The doc file is not converting to pdf OR html with Figures 3.1, 3.2, 3.3, 3.4, 4.3, 7.1

Hence this pdf Version is without these figures.

CHAPTER 1

INTRODUCTION

Recently, a lot of effort has been spent on the promotion of alternative sources of energy in various fields. The development of manned as well as unmanned solar powered aircraft has been carried out by several agencies over the past decade because of their promising potential in several military and civilian applications. The possibility of designing and constructing a fully solar powered miniature aircraft is being explored in the present project.

1.1 Aim and Scope of the Project

The present project aimed at achieving sustained, controlled flight or hop of a Radio Controlled (R/C) model aircraft using only solar energy. Since this was the first attempt of its kind at IIT Bombay, in India, no information or experience regarding the design and manufacturing of this class of aircraft was available. Therefore, no particular mission or flight profile was prescribed. Improvements and optimization in the design were carried out only to the extent that it did not interfere with the task of building and flying the aircraft within the time frame and resources available. The project will be termed a success only if the model is designed, built and flown within a two semester period, also keeping into mind the time required for procurement of hardware components from overseas, if required.

The tasks to be completed towards the success of the project were divided in two stages as shown in Table 1.1.

Stage I	**Stage II**
Conceptual and feasibility study	Tests on individual hardware components
Literature survey	Procurement of solar panel(s)
Basic design and glider tests	Detailed design of the model
Hardware identification and procurement	Fabrication and Flight testing

Table 1.1: Break-up of Tasks in Two Stages.

It was decided that no onboard energy storage system would be provided i.e. the solar cells would directly be connected to the power-plant and control components. Thus, the aircraft was expected to be capable of flying only if sufficient sunlight was available. Also, the aircraft was expected to fly in a reasonably calm atmosphere.

1.2 Introduction to Unmanned Aerial Vehicles (UAV's)

Unmanned Aerial Vehicles or UAV's are a special class of aircraft that are designed to fly without any onboard pilot or crew. UAV's can be broadly classified into:

1) Remotely Piloted Vehicles (RPV's), which are externally controlled by an agent from outside like a pilot or control unit.
2) Autonomous Vehicles, which are designed to operate on preprogrammed missions and calculate their flight path using onboard resources. No intervention from any external agent is required.

UAV systems have achieved great attention from researchers worldwide because of their main advantage of carrying out missions without risking the life of a human operator and enabling flights of longer durations than manned flights. UAV systems are being used for several military and civilian applications as stated below:

1) Reconnaissance and surveillance in hostile environments.
2) Delivery of equipment.
3) Aerial targets.
4) Detection platform for Nuclear, Biological or Chemical warfare.
5) Actual combat missions.
6) Communications relay platforms.
7) Traffic monitoring and control.
8) Geological and Atmospheric surveying and resource assessment.
9) Search and Rescue operations.
10) Long Endurance missions.

An effort to minimize the size of UAV's has also gained immense interest in the recent years. Micro Air Vehicles (MAV's), as they are called, are a special class of miniature UAV's appreciated because of their potential in military reconnaissance missions. MAV's, because of their small size, are almost undetectable by RADAR, visible or audible signals and are able to operate in a constrained or hostile environment with little risk to human operators or expensive equipment.

1.3 Introduction to Solar Powered Aircraft

Solar Powered Aircraft have recently been one of the hottest research topics in leading research agencies such as NASA[1] and Aerovironment Inc[2]. As is the advantage with most solar powered systems, these aircraft are not required to carry any onboard fuel except for contingency purposes. This makes them very suitable as platforms for long endurance missions that may last several days/months or even up to a year. The advantages of solar powered aircraft over other aircraft are as follows:

1) In the case of solar energy, the solar power density increases with altitude from $80 mW/cm^2$, on ground level, to $136.7 mW/cm^2$ in space[3]. This is the main advantage as compared to other sources of energy.

2) In contrast to air-breathing engines, solar powered aircraft have electric power-plants. So the need to compress oxygen-lean air at high altitudes is eliminated[3]. Besides, the danger of a catastrophic explosion due to the presence of onboard fuel can be eliminated.
3) Electric power-plants have no exhaust and therefore do not contaminate the environment.
3) Solar powered aircraft are especially suited for reconnaissance or surveillance missions where high speed is not a requirement[3]. An electric power-plant also ensures little infrared signature, low noise level, better reliability and low vibration levels.
4) Besides, these aircraft have a lower number of moving parts, which makes them more reliable. Also, the life expectancy of the solar panels is about 25 years with little maintenance requirements[4].

However, solar powered aircraft that are supposed to fly overnight have to carry an onboard power storage system in the form of fuel cells or onboard batteries, which can be charged during daylight and used to propel the aircraft throughout the night.

CHAPTER 2

RELATED WORK IN THIS FIELD

Solar powered aircraft and related systems are being developed by several agencies worldwide. Solar powered aircraft have been regularly breaking altitude records set by other types of aircraft and their own predecessors. Some relevant developments have been cited in this chapter.

2.1 Projects Undertaken by Astro Flight Inc.

In January 1971 Astro Flight Inc.[5] demonstrated the world's first practical Radio Controlled Electric Airplane. In 1974 Astro Flight was awarded a contract through Lockheed to build the world's first solar powered airplane, Sunrise I. Sunrise I made its first flight on November 4th, 1974. It was powered by two Astro 40 Ferrite motors connected to a 36x24 wood propeller through a 6:1 gear drive. Over 1000 solar cells mounted on the wings produced 450 W of power. Sunrise I had a wingspan of 9.76 m (32 ft), weighed 12.25 kg (27 lb) and had a service ceiling of about 6.1 km (20,000 ft) in June, on a clear day

In June of 1975 a project for an improvised version of Sunrise I was awarded and on September 27th 1975, Sunrise II made its first solar powered flight. It had a 600 W panel of 4480 solar cells, which powered a single Astro Cobalt 40 motor connected to a 38 x 24 propeller through a 6:1 belt drive. Sunrise II had a wingspan of 9.76 m (32 ft) and weighed 10.21 kg (22.5 lb).

Figure 2.1: Sunrise II, made by Astro Flight Inc. in 1975 [5].

2.2 Projects Undertaken by Dupont and Dr Paul MacCready

Shortly after the success of Gossamer Albatross and Gossamer Penguin as human powered aircrafts, Dupont sponsored Dr MacCready in an attempt to modify the Gossamer Penguin into a man carrying solar plane[1, 5]. On May 16th, 1980 the Gossamer Penguin made its first flight with the same solar panel and motor used on Sunrise II.

Figure 2.2: The Gossamer Penguin in Flight, Powered by Solar Cells [1].

The Gossamer Penguin was designed to fly at heights of a few feet and hence was not safe for a pilot flying at higher altitudes. In 1980, Dupont sponsored Dr MacCready for building a solar plane that would cross the English Channel. This plane, the Solar Challenger, was powered by a 2700 W solar panel having 16,128 cells. On July 7, 1981 the Solar Challenger successfully crossed the English Channel covering 163 miles in 5 hours and 23 minutes, with solar energy as its sole power source and no onboard energy storage system. The Solar Challenger had a wingspan of 14.18 m (46.5 ft), weighed 152.8 kg (336 lb) and displayed various improvements in structural design and airfoil selection[6].

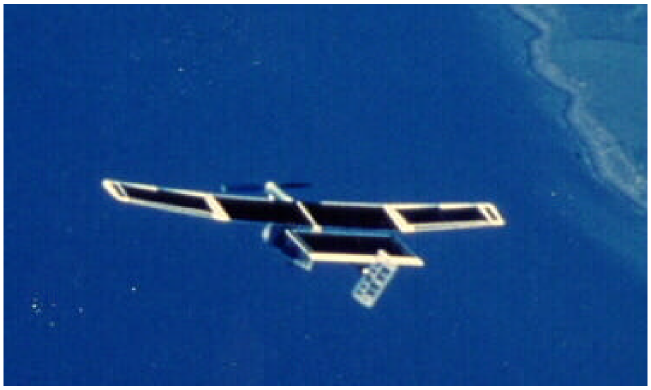


Figure 2.3: The Solar Challenger in Flight Over the English Channel [2].

2.3 Projects Undertaken by NASA

Following the success of the Solar Challenger in 1981, Aerovironment Inc. (also founded by Dr MacCready) received funding from the government to study the feasibility of long duration, solar-electric flight above 19.812 km (65,000 ft). An aircraft designated HALSOL (High Altitude Solar Energy) was built and a prototype was flown establishing the aerodynamics and structural concepts[1]. But subsystem technologies like the energy storage system proved inadequate for long endurance missions. The HALSOL was thus mothballed, only to emerge later as the Pathfinder, which flew in 1993 at the Dryden Flight Research Center, California. Later, the Pathfinder became a part of NASA's Environmental Research Aircraft and Sensor Technology (ERAST) program[7] and the resulting High Altitude Long Endurance (HALE) aircraft have been regularly shattering altitude records with the latest record being that of Helios that set an unofficial altitude record of 29.4 km (96,500 ft) on August 13th, 2001[2]. A comparison of various solar powered aircraft flown by NASA is given in Table 2.1.

Aircraft	**Pathfinder**	**Pathfinder Plus**	**Centurion**	**Helios**
Wingspan (m/ft)	30/98.4	36.89/121	62.78/206	75.29/247
Length (m/ft)	3.65/12	3.65/12	3.65/12	3.65/12
Wing chord (m/ft)	2.44/8	2.44/8	2.44/8	2.44/8
Gross weight (kg/lb)	254.09/ 560	317.6/700	862.07/ 1,900	750/1,653
Payload (kg/lb)	45.37/100	68.06/150	272.7/600	284.03/626
Cruise Airspeed (kmph)	27.4-33.8 /17-21	27.4-33.8/ 17-21	27.4-32.19 /17-21	30.58-40.2/19-25
Power (W)	7,500	12,500	31,000	35,000
Motors	1.25 kW x 6	1.5 kW x 8	2.2 kW x 14	1.5 kW x 14
Remarks		Transition between Pathfinder and Centurion	Onboard Lithium Battery	Onboard Energy storage system for night flying

Table 2.1: Comparison of NASA's HALE Aircraft [1].

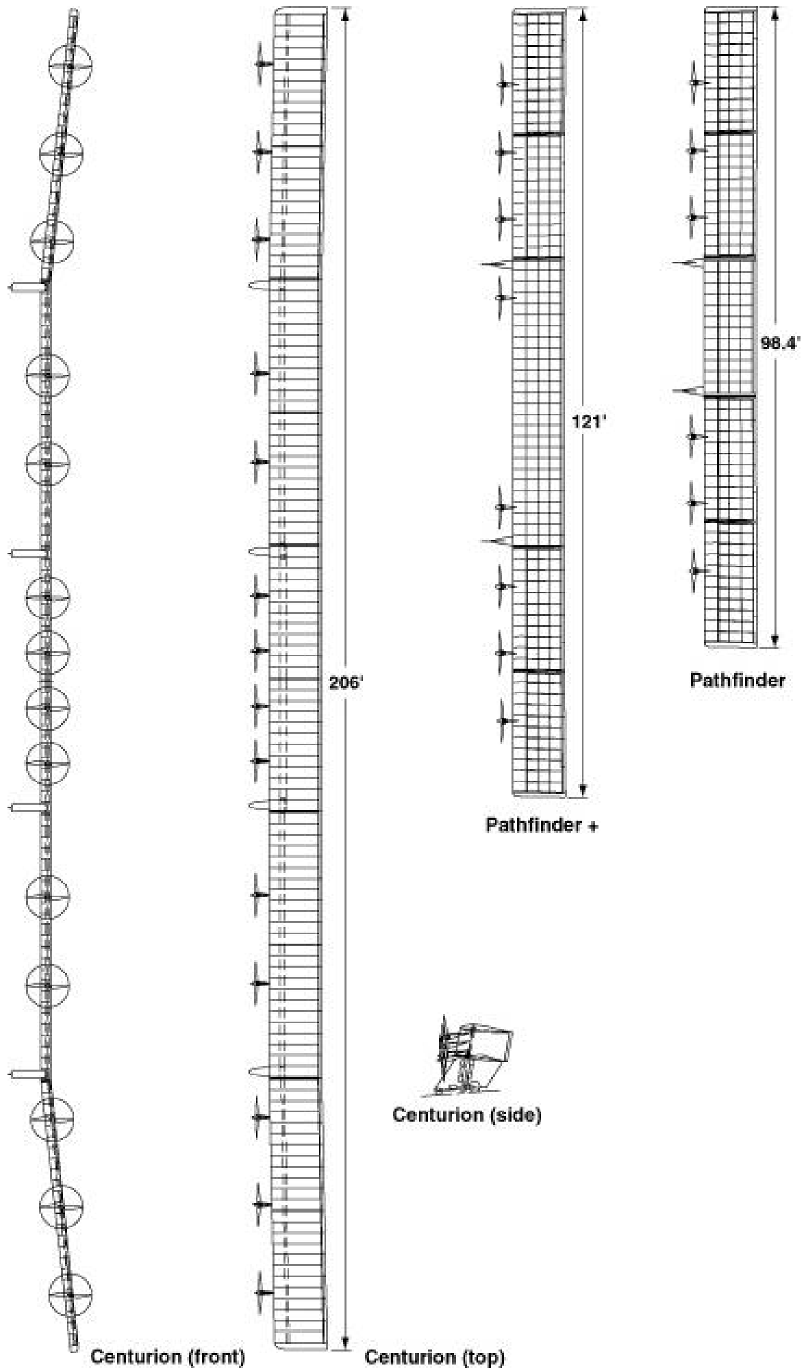


Figure 2.4: NASA's Solar Powered HALE Aircraft[1].

All these aircraft are flexible flying wings fabricated primarily from carbon fiber, graphite epoxy composites and Kevlar. Aerovironment Inc. has already launched SkyTower Telecommunications aimed at providing an alternative to communications satellites through their solar powered long endurance aircraft, Helios. Some of the advantages proposed by SkyTower Telecommunications[2] are shown in Table 2.2.

Versus Satellite Systems	Versus Terrestrial Systems
Lower cost, scalable	Lower Cost
Higher coverage angles	Higher coverage angles
Higher capacity	Faster/easier deployment
Lower power requirements	Instantaneous coverage
Maintainable / upgradeable	Minimal backhaul requirements
Relocateable	Easier to maintain
Lower launch risk	Easier to upgrade
	Easier to relocate
Versus Solar Airships / Blimps	**Versus Jet/Piston-powered Aircraft**
Technical viability	Economic viability
Station keeping in winds	Reliability / flight duration

Table 2.2: Some Benefits of Solar Powered Aircraft [2].

Another novel project taken up by University of California, Los Angeles (UCLA) in collaboration with NASA and Rockwell Corp. was the Speyer Flyer[8]. The project aimed at demonstrating the flight of a High Altitude Solar Powered Ultralight Aircraft, several of which would fly autonomously in a `V' formation over a city as a communications hub relaying calls between cell phone users.

2.4 Projects Undertaken in Germany

In Germany, the first solar powered model, Solaris, flew in August 1976 and demonstrated the feasibility of solar powered flight using conventional material and design principles[9]. In 1980, Günter Rochelt designed and flew Solair I and later in 1996, Solair II with the main intention of making solar powered flight suitable for everyday use[10].

Dr Seighard Deilin from Germany flew his first solar powered Radio Controlled model, the Mikro Sol in 1995[11]. The Mikro Sol weighed 198 gm and was the first solar powered model to be capable of take off using only solar cells. Following the success of Mikro Sol, Dr Deinlin constructed the lightest solar powered model plane, the Nano Sol, which was included in the Guinness Book of World Records in 1997 as the lightest solar powered model aircraft[11]. The Nano Sol weighed 159.5 gm and had a wingspan of 1.11 m. The power consumption was 8.64 W. Dr Deinlin has also constructed a lighter model called Pico Sol, which weighs 127 gm and has to be hand launched. A commercially viable version, Pico Sol II, is also available.

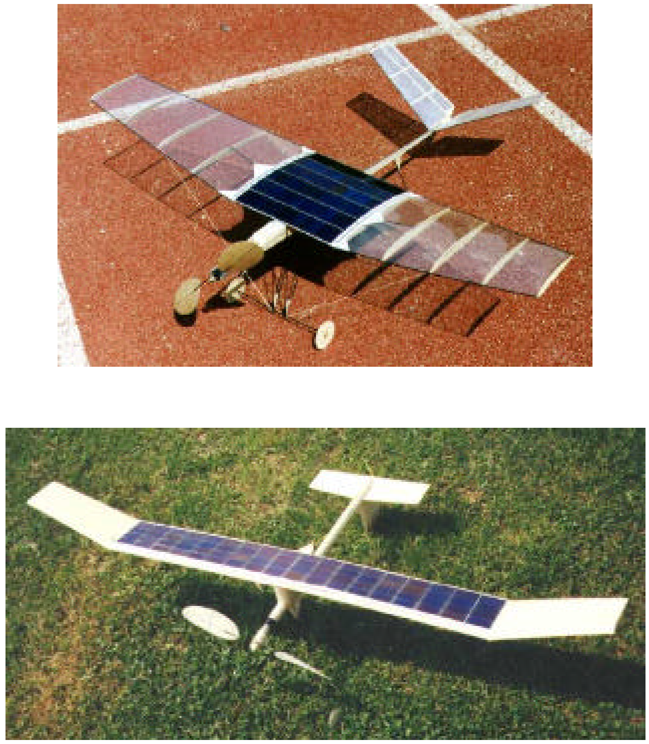


Figure 2.5: Mikro Sol and Nano Sol[11].

Besides these, research work on solar powered aircraft was also carried out at the University of Stuttgart, Germany resulting in a modified sailplane called Icare and the world's first solar powered airship, Lotte[9]. Several universities in United States of America have also been participating regularly in national level solar-car races and putting forth new designs. Interest in solar powered aircraft will surely increase in the coming years and, with preliminary studies on a solar powered Very Long Range (VLR) airship, SUNSHIP already under way, a round-the-globe flight based only on solar power does not seem to be a distant reality[12].

CHAPTER 3

SOLAR POWER FOR AIRCRAFT

Scientific investigation of the photovoltaic effect started soon after its discovery in 1839 by a French scientist named Henri Becquerel. Practically applicable solar cells were made by 1950's and since then their use as an inexhaustible and innocuous energy source has being widely studied for their use in household applications to aircraft designed for flights on planets like Mars[22].

3.1 Solar Cells

The effects of solar cells and their compatibility with aircraft have been studied in detail by various researchers over the years. Most solar cells are made of Silicon, Si, or GaAs/Ge combinations. For the present aircraft, it was decided that Si based solar cells be used. Each solar cell has a positive and a negative side and electric current is generated when sunlight strikes the positive side of the silicon cell and the electrons are activated, as illustrated in Figure 3.1.

Figure 4.1

Figure 3.1: Functioning of a Solar Cell.[23]

Several cells can be connected to make a photovoltaic (PV) module. The power available from a PV module depends on some of the factors listed below[23]:

1) The type and area of the material of the cells
2) Efficiency of the solar cells.

3) The intensity of sunlight, which depends on the orientation of the module, location, day of the year and time[24]. The effect of time of day and location can be seen in Figure 3.2.

Figure 3.2: Effect of Time of Day and Location on Solar Cell Output.[24]

For the same panel tracking sunlight can have a significant impact on the power output, as shown in Figure 3.3.

Figure 3.3: Effect of Tracking on Solar Cell Output.[23]

4) The wavelength of the sunlight impinging on it.
5) Miscellaneous factors like clouds, shade, temperature, winds etc.

Si based solar cells are available in three types: Amorphous, Polycrystalline and Monocrystalline. Out of these, the monocrystalline cells are the most efficient (up to 25%) while polycrystalline cells provide efficiencies between 10 – 12%.

3.2 Effects of Solar Cell Characteristics on Aircraft

Most of the literature available provided information on the effects of various parameters related to the use of solar power on aircraft obtained through computer-generated models[24, 25, 26]. Some of the observations that were made from these studies are:

1) For long endurance aircraft with an onboard energy storage system, the efficiency of the storage system has the greatest impact on aircraft size and performance as compared to other parameters related to solar panels themselves[25]. Hence, considering the time and resources available for this project, the decision of not having an onboard energy storage system is justified.
2) The time of the year has a significant impact on the power output of the PV module[24]. The winter solstice in the northern hemisphere, December 21, represents the minimum available solar energy and hence a module designed for these conditions will be operational at any time of the year.
3) As the location of the cells moves farther from the equator (latitude increases), the impinging solar power density decreases. It is preferable to fly at lower latitudes with high intensity of sunlight and shorter daytime periods than to fly at higher latitudes where days are longer[25].
4) The time of the day also influences the power output. Maximum power output is expected from 10:00 am to 3:00 pm when peak sunshine is available, as can also be seen from Figure 3.3.
5) As the altitude of flight increases, the aircraft size increases significantly because the decrease in air density requires an increase in wing area to generate the same lift. This is illustrated in the Figure 3.4. This is the reason why high altitude aircrafts by NASA have been having higher and higher aspect ratios (AR) over the years.

Figure 3.4: Effect of Latitude and Wingspan on Maximum Altitude.[24]

6) The mass of the solar panel also has an effect on the size of the aircraft but its impact is lower than that of cell efficiency.

In full overhead sun (1000 W/m^2 at Sea Level) the solar cells produce their rated power. Each cell produces 0.4 – 0.5 volts and the voltage output remains fairly constant with the change in intensity of sunlight. For a comparison of PV modules, their current ratings should be used[4].

CHAPTER 4

SIZING AND DESIGN OF THE AIRCRAFT

As stated in Chapter 1, this project aimed at making a flyable model capable of sustained level fight using only solar power, within the time frame and resources available. No particular mission was expected to be performed by the aircraft. The design of the model as well as the construction techniques were expected to have a major influence on the success of the project. Because solar cells were to be used as the only source of energy, it was imperative that the model be electric powered. A rough estimate of the targeted size of the aircraft was required at the early stages of design.

As the size of the aircraft increases, the weight and power requirements increase. An increase in power requirement translates into an increase in the number of solar cells and hence in the cost of the solar panel. In order to keep the costs affordable, it was necessary that the aircraft be as small as possible. This would also ensure that the aircraft is flyable in IIT itself. Choice was thus on the smallest possible size. On the other hand, Micro Air Vehicle (MAV) sized aircraft were not feasible because the required micro hardware is costly and not easily available, and also the design data readily available did not address these tiny sizes.

The weight of the model and the power requirements from the propulsion as well as controls side would have to be kept as low as possible. The weight and size of the components available put a lower limit on the aircraft size. A quick survey of commercially available miniature servos, electric motors and receivers revealed the weight, size, power requirements, cost and other specification of these components. The components used on a 42 gm R/C helicopter, Pixel, found to be the best suited in terms of their size and weight, and were selected as the onboard equipment[13]. Another search for R/C models that use these components led to Electric Slow-flyer aircraft. These R/C models had the following broad specifications[14, 15],

- Wing span of 0.5 m to1.0 m
- Weight: 100 gm to 150 gm
- Power requirement of 4 - 8 Watt
- Three channel control: One each for elevator and rudder control through servos (roll control is achieved through the rudder). The third channel is for throttle control through an electronic speed controller.

The above data on electric slow flyer formed a baseline design and it was thus decided that the model be made similar to Electric Slow-flyer aircrafts, which have recently gained popularity in the aeromodelling community. Specifications of various R/C model aircraft supplied by the manufacturers of the selected equipment, WES Technik, and others were also collected[14, 15, 16]. This data was used to obtain an estimate of the weight and size of the model. The power requirements of these models (4 W to 10 W), ascertained the feasibility of the model and that the aim of level flight could be achieved.

The hardware selected for propulsion and control was mainly based on the R/C models supplied by WES Technik. The individual components are listed in the appendix and a detailed specification of each is available from the manufacturer[15]. The power requirements and weights of the components are given in Table 4.1.

Component	**Voltage (V)**	**Current (A)**	**Power (W)**	**Weight (gm)**
Controls Unit				
Servo	5	0.1	0.5	3
	5	0.1	0.5	3
Speed controller	7	0.05	0.35	1.9
Receiver	5	0.05	0.25	4.3
Propulsion Unit				
Motor and Gear	6	0.55	3.3	14.2
Propeller				3.4
Total			**4.9**	**29.8**

Table 4.1: Power Break-up and Weights for the Hardware Components

4.1 Initial Sizing

Use of solar energy as an alternative to fossil fuels has been widely studied for applications ranging from household appliances to aircraft. The initial sizing of the aircraft was done by combining the information available from electric flying models and solar cell information available on the Internet[14, 15]. An estimate of the weight and power requirements of the propulsion and control units based on the data available from the manufacturer is listed in Table 4.1[15]. Using preliminary data available for typical solar cells, a first cut estimate of the solar panel area and weight were found[17].

A single solar cell gives a voltage of 0.4 V to 0.5 V and several of them can be connected to make a photovoltaic (PV) module fulfilling the desired power requirement. Silicon, Si, based solar cells are the most popular and are available in three types: Amorphous, Polycrystalline and Mono-crystalline, with mono-crystalline cells being the most efficient with efficiencies of 12% to 25%. From typical commercially available solar cells, a rough estimate of the solar panel weight and power output is as follows[17]

- Area = 100 cm^2
- Wattage = 1 Watt
- Weight = 10 gm

Power requirement of 4.9 Watts thus required solar cell area of 490 cm^2 and was expected to weigh 49.0 gm. This assumed a bright day with overhead sun and without cloud cover. From the typical weights of the models, the desired weight of the aircraft was targeted to be 125 gm[11, 13, 15]. This weight estimate, considering the fixed weight of the propulsion and control units (29.8 gm) and the solar panel (49 gm) left about 46.2 gm (125 – 29.8 – 49 = 46.2 gm) for the entire airframe and control linkages. The construction of such a light airframe was a challenging task and the design of the airframe also had to incorporate the safety of the solar cells and the onboard components in the event of a crash. It was expected that, as is the case with several aircraft in this category, the handling loads may exceed the in-flight loads. The design features related to these aspects were incorporated in the airframe construction. Figure 4.1 gives the weight break up.

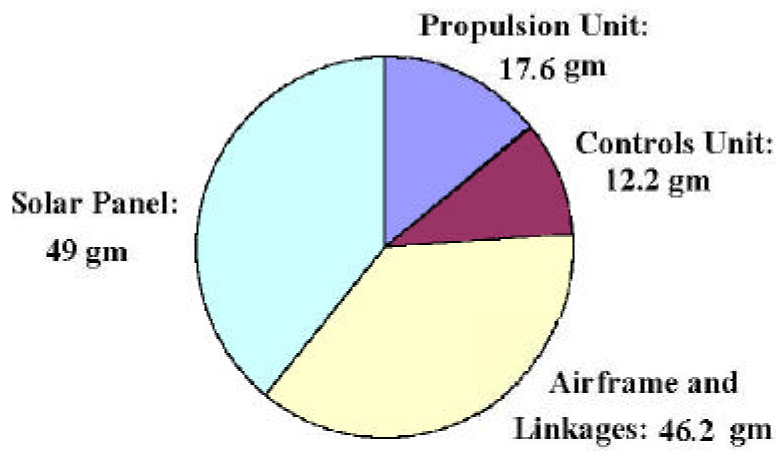


Figure 4.1: Weight Break-up for the Aircraft.

4.2 Wing Design

Wing loading of successfully flown R/C aircraft, W/S_W = 0.183 gm/cm^2 = 6 oz/ft^2, was taken as a ballpark figure[15]. For a weight, W = 125 gm, the wing area, S_W = 700 cm^2. Wing was the preferred location for mounting solar cells as it offered the flattest and contiguous surface and was naturally exposed to the sun during normal flying conditions. 490 cm^2 area of the wing was kept as flat as possible to receive the solar cells. In this aircraft, the wing performed two critical functions. The primary function of generating lift force and the other of providing a surface for mounting the solar panel that generates the power required for the entire aircraft. Most of the modern solar powered aircraft have aspect ratios, AR, in the range of 20 to 30 [1]. A higher aspect ratio directly manifests itself as a reduction in the power required for sustained level flight. However, a higher aspect ratio was difficult to achieve structurally because the stiffness of the wing would have to be maintained as the slenderness increases. Besides on this aircraft, the size of the solar panel also posed a constraint on the aspect ratio achievable because, for a given wing area, the chord could not be minimized below the width of the solar panel. A realistically achievable aspect ratio, AR = 7 was used in the calculations. Using this, the planform dimensions of the wing were obtained as:

Span, $b = (AR \times S)^{1/2} = 70$ cm ………………………………………. (4.1)

For a rectangular wing, the chord, $c = S / b = 10$ cm

Several iterations later a wing layout at Figure 4.2 was arrived at. Note the 50 cm mid span area aft of front spar was reserved for solar cells. Commercially available models have a full-length dihedral, which could not be provided here because the central portion of the wing was to be flat for the solar cells to get uniform sunlight. The rectangular mid section of the wing was the mounting platform for the solar panel, which was mounted in the shaded area in Figure 4.2. Wing tips over the last 13 cm were given a tip dihedral of 18.3 degrees. This arrangement was expected to have the following advantages:

1) The tips would provide lateral stability.
2) The dihedral would make sure that rudder control is effective for rolling.
3) The tip sections would also provide a surface for mounting extra solar panels, if required.
4) The mid section could be kept flat and its construction was made as simple as possible.

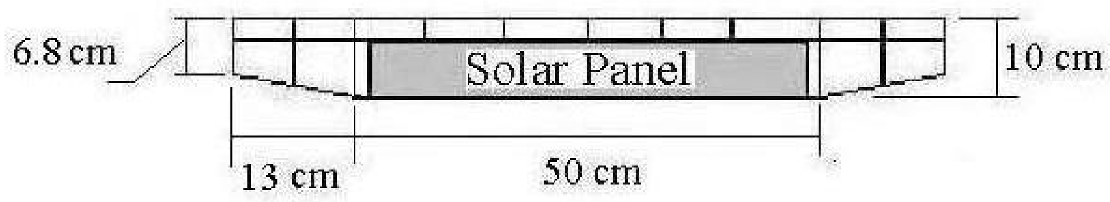


Figure 4.2: Wing Layout Proposed During Preliminary Design.

Wing had following specifications,

Span	=	76 cm
Spanwise extent of constant chord	=	50 cm
Constant chord	=	10 cm
Tip chord	=	6.8 cm
Dihedral angle of the tip	=	18.3 ^{o}
Wing area, S_W	=	700 cm^2
Aspect Ratio	=	7

The wing was made with a box type construction consisting of ribs and spars covered by a thin Mylar/tissue skin. The mounting of solar cells on the wing entailed a significant effort in the airfoil selection and construction, and is discussed below.

The flight speed for the aircraft was expected to be in the range of 2 to 5.4 m/s (for C_L of 0.1 to 0.8). The Reynolds number range corresponding to this was 13,700 to 37,000. The Reynolds number range suggested that the airfoil be a specifically designed one keeping very Low Reynolds number applications in mind. The airfoils used on the existing solar powered models or even popular model aircraft could not be simply scaled down geometrically. It was found that, at such Low Reynolds numbers, a curved plate gives a better aerodynamic performance than a flat plate or a two surface airfoil[18]. Figure 4.3 illustrates this observation.

Figure 4.3: Comparison of Low Reynolds Number Airfoils [17].

Data of “8.7 % Gewolbren Platte” airfoil was available from literature[27] and hence it was decided to make the airfoil according to that contour. This also revealed the possibility that the solar panel itself could act as the wing surface over most part of the wing. Recent empirical developments in Low Reynolds number aerodynamics brought to notice another peculiarity of the airfoil section to be selected. It was seen that, contrary to commonly used airfoils, it was favorable, in this case, to have a sharp leading edge and a blunt trailing edge[19, 20].

Figure 4.4 shows a rib section that gives shape to the single surface airfoil.

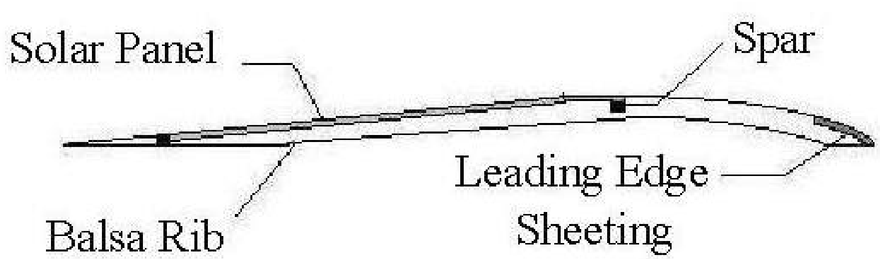


Figure 4.4: Rib Section Proposed During Preliminary Design.

The leading edge sheeting in Figure 4.4 was intended to provide the wing with a straight and sharp leading edge. Besides, the sheeting would crumple in the event of impact and absorb energy so that the solar cells were not damaged.

4.3 Empennage Sizing

The tail surfaces used are conventional ones consisting of a horizontal tail and elevator and a vertical tail with a movable rudder. The tail surface areas were calculated using the thumb rules given by Andy Lennon[21] as follows:

Horizontal Tail

Horizontal Tail Area, HTA = 17% S = 0.17 x 700 = 119 cm^2 ………. (4.2)

The Tail Moment Arm, TMA, was calculated as

$$TMA = \frac{2.5 \text{ x } MAC \text{ x } 0.2 \text{ x } S}{HTA} \quad \text{…………………………………..… (4.3)}$$

Where all linear dimensions are in inches and areas in square inches. MAC is the mean aerodynamic chord, c, in this case. Substituting these values, we get TMA = 31.76 cm or 12.505 in.

With an aspect ratio (Horizontal Tail), AR_{HT} = 4, we get the span, b_{HT} = 21.81 cm and chord, c_{HT} = 5.45 cm

The elevator area was taken as 40% of the Horizontal Tail Area.

Vertical Tail

Similarly, the Vertical Tail Area, VTA, was calculated as

VTA = 8% S = 0.08 X 700 = 56 cm^2 ………………….……………… (4.4)

And an aspect ratio (Vertical Tail), AR_{VT} = 2, we get height, h_{VT} = 10.58 cm and chord, c_{VT} = 5.29 cm

The rudder area was taken as 35% of the Vertical Tail Area.

The specifications for the tails surfaces are provided in Table 4.2.

Horizontal Tail		**Vertical Tail**	
Area	119 cm^2	Area	56 cm^2
Span	19 cm	Height	9.5 cm
Root Chord	6.8 cm	Root Chord	6.8 cm
Tip Chord	5.8 cm	Tip Chord	5 cm
Aspect Ratio	3	Aspect Ratio	2
Tail Moment Arm	31.76 cm	Tail Moment Arm	31.76 cm

Table 4.2: Empennage Specifications for Preliminary Design

4.4 Configuration

It was decided that the aircraft would be conventional in design with a fixed main wing and tail surfaces and movable control surfaces in a tractor configuration. Two configurations shown in Figure 4.5 were considered. Twin boom configuration was expected to be structurally lighter and would allow ease of aligning the horizontal tail with the wing. However, if the twin boom configuration from Figure 4.5 was chosen, the possibility of two vertical tails might have had to be considered. In that case both the rudders would have to be connected to the servo actuator using different control linkages (push rods or pull-pull cables). Routing of control linkages in twin boom configurations was complex, and hence twin boom was not preferred. The single boom configuration was chosen.

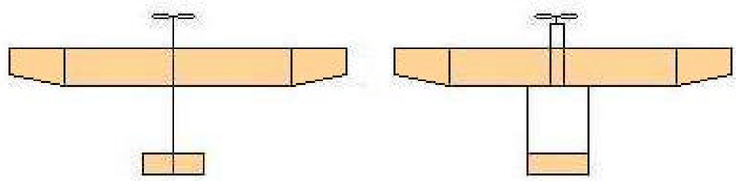
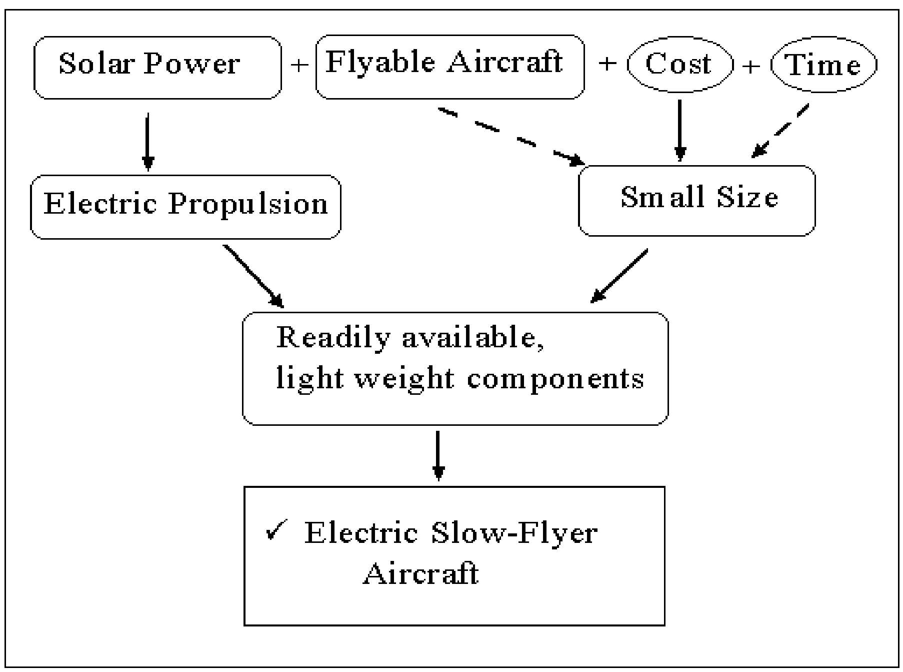


Figure 4.5: Configurations Considered

The overall design methodology used for the aircraft is summarized in Figure 4.6.



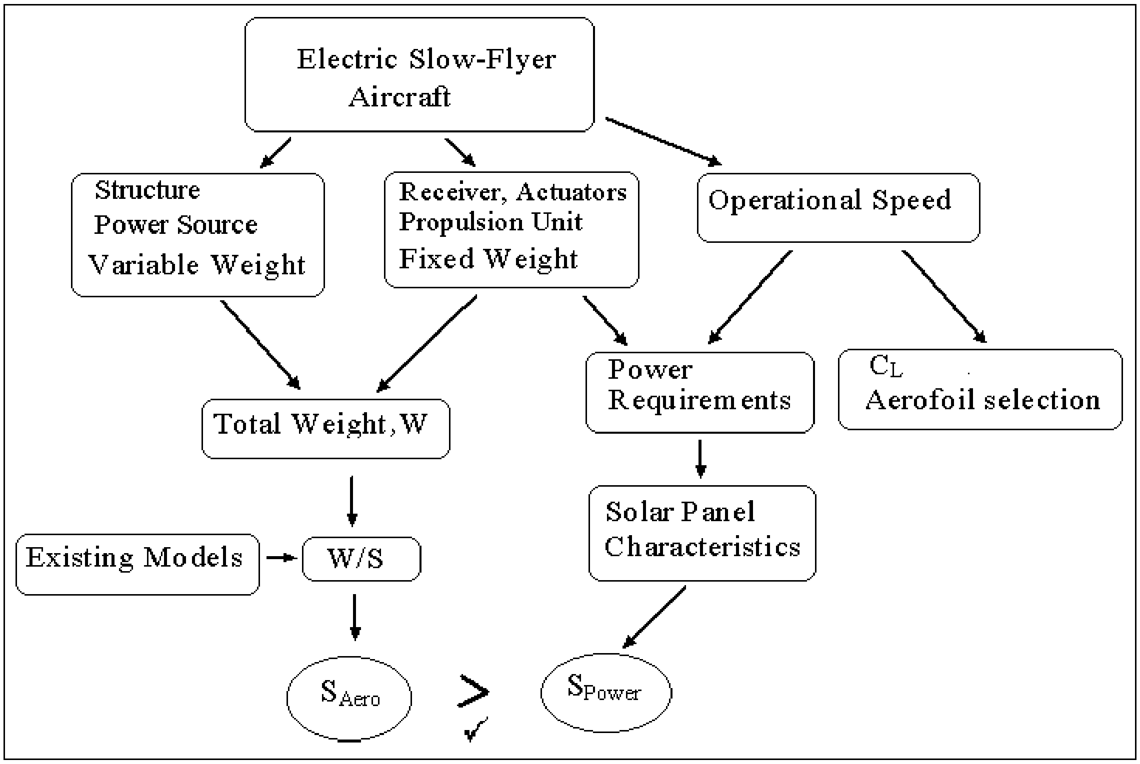


Figure 4.6: The Overall Design Methodology Used.

CHAPTER 5

TESTS AS AN UNPOWERED GLIDER

After the sizing of the aircraft was done as shown in Chapter 4**,** it was decided that a preliminary model be made so that an estimate of the aerodynamic characteristics could be obtained. Also, it was decided that the airframe would be made as light as possible its weaker portions be reinforced appropriately in the subsequent attempts. This would also ascertain if the task of building the airframe could be successfully completed using the known construction techniques and materials.

The dimensions and areas obtained in Chapter 4 were used to construct a glider airframe using conventionally used balsa wood. A circular arc airfoil with 5% camber was used on the wing. The wing and tail surfaces were then covered with kite-paper, which was later shrunk with the help of moisture to obtain an adhering skin contour. A suitable fuselage consisting of thermocole and balsa was constructed and appropriate ballast put to ensure that the Center of Gravity, CG, location was maintained. A picture of the glider is shown in Figure 5.1.

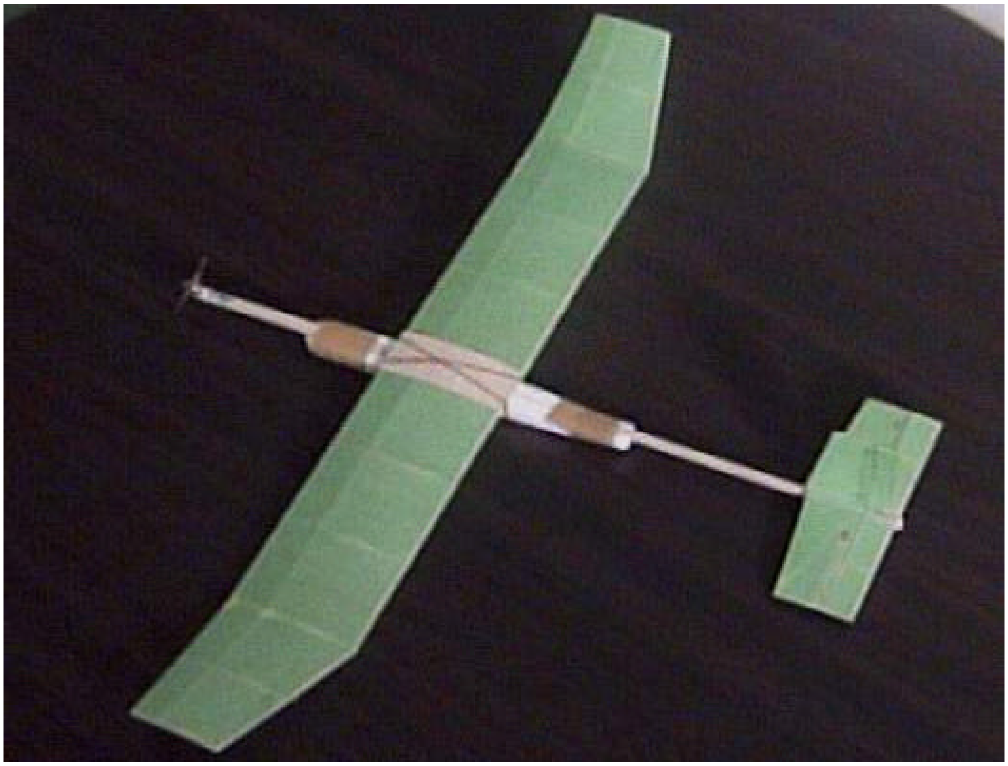


Figure 5.1: The Glider Used for Initial Tests.

The weight of the individual parts was recorded as given in Table 5.1.

Part	**Weight before covering (gm)**	**Weight after covering (gm)**
Wing	6.94	10
Tail surfaces	3	5
Fuselage	16	16
Ballast	10	10
Total	**35.94**	**41**

Table 5.1: Weight Break-up for the Glider.

The glider was then hand launched and it was found that, when launched from a height of 1.21 m (4 ft), the glider typically traveled a distance of 9.14 m (30 ft). This was used to calculate the Lift to Drag ratio, L/D, for the model. Figure 4.2 illustrates the trajectory.

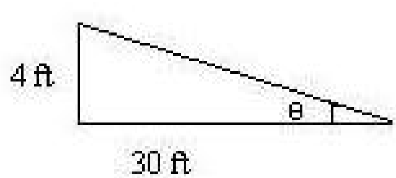


Figure 5.2: Calculation of the Glide Angle

Therefore, $\tan\theta = D/L = 1.21 / 9.14$ and $\theta = 7.6^o$ ………………… (5.1)

From Equation (5.1) it was seen that L/D = 7.5. This ratio was then used to calculate the drag and hence the thrust required for level flight as follows.

For steady level flight, the lift and weight should be balanced

Therefore, $L = W$ …………………………………………………… (5.2)

$T = D$ ……………………………………………………….. (5.3)

From Equations (4.2) and (4.3), we got

$$T = \frac{W}{L/D} = \frac{125}{7.5} = 16.667 \text{ gm} \quad \ldots\ldots\ldots \quad (5.4)$$

Thus, the thrust requirement was estimated to be T = 17 gm approximately.

5.1 Results of the Glider Tests

The following results were obtained from the glider tests:

1) The parts weight shown in Table 5.1 ascertained that it would be possible to construct the airframe within the desired weight of 46 gm. The total weight of the glider, 41 gm included 10 gm ballast weight, which could be the propulsion unit itself. So, effectively a margin of 15 gm still remained for reinforcements and control linkages.

2) The glider was tail heavy and ballast was required for maintaining CG. The weight of the solar panel would even worsen the problem. It would be necessary that the subsequent models have lighter tail surfaces.
3) The kite-paper used for covering was quite heavy as compared to the wing itself. Hence a suitable material like lightweight Mylar should be used. This was also found to be available from WES Technik. Besides, this material would also be stronger than kite-paper.
4) The construction of the glider, though without any reinforcements, was stiff enough to sustain the flight loads. Hence, suitable reinforcements were required only to prevent the solar cells from getting damaged.
5) The required thrust was T = 17 gm and from the manufacturer's data[15] showed that the available thrust will be about 43 gm. Thus, the thrust requirement could be fulfilled easily.
6) The glider was very sensitive to winds. Hence, ideally the model should be flown indoors. But this would be impossible because of the sunlight requirements. Thus, an open space with very low winds (< 1 m/s) would be required.

CHAPTER 6

TESTS ON HARDWARE COMPONENTS

When the propulsion and controls units arrived, an attempt was made to use them with batteries and verify their working. In the first attempt, Li CR2032 batteries were used but they did not provide enough power to run the components. Hence, a regular 9 V battery was used and, because the receiver required a voltage between 4.8 and 6 V, a regulator was used to convert the 9 V to 5 V for the receiver. The motor still operated at 9 V.

6.1 Thrust Measurement

An idea of the actual thrust provided by the propulsion unit was required. Hence, a setup for thrust measurement was designed. The propulsion unit was powered using a battery and was clamped on an aluminum pipe mounted vertically on a thermocole base, as shown in Figure 6.1.

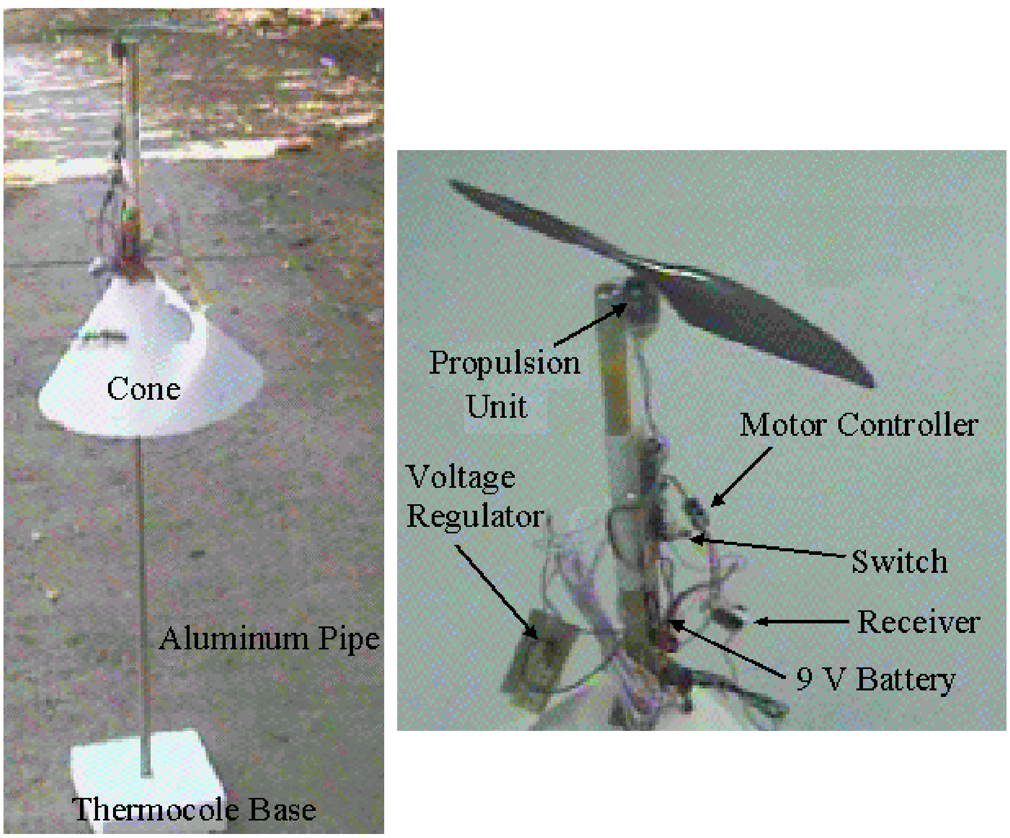


Figure 6.1 Thrust Measurement Setup

The whole setup was then put on a digital weighing scale and the difference in the weight before and after turning on the motor was recorded. Since the prop-wash was facing downwards, there was a possibility that the readings would be affected because of ground effect. However, this was avoided by keeping the propeller at a considerable distance from the ground and deflecting the prop-wash away from the weighing scale, by a cone arrangement. The thrust readings were taken by supplying power through a battery as shown in Table 6.1.

	5 V Battery	9V Battery
Weight (Before)	262.0 gm	284.6 gm
Weight (After)	240.6 gm	242.0 gm
Thrust Available	21.4 gm	42.6 gm

Table 6.1: Calculation of Thrust Available

During the first attempt, the RPM vs Current readings were taken and are shown in Figure 6.2.

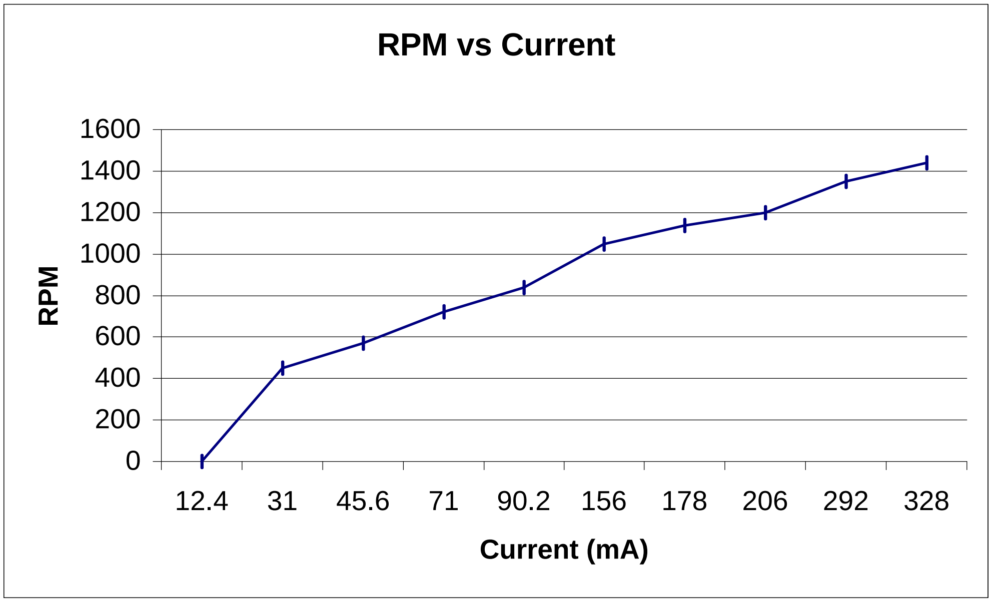


Figure 6.2: RPM vs Current Curve for the Propulsion Unit.

6.2 Measurement of Current Requirements of Servo Actuators

Servos were tested, one at a time, to get an idea of the power requirements under various loads. The input voltage is given to the servo through the receiver and hence was 5 V. Different weights were attached to the servo arm and lifted by the servo. Table 6.2 shows the weight and current readings. It was seen that the servo drew a significant amount of current in the transient state as it was moved. This current ranged from 250 to 285 mA.

Weight (gm)	**Steady State current (mA)**
10	13.9
20	13.9
30	13.9
40	12.96
50	12.96
60	12.96

Table 6.2: Steady State Current Drawn by the Servo Actuator.

CHAPTER 7

SOLAR PANEL FOR THE AIRCRAFT

This chapter discusses how the requirements for the solar panel were worked out based on Chapters 3 and Chapter 4. The specifications of the solar panels provided by the manufacturer, Tata-BP Solar, Bangalore are also provided.

7.1 Requirements Capture for the Solar Panel

Some requirements for the solar panel required for the project were worked out on the basis of the available power requirements and the tests done on the components as follows:

1) The PV module should satisfy the power requirement of all the onboard components.
2) The solar cells should be highly efficient, preferably monocrystalline Si cells.
3) The weight of the panel should be as low as possible.
4) Most of the commercially manufactured modules are for terrestrial applications. The module in this case should be flexible and may have to be custom-made.
5) The panel should be modular in nature. Different but identical modules for receiver, servos and propulsion unit will be necessary. This will ensure that the entire panel need not be replaced in the event of damage to a certain module.
6) The cells should be covered with a suitable material to make them resistant to impact and moisture.

Based on these requirements, a manufacturer, Tata-BP Solar (Bangalore) were contacted and it was decided that solar panels be procured from them. The following requirements were given to them:

1) Total power source of 8.5-9V current 600mAmp.
2) The power source could be split in two modules: a) Two connected in series 4.5 volts and 500mAmp b) Two connected in parallel 9 volts 250mAmp each.
3) Following specifications were given for the solar cells:

   Area available: 7 x 70 cm = 490 square cm

   Weight: Below 55 gm

   Output Voltage: 8.5-9 V

   Power (on December afternoon sunlight): 5.5 W

7.2 Solar Panel I

Based on the given requirements, the manufacturer first attempted to supply a panel that would give an idea of what could be possible. The first attempt by the manufacturer arrived towards the end of January 2002. The specifications of the panel were:

Power: 2.5 W

No of Cells: 10

Cell efficiency: 13.6 %

Fill factor: ~ 0.6

Size: 35 x 7 cm

Weight: 46 gm

Voltage (open circuit): 5.35 V

Performance under loads:

10.2 : 4.45 V & 306 mA

14.7 : 4.75 V & 195 mA

The following observations were made from the panel:

1) The weight of the panel was nearly double of the required weight. This was mainly because of a heavy substrate used by the manufacturer as the base for laying the solar cells. It was suggested that a lighter substrate be used for the future panels.
2) The panel could not satisfy the power requirements of the components. Especially the high transient currents drawn by the servos led to the voltage dropping to such an extent that the digital micro controller, which needed at least 4.8 V, stopped functioning when servo inputs were given. This led to a complete loss of thrust which could be adjusted only after fully lowering the thrust and keeping it at "zero" position for one second, for the controller to start functioning again.

7.3 Solar Panel II

Following the tests on Solar Panel I, the manufacturer was asked to supply the entire panel to be used on the final design. The second solar panel was available by the end of February 2002. The panel's specifications were:

Power: 4.43 W

No of Cells: 18

Cell efficiency: 13.7 %

Fill factor: ~ 0.68

Size: 60 x 7.5 cm

Weight: 70 gm

Voltage (open circuit): 10.87 V

Current (short circuit): 593 mA

Performance under loads:

10.2 : 4.87 V & 480 mA

14.7 : 8.08 V & 310 mA

20.4 : 7.6 V & 360 mA

The second panel was comparatively lighter and more flexible than the previous one. The panel, however, was slightly curved and hence the mounting on the wing had to be done carefully. The Voltage vs Current graph for the panel is shown in Figure 7.1.

Figure 7.1: Current vs Voltage Curve for Solar Panel II.

It was seen that the panel exceeded the weight limit by 20 gm (~ 36 %). This increase in the weight of the panel made it necessary that the airframe be made as light as possible. The panel still did not deliver the power that was specified in the requirements. However, attempts to fly the model with this same panel were made. It was observed from the panels that it was not entirely possible for the suppliers to manufacture panels suitable for the project's requirements because they were not familiar with manufacturing such panels. However, an appreciable amount of effort was spent by them to manufacture the panels that would be most suitable for the given requirements.

CHAPTER 8

ATTEMPTS AT POWERED FLIGHT

After the solar panels were procured, their mounting on an aircraft wing and powered flights were explored. This chapter provides the details of the designs which were tested, the observations made and the changes that were incorporated.

8.1 Powered Glider, Design I

The glider used for testing and described in Chapter 5 was used for the first powered flight trials also. The heavy tail surfaces used in the glider were replaced by lighter ones and a carbon fiber rod was used for the fuselage because it proved lighter and stronger than the previous thermocole and balsa combination. The Solar Panel I was mounted on the wing of the glider with the help of cotton thread and tape. As mentioned in Chapter 7, Panel I was quite stiff and hence a portion of the wing, which had a circular arc airfoil, was flattened to mount the solar panel. The Propulsion and controls units were mounted on a balsa body and attached to the carbon-rod fuselage. Carbon fiber rods were used as pushrods for the control linkages. Since the glider was tail-heavy, the propulsion unit was mounted so as to adjust the Center of Gravity, CG, at 25 % of the Mean Aerodynamic Chord, MAC. A sketch of the powered glider and its picture are shown in Figure 8.1.

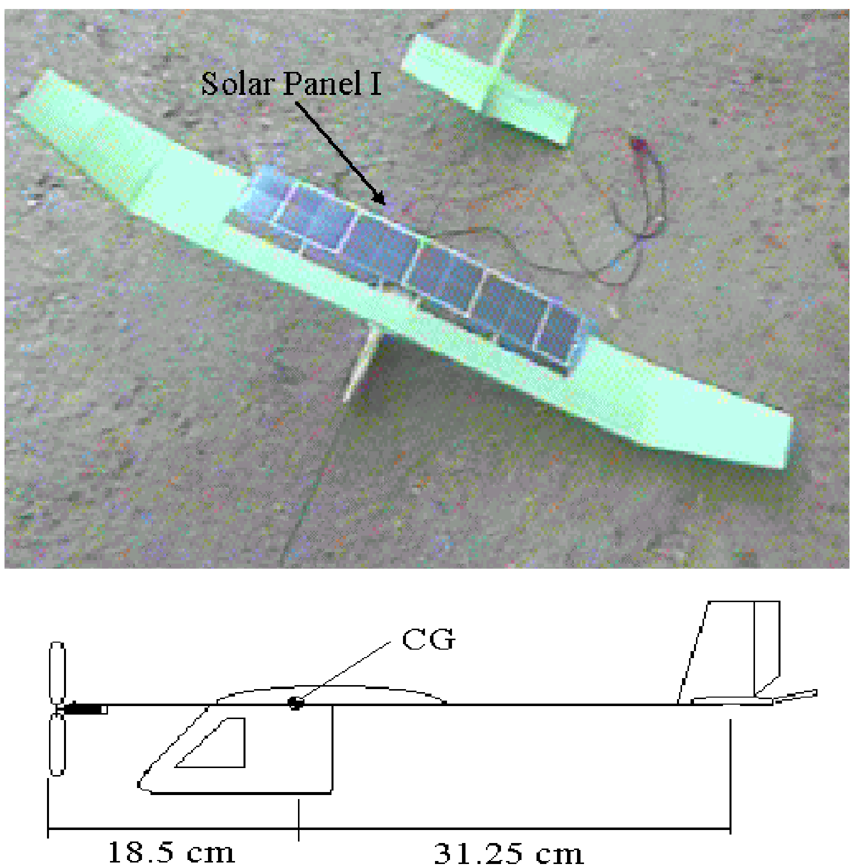


Figure 8.1: Side View Sketch and Photograph of Deign I.

Observations from the powered tests were as follows:

1) Though the motor and servos functioned properly when tested individually the transient current drawn by the servos caused the voltage to drop to such an extent that the digital micro controller stopped functioning. This presented a major problem because the throttle had to be again brought back to the minimum level and held for 1 second for the controller to start functioning again. Any movement of the control surfaces would thus cause the propulsion unit to stop.
2) An incidence angle of about 2^o was provided while mounting the wing but proved to be insufficient. The need for a wing with a changeable incidence, or a movable stabilizer, was felt but could not be incorporated in the glider.
3) The aircraft showed virtually no effort to lift itself even when hand launches were tried at sufficient speeds by running with the aircraft. The glider needed higher speed or more area to generate the required lift.
4) The solar panel thickness led to a protruding surface on the airfoil possibly disturbing the air flow and hence the lift. It was thus decided that the panel must be mounted after providing a groove on each rib so that it sits flush with the airfoil contour.
5) Since Panel I supplied only 2.4 W of power, the thrust generated was inadequate for the aircraft to be able to sustain flight.
6) However, the total weight of the aircraft turned out to be only 100 gm leading us to a confidence that the structural weight would remain within the limits. The structural weight of the model was only 100 – 46 – 29.8 = 24.2 gm.

8.2 Modified Design, Design II

The same glider was modified and flown again after some deliberation about the possible sources of the problems. In most commercially available models the propulsion unit was mounted fairly close to the wing LE and the Tail Moment Arm was also quite small as compared to the glider used for testing. Also, the deployable control surfaces were quite large as compared to the ones used here. This could lead to the following effects:

1) When the Propeller is close to the wing LE, the slipstream will effectively increase the air velocity seen by the wing. This could lead to an increase in the lift generated by the portion of the wing facing the slipstream.
2) A smaller fuselage length would obviously reduce weight by a small amount.
3) In mounting the propulsion unit farther away from the CG, as ballast, the Moment of Inertia in the pitch axis increases considerably because it depends on the square of the distance. Though this does not affect the static performance of the aircraft after it has reached a certain trim, it does affect its ability to reach the trim from the position when it is hand launched. This could be a factor rendering the elevator less effective.

The fuselage was shortened and the hardware was rearranged so as to keep the CG at the same position as earlier. A sketch of the modified glider is provided in Figure 8.2.

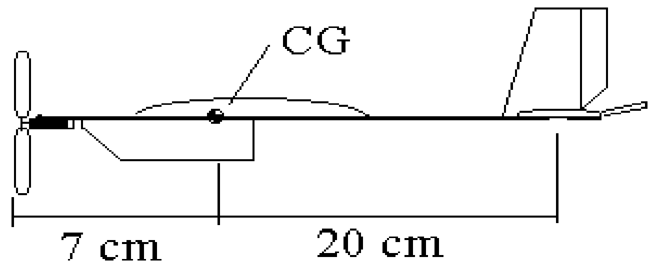


Figure 8.2: Side View Sketch of Design II.

The aircraft was again tested by a hand launch. Though problems of limited power and thrust capability remained, at one instant the glider did show a tendency to lift when it encountered a gust. Performance against the wind did provide a confidence that, at sufficient speeds, flight could be possible. The weight of this design reduced to 95 gm.

8.3 The Current Design, Design III

The third aircraft was designed keeping in mind the new solar panel. The size of the wing had to be increased because the solar panel was bigger than the required size. Besides, observations of the first two designs also called for an increase in wing area. The wing was made to the following specifications and is shown in Figure 8.3.

Span	= 90 cm
Spanwise extent of constant chord	= 62 cm
Constant chord	= 12 cm
Tip chord	= 8.0 cm
Dihedral angle of the tip	= 16^o
Wing area, S_W	= 1013 cm^2
Aspect Ratio	= 8
Weight, W	= 140 gm
Wing Loading, W/S_W	= 0.138 gm/cm^2

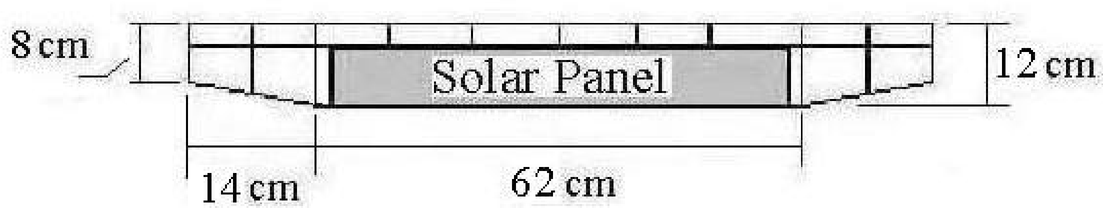


Figure 8.3: Wing Planform for Design III.

A generous dihedral of 16^o was given with the intention of making the glider more responsive in roll when rudder was applied. Two of the spars of the wing were reinforced with carbon fiber strips. This increased the strength significantly because the “composite” spar was an effective inverted `T’ section. The LE spar was made with thin 1/16 inch square balsa. As explained in Chapter 4, the LE spar was intended to absorb energy while breaking so that the cells do not get damaged. The wing tips were also made without any reinforcements for the same reason. The rib section was slightly different from the one in Figure 4.4 and is shown in Figure 8.4. A dowel and rubber-band arrangement was made for mounting the wing over the fuselage. This ensured that the angle of incidence could be changed easily and that the wing would easily detach from the aircraft in the event of a crash. The solar panel was mounted after making a groove on each rib so that it remained flush with the covering and did not disturb the flow. The middle-spar was bent chordwise slightly so that it provided an edge to firmly glue the curved solar panel.

The tail surfaces were made according to the estimates from the second design. And the details are given in Table 8.1. The elevator area and rudder area were increased to 50% HTA and 40% VTA respectively because the Tail Moment Arm had been scaled up from Design II.

Horizontal Tail		**Vertical Tail**	
Area	202.6 cm^2	Area	101.3 cm^2
Span	24.5 cm	Height	11.5 cm
Root Chord	9.5 cm	Root Chord	10.3 cm
Tip Chord	7.5 cm	Tip Chord	8.2 cm
Aspect Ratio	3	Aspect Ratio	1.3
Tail Moment Arm	25 cm	Tail Moment Arm	25 cm

Table 8.1: Empennage Specifications for Design III.

Figure 8.4 gives the side view of the aircraft showing the TMA and the distance between the CG and the propeller.

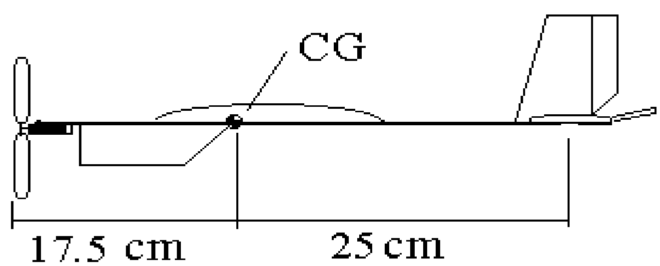


Figure 8.4: Side View Sketch of Design III.

8.4 New Hardware Added to Design III

The voltage requirements of the components required that certain other electronic hardware be added for their proper functioning. The details for these additions and their effects are provided below.

1) Solar Panel II supplied an open circuit voltage of 10.87 V. However, the receiver and the servos required an operating voltage of 4.8 to 6 V. Hence a voltage regulator, LM 2940CT, was added as an onboard component and it converted the solar panel output voltage to 5 V for the receiver. This led to a weight penalty of 4 gm.

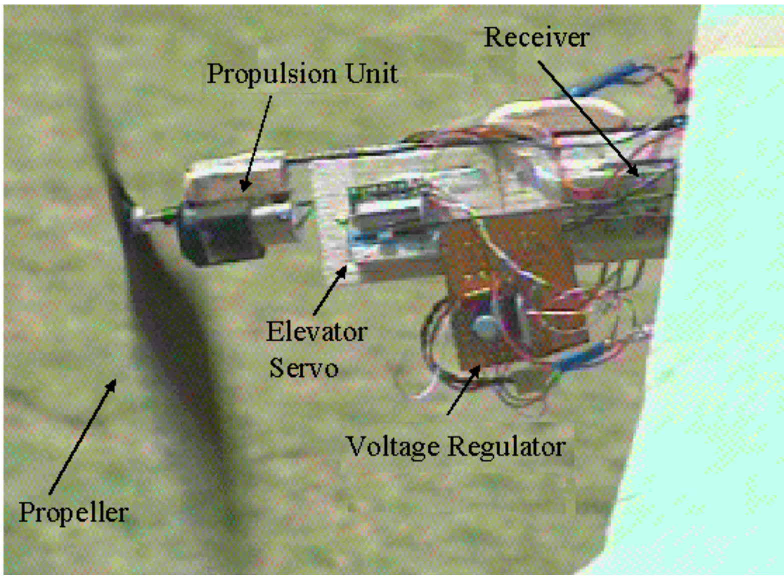


Figure 8.5: Photograph Showing the Voltage Regulator.

A photograph showing the regulator is shown in Figure 8.5.

2) The micro controller (throttle control) operates in a voltage range of 6 to 10.8 V. If voltage drops below 4.8 V, a reduction in RPM is found and the throttle has to be brought back to minimum in order to start the controller's functioning again. As shown in Figure 7.1, the output voltage of the solar panel drops significantly beyond a certain voltage range. In this case, the high transient currents drawn by the servos led to a drop in the voltage to an extent that the motor stopped working when servo inputs were given. In order to overcome this problem capacitances were added in parallel to the solar panel output. These capacitances would store charge and dissipate it providing adequate current to cater to the transients. The problem was solved to a large extent only after the addition of two 1000 mF capacitances, one of which is shown in Figure 8.6.These capacitances however, added about 6 gm to the weight of the model. Since, rudder as well as elevator control were crucial, it was decided that the weight penalty be accepted in order to ensure the functioning of all controls.

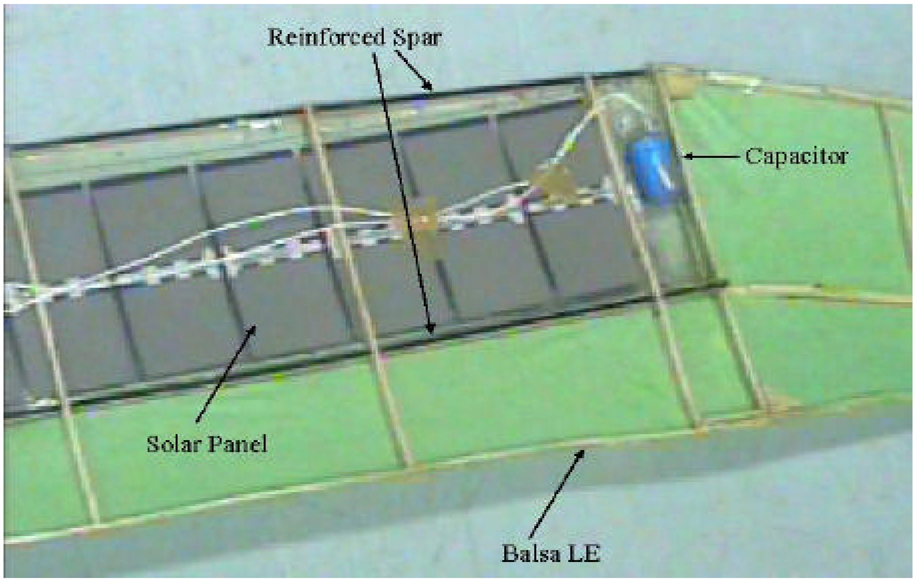


Figure 8.6: Photograph Showing the Capacitor Mounting.

The addition of new hardware components as well as the fact that the solar panel was 20 gm heavier than expected changed the weight breakup significantly as compared to Figure 4.1. The new weight break-up is shown in Figure 8.7. The total weight of the aircraft, Design III, was 140 gm.

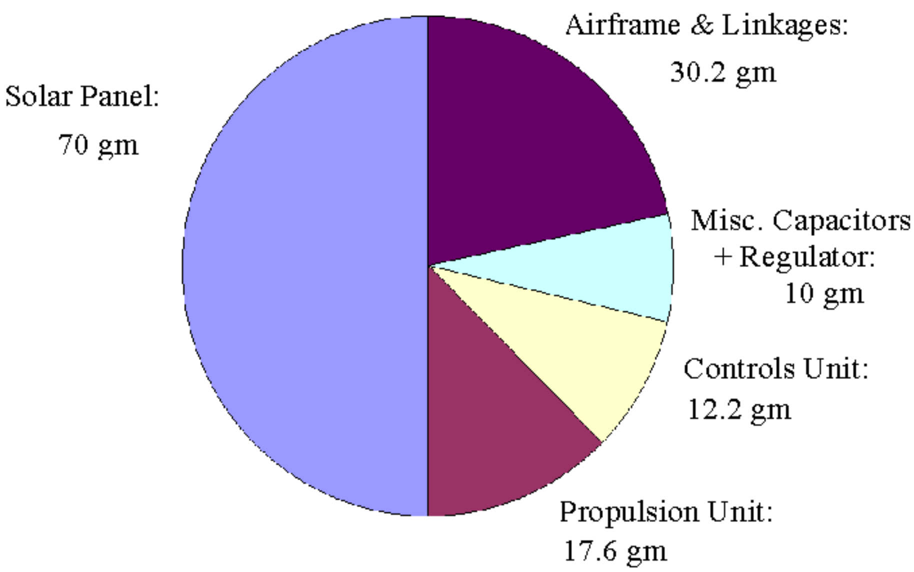


Figure 8.7: Weight Breakup for Design III.

It can be seen that the major part of the aircraft was the solar panel itself. The weight of the airframe was significantly lower as compared to the hardware components put onboard. The use of balsa and carbon fiber reinforcements led to such a low weight airframe. However, the target weight of 125 gm which was set in Chapter 4 was exceeded. But this was compensated by an increase in the wing area that reduced the wing loading to 0.138 gm/cm^2 (4.52 oz/ft^2) , which was even lower than the wing loading expected in Chapter 4. Photographs of Design III are given below in Figure 8.8.

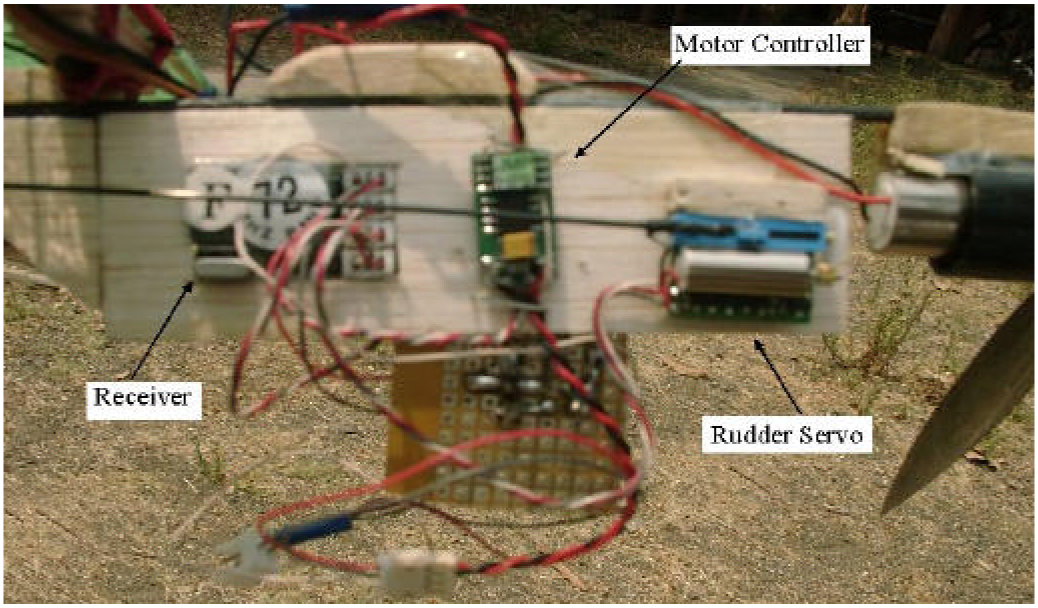


Figure 8.8: Design III.

8.5 Flight Tests of Design III

This design was also tested by a hand launch with full throttle. The following observations were made from the tests:

1) The glider test of Design III showed, by a calculation similar to Chapter 5, that the L/D ratio was 5.7 and the required thrust, for a weight of 140 gm, would be 24.56 gm.

2) The aircraft did show enough lifting capacity and it ballooned significantly if launched at a higher speed than necessary. The wing's lifting capacity was adequate but it was seen from the videos that the aircraft slowed down after being launched. The thrust available was not adequate to keep the aircraft flying at the level flight speed. The RPM range achieved was 1500-1600 which corresponds to a thrust of about 20 gm. Hence an increase in thrust was required.
3) The solar panel was mounted on the aft portion of the wing as shown in Figure 8.3 because it was the flattest portion available. This however, put the panel at an angle with the sunlight during level flight reducing its output to some extent. If the aircraft was flown in a direction so that the panel faced the sun, it was advantageous in terms of the power output but on most days, the wind direction did not permit flying the aircraft in such an advantageous direction.
4) The aircraft was still extremely sensitive to winds. Especially, since aileron control was not provided, it was not possible to get the aircraft level, using rudder alone, once it had banked due to the winds. The coupling between roll and yaw axis was not adequate to control the aircraft against a roll induced due to the winds. Hence, it was decided that the dihedral angle of the tips be reduced to make it less responsive to lateral gusts. The dihedral angle was reduced from 16^o to 8^o. However, the problem was not overcome entirely.
5) Since the flight speed of the aircraft was quite low (less than 5 m/s), the wind speeds it faced were comparable to its forward speed. Hence it was necessary that the aircraft be flown at a large yaw angle. However, it was not possible to obtain such high angles using the rudder. It was necessary that the aircraft be flown in almost "nil-wind" conditions.
6) The use of solar power presented a special problem aggravating the problem listed above. In order to get the maximum sunlight, all tests were conducted as close to noon as possible. In that case, any roll angle induced by the winds changed the angle at which the solar panel faced the sun. Because of this, the power output of the panel decreased leading to a drop in the thrust and the controls were also affected. In order to maintain sufficient power output from the panel a roll angle of less than 10 degrees could only be allowed.
7) The features incorporated to reduce impact damage, like non-reinforced wingtips, rubber-band attachment and weak Leading Edge, LE, did prove to be useful in preventing damage to the solar panel as well as the other parts of the aircraft.

CHAPTER 9

SCOPE FOR FUTURE WORK

The flight tests of Design III showed that the aircraft was not able to sustain level flight on its own power. This chapter lists the problems currently faced, possible solutions and scope for future work in this field.

9.1 Problems with the Current Design

The observations from the flight tests of the various designs revealed some problems which are listed below.

1) The solar panel was not able to provide enough power to enable the propulsion unit to give adequate thrust required for level flight. An increase of about 5-6 gm in the available thrust would improve the situation to a large extent.
2) The Current vs Voltage curve of the solar panel was not designed keeping in mind the high transient currents drawn by the servo actuators. The capacitances added in order to solve this problem added to the weight of the aircraft and the wires used for their connections increased the drag.
3) All the three designs were very sensitive to winds and, because their flight velocities were very low, it was difficult to control them even in slight winds. It was not possible to find a time and place where adequate sunlight was available with no winds.
4) The solar panel was mounted on the aircraft with level flight in mind. Because of this, any bank induced either by winds or control inputs led to the panel facing the sunlight at an angle. This reduced the power output of the panel significantly, dropped the propeller RPM leading to a reduction in thrust, and hence flight speed. This further reduced the rudder effectiveness and it became almost impossible to recover from a bank.
5) Roll control on the designs was provided with rudder which was a “secondary” effect. It was observed that adequate roll response was not available using rudder alone.
6) The aircraft structure was made so light that extreme care was required to handle and launch the aircraft. Parts of the aircraft like the wingtips and the LE spars, which were designed to break on impact, required frequent repairs and replacement.

9.2 Changes Possible on Design III

The overall design process and flight tests do provide a feeling of confidence that if the problems mentioned in the previous section are overcome, sustained flight could be achieved. The following improvements are suggested on the current designs and are easily implementable.

1) The capacitors used were 25 V capacitors and replacing them with smaller 10 V capacitors would reduce weight to some extent. A smaller and lighter regulator and thinner connecting wires can be used.

2) During the initial stages of hardware procurement, light weight batteries suitable for the model should have been procured. Flying the aircraft with regular battery power would enable us to solve the trimming problems without having to bother about the effects of the solar panel.
3) The possibility of using ailerons for roll control can be studied.
4) A suitable enclosed space with adequate sunlight can be found for further tests.
5) A solar panel which is able to provide more power and satisfy the transient current requirements can be procured. A modular solar panel would be suitable because it would enable us to provide power to the propulsion unit and the controls unit separately eliminating the effect of the servo transient currents on the motor performance.

9.3 Suggestions for Future Work

The field of UAV/MAV and Solar Powered Aircraft shows a tremendous potential for progress in the coming years. The advantages of these types of aircraft, mentioned in Chapter 1 and Chapter 2, have attracted a large number of research establishments to pursue research in this field. On the basis of the present project, some suggestions are made below in regard to solar powered aircraft, with the hope that further research will be continued in this direction. The following activities/suggestions can be taken up in the future as complete projects or in parts.

1) The effect of location, time of day, date and panel inclination on solar panel output should be quantifiably studied and the design of the solar panel should be incorporated with the design of the aircraft.
2) It was observed that, in the present project, the interaction with the solar cell manufacturer was insufficient to ensure the delivery of a panel that fully satisfied all requirements. Because the panel was the most critical component for the aircraft and the fact that it was the first attempt by the manufacturer to fabricate a panel for such an application, it was necessary that the design of the solar panel as well as the aircraft should have been concurrent with both the teams being aware of the design process. Some aspects of the design of the solar panel should have been studied in detail and not left to the manufacturer alone.
3) The possibility of having an onboard energy storage system should be explored to ensure that future aircrafts are not affected by temporary disruptions in direct sunlight. The possibility of having a movable panel to enable tracking of sunlight or putting solar panels on the tail surfaces/fuselage can also be worked on.
4) Though the project had no mission profile specified, projects with specific missions can be taken up when adequate knowledge and experience about solar powered aircraft is achieved.
5) The hardware procured, literature collected and experience gained as a part of this project provide a good staring point and should be effectively used in the design and construction of Miniature or Micro Air Vehicles.

CHAPTER 10

CONCLUSIONS

The tasks leading to the successful completion of the project were implemented as planned in Table 1.1. The following conclusions can be drawn regarding Stage I of the project, which was completed by December 2001.

1) Studies regarding the feasibility of the project were conducted and relevant literature survey performed.
2) The type of aircraft, Electric Slow-Flyer, was selected.
3) The necessary hardware components were identified and procured.
4) Preliminary sizing and weight analysis were carried out.
5) A model was tested as a glider in order to verify the suitability of known construction techniques and estimate power requirements.
6) The requirements capture for the solar panel was completed and the manufacturer was contacted in order to ascertain the availability of the required type of solar panel.

The tasks completed in Stage I have helped in gaining confidence regarding the completion of the tasks scheduled for Stage II. As shown in Table 1.1, Stage II comprised of Solar Panel procurement, detail design of the aircraft, fabrication and flight-testing. The following points can be noted about Stage II:

1) The tasks of hardware testing, procurement of solar panel, detail design and flight testing were completed.
2) The problems faced in the completion of the above tasks were identified and corrective actions were taken. It was necessary to modify the design of the aircraft in order to achieve the desired goal of sustained level flight.
3) Because the aircrafts could not maintain level flight, an attempt was made to understand the causes of the failure and review the decision making process. Suggestions for further work in this field have been presented.

With the world record for a fully solar powered aircraft being at a weight of 159.5 gm and a wingspan of 1.11 m, making an aircraft both lighter (125 gm) and smaller (<1 m) proved to be an ambitious task. Though level flight could not be achieved, the problems were identified and the necessary modifications were suggested. The project can be termed as partially successful but ensures a high chance of success in further attempts.

Appendix

No.	Component	Product name	Remarks	Quantity required
1	Propulsion Unit	DC 5-2.4	Motor + 11.8:1 gear	1
2	Propeller	25 / 12 carbon fiber	Twin blade carbon fiber	1
3	Servo actuator	LS 3.0	For elevator and rudder control	2
4	Receiver	R4P JST 72 MHz	Control input detection	1
5	Speed controller	JMP-HF9	For throttle control	1
6*	Voltage Regulator	LM 2940CT	For regulating receiver input to 5 V	1
7*	Transmitter	Futaba 72 MHz	To transmit the pilot's commands	1

* Not shown in the Figure A1

Table A1: List of Onboard Components

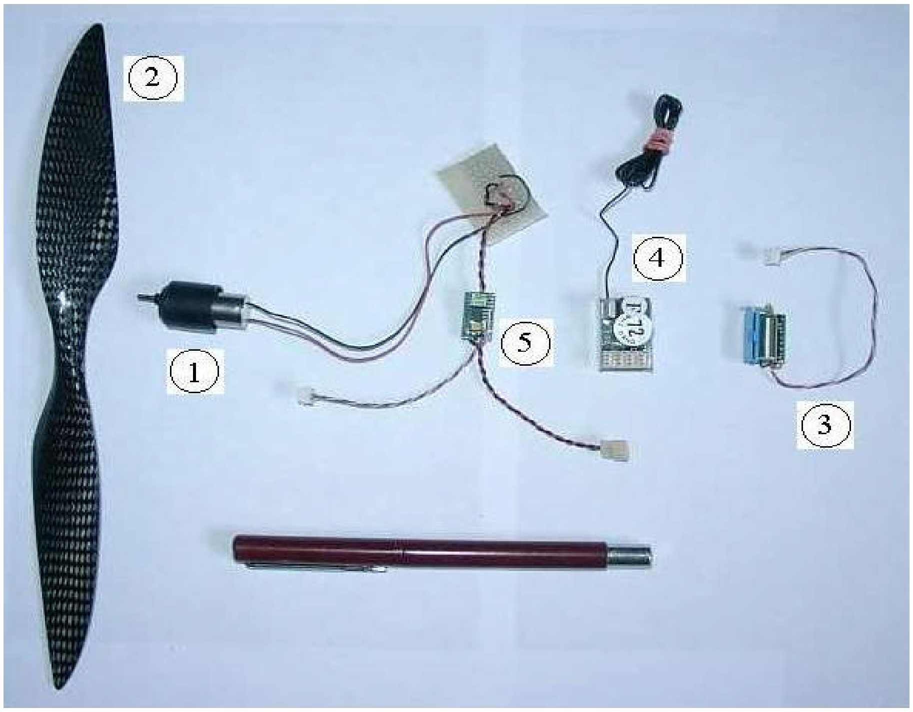


Figure A1: Photograph Showing the Size of Hardware Components

COMPONENT SPECIFICATIONS

This appendix contains the data and description on the individual components as available from the manufacturer.

Propulsion unit:

Motor: DC 5-2.4

Gear ratio: 11.8:1

Weight: 14.2 gm

Diameter of output drive shaft: 2 mm

Gear efficiency: 93%

Rated voltage (V)	5
No load current (mA)	35
No load speed (1/min)	21000
Stall current (mA)	2000
Max. output power (W)	2.42
Max. Efficiency (%)	75.3
Diameter/ length (mm)	12 x 17.9

Table A2: Motor Specifications

Voltage (V)	Current (A)	Power (W)	Thrust oz (g)	RPM
4	0.29	1.2	0.71 (20)	1665
5	0.40	2.0	1.1 (30)	1990
6	0.55	3.3	1.5 (43)	2340
7	0.70	4.9	1.9 (55)	2650
8	0.85	6.8	2.4 (69)	2900
9	1.01	9.1	2.9 (82)	3150
10	1.17	11.7	3.4 (95)	3380

Table A3: Power and Thrust Characteristics of the Propulsion Unit.

Propeller:

Manufacturer: WES technik

D=25, H=12

Weight: 3.4 gm

Carbon fiber

Servo: (2 required)
Manufacturer: WES technik
Model: LS-3.0
Max deflection: 14 mm
Time to full deflection: 0.15sec
Max output force: 200gm
Operating voltage: 3-5 V
Load current <100 mA

Receiver:
Supplier: WES technik
Model: R4P JST
Frequency: 72 MHz, 4 Channel
Weight: 4.3 gm
Power requirement: 0.25 W

Speed controller:
Supplier: WES technik
Model: JMP-HF9

Technology:	RISC Micro controller
Functions	Auto-Calibration, Safe-Start
Operating Voltage:	6 to 10.8 V (13V max)
Clock-Frequency:	133 kHz
Response time	0.2 sec
Rpm reduction:	at < 4.8V
Current (A): Cont./ Max.	1.5/ 3
Dimensions (mm):	4 x 8 x 17
Mass (without wires):	0.9 g
Mass (with wires):	1.9 g

Table A4: Specifications of the Motor Controller

Voltage Regulator:
Model: LM 2940CT
Dropout Voltage: 0.5 V
Maximum Current: 1 A
Operating Current: 30 mA