Comparing Solar Arrays for Autonomous, Fixed-Wing HALE Aircraft using the Metric of Absolute Ceiling

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Solar-powered fixed-wing high-altitude long-endurance pseudosatellites (HALE or HAPS) are being developed to provide telecommunication and related services to markets across the globe. For technical, operational and regulatory reasons, these aircraft need to operate continuously at stratospheric altitudes, which is a challenge given the low air densities and limited electrical energy available aloft. We start with a formulation of absolute ceiling of fixed wing aircraft in terms of basic aerodynamic and propulsive power parameters, and use it as a yardstick to derive an equivalency between solar arrays of differing power conversion efficiencies and area densities. We also present numerical values for selected solar technologies parameters and their effect on performance.

Nomenclature

η	Solar cell conversion efficiency
S_s	Area of solar array (m^2)
F	Solar cell packing factor
W_e	Weight of basic aircraft (no solar) (N or kg)
W_s	Weight of solar array (N or kg)
R	Ratio of solar weight per unit area to wing loading of basic aircraft
P_{solar}	Power delivered by solar array under standard test conditions (W)
P_{prop}	Net propulsive power (W)
k_s	Constant of proportionality relating P_{solar} and P_{prop}

Other symbols have their usual meanings.

I. Introduction

The burgeoning commercial demand for low-latency data connectivity and high-resolution imagery is driving the development of fixed-wing solar-powered high-altitude long-endurance pseudosatellite platforms (HALE or HAPS) as an alternative to satellite platforms.[1] The development of HALEs is being enabled by a convergence of advances in electric propulsion, battery and solar technologies, as well as shrinking payload sizes. In order to maximize commercial utilization, it is desired that these platforms be able to remain aloft and on station for months.

Solar is the only form of energy that can match the autonomous mode of operation and extended mission profiles of these platforms. Although indirect utilization of solar energy through thermals can be valuable and is being actively explored,[2] direct conversion of sunlight to electricity through wing- and body-mounted solar arrays is likely to be the dominant source of power for these platforms, providing forward propulsion for station keeping as well as altitude gain and battery charging for flight through the night. Although flight at lower altitudes (i.e., higher air densities) requires lower power, operators of high-altitude long endurance pseudosatellites (HALE or HAPS) require a high altitude of 20-25 km to place the aircraft in a zone of low wind speed and well above most air traffic. Sustaining flight under these low air density conditions is

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challenging and requires optimization of the aircraft design, including the choice and application of solar technology.

In the early days of solar powered aircraft there was only one solar technology, crystalline silicon solar cells, available in sufficient quantity and quality. This technology was used for many pioneering designs, including the Helios platform developed by NASA and Aerovironment. In the last twenty years, however, a number of new solar cell technologies have been developed and commercialized (see Fig. 3).

One question that naturally arises from the wide choice of solar technologies is whether a lower efficiency technology might be able to perform just as well for a HALE not on the basis of conversion efficiency, but due to a superior power to weight (or mass) ratio (watts per kilogram). The motivation for examining this in more detail may arise from the fact that the solar technology being considered offers reduced cost, increased robustness, or ease of integration during manufacture of the HALE.

In this paper we assume that the solar arrays are integrated into the wings, and we derive an equivalency between solar arrays of differing power conversion efficiencies and area densities, using absolute ceiling as the metric of comparison. Intuitively, we know that the weight of the solar array would be more important in designs where the aircraft weight is small for its size, i.e. in designs where the basic wing loading is low. Therefore, the analysis references the weight of the solar relative to that of the basic aircraft itself.

We recognize that this model has a number of simplifications, both in aircraft performance and in the design and weight of the solar array. In particular, higher order design effects resulting from battery and structural sizing are not addressed. Since batteries tend to form a significant fraction of total weight, battery selection must be carefully examined in any practical high-altitude aircraft design.

II. Equivalence of Solar Arrays with Different Efficiency, Weight, and Packing Factor

The HALE platforms are being designed to operate in the lower portion of the stratosphere between altitudes of roughly 20 to 30 km (approximately 60,000 to 90,000 feet). This would place the platform well above most air traffic, and in an atmospheric zone of low wind speeds (see Fig. 1), important for efficient station-keeping. Maintaining these high altitudes while operating on solar power and while carrying a meaningful payload is challenging. In this paper we therefore assume that the aircraft's ceiling is the most important metric of performance.

Assuming a minimum drag configuration, we arrive at a simple expression for the mininimum air density at which flight can be sustained. Basic equations are from Raymer.[3]

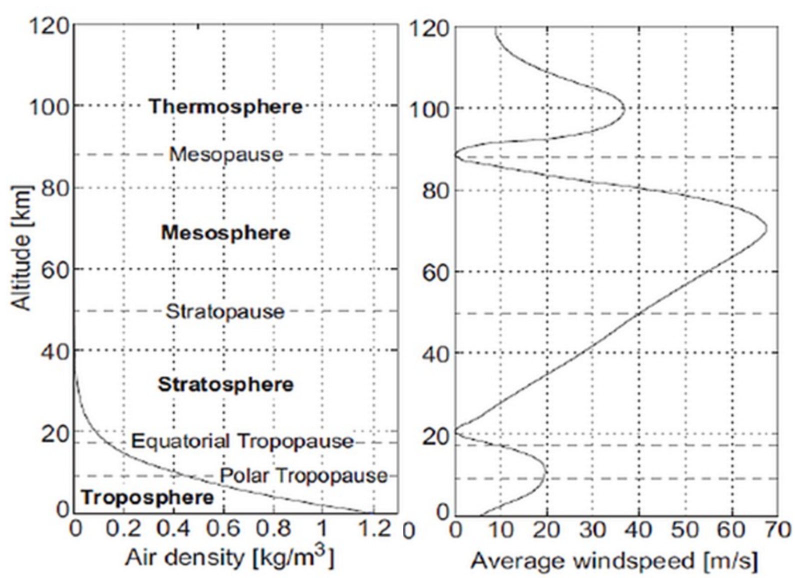

Figure 1. Air density and average wind speeds vs. altitude. The preferred altitude for high-altitude pseudosatellites is 20-30 km, in the zone of low wind speeds. Air density at 20km is roughly 1/10 that at sea level.[3]

II.A. Absolute Ceiling

Vertical speed (rate of climb) can be written as

$$V_v = \frac{P_{prop} - DV}{W} \tag{1}$$

where P_{prop} is the net propulsive power. Since the rate of climb is zero at the aircraft ceiling,

$$P_{prop} = DV \tag{2}$$

We also have the expression for minimum drag, which corresponds to best climb:

$$D_{min} = 2qSC_{d0} \tag{3}$$

Therefore

$$P_{prop} = 2qSC_{d0}V_{mt} \tag{4}$$

where V_{mt} is the speed for minimum thrust (best climb). Substituting $q = \frac{1}{2}\rho_0 V_{mt}^2$, we get

$$P_{prop} = \rho_0 SC_{d0}V_{mt}^3 \tag{5}$$

Raymer gives expression for V_{mt}:

$$V_{mt} = \sqrt{\frac{2W}{\rho S}\sqrt{\frac{K}{C_{d0}}}} \tag{6}$$

Substituting the expression for V_{mt} into that for P_{prop} above and rearranging, we get,

$$\rho_0 = \frac{S^2 C_{d0}^2 \left(\frac{2W}{S}\sqrt{\frac{K}{C_{d0}}}\right)^3}{P_{prop}^2} \tag{7}$$

Looking this density up in the atmospheric tables (e.g. MIL-HDBK-310) gives us the corresponding absolute ceiling in meters or feet. For simplicity, however, we continue to use the minimum air density ρ_0 as a proxy for ceiling in the following comparison.

II.B. Different solar array metrics, same absolute ceiling

Solar array are typically constructed of many individual solar cells interconnected in series and parallel, and then encapsulated in protective coatings or laminates to prevent mechanical damage or water ingress into the solar cells. The degree of protection required depends strongly on the type of solar cell, and also on design life of the HALE platform. The solar cells, interconnections and protective encapsulation together make up the mass of the solar array.

Now we compare two solar airplanes ("1" and "2") that use different solar technologies, but reach the same ceiling. The comparison assumes that there is no difference in aerodynamics between the two cases. We use the standard wing loading ratio (W/S weight divided by wing area, called W_e in the analysis), the ratio W_s for weight of solar per unit area and introduce a ratio of ratios R. R is the ratio of solar cell weight per unit area, to the non-solar aircraft weight per unit area. Also, we use a non-unity packing factor F (the fraction of the wing area covered by solar). The tradeoff can be calculated using equation 20 and, in the case of equal packing factors, equation 21.

Equal absolute ceilings means

$$\rho_{01} = \rho_{02} \tag{8}$$

Assuming no difference in aerodynamics, therefore,

$$\frac{S^2 C_{d0}^2 \left(2\left(\frac{W}{S}\right)_1 \sqrt{\frac{K}{C_{d0}}}\right)^3}{P_{prop1}^2} = \frac{S^2 C_{d0}^2 \left(2\left(\frac{W}{S}\right)_2 \sqrt{\frac{K}{C_{d0}}}\right)^3}{P_{prop2}^2} \tag{9}$$

which simplifies to

$$\frac{\left(\frac{W}{S}\right)_1^3}{P_{prop1}^2} = \frac{\left(\frac{W}{S}\right)_2^3}{P_{prop2}^2} \tag{10}$$

Let us assume that the power P_{prop} delivered by the propeller over the course of a 24 hour period is proportional to the power rating P_{solar} of the solar array, with a constant of proportionality k_s encompassing flight path, the length of day and other solar yield parameters.

$$P_{prop} = k_s P_{solar} \tag{11}$$

A property of the solar arrays is the geometric ratio referred to as packing factor F, or how much active solar cell area can be integrated into a given airframe surface area. The size and shape of the solar cells, and the geometry of the cell-to-cell interconnections affect this ratio (see Fig. 2). Making the above equation

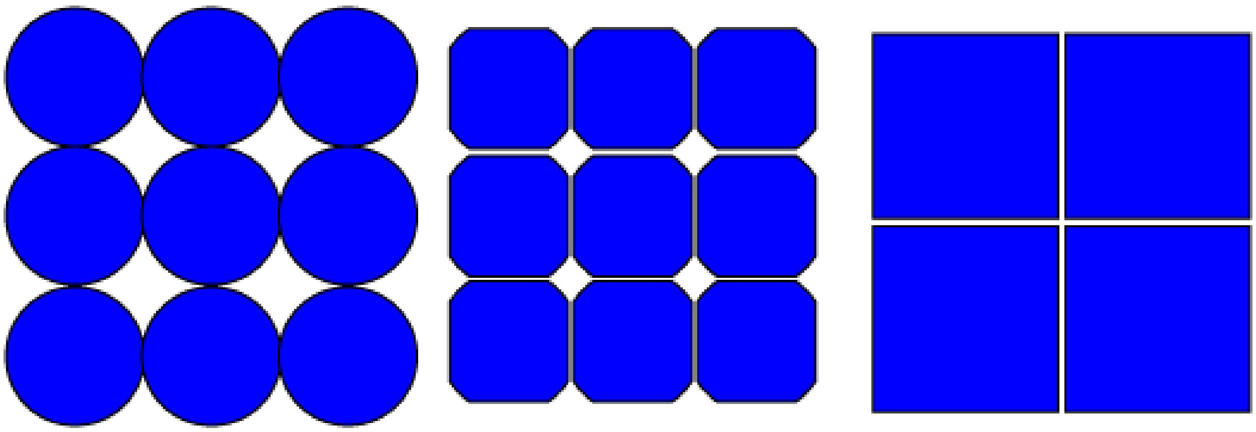

Figure 2. Solar cell packing factor, the ratio of actual solar cell area to the available area, depends on cell and interconnect design. Wing shape can also be a factor.[4]

explicit for cell efficiency, packing factor and wing area, we get,

$$P_{prop} = \eta k_s F S \tag{12}$$

Also, aircraft weight W is the sum of non-solar aircraft weight W_e and solar weight W_s. We can write overall wing loading as

$$\frac{W}{S} = \frac{W_e}{S} + \frac{W_s}{S} \tag{13}$$

By definition, the packing factor is the ratio of the area occupied by the solar cells to the overall wing area. Therefore overall wing area can be written as:

$$S = \frac{S_s}{F} \tag{14}$$

Substituting into the previous equation, overall wing loading is

$$\frac{W}{S} = \frac{W_e}{S} + F\frac{W_s}{S_s} \tag{15}$$

If R is the ratio of solar array weight per unit area, to non-solar aircraft weight per unit area, i.e. if,

$$\frac{W_s}{S_s} = R\frac{W_e}{S} \tag{16}$$

then overall wing loading can be written as

$$\frac{W}{S} = \frac{W_e}{S}(1 + RF) \tag{17}$$

Substituting this, and the expression for P_{prop} (Eq. (12)) back into the relationship for equal absolute ceilings (Eq. (10)), we get,

$$\frac{(\frac{W_{e1}}{S_1})^3(1 + R_1F_1)^3}{k_{s1}^2 F_1^2 S_1^2 \eta_1^2} = \frac{(\frac{W_{e1}}{S_1})^3(1 + R_2F_2)^3}{k_{s2}^2 F_2^2 S_2^2 \eta_2^2} \tag{18}$$

Assuming two aircraft have the same basic design and fly the same altitude and path, i.e. if $L_{e1} = L_{e2}$, $k_{s1} = k_{s2}$, $W_{e1} = W_{e2}$ and $S_1 = S_2$, we get,

$$\frac{(1 + R_1 F_1)^3}{F_1^2 \eta_1^2} = \frac{(1 + R_2 F_2)^3}{F_2^2 \eta_2^2} \tag{19}$$

Rearranging and taking square roots,

$$\frac{\eta_1}{\eta_2} = \frac{F_2}{F_1} \left(\frac{1 + R_1 F_1}{1 + R_2 F_2} \right)^{\frac{3}{2}} \tag{20}$$

In the case where the two solar arrays achieve the same packing factor on the wing, i.e. where $F_1 = F_2 = F$,

$$\frac{\eta_1}{\eta_2} = \left(\frac{1 + R_1 F}{1 + R_2 F} \right)^{\frac{3}{2}} \tag{21}$$

Therefore, in our simplified case[a], the equivalent efficiency of a solar technology for reaching an absolute ceiling does not depend on aerodynamic parameters or on the absolute weight of the aircraft, but only on the packing factor F and ratio of solar weight per unit area to the weight of the non-solar aircraft, R.

III. Discussion

There are over twenty photovoltaic technologies being actively pursued by various manufacturers and research groups (See Fig. 3). Broadly, they can be divided into wafer-based and thin-film types. Here we discuss only those technologies which can readily be produced in a form suitable for UAV integration. Relevant wafer-based technologies are crystalline silicon (c-Si) and gallium-arsenide (GaAs). Relevant thin-film technologies are amorphous silicon, copper-indium-gallium-selenide and thin gallium-arsenide based photovoltaics.

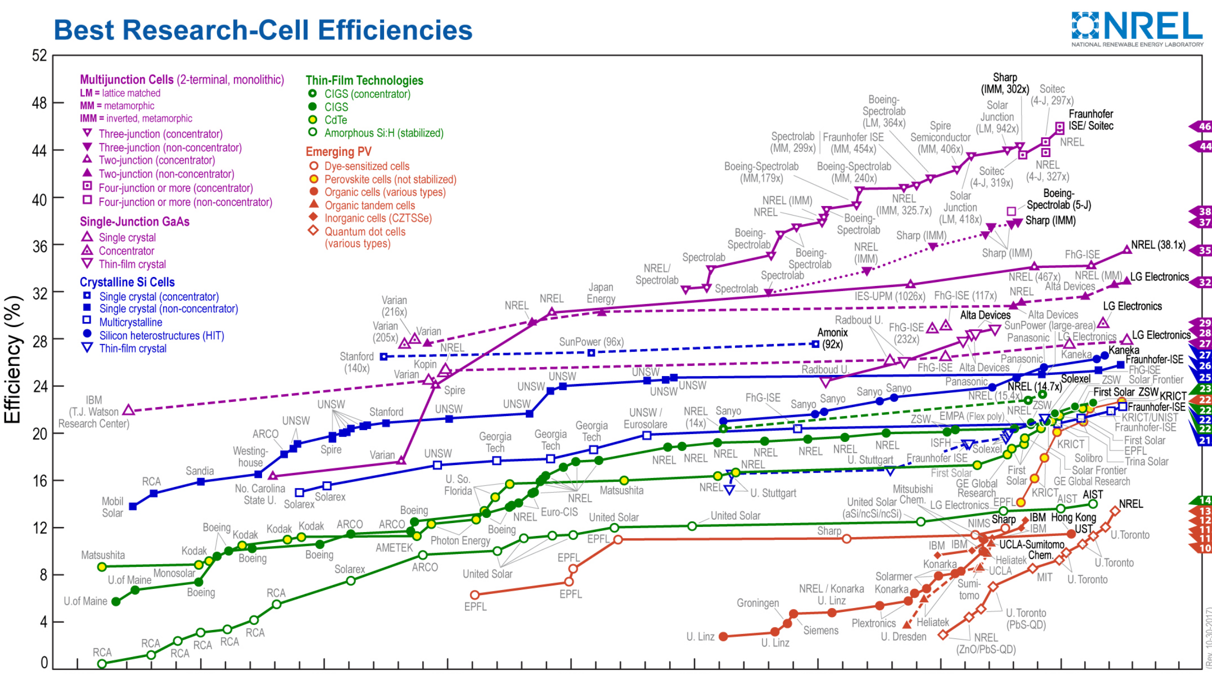

Figure 3. Evolution of solar cell technologies and efficiency records. There are now over 20 cell technologies being actively developed by researchers globally, and being tracked by the National Renewable Energy Laboratory.[5]

The most common type of solar cell technology is crystalline silicon, which has been in manufacture in various forms since the 1950s, and is currently manufactured on a gigawatt scale annually. As seen

[a]As discussed briefly below, finite battery energy density is a major reason why the real situation is more complicated and must be examined with a full accounting of energy storage.

in the figure above, crystalline silicon has achieved light-to-electric conversion efficiencies of 25-26% and is available in production at efficiencies in the 21-22% range. The ultimate efficiency of a solar cell is limited by its semiconductor properties such the nature of its bandgap and losses from various fundamental mechanisms such as phonon losses as well as defect-driven losses that manufacturers can try to minimize through production improvements. How much semiconductor thickness is required to absorb incident light is also a function of these properties. In the case of crystalline silicon, a thickness of approximately 150 microns is required to achieve highest efficiency. Since the density of silicon is 2.33 g/cm^3, this thickness translates to a limiting weight per unit area of about 350 g/m^2. In practice, weights are slightly higher due to the need for metallic contacts and other ancillary layers to electrically and mechanically support the semiconductor itself. This is also generally true in the case of the so-called III-V solar cells, which can produce high efficiencies but are prepared on a germanium wafer often about 140 microns in thickness.[6] Germanium is more dense than silicon (5.3 g/cm^3), resulting in a basic cell weight of around 740 g/m^2. The increased weight directly reduces payload capacity or available battery size.

There are other semiconductor materials, such as amorphous silicon (a-Si), cadmium-telluride (CdTe) and copper-indium-gallium-diselenide (CIGS) with higher light absorption coefficients that can achieve good light absorption with layers as thin as 1 micron.[13] These cells are typically made by depositing the semiconductor layers on to a support layer called a substrate. It can be appreciated that in such a case, many of the ultimate properties of the finished solar cells are determined not by the semiconductor, but rather by the mechanical and electrical support materials. The choice of substrate material can also constrain the processing envelope available to the designers of the solar cells, often limiting the conversion efficiency. For example, the thin-film material copper-indium gallium diselenide (CIGS) has been used to attain conversion efficiencies in excess of 22%. Those efficiencies are typically attained using a glass substrate, because glass is dimensionally stable and allows processing at temperatures in excess of 500°C. Of course, glass is heavy and rigid, making such cells generally unsuitable for use on aircraft platforms. Solar cell technologists have therefore developed CIGS solar cells on other, lighter substrates such as thin metal foils or plastics. However, processing limitations of such substrates necessitate use of suboptimal semiconductor growth regimes, resulting in efficiencies that are often 50% lower than the on-glass records. Some thin-film technologies, such as cadmium-telluride can only be effective on optically transparent substrates ("superstrates") and require processing at temperatures around 400°C,[14] basically eliminating the possibility of using metal foils or most high-temperature plastics films such as polyimides (which generally transmit only yellow and red light).

Chemical sensitivity of the semiconductor materials to moisture, oxygen or other atmospheric components also influences the overall weight of the solar array, as does susceptibility to damage from mechanical stresses. For example, it is well known[15] that CIGS is extremely sensitive to moisture. CIGS solar cells must therefore be encapsulated within special multilayer barrier materials (or within glass), resulting in significant weight increase even if the basic solar cell is itself quite light on an area basis. Solar cells made from thick semiconductor wafers, such as crystalline silicon, tend to be susceptible to mechanical stresses from impact and bending, and must sometimes be protected from such forces with a relatively thick (and heavy) protective membrane.

Of note is a relatively new class of solar cells that utilizes highly efficient III-V semiconductor materials like gallium-arsenide (GaAs) but lifts the solar layers off the wafer substrate, transferring them to a plastic carrier film and reusing the wafers to reduce manufacturing cost. Since the cells are not fabricated on the plastic film but are transferred in a secondary step, a wider range of processing conditions and thin plastics can be used. These cells are able to retain the high efficiencies of classic III-V solar cells, but are also light, thin and flexible. Since III-V semiconductors are not susceptible to attack by moisture or oxygen, thin encapsulation is generally sufficient for protection of the cell, resulting in low product weight. Large area efficiencies in excess of 25% have been demonstrated using these cells.[7]

Table I: Mass and Area Performance of Selected Solar Technologies

Technology	Record η	Large-Area η	Bare g/m^2	Finished g/m^2	Finished W/m^2	W/kg
c-Si	25.8[7]%	21%[8]	445[9]	600	252	420
a-Si	14%	4%[10]	n/a	700[10]	48	68
Flexible CIGS	22.6%	11%	140	500	132	264
Thin 1J GaAs	28.8[7]%	25.1%[7]	112[11]	280[12]	301	1075

Table I lists the record efficiencies, large area production efficiencies, areal weights, and power to weight ratios of some of these cell technologies. When calculating final power (W/m^2) and specific power (W/kg), a multiplication factor of 1.20 is uniformly applied to the power output of all the technologies to account for increased solar radiation aloft. Geometric packing factor is not accounted for in the table. For silicon, we assume a 155 g/m^2 front laminate film. For CIGS, we assume a large-area efficiency of 11%, 100 micron total cell and substrate thickness (average density of 1.4 g/cm^3), a multilayer optically clear barrier film[16] on the front and a 120 g/m^2 metallized film moisture barrier on the back.

Table II lists design parameters of a hypothetical high-altitude aircraft design. The constant of proportionality k_s is assumed to be 0.20. A given location over the earth is illuminated with sunlight (intensity approx. 1367 W/m^2) for 1/4 of the time on average[b], and an additional power system derating of 20% is assumed, for an overall factor of 0.20.

Table II: Hypothetical High Altitude Aircraft

Aspect Ratio (A)	Area (S)	Min Drag Coeff (C_{d0})	Non-Solar Weight (W_e)	Oswald Factor (e)	Solar Constant of Proportionality, (k_s)
20	100 m^2	0.05	380 kg	0.8	0.20

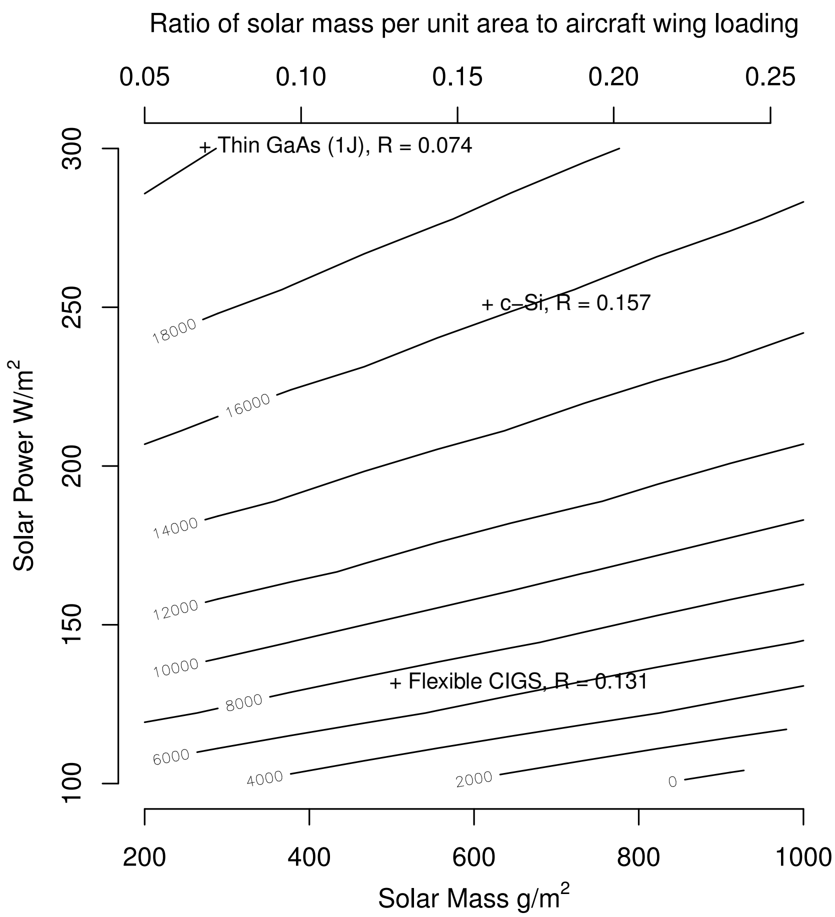

Figure 4. Absolute ceiling contours (in meters) for the aircraft of Table II (using Eq. 7), as areal density and efficiency of solar array is varied. Contours are every 2000 meters. Selected solar technologies from Table I are included for reference and the ratio R is listed in each case. Obviously, in addition to limiting aircraft performance, use of a heavier solar technology will also reduce the useful payload or battery capacity. These considerations will be importance in any practical application.

We then apply Eq. 7 to these parameters to calculate the minimum air density at which flight would

[b]The Earth's disk area as presented to the Sun is πr^2 and its surface area is $4\pi r^2$. The ratio is 1/4, or 6 hours per day on average, varying with latitude and season. Weather also modulates this at lower altitudes.

be supported, for a range of solar efficiencies and solar mass. Note that the table lists weight in kilograms, but this must be converted to newtons to obtain the correct numerical values. The air density is converted to a nominal altitude in meters by looking them up in MIL-HDBK-310 and plotted in Fig. 4. The contour lines represent loci of equal absolute ceilings for different combinations of solar power (in W/m^2) and solar mass (as weight per unit area, g/m^2). A packing factor (F) of unity is assumed. R, the ratio of solar weight per unit area to the wing loading of the basic (non-solar) aircraft of Table II is also shown at the top of the figure). Solar technologies from Table I are included for reference and the ratio R is listed in each case. Assuming a packing factory of unity, we see that using the crystalline silicon technology of Table I, the aircraft is able to attain an altitude of approximately 16,000 m. The CIGS technology, though about 100 g/m^2 lighter (-10 kg for the entire, 100 m^2 aircraft), is only able to support an altitude of approximately 7000 m on account of its lower power output. The thin gallium-arsenide (GaAs) technology is able to support an altitude of over 20,000 meters due to its combination of high power output and low weight (32 kg lower than the crystalline silicon solar array, and 22 kg lower than the CIGS array).

Since energy is available from the sun for only a fraction of the day, power for night flight must come from stored energy, usually batteries[c] that are charged from the solar array during the day (this is largely the reason for the low value of k_s, 0.20). Obviously, in addition to limiting aircraft performance, use of a heavier solar technology will also reduce the useful payload or battery capacity. For example, even if we were to accept the lower operating ceiling, the use of crystalline silicon (c-Si) instead of thin gallium-arsenide (GaAs) would reduce useful payload by 32kg (about 8% of aircraft weight) for the aircraft of Table II, quite possibly rendering the design unviable even with the availability of high energy batteries[d]. Further, if we were to accept a lower ceiling, we could actually use a reduced amount of a higher performance technology or carry a much higher weight[e]. It can be seen from Fig. 4 that only about 220 W/m^2 is needed of the thin GaAs solar to reach the ceiling attained with 100% coverage of crystalline silicon solar. This could reduce amount of solar needed and simultaneously allow use of a larger battery, or possibly allow use of a smaller wing. These considerations will be importance in any practical application.

IV. Conclusion

Designers of high-altitude long-endurance platforms can choose from a wide range of available solar cells and must carefully evaluate the significant performance tradeoffs resulting from that choice. In this paper we have presented a simplified method of comparing the power and mass densities of solar technologies, relative to wing loading of the basic aircraft. The method places a premium on attaining a high operating ceiling.

We have found that in the first order, the equivalent efficiency of a solar technology for reaching an absolute ceiling does not depend on aerodynamic parameters or on the absolute weight of the aircraft, but only on the packing factor F and ratio of solar weight per unit area to the weight of the non-solar aircraft, R.

We have also reviewed the performance of selected solar technologies and the drivers of weight in each case, and have presented these parameters in the context of absolute ceiling.

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[c]Another form of energy storage, not discussed in detail here, is potential energy in the form of excess altitude. For an aircraft weighing 380 kg (as in Table II) available potential energy per 1000 m of altitude is about 3.8 MJ, or about 1 kWh.

[d]Battery energy density currently limits the seasonal envelope of practical high altitude aircraft. Batteries needed to fly through long nights (such as winter in the temperate zones) are currently too heavy for the design space to close. Sustained flight is only possible close to the equator, where the days are long (k_s significantly greater than 0.20)

[e]Extrapolating the 16,000 m contour in Fig 5, or by using Eq. 21, we see that if R for crystalline silicon is 0.157 for this aircraft, the allowable R for the thin GaAs is 0.306, much greater than the actual 0.074.

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