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Effect of Power System Technology and Mission Requirements on High Altitude Long Endurance Aircraft

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ABSTRACT

An analysis was performed to determine how various power system components and mission requirements affect the sizing of a solar powered long endurance aircraft. The aircraft power system consists of photovoltaic cells and a regenerative fuel cell. Various characteristics of these components, such as PV cell type, PV cell mass, PV cell efficiency, fuel cell efficiency and fuel cell specific mass, were varied to determine what effect they had on the aircraft sizing for a given mission. Mission parameters, such as time of year, flight altitude, flight latitude and payload mass and power, were also altered to determine how mission constraints affect the aircraft sizing. An aircraft analysis method which determines the aircraft configuration, aspect ratio, wing area and total mass, for maximum endurance or minimum required power based on the stated power system and mission parameters is presented. The results indicate that, for the power system, the greatest benefit can be gained by increasing the fuel cell specific energy. Mission requirements also substantially affect the aircraft size. By limiting the time of year the aircraft is required to fly at high northern or southern latitudes a significant reduction in aircraft size or increase in payload capacity can be achieved.

SYMBOLS

AR	Aspect ratio
b	Wing Span (m)
c	Solar Cell Spc. Mass (kg/m^2)
e	Oswald's Efficiency Factor
f	Friction Factor
FS	Factor of Safety
m_{af}	Airframe Mass (kg)
m_{con}	Controls Mass (kg)
m_{cov}	Covering Mass (kg)
m_{eng}	Engine Mass (kg)
m_{fce}	Fuel Cell & Elect. Mass (kg)
m_{fcs}	Fuel Cell System Mass (kg)
m_{fus}	Fuselage Mass (kg)
m_{h2}	Hydrogen Mass (kg)
m_{h2t}	Hydrogen Tank Mass (kg)
m_{le}	Leading Edge Mass (kg)
m_{pl}	Payload Mass (kg)
m_{plr}	Propeller Mass (kg)
m_{prop}	Propulsion System Mass (kg)
m_{o2}	Oxygen Mass (kg)
m_{o2t}	Oxygen Tank Mass (kg)
m_{rib}	Rib Mass (kg)
m_{sc}	Solar Cell Mass (kg)
m_{spar}	Spar Mass (kg)
mte	Trailing Edge Mass (kg)
m_{tot}	Total Mass (kg)
m_{ts}	Tail Structure Mass (kg)
n	Day Number
P_{pay}	Payload Power Required (W)
P_{req}	Power Required to Fly (W)
P_{res}	Reserve Power (W)
P_{sc}	Solar Cells Output Power (W)
r	Earth Sun Distance
r_m	Mean Earth Sun Distance (m)
RC	Aircraft Rate of Climb (m/s)
S	Unit Area (m^2)
S_i	Solar Intensity (W/m^2)
S_{ia}	Avg.Solar Intensity (W/m^2)
SP_{fce}	Fuel Cell & Elect. Spc. Power (W/kg)
T	Reactant Temperature (°K)
t	Time (hours)
t_d	Discharge Time (hours)
t_{sr}	Sunrise Time (hours)
V	Aircraft Flight Velocity (m/s)
V_{min}	Velocity for Min. Power (m/s)
δ	Earth's Declination Angle
ε	Earth's Orbit Eccentricity
ϕ	Latitude
η_{fc}	Fuel Cell Efficiency
η_{prop}	Propulsion System Efficiency
η_{sc}	Solar Cell Efficiency
θ	Solar Elevation Angle
ρ	Atmospheric Density (kg/m^2)
ρ_t	Reactant Tank Density (kg/m^3)
σ_t	Reactant Tank Yield Strength (Pa)
τ	Solar Attenuation Factor
$\omega(t)$	Hour Angle

INTRODUCTION

The ability to fly nonstop for extended periods of time at very high altitudes has been an ongoing goal of the aeronautics community. The characteristics of such an aircraft would allow it to perform a variety of unique missions for a host of possible customers. These missions could include a geographically stationary communications platform for mobile phone links, high data rate video transmissions, environmental monitoring of selected regions of the Earth or atmosphere, military reconnaissance, or border patrol. This type of aircraft would be capable of performing many of the tasks that small low earth orbit satellites presently perform but at a substantially reduced cost and with the versatility of being capable of being retrieved and repaired or refitted with new or improved monitoring equipment.

Due to the high altitude at which these aircraft will be required to fly (20 km or higher) and the required endurance (from a few weeks to a year) the method of propulsion is the major design factor in the ability to construct the aircraft. One method of supplying power for this type of aircraft is to use solar photovoltaic (PV) cells coupled with a regenerative fuel cell. The main advantages to this method over others such as open cycle combustion engines or air breathing fuel cells is that it eliminates the need to carry fuel and to extract and compress air at altitude which can be a significant problem both in gathering the required volume of air and in rejecting the heat of compression.

This analysis was performed to determine what combination of solar cell and fuel cell characteristics produces the greatest benefit to the development of a high altitude long endurance aircraft and to examine how mission requirements affect the aircraft design. This was accomplished by examining how the variation in mass and efficiency of the solar arrays and fuel cells affects the aircraft configuration or size over a range of flight latitudes and mission times.

ANALYSIS

The method of analysis is based on the work performed by NASA Langley Research Center and Lockheed Missiles and Space Company on the high altitude powered platform (HAPP)[1,2,3,4]. In order for a solar powered aircraft to be capable of continuous flight, enough energy must be collected and stored during the day to both power the aircraft and to enable the aircraft to fly throughout the night. In order to determine if this is possible and, if it is, how much energy is available to the aircraft for propulsion, an energy balance diagram is used. Since aircraft size is not yet known, the energy balance must be set up irrespective of wing area. This is accomplished by plotting the solar energy produced by the solar cells per unit area, P_{sc}/S, as a function of time which is given by the following equation and is shown in figure 1.

$$P_{sc}/S = S_i \tau \eta_{sc} \sin(\theta) \quad (1)$$

The solar intensity (S_i) value varies slightly throughout the year and is

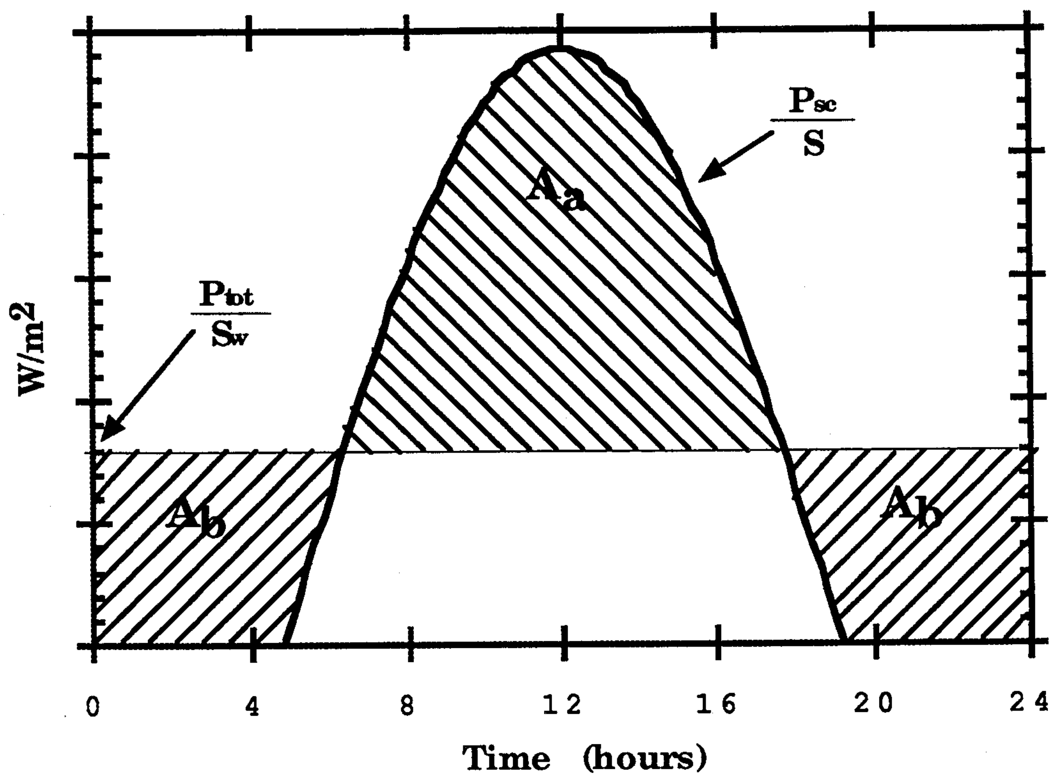

Figure 1 Energy Balance Diagram

determined by the actual distance the earth is from the sun, r, at a selected time of year. This is given by the following equations:

$$r = r_m(1-\varepsilon^2)/(1+\varepsilon\cos(\alpha)) \quad (2)$$

$$\alpha = 2\pi(n-4)/365 \quad (3)$$

$$S_i = S_{ia}\, r_m{}^2/r^2 \quad (4)$$

where r_m = 1.496E8 km, ε = 0.017 and S_{ia} = 1352 W/m^2.[1,5] For this analysis the solar attenuation factor due to the atmosphere, τ, is assumed to be 0.85. The solar elevation angle above the horizon, θ, is expressed by the following relation.

$$\theta = 90 - \cos^{-1}[\sin(\phi)\ \sin(\delta) + \cos(\phi)\ \cos(\delta)\ \cos\omega(t)] \quad (5)$$

where $\omega(t)$, the hour angle as a function of time of day, is defined as being zero at solar noon, positive before noon and negative after noon with 1 solar hour equaling 15° of rotation. The expression for $\omega(t)$ is given by:

$$\omega(t) = -\pi t/12 + \pi \qquad (6)$$

The area under the P_{sc}/S curve and above the P_{tot}/S_w value is the specific energy generated during daylight hours which can be used to power the aircraft during the night. The energy required by the aircraft during night time is 2 A_b. This energy balance is expressed by the following relation:

$$\eta_{fc} A_a = 2 A_b \qquad (7)$$

The areas A_a and A_b are shown on figure 1 and expressions for them are given below. Let $a = P_{tot}/S_w$.

$$A_a = 2[(S_i \tau \eta_{sc} \cos(\phi) \cos(\delta)\, 12/\pi) \sin(\pi a/12) + (12 - a) S_i \tau \eta_{sc} \cos(\phi) \cos(\delta) \cos(\pi a/12)] \qquad (8)$$

$$A_b = S_i \tau \eta_{sc} \sin(\phi) \sin(\delta)\, t_{sr} + S_i \tau \eta_{sc} \cos(\phi) \cos(\delta)\, 12/\pi\, [\sin(\pi a/12) - \sin(\pi t_{sr}/12)] - a S_i \tau \eta_{sc} \cos(\phi) \cos(\delta) \cos(\pi a/12) \qquad (9)$$

The sunrise time in hours, t_{sr}, and is given by the following relation:

$$t_{sr} = \cos^{-1}[(\sin(\phi) \sin(\delta))/(\cos(\phi) \cos(\delta))]\, 12/\pi) \qquad (10)$$

By iteratively solving equations 8 and 9 "a"; i.e. ,P_{tot}/S_w, is obtained. This value represents the output power per unit wing area which can be used to fly the aircraft and run any necessary equipment.

Aircraft power requirements based on mass and flight altitude must now be determined. This is done by using the power conservation equation given below:

$$P_{tot}/S_w = [P_{req} + P_{pay} + P_{res}]/S_w \qquad (11)$$

The expression for the power required to fly, P_{req}, is obtained by using the velocity for minimum power or maximum endurance, V_{min} given by:[6,7]

$$V_{min} = [4\, (m_{tot}\, 9.81)^2/(\pi\, 3\, f\, e\, (\rho\, b)^2)]^{.25} \qquad (12)$$

where the friction factor, f, for the aircraft is given by:

$$f = 0.0117\, S_w \qquad (13)$$

The above expression assumes a coefficient of friction for the aircraft of 0.005 and

that the tail area is 33% of the wing area. This value was used based on an approximation of the required tail volume. Oswald's efficiency factor, e, is assumed to be 0.8. [6]
Using the velocity for minimum power and the following expression for required power [6,7]

$$P_{req} = \rho f V^3/2 + 2(m_{tot}\,9.81)^2/(\pi\, e\, AR\, S_w\, \rho\, V) \qquad (14)$$

which yields an expression for the minimum power required to fly.

$$P_{req} = [2.48\,[(m_{tot}\,9.81)^2/(\pi\, e\, AR\, S_w)]^{.75} f^{.25} \rho^{-.5}\,]/\eta_{prop} \qquad (15)$$

The payload power is a constant which can be changed depending on the mission being considered. Reserve power, P_{res}, is power required for maneuvering or gaining altitude. It is given by the following expression: [6,7]

$$P_{res} = m_{tot}\,9.81\,RC \qquad (16)$$

Equations 15, 16 and the constant for payload power can be substituted into equation 11 to yield an expression for P_{tot}/S_w in terms of m_{tot}, S_w and AR. Now an expression must be obtained for m_{tot} in terms of S_w and AR. The total aircraft mass, m_{tot}, can be broken down as follows;

$$m_{tot} = m_{af} + m_{prop} + m_{sc} + m_{fcs} + m_{pl} \qquad (17)$$

The airframe is composed of various structural components. The airframe mass can therefore be represented as a sum of the masses of these components. This representation is given by the following equation.

$$m_{af} = m_{spar} + m_{le} + m_{te} + m_{cov} + m_{rib} + m_{con} + m_{fus} + m_{ts} \qquad (18)$$

Sizing equations for these components are given in reference 8 and are shown below.

Spar Mass: $$m_{spar} = 0.0026\, AR^{.9}\,(1 + 0.008\, AR)\, m_{tot} \qquad (19)$$

Leading Edge Mass: $$m_{le} = 0.9415\, S_w/AR^{.5} \qquad (20)$$

Trailing Edge Mass: $$m_{te} = 0.0998\, AR\, S_w \qquad (21)$$

Covering Mass: $$m_{cov} = (0.2055 + 0.0028\, AR)\, S_w \qquad (22)$$

Rib Mass: $m_{rib} = 1.033\, S_w^{.6}$ (23)

Control Mass: $m_{con} = 0.3006\, S_w / AR^{.5}$ (24)

Fuselage Mass: $m_{fus} = .0079\; m_{tot} (\rho\, V^2 S_w)^{.9} / S_w$ (25)

Tail Structure Mass: $m_{ts} = 0.4078\, m_{tot}^{.87} (AR/S_w)^{.36}$ (26)

The propulsion system consists of an electric motor, gear box and propeller. The combined efficiency, η_{prop}, for these components was set at 75%. A diagram of the propulsion and power system is shown in figure 2.

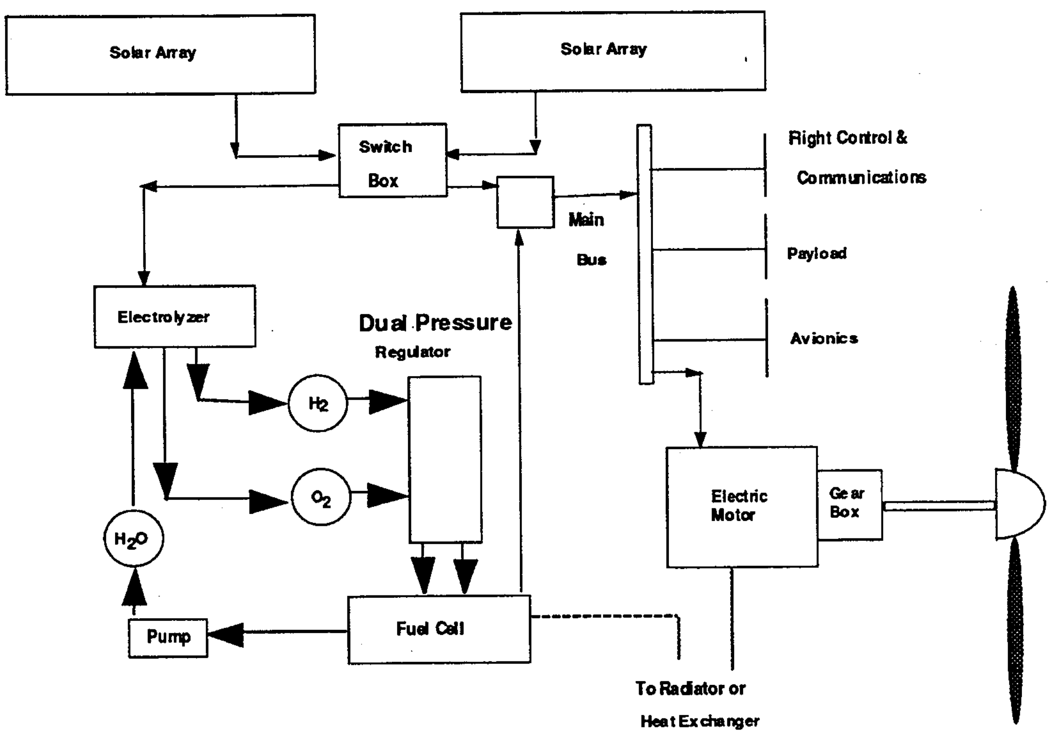

Figure 2 Propulsion and Power System

The propulsion system mass, m_{prop}, consists of the engine and the propeller.

$$m_{prop} = m_{eng} + m_{plr} \tag{27}$$

The equation for engine mass and propeller mass[8] are given below, where the constant 0.0055 is the specific mass in kg/W for a samarium cobalt electric motor.[4]

Engine Mass: $m_{eng} = 0.0055\ P_{req}$ (28)

Propeller Mass: $m_{plr} = 10.27\,(m_{tot}/S_w)^{.5}$ (29)

The solar cell mass, m_{sc}, is given by;

$$m_{sc} = c\ S_w \tag{30}$$

For this analysis it was assumed that the solar cells were only on the wing and that their packing factor was 100%. This was done to simplify the calculations. In an actual aircraft the cells would also be located on the tail surface which is estimated to be 33% of the wing surface area. Therefore the effective packing factor for this analysis, assuming that PV cells could be placed on the horizontal tail surface, is 75%.

The fuel cell system mass, m_{fcs}, consists of the fuel cell and electrolyzer, reactants and tankage.

$$m_{fcs} = m_{fce} + m_{o2} + m_{h2} + m_{o2t} + m_{h2t} \tag{31}$$

Fuel Cell / Electrolyzer Mass: $m_{fce} = P_{tot}/SP_{fce}$ (32)

Oxygen Mass: $m_{o2} = P_{tot}\,t_d/(4020.83\,\eta_{fc})$ (33)

Hydrogen Mass: $m_{h2} = P_{tot}\,t_d/(32166.67\,\eta_{fc})$ (34)

Oxygen Tank Mass: $m_{o2t} = 0.0617\,\pi\,\rho_t\,P_{tot}\,t_d\,T\,FS/(\sigma_t\,\eta_{fc})$ (35)

Hydrogen Tank Mass: $m_{h2t} = 0.5335\,\pi\,\rho_t\,P_{tot}\,t_d\,T\,FS/(\sigma_t\,\eta_{fc})$ (36)

The reactant temperature, T, is assumed to be the ambient temperature at altitude.

The final term in the total mass equation is the mass of the payload, m_{pl}, which is a constant. Substituting equations 18 through 36 into equation 17 yields an implicit expression for m_{tot} in terms of wing area and aspect ratio.

Power and mass equations having been expressed in terms of wing area and aspect ratio can now be solved iteratively to determine aircraft dimensions and characteristics required to fly at the selected time of year and altitude. Figure 3 presents a flow chart for the computer code which was used to iteratively solve the above equations.

RESULTS

The analysis generates curves (e.g. Figure 4) that represent maximum endurance (or minimum power, see eq. 11 and 12) points for the specified aircraft power system configuration and mission requirements.

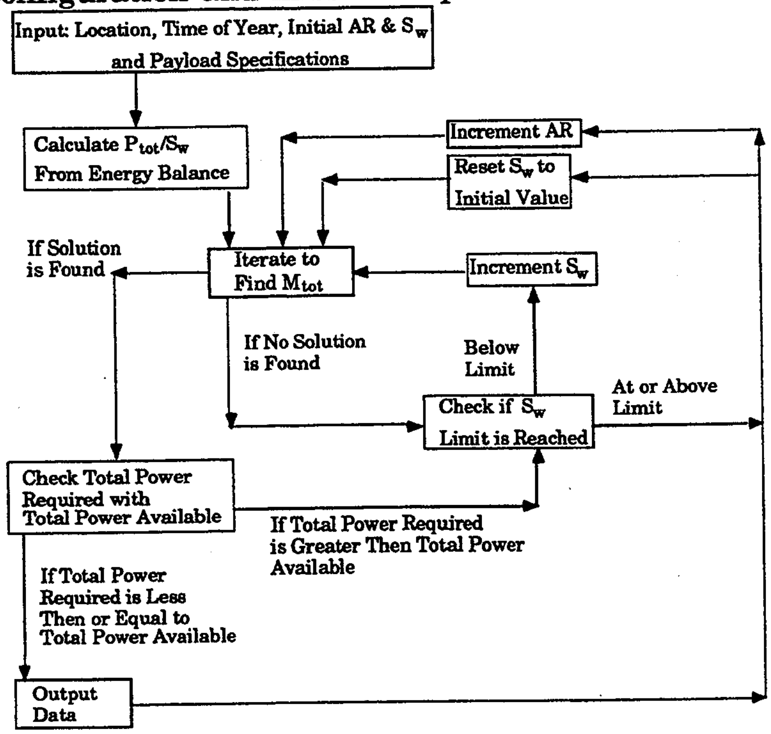

Figure 3 Computer Code Flow Chart

Any wing area / aspect ratio combination below the curve will not be capable of collecting enough energy during the day to fly continuously, whereas any point on or above the curve should be capable of collecting enough energy for continuous flight. These curves were generated in order to determine what effect the PV cell characteristics, fuel cell performance and mission requirements had on the required aircraft size. The baseline values of these quantities are as follows:

Baseline Values for Power System and Mission

Photovoltaic Array:	CLEFT GaAs	Efficiency 20% Specific Mass 0.361 kg/m^2
Fuel Cell / Electrolyzer System:	Efficiency 67%, Specific Energy 400 W-hr/kg	
Mission Characteristics:	Flight Altitude 20 km Flight Latitude; 32°N Flight Date; December 22 Payload Mass; 100 kg Payload Power; 100 W	

For each curve generated there is a wing area / aspect ratio combination that has a minimum wingspan. This minimum wingspan point is considered the design point and the performance results are given for the aircraft configuration which has the maximum endurance and the minimum wingspan or equivalently the minimum power as set by equation 12.

The results which were obtained can be broken down into two categories. The first deals with the effect of varying the power system components. The second involves varying the mission requirements.

Effect of Power System Component Variations

The baseline CLEFT GaAs array was compared with three other types of arrays. The characteristics for all four array types are shown in table 1 [8,9]. Figure 4 shows the wing area versus aspect ratio curves for the four types of PV cells.

PV Cell Type	Efficiency %	Cell Thickness (µm)	Array Specific Mass (kg/m^2)	Technology Status
Amorphous Silicon	10.0	2.0	0.022	Future
CLEFT GaAs	20.0	20.0	0.361	Near-Term
GaAs/Ge	22.0	~250.0	.064	Near-Term
Silicon	14.5	250.0	0.427	Present

Table 1 Photovoltaic Array Specifications

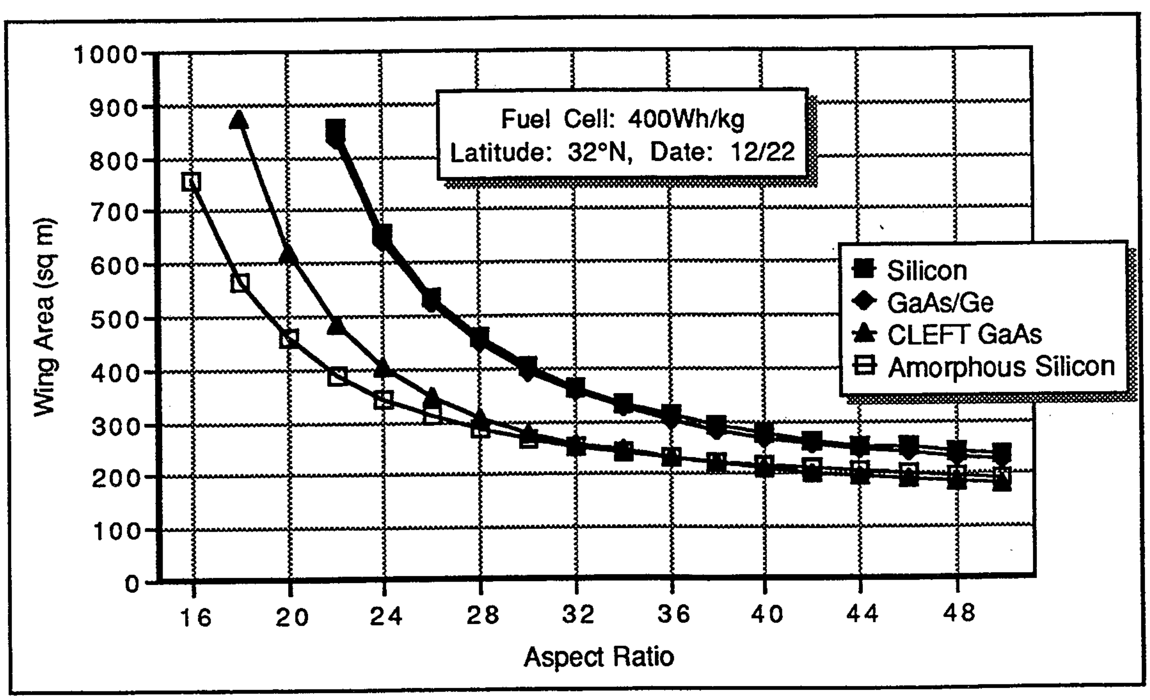

Figure 4 Maximum Endurance Curves for Various PV Array Types

By examining these results it is shown that the aircraft with amorphous silicon cells performed better then the CLEFT GaAs powered aircraft at lower aspect ratios and both amorphous silicon and CLEFT GaAs performed significantly better then the GaAs/Ge and silicon powered aircraft. For the remainder of the results CLEFT GaAs is used as the baseline PV cell. Figure 5 shows the wingspan vs aspect ratio for the curves in figure 4. The minimum wingspan point of each curve in figure 5 is used as the design point for that case. The design points for the curves in all the subsequent figures were obtained in the same manner. A listing of the design point specifications for all the figures is given in tables 2 and 3.

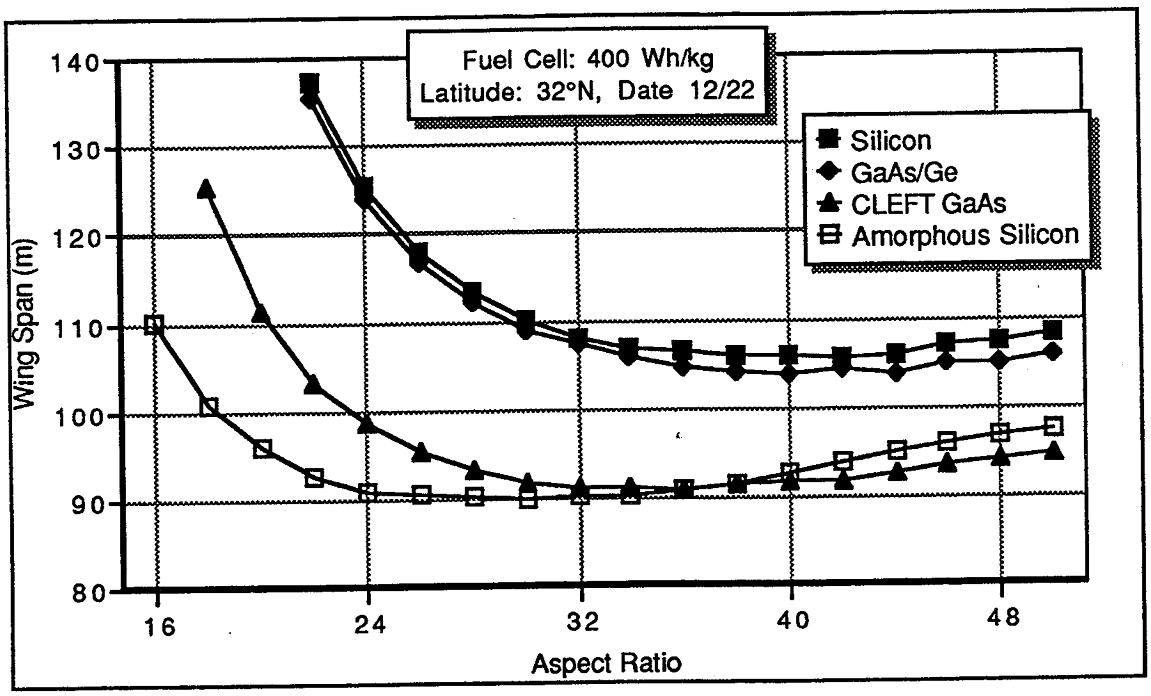

Figure 5 Wing Span vs Aspect Ratio for the Maximum Endurance Curves of Figure 4

Figures 6 and 7 show the influence PV cell efficiency and mass have on the aircraft sizing. From Figure 6 it can be seen that the benefit of increasing PV cell efficiency is not linear with the reduction in aircraft size. As the efficiency increases, the corresponding reduction in aircraft size decreases. This suggests that, with all other parameters held constant, the benefits of increasing solar cell efficiency is limited and approaches a level in which an increase in efficiency no longer effects the aircraft size. Based on Figure 7, a reduction in PV cell specific mass is directly proportional to a reduction in aircraft size.

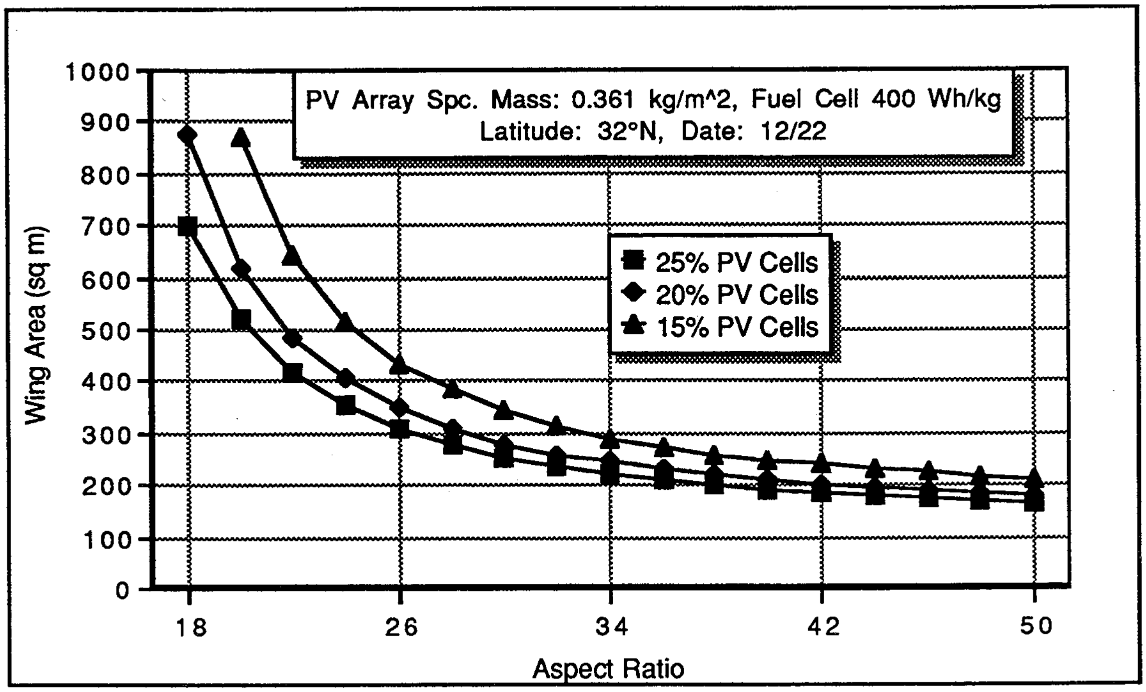

Figure 6 Effect of PV Cell Efficiency on Maximum Endurance Curves

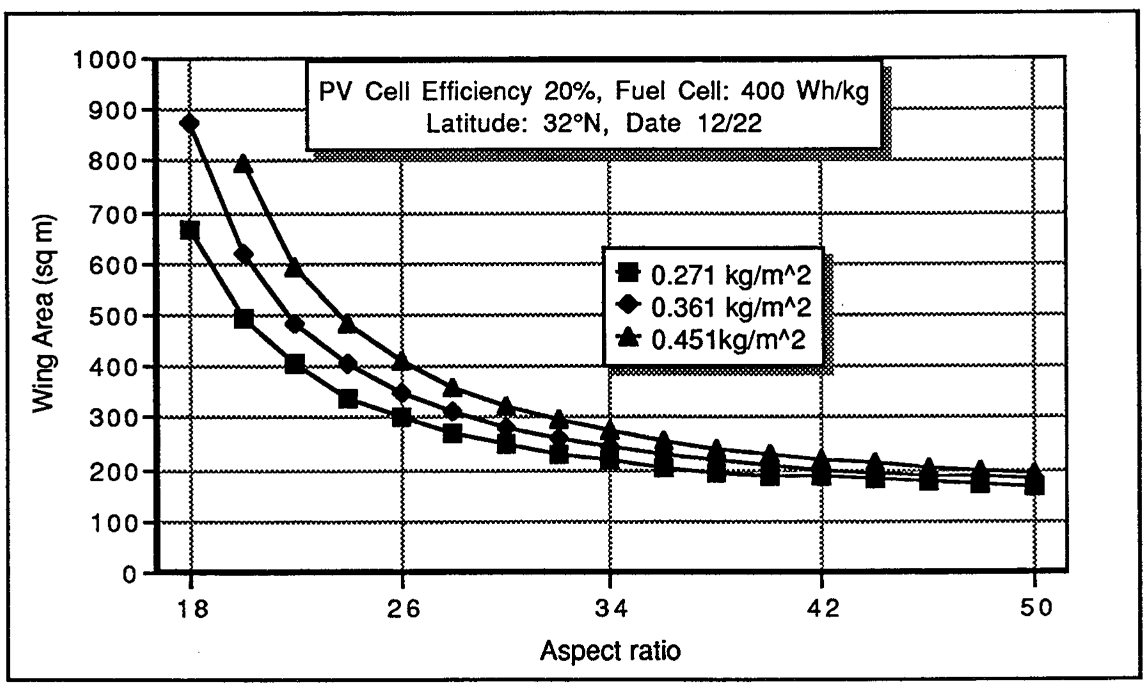

Figure 7 Effect of PV Cell Specific Mass on Maximum Endurance Curves

The effects of fuel cell efficiency and specific energy are shown in Figures 8 and 9. These two quantities are used to set the performance and technology level of the fuel cell. The three cases which are shown in the figures represent estimates for present day, near term and far term technology respectively with increasing specific energy and efficiency.

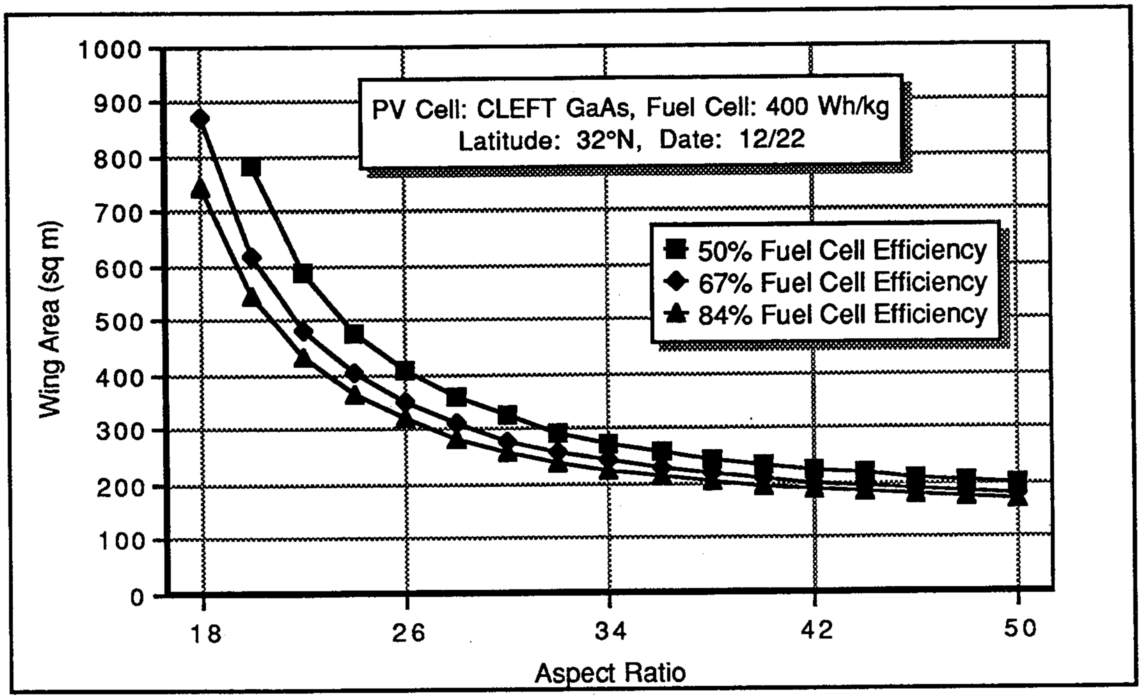

Figure 8 Effect of Fuel Cell Efficiency on Maximum Endurance Curves

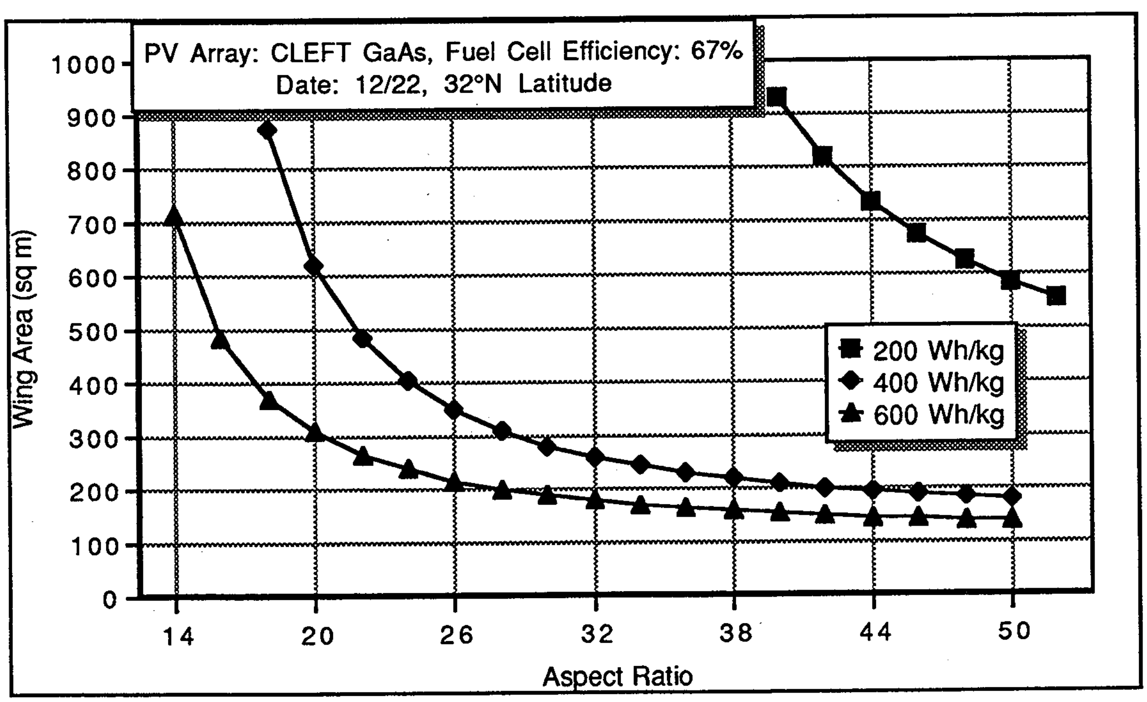

Figure 9 Maximum Endurance Curves for Various Fuel Cell Specific Energy Values

From the above figures it can be concluded that fuel cell performance has a significant impact on size and performance of a solar powered aircraft. There are modest size reductions with increasing fuel cell efficiency; however, the size reductions which are gained by an increase in the specific energy of the fuel cell are substantial. As with solar cell efficiency, the rate of aircraft size reduction is not linear with increasing fuel cell specific energy or efficiency. However, considering a present day fuel cell technology level of approximately 200 W hr/kg, any increase in this fuel cell specific energy value would provide significant decreases in the required aircraft size necessary to perform a specified mission.

Table 2 shows the specifications for the maximum endurance/ minimum wing span points from figures 4 through 9.

Variation from Base Aircraft Configuration	Mass (kg)	Aspect Ratio	Wing Span (m)	Required Power (kW) for Cruise	Minimum Cl for Cruise
CLEFT GaAs Cells	768	36	91.0	5.6	0.18
Amorphous Silicon Cells	546	30	90.0	3.7	0.17
GaAs/Ge Cells	884	40	103.9	6.0	0.19
Silicon Cells	823	42	105.5	5.3	0.20
15% Efficient PV Cells	844	34	99.3	6.0	0.18
25% Efficient PV Cells	773	34	86.5	6.0	0.18
0.271 kg/m^2 PV Cell	774	32	85.8	6.1	0.17
0.451 kg/m^2 PV Cell	877	38	95.5	6.4	0.19
84% Efficient Fuel Cell	808	34	87.5	6.3	0.18
50% Efficient Fuel Cell	775	38	96.5	5.3	0.19
600 W hr/kg Fuel Cell	589	32	75.9	4.6	0.17
200 W hr/kg Fuel Cell	1510	48	167.6	8.2	0.21

Table 2 Maximum Endurance / Minimum Wingspan Data for Various Power System Component Specifications

Effects of Mission Specification Variations

The next series of figures pertains to the type of mission and payload that the aircraft would be required to fly. The variables which represent these are flight altitude, time of year, latitude and payload mass and power. The flight altitude range which encompasses most of the scientific, environmental and reconnaissance missions is between 20 and 30 km.[10] The effect of required mission altitude is shown in Figure 10. From this figure it can be seen that aircraft size increases significantly with increasing altitude. Using the base aircraft power system and mission configuration, no wing area / aspect ratio combination was found that would allow the aircraft to fly continuously at an altitude of 30 km.

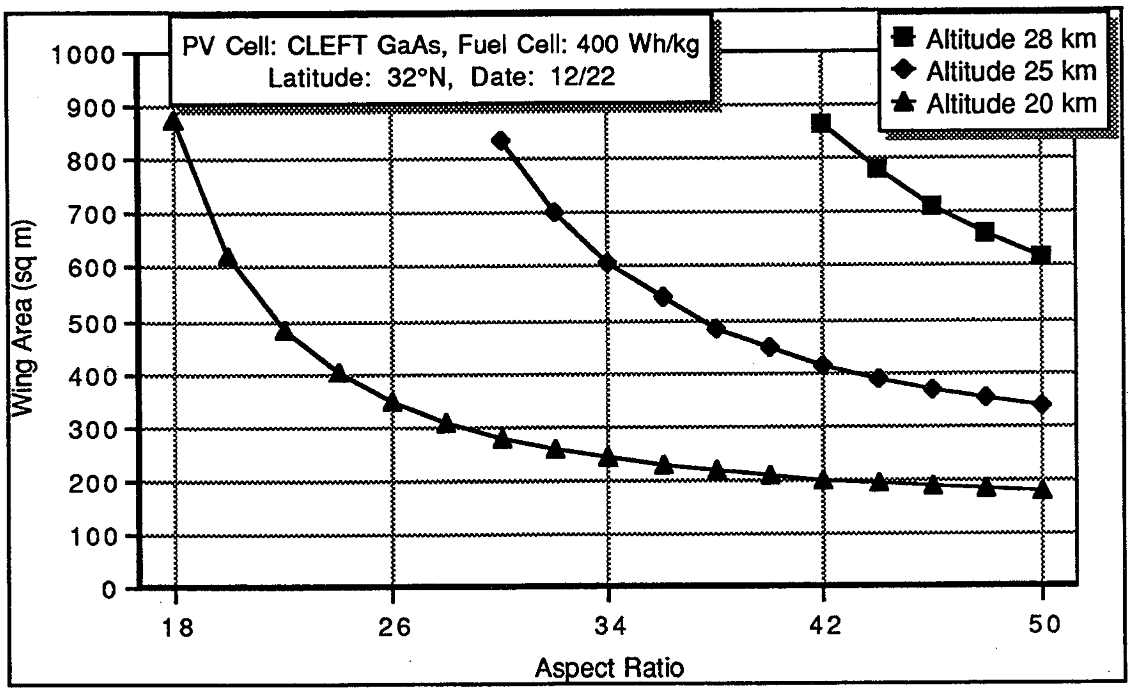

Figure 10 Maximum Endurance Curves for Various Flight Altitudes

The specified time of year (date) and latitude determines the charge/discharge period for the energy storage system as well as the amount of total solar energy available. The winter solstice, December 22, is the date with the longest discharge period and smallest amount of available solar energy. This date was chosen as the baseline because it is the time of lowest daily average solar flux in the northern hemisphere and therefore represents a worst case situation. Any aircraft power system and mission configuration which is feasible at this date would be capable of operating throughout the year. However, by varying the required latitude throughout the year, aircraft size can be reduced. Figures 11 and 12 show the maximum endurance curves for the base aircraft at various latitudes and times of year. From these graphs it can be concluded that a fairly significant reduction in aircraft size can be obtained by restricting the time of year that the aircraft is required to fly at high northern latitudes.

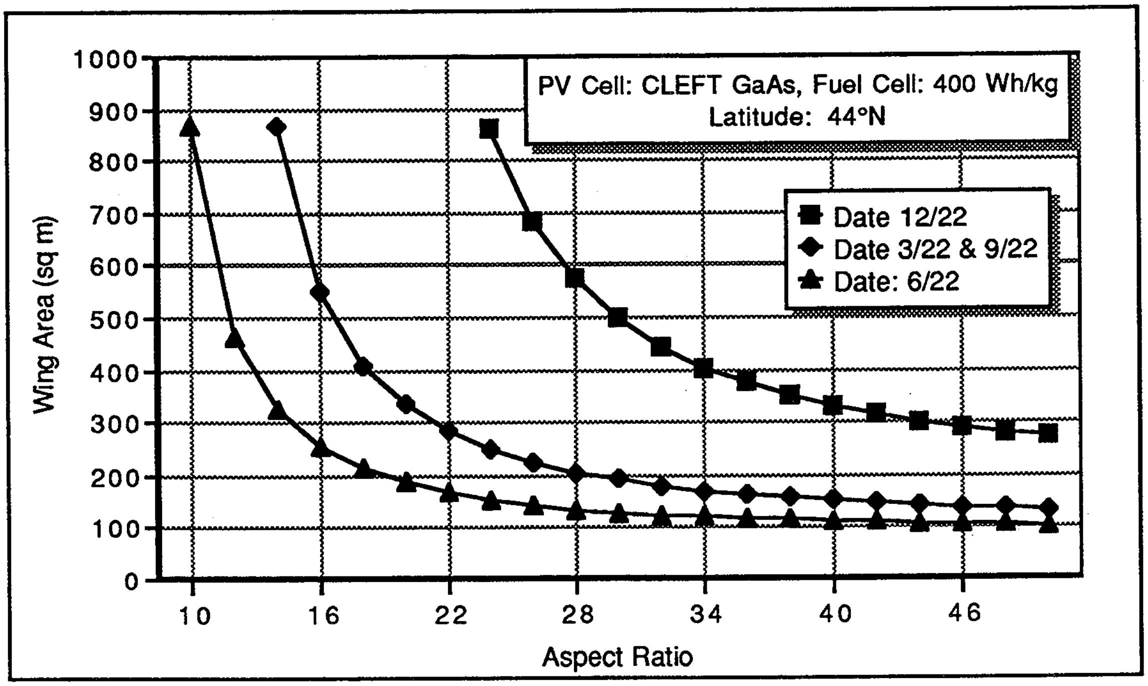

Figure 11 Maximum Endurance Curves for Various Times of the Year

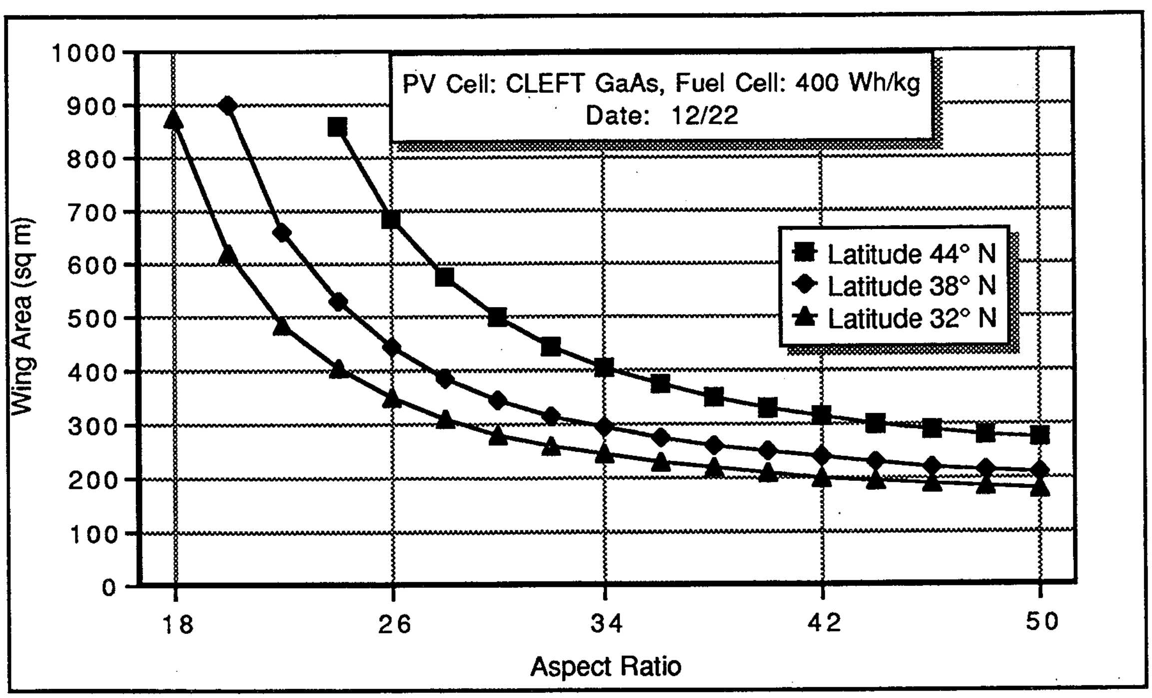

Figure 12 Maximum Endurance Curves for Various Latitudes

For example, an aircraft with an aspect ratio of 34 and a wing area of 300 m^2 would be capable of flying year round at 38° N latitude. However this same aircraft would also be capable of continuous flight for more than half the year at a latitude of 44° N. In contrast, an aircraft capable of year round continuous flight at a latitude of 44° N with the same aspect ratio would require a wing area of approximately 400 m^2.

Payload and payload power required also has an effect on the aircraft size. Mission requirements will mostly determine the amount and type of payload. In most situations lightweight, low power instruments, similar to satellite equipment, will need to be used. The effect of increasing payload mass and power is shown in Figure 13. From this figure it can be concluded that the increase in payload power has very small effect on the aircraft size. This is reasonable since payload power, which is of the order of hundreds of watts, is significantly less then the power required to fly the aircraft. An increase in payload mass, however, has a significant effect on the aircraft size. Therefore, any reduction in payload mass, such as through component miniaturization, is worthwhile.

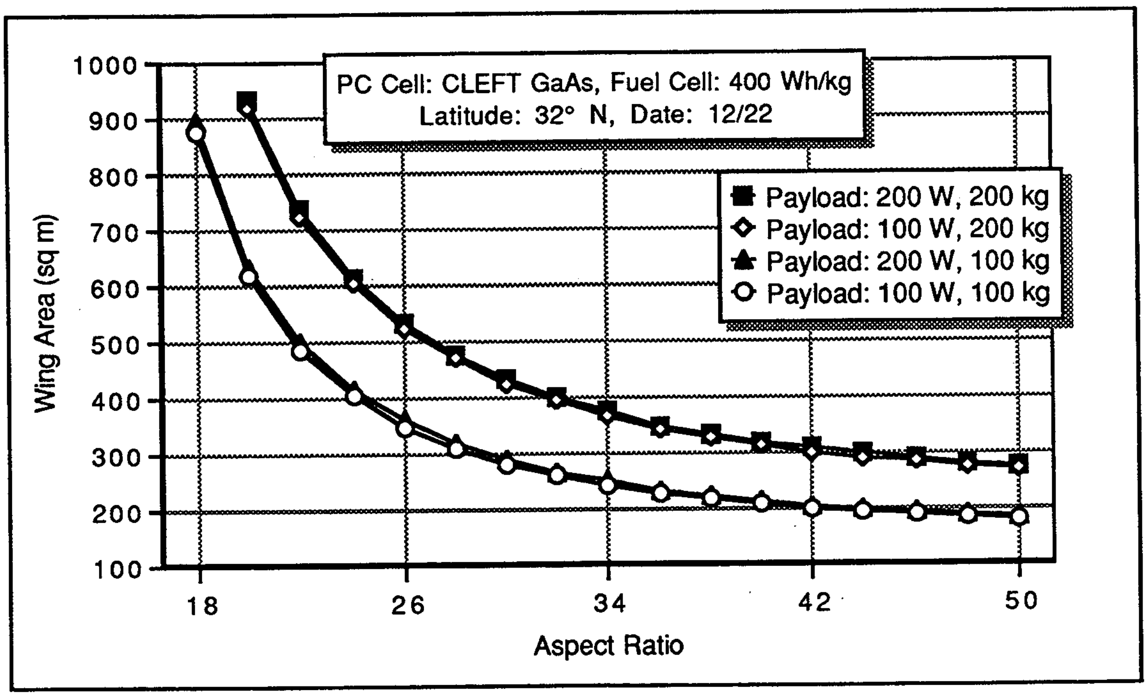

Figure 13 Maximum Endurance Curves for Various Payload Specifications

Table 3 shows the specifications for the maximum endurance/ minimum wing span points from figures 10 through 13.

Base Aircraft Mission Requirement	Mass (kg)	Aspect Ratio	Wing Span (m)	Required Power (kW) for Cruise	Minimum C_l for Cruise
Flight Date 3/22 or 9/22	667	30	74.5	5.7	0.17
Flight Date 6/22	679	24	64.8	6.9	0.15
Flight Altitude 25 km	1073	46	130.5	8.8	0.21
Flight Latitude 38°N	800	38	99.4	5.4	0.19
Flight Latitude 44°N	865	40	114.9	5.3	0.19
Payload Mass 200 kg	1287	34	111.4	9.9	0.18
Payload Power 200 W	800	36	92.0	6.0	0.18
Payload Mass 200 kg Power 200 W	1260	36	112.3	9.5	0.18

Table 3 Maximum Endurance / Minimum Wingspan Data for Various Mission and Payload Requirements

CONCLUSION

Results of this study show that increasing efficiency of power system components can have a significant effect on reducing aircraft size necessary to carry out a particular mission. The most significant reduction in aircraft size occurs by increasing fuel cell specific energy. An increase in either efficiency or performance of power system components can also be used to expand mission parameters such as flight range or altitude for a fixed aircraft size. The flight range or maximum latitude in which the aircraft is to operate also has a significant effect on aircraft size. By reducing the required year long flight latitude, aircraft size and weight can be reduced. Also if the required duration of flight is restricted to summer months, then a smaller than indicated aircraft could be used for the desired latitude. This effect is more pronounced the more northern the latitude. The PV cell used throughout most of the analysis was CLEFT GaAs. However, results with the amorphous silicon PV cell also indicate that it would be useful for this type of application. If very light weight amorphous silicon arrays or any thin film array of similar performance can be mass produced, they would have significant advantages over individual-celled rigid arrays. The main advantage would be their incorporation onto the wings of the aircraft. Since they are flexible and can be made in large sheets they can conform to the shape of the wing. This allows for fairly easy installation directly over the wing surface. Also there would be no need to wire each individual cell together as is necessary with individual rigid cells. In order to make the commercial construction and maintenance of this type of aircraft practical it is the belief of the author that light weight, flexible PV arrays will need to be used.

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13. ABSTRACT ***(Maximum 200 words)***

An analysis was performed to determine how various power system components and mission requirements affect the sizing of a solar powered long endurance aircraft. The aircraft power system consists of photovoltaic cells and a regenerative fuel cell. Various characteristics of these components, such as PV cell type, PV cell mass, PV cell efficiency, fuel cell efficiency and fuel cell specific mass, were varied to determine what effect they had on the aircraft sizing for a given mission. Mission parameters, such as time of year, flight altitude, flight latitude and payload mass and power, were also altered to determine how mission constraints affect the aircraft sizing. An aircraft analysis method which determines the aircraft configuration, aspect ratio, wing area and total mass, for maximum endurance or minimum required power based on the stated power system and mission parameters is presented. The results indicate that, for the power system, the greatest benefit can be gained by increasing the fuel cell specific energy. Mission requirements also substantially affect the aircraft size. By limiting the time of year the aircraft is required to fly at high northern or southern latitudes and significant reduction in aircraft size or increase in payload capacity can be achieved.

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