500{,}000/Re)^{0.2}$$

where Re is the candidate aircraft's planned operating Reynolds number based on mean aerodynamic chord. The aircraft's minimum drag coefficient C_{D0} is then predicted by multiplying C_e by the aircraft's estimated wetted area. The aircraft's aerodynamic characteristics are represented by a drag polar of the form

$$C_D = C_{D_0} + K_1(C_L)^2 + K_2 C_L \tag{1}$$

In the expression for the drag polar, K_1 represents the estimated parabolic variation of drag due to lift. K_2 has the effect of shifting the value of C_L for which drag is a minimum. Its value is calculated so that minimum drag occurs at an angle of attack appropriate for the airfoils and configuration being evaluated. Note that K_1 is normally negative and that as its magnitude increases, the magnitude of C_{D_0} also increases so that the minimum value of C_D remains constant.

Master Equation

Once the aerodynamic characteristics of a candidate configuration are estimated, the ability to predict its capabilities depends on representing the relationships among the various forces that act upon it. This is accomplished by developing a master equation that begins with the traditional expression for specific excess power P_s:

$$P_s = \left(\frac{P - DV}{W}\right) = \frac{\mathrm{d}}{\mathrm{d}t}\left(h + \frac{V^2}{2g}\right) \tag{2}$$

or, letting h_e represent the expression for energy height in the right-hand parentheses in Eq. (2):

$$P_s = \frac{\mathrm{d}h_e}{\mathrm{d}t} \tag{3}$$

Next, following Mattingly et al.,[8] we manipulate the traditional equations for lift and drag to express them in terms of takeoff wing loading:

$$L = nW = C_L qS \quad \text{and} \quad D = C_D qS$$

Defining $W = \beta W_{\mathrm{TO}}$, we can write

$$C_L = \frac{nW}{qS} = \frac{n\beta}{q}\frac{W_{\mathrm{TO}}}{S} \tag{4}$$

then using Eq. (1), the expression for the thrust power required can be written

$$C_D = C_{D_0} + K_1(C_L)^2 + K_2 C_L \tag{5}$$

then the expression for the thrust power required can be written

$$DV = qSV[K_1(C_L)^2 + K_2 C_L + C_{D_0}]$$

Now it is necessary to represent the thrust power available from the solar-electric propulsion system. First, assuming the aircraft is flying high enough to avoid clouds, the total solar energy incident upon the aircraft solar cells will depend on the total area of the solar cells, latitude, time of year, and the orientation of the cells. Following Morgan we define a capacity factor cf which represents the daily average of solar energy available from the sun per hour divided by the peak solar intensity. A nominal value of $cf = 0.3$ will be used for the analyses in this article. The average thrust power available TP_{avail} during a 24-h period is then given by

$$\text{Avg } TP_{\mathrm{avail}} = S \cdot R \cdot \eta \cdot I_{\max} \cdot cf$$

where $I_{\max}$ is the maximum solar intensity in watts per unit area, R is the ratio of solar cell area to reference planform area, and η is the total system efficiency given by

$$\eta = \eta_{\mathrm{prop}}\eta_{\mathrm{gears}}\eta_{\mathrm{motor}}\eta_{\mathrm{cells}}(0.45 + 0.55\eta_{\mathrm{batt}})$$

Note that the efficiency of the batteries or equivalent energy storage systems is only applied to just over half the total energy collected, because it is assumed that almost half of the energy will be consumed by the motor when collected and the other part will be stored and consumed at night. For the examples given in this report, a total system efficiency of 0.1 was assumed.

The master equation describing the problem is now written by substituting the expression for average thrust power available into the equation for specific excess power:

$$SR\eta I_{\max} cf = DV + W\frac{\mathrm{d}h_e}{\mathrm{d}t} \tag{6}$$

or, solving for R and substituting the expressions for drag and lift coefficient, we obtain the master equation

$$R = \frac{qV}{\eta I_{\max} cf}\left[K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right)^2 + K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right) + C_{D_0}\right] + \frac{\beta}{\eta I_{\max} cf}\frac{W_{TO}}{S}\frac{\mathrm{d}}{\mathrm{d}t}\left(h + \frac{V^2}{2g}\right) \qquad (7)$$

The master equation expresses R as a function of takeoff wing loading and can easily be applied to a variety of flight conditions and maneuvers. The solar cell area required to achieve various mission requirements can be determined for a range of takeoff wing loadings. This analysis is called constraint analysis and forms an important step in the conceptual design of the vehicle.

Constraint Analysis

The constraints to be analyzed depend on the mission requirements for the vehicle. For a solar-powered aircraft used as a platform for surveillance, pollution, and resources monitoring, or theater ballistic missile defense, the nominal mission requirements might include the following:

1) A sustained cruise altitude high enough to provide a wide surveillance area: A cruise altitude of 70,000 ft was set as a goal for this analysis.

2) A cruise velocity at this altitude that will, as a minimum, permit effective stationkeeping against higher than normal winds and will allow relatively quick redeployment of the vehicle with normal winds: For this analysis a cruise velocity of 100 kn = 169 ft/s was chosen as the target.

3) An absolute ceiling requirement that is well above the cruise altitude requirements: Flying at altitudes above the nominal cruise altitude will permit two favorable operational techniques to be employed. The first of these is using altitude and the variations in wind velocity as a function of altitude to optimize the ground speed and ground track of the vehicle. This capability is important in both stationkeeping and redeployment and is well-known and effectively utilized by hot air balloons. Additionally, a ceiling higher than cruise altitude permits the vehicle to gain altitude (and potential energy) during the day so that it can glide back to the nominal cruise altitude during the night. This technique allows solar energy to be used directly without the inefficiencies of battery storage. It also decreases the required battery storage capacity and battery system weight. The master equation analyzes an absolute ceiling requirement in the same way as a cruise leg (change in energy height is zero), except that the vehicle will fly at the velocity for minimum power required rather than at a prescribed velocity. For this vehicle, an absolute ceiling goal of 80,000 ft was established.

4) A turn capability at cruise altitude that enables the vehicle to fly a racetrack pattern above the surveillance area: The goal set for turn capability was to be able to sustain a 1.1-g (25-deg bank angle) coordinated turn at cruise altitude. Again, the rate of change in energy height is zero for a level turn. One simply uses the desired load factor in the first term of the master equation to ensure that there is sufficient power available to achieve the turn.

5) A rate of climb capability that enables the aircraft to reach its cruise altitude within 12 h of a sea level takeoff. This requirement can be modeled most simply by requiring a rate of climb of 1.7 ft/s at 35,000 ft. If that rate of climb approximately equals the average rate of climb between sea level and 70,000 ft, the vehicle should arrive at its cruise altitude in approximately 12 h. In this constraint the rate of change of energy height in the master equation is not zero, but equals the desired rate of climb. The climb speed is the velocity for minimum power required that for propeller-driven aircraft produces the maximum rate of climb.

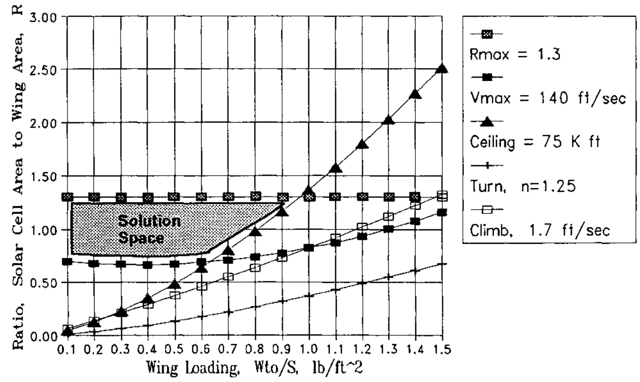

Fig. 1 Example constraint diagram.

For the constraints discussed, a nominal constraint diagram is constructed as shown in Fig. 1. In constructing this diagram two departures were made from the simple application of the master equation discussed previously. For the climb constraint, it is assumed that this climb occurs after the initial launch of the aircraft, and that the vehicle takes off during the daylight morning hours with batteries fully charged, so that the climb will be made using power from the collectors, not stored energy, and no collected energy needs to be routed to storage. For this constraint, therefore, total system efficiency can exclude battery efficiency, and capacity factor can be based on the number of hours required to climb to altitude. For the climb constraints in this article, a total system efficiency of 0.14 and a capacity factor of 0.6 are used. The same values of these two parameters are used for the absolute ceiling constraint since the aircraft would not be expected to sustain the absolute ceiling for more than a few hours.

In Fig. 1, the most restrictive constraint for very low wing loadings is the cruise speed requirement. At a wing loading of 0.9 the minimum sustained altitude constraint becomes more restrictive. The solution space is bounded above by the maximum solar cell collector area that can be achieved. A design point must be chosen that establishes the takeoff wing loading and the solar cell area that will be used to continue the analysis. The best choice for very light vehicles such as this one is typically the upper right corner of the solution space, $W_{TO}/S = 1.2$ and $R = 1.3$. Once this critical decision has been made, the design process can continue to the sizing of the vehicle.

Sizing

The sizing equation is developed by first writing parametric expressions for the weight of the various components. These weight expressions are primarily based on the presentation by Morgan. For the weight of the electric motor, propeller, and nacelle, the equation for weight is based on the maximum shaft power output $SP_{\max}$ required from the motor, which typically either occurs for the maximum rate of climb or for the maximum level speed, or as we will assume, will be equal (by choosing appropriate values for the constraints) for both cases. The shaft power requirement is in turn based on the thrust power required from the propeller and the propeller efficiency factor η_{prop}. The maximum thrust power required is expressed as the drag coefficient at the aircraft's maximum speed, multiplied by the dynamic pressure, velocity, and reference planform area. Nominal values for weight parameters for the various components were obtained from Refs. 2, 3, 4, and 9:

$$W_{\text{motor, nacelle, prop}} = \frac{SP_{\max}}{31\ \text{W/lb}} = \frac{SP_{\max}}{22.855\ \text{ft-lb/s/lb}} = \frac{(TP_{\text{req}})_{\max}}{22.855\eta_{\text{prop}}} = \frac{C_{D_V} q_{\max} V_{\max} S}{22.855\eta_{\text{prop}}}$$

Once a weight expression is written, the component wing loading portion, Ω_m for the motor, nacelle, and propeller, can be written by dividing by the reference area:

$$\Omega_m = \frac{W_{\text{motor,nacelle,prop}}}{S} = \frac{C_{D_V} q_{\max} V_{\max} S}{22.855\eta_{\text{prop}}}$$

One should note that a more conservative approach to motor weight estimates would be to size the motors to absorb the maximum power available from the collectors. This would permit higher maximum speed and rate of climb constraints (one could use a capacity factor of 1 and state that those constraints could only be achieved with the sun at zenith), but it results in heavier motors that are not operated at their most efficient power levels for most of the flight. Unless greater maximum performance is required, this approach should be avoided.

The energy storage components, be they batteries, fuel cells, flywheels or other systems, are represented parametrically by their energy density in watt-hours per pound. The energy storage system is sized by the nighttime endurance problem, which is flown at the speed for minimum power required. These conditions result in a minimum shaft power required from the motor $SP_{\min}$. For this problem, not only the efficiency of the propeller, but also the efficiency of the motor and gears in converting electrical energy into shaft power, η_{motor} and η_{gears}, must be considered. If we assume that the storage system must provide energy for 13 h of each day, and that an energy density of 100 Wh/lb can be achieved, then

$$W_b = \frac{SP_{\min}}{\eta_{\text{motor}}} \frac{13\ \text{h}}{(100\ \text{Wh/lb})} = \frac{SP_{\min}}{5.67\eta_{\text{motor}}}$$

which gives weight in pounds for shaft power required given in foot pounds per second. In terms of thrust power required

$$W_b = \frac{TP_{\min}}{5.67\eta_{\text{prop}}\eta_{\text{gears}}\eta_{\text{motor}}} = \frac{C_{D_P} q_P V_{P\min} S}{5.67\eta_{\text{prop}}\eta_{\text{gears}}\eta_{\text{motor}}}$$

and so, letting $\eta_{p,g,m} = \eta_{\text{prop}} \cdot \eta_{\text{gears}} \cdot \eta_{\text{motor}}$

$$\Omega_b = \frac{C_{D_P} q_P V_{P\min}}{5.67\eta_{p,g,m}}$$

A value for $\eta_{p,g,m} = 0.8$ is used in the examples in this article. Note that the required weight of batteries can be significantly reduced by the practice of climbing during the daytime and gliding during a portion of the night, thus reducing the number of hours of battery power required.

The wing loading portion for the solar cells or equivalent solar collectors is determined by the design point value of R and the weight per square foot of cells, which we assume is 0.07 lb/ft^2:

$$W_c = 0.07S_c = 0.07S \cdot R$$

$$\Omega_c = 0.07R$$

At this level of analysis, the weight of wire, transformers, controllers, etc., is given the subscript r and accounted for by a simple 10% addition to the weights of the other electrical components:

$$W_r = 0.1(W_{\text{batt}} + W_{\text{motor,prop,nacelle}} + W_c)$$

$$\Omega_r = 0.1(\Omega_b + \Omega_m + \Omega_c)$$

The weights of airframe components are estimated by parameters that represent both an average from historical data and a target for the structural design of the aircraft. For the examples in this article, the weight per unit area for aerodynamic surfaces is assumed to be 0.2 lb/ft^2 of planform area.[1-3,9] The weight of fuselages is taken to be 0.14 lb/ft^2 of wetted area, consistent with the relative magnitudes of historical averages for conventional aircraft fuselages listed by Raymer:

$$W_w = 0.2S$$

$$\Omega_w = 0.2$$

$$W_f = 0.14S_{\text{wet}_f}$$

or, letting R_f represent the ratio of the wetted area of the fuselage to the aircraft reference planform area:

$$\Omega_f = 0.14R_f$$

Using similar notation for the vertical and horizontal tail surfaces:

$$W_v = 0.2S_v = 0.2R_v S$$

$$\Omega_v = 0.2R_v$$

$$W_h = 0.2S_h = 0.2R_h S$$

$$\Omega_h = 0.2R_h$$

Note: weights of controls are included in surface weight.

Once all aircraft components except the fixed load have been represented as wing loadings, the sizing equation is written by summing the wing loadings plus the fixed load divided by reference area and equating them to the design point wing loading. If we assume the fixed weight of mission equipment and payload is 200 lb:

$$W_{\text{fixed}} = 200\ \text{lb}$$

$$\Omega_{\text{fixed}} = 200\ \text{lb}/S$$

then

$$\Omega = \frac{W_{\text{TO}}}{S} = \Omega_w + \Omega_f + \Omega_v + \Omega_h + \Omega_{\text{fixed}} + 1.1(\Omega_c + \Omega_b + \Omega_m)$$

$$\Omega = 0.2 + 0.14R_f + 0.2R_v + 0.2R_h + \Omega_{\text{fixed}} + 1.1\left(K_c R + \frac{C_{D_P} q_P V_{P\min}}{5.67\eta_{p,g,m}} + \frac{C_{D_V} q_{\max} V_{\max} S}{22.855\eta_{\text{prop}}}\right)$$

or, representing all the component wing loadings as a total aircraft empty wing loading, Ω_a:

$$\Omega = \Omega_a + 200\ \text{lb}/S$$

The resulting equation is then solved for reference planform area:

$$S = 200/(\Omega - \Omega_a)$$

Then, using the design point wing loading, the aircraft takeoff gross weight is given by

$$W_{\text{TO}} = \Omega S$$

It is important to note that although for simplicity in this analysis the parameters describing component weight per unit area, energy density, fixed load weight, etc., were given as specific values rather than as variables, all of these quantities can be changed to determine the impact of technology and mission requirements on aircraft size. The previous analysis also assumes that as the aircraft size changes, the relative sizes of the components stay constant. If this rule is not followed, the aircraft's drag polar will change. This situation can be handled in the same way that Reynolds number effects are included, by redrawing the configuration to its new scale and iterating through the aerodynamic, constraint, and weight analysis until a converged value of S is determined.

Case Studies

The methodology just described is applicable with some modifications to both heavier-than-air and lighter-than-air aircraft. The following four case studies of candidate configurations illustrate how the methodology is applied.

Case 1, Rectangular Flying Wing

As a first example of how the method is applied, we will evaluate a simple rectangular planform flying wing with an 8-ft chord, a 200-ft span, and twin propellers as depicted in Fig. 2. Since the airfoil for such a configuration must have zero pitching moment at the trim flight condition, the allowable camber that the airfoil can have is very small. As a consequence, the minimum drag lift coefficient for the airfoil is also quite low. On the other hand, the simplicity of the configuration results in a very low wetted area, and therefore, very low skin friction drag coefficient. The drag polar that our aerodynamic analysis method predicts for this concept is

$$C_D = 0.011 + 0.0212(C_L)^2 - 0.0039C_L$$

Using the above drag polar, a constraint diagram for the configuration is constructed for the design mission and displayed as Fig. 3. The value of R_{max} used in this analysis is based on the assumption that 95% of the flying wing's wetted area is covered with flexible solar cells, and that those cells not exposed to direct sunlight receive reflected light. The intensity of the reflected light is assumed to average 30% of the direct sunlight intensity. Note that the sustained altitude constraint for this configuration limits the maximum allowable wing loading for the aircraft to 0.85. Substituting the appropriate values into the sizing equation reveals that the available payload fraction is negative, so that an aircraft of this configuration that meets these constraints is not feasible. A sensitivity study of this constraint is shown in Fig. 4. Note that reducing the required sustained altitude to 50,000 ft allows a design wing loading of just over 1 lb/ft^2. Since the maximum

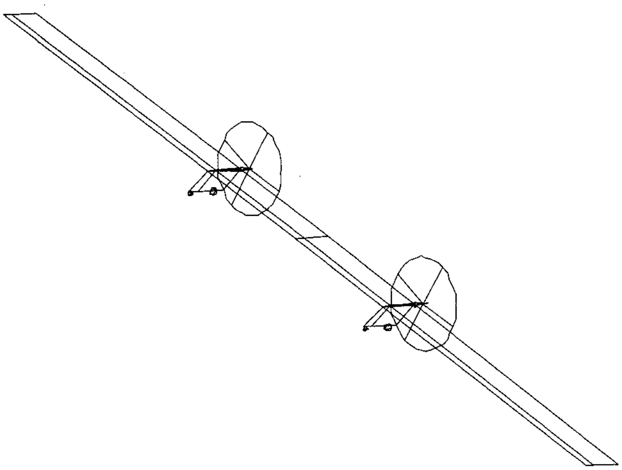

Fig. 2 Flying wing configuration.

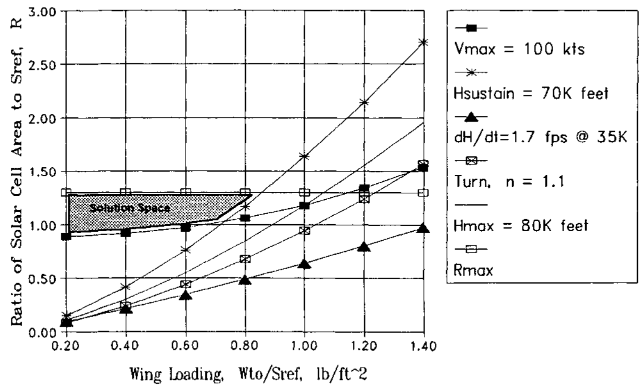

Fig. 3 Flying wing constraint diagram.

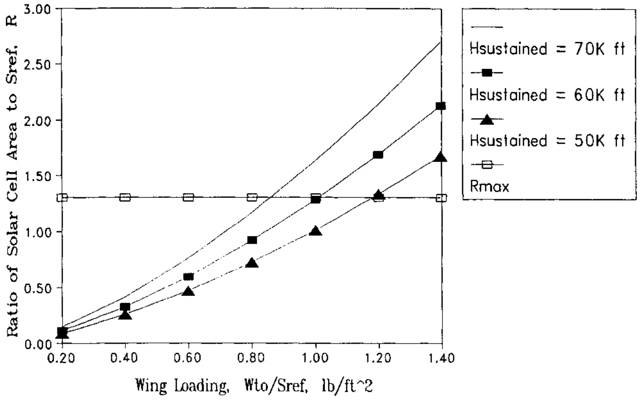

Fig. 4 Flying wing sustained altitude sensitivity.

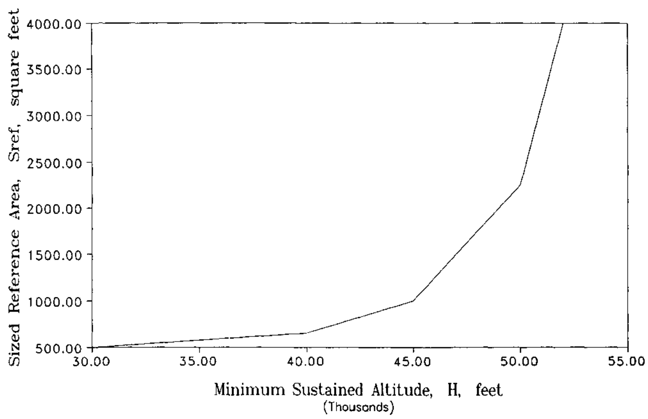

Fig. 5 Wing area sensitivity to sustained altitude.

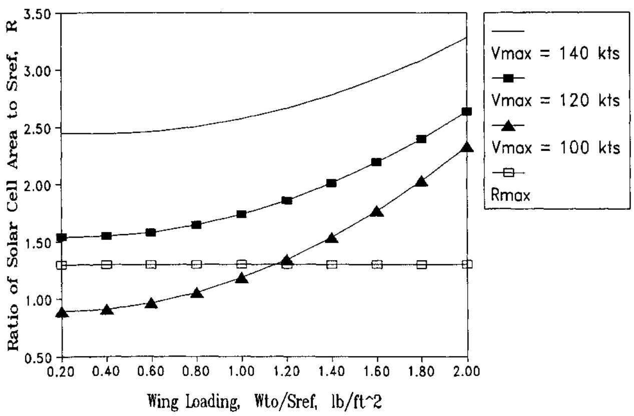

Fig. 6 Flying wing maximum speed sensitivity.

altitude constraint on Fig. 3 permits a maximum altitude above 80,000 ft at this wing loading, it appears that it would be possible for the aircraft to climb to 80,000 ft in the daytime and glide down to 50,000 ft at night, thus saving over 3 h of battery time. By reducing the required battery duration in the sizing equation to 10 h, a workable configuration is possible. Figure 5 shows the sensitivity of sized wing area to the nighttime sustained altitude for this configuration. Figure 6 is an example of sensitivity analysis for the maximum speed constraint. Note how little margin there is for any significant maximum speed increase. For the assumed component efficiencies and mission constraints, this flying wing configuration appears to be feasible only if reduced performance capabilities are accepted. Since the trim requirements of a flying wing place a rather severe constraint on choice of airfoils, it may be useful to study configurations that allow the use of much more highly cambered and possibly more efficient airfoils. The tandem wing is one such configuration.

Case 2, Tandem Wing

Consider next a tandem wing aircraft with twin pusher propellers as depicted in Fig. 7. A constraint diagram for a configuration of this sort is shown in Fig. 8. The reference area for this analysis is taken as the area of a single wing, not the sum of the two wing areas. As with the previous example, it is assumed that 95% of the aircraft wetted area will be covered with flexible solar cells which, with a 30% intensity for reflected light, yields an effective $R_{max} = 2.6$. The operational technique of climbing to 80,000 ft in the daytime and gliding to 50,000 ft before using batteries at night is assumed for this analysis.

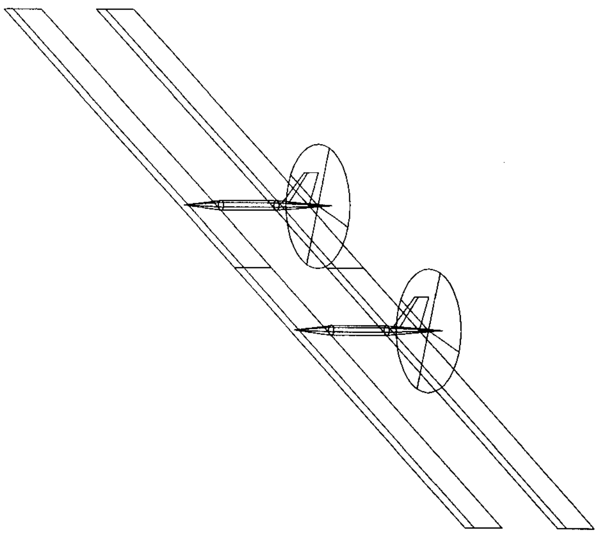

Fig. 7 Tandem wing aircraft example.

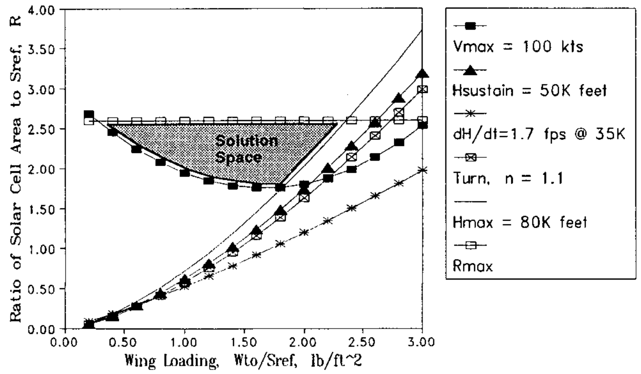

Fig. 8 Tandem wing concept constraint diagram.

While the constraint diagram looks very encouraging, sensitivity analysis reveals that only extreme aerodynamic characteristics make it so. One of the chief advantages of the tandem wing configuration over the flying wing is the ability to utilize high-lift, low-drag airfoils that typically are highly cambered. Figure 9 shows a sensitivity to airfoil camber. The analysis assumes airfoil shape is optimized for each camber value to maximize airfoil lift-to-drag ratio. Figure 10 is a sensitivity diagram showing the effect of aspect ratio on the required wing area predicted by the sizing equation. As can be seen, quite large aspect ratios are needed to obtain reasonable wing sizes, even with highly cambered airfoils. The form of the sizing equation suggests a direct proportionality between the fixed load and the sized aircraft weight. At a fixed load value of 200 lb, the sensitivity of sized aircraft gross weight to fixed load, $\partial W_{TO}/\partial W_{fixed}$ is almost 10 lb/lb. This is significantly higher than the sensitivity exhibited by the "lighter than air" craft considered next.

Case 3, Lighter than Air

The basic master equation can be applied to a lighter than air example by simply setting $C_L = 0$. Consider the semirigid airship configuration shown in Fig. 11. The design concept for such an aircraft would be to plan for it to reach its fully inflated state only when the helium inside it expands as the vehicle reaches its design altitude. The gas envelope would

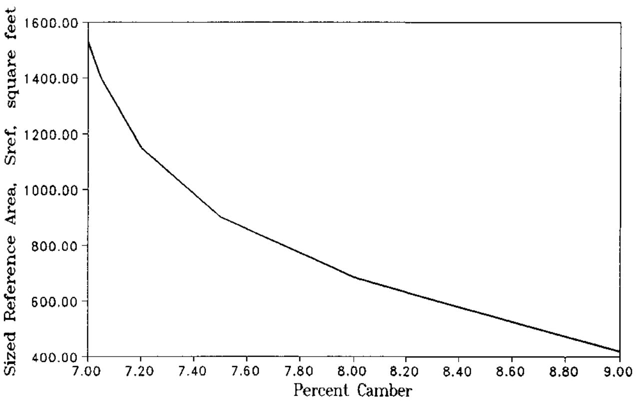

Fig. 9 Tandem wing size sensitivity to camber.

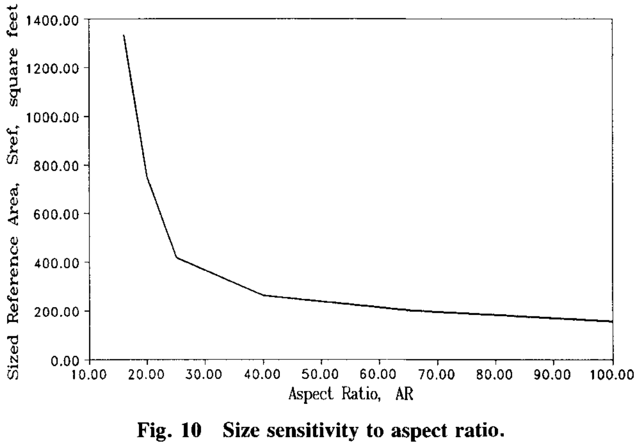

Fig. 10 Size sensitivity to aspect ratio.

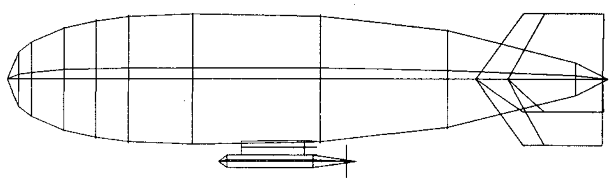

Fig. 11 Airship concept.

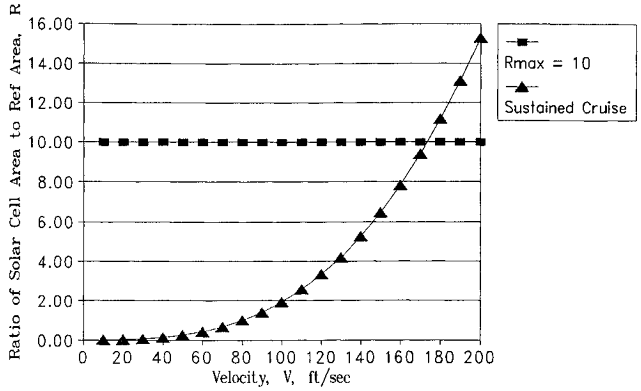

Fig. 12 Airship concept speed sensitivity.

be only slightly inflated at sea level, so the craft would have to ascend vertically using buoyant lift until reaching its design altitude. This concept is described in more detail by Ward and Taluy.[11] Because of this novel launch method, and use of buoyant lift for altitude maintenance, the analysis problem for this concept is essentially reduced to the following questions: How big does it have to be to carry 200 pounds? How fast is it? Can it keep station against winds?

The master equation, when applied to the cruise speed problem with $C_L = 0$ simplifies to

$$R = qVC_{D_0}/\eta I_{max}cf$$

For the configuration shown, the reference area used to define R and C_{D_0} was the planform area of the horizontal tail. As a result, both R_{max} and C_{D_0} have very high values. As with previous examples, it is assumed that 90% of the wetted area of the vehicle is covered with thin flexible solar cells, and that the Earth's albedo provides 30% of the solar intensity as reflected light incident on the under-surfaces of the vehicle. With these assumptions, $R_{max} = 10$ for the configuration under consideration. The constraint diagram for this case is trivial when plotted with W_{TO}/S on the x axis. However, when the cruise velocity V is used as the independent variable, the resulting sensitivity diagram is very useful, as shown in Fig. 12. Note that the maximum speed capability at the design altitude for $R = 10$ is comparable to the top speed of the "heavier than air" concept.

To size the concept, some changes in the sizing equation are required. First, since the airship's minimum power required to maintain altitude is zero, it would function with only enough batteries aboard to energize onboard systems. However, in order to keep station against winds at night, some batteries should be carried. For initial calculations we assume it is desired to carry enough batteries to maintain 100 ft/s through the night. With this assumption, using $\eta_b = 0.55$, we calculate $\Omega_b = 0.748$. If $R_{max} = 10$, then $\Omega_c = 1.05$ and $\Omega_m = 0.89$. A survey of modern airships in Ref. 12 suggests a conservative Ω for all the airship structure of 5 psf based on using the horizontal tail planform as the reference area.

All features of the airship scale with horizontal tail surface area except the internal volume available for lifting gas. Volume scales as the cube of dimensions or as areas to the 3/2 power. The lifting ability of helium in a standard atmosphere is approximately 0.066 lb/ft^3 at sea level and about 1/17 that value at altitude. If we treat the helium lift as a negative wing loading portion, and scale lifting gas volume with S to the 3/2 power, we can now write and solve the sizing equation. Using the lifting gas volume of 250,000 ft^3 for the airship concept as drawn, and a horizontal tail area of 1800 ft^2 as reference values, and letting Ω_s represent the wing loading portion for the entire airship structure:

$$\Omega = 0 = \Omega_s + 1.1(\Omega_b + \Omega_c + \Omega_m) - 0.000015S^{1.5} + 200/S$$

The above equation has a solution for $S = 7{,}190$ ft^2. This equates to an airship that is over 380 ft long. Since our choice of Ω_s was very conservative, it is important to note that the sized planform area sensitivity to the assumed value of Ω_s, $\partial S/\partial\Omega$, is approximately 500 ft^2/psf. On the other hand, it is interesting that the sensitivity of this concept's sized planform area to fixed weight $\partial S/\partial W_{fixed}$ is less than 0.3 ft^2/lb. This strongly suggests that such a vehicle should be built to carry a much heavier payload, since the modest increase in size would result in a much more cost-effective vehicle.

Conclusions

A methodology has been described and demonstrated that provides a simple means for defining, evaluating, and sizing conceptual designs for high-altitude, long-endurance solar-powered aircraft. Three example concepts were studied; flying wing and tandem wing heavier-than-air aircraft, and a lighter-than-air airship configuration. The two heavier-than-air aircraft were found to be unable to sustain the desired altitude when estimates for current solar–electric propulsion and energy storage systems efficiencies and energy densities were used. However, the use of a mission profile that involved climbing during daylight hours to 80,000 ft and then gliding at night to 50,000 ft before using stored energy to sustain altitude, allowed both aircraft to meet the other mission requirements. Both aircraft showed improvement with higher aspect ratios, with aspect ratios of 25 providing concepts that could sustain 50,000 ft altitude. The airship studied showed very good performance, but its large size would make it extremely expensive.

The methodology presented has been shown to be a quick and useful tool in conceptual design. The results of the various analyses suggest that high-flying, long-endurance solar-powered aircraft are feasible.

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