NASA-CR-197179

NASw-4435

Utah State University

MAE 595
Space Systems design

IN-18-CR
26153
44 P

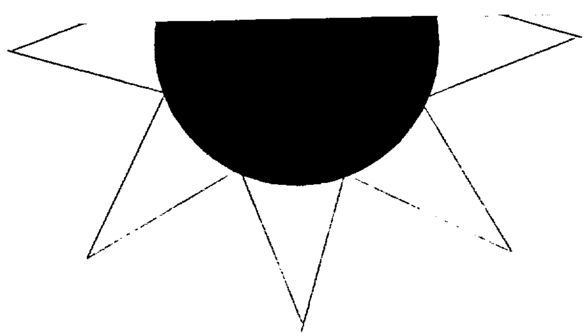
NASA-USRA
Advanced Design Program

(NASA-CR-197179) SolSTUS: SOLAR SOURCE THERMAL UPPER STAGE (Utah State Univ.) 44 p

N95-13581

Unclas

G3/18 0026153



SolSTUS
Solar Source Thermal Upper Stage

A last stage LEO to GEO or interplanetary orbit booster using a solar thermal propulsion system.

Fall 1994

Reference mission and design considerations.

Contents

0.0 Abstract:

This paper was written by members of the Utah State University (USU) Space Systems Design class, fall quarter 1993. The class is funded by NASA and administered by the University Space Research Association (USRA). The focus of the class is to give students some experience in design of space systems and as a source of original ideas for NASA.

This paper is a summary of the work done by members of the Space Systems Design class during the opening phase of the course. The class was divided into groups to work on different areas of the Solar Thermal Rocket (STR) booster in order to produce a design reference mission that would identify the key design issues. The design reference mission focused upon a small satellite mission to Mars.

There are several critical components in a Solar Thermal Rocket. STR's produce a very low thrust, but have a high specific impulse, meaning that they take longer to reach the desired orbit, but use a lot less fuel in doing it. The complexity of the rocket is discussed in this paper. Some of the more critical design problems discussed are:

1. The structural and optical complexity of collecting and focusing sunlight onto a specific point,

2. Long term storage of fuel (liquid hydrogen),

3. Attitude control while thrusting in an elliptical orbit and orienting the mirrors to collect sunlight,

4. Power and communications for the rocket and it's internal systems.

The design reference mission discussed here is a very general mission to Mars. A first order trajectory design has been done and a possible basic science payload for Mars has been suggested.

This paper summarizes the design reference mission (DRM) formulated by the USU students during fall quarter and identifies major design challenges that will confront the design team during the next two quarters here at USU.

1.0 Reference mission

1.1 Systems group

1.1.1 General description

A solar thermal rocket (STR) has several advantages over standard types of rockets, but also several disadvantages. To provide thrust an STR must collect energy from the sun and focus it to one point (see figure 1.1.1). This focused energy then heats some type of propellent. When the propellent is heated it expands providing thrust. To focus the sunlight a large concave mirror or lens must be used. An STR is very efficient, but has very low thrust. An STR could never even begin to provide the thrust necessary for an earth launch and must therefore begin it's operation in space. This paper examines a Solar Thermal Rocket carried into a low earth orbit (LEO) of about 400 to 500 km. The STR could be carried into LEO by either the space shuttle, or some other less expensive small rocket like a Pegasus or Taurus launch vehicle (volume and mass permitting). From LEO the STR would begin operation and move a payload either out to a geosynchronous orbit (GEO) or into an interplanetary orbit.

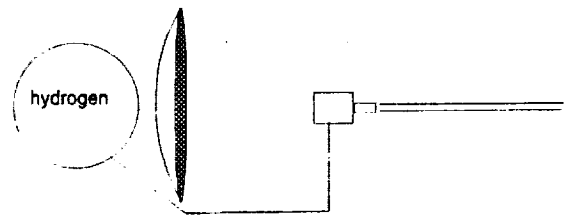


figure 1.1.1 Example of mirror collection system.

1.1.2 Mission

One mission that an STR could be used for is a scientific mission to Mars. The design reference mission that we have worked on fits into today's emphasis on "smaller, cheaper, faster". The mission we envision is a small mission of under 300 to 400 kg. We picture the payload being launched into low earth orbit along with a final stage solar thermal rocket. The STR would be used to raise the orbit and launch the Mars mission into a heliocentric Homann transfer orbit to Mars.

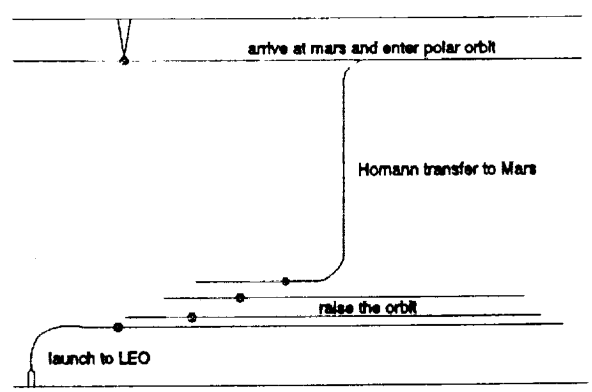


figure 1.1.2 Design reference mission scenario

Once the mission reaches Mars the orbiter would have to be slowed and placed into a polar orbit. The polar orbit allows for complete coverage of the Martian surface and allows the polar caps to be studied. The studies at Mars would include some of the studies that were to have been performed by the Mars observer, but not all. Studies of the Martian surface composition and geography, internal thermal activity, and atmospheric composition and structure would be conducted.

1.1.3 Payload

The actual instruments to be used in this reference mission were taken directly from the Mars observer. Three of the smaller instruments were selected, hopefully to be capable of giving the most data for the least amount of mass. The instruments selected are given below [1]:

1.) Medium and low resolution camera: This instrument would not be as fine as the Mars observer because of mass requirements for very high resolution cameras. It would give a good picture of the Martian geography and a general understanding of the surface structure and geologic activity along with better maps of the Martian surface.

mass: 10-15 kg power: 5-8 W data rate: 700-10000 bps

2.) Thermal Emissions Spectrometer: A thermal emissions spectrometer measures the heat radiation from the surface. With this data and a knowledge of the properties of materials, scientists should be able to determining surface and atmospheric composition to a reasonable degree of accuracy.

mass: 13 kg power: 14.3 W average 23 W peak
data rate: 700-4000 bps

3.) Radio science experiment: The radio science experiment uses the already existing communications antenna with the addition of a small ultrastable oscillator. While passing behind the Martian atmosphere the satellite would send a signal to earth which, when interpreted should give a good structural model of the Martian atmosphere and exact positioning data of the satellite and Mars.

mass: 1.6 kg (addition to existing communications system)
power: 2.8 W (additional)

These are only possible instruments for a Mars mission payload. We are engineers, not scientists, and our understanding of what is needed may not be the best. The total mass of 300 to 400 kg is also an estimate of what a payload may require. Communications and power requirements are two more of the driving factors in the size and mass of the mission, these are discussed later in the report.

1.2 Design Reference Mission trajectory

1.2.1 Orientation of Initial Orbit

The solar thermal rocket (STR) is initially assumed to be in a counterclockwise Low Earth Orbit (LEO) at 300 km. This orbit is obtained by launch from a Pegasus (or comparable) launch vehicle.

1.2.2 Orientation of Final Orbit

The final orbit before a Homann transfer to Mars is a High Elliptic Earth Orbit (HEEO). At the perigee of this final orbit, a delta V is given to escape Earth and give the craft the necessary excess velocity to reach Mars.

1.2.3 DRM Trajectory

The characteristics of the solar thermal rocket itself limit the possible techniques to move from LEO to a highly elliptical Earth orbit. It can be shown that an impulsive thrust at perigee is more efficient than a spiral orbit.[1] Because the STR is not a truly impulsive thruster, a technique to approximate this must be used. Therefore, the STR thrusts tangentially to the orbit across an arc centered at the perigee to give an effect comparable to an impulsive thrust. The orbit is gradually changed from the initial circular orbit to an elliptical orbit of high eccentricity.

When the portion of the orbit in which the STR does not thrust becomes large enough to inhibit storage of the propellant boil off, an impulsive thrust at perigee by a chemical thruster will move the STR to its final orbit. The STR is separated from the payload and retrieved for further use prior to this maneuver.

In order to give a final delta V at perigee and have the excess velocity in the direction of the Earth's velocity, the direction of travel must be reversed. A minimum delta V to reverse the orbit can be given at apogee. The payload will then be given the final delta V at the perigee to achieve transfer velocity.

An approximate trajectory was simulated by solving the equations using a Runge-Kutta fourth order solution. The apogee before jettison of the STR was on the order of 200 Earth radii and the eccentricity approached one. With the thrust of the STR alone, this maneuver occurs in a matter of months.[2]

To maximize efficiency of the escape trajectory, the orbit was oriented such that a final burnout at perigee would align the momentum of the Earth and the momentum of the space craft in the direction of the desired Hohmann transfer. The orientation of this orbit is such that the major axis is at an angle of about 151° counterclockwise from the velocity vector of the Earth at the time of burnout. see figure 1.2.1

1.2.4 Other Considerations

Assuming that the desired HEEO can be attained in about one month by thrusting through an arc of 120°, the Earth will rotate about 1/12 of a revolution around the sun. During this time the orbit of the STR remains fixed with respect to the sun. This will cause the direction of the Earth's velocity vector to rotate about 30° before escape velocity is achieved. Therefore, to align the perigee correctly, the orientation of the major axis must be increased from 151° to 181°.

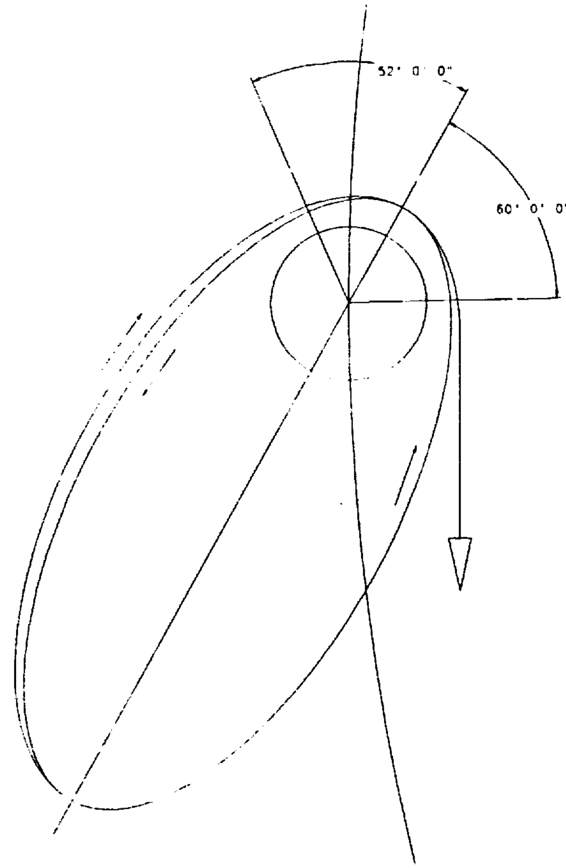


figure 1.2.1 final orbit

1.2.5 Relevant Data

The nature of the solar thermal rocket requires that it be in direct sunlight to thrust. Measuring from the velocity vector of the Earth, the angle in which the rocket is able to thrust was found to be -17° to 197° for the LEO. In relation to the STR, plenty of sunlight is available on one side of the craft. However, only 16° (of the 60° desired) is available on opposite side. As the Earth moves about the sun and as the orbit becomes more eccentric, the angle of available sunlight will increase from 16° to 52°.

For a Mars reference mission, a velocity of 10.9 km/sec is required to escape the Earth's gravitational influence. An additional V_∞ of 2.98 km/sec is required for a Hohmann transfer to Mars. Thus, the total velocity required at the final burnout will be 13.9 km/sec. Perigee thrusts using the STR will increase the velocity to about 10.4 km/sec when the STR is separated from the craft. The remaining delta V will be provided by chemical thrusters.

1.2.6 Tradeoffs

1.2.6a Propellant Boil Off

When designing the final trajectory, the ability of the STR to store propellant boil off is an important consideration. Any orbit which leads to boil off of more propellant than can be stored or used results in the loss of propellant. This type of orbit must be avoided in order to use the fuel efficiently.

1.2.6b Structure

The orientation of the orbit with respect to the sun significantly affects the required direction of thrust and thus the relative positions of the thruster and mirror.

1.2.6c Steering

The steering system must be capable of keeping the relative positions of the mirror and spacecraft such that the mirror is always facing the sun. Thus, the steering system must actively control this orientation whenever the STR is thrusting. An increase in the necessary arc angle of thrust may be limited by control system capabilities.

1.2.7 Alternative Trajectories

1.2.7a Inclination Change

Several methods were considered to efficiently achieve transfer velocity. The orientation of the ellipse could be changed to a more inclined orbit allowing more time in direct sunlight. This maneuver, however, required a large delta V and was inefficient.

1.2.7b Major Axis Orientation

The major axis could be oriented to allow an arc of thrust greater than 180°. This provides a more efficient use of the STR. However, the mirrors must be moved across the path of the thrust nozzle which increases the structural complexity. This also requires that the final delta V be given before perigee which reduces the efficiency.

1.2.8 Summary

The Design Reference Mission as outlined provides a possible trajectory for a mission to test the solar thermal rocket. It is apparent that several alternative trajectories still exist and further iterations and analyses may provide a more suitable mission.

2.0 Design issues

2.1 Solar Thermal Rocket

2.1.1 Introduction

Several designs of solar thermal rockets have been proposed but the optimum design is still a matter of opinion. In an optimum design, two major components to be considered are the rocket motor and the mirrors or lenses used to concentrate solar radiation.

2.1.2 Rocket Motors

Proposed designs for solar thermal rockets fall into variations of two design principles. Because hydrogen(the most efficient propellant) is transparent to solar radiation, the hydrogen must either be seeded with a solar absorbent or absorb heat from a medium which has been heated by the solar radiation.

Proposed rocket motor designs include:

1. Solar Radiation Absorber[4]
   The propellant is seeded with an alkali vapor which absorbs solar radiation heating both the alkali vapor(potassium) and the hydrogen.

2. Solar Thermal Propulsion Engine[5]
   Solar heat is absorbed in "an ogive collection cavity" with a heat exchange medium between its inner and outer walls through which the propellant passes and is heated.

3. Refractory Hollow Unit[6]
   A series of "coiled refractory hollow metal tubes" form a collection cavity. Propellant passes through the tubes and is heated.

4. Absorption Rocket[7]
   Propellant passes through a series of "circumferentially spaced injection ports" and "stacked radiation absorber material" to be heated.

Two advantages to designs which seed the propellant (case 1) include higher receiver efficiency than a black body absorber and higher operating temperatures. Disadvantages may include: reflection from chamber window, more complicated design requirements, and opaqueness of the window to certain frequencies.

Advantages to designs passing propellant through an absorbing medium (cases 2,3,4) can include: simple design requirements, no reflective window, and trapping of all incident solar radiation. Disadvantages may include: development of special materials capable of withstanding high temperatures and hydrogen embrittlement, limitations on operating temperatures, and reradiation at the front of the rocket motor.

2.1.3 Mirrors and Lenses

Focusing devices available for use in solar thermal propulsion systems can be divided into five categories. Table I list these categories and the corresponding specific mass(mass per m^2 of reflective surface).

Table I

Reflector type	Specific Mass(kg/m^2)
Adaptive	20-100
Rigidized	1-2
Inflatable	0.02-1.0
Deployable	0.875
Fresnel Lenses	1.21

The adaptive reflector consists of many small reflectors mounted on a truss structure and is generally rejected because of mass constraints. Rigidized reflectors are inflated. However, the structure then becomes rigid, and the propellant can be removed. Inflatable optics are also inflated, and the shape is maintained by the pressure of the inflatant. Inflatable mirrors are impractical for small structures because punctures caused by the space environment require the use of large structures that can be maintained at low pressures in order to minimize leakage in inflatable structures.

Deployable mirrors are perhaps the best option because they are light and efficient. A 16 m^2 has been developed by Hercules [3]. Two of Hercules mirrors are capable of delivering the power required by the solar rocket motors. An evaluation of focusing and tracking requirements show that a good mirror design would have a 3m focal length and a maximum tracking error of +/-.31°.

Fresnel lenses are an attractive option because they can be placed between the sun and the rocket motor and structural blockage can be cut to a minimum. The biggest obstacle facing the use of Fresnel lenses in space in the degradation of the lens material due to collisions with atomic hydrogen. This may be overcome if the casting of "sol-gel" Fresnel lenses being developed by McDonnel Douglas pass space environment tests [8]. McDonnel Douglas has developed a solar dome lens capable of delivering the power required. It has an 7.2m focal length and a tracking error of 0.25° gives an efficiency of 87%. An effective area is 34.3m^2 delivers 38.2 kW to the rocket motor.

2.1.4 Conclusions

Deciding on the best rocket motor and mirrors to use for an optimum solar thermal propulsion unit is clouded by the claims of designers. Each designer claims his design works best while pointing out the disadvantages of other designs. In reality his design may have some of the same disadvantages. Because working prototypes have not been tested to our knowledge, an extensive analysis of the rocket motor designs must be carried out in order to determine the overall efficiency of each design. The two major design considerations to be examined in determining the overall efficiency are:

1. The efficiency of the rocket motor.
   What percentage of the solar power delivered to the motor is converted into thrust? And what specific impulse can the rocket motor maintain?

2. The efficiency of the mirrors.
   What is the specific mass of the mirrors?, What is the reflective efficiency of the mirrors?, And what is the focal length and focusing requirements of the mirrors?

At the present time the Hercules solar propulsion unit appears to be the most favorable. The mirrors for the unit have been developed and can delivered the power needed for the rocket motors. A trade off between the use of a parabolic which focuses the light directly in front of it and a section of a parabola which focuses the light to the side of the mirror must be determined.

Hercules claims that 95% of the solar radiation that enters the rocket motor will be converted to kinetic energy. This number is well above any other reported efficiencies and must be examined to determine its accuracy. Specifically the efficiency of a black body is below 80% at the rocket motors operating temperature, and the Hercules' rocket motor design appears to incorporate a black body absorption chamber.

If the Hercules' claims hold, the Hercules' rocket motors and mirrors (3m focal length, placed directly in front of the rocket motors) will form a desirable solar thermal propulsion unit.

2.2 Steering

2.2.1 Overview

The Steering System of the SolSTUS will provide the following services:

- stabilization of craft
- orientation of mirrors to the solar flux (.3 Deg Accuracy)
- orientation of thrust vector to orbital trajectory
- orientation of communication antenna (.5 Deg Accuracy)

The following problem arises from this setup:

- As the mirrors and the thruster align themselves with their respective orientations, angular momentum must be conserved. This will be referred to as the problem of conservation of angular momentum.

2.2.2 Problem of conservation of angular momentum

The conservation of angular momentum dictates that a system of interlocking rotating platforms will conserve angular momentum. That is, if one section is rotated, then the other section will compensate by rotating in the opposite direction.

The SolSTUS will consist of two platforms: the mirror section and the thruster section. While the main thruster must align itself with the orbital trajectory, the mirrors of SolSTUS must be constantly aligned with the solar flux. Owing to the conservation of angular momentum in space, maintaining the orientation of one platform while the other turns is difficult.

Assuming the two platforms are joined by a near frictionless joint, the following solution is proposed. A three axis hydrozine thruster system will be used to orientate the main thruster. Because the joint is only near frictionless, some momentum will be transferred to the mirrors. A momentum wheel, that is attached to the mirror structure, will be used to absorb or release any momentum needed in order to maintain the mirrors orientated to the solar flux. With this system, the two platforms will be able to maintain an orientation in space.

2.2.3 Stabilization of Craft

After SolSTUS is delivered into orbit, sensors will run an algorithm to stabilize the craft. This will be initiated by the ground station.

2.2.4 Hardware of SolSTUS

The sensors of SolSTUS consists of the following:

- Star Sensor
- Horizon sensor
- Sun Sensor
- Rate and/or position sensors

The actuators of SolSTUS consists of the following:

- Hydrozine thrusters.
- Momentum wheel
- Tapping the hydrogen fuel

2.2.5 Tapping of hydrogen fuel

There exist the possibility of designing a system to tap the hydrogen fuel used for the main thruster and use it for the attitude control thrusters. This is facilitated by the fact that every so often hydrogen will need to be vented in order to prevent rupturing of the containment vessel. Harnessing this power into attitude control could be beneficial. This should be explored as more research is done on the use of heated hydrogen as a fuel.

2.2.6 Mass Configuration of Craft

Under the suggested solution to the problem of conservation of angular momentum, a near frictionless joint links the two platforms together. It is essential that the center of mass of each platform share the same location, but be independent of each other.

This presents a design problem. The mirrors must be the focal length away from the main thruster. In order to keep the center of mass of the mirror platform at the near frictionless joint, counterbalancing the mirrors in the opposite direction is necessary. Perhaps that counter balance could consist of fluid tanks or scientific payload.

Also, note that as the pressure of the hydrogen gas changes, the position of the storage tank and the pneumatics becomes critical, possibly changing the center of mass of SolSTUS. The change of the center of mass of the system will induce an internal moment, leading to destabilization of the craft. Final design of SolSTUS must consider this quandary.

2.3 Fuel form and properties

2.3.1 Hydrogen states

There are four different forms of hydrogen which were looked at for use with this mission: liquid (LH_2), solid (SH_2), slush (SLH_2), and supercritical (SCH_2). The areas used to decide which form to use were duration of mission (length of time that there would be a need for hydrogen), technical risk and complexity. LH_2 provides a relatively long duration of mission, while still being relatively easy to work with and inexpensive compared to the other forms. The other three forms of hydrogen all have a shorter mission duration than LH_2 and are increasingly more difficult to use and have a greater risk associated with their use. For this mission it was decided to use LH_2 as the fuel.

For an initial the hydrogen tank pressure was taken as standard atmospheric. The boiling point of LH_2 at atmospheric pressure is 20.28 Kelvin. The calculation used for this mission will be made using LH_2 at 20.0 Kelvin. The density at 20.0 Kelvin is $\rho = 71.098\ kg/m^3$.

2.3.2 Tank Design

Two different tank designs were looked at for storing the hydrogen: storing all the hydrogen in a single tank, and storing the hydrogen in two separate identical tanks. The fuel required to escape the earths sphere of influence is ≈ 45 kg. This equals .6329 m^3. A 5% factor will be added to the volume to allow room for the gaseous hydrogen. This makes the volume = .6645 m^3.

The single tank used has dimensions of d=.9 m, l=1.045 m, and a thickness of t=.003 m. If made out of aluminum, this tank would weigh 35.34 kg. The tank would be supported by eight support struts, four on each end.

The double tank used has dimensions of d=.65 m, l=1 m, and t=.003 m. If made out of aluminum, these two tanks would weigh 45.33 kg. Each tank would be supported by six support struts, three on each end.

One of the major considerations in the choice of the tank design is the center of mass of the satellite. Since the fuel is the only dynamic mass on the satellite, it must be placed as close to the center of mass as possible. With one tank this would mean placing it along the center axis. As fuel is depleted, this could change the center of mass in two different directions. With two tanks, as long as both tanks are at equal distances from the center axis, the tanks would only have to be aligned with one axis. With two tanks it would be necessary to drain both tanks equally. This would require more hardware than for one tank.

The tanks will be insulated using double aluminized Mylar (DAM)/single tissue glass multi-layer insulation (MLI). The MLI has a density of $\rho_{MLI} = 51.5\ kg/m^3$ and a thermal conductivity of k=.16 mW/mK. 2 cm of MLI was used in the heat loss calculations.

2.3.3 Thermodynamic Properties

The amount of LH_2 which turns to gaseous hydrogen is a function of the temperature difference between the hydrogen and the surroundings, and the thickness and thermal conductivity of the MLI.
For a 2 cm thick layer of MLI, the heat loss due to the temperature difference is
q = 1.1475 W. This gives a mass boil off of .2225 kg/day/tank.

The heat loss due to the support struts is a function of the area, length, and thermal conductivity of the material. For support struts which are .01 x .01 x .5 m, the heat loss is q = .696 W/support. This gives a mass boiloff of .1349 kg/support/day. For the single tank this would correspond to 1.0792 kg/day. For the double tank this would correspond to 1.6188 kg/day

2.3.4 Fuel Extraction

During the time in which hydrogen will be used as a fuel, there will need to be two different ways of extracting the hydrogen from the tanks. At the beginning of the mission, when the length of time between thrusts is relatively small, not enough hydrogen will be boiling off to supply the thrusters with gaseous hydrogen. In contrast, near the end of the thrusting, the length of time between thrusts will get quite long. During this time to much hydrogen will be boiling off and must either be stored or vented off.

If the pressure inside the tanks is greater than the surrounding pressure, this will force the gaseous hydrogen out of the tanks and into the fuel lines. The tanks will be designed to hold a greater pressure than just over the surrounding pressure, to allow for storing more of the gaseous hydrogen.

When there is not enough gaseous hydrogen to supply the thrusters, the LH_2 will be extracted using a Liquid Acquisition Device (LAD). This device will remain in contact with the LH_2 and will have a pump to deliver the fuel to the thrusters. As the LH_2 gets closer to the thrusters, it will change to gaseous hydrogen.

3.0 Support systems

3.1 Structure

3.1.1 Structural Requirements

The success of this mission depends on the pointing accuracy of the mirrors. This accuracy is required to develop the heat to expand the hydrogen and provide the desired thrust. The pointing accuracy of the mirrors to the solar flux will need to be in the range of 0.3°. This range is a combination of the losses due to vehicle orientation, an attitude control problem, and due to the flexure of the mirror support. The thrusters must be aligned with the mirrors to even better accuracy which is solely a structures problem. The objective of structural design is to support the mirrors and thrusters rigidly enough to provide this pointing accuracy. The necessary stiffnesses depend upon the final vehicle configuration.

The design mission requires that SolSTUS fit in the smallest launch vehicle possible in order to be economic. A Pegasus or Titan launch vehicle is preferable. Thus many subsystems will need to be deployable. The mirrors are the largest subsystem which must be deployed. Hercules Corp. has a deployable mirror design which could satisfy our application. The parabolic communications antenna and the support booms for the mirrors and thrusters must also be deployable. The basic trade-off for a deployable system is that the stiffer the system, the harder it is to deploy.

Possible concepts for the support booms include systems based upon material deformation and hinged materials. True inflatable structures are light and easily packaged but are prone to leak due to punctures if inflated to a high enough pressure to give the necessary stiffness. Inflatable rigidized by foam have not yet been developed with enough dimensional stability and uniformity of properties. (Research in inflatable, rigidized structures is presently being conducted by USU and Thiokol Corporation under a NASA In-Step contract.) Furlable systems have been developed which unwind like a carpenter tape. They require heavy mechanisms and have not been developed for structural support. However, they could be used as the actuator for a stiffer telescoping strut. A more complex mechanical system involving hinged trusses would be the stiffest alternative. The actual system design is to be determined but could be as simple as pivoting an arm into place or as complex as unfolding the truss system and moving it into place with a system to lock the joints from further movement. Deployable systems such as these are topics of current research. The system used on SolSTUS will probably require some research and modeling.

3.1.2 Structural Components

One major concern for development of the spacecraft is maintaining the center of mass at the point through which the thrust vector passes. This must be accomplished even as the fuel level decreases. Two possibilities have been considered. First, two tanks balanced across the center of mass could be used provided that they empty at the same rate. Under the scenario of using the hydrogen gas bleed off to feed the thrusters rather than vent to space, both tanks would have to heat uniformly. Second, one tank could be used if the center of mass of the entire system passes through its own center of mass. This way the center of mass will remain the same as the fuel decreases and expands to fill the tank.

The hydrogen fuel tank is addressed as a major concern because it will require a large portion of the volume and mass of the spacecraft. Due to the requirement that the center of mass be maintained, the other subsystems will most likely be built around the fuel tank.

Other design factors which affect the spacecraft structure include maintaining the thrusters of the absorption chambers at the center of mass and positioning these at the correct distance and angle to the mirrors. This requires maintaining the pointing accuracy between these two subsystems as previously indicated. The mirrors will need to be attached to the spacecraft and also to the absorption chambers, without blocking large amounts of incoming solar flux. The possibility of using a swivel joint to separate the mirrors and thrusters from the craft to allow for independent rotation has also been considered. The three systems just mentioned; fuel tanks, thrusters, and mirrors, constitute the foundation of the spacecraft. From there the other subsystems are included to aid in accomplishing SolSTUS's mission. Placement of solar arrays or batteries must be considered in order for the craft to obtain the required power. Communications requires that an antenna or phased array be placed such that the craft can communicate with ground control. The payload must be placed for the most effective method of separation and transfer to Mars to be accomplished. Also Hydrazine thrusters, if used for attitude control, must be oriented so that they can most efficiently control the mirrors and thrusters or payload, whatever the case may be. As each subsystem is located its mass must be taken into consideration so that the center of mass for the entire system can be maintained running through the fuel tank and the thrusters.

Another concern in our design is that of obtaining thermal isolation of the hot and cold systems. The heat from the absorption chambers must not cause all of the hydrogen to boil off before it can be used for thrusting. Therefore, the materials and structures used to support the thrusters and fuel tank must be taken into consideration. The most desirable design will include a material with a low thermal conductivity and high yield strength and a structure with a minimal cross sectional area with preferably longer sections.

3.1.3 Payload Deployment

SolSTUS is actually two rocket systems. Mission planning requires separation of the STR used for initial velocity gain and the system headed to Mars at the final orbit's apogee. At this point the payload system will be given a reverse velocity. The system's placement should be at the rear of the spacecraft. It must be supported before release and then released by some mechanism such as a spring. A chemical thruster must be located on this rocket give the necessary reverse velocity and final perigee thrust to the payload system.

3.2 Power

3.2.1 Requirements

The basic specifications and requirements for SolSTUS's electrical power supply follow. First, we assume that the spacecraft will require 50 Watts continuous power, both in Low Earth Orbit (LEO) and in Mars orbit. Since 28 Volts is the aerospace standard for small power supplies, i.e. power supplies which generate less than 1 kiloWatt, we also assume that SolSTUS's power supply voltage specification will be 28 Volts. We also assume a solar cell duty cycle of 2/3 (16 hours of sunlight, then 8 hours darkness). Finally, we assume that the LEO part of the mission will last less than 2 months, and that the Mars part of the mission will last less than 3 years. This is important to consider because solar winds damage solar cell arrays. Over long periods of time, the cumulative damage will cause a noticeable decrease in solar cell array power output. Fortunately, we can ignore this because where solar wind-induced cell degradation is concerned, 3 years is a relatively short period of time; damage to the array will be insignificant.

3.2.2 Topology

The power subsystem topologies to choose between are Direct Energy Transfer (DET) and Peak Power Transfer (PPT). PPT was chosen because PPT os more efficient than DET when eclipse is significant fraction of orbit, as in the case of a LEO [9].

- Direct Energy Transfer (DET)
  - Solar array, batteries, electronics, and loads on common bus.
  - Battery provides power directly to bus during eclipse and whenever power demand exceeds solar array capability.
  - Solar array provides power directly to bus for loads and battery charging during sunlit portion of orbit.
  - Power is shunted whenever solar array output exceeds power demand.

- Peak Power Tracking (PPT)
  - Solar array regulated at different voltage than battery load bus.
  - Battery provides power to load bus during eclipse and whenever power demand exceeds solar array capability.
  - Solar array is operated at maximum power output at sunrise when the cold array has additional power for the loads and recharging batteries.
  - Solar array is operated at less than maximum power when its capability exceeds the power demand.

3.2.3 Solar Cells

There are two basic solar cell technologies currently available which are suitable for use in spacecraft. One is Gallium Arsenide technology (GaAs), and the other is Silicon technology (Si). The major advantage of GaAs cells is that they are more efficient; given the same amount of sunlight, a GaAs cell will generate more power than an Si cell. GaAs cells are about 18.5% efficient, while a typical Si cell is about 14% efficient. GaAs cells also tend to be more heat-resistant than Si cells. Solar cells become less efficient the hotter they get, but the rate of power decrease with temperature increase for GaAs cells is about half that for Si cells.

Some characteristics of Si and GaAs cells are similar. GaAs cells and Si cells are virtually identical in mass per unit area. For example, the mass of a 2 square meter array of GaAs cells is virtually identical to the mass of a 2 square meter array of Si cells. (Roughly 1 kilogram per square meter). We estimate that cell support structure will add an additional 0.5 kilograms per square meter, for both technologies. The packing factors for both cell types are similar. The packing factor is simply the ratio of the actual area of solar cells to the total area of the array. It is less than 1, because there must be some space between individual cells to allow for electrical interconnects and other mounting-related concerns. A typical packing factor is 85%.

Si and GaAs radiation-degradation characteristics differ somewhat. Given a low level of radiation, GaAs cells degrade more slowly than Si cells. At high radiation levels the reverse is true; GaAs cells degrade more rapidly than Si cells. Since SolSTUS's solar cell arrays will not be exposed to radiation for very long, degradation will be insignificant. We will therefore probably not consider radiation-degradation characteristics when we decide which solar cell technology to use.

Si solar cells are cheaper than GaAs cells, on a cost per unit area basis. The bottom line however is that because GaAs cells are more efficient, given a specific power generation requirement a GaAs solar cell array will be smaller and lighter than an Si solar cell array. This offsets cell costs in two ways; first, even though Si cells are cheaper per unit area, the total necessary area of a GaAs cell array is less than the necessary area of an Si cell array. Second, a GaAs cell array will be less expensive to put into LEO because it is lighter. There may be a third consideration as well; if payload volume of the launch vehicle is close to that of SolSTUS's total packaged volume, then using a small GaAs cell array instead of a larger Si cell array could allow us to use a small, inexpensive launch vehicle instead of a larger, more expensive one.

3.2.4 Batteries

The two most feasible technology choices are Nickel Cadmium (NiCd) and dependent pressure vessel (DPV) Nickel Hydrogen (NiH_2) batteries.

Design considerations for battery packs:

Depth of discharge is how deeply a battery can be discharged without causing permanent damage to the cell. A Nickel Hydrogen battery has a greater depth of discharge than does a Nickel Cadmium battery.

Energy density is the amount of power a cell will sustain over a given length of time. The Nickel Hydrogen battery has an energy density of seventy watt hours per kilogram; twice that of the Nickel Cadmium battery.

Cycle life is the number of times a battery can be fully charged and then discharged. The Nickel Hydrogen battery has a life of 10,000 cycles; ten times that of the Nickel Cadmium battery.

Batteries must be able to withstand stresses associated with space. Nickel Cadmium batteries have been used extensively in space applications in the past. Nickel Hydrogen batteries are space worthy but because they are a new technology, actual mission performance data are not yet available.

3.2.5 Summary

Because an array of GaAs cells will be smaller and lighter than an array of Si cells, we suggest using GaAs cells in the solar arrays. Given 50 Watts continuous power requirement, a 100 Watt solar array will provide energy necessary to satisfy spacecraft power requirements and recharge the batteries, with a safety margin. (A 100 Watt solar cell array allows a 1/2 duty cycle, 12 hours of sunlight for every 12 hours of darkness, whereas we expect 16 hours sunlight for every 8 hours darkness. This is our safety margin).

For LEO (1 Au), a 100 Watt GaAs solar array will measure 0.7 square meters in area, and will mass 1.2 kilograms, including support structure. For Mars orbit (1.5 Au), a 100 Watt GaAs solar array will measure 1.6 square meters, and will mass 2.4 kilograms.

NiH_2 batteries have twice the energy density of NiCd batteries and hence half the weight. NiH_2 batteries have a much longer cycle life than NiCd batteries, and also tolerate a deeper discharge. For these reasons we feel that Nickel Hydrogen batteries are a good choice for a mission of this size.

Appendix A contains data on the Power subsystem in the following areas:

- Topology block diagram.
- Solar cell parametric curves.
- Solar cell characteristics.
- Battery parametric curves.
- Battery characteristics.

3.3 Communications Subsystem

3.3.1 Introduction

The communications strategy is composed of two separate divisions: The earth orbit ferry and the Mars science payload communication schemes.

While the ferry decreases the geocentric orbit energy to near zero, the only spacecraft data returned to earth applies to the attitude and control of the spacecraft; no science data is returned while in earth orbit. Also, many commands will need to be sent to the ferry portion of the spacecraft while it makes its perigee thrusts. After payload release to orbit mars (PROM), the option remains to bring the ferry back to LEO. This option will require a separate communication system for the payload and ferry.

The Mars orbit insertion will require large amounts of Tracking, Telemetry, and Command (TT&C) between the spacecraft and mission control. The subsequent science mission will also require high levels of telemetry from the spacecraft to mission control, but not visa-versa.

This profile clearly divides the communications architecture into two complete and separate systems. The two communication schemes require independent communication subsystems.

3.3.2 Spacecraft Antenna Options

There are a variety of antenna types to consider in this mission. Primarily, the most recommended antennas are the parabolic dish, phased array, microstrip, and omnidirectional. Parabolic dish antennas possess the best gain of these types and omnidirectional the worst.

High gain is very important to TT&C having low power over great distances, and communication would best be served by a parabolic dish antenna or a phased array antenna. For command, where high data rates are not required, lower gain is acceptable. A grouping of microstrip or omnidirectional antennas would provide sufficient communication capabilities for earth orbit, and if placed properly, will not require mechanical pointing. Omnidirectional antennas, which have the worst gain of all antenna types are capable of servicing the mission and provide the benefit of receiving commands regardless of attitude. Omnidirectional antennas provide an excellent redundancy for unforseen contingencies.

<u>Parabolic Antenna</u>. The parabolic antenna is the most commonly used in deep space communication systems. The main reason is its high gain, which is a measure parabolic reflectors focusing power into a narrow beam that can be pointed in one direction. This effect is extremely desirable when transmitting long distances through space.

Considering gain first, for a parabolic, reflector antenna the gain is expressed as [12][13]:

$$G=\frac{4\cdot A\cdot\eta\cdot\pi}{w^2}$$

where A is the cross-sectional area in meter-squared, ⁿ is the antenna efficiency, (typically 0.55), and w is the wavelength of the signal. This gain however, is usually expressed in decibels. The equation therefore becomes:

$$G=20.4+20\log_{10}f+20\log_{10}D+10\log$$

Nomenclature

M	Margin for Error
E_b	Energy per bit
N_o	Noise
P	Transmitter Power
G_t	Gain of transmitter
G_r	Gain of receiver
L_l	Loss over line
L_s	Loss over space
L_a	Loss in atmosphere
k	Boltzman's Constant
T_s	Temperature of System
R	Data Rate
c	Speed of Light in Vacuum
s	Distance
f	Carrier Frequency
D	Diameter of Antenna
n	Antenna Efficiency

The constant 20.4 dB/m^2GHz^2 is equal to $20\log_{10}(\pi/c)$ where c is the speed of light. Using this formula, an efficiency of 0.55, a diameter of 1 meter, and a X-band frequency of 12 GHz, the gain of the parabolic antenna is 39.4 Db. The gain will vary with corresponding changes in antenna size and transmitting frequencies.

Another variable considered in antenna choice is its beam width. A useful definition of beam width is the angle of the signal when the power has dropped to one-half its peak value or its 3-Db drop. This angle is expressed as:

$$\theta=\frac{21}{f\cdot D}$$

where f is in Ghz and D is diameter of the parabolic antenna in meters. Using the same antenna parameters listed above the 3-dB drop beam width angle is 1.75 degrees. This angle also varies with changes in frequency and size.

Phased Array Antenna. Phased array antennas have electrical beam steering, a property of the multiple-plate design. The phased array antenna considered is steerable to 45 degrees from center. It is approximately 41x41x8 cm and may weigh as much as 4.5 kg for each one (it will take about 5 antennas for full coverage). This phased array antenna has a gain of 16 dB. [11]

Microstrip Array. Because of the lower requirements for gain and the relatively short distance of earth orbit, a microstrip array antenna is sufficient for TT&C. This antenna design is small, lightweight, relatively inexpensive, and proven in satellite applications. [10][16][17]

The antenna uses a constrained series space feed to distribute the RF signal from the input port to array elements. This provides highly accurate and efficient power distribution with low side lobes. A total of four arrays, each measuring less than ten inches square and less than an inch thick, are mounted flush on orthogonal surfaces of the satellite. This arrangement allows the satellite to transmit or receive from any attitude.

The primary liability of the microstrip design is durability. However, material selection minimizes this liability, giving the microstrip comparable reliability to other space antenna designs. The substrate normally chosen for this design is known as RT/Duroid 5800; it has a resistivity of 2.33 and withstands the wide temperature variations typical of the space environment. This substrate has a thickness of 0.028 inches and a conductor thickness of 0.0014 inches. Gallium arsenide chips have also been considered for this purpose, but they have a much lower efficiency and cause significant thermal problems [16].

There are two main classes of microstrip antennas: Hybrid and monolithic. Hybrid microstrip antennas are multilayered and have greater weight and volume than the monolithic class, by a factor of ten or more. The monolithic antennas are ten to twenty times the price of the hybrid antenna ($2,000 to $10,000 vs. $200 to $500), but are less prone to mechanical failure and require a simpler feed network than the hybrid antenna. With the hybrid antenna, power output of several hundred watts at L band, 50 to 100 watts at S band and 10 to 20 watts at C band could be expected. Above C band and X band, monolithic amplifiers with outputs of 3 to 5 watts per chip is a practical choice. On the other hand, if monolithic devices are used, then power levels of 1 to 5 watts are reasonable from L to X band. Transmitter/receiver (T/R) gains of 20 to 30 dB are common in T/R modules attached to the monolithic antennas to compensate for this low power [17].

It is recommended the monolithic phased array antenna be used, because of the great advantages in weight, size, and reliability. The lower power of the monolithic antenna can be compensated with T/R modules, and the available frequency range for the antennas are comparable. The only advantage of the hybrid antenna is original cost [14].

<u>Omnidirectional Antenna</u>. An antenna of this type is very simple. It is simply a rigid wire extended from the vehicle which broadcasts and receives in a full hemisphere (giving it a gain of 1). To prevent satellite blockage, multiple antennas are placed about the spacecraft to give it full coverage at any attitude. An omnidirectional antenna system is frequently used as a redundancy in the communications subsystem.

3.3.3 Deep Space Network

The NASA Deep Space Network (DSN) is capable of providing communications services for both schema of the SolSTUS mission. The DSN, maintained by Jet Propulsion Laboratory in Pasadena, California has complexes located in three locations throughout the world. They are: Goldstone, California; Madrid, Spain; and Canberra, Australia. These location are approximately 120° apart to provide full 24-hour coverage. [18]

The DSN communicates across two bands, the S-band (2.0 to 2.3 GHz) and the X-band (7.1 to 8.5 GHz). Typically long range missions utilizing a parabolic dish antenna use the X-band. Although the X-band is subject to greater space loss and atmospheric loss, the advantage lies in the increased gain of a parabolic dish antenna. For communications not using the parabolic dish (or any other antenna not benefitted by the higher frequency), S band is more advantageous because of its favorable atmospheric penetration.

The DSN has two antenna sizes at each of the three sites. These parabolic dish antennas have diameters of 34 meters and 70 meters. For missions like SolSTUS, the 34 meter antennas are used. The 70 meter antennas are generally used only for unique events and not for continuous data collection; time on these antennas is very limited. The 70 meter antenna can provide additional gain for payload separation, Mars orbit insertion, and any contingencies which may arise.

These antennas are not highly efficient (η=0.45) but are liquid cooled and reduce the system temperature significantly (T_s= 26K against star background) [18]. Using the following gain equation for parabolic dish antennas:

$$G=20.4+20\log_{10}f+20\log_{10}D+10\log_{10}\eta$$

The 34 meter antenna receiving telemetry in the S-band (2.0 GHz) has a gain of 54 while the 70 meter antenna has a gain of 60. In the X-band (8.0 GHz), the 34 meter antenna has a gain of 66, and the 70 meter antenna has a gain of 72.

3.3.4 Computer System, Transmitter, and Receiver

The computer system for the ferry has minimal requirements. As the spacecraft only needs to receive commands relating to thrusts and interpret data for attitude control, the computation and memory requirements are not as stringent as those of a scientific spacecraft.

A computer capable of supporting this leg of the mission is the AlTech (OSC) 950 computer with 4 million instructions per second (MIPS) and 500k bytes memory. It has a mass of 4 kg and power requirement of 8 watts. Another candidate is the Honeywell

DSBC with 1.2 MIPS and 256k bytes memory. This smaller computer has a mass of 1.2 kg and a power consumption of 4 watts. [15]

For the Mars payload of the mission, one possible alternative for continuous telemetry (which is not very feasible unlike the ferry spacecraft) is store-and-forward. To reduce complexity of the spacecraft, the parabolic antenna could not be steerable with respect to the spacecraft; instead, the spacecraft could point the antenna by attitude adjustments to the whole spacecraft. This, of course, will place greater requirements on the attitude control system. In this condition, or in the position where the antenna is behind Mars, the science data will have to be stored in computer memory until the antenna can be pointed toward earth.

Initial research identifies the Small Explorer Data System (SEDS), developed for the Small Explorer (SmEx) Division at Goddard Space Flight Center. This computer system uses the INTEL 80386/80387/82380 chipset for space applications with the fiber optic MIL-STD 1773 data bus architecture. This system provides computation at 1.2 MIPS with 7 megabytes random access memory (RAM) as well as telemetry, tracking, and communications (TT&C) standard interfaces. This computer consumes 10.5 watts average power and weighs approximately 7.5 kg. SAMPEX successfully used SEDS without any message loss. [19]

3.3.5 Payload and Data Rates

As stated in the payload section, the communications system must be able to process data ranging from 688 bps to 34,252 bps (bits per second). Using the data store and forward system described above, data will be stored when the spacecraft is unable to transmit. Once transmission line to earth is available, data will be transmitted at rates as low as 6 Kbps when Mars and Earth are at greatest distances or much faster rates when the Earth is closer to Mars.

The communication system will also be designed to receive control instructions from earth. Because these control signals do not contain enormous amounts of information, the receiving data rate should be approximately 64 bps. This estimate may vary if more data is to be received.

3.3.6 Power Requirements

The power required for the transmission and reception of data is summarized with the following equation [15]:

$$P=-\frac{E_b}{N_o}-M+G_t+G_r+L_l+L_s+L_a-10\log_{10}k-10\log_{10}T_s-10\log_{10}R$$

Where the space loss is calculated with the following:
The following are examples of power calculation for the SolSTUS mission.

$$L_s = 20\log_{10}c - 20\log_{10}4\pi - 20\log_{10}s - 20\log_{10}f$$

Ferry example. First consideration is the ferry segment of the spacecraft. For this segment, the example will use the S-band frequency (2 GHz) with an omnidirectional antenna. For the ferry segment of the SolSTUS mission, using the equation for space loss, we determine L_s=-158 dB for LEO ($s=1\times10^6$ m) and L_s=-208 dB for the apogee of the final orbit ($s=287\times10^6$ m).

To calculate power needed to transmit a command to the ferry, the following parameters will be used: E_b/N_o=10.3 dB, this signal-to-noise ratio assumes differential phase shift keying (DPSK); M=5 dB, which includes 3 dB pointing error (half of the half-power beam width) and 2 dB for filtering and timing errors; G_t=54 dB, assuming 34m DSN antenna on S-band; G_r=1 dB, assuming an omnidirectional antenna; L_l=-1 dB, which is a standard approximation; L_s=-208, as above; L_a=-4.5 dB, another standard approximation for clear weather (-0.5) plus a -4 dB margin for rainfall or cloudy weather in S-band; $k=1.38\times10^{-23}$ J/K, the Boltzman constant; T_s=1295 K, the high temperature of the omnidirectional antenna; R=2000 bps, an acceptable data rate for TT&C. These variables, using the link equation gives a final power requirement of 9.33 watts for a near-worst-case scenario.

Payload example. Power requirements become much greater for the Mars payload, and it becomes advantageous to use the X-band (8 GHz). The space loss varies greatly as the distance from Earth and Mars varies (s=55800016700 m to 399575990900 m). Using the space loss equation, the loss ranges from -265 dB to -283 dB.

For the Mars orbit telemetry, the following parameters will be used: E_b/N_o=4.4 dB, this signal-to-noise ratio assumes binary phase shift keying (BPSK) and quadriphased phase shift keying (QPSK) modulation with rate-1/2 Viterbi decoding; M=5 dB, which includes pointing error and filter/timing errors; G_t=36 dB, a 1 meter diameter antenna with 0.6 efficiency; G_r=66 dB, assuming the 34 meter DSN antenna on X-band; L_l=-1 dB, which is a standard approximation; L_s=-283, as above; L_a=-6.5 dB, another standard approximation for clear weather (-0.5) plus a -6 dB margin for rainfall or cloudy weather in the X-band; $k=1.38\times10^{-23}$ J/K, the Boltzman constant; T_s=26 K, the temperature against the moon (a good worst case); R=6000 bps, a less-than-desirable data rate for telemetry to Earth. These variables, using the link equation gives a final power requirement of 21.23 watts.

3.3.7 Recommendations

The SolSTUS mission clearly requires two separate communication architectures because of differing requirements for the separate ferry and payload. Additionally, to recover the ferry to LEO, it needs to maintain a communications system after payload separation. It is recommended that, with respect to the communications system, the ferry and payload be treated as separate spacecraft.

Because of the relative close proximity of the ferry to earth, it may be possible to meet all communication requirements without the use of the NASA DSN. Using a smaller ground station is plausible and may significantly reduce the cost of the mission.

It is not likely the demands of the payload, once it reaches Mars orbit insertion, can be met by a system with capabilities less than the DSN. Certainly significant data transfer can occur with the 34 meter antennas, and circumstances requiring a temporary increase in data communications may be met with succinct use of the 70 meter DSN.

1.1 References

1. A. L. Albee and F. D. Palluconi, The Mars Observer, quarterly news letter Winter 1991, JPL publication

1.2 References

2. Zuppero, A., Private communication, November 17, 1993.

3. Zuppero, A., et al., Small Satellite Missions to Deep Space Enabled by Solar Thermal Rockets, Idaho National Engineering Laboratory.

2.1 references

4. Beall, M. W., et al, Solar Thermal Propelled Spacecraft Mars Cargo Missions, Case for Mars IV, Boulder, CO, June 1990.

5. Pande, John B., Solar Thermal Propulsion Engine, U.S. Patent# 5,138,832, Hercules, Aug 1992.

6. Vrolyk et al, U.S. Patent# 4,815,443.

7. Shoji, James M., Solar Thermal Propulsion Unit, U.S. Patent# 4,781,018, Rockwell International Corp., Nov. 1988.

8 O'Neal, M. J., Conceptual Design Study of a 5 Kilowatt Solar Dynamic Braxton Power System Using a Dome Fresnel Lens Solar Concentrator, NASA, Springfield, VA, 1989

3.2 References

9. Day, John H., Small Explorer Power Subsystems Overview, AIAA Small Satellite Subsystems Symposium, NASA/Goddard Space Flight Center, June 1993

3.3 References

10. Baker, Steven Max. A Microstrip Phased Array Antenna For Satellite Application. Logan, Utah: Utah State University Masters Thesis, 1989.

11. Bate, Roger R., Donald D. Mueller, and Jerry E. White. Fundamentals Of Astrodynamics. New York, New York: Dover Publications, Inc. 1971.

12. Copernicus - Lunar Surface Mapper. Logan, Utah: Utah State University, Advanced Design Program report 1992.

13. Damon, Thomas D. Introduction to Space The Science of Spaceflight. Malabar, Florida: Orbit Book Company, 1989.

14. Fortescue, Peter and John Stark, eds. Spacecraft Systems Engineering. Chichester, England: John Wiley and Sons, 1991.

15. Gordon, Gary D. and Walter L. Morgan. Principles Of Communications Satellites. New York, New York: John Wiley & Sons, Inc. 1993.

16. Griffin, Michael D. and James R. French. Space Vehicle Design. Washington, D.C.: American Institute of Aeronautics and Astronautics, Inc., 1991.

17. Larson, Wiley J. and James R. Wertz, eds. Space Mission Analysis and Design. Dordrecht, The Netherlands: Kluwer Academic Publishers, 1991.

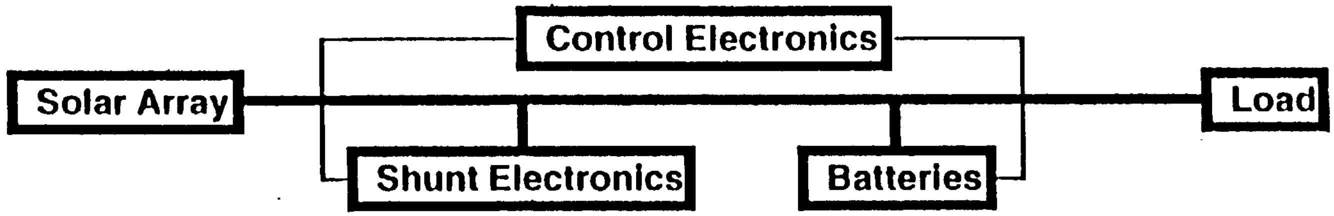
18. Mailloux, Robert J. Antenna Array Architecture, Proceedings Of The IEEE, vol. 80, No.1, pp 163 - 172, January 1992.

19. Tang, Raymond and Richard W. Burns. Array Technology, Proceedings Of The IEEE, vol. 80, No.1, pp 173+, January 1992.

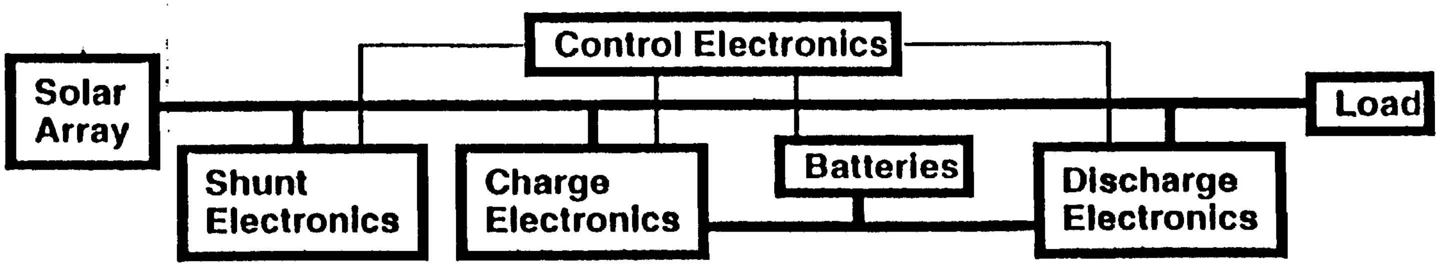
20. Yuen, Joseph H., ed. Deep Space Telecommunications Systems Engineering. Jet Propulsion Laboratory: California Institute of Technology, 1983.

Appendix A

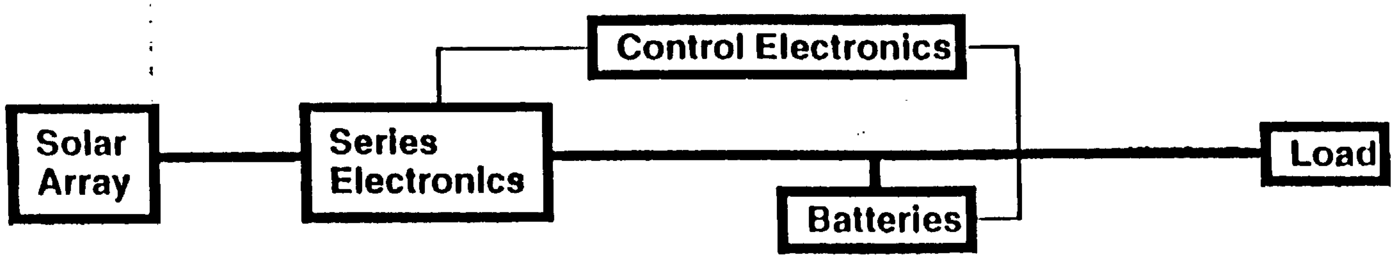
POWER SUBSYSTEM TOPOLOGY DIAGRAMS



DIRECT ENERGY TRANSFER UNREGULATED BUS

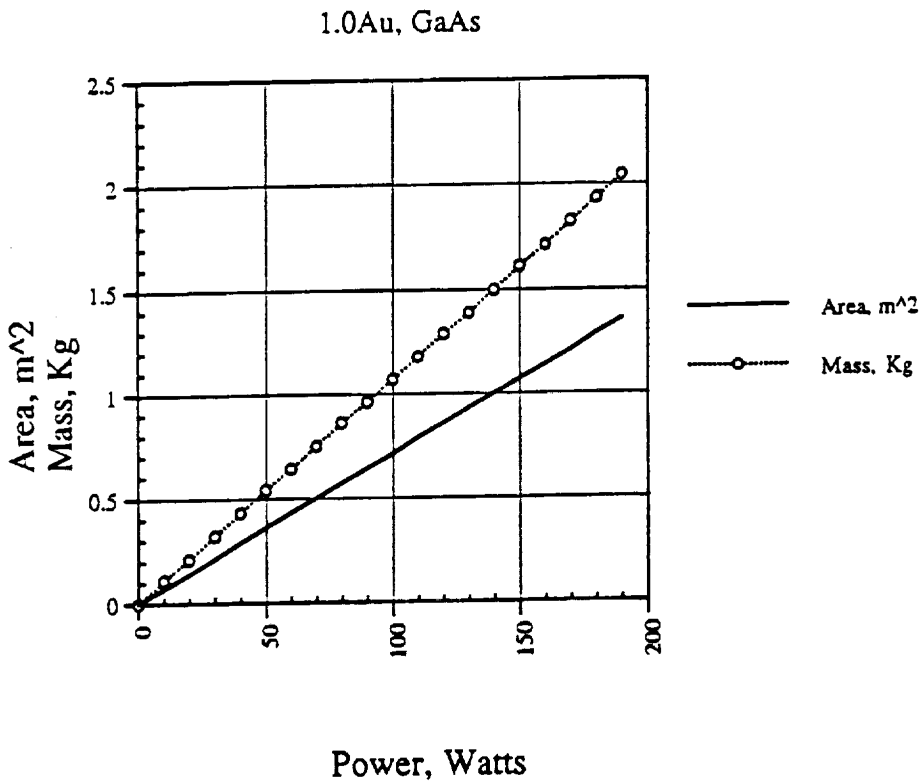


DIRECT ENERGY TRANSFER REGULATED BUS

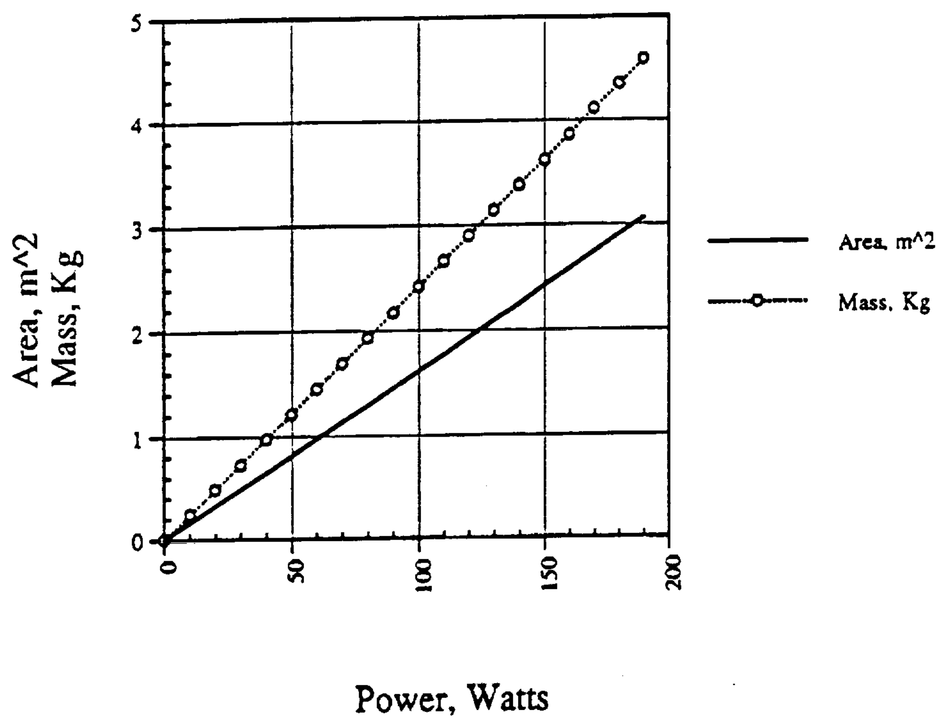


PEAK POWER TRACKER UNREGULATED BUS

Energy in W/(m^2) @1 AU:	1350								
Distance from Sun, AU:	1.5								
Total Power Necessary, W:	100								
Max. Angle of Incidence, deg:	30								
Name	**Parameters**					**Results (totals)**			
	Eff'cy (28°C)	Degr'dn fctr	Pack. factor	Size, cm^2	g/(m^2)	$/(m^2)	Area, m^2	Mass, Kg	#/cells	$, Dollars
GaAs ON	0.185	0.75	0.86	16	1021.5		1.61	1.65	867	0
Si, BSR	0.127	0.78	0.86	64	1031.2		2.26	2.33	304	0
(10 Ohm-cm)										
Si, BSR	0.136	0.73	0.86	64	1031.2		2.25	2.32	303	0
(2 Ohm-cm)										
Si, BSF/R	0.148	0.68	0.86	64	1031.2		2.22	2.29	299	0
(10 Ohm-cm)										

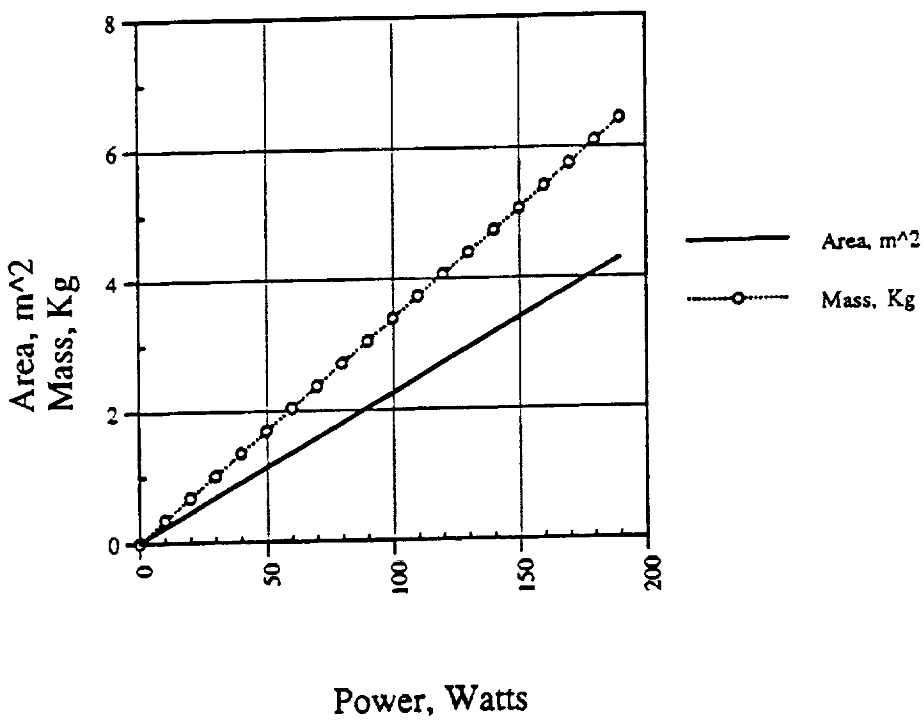

1.0Au, GaAs
Area, m^2
Mass, Kg
0
0.5
1
1.5
2
2.5
0
50
100
150
200
Power, Watts
Area, m^2
Mass, Kg

1.5Au, GaAs



Area, m^2
Mass, Kg
0
1
2
3
4
5
0
50
100
150
200
Area, m^2
Mass, Kg
Power, Watts

1.5Au, Si (BSR,10ohm)



WHAT'S AVAILABLE IN SOLAR CELLS

CELL TYPE	SILICON			ADVANCED SILICON	GaAs ON		
	2ohm cm BSR	10ohm cm BSR	10ohm cm BSF/R	10ohm cm BSF/R	GaAs	Ge	Ge Cascade
Watts/Sq Ft 28°C AMO BOL	14.7W	13.7W	16.2W	18.9W	19.8W	19.8W	
Efficiency (AMO) BOL	13.4%	12.5%	14.8%	17.0%	18.5%	18.5%	
Max Cell Size Available (cm)	8 x 8	8 x 8	8 x 8	8 x 8	4 x 4	6 x 6	2 x 4
Nominal Voltage (VOC)	0.60	0.55	0.61	0.65	1.0	1.0	2.0
Thickness Nominal (mils)	8.0	8.0	8.0	8.0	12	8.0	8.0
Thickness As Low As (mils)	2.5	2.5	2.5	8.0	8.0	3.5	4.0
Quantity Delivered (2 x 4 cm EQUIVALENT)	>2M	>2M	>1M	--	>200K	>200K	--
ASEC production History	30 yrs	30 yrs	15 yrs	New	8 yrs	4 yrs	New
Availability	Now	Now	Now	1-92	Now	Now	1-94

Note: 100 2 x 4cm cells per square foot
Coverglass used in calculating is 6mil CMX with AR coating
Cell thickness 0.008"

08/91

BSR - Back Surface Reflector, provides high reflectance for unabsorbed sunlight, thereby decreasing orbital cell temperature.

BSF/R - A Back Surface Field is included to increase cell output. The BSF formation still retains full BSR advantages.

For information, please contact:
J. Capitano, Director of Marketing [818] 968-6581
D. Kozak, Marketing Manager [818] 968-6581

FAX: [818] 336-8694

APPLIED SOLAR ENERGY CORPORATION
P.O. Box 1212, City of Industry, California 91748

JOE4.CDR

Comparison of Silicon versus GaAs/Ge One Square Foot Solar Panel Power

CELL TYPE	BOL, AMO, 0°C Power/Sq Ft (W)	EOL, 1 MeV ELECTRON AMO, 0°C Power/Sq Ft 3×10^{14} e/cm² (W)	EOL, 1 MeV ELECTRON AMO, 0°C Power/Sq Ft 1×10^{15} e/cm² (W)	BOL, AMO, 28°C Power/Sq Ft (W)	EOL, 1 MeV ELECTRON AMO, 28°C Power/Sq Ft 3×10^{14} e/cm² (W)	EOL, 1 MeV ELECTRON AMO, 28°C Power/Sq Ft 1×10^{15} e/cm² (W)	BOL, AMO, 60°C Power/Sq Ft (W)	EOL, 1 MeV ELECTRON AMO, 60°C Power/Sq Ft 3×10^{14} e/cm² (W)	EOL, 1 MeV ELECTRON AMO, 60°C Power/Sq Ft 1×10^{15} e/cm² (W)
Silicon 2 Ohm-cm, BSR	16.6	13.8	12.1	14.7	12.3	10.8	12.6	10.6	9.2
Silicon 10 Ohm-cm, BSR	15.4	13.6	12.1	13.7	12.1	10.8	11.8	10.4	9.1
Silicon 10 Ohm-cm, BSF/R	18.3	14.0	12.3	16.2	12.5	11.0	13.9	10.7	9.3
GaAs/Ge	21.0	18.5	16.0	19.8	17.4	14.8	18.4	16.1	13.6

Note: 100 2 x 4cm cells per square foot
Coverglass used in calculations is 6mil CMX with AR coating
Cell thickness 0.008"

BSR - Back Surface Reflector, provides high reflectance for unabsorbed sunlight, thereby decreasing orbital cell temperature.

BSF/R - A Back Surface Field is included to increase cell output. The BSF formation still retains full BSR advantages.

08/91

For information, please contact:
J. Capitano, Director of Marketing [818] 968-6581
D. Kozak, Marketing Manager [818] 968-6581

FAX: [818] 336-8694

APPLIED SOLAR ENERGY CORPORATION
P.O. Box 1212, City of Industry, California 91748

J0IXCDA

Gallium Arsenide on Gallium Arsenide Solar Cells

CELL SPECIFICATIONS

Test Conditions: AMO, 135.3 mW/cm², 28°C

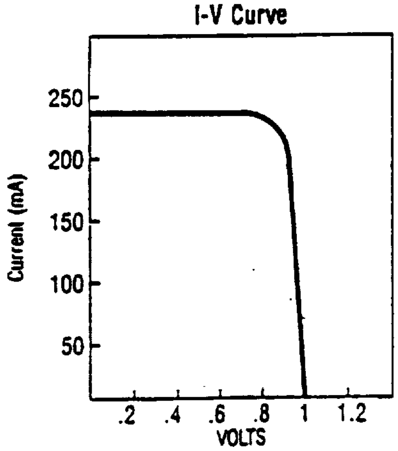
Parameter	Symbol	Unit	22 GaAs-300	24 GaAs-300	24 GaAs-200	44 GaAs-200
Cell Size	—	cm	2 x 2	2 x 4	2 x 4	4 x 4
Cell Thickness	—	μm	300	300	200	200
Efficiency	η	%	18.5	18.5	18.5	18.0
Open Circuit Voltage	V_{oc}	V	1.000	1.000	1.000	1.000
Short Circuit Current	I_{sc}	mA	121	242	242	484
Optimum Load Voltage	V_L	V	0.870	0.870	0.870	0.870
Load Current	I_L	mA	115	230	230	450
Maximum Power	Pmax	mW	110	200	200	391
Weight	—	gm	0.670	1.343	0.950	2.686
Solar Absorptance**	α_s	—	0.830	0.830	0.830	0.830

**350 nm filter

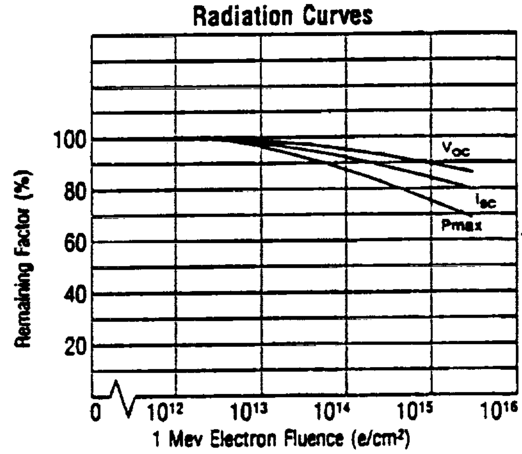
CELL CHARACTERISTICS

Test Cell: 2 x 4 cm² x 200 μm

I-V Curve



Radiation Curves



Temperature Coefficients (28°C to 60°C)

Parameter	%/°C
Voc	-0.194
Isc	+0.056
V_L	-0.232
Pmax	-0.223

Applied Solar Energy Corporation
15251 East Don Julian Road ▪ City of Industry, California 91749
Phone (818) 968-6581 ▪ FAX (818) 336-8694

Silicon Back Surface Reflector (BSR) Solar Cells

CELL SPECIFICATIONS

Test Conditions: AMO, 135.3 mW/cm², 28°C

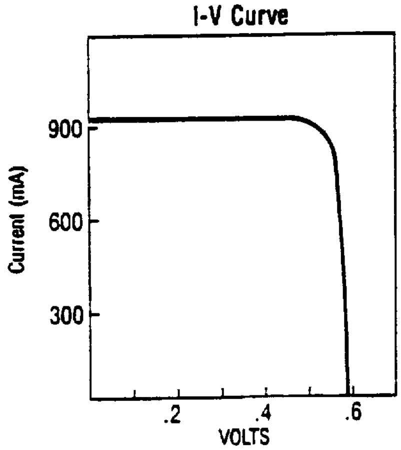
Parameter	Symbol	Unit	24 BSR2-200	24 BSR10-200	46 BSR2-200	46 BSR10-200
Cell Size	—	cm	2 x 4	2 x 4	4 x 6	4 x 6
Cell Thickness	—	µm	200	200	200	200
Resistivity	—	Ω - cm	2	10	2	10
Efficiency	η	%	13.4	12.5	13.4	12.5
Open Circuit Voltage	Voc	V	0.590	0.540	0.590	0.540
Short Circuit Current	Isc	mA	308	312	924	936
Optimum Load Voltage	V_L	V	0.50	0.455	0.50	0.455
Load Current	I_L	mA	290	297	870	890
Maximum Power	Pmax	mW	145	135	435	406
Weight	—	gm	0.416	0.416	1.248	1.248
Solar Absorptance**	α_s	—	0.68	0.68	0.68	0.68

**350 nm filter

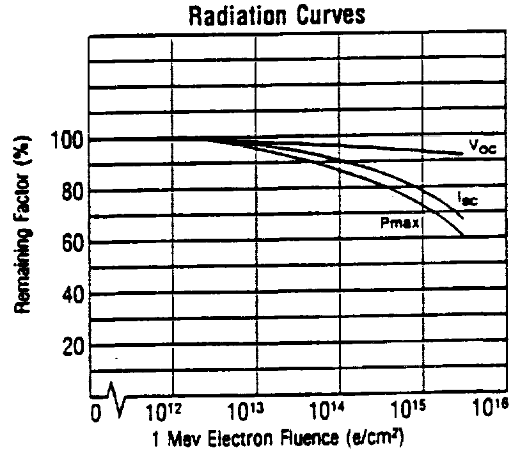
CELL CHARACTERISTICS

Test Cell: 4 x 6 cm x 200 µm 2 Ω - cm BSR

I-V Curve



Radiation Curves



Temperature Coefficients (28°C to 60°C)

[illegible]	[illegible]
Voc	-0.370
Isc	0
V_L	-0.443
Pmax	-0.445

Applied Solar Energy Corporation
15251 East Don Julian Road • City of Industry, California 91749
Phone (818) 968-6581 • FAX (818) 336-8694

NICKEL CADMIUM VS. NICKEL HYDROGEN BATTERIES

Assumptions:

The sun will be available for 12 hour (50% of the time) stretches.
Average power need in LEO is 50 Watts Continuous.

Scientific payload power requirements:

peak: 39.5 Watts
ave: 24.8 Watts

Battery mass while in LEO:

NiCads
50_W*12_hours = 600_Wh   600_Wh/35_Wh/kg = 17.14_kg
NiH_2
50_W*12_hours = 600_Wh   600_Wh/55_Wh/kg = 10.90_kg

Battery mass of scientific payload(using peak requirements):

NiCads
39.5_W*12_hours = 474_Wh   474_Wh/35_Wh/kg = 13.54_kg
NiH_2
39.5_W*12_hours = 474_Wh   474_Wh/55_Wh/kg = 8.61_kg

Total battery mass:

NiCads
17.14_kg + 13.54_kg = 30.68_kg
NiH_2
10.90_kg + 8.61_kg = 19.51_kg

$\Delta mass = m_{NICd} - m_{NIH2} = 11.17\_kg$

NICKEL CADMIUM BATTERIES:

Working Voltage: 1.25_V DC

Energy Density: 35_Wh/kg

Watt Hour (Wh): Practical unit of electrical energy equal to a power of one watt absorbed continuously for one hour.

Reason for choosing NiCads:

1. *High power capability.*
2. *Long cycle life (over 1000 cycles on deep discharges; many more on partial discharges.)*
3. *Can be recharged rapidly.*
4. *Good low-temperature performance.*
5. *Sealed cells.*

Assuming that the solar cells will be in the dark 50% of the time (12 hours) and that our power need is 50 watts continuous the weight of the battery array can be determined as follows:

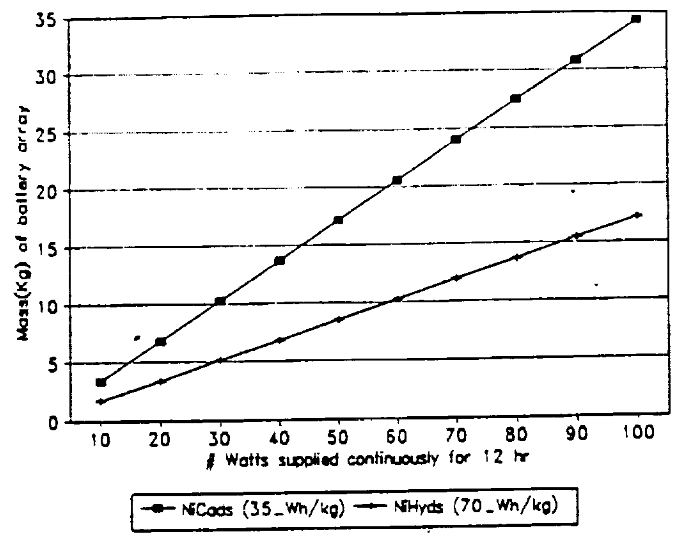
*50_Watts*12_hours = 600_Wh*

600_Wh\(35_Wh\kg) = 17.14_kg

To use the Air force 28_V standard:

*23_batteries*1.25_V/battery = 28.75_V (Series Config)*

			Theoretical battery		Practical battery		
						Capacity	
Battery system	Anode	Cathode	Voltage, V	Capacity,* Wh/kg	Typical working voltage, V	Wh/kg	Wh/L
Secondary batteries							
Lead-acid	Pb	PbO_2	2.1	120	2.0	35	80
Edison	Fe	Ni oxide	1.4	224	1.2	30	60
Nickel-cadmium	Cd	Ni oxide	1.35	181	1.2	35	80
Silver-zinc	Zn	AgO	1.85	283	1.5	90	180
Nickel-zinc	Zn	Ni oxide	1.73	215	1.6	60	120
Nickel-hydrogen	H_2	Ni oxide	1.5	289	1.2	55	60
Silver-cadmium	Cd	AgO	1.4	227	1.1	60	120
Zinc-chlorine	Zn	Cl_2	2.12	394	1.9	100	130§
High temperature	Li(Al)	FeS	1.33	345	1.2	60	100§
High temperature	Na	S	2.1	377	1.7	100	150§



Mass(Kg) of battery array
35
30
25
20
15
10
5
0
10 20 30 40 50 60 70 80 90 100
Watts supplied continuously for 12 hr
NiCads (35_Wh/kg)
NiHyds (70_Wh/kg)