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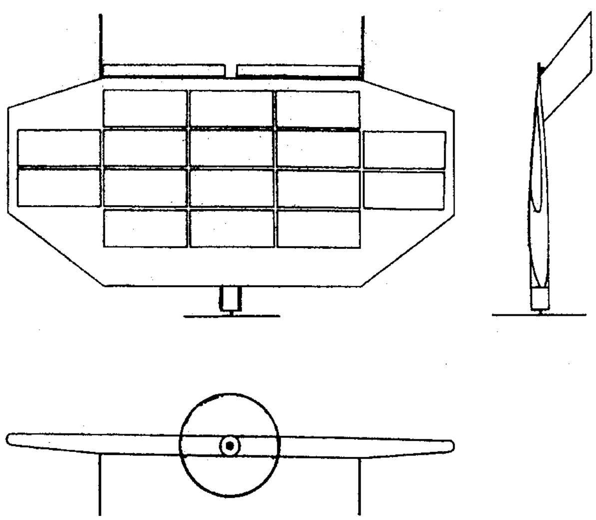

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Development of a Solar Powered Micro Air Vehicle

by

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ABSTRACT

The design of a solar powered micro air vehicle is presented. The paper describes the process followed and assumptions made during the design. The study concludes that it is feasible to build a solar powered micro air vehicle with a maximum linear dimension less than 0.38 m (15 in.). Construction of and flight tests of the prototype vehicle (Sunbeam I) are described. The battery-powered prototype had a mass of 71 gm, surface area of 0.051 m^2, and flew at a velocity of less than 10 m/s with 3.5 W of power. It is projected that on a good day (near noon during the summer months) the solar version of the plane will be able to fly on solar power alone.

INTRODUCTION

Solar aircraft have been flying for about 3 decades. Micro air vehicles (MAVs) are much newer. A reasonable first step in designing a solar powered MAV would be to review what has been done in the past related to both solar aircraft and MAVs. Below, a summary of past technologies will be presented. First solar aircraft (of all sizes) will be studied. Second, some electric MAVs will be investigated.

The first solar aircraft was Sunrise I (Fig. 1) built by Astro Flight and flown during the winter of 1974-75[1]. It weighted 27.5 lbs, had a 32 ft wing span and was powered by 450 watts of power from the solar cells. It was damaged by a sand storm in the spring of 1975.

By the fall of 1975, Astro Flight constructed an improved version called Sunrise II (Fig 2)[1]. It had over 600 watts of power, weighed 22.5 lbs and had 90 ft^2 of surface area (32 ft span).

The 600 watts of solar panels from Sunrise II were used to power Gossamer Penguin (Fig. 3) (541 W of power were available from the cells). Gossamer Penguin weighed from 170 to 250 lbs with the pilot (68 lbs without) and had a wing span of 71 ft. Numerous flights were made between May and August of 1980 under solar power.

The next airplane in the series was the Solar Challenger (Fig. 4) It was designed to cross the English channel. The solar cells could deliver over 4000 W at altitude and 2500 W at sea level. It had a wing span of 46.5 ft, weighed 336 lbs (with pilot). On July 7, 1981 the Solar Challenger flew across the English channel.

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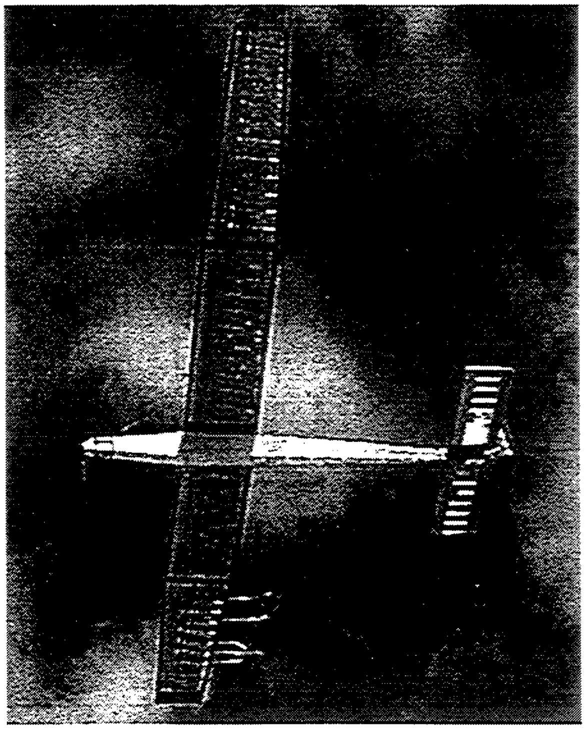


Figure 1: Sunrise I

Figure 2: Sunrise II

Following Solar Challengers, AeroVironment was funded to work on a classified project for the U. S. government[2]. They built an airplane designated HALSOL (High-Altitude Solar Energy). Three subscale models and a final prototype were built. HALSOL was mothballed. A little over a decade passed before NASA's Environmental Research Aircraft and Sensor Technology (ERAST) program became interested in solar aircraft. In 1997 Pathfinder (Fig. 5) flew to an altitude of over 67,000 ft. Pathfinder had a wing span of 98 ft, weighed 486 lbs and was powered by 8000 W from the solar cells. Pathfinder was modified, increasing the wing span to 121 ft. This version was flow in 1998 to an altitude of 80,000 ft. It was called Pathfinder Plus (Fig. 6)

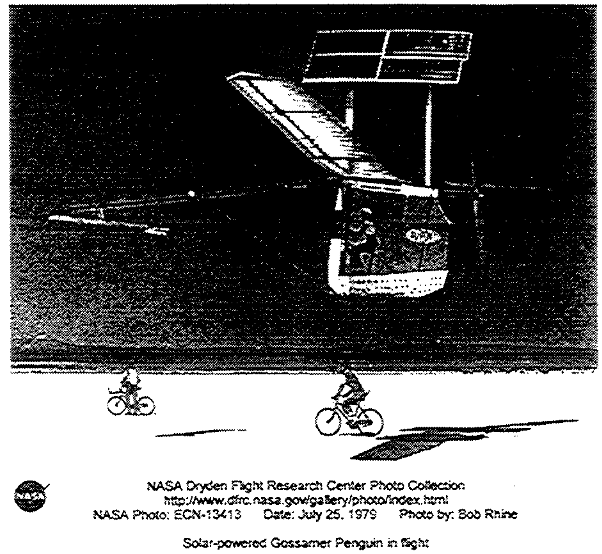


Figure 3: Gossamer Penguin

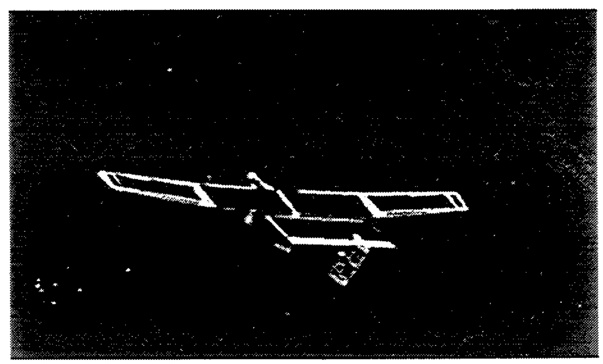


Figure 4: Solar Challenger

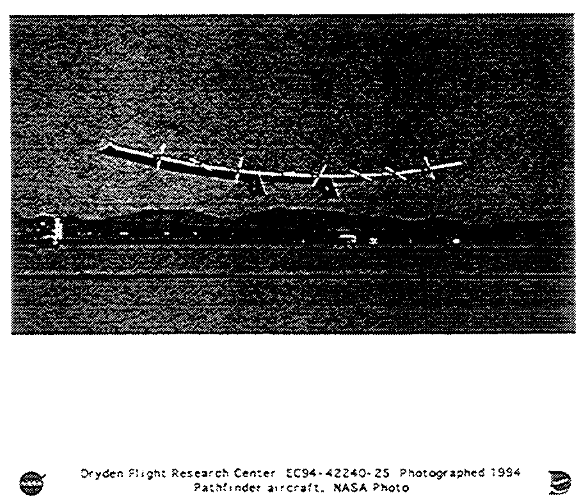
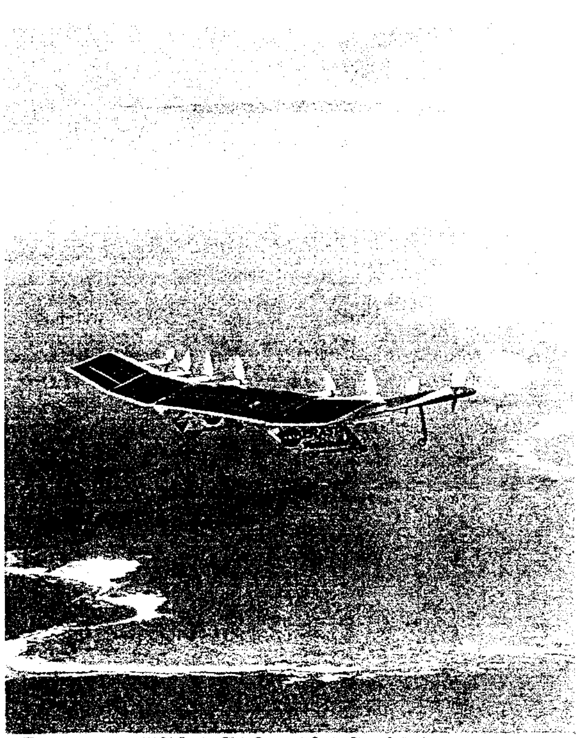


Figure 5: Pathfinder

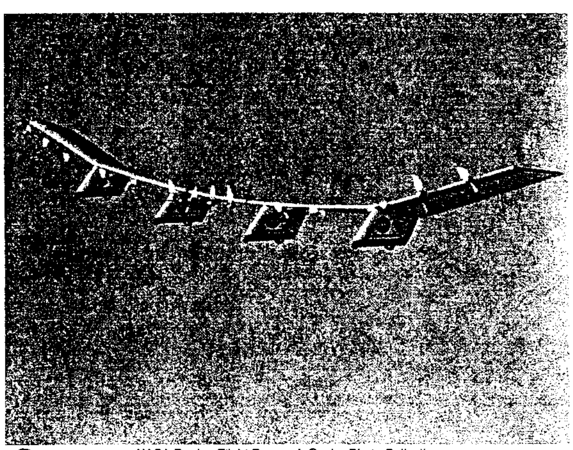


NASA Dryden Flight Research Center Photo Collection
http://www.dfrc.nasa.gov/gallery/photo/index.html
NASA Photo. EC98-44621-25 Date: 17 Jun 1998 Photo by: Nick Galante

Pathfinder-Plus on a flight over Hawaiian island N'ihau

Figure 6: Pathfinder Plus

The next in the series of aircraft was Centurion (Fig. 7). Centurion had a wing span of 206 ft and a gross weight of almost 1300 lbs. It was powered with 31 kW of solar power, and flew in 1998.



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NASA Photo: EC98-44803-115 Date: November 1998 Photo by: Carla Thomas

Centurion in Banked Flight

Figure 7: Centurion

The latest of the non-micro solar powered aircraft built by AeroVironment was Helios (Fig. 8). In the summer of 2001, Helios set the impressive altitude record reaching 96,500 ft. Helios has a wing span of 247 ft, weighs over 2000 lbs and is powered with 42 kW of solar cells.

NASA Dryden Flight Research Center Photo Collection
http://www.dfrc.nasa.gov/gallery/photo/index.html
NASA Photo: ED01-0230-3 Date: August 13, 2001 Photo by: Carla Thomas
The Helios Prototype aircraft in a northerly climb over Niihau Island, Hawaii, at about 8,000 feet above sea level.

Figure 8: Helios

While AeroVironment and NASA have been making larger and larger solar aircraft, others have been working on a smaller scale. Several individuals and colleges have tackled the challenges of solar powered flight. In 1993 Mike Garton built and flew a solar plane weighing 4 lbs, having a wing span of 18 ft and powered by 80 W of solar power.

An enthusiast for solar powered R/C aircraft is Dave Beck. He has built at least two solar aircraft, one being Solar Solitude (Fig. 9). Solar solitude has a wing span of 106.6 in, weighs 4.4 lbs and is powered by 63 W of solar cells and was flown in 1998.

Dave Beck maintains a web sited with useful information and links related to solar flight. From the site, information was obtained about Dave's first solar aircraft, Solar Excell; Wolfgang Schaeper; Oklahoma State University's Helios, University of Stuttgart's Icare 2 (Fig. 10), Bernd Bobmann's Trosollmuffel; Todd Heimerk's Simple; and Bob Boucher's Solaris I (Fig. 11) and Stardust.

The last solar aircraft to be discussed will be the small guys. These aircraft have the goal of being the smallest solar powered aircraft. The first is MikroSol (Fig. 12). MikroSol was built in 1996 by

Dr. Sieghard Dienlen and weighed 1.9 N. The next year, a smaller version was built. It was called NanoSol and was declared the smallest solar aircraft at the time (1997) by the Guinness Book of Records. NanoSol (Fig. 13) has a wing span of 1.11 m, weighs 1.56 N and is powered by 8.64 W of solar power. The latest version is PicoSol weighing in at 1.24 N and having a wing span of 0.99 m (Fig. 14).

Figure 9: Solar Solitude

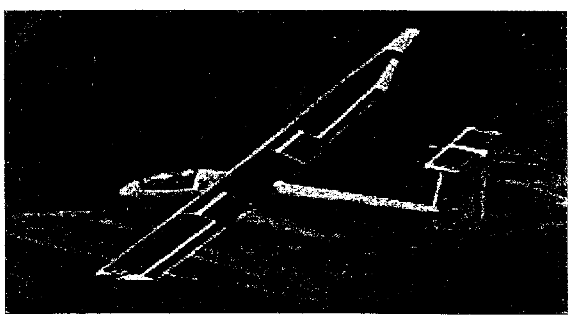

Figure 10: Icare 2

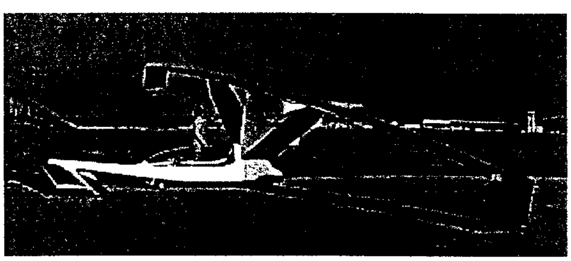

Figure 11: Solaris I

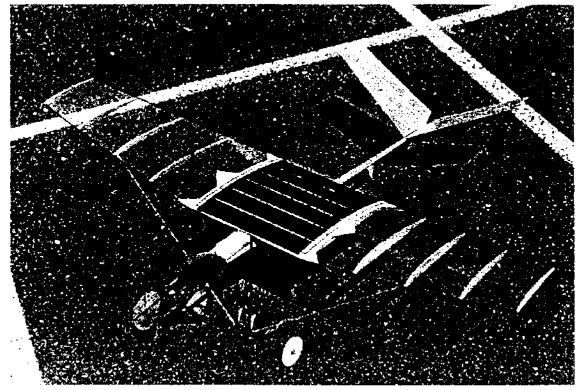

Figure 12: MikroSol

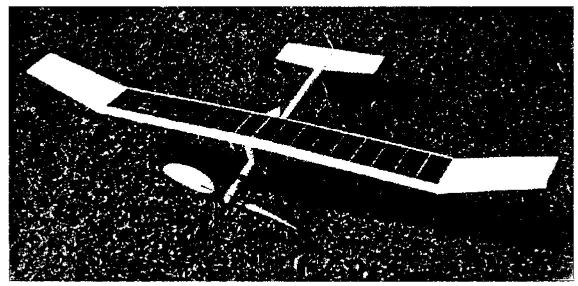

Figure 13: NanoSol

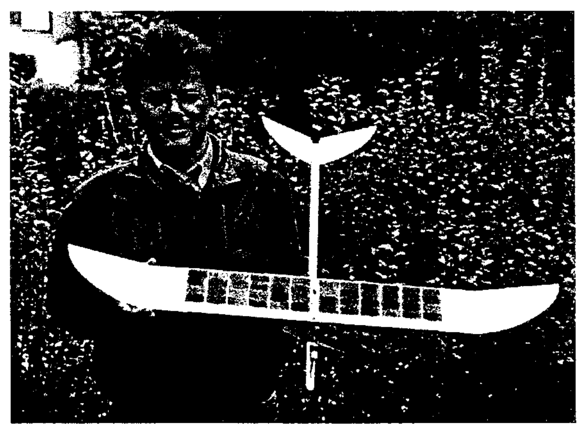

Figure 14: PicoSol

A smaller plane has been designed but not successfully flown by students and a faculty member at the University of Bristol[3] (Fig. 15). It has a wing span of 0.5 m, weighs 1.2 N and is powered by only 1 W. The weight does not include any controls since the plane is intended for free flight. Battery tests of a prototype indicated there would be insufficient solar power for the solar version of the aircraft.

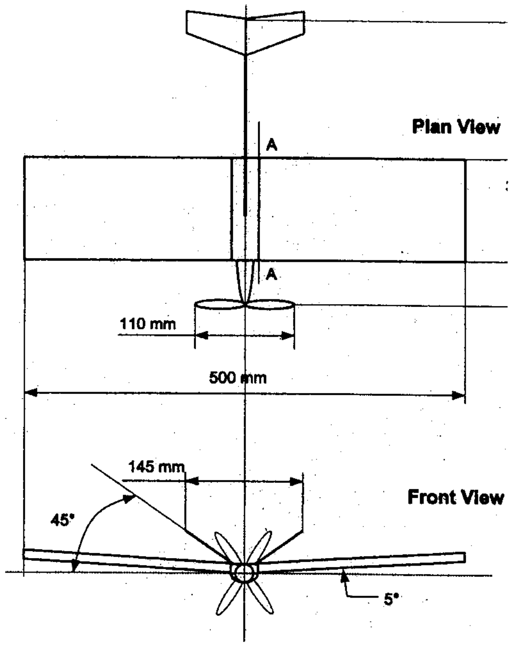

Figure 15: University of Bristol Aircraft[3]

Besides the solar planes discussed above, studying successful electric MAVs is useful when designing a solar MAV. Lockheed Martins's MicroSTAR[4] has a mass of 85 g (weights 3 oz.) 11.2 m/s (25 mph) cruise speed, and 22.9 cm (9 in.) wingspan. AeroVironment's Black Widow[5] (Fig. 16) has a maximum linear dimension of 15.2 cm (6 in.), a mass of 80 grams (2.8 oz.), and a flight time of 30 minutes. Black Widow utilizes 4.35 W of power from the batteries. AeroVironment has shown that propeller efficiencies of 80% or greater are possible at the MAV scale, motor efficiencies of 70% are also possible, and that an airfoil with a maximum linear dimension of 6 inches could fly at the low Reynolds numbers associated with small airfoils and low speeds with a lift to drag ratio of 10. For the Black Widow, the 80 grams of total mass are divided into about 50 grams for propulsion, 10 grams for payload, 7 grams for controls and 14 grams for structure.

In summary, it is useful to compile some basic design parameters for solar and electric MAVs. Table 1, lists the wing loading and power to weight ratio for many of the aircraft discussed above.

Having reviewed several solar and electric MAVs, ball park values for some of the critical design variables were found. With this information,

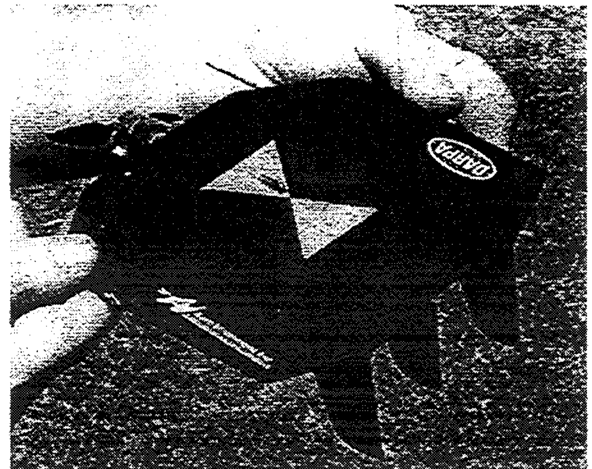

Figure 16: Black Widow[5]

a preliminary design study was conducted for a solar powered MAV. Below, the results of the preliminary design study will be presented. Next, specific components were selected and a more detail design study was conducted with refined values of component masses and propulsion system power and efficiency. These will be presented as well. A battery-powered prototype was constructed and flight tested. Results of the flight tests will be presented. Lastly, a solar powered version of the vehicle was built. Its performance will be discussed.

Plane	W/S(N/m^2)	P/W(W/N)
Sunrise I		3.74
Sunrise II	11.6	6.13
Gossamer Penguin	19.8	0.77
Solar Challenger		1.24
Pathfinder	29.5	3.7
Centurion	22.7	5.93
Helios	25	5.89
Garton	11.3	3.37
Solar Solitude	35.6	3.21
Trosollmuffel	18.1	4.28
NanoSol	9.34	5.54
PicoSol	8.37	
University of Bristol	19.4	0.82
MicroStar	~30	
Black Widow	34	5.5

1 N/m^2 = 0.0209 lb/ft^2

Table 1: Summary Data for Solar and Micro Electric Planes

PRELIMINARY DESIGN

The design procedure outlined by Anderson's in the text "Aircraft Performance and Design"[6] was used as a guide. A set of input variables was established. From the input variables, the weight and surface area of the aircraft were found. Next, the drag polar for the aircraft was estimated. Lastly, the power available from the solar cells and the power required for flight were found and compared. Any design with excess power was determined to be a potential candidate aircraft. In the paragraphs that follow, more details about the process will be given.

As mentioned above, first a set of input variables was established. The first two variables were the stall speed (V_{stall}) and the maximum speed (V_{max}). A low stall speed will result in a larger aircraft and a high maximum speed will require more power. Because the goal of this study was to design a small aircraft with a minimum amount of available power, the design velocity of the aircraft had a large impact on size and feasibility of the design. Initially, the baseline stall speed was selected as 6 m/s and the maximum speed was 10 m/s. These values were best guesses, coming from experience working with similar size aircraft. Later, it will be shown that the 6 m/s stall speed had to be increased to decrease the final size of the solar aircraft. The maximum speed of 10 m/s was a good estimate.

The next input variables were the motor efficiency (η_m), the propeller efficiency (η_p), the solar cell efficiency (η_{cell}), the available solar flux (q_{sun}), a solar cell spacing efficiency factor ($\eta_{spacing}$), and an average solar incidence angle (θ_{inc}). These variables are all related to energy capture by the solar cells and energy conversion by the motor and propeller. The baseline value for the motor efficiency was 62.5%. Maximum motor efficiency for miniature motors can be as high as 80%[5]. 62.5% was assumed to leave a margin for future improvement. Propeller efficiency was assumed to be 62.5%. This gave a combined motor-propeller efficiency of 39%. AeroVironemnt reported a combined motor-propeller efficiency of 56% for their Black Widow aircraft[5]. Thus, the 39% value seemed feasible. The solar cell spacing efficiency factor defined what percent of the aircraft plan form area could be covered by solar cells. 90% was assumed as the baseline value. The solar energy flux was assumed to be 800 W/m^2. This value was on the high end of what can be achieved in Provo, Utah where the aircraft was being built. The average solar incidence angle was assumed to be 20° (this value turned out to be optimistic for Provo, Utah which is located at 40° N latitude). The solar cell efficiency was assumed to be 16%. This is a high value for solar cell efficiency, but values as high as 30% have been reported. Several solar cells donated for the project by Sun Power were tested and shown to have efficiency as high as 16%.

Three other input variables were air density, wing aspect ratio and maximum wing lift coefficient. The air density was selected for a standard day in Provo, Utah as 1.117 kg/m^3. Because small size was the goal of the design, a small aspect ratio of 2.5 was selected as the baseline value. The maximum lift coefficient was selected as 0.5. This value has been achieved by similar aircraft (size and Reynolds number)[5].

Once the above set of variables was selected, the weight and surface area of the aircraft were determined. These two values were interrelated because for larger areas, more structure and solar cells were required which increased aircraft weight. The larger the weight, the more the area required. Weight and area were found in the following manner. The weight of the aircraft was divided into the weights of the motor, propeller, battery, controls, miscellaneous hardware, solar cells and structure. Table 2 below shows the baseline values for each of these components.

Component	Weight (N)
Motor	0.2
Propeller	0.05
Battery	0.1
Controls	0.1
Miscellaneous	0.1
Solar Cells	5.08 N/m^2
Structure	2.0 N/m^2

Table 2: Summary of Baseline Component Weights

By examining the table, it can be seen that the solar cells weight and the structure weight are assumed to be functions of the aircraft area. The values for solar cell and structure weight came from weighing the solar cells available and by building and weighing foam and balsa/tissue paper structures of similar size aircraft.

Aircraft (wing) surface area came from the equation:

$$W = L = \frac{1}{2}\rho S C_{L,max} V^2 \quad \text{or} \quad S = \frac{2W}{\rho C_{L,max} V^2}$$

where W is total weight, L is lift, ρ is air density, S is surface area, $C_{L,max}$ is the maximum lift coefficient, and V is the aircraft velocity. Iterating by first summing the component weights to find the total aircraft weight, and then calculating the required surface area, the surface area and weight of the aircraft were estimated. Once these were found, the aircraft wing loading (W/S), wing span (b) and average wing chord (c) were found. Wing span and chord were found from the relations:

$$b = \sqrt{S\,AR} \quad \text{and} \quad c = \frac{S}{b}$$

where AR is the wing aspect ratio.

The next step in the design procedure was to estimate the drag polar for the aircraft. The drag polar has the general form:

$$C_D = C_{D,o} + KC_L^2$$

where C_D is the drag coefficient, $C_{D,o}$ is the zero-lift drag coefficient and KC_L^2 is the induced drag. A conservative estimate of the zero-lift drag coefficient was made, estimating it to be 0.05. This number came from considering the friction coefficient for laminar and turbulent flow for a flat plate at a Reynolds number of 107,000 (0.004 and 0.006 respectively). The friction coefficient was multiplied by the ratio of the wetted surface area of the aircraft to the wing area, which was estimated to be 2.2. Also, the minimum drag coefficient for the EH 1590 airfoil was considered. It's value was 0.009. The largest of all these values for $C_{D,o}$ was 0.016. Reference 1 states that adding solar cells to the wing of Sunrise I doubled the drag on the aircraft. Finally, the value of 0.05 was selected as a conservative estimate for $C_{D,o}$. K was found from the relations:[7]

$$K = \frac{1}{\pi\, AR\, e_0} \quad \text{where}$$

$$e_o = 1.78\left[1 - 0.045(AR)^{0.68}\right] - 0.64$$

Once the drag polar, aircraft weight and wing area had been found, the available power (P_a) and required power (P_{req}) could be determined. These values were defined as the power out of the solar cells and the power into the motor, respectively. Available power was found from the relation:

$$P_a = q_{sun} S \eta_{spacing} \eta_{cell} \cos(\theta_{inc})$$

where q_{sun} was the specified solar energy flux, $\eta_{spacing}$ and η_{cell} were the efficiency of solar cell spacing and the energy conversion efficiency of the solar cells, and θ_{inc} was the incidence angle of the solar radiation on the wing. The required power was found with the relation:

$$P_{req} = \frac{T_{req} V}{\eta_p \eta_m} \quad \text{where } T_{req} = \frac{1}{2}\rho V^2 S\left(C_{D,o} + KC_L^2\right)$$

where T_{req} is the required thrust. The power required was found for velocities between V_{stall} and V_{max}. At each velocity, the lift coefficient required for flight had to be found by setting the aircraft weight equal to the lift produced by the wing.

$$L = W = \frac{1}{2}\rho V^2 S C_L$$

PRELIMINARY DESIGN VARIABLES STUDY

As described above, an initial set of input variables was selected for the design study. The values used are summarized in Table 3. For the design study, each of the variables was varied from a low value to a high value. Only one variable was changed at a time. The other variables were held constant at the baseline values. The effect of varying the different parameters was studied by monitoring the size of resulting aircraft and the excess power at V_{stall}. The study helped highlight the impact of each variable on the final design.

The first parameter studied was V_{stall}. Aircraft weight, surface area and span were found for different values of V_{stall}. Figure 17 shows how V_{stall} affects the required surface area of the aircraft. It should be no surprise that the higher the stall speed, the smaller the aircraft. Excess power was also calculated for each design. All of the aircraft had sufficient solar energy as long as V_{stall} was less than 10 m/s. As V_{stall} increased above 10 m/s, the surface area for energy collection became to small to meet the power requirements of the aircraft.

The second variable studied was motor efficiency. It was varied from 0.3 to 1.0. When motor efficiency dropped below 35%, excess power became negative. Because motor efficiency and propeller efficiency have the same impact, this finding also holds for propeller efficiency.

Variable	Low Value	Baseline Value	High Value
V_{stall}	5 m/s	8 m/s	10 m/s
Propeller Efficiency	0.3	0.625	1.0
Motor Efficiency	0.3	0.625	1.0
% Solar Cell Area Coverage	50%	90%	100%
Solar Cell Efficiency	8%	16%	24%
Solar Energy Flux	500 W/m^2	800 W/m^2	1200 W/m^2
Solar Incidence Angle	60°	20°	0°
Aspect Ratio	1.0	2.5	5.0
C_{Do}	0.005	0.05	0.05
$C_{L,MAX}$	0.2	0.5	0.9

Table 3: Parameters Used for Design Study

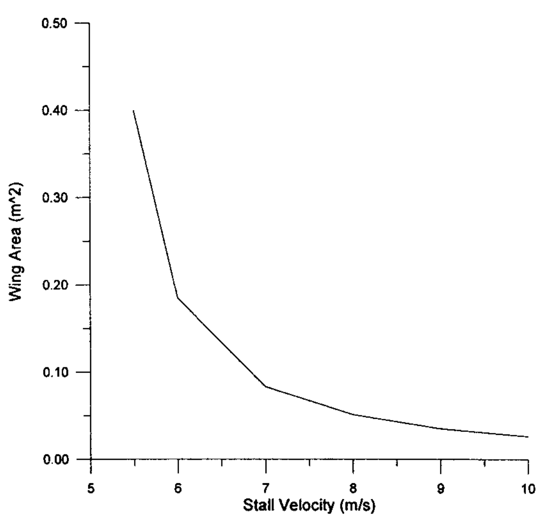

Figure 17: Surface Area Vs. V_{stall}

The propulsion efficiency is the product of the motor and propeller efficiency. It was found that this product must be greater than 22% for excess power to exist.

The third set of parameters studied were solar cell efficiency, % solar cell coverage, and solar energy flux density. Excess power was a strong function of each variable. Excess power became negative for solar cell efficiencies less than 9%, % coverage less than 50%, and solar energy flux less than 500 W/m^2. It was felt that these lower limits are achievable with current technology.

Next, aspect ratio was investigated. As aspect ratio was varied from 1 to 5 the excess power increased with higher aspect rations, but the aircraft span also increased. All aspect ratios gave feasible designs. Because of the design goal to build a small aircraft, smaller aspect ratio met the goal best.

When parasite drag was investigated, the zero lift drag coefficient was varied from 0.005 to 0.05. It was found that when C_{Do} exceeded 0.05, the excess power became negative. This placed a limit on the permissible parasite drag.

Lastly, $C_{L,MAX}$ was studied. It was varied from 0.2 to 0.9. As maximum lift coefficient was increased, the required surface area (and thus span) decreased, but the excess power decreased. For lift coefficients greater than 0.9, the aircraft became so small that excess power became negative. Reasonable values for maximum lift coefficient are about 0.4 to 0.5 for low aspect ratio, low Reynolds number applications.

PROTOYPE AIR VEHICLE

Description

From the above parameter variation study, it was concluded that a solar powered MAV was technologically feasible. Conservative values of each design variable were selected. They were used to define the first aircraft. These variables are listed as the baseline values in Table 3. Table 4 below summarizes the dimensions of the first aircraft, which was named Sunbeam I.

Within the above guidelines, a prototype aircraft was constructed. Figure 18 is a drawing of the aircraft. The plan form was selected to allow for a maximum number of solar cells. An EH 1590 airfoil was selected. It provided the required thickness to accommodate the controls components inside the wing, and had the positive pitching moment required for longitudinal stability. 5° of dihedral were added near the wing tips to improve roll stability. The outer sections of the wing were swept 35° to move the neutral point back. The chord for the majority of the wing was 18 cm. The wing tip chord was 8.5 cm. The neutral point was found to be 5.07 cm from the wing leading edge. About 1/3 of the aircraft weight was from the solar cells. These had to be distributed over the entire wing surface. In order to have positive static margin, the motor had to be cantilevered in front of the wing leading edge. Varying this distance proved to be an effective way of changing the static margin during flight tests. Two vertical tails were added to the aircraft. Their size and location resulted in a tail volume ratio of 0.03.

Parameter	Design Value	As Built (Battery)	As Built (Solar)
Motor Weight	0.2 N	0.17 N	0.17 N
Propeller Weight	0.05 N	0.01 N	0.01 N
Battery	0.1 N	0.24 N	0
Controls	0.1 N	0.17 N	0.13 N
Misc.	0.1 N	0.04 N	0.01 N
Solar Cells	0.26 N	N/A	0.22 N
Structure	0.10 N	0.08 N	0.08 N
Total Weight	0.91 N	0.71 N	0.62 N
Area	0.0501 m^2	0.055 m^2	0.055 m^2
Span	0.36 m	0.38 m	0.38 m
Power Required @ V_{stall}	3.1 W	3.3 W	TBD
Power Available	5.5 W		TBD

Table 4: Basic Design Parameters

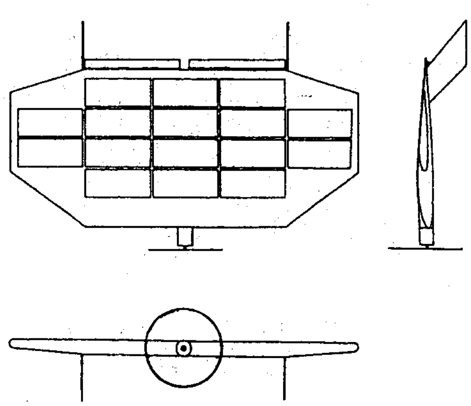

Figure 18: Sketch of Sunbeam I

FLIGHT TEST RESULTS

The first flight tests were glide tests. A model (without motor, propeller and solar cells) was taken to the top of a small hill and launched into the wind. Using radio controls, the model was flown down the hill. Four tests were conducted. Stable flight was achieved with the plane gliding smoothly down the hill. It was shown that the aircraft could turn both right and left. For the last two glide tests, the weight of the aircraft was increased to close to total design weight. No degradation in flight performance was noticed.

The second flight tests were conducted with a battery-powered version of the aircraft. A battery pack made from six 1/3 AAA NiCad batteries wired in series was made. Its weight was 0.26 N, which was close to the projected weight of the solar cells. The battery pack was mounted in the wing, centered where the center of mass of the solar cells would be. The power output of the battery pack was measured to be 10 W decreasing to 8 W over 3 minutes. This was higher than the projected power from the solar cells (5.5 W). For the first flight, the aircraft center of gravity was selected to give 2% static margin. Stable flight was achieved but the small static margin made it difficult to control the aircraft in the longitudinal axis. A small weight was added to the nose of the aircraft, increasing the static margin to 3%. This improved the longitudinal stability, making the aircraft easier to fly. The aircraft turned slow and smooth while under power. During the flight tests, the power was decreased to about ½ full power to see if the aircraft would still fly. The ability of the aircraft to climb was reduced, but it could still fly in close to level fight with a slight rate of climb.

A third flight test was conducted with a battery pack made from six, 1/3 AAA Nickel Metal Hydride batteries. These were slightly lighter than the NiCad batteries used for flight test two, weighing 0.24 N, and supplied the motor with 7.5 W of power decreasing to 3.3 W over 5 minutes. The aircraft was able to maintain level flight at the 3.3 W of power. This was very close to the 3.1 W of required power predicted during the aircraft design.

The fourth tests were of the panel of photo voltaic cells selected to power Sunbeam I. Sun Power donated some high efficiency, light weight solar cells for use on the project. On November 1, 2001 the cells were each tested with a 0.5 Ω load. On the average, each cell produced 0.045 W of power. The designed called for 16 cells in series, which would supply only 0.72 W of power to the Sunbeam I motor. This was not enough power to fly the plane (3 to 3.5 W are required). The lower than expected power level from the cells caused by the low sun angle late in the fall. A second test was conducted by tilting each cell so it was more perpendicular to the sun. The power output of each cell increased to 0.188 W, or 3.0 W for 16 cells in series. This was still less than the 5.5 W of power output expected from the cells. One possible explanation is the increased atmosphere thickness associates with the low sun angle. As a result of the above tests, flight test of the solar version of Sunbeam I have been delayed until late spring, or early summer of 2002.

While waiting for summer, work was done to reduce the required power for the aircraft. The motor propeller combination was tested in a wind tunnel. Their combined efficiency under typical flight conditions (8 m/s wind speed, 3 W input power) was measured to be 21%. Because the motor was 69%, the propeller efficiency must only be about 30%. The propeller used was a U-80 propeller obtained from a model R/C aircraft store. A more efficient prop was designed to match the motor power, RPM, aircraft velocity. A program called JAVAPROP was used. Several unsuccessful attempts were made to fabricate the new prop using BYU's rapid prototyping capabilities. Finally a U-80 prop was modified to closely match the JAVAPROP design. The modified U-80 was tested in the wind tunnel and found to have an efficiency of 58%, giving a combined motor/prop efficiency of 45%. The improved propeller should improve the chance of a successful solar powered flight for Sunbeam I.

Further improvements were sought in the aerodynamic performance of the aircraft. Earlier is was assumed that the zero lift drag coefficient (C_{Do}) was 0.05. To minimize drag for the final design, the antenna wire was imbedded in the aircraft structure, not trailed behind the aircraft, as was done in the battery powered prototype testing.

CONCLUSIONS

A solar powered, micro air vehicle has been designed. The aircraft designated 'Sunbeam I' has a wing span of 0.38 m, a mass of 71 gm, is powered by 3-5 W of solar power. It's design velocity is 8-10 m/sec. A battery powered prototype has been successfully flown at a power level similar to the expected solar cell power output. Flight testing of the solar version has been delayed until the summer of 2002 when the sun angle is more favorable.

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Note: Much of the data and many of the photos were obtained from the Internet. Because websites often change, none are listed here. An Internet search is the most reliable method of obtaining access to much of the data presented in this paper. One source that should be reliable and where the largest number of photos were obtained was the NASA photo gallery: http://www.dfrc.nasa.gov/gallery/photo/index.html

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