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Conceptual design of a solar power beaming space system

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ABSTRACT

CONCEPTUAL DESIGN OF A SOLAR POWER BEAMING SPACE SYSTEM

by Tuyet N. Le

The concept of Space-Based Solar Power (SBSP) is a global solution for the world energy crisis. SBSP has been discussed for decades; however, there still has not been a single watt transmitted down from orbit. A conceptual SBSP demonstration design has been developed for a system that will beam 300W of power to the Earth's surface. This demonstration is estimated to be at 25% efficiency due to atmospheric losses and laser conversion losses. A 2200W laser is a modular subsystem of the100 kg payload flight demonstration. All of the technologies needed for this demonstration already exist. The demonstration includes the following modular subsystems: the laser system, the acquisition, tracking, and pointing system, the safety and control system, and the ground segment/receiver system. The ISS demonstration is estimated to cost approximately 12 million dollars. Tradeoff design studies and systems engineering evaluations were completed in order to demonstrate the feasibility of this system. An Excel database was developed to help calculate some basic dynamics, creating an SBSP preliminary systems design tool for the demonstration.

DEDICATION

This is dedicated to my wonderful and brave parents Chau Nguyen and Dan Le.

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Chapter 1: Introduction

Every day the world population increases in number and puts a greater strain on the Earth's finite supply of resources. As fossil fuels are depleted by today's demanding economies and industries, the need for alternative sources of energy increases exponentially. For example, according to the India Planning Commission, India must generate 700,000 additional megawatts of power to keep pace with its frantically growing economy and population (Farrar, 2008). Many villages exist with limited power or no power at all. In order to keep pace with population expansion, India must develop new sources of energy to provide power to these villages and bring them in line with the more developed regions of the country. One solution to this looming energy crisis is to look to the stars. Solar power is one source of clean, virtually unlimited energy. An ideal solution would be to develop a method to harvest this cheap solar energy twenty four hours a day. One such solution is the concept of Space-Based Solar Power (SBSP). SBSP requires the assembly of an expansive network of solar panels in geosynchronous orbit about the Earth. Placed in a high orbit where solar energy is intense, these solar cells would gather the sun's energy almost twenty four hours a day and 365 days a year. Once collected by the solar panels, this endless supply of energy could be beamed down to ground stations all over the world, including rural, undeveloped areas in third world countries.

The advantages of Space Based Solar Power are many. This method of harvesting clean, limitless energy reduces the need for the destruction of the environment for the purpose of meeting increasing energy demands. The need for development of polluting coal power plants and drilling for oil would be greatly reduced or eliminated. An SBSP network would allow the world to detach itself from the dependence on a finite supply of fossil fuels. The reduction of competition for limited resources would reduce tension between world powers and relieve worries over energy shortages. SBSP would allow for global expansion and development without inciting fears over an energy supply that cannot keep up with increasing demand. A future powered by the sun would allow economies and innovation to thrive around the globe. Small villages in third world countries such as India would be transformed into thriving communities with higher living standards and significant contributions to the global economy. The United States, Russia, China, Japan, Canada, and the members of the European Union, are all intrigued by the idea of SBSP for domestic and commercial purposes. The early pioneers of SBSP technology will be able to assert themselves as global energy leaders for decades to come.

In 2007 the National Security Space Office (NSSO) produced an Space-Based Solar Power study stating that the United States government should allow for and facilitate the development of an SBSP project to meet current environmental and energy challenges in order to create an energy source that is renewable and environmentally friendly. The NSSO called for a letter of support for SBSP to be sent from Congress to the Department of Defense, planning for an SBSP demonstration by 2013. Responsive to that request, this thesis provides a conceptual design for demonstration of a Space-Based Solar Power Beaming System and its financial feasibility. Chapter 2 offers basic background information on SBSP to enable better understanding of the conceptual design of a solar power system. Chapter 3 takes a systems engineering approach to analyzing all the sub-systems of SBSP. Chapter 4 reviews the tradeoff studies and presents the preliminary SBSP demonstration. Chapter 5 contains the systems database and preliminary design calculations. Chapter 6 concludes the study and suggests future work.

Chapter 2: Background

The sun radiates billions of terra-watts of energy to all corners of the solar system. Only a small portion of this energy strikes the earth's surface due to a relatively small surface area, night and day cycles, the atmosphere and seasonal trends. Space-based solar power (SBSP) involves the concept of placing solar cell collectors in a chain in geosynchronous earth orbit. Energy is captured using photovoltaic or solar cells and is beamed down to earth. The beamed down energy is captured via ground solar cells and converted into electricity for the grid. The placement of the SBSP platform will require numerous delivery and deployment flights to orbit resulting in the need to drive down the cost of launch services.

This thesis proposes a SBSP demonstration to beam down power from orbit. The following are methodology and approaches necessary to illustrate the achievability of beaming down 300W of power from the International Space Station to a ground station on earth. Included in this thesis is a draft of the SBSP architecture that defines the different components of the SBSP project and systems. Orbit dynamics calculated for the demonstration via an Excel database. A systems design tool was generated for archive purposes. Finally the financial feasibility of the systems is established. The demonstration is designed to be modular and compatible with the International Space Station. The next few sections will give additional background information on the concept of demonstrating SBSP. Background on solar energy and sunlight are discussed in Section 2.1. Photovoltaic and solar cells are presented in Section 2.2. Microwave and laser theories are compared in Section 2.3. GEO and LEO are explained is Section 2.4. Launch vehicles are noted in Section 2.5. Lastly, Section 2.6 introduces the International Space Station as a test bed for this SBSP demonstration.

2.1 Solar Energy and Sunlight

Solar energy can be harvested by using solar radiation from the sun to generate electrical power. Solar radiation can be captured via photovoltaic cells and converted directly into electricity. Photovoltaic cells can be seen on home rooftops for power generation or in large fields connected to the utility grid (Tanton, 2008). Photovoltaic cells can convert solar energy directly into electricity. Figure 2.1 shows a block diagram with basic solar energy conversion system.

Figure 2.1: Photovoltaic cells to convert solar energy directly into electricity

The sun emits energy as electromagnetic radiation. Outside the Earth's atmosphere, solar radiation is constant and intense. On Earth, sunlight is filtered through the atmosphere, and solar radiation can be seen during the daytime as light. When clouds block direct radiation, The Earth does not experience direct sunshine. Sunshine is the combination of bright light and heat. The World Meteorological Organization defines sunshine as direct irradiance from the sun measured on the ground at a minimum of 1120 $W \cdot m^{-2}$ (Badescu, 2008). Table 2.1 summarizes the important characteristics of the sun.

Table 2.1: Characteristics of the sun
(Reprinted with permission from William Stine at www.powerfromthesun.net)

Present age	4.5×10^9 years
Life expectancy	10×10^9 years
Distance to Earth	
Mean	1.496×10^{11} m = 1.000AU
Variation	1.016735 to 0.98329 AU
Mass	1.987×10^{30} kg
Density	
Mean	14.1 kg/m^3
Center	1,600 kg/m^3
Composition	
Hydrogen	73.46%
Helium	24.85%
Oxygen	0.77%
Carbon	0.29%
Iron	0.16%
Neon	0.12%
nitrogen, silicon, magnesium, sulfur, etc.	<0.1%
Solar radiation	
entire Sun	3.83×10^{26} W
Unit area of surface	6.33×10^7 W/m^2
at 1 AU (i.e. the solar constant)	1,367 W/m^2
Temperature	
Center	15,000,000 K
Surface (photosphere)	6,050 K
Chromosphere	4,300-50,000 K
Corona	800,000-3,000,000 K
Rate of mass loss	4.1×10^9 kg/s

The intensity of the sun is approximately $6.33 \times 10^7 W/m^2$ and the average Earth-sun distance is 1.496×10^{11} m or 1.0 AU. The solar constant is the amount of energy received at the top of the Earth's atmosphere. The currently accepted value for the solar constant is $I_{sc} = 1367\left(W/m^2\right) = 1.367\left(kW/m^2\right)$ (Stine & Geyer, 2001). Figure 2.2 shows the US annual average solar energy received by a latitude-tilt photovoltaic cell.

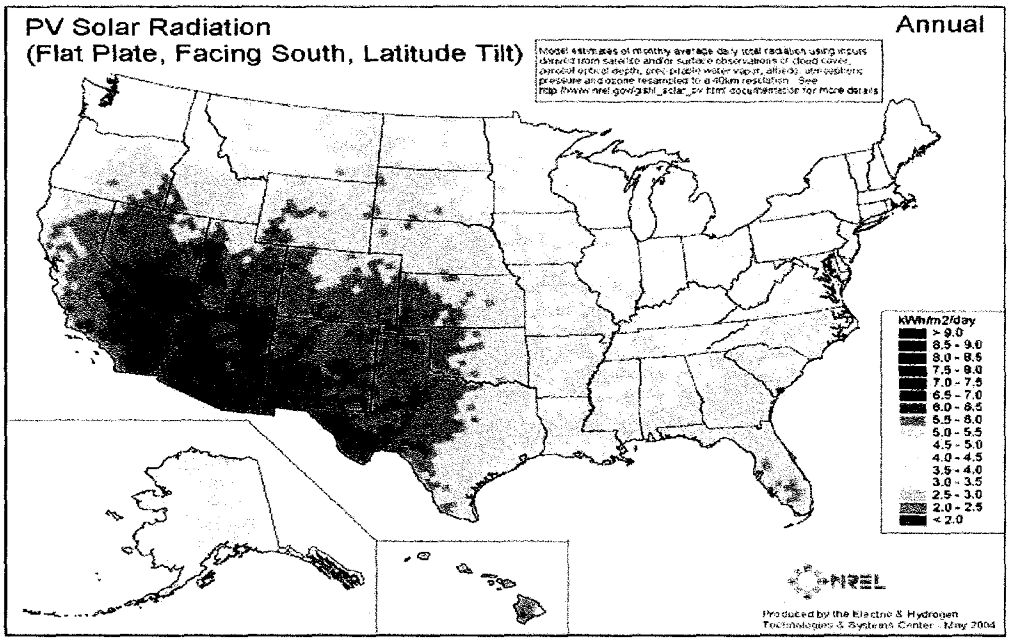

Figure 2.2: US Annual Average Solar Energy
(Courtesy of NASA)

Solar energy received is between 2.0 and 9.0 (kWh/m^2/day), taking the average value of solar energy received at 5.5 (kWh/m^2/day), then multiplying by 1 day/24 hours to get 0.299 (kW/m^2). The solar constant at the top of Earth's atmosphere is 1.367 (kWh/m^2). Equation 1 calculated 17% as the average solar constant captured on earth.

$$\frac{0.229\left(kW/m^2\right)}{1.367\left(kW/m^2\right)} = 0.168 = 17\% \tag{1}$$

As solar radiation passes through the Earth's atmosphere, it is absorbed, reflected, scattered, and transmitted directly. On a cloudy or foggy day the direct component of solar irradiance is essentially zero, and there are no shadows. The scattered component of solar irradiance is how we see shade. If there were no scattering component of solar irradiance, the sky would appear black as at night and stars would be visible throughout the day. The amount of this scattering light depends on the amount of water and dust in the atmosphere and the altitude of the observer above sea level (Stine & Geyer, 2001). Many other complexities are involved when trying to collect energy from the sun on Earth; the problem becomes simpler if solar energy is collected in orbit and then transmitted to earth.

2.2 Photovoltaic/Solar Cells

Sunlight is converted into electricity via the photoelectric effect with the use of photovoltaic cells also known as solar cells. Small packets of energy called photons in sunlight strike a photovoltaic cell filled with charge carriers, such as electrons. Photons may either be absorbed, reflected, or pass through the cell. In the case of absorption, the energy is passed to an electron in the cell. If the photon has sufficient energy, then the electron's energy level to the conduction level will be raised and it will move, thus creating an electrical current. In order to induce an electric field in the photovoltaic cell and increase electron flow, cells are made of two separate semiconductors, a p-type material and an n-type material. N-type materials are composed mainly of electrons, and p-type materials are composed of holes (charge carriers that combine with electrons). Semiconductors can be "doped" in order to create excess electrons or to create "holes" in the outer electron layers of the atoms, thus increasing current flow. Even though both materials are electrically neutral, n-type material has excess electrons and p-type material has excess holes. Sandwiching these together forms the p/n junction and creates an electric field. This electric field contributes to the movement of electrons during the photovoltaic effect (Lenardic, 2008).

Photovoltaic theory states that when light, in the form of photons, strike photosensitive material with an energy value higher than the band gap of semiconductor material of the solar cell, electrons are excited and begin to move. The built-in electric potential of the solar cell creates a current flow. Voltage and current create power. The amount of solar power collected is affected by the Beta angle, the angle between the orbital plane and the light of the Sun, which can be seen in equation 2 (Lenardic, 2008).

$$\sin(\beta) = \sin(i)\cos(\delta)\sin(\theta - \theta_S) + \cos(i)\sin(\delta) \qquad (2)$$

Where: i is the inclination of the orbit

δ is the declination of the orbit

θ is the right ascension of the orbit

θ_S is the right ascension of the Sun

Beta is the angle between the orbit plane and the vector from the Sun shown in Figure 2.3.

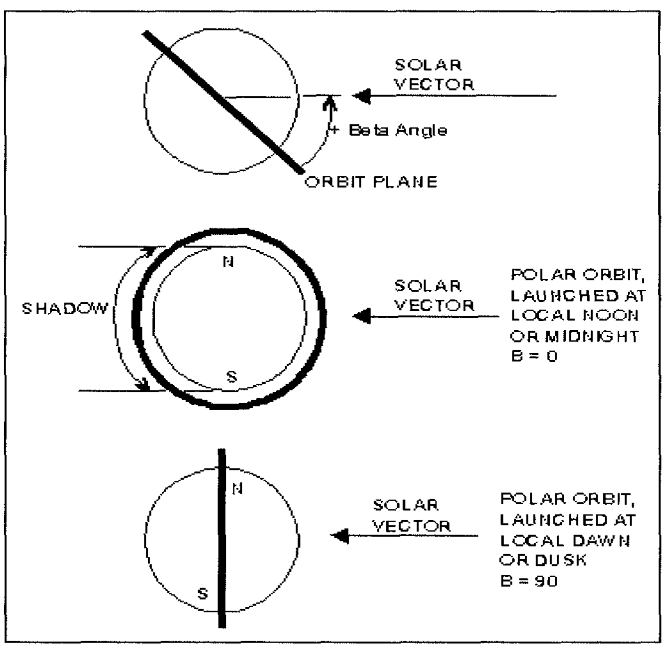

Figure 2.3: Beta Angle from the Sun
(Reprinted with permission from Tim Kelly at K&K Associates
http://www.tak2000.com)

A solar cell model consists of a diode and current source connected in parallel. Solar radiation is directly proportional to the current source. A diode represents the p-n junction of a solar cell. A diode is a device that either allows or prevents current flow between the p and n type material depending on the voltage applied at each end. Equation 3 represents the ideal solar cell model.

$$I = I_{Ph} - I_s \left(\frac{V}{e^{m \cdot V_T}} - 1 \right) \tag{3}$$

Where: I_{Ph} is photocurrent (A)

I_s is reverse saturation current (A) (approximately $10^{-8}/m^2$)

V is diode voltage (V)

V_T is thermal voltage (see Equation 4 below)

$V_T = 25.7\ mV$ at 25°C, m assuming diode ideality factor = 1 (4)

Thermal voltage V_T can be calculated with Equation 5.

$$V_T = \frac{k \cdot T}{q} \tag{5}$$

Where: k is the Boltzmann constant = 1.38 x10^{-23} (J/K)

T is temperature in degrees Kelvin (K)

q is the charge of an electron = 1.6x10^{-19} (A)

There is a specific type of solar cell called “thin-film cell”. Thin-film cells are approximately four-one-hundred-thousandths of an inch thick. The advantages of thin-film solar cells include their light weight and fewer required materials for fabrication, which reduce the cost per solar cell. Table 2.2 compares the different solar cell types available, along with their thickness, efficiency, colors, and features (Lenardic, 2008).

Table 2.2: Overview of Solar Cell Materials

(Reprinted with permission from Denis Lenardic at www.pvresources.com)

Material	Thickness	Efficiency	Colors	Features
Monocrystalline Si solar cells	0.3 mm	15 - 18 %	Dark blue, black with AR coating, grey without AR coating	Lengthy production procedure; wafer sawing necessary. Best researched solar cell material - highest power/area ratio.
Polycrystalline Si solar cells	0.3 mm	13 - 15 %	Blue with AR coating, silver-grey without AR coating	Wafer sawing necessary. Most important production procedure at least for the next ten years.
Polycrystalline transparent Si solar cells	0.3 mm	10 %	Blue with AR coating, silver-grey without AR coating	Lower efficiency than monocrystalline solar cells. Attractive solar cells for different BIPV applications.
EFG	0.28 mm	14 %	Blue, with AR coating	Limited use of this production procedure Very fast crystal growth, no wafer sawing necessary
Polycrystalline ribbon Si solar cells	0.3 mm	12 %	Blue, with AR coating, silver-grey without AR coating	Limited use of this production procedure, no wafer sawing necessary. Decrease in production costs expected in the future.

Apex (polycrystaline Si) solar cells	0.03 to 0.1 mm + ceramic substrate	9,5 %	Blue, with AR coating, silver-grey without AR coating	Production procedure used only by one producer, no wafer sawing, production in form of band possible. Significant decrease in production costs expected in the future.
Monocrystaline dendritic web Si solar cells	0.13 mm incl contacts	13 %	Blue, with AR coating	Limited use of this production procedure, no wafer sawing, production in form of band possible.
Amorphous silicon	0.0001 mm + 1 to 3 mm substrate	5 – 8 %	Red-blue, Black	Lower efficiency, shorter life span. No sawing necessary, possible production in the form of band.
Cadmium Telluride (CdTe)	0.008 mm + 3 mm glass substrate	6 – 9 % (module)	Dark green, Black	Poisonous raw materials, significant decrease in production costs expected in the future.
Copper-Indium-Diselenide (CIS)	0.003 mm + 3 mm glass substrate	7.5 – 9.5 % (module)	Black	Limited Indium supply in nature. Significant decrease in production costs
Hybrid silicon (HIT) solar cell	0.02 mm	18 %	Dark blue, black	Limited use of this production procedure, higher efficiency, better temperature coefficient and lower thickness.

2.3 Microwave Transmission/Laser Beam

Microwaves are electromagnetic waves with wavelengths ranging from 1 mm to 1 m, or frequencies between 0.3 GHz and 300 GHz. Microwave power transmissions (MPT) are the use of microwaves to transmit power through outer space or the atmosphere wirelessly. Microwaves are coherent and polarized in contrast to visible waves (apart from lasers). They obey the laws of optics and can be transmitted, absorbed or reflected depending on the type of material. A rectenna is a rectifying antenna that is used to directly convert microwave energy into DC electricity. In principle, the rectenna is capable of very high conversion efficiencies - over 90% in optimal circumstances. A rectenna can be used to capture transmitted microwaves (Hill, 2000).

A loss to the oscillating electric field is related to the absorption of microwaves material's complex permittivity ε in equation 6.

$$\varepsilon = \varepsilon_0 \left(\varepsilon' - i\varepsilon'' \right) \tag{6}$$

Where: ε_0 is the permittivity of free space ($\varepsilon_0 = 8.86 \times 10^{-12}$ F/m)

ε' is the relative dielectric constant

ε'' is the effective relative dielectric loss factor.

Commonly used to describe the losses is the loss tangent $(\tan\delta)$ in equation 7.

$$\tan\delta = \frac{\varepsilon''}{\varepsilon'} = \frac{\sigma}{2\pi f \varepsilon_0 \varepsilon'} \tag{7}$$

Where: σ is the total effective conductivity (S/m) caused by ionic conduction and displacement currents

f is the frequency

The power absorbed per unit volume is described by equation 8.

$$P = \sigma|E|^2 = 2\pi f \varepsilon_0 \varepsilon' \tan\delta |E|^2 \qquad (8)$$

Where: $|E|$ is the magnitude of the internal electric field (V/m)

Figure 2.4 shows the electromagnetic spectrum. Laser is infrared (IR) and microwave wavelength is to the right of IR (Electromagnetic Radiation, 2009).

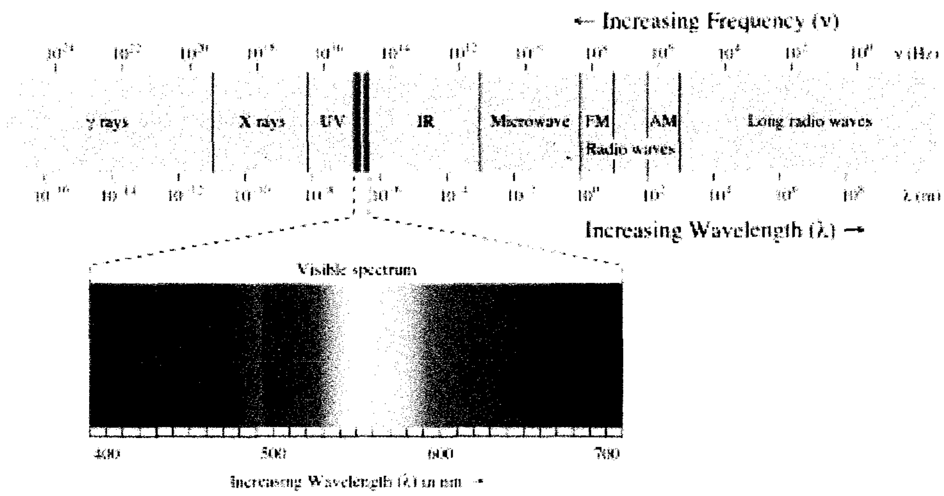

Figure 2.4: Electromagnetic Spectrum with Visible Light Highlighted
(Courtesy of NASA from wikipedia.org)

Laser is an acronym for Light Amplification by Stimulated Emission of Radiation. A laser is a device that emits light (electromagnetic radiation) through a process called stimulated emission. In optics, stimulated emission is the process by which an electron, agitated by a photon having the correct energy, may drop to a lower energy level resulting in the creation of another photon. Electromagnetic radiation takes the form of self-propagating waves in a vacuum or in matter. Infrared radiation (IR) is electromagnetic radiation whose wavelength is longer than that of visible light, but shorter than that of terahertz radiation and microwaves. Infrared radiation has wavelengths between about 750 nm and 1 mm, spanning three orders of magnitude. The color of light is determined by its frequency or wavelength. The shorter wavelengths are the ultraviolet and the longer wavelengths are the infrared. The smallest particle of light energy is described by quantum mechanics as a photon. The energy of a photon is shown in equation 9.

$$E = h \cdot \nu \tag{9}$$

Where: ν is the frequency

h is Planck's constant

The wavelength of light is related to the frequency as shown in equation 10.

$$\lambda = \frac{c}{\nu} \tag{10}$$

Where: λ is the wavelength of light

c is the velocity of light in vacuum ($300 m/s^2$)

Table 2.3 shows the various types of material currently used for lasing and the wavelengths that are emitted by that type of laser. Note that certain materials and gases are capable of emitting more than one wavelength. The wavelength of the light emitted in this case is dependent on the optical configuration of the laser.

Table 2.3: Common Lasers and Their Wavelengths

(Reprinted with permission from Josee Sansoucy at www.mcgill.ca)

Laser Type	Active Medium	Wavelength, nm
Excimer Gas Lasers	Argon Fluoride	193 nm
	Krypton Fluoride	248 nm
	Xenon Chloride	308 nm
	Xenon Fluoride	351 nm
Gas Lasers	Nitrogen	337 nm
	Helium Cadmium	325 nm, 442 nm
	Argon	488 nm, 514 nm
	Krypton	647 nm
	Helium Neon	633 nm
	Carbon Dioxide	10,600 nm
Solid State Lasers	Doubled Nd:YAG	532 nm
	Nd:YAG	1,064 nm
	Ruby	694 nm
	Ti:Sapphire	700-1,100 nm
Dye Lasers	Rhodamine 6G	570-650 nm
Semiconductor Lasers	Galium Arsenide (GaAs)	850 nm, 905 nm
	InGaAlP	670 nm
	GaAlAs	750-900 nm
	InGaAsP	1300-1600 nm
Fiber lasers	Er:doped optical fiber	1550 nm

Laser light has three unique characteristics, which make it different than ordinary light. It is monochromatic, directional, and coherent. Monochromatic consists of one single color or wavelength. Even through some lasers can generate more than one wavelength, the light is extremely pure and consists of a very narrow spectral range. Directional means that the beam is well collimated and travels over long distances with very little spread. Coherent means that all individual waves of light are moving precisely together in phase strongest of the light waves (Aldrich, 2008).

2.4 Geostationary Orbit (GEO)/Lower Earth Orbit (LEO)

A geosynchronous orbit directly above the Earth's equator is the Geostationary orbit (GEO). The GEO period is equal to the Earth's rotational period and has an orbital eccentricity of approximately zero. On earth, geostationary objects appear motionless in the sky, making the GEO an orbit of great interest to operators of communications and weather satellites. Due to the constant 0° latitude and circularity of geostationary orbits, satellites in GEO differ in location by longitude only.

A Geosynchronous Transfer Orbit or Geostationary Transfer Orbit (GTO) is the Hohmann transfer orbit around the Earth between lower Earth orbit (LEO) and geosynchronous orbit (GEO). Figure 2.5 shows that a Hohmann transfer orbit is an ellipse where the perigee is a point on a LEO and the apogee has the same distance from the Earth as the GEO (Hohmann Transfer Orbit, 2009).

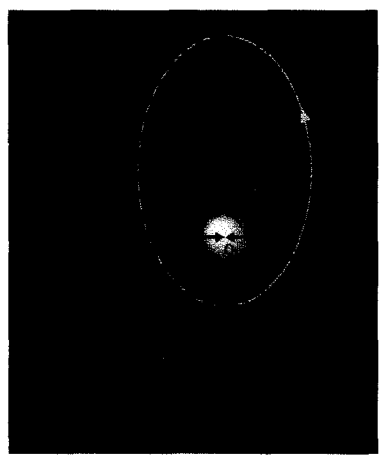

Figure 2.5: Hohmann Transfer Orbit
(Courtesy of NASA from wikipedia.org)

Heavy Lift Launch Vehicles are the only rockets capable of moving heavier satellites into geostationary or geosynchronous orbits. After a typical launch the inclination of the LEO is determined by the latitude of the launch site and the direction of launch. The GTO typically inherits the same inclination. The inclination must be reduced to zero to obtain a geostationary orbit. Most of the delta-v (ΔV) for this inclination change is done at the GEO distance because that requires less energy than at LEO. This is because the required

ΔV for a given inclination change Δi is directly proportional to orbit velocity V which is lowest in its apogee. The required ΔV for an inclination change in either the ascending or descending orbital node of the orbit is calculated from equation 11:

$$\Delta V = 2V \sin\frac{\Delta i}{2} \tag{11}$$

For a typical GTO with a semi-major axis of 24,582 km, the perigee velocity of a GTO is 9.88 km/s while the apogee velocity is at 1.64 km/s. Therefore it is most efficient to change inclination at GEO. However, note that in actual operation, the inclination change is combined with the orbital circularization burn, and considerably less ΔV is required than the above calculation would imply. For a small-scaled Space-Based Solar demonstration, it will only be operating in a LEO orbit, which is where the ISS is orbiting. However, for SBSP full-scale operation, we will need to consider GTO and GEO. Many recent SBSP studies suggested GEO as an operational orbit to house the large solar panels. As stated above Hohmann transfer should be the method used to transfer a payload from LEO through GTO to GEO (Hohmann Transfer Orbit, 2009).

2.5 Launch Vehicles

A launch vehicle is a rocket used to carry a payload from the earth's surface into outer space. There are two types of launch vehicles. Expendable launch vehicles are designed for one-time use. They usually separate from their payload, and may break up during atmospheric reentry. Reusable launch vehicles, on the other hand, are designed to be recovered intact and used again for subsequent launches (Launch Vehicle, 2009). A trade-off study was conducted to compare expendable vs. reusable launch vehicles. More differences between them and cost analysis of the various systems is discussed later in this paper.

2.6 International Space Station (ISS)

The International Space Station (ISS) is a research facility currently being assembled in outer space in LEO orbit. The space station is a joint project among the space agencies of the United States (NASA), Russia (RKA), Japan (JAXA), Canada (CSA) and eleven European countries (ESA). Assembly began in 1998, and as of July 2008 the station is approximately 85% complete. The source of electrical power for the ISS is the sun: light is converted into electricity through the use of solar arrays. Before the Space Shuttle mission STS-97, (November 30, 2000) the only power source was the Russian solar panels attached to the Zarya and Zvezda modules: the Russian segment of the station uses 28 volts DC. In the remainder of the station, electricity is provided by the solar arrays attached to the truss at a voltage ranging from 130 to 180 volts DC. The power is then stabilized and distributed at 160 volts DC, before finally being converted to the user-required 124 volts DC. The high-voltage distribution line allows for smaller power lines, reducing weight. Power can be shared between the two segments of the station using converters, and this feature is essential because of the cancellation of the Russian Science Power Platform. Russian segment will depend on the US built solar arrays for future power supplies.

Figure 2.6 shows the ISS in 2001, showing the solar panels on Zarya and Zvezda.

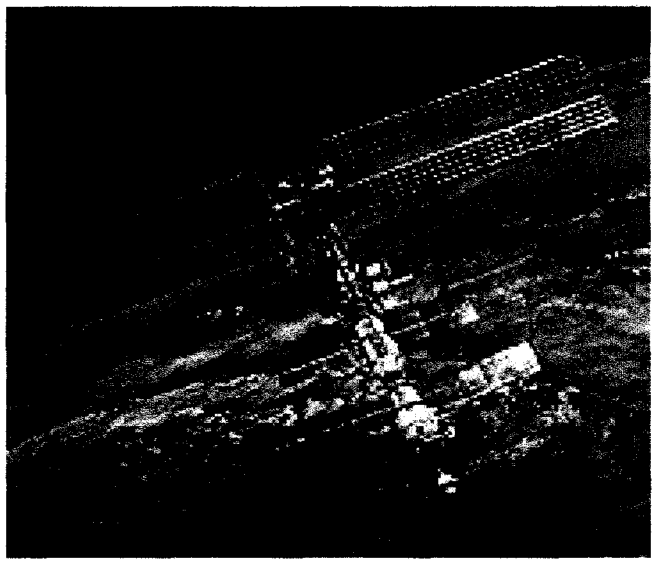

Figure 2.6: The ISS in 2001, Showing the Solar Panels on Zarya and Zvezda
(Courtesy of NASA from wikipedia.org)

The solar array normally tracks the sun to maximize the amount of solar power. The array is about 375 m^2 (450 yd^2) in area and 58 meters (190 ft) long. In the fully-complete configuration, the solar arrays track the Sun in each orbit by rotating the alpha gimbal; while the beta gimbal adjusts for the angle of the Sun from the orbital plane (International Space Station, 2009).

2.7 Summary of Background

Some background information on solar energy and sunlight, photovoltaic and solar cells, microwaves vs. laser, geostationary orbit (GEO) and lower Earth orbit (LEO), launch vehicles, and the International Space Station (ISS) should provide a better understanding of Space-Based Solar Power. The Sun radiates solar energy, and photovoltaic cells make it possible to convert solar energy into electricity. The two methods of transmitting solar power are microwave transmission and laser beaming. Ideally SBSP should be brought to GEO for operation, and a demonstration will be performed in LEO. There are two types of launch vehicles, expendable vs. reusable, which can take payload into orbit. Launch vehicles are a major constraint for space access. A cost analysis of affordable Launch vehicles will be discussed later in this paper. The International Space Station National Laboratory is a great test bed for this type of technology demonstration. This thesis proposes a conceptual design of an solar power beaming space system demonstration that fits well with the goals and capabilities of the Space Station. The next chapter presents the systems engineering of the SBSP demonstration

Chapter 3: Systems Engineering

Lowering cost to space access and supporting infrastructure is the first step to enabling a viable system of Space-Based Solar Power (SBSP). The concept of SBSP consists of an assembly of large arrays of connected solar panels in high Earth orbit. The arrays collect solar energy and then convert it to electricity and transmit it wirelessly to an earth-based ground station via laser or microwave transmission. Once this energy is beamed down to earth, it can then be plugged into the existing energy grid and distributed as electrical power to customers.

Solar Power Satellites (SPS) have been studied for over thirty years by NASA and the Department of Energy. In the 1970's SPS was studied using then-current technologies that showed technical feasibility. NASA looked into Space Solar Power from 1999 to 2001. As the time the National Research Council found the program to have a solid foundation but it required significant funding increases. The cost was too high. All funding for Space Solar Power was canceled after September 2001, and no Research and Development work has been done by NASA since. The two main factors are the cost program delivered watt of the solar power components, and the cost per delivered watt of getting those components to their final destination in space. The cost of components is the first problem; current prices for solar electric power systems are about $2.50 per peak Watt. The day/night cycle, non-deal Sun angles, weathering, and cloud cover reduce power output enough to make the final cost per average watt $10 or more. In space you can get peak power constantly whereas on earth, compensation is required due to loss in transfer. Component costs are potentially much closer to wholesale utility requirements for space solar power than they are for terrestrial solar power. The other cost of concern is delivery to orbit. Typical communications satellite solar panels have a mass per kW of about 20 kg; therefore, a current launch cost at $10,000/kg is $200/W. In order to bring that number down, improvements in both mass per kW and cheaper access to space are required. Component and launch will not be the only costs. Improved robotics and computational capabilities will also make SBSP cheaper (Aldrich, 2008).

Peter Glaser first wrote about Solar Power Satellites in 1968. William Brown proved the potential of wireless power transmission about that time, and solar power from space was an important part of physicist Gerard O'Neill's inspiring call to space in his 1977 book "The High Frontier". Organizations such as the Space Studies Institute and the National Space Society (NSS) continue to see space solar power as part of their vision of a space-faring future. In the following section I will discuss systems engineering design for the conceptual solar power beaming space system (Smith, 2003).

Systems engineering is a methodical, disciplined approach for the design, technical management, operations, and retirement of a system. A "system" is a construct or collection of different elements that together produce results not obtainable by the elements alone. Systems engineering is the art and science of developing an operable system capable of meeting requirements within often opposed constraints (National Aeronautics and Space Administration, 2007).

3.1 Functional Flow Block Diagrams (FFBDs)

There are many techniques to perform functional analysis. Some of the more popular include:

(1) Functional Flow Block Diagrams (FFBDs) to depict task sequences and relationships,

(2) N^2 diagrams (or N x N interaction matrix) to identify interactions or interfaces between major factors from a systems perspective, and

(3) Timeline Analyses (TLAs) to depict the time sequence of time-critical functions.

The primary functional analysis technique is the functional flow block diagram (FFBD). The purpose of the FFBD is to indicate the sequential relationship of all functions that must be accomplished by a system. When completed, these diagrams show the entire network of actions that lead to the fulfillment of a function (National Aeronautics and Space Administration, 2007).

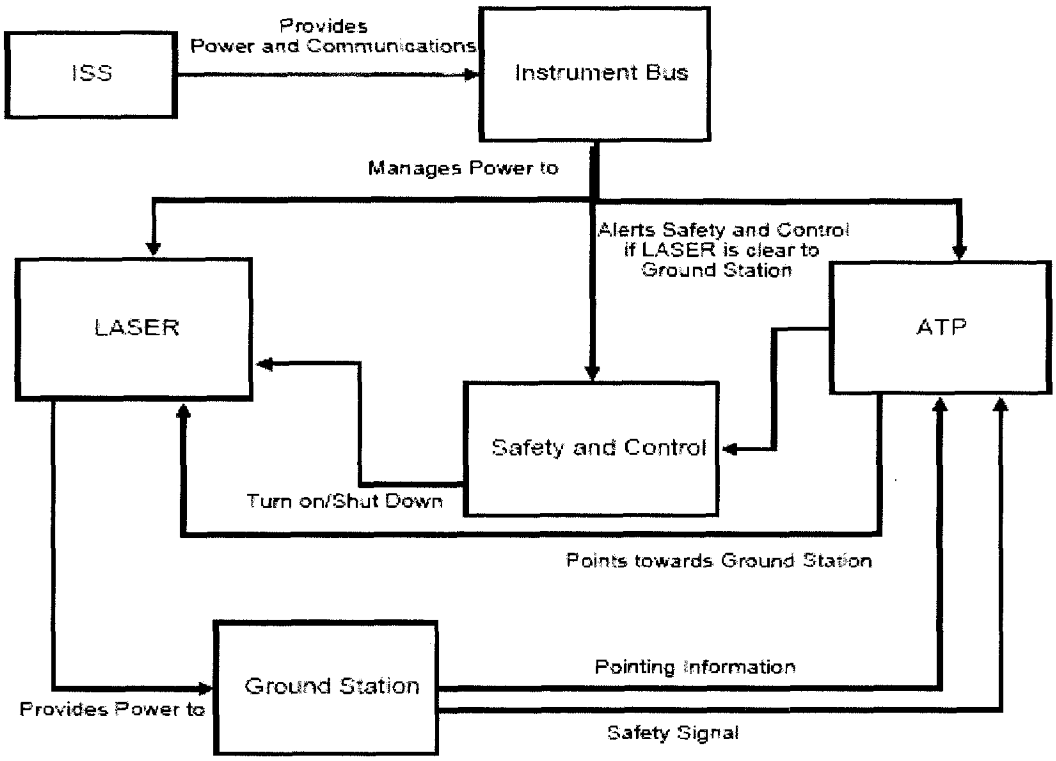

Figure 3.1: SBSP Demonstration Functional Flow Block Diagrams

The functional flow block diagram shown in figure 3.1 illustrates that the ISS provides power and communications to an Instrument Bus. The Instrument Bus manages power to Safety and Control, Laser, and ATP. Instrument Bus alerts Safety and Control if Laser is clear for Ground Station. Safety and Control controls turn on and shut off power for Laser. The ATP system works along side with Safety and Control to control and point Laser towards Ground Station. Ground Station provides pointing information and safety signal to ATP. Laser provides power to Ground Station.

3.2 N-squared Diagram

The N-squared (N2 or N^2) diagram is used to develop system interfaces. The system components are placed on the diagonal; the remainder of the squares in the N x N matrix represents the interface inputs and outputs. Where a blank appears, there is no interface between the respective components. The N2 diagram can be taken down into successively lower levels to the component functional levels. In addition to defining the interfaces, the N2 diagram also pinpoints areas where conflicts could arise in interfaces, and highlights input and output dependency assumptions and requirements (National Aeronautics and Space Administration, 2007).

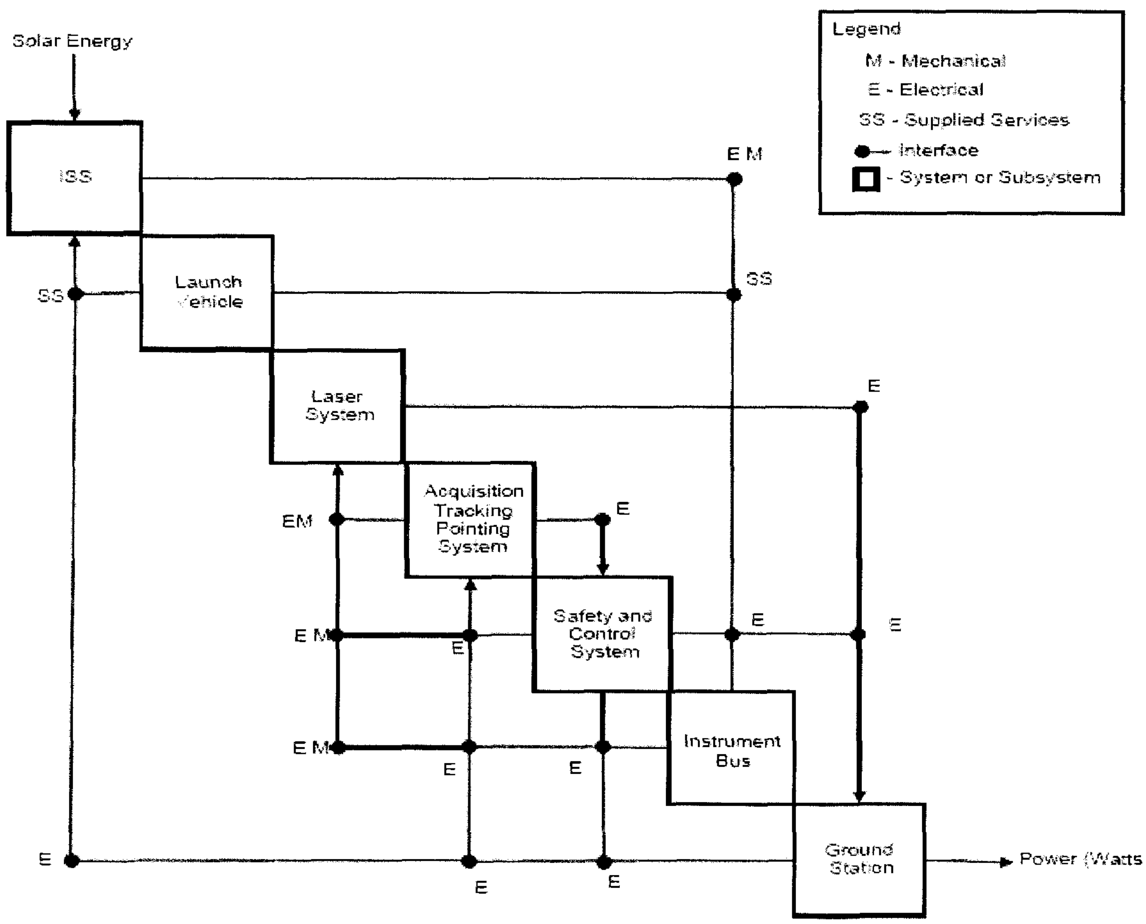

Figure 3.2: SBSP Demonstration N^2 Diagram

Figure 3.2 shows the International Space Station (ISS) collects solar energy from the Sun. The ISS is connected to the Instrument Bus via electrical and mechanical interfaces. The

ISS houses the Instrument Bus and provides the instrument bus with power and communications. Launch Vehicles deliver the Instrument Bus to the ISS and it is interfaced via supplied services. A Laser system beams down power to Ground stations and therefore, is communicating with Ground Station via an electrical interface. Acquisition, Tracking, and Pointing System help to track safety signals from Ground Stations, which alerts the Safety and Control System if the link is impeded. The ATP System supports tracking and pointing of the laser toward the Ground Station. The Safety and Control System has a laser curtain component to protect against objects potentially flying through the laser. ATP as well as Safety and Control System interface with Instrument Bus and Ground Station via electrical interface.

The Ground Station feedback is linked to the ISS via electrical interface. Ground Station signals to Safety and Control System to assure laser path to Ground is unimpeded. Ground Station also has a communication component which assists with the pointing and tracking of the laser. Instrument Bus is interfaced with Laser System via electrical and mechanical Interfaces. The Instrument Bus manages power to Safety and Control System, manages power to communications to the ATP System, and manages power to the laser system via electrical interface.

3.3 Product Breakdown Structure (PBS)

The top-level requirements and expectations are initially assessed to understand the technical problem to be solved and establish the design boundary. This boundary is typically established by performing the following activities:

(1) Defining constraints that the design must adhere to or how the system will be used. The constraints are typically not able to be changed based on tradeoff analyses.

(2) Identifying those elements that are already under design control and cannot be changed. This helps establish those areas where further trades will be performed to narrow potential design solutions.

(3) Establishing physical and functional interfaces (e.g., mechanical, electrical, thermal, and human) with which the system must interact.

(4) Defining functional and behavioral expectations for the range of anticipated uses of the system as identified in the ConOps. The ConOps describes how the system will be operated and the possible use-case scenarios.

A complete set of the project requirements includes the functional needs requirements, performance requirements, and interface requirements. For space projects, these requirements are decomposed and allocated down to design elements through the PBS (National Aeronautics and Space Administration, 2007).

The SBSP Demonstration "A" in figure 3.3 below was broken down into three sub-systems:

- A1.Space
- A2.Safety and Control
- A3.Ground Station

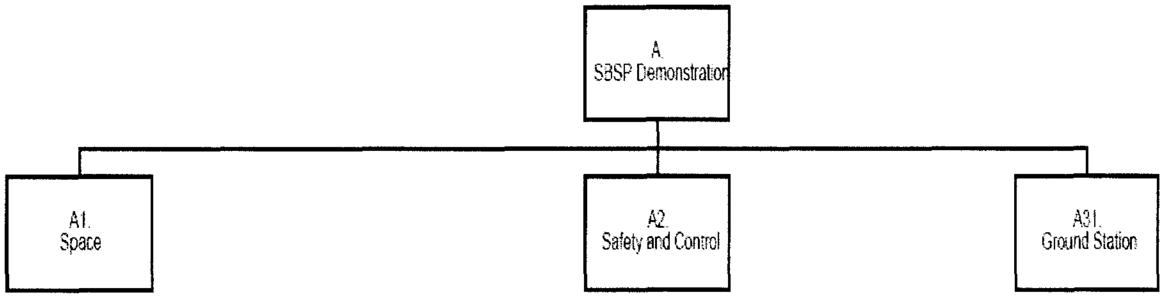

Figure 3.3: SBSP Demonstration "A" Product Breakdown Structure

Each of the Space, Safety & Control, and Ground Station sub-systems will then be broken down furthermore into individual sub-systems.

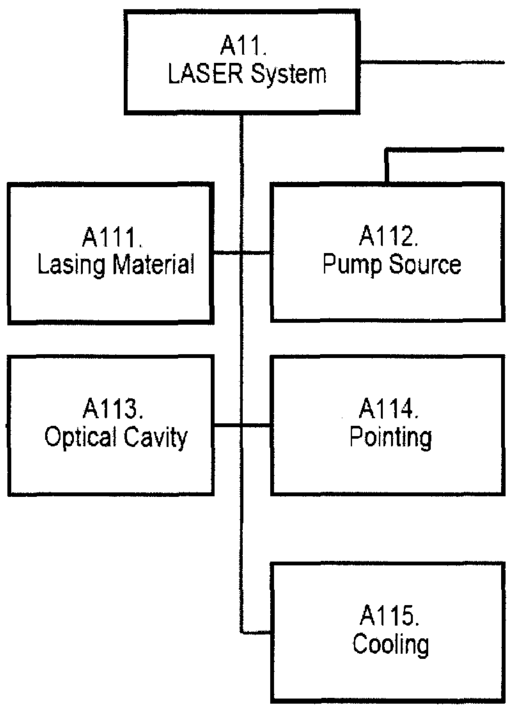

Figure 3 4: SBSP Demonstration "A11" Laser System

For example A1.Space subsystem is Laser System A11 as seen in figure 3.4. Laser system was broken down into its individuals sub-systems:

- A111.Lasing material
- A112.Pump Source

- A113.Optical Cavity
- A114.Pointing
- A115.Cooling

Laser as a system requires all its sub-systems in order to be functional. Likewise, figure 3.4 illustrate A12., the Acquisition, Tracking, Pointing (ATP) System. This was broken down to:

- A121.Tracking
- A122.Acquisition
- A123.Pointing
- A124.Isolation and Stabilization

A123.Pointing as a sub-system can also be broken down to it individual sub-system.

- A1231.Gimbal

A124.Isolation and Stabilization can also be broken down to individual sub-system

- A1241.Gyros

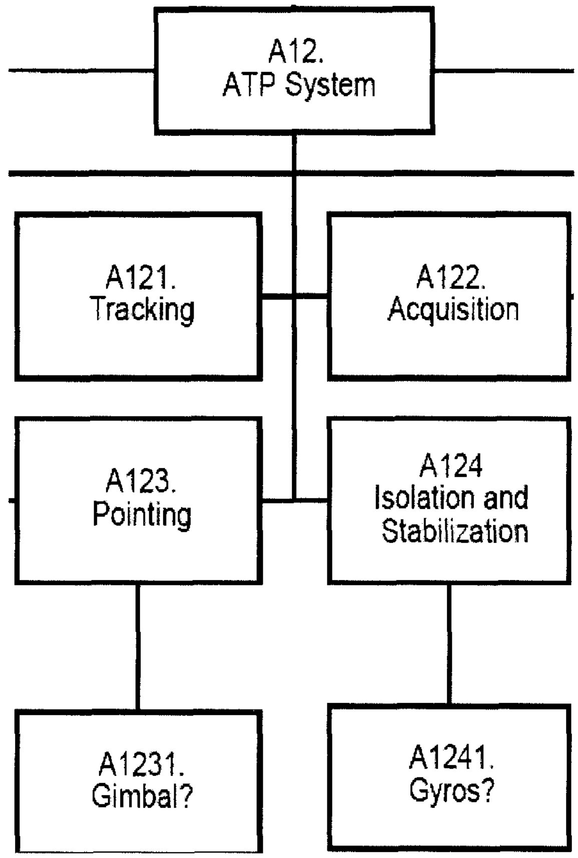

Figure 3 5: SBSP Demonstration "A12" ATP System

In addition, figure 3.6 shows A13.Instrument Bus System broken into three sub-systems:

- A131.Power Management
- A132.Command and Control
- A133.Communications

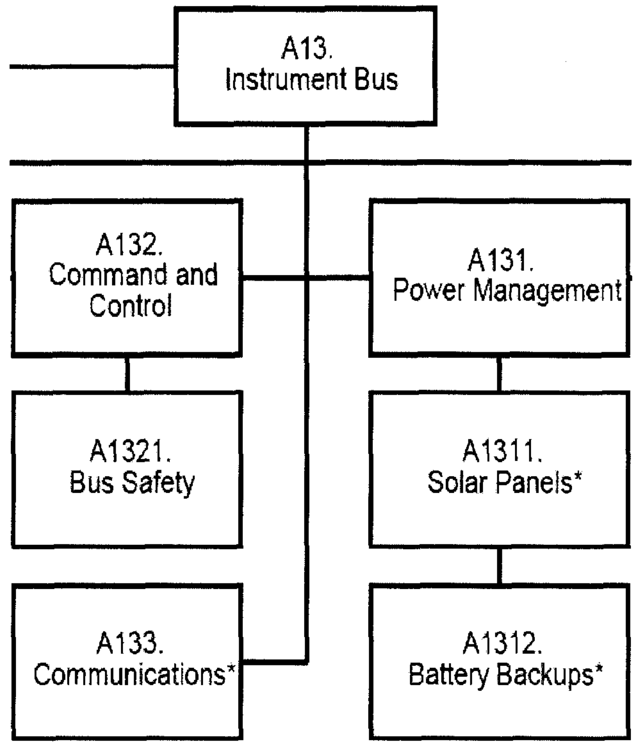

Figure 3.6: SBSP Demonstration "A13" Instrument Bus

A131.Power Management was then broken into two individual sub-systems:

- A1311.Solar Panels
- A1312.Battery Backups.

A132.Command and Control was broken down to

- A1321.Bus Safety

The completed Product Breakdown Structure can be seen in figure 3.7. A2.Safety and Control and A3.Ground Station systems can also be seen broken down to their individual sub-systems in figure 12. In addition, B1.Launch Vehicle is the SBSP system constraint, where B11 is Payload and B12 is Launch Vehicle Environment.

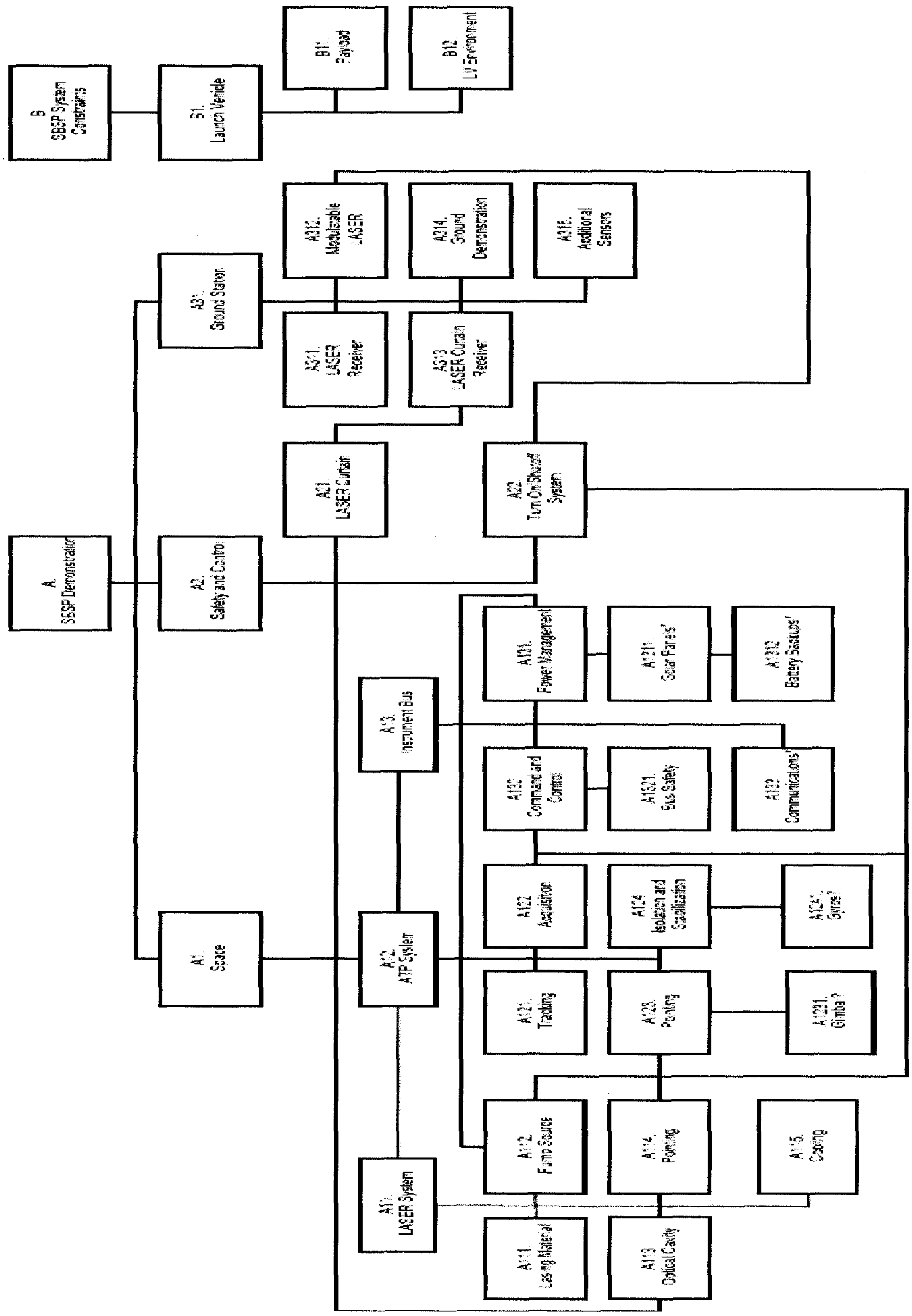

Figure 3.7: SBSP Demonstration Completed Product Breakdown Structure

Chapter 4: Tradeoffs Studies

The following sections will analyze the tradeoffs studies for expendable vs. reusable launch vehicles, microwave vs. laser, and present the preliminary SBSP demonstration. In addition section 4.3 will explain the reasoning behind the down select of LEO for demonstration, GEO for operation, and Thin Film Solar Cells.

4.1 Launch Vehicles (Expandable vs. Reusable) Cost Analysis

There are about twenty countries with advanced-launch capabilities. Only a few out of those twenty countries have developed reusable launch vehicles. My launch vehicles tradeoff study in Appendices I and II shows that reusable launch vehicles cost less than expendable launch vehicles. For example, I compared the Russian Proton, the Chinese Long March, and the United State, SpaceX Falcon 9 Normal/Heavy launch vehicles below (Wertz, Eonomic Model of Reusable vs. Expendable Launch Vehicles, 2000).

Table 4.1: Compare existing expendable vs. Reusable launch vehicles cost

Launch Vehicle	Country	LV Type	Launch Cost	Payload to Orbit	Cost per kg
Proton	Russia	Expendable	$85 Million	4,600 kg to GTO	$18,350
Long March	China	Expendable	$60 Million	5,200 kg to GTO	$11,500
Falcon 9 Normal	USA	Reusable	$35 Million	4,900 kg to GTO	$10,500
Falcon 9 Heavy	USA	Reusable	$78 Million	12,000 kg to GTO	$8,200

Table 4.1 compares existing launch vehicles cost for both expendable and reusable. Launch cost for the Russia Proton expendable vehicles is highest at $85 million. Launch cost for Falcon 9 Normal reusable vehicles is lowest at $35 Million. A payload capacity

to orbit for Proton is lowest at 4,600 kg to GTO whereas Falcon 9 Heavy payload to orbit is 12,200 kg to GTO. The costs per kg are higher for expendable than reusable launch vehicles: Proton at $18,350 per kg, Long March at $11,500 per kg, Falcon 9 Normal at $10,500 per kg, and Falcon 9 Heavy at $8,200 per kg. Therefore, for low cost access to space, reusable launch vehicles are the way to go.

4.2 Microwave vs. Laser

One of the major challenges of Space-Based Solar Power is not having an actual on-orbit demonstration of watts beamed down from orbit to measure the losses. This thesis proposes a demonstration of a 1200W laser system on the International Space Station. The reason for laser beaming over microwave transmitting is because laser transmission allows for components that are practical to use.

4.3 SBSP Demonstration

The demonstration shown in figure 4.1 is the SBSP design architecture.

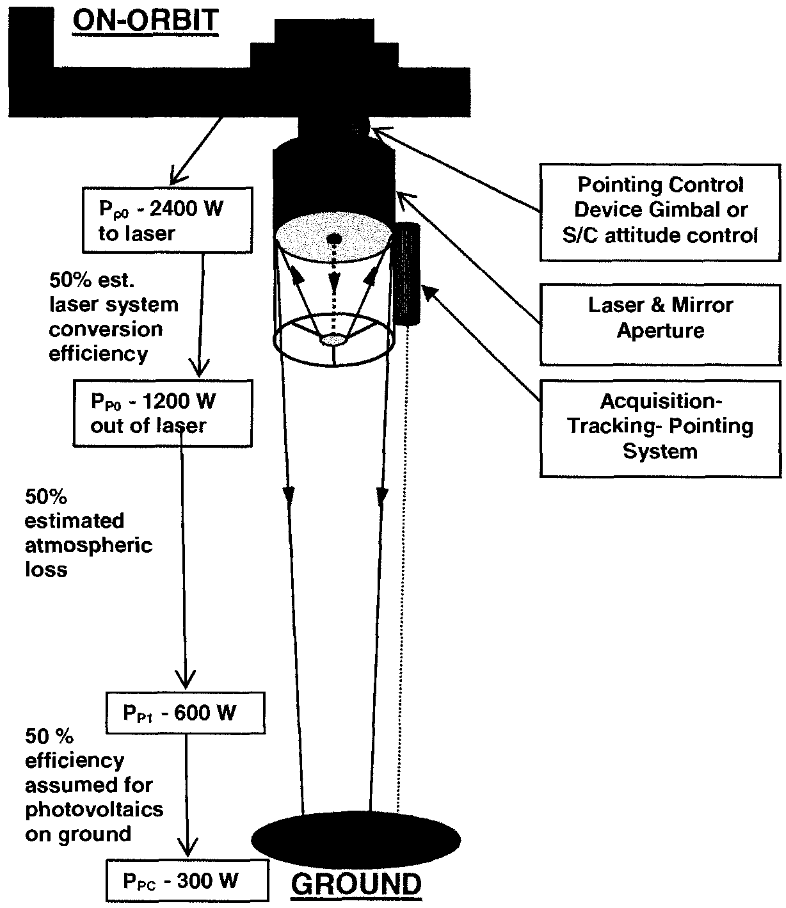

Figure 4.1: SBSP Demonstration Concept

This is an attached payload demonstration on the International Space Station (ISS). A satellite mass payload of approximately 100kg payload including: a pointing control device (perhaps a gimbal or spacecraft attitude control), a laser with mirror aperture, and an acquisition, tracking, pointing (ATP) system. The ground station receiver on earth will be approximately 10 meters in diameter of photovoltaic including ground support systems. The demonstration concept is to either be attached to the U.S FRAM or Japan JEM. It is being assumed that the ISS provides electrical power and many of the spacecraft functions. The demonstration will required approximately 2500W from the ISS, allow 100W to support systems, and 2400W for laser system. With a prediction of 50% estimated laser system conversion efficiency 2400W to laser will be reduced to 1200W out of laser. Another 50% estimated atmospheric loss will reduce power to 600W and finally an anticipated 50% efficiency assumed for photovoltaic on the ground. The laser power demonstration is 300W on the ground (Grady, 2008).

Figure 4.2 shows the receiver sizing from sunsats in GEO with PV/microwave and PV/laser. Microwave beam diffraction is limited by $\lambda \approx 0.12m,\ D_1 = 1km$; therefore, the diameter spot is $d_{spot} = 2(\lambda / D_1) = h$, where h for microwave is 40,000 km. The calculated receiver size is then 10km.

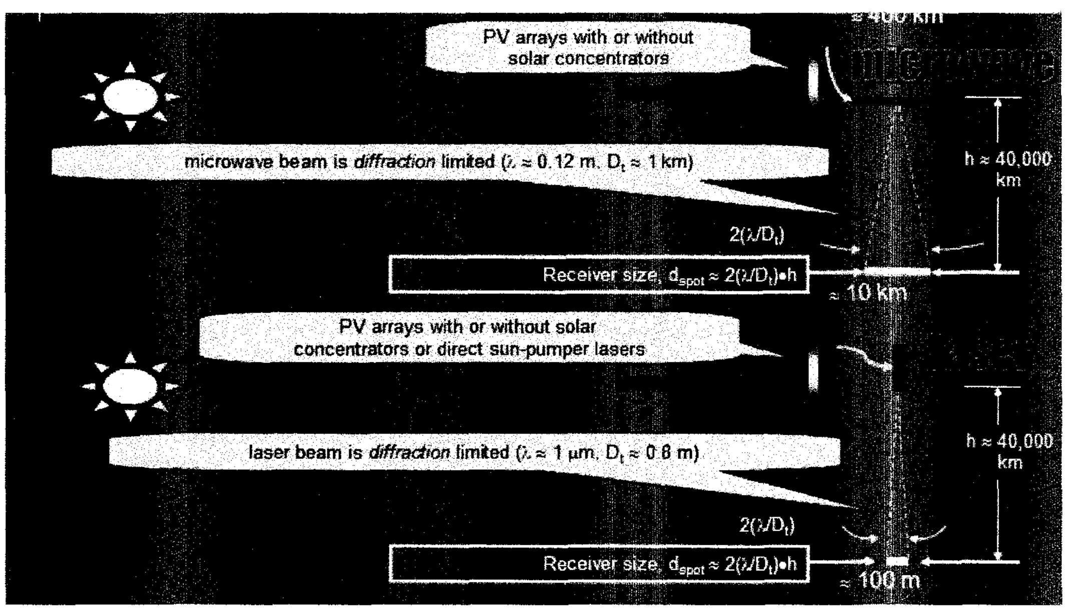

Figure 4.2: SBSP Demonstration Microwave vs. Laser
(Reprinted with permission from Eric Hoffert from www.sspi.gatech.edu/ssp_fundable_demo_hoffert.ppt)

Laser beam is diffraction limited by $\lambda \approx 0.1\mu m,\ \ D_1 = 0.8m$; therefore, if diameter spot is $d_{spot} = 2(\lambda / D_1) = h$, where h for laser is 40,000 km. The PV arrays with or without solar concentrators with direct sun-pumper laser and the laser receiver size is 100m. (Hoffert & Hoffert, 2008).

Figure 4.3 shows the SBSP demonstration estimated value for the laser beam ground spot.

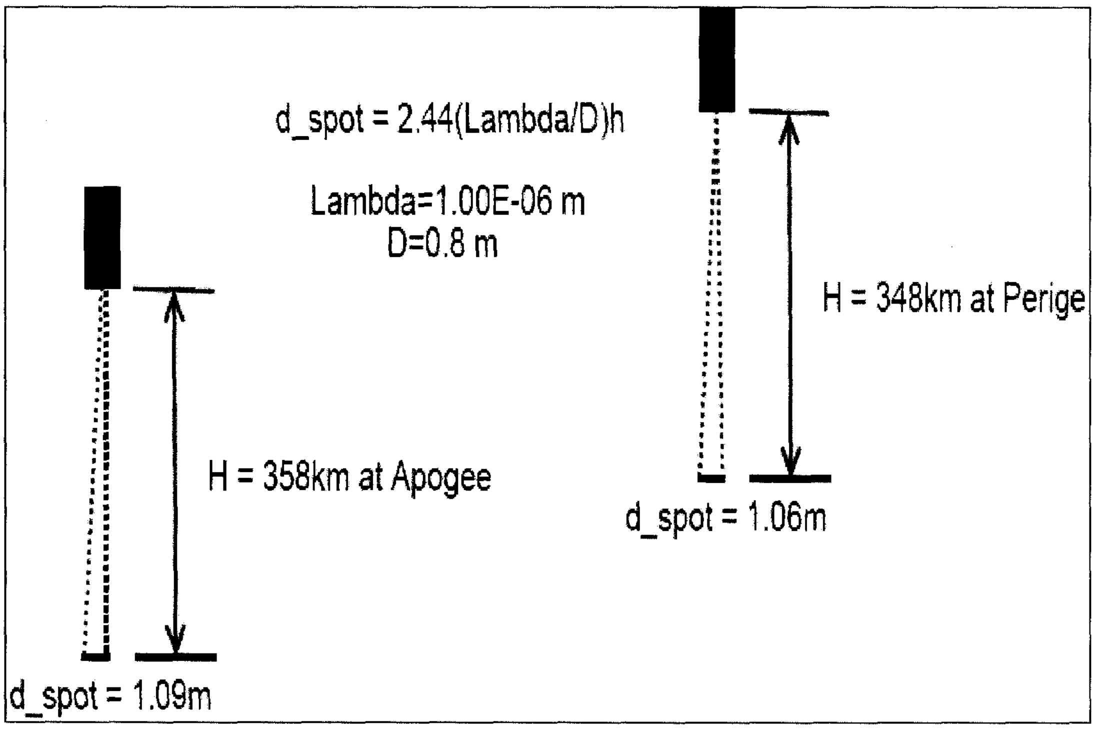

Figure 4.3: Laser Beam Ground Spot

The d_spot was calculated for the International Space Station orbit at apogee and perigee. As a safety factor the d_spot be multiplied by 10 meters on the ground.

4.4 Demonstration on LEO and Operational on GEO

The International Space Station (ISS) is the perfect test bed for this technology, a small scaled SBSP demonstration using 2400W. The launch cost for lower Earth orbit (LEO) is cheaper then GEO on the SpaceX Falcon 9. A payload of 100kg can be delivered to the ISS for a couple of million dollars in launch costs. A demonstration in LEO is workable for a small-scale experiment however for a full-scale SBSP mission, GEO would be better for operational.

4.5 Thin-Films Solar Cells

Thin-film PV panels are strongest where traditional crystalline silicon PV panels are weakest and are cheapest. The crystalline silicon fabrication process of forming a rod of pure silicon and sawing it into wafers is inherently expensive. Thin-films are usually deposited, not sawed, and they use a fraction of the material used in crystalline panels. In recent years, the energy market has begun to understand their potential, and growth rate of thin-films made out of Cadmium Telluride (CdTe) and Cadmium Indium Gallium Diselenide (CIGS) has exploded (Knight, 2008). The next chapter discusses the systems databases.

Chapter 5: Systems Database

The systems database was written as an input output Excel program. The objective of the database is to create a systems design tool for a specific space based solar power system. The N2 Diagram is used as a cover sheet for the design tool, Figure 5.1.

Solar Power
↓

Instrument Bus	Manages power to Laser System	Manages power and comm. To ATP System/Manages power to Saftey and Control System		
(m, T, W)	Laser System	Tracks Safety Signal from Ground Station, alerts Safety and Control if link is broken (λ)	Beams Watts to Ground Station (W, ε)	
	Points the Laser to the Ground Station	ATP System/Safety and Control System	Tracks the Ground Station and points the laser towards it/ Laser curtain to protect against object potentially flying through laser	
	Provides the enable for the Laser to fire if nothing is impeding the safety laser curtain	Comm. To assist with the pointing and tracking/ Signal to assure Laser path to Ground Station is umimpeded	Ground Station	→ Watts

Figure 5.1: N2 Diagram

The N2 diagram demonstrates four important sub-systems: Instrument Bus, Laser System, ATP/Safety and Control System, and Ground Station. These systems work together from top to bottom and vice versa; the diagram should be read clockwise. Also included in the diagram are each sub-systems constraint variables. The database that follows the N2 Diagram is the Instrument Bus calculation (see Table 5.1).

Table 5.1: Instrument Bus Electromagnetic Spectrum

Electromagnetic Spectrum	**input**	**Units**	**output**	**units**
Payload Temperature (Absolute) = T	612.8680292	**K**	339.7180	**C**
Wavelength = λ	1.07E-06	**M**		
Frequency	1.00E+05	**Hz**		
		Equations		
Spectral Irradiance = E_λ	[(2πhc^2)/(λ^5)]*[1/(e^(ch/kTλ)-1)]	9-2 Pg 256	83281.20	**W/m^2/μ**
Total radiant emittance = W_b	σT^4	9-3 Pg 256	8000.00	**W/m^2**
Wavelength of peak emittance = λ_max	2,898/T	9-4 Pg 257	4.7286	**μm**

The electromagnetic spectrum of the payload temperature (in absolute value) can be calculated from equation 12

$$W_b = \sigma T^4 \left[\frac{W}{m^2}\right] \rightarrow T = \left(\frac{W_b}{\sigma}\right)^4 = \frac{8000}{5.67051\times10^{-8}} \tag{12}$$

Where: Stefan-Boltzmann's Law is $\sigma = 5.67051\times10^{-8}\left[W\cdot m^{-2}\cdot K^{-4}\right]$

Equation 13 is the spectral energy distribution of a blackbody is given by Planck's Law:

$$E_\lambda = \frac{2\pi hc^2}{\lambda^5}\frac{1}{e^{ch/kT\lambda}-1} \tag{13}$$

Where: λ is the wavelength

h is the Planck's constant ($6.626075\text{x}10^{-34}$)W*s^2

T is the Absolute temperature K

c is the speed of light (2.99 x 10^8) m/s

k is the Boltzmann's constant (1.380658 x 10^{-23})W*s/K

Equation 14 solves for the wavelength of peak emittance.

$$\lambda_{max} = \frac{2,989}{T} \tag{14}$$

For this specific SBSP demonstration on the International Space Station, the laser payload temperature profile is 612.87 °K or 339.72 °C and its wavelength is 1.07E-06. The spectral irradiance outputted 83,281.20W/m^2/μ. The radiant emittance is 8,000W/m^2 and the wavelength of peak emittance 4.728587 μm (Bate, Mueller, & White, 1971).

A code was written to design the laser optic system. The systems database accepts inputs for the variables from equations 15-16. The three options for the inputs are: (1) the variables can be inputted directly, (2) the variables can be calculated, and (3) the default variables are from Southampton Photonic Laser data (see Table 5.2).

Table 5.2: Laser System

Constants		Input	output	unit
Earth Radius			**6371000**	**m**
Orbit Apogee Altitude			**405696000**	**m**
Parking Orbit Altitude			**100000**	**m**
P_laser = (1400/R_sl,au^2)*ε_las*π*R_array^2				
Power-beam Laser	**P_laser**		**800**	**W**
Solar array radius	**R_array**		**2.42521818**	**m**
Efficiency of sunlight conversion to a collimated EM radiation beam	**ε_las**		**0.075**	
Separation between the Sun and the solar-pumped laser power station	**R_sl,au**		**1**	**Au**
2.44*λ_laser/D_las-tran = D_receiver/D_las-rec,max				
Laser wavelength	**λ_laser**		**1.07E-06**	**m**
Diameter of the Laser-transmitting optics	**D_las-tran**		**0.75**	**m**
Separation between the laser power station and the receiver	**D_las-rec, max**	358000	**358000**	**m**
Sail receiver	**D_receiver**		**1.246221867**	**m**

The size of the power system can be estimated using equation 15

$$P_{laser} = \frac{1400}{R_{sl,au}^2} \varepsilon_{las} \pi R_{array}^2 \,(W) \tag{15}$$

This equation, R_{array} is the radius of a disc-shaped solar array, in meters. The efficiency of the power system converting sunlight to laser power is ε_{las} and the separation between the sun and laser power station is $R_{sl,au}$, in Astronomical Units. The 1,400Wm^2 is the solar irradiance on an object 1AU from the Sun. The size of the laser optics can calculated with the relationship in equation 16.

$$\frac{2.44\lambda_{laser}}{D_{las-tran}} = \frac{2R_{sail}}{D_{las-ship,\max}} \tag{16}$$

The laser wavelength is given by λ_{laser} and the diameter of the laser transmitting optics, $D_{las-tran}$, both in meters. The maximum separation between laser power station and the payload is $D_{las-ship,max}$, in meters. The radius of the solar sail is given by R_{sail}, in meters (Bate, Mueller, & White, 1971).

The Acquisition, Tracking, and Pointing & Safety System assume an ISS payload and ellipse orbit altitude. The value used for Earth radius is 6378.14 km and the Earth gravitational constant is 398,600.5 km^3/sec^2, see Table 5.3.

Table 5.3: Classic Orbit Elements

Classic Orbit Elements	input	output elements	output	units
Semi-major axis = a	(rA+rP)/2	""	6731.14	**km**
Inclination = i	51.6428	""	51.6428	**deg**
Eccentricity = e	(rA/a)-1	""	0.000742816	
Perigee altitude = H	348.00	Radius Perigee = rP	6726.14	**km**
Apogee altitude = Ha	358.00	radius Apogee = rA	6736.14	**km**

Equations 17-21 are the classic orbit elements:

$$a = \frac{-\mu}{2\varepsilon} = \frac{(r_A - r_P)}{2} \tag{17}$$

Where: a the Semi-major axis is describes the size of the ellipse

$$e = |e| = 1 - \left(\frac{r_P}{a}\right) = \left(\frac{r_A}{a}\right) - 1 \tag{18}$$

Where: e the Eccentricity is describes the shape of the ellipse

$$i = \cos^{-1}\left(\frac{h_Z}{h}\right) \tag{19}$$

Where: i the Inclination of the angle between the angular momentum vector and the unit vector in the Z-direction

$$r_P = a(1-e) \tag{20}$$

Where: r_P is the radius of perigee

$$r_A = a(1+e) \tag{21}$$

Where: r_A is the radius of apogee

The semi-major axis is 6731.14 km, inclination is 51.64 deg, eccentricity is 0.000742816, perigee altitude is 6726.14 km, and apogee altitude is 6736.14 km.

The basic orbit dynamics, the orbit period, orbit revolutions per earth day, orbit energy, average orbit angular velocity, average ground velocity, and satellite velocity at perigee and apogee are calculated in Table 5.4 (Wertz, Space Mission Geometry, 2004).

Table 5.4: Basic Orbit Dynamics

Basic Dynamics	Input		output	
Orbit period = P	2*pi(a^3/mu)^(1/2)	*Table 6-2, Pg 137	91.59940099	**min**
Orbit revolutions per Earth day	(# of min per day)/orbit period		15.72062682	**revs/day**
Orbit energy =ε	[-] mu/2a	*Eqn 6-4, Pg 134	-29.60869184	**km^2/sec^2**
Average orbit angular velocity = n	(mu/a^3)^(1/2)	*Eqn 6-13, Pg 139	0.001143236	**rad/sec**
Average ground velocity = Vg	2*pi*R_E/P	*Eqn 5-32, Pg 116	7.291720816	**km/sec**
Satellite velocity (at perigee) = V_P	(ε+mu/rP)*2)^(1/2)	*Eqn 6-4, Pg 134	7.70	**km/sec**
Satellite velocity (at apogee) = V_A	(ε+mu/rA)*2)^(1/2)	*Eqn 6-4, Pg 134	7.68956915	**km/sec**

The Basic Orbit Dynamics are shown in equations 22-27

$$P = 2\pi\left(\frac{a^3}{\mu}\right)^{1/2} \cong 84.489\left(\frac{a}{R_E}\right)^{3/2} \text{min} \cong 0.00016587a^{3/2} \text{ min}, a \text{ in km} \tag{22}$$

Where: P is the Orbit period

$$\varepsilon = \frac{V^2}{2} - \frac{\mu}{r} = -\frac{\mu}{(2a)} \tag{23}$$

Where: ε is the total specific mechanical energy or mechanical energy per unit mass for the system and is the sum of the kinetic energy per unit mass and potential energy per unit mass. (Orbit energy)

$$n \equiv \left(\frac{\mu}{a^3}\right)^{1/2} \cong 36{,}173.585a^{-3/2}\left[\frac{\deg}{\sec}\right] = 8{,}681{,}660.4a^{-3/2}\left[\frac{rev}{day}\right]$$
$$n \cong 3.1252977\times10^9 a^{-3/2}\left[\frac{\deg}{day}\right] \tag{24}$$

Where: n is the mean motion or average angular velocity

$$V_g = 2\pi\frac{R_E}{P} \le 7.905\left[\frac{km}{\sec}\right] \tag{25}$$

Where: V_g is the ground tracking velocity

Equation 26 is the Velocity at perigee

$$\varepsilon = \frac{V^2}{2} - \frac{\mu}{r} \rightarrow V_P = \left(2\cdot\frac{\varepsilon\mu}{r_P}\right)^{1/2} \tag{26}$$

Equation 27 is the Velocity at apogee

$$V_A = \left(2\cdot\frac{\varepsilon\mu}{r_A}\right)^{1/2} \tag{27}$$

In summary, the orbit period is found to be 91.599 min, orbit revolution per Earth day is 15.72 revs/day, orbit energy is – 29.60 km^2/sec^2, average orbit angular velocity is 0.001143 rad/sec, average ground velocity is 7.2917 km/sec. Satellite velocity (at perigee) is 7.70 km/sec, and satellite velocity (at apogee) is 7.6895 km/sec.

The minimum elevation angle and maximum nadir is calculated in Table 5.5.

Table 5.5: Orbit Geometry

Orbit Geometry	input		output	units	ouput	units
Minimum elevation angle = angle ε	* minimum angles for IR payload	*Eqn 5-26 pg. 113	0.35	**rad**	20.00	**deg**
Maximum nadir angle = η	asin[cos ε)(sin ρ)]	*Eqn 5-26 pg. 113	1.10	**rad**	63.0086	**deg**

The Orbit Geometry, where ε is the Grazing angle or spacecraft elevation angle measured at the target between the spacecraft and the local horizontal.The Minimum angles for IR payload are 20 degrees which is 0.35 radians. Where η is the Nadir angle measured at the spacecraft from the sub-satellite point to the target and $\sin\eta = \cos\varepsilon \sin\rho$. The general coverage characteristics at perigee and the general coverage characteristics at apogee is shown Table 5.6 and Table 5.7.

Table 5.6: General Coverage Characteristics

General Coverage Characteristics (at Perigee)	input		output	Units	ouput	units
Earth angular radius = ρ	asin(R_E/R_E+H)	*Eqn 5-16 Pg 111	1.247714772	**Rad**	71.489	**deg**
Maximum Earth central angle = λ0	90-ρ	*Eqn 5-17 Pg 111	0.323081555	**Rad**	18.511	**deg**
Range to horizon = D_max	R_E*tan(λ0)	*Eqn 5-18 Pg 111	2135.483421	**Km**		
Swath width for overlapping equatorial coverage = S	[-](sin^-1)(sin24.8)(sin55)	Eqn 9-30 Pg 293	0.315095461	**Rad**	18.054	**deg**
Swath width = λ	2*(90-λ-η)	Eqn 5-27 Pg 113	0.24404708	**Rad**	13.98	**deg**
Swath width			1556.566444	**Km**		
Maximum field of view	2 times maximum nadir angle		2.199413873	**Rad**	126.02	**deg**
Slant range to edge of swath = D	R_E(sinλ/sinη)	Eqn 5-28 Pg 113	871.2552651	**Km**		
where K_A =	2.56E+08	**km^2**				
Instantaneous access area = IAA	K_A(1-cosλ)	Eqn 7-6 Pg 167	1900580.479	**km^2**		
Area access rate = AAR	(2*K_A*sinλ)/P	Eqn 7-10 Pg 169	1.13E+04	**km^2/sec**		

Table 5.7: General Coverage Characteristics

General Coverage Characteristics (at Apogee)	input		output	Units	ouput	units
Earth angular radius = ρ	asin(R_E/R_E+H)	*Eqn 5-16 Pg 111	1.243309826	**Rad**	71.236	**deg**
Maximum nadir angle = η	asin[cos ε)(sin ρ)]	*Eqn 5-26 pg. 113	1.10	**Rad**	62.842	**deg**
Maximum Earth central angle = λ0	90-ρ	*Eqn 5-17 Pg 111	0.327486501	**rad**	18.764	**deg**
Range to horizon = D_max	R_E*tan(λ0)	*Eqn 5-18 Pg 111	2166.774617	**km**		
Swath width for overlapping equatorial coverage	[-](sin^-1)(sin24.8)(sin55)	Eqn 9-30 Pg 293	0.315095461	**rad**	18.054	**deg**
Swath width = λ	2*(90-λ-η)	Eqn 5-27 Pg 113	0.249859766	**rad**	14.32	**deg**
Swath width			1593.640568	**km**		
Maximum field of view	2 times maximum nadir angle		2.193601187	**rad**	125.68	**deg**
Slant range to edge of swath = D	R_E(sinλ/sinη)	Eqn 5-28 Pg 113	893.2259393	**km**		
Instantaneous access area = IAA	K_A(1-cosλ)	Eqn 7-6 Pg 167	1992075.103	**km^2**		
Area access rate = AAR	(2*K_A*sinλ)/P	Eqn 7-10 Pg 169	1.16E+04	**km^2/sec**		

Where ρ is assumed a spherical Earth, the line from the spacecraft to the Earth's horizon is perpendicular to the Earth's radius, and therefore Earth angular radius in equation 28 (Boden, 2004).

$$\sin\rho = \cos\lambda_0 = \frac{R_E}{R_E + H} \tag{28}$$

Where R_E is the radius of the Earth and H is the altitude of the satellite in equation 29.

$$\rho + \lambda_0 = 90\deg \tag{29}$$

Where: λ_0 is the Maximum Earth central angle

D_{max} is the distance to the horizon. (Range to horizon)

$$D_{max} = \left[(R_E + H)^2 - R_E^2\right]^{1/2} = R_E \tan\lambda_0 \tag{30}$$

S is the Swath width. The perpendicular separation between the grounds tracks. The perpendicular separation between the orbits at the equator is 20.1 deg. Because the swath width is 24.2 deg, we now have some overlap margin even at the equator and substantial margin at higher latitudes, which are the primary areas of interest (ee equation 31).

$$S = \sin^{-1}(\sin 24.8\deg \sin 55\deg) 20.1\deg \tag{31}$$

The Maximum field of view is two times maximum nadir angle

$$D = R_E\left(\frac{\sin\lambda}{\sin\eta}\right) \tag{32}$$

Where D is the Slant range to edge of swath

$$IAA = K_A(1-\cos\lambda) \tag{33}$$

Where IAA = the instantaneous access area

$K_A = 2.55604187 \times 10^8$ for area in km^2

$$AAR = \frac{(2K_A \sin\lambda)}{P} \tag{34}$$

Where AAR = the area access rate as the satellite sweeps over the group for the access area. Lastly, the Gravitational Perturbations are calculated in Table 5.8.

Table 5.8: Gravitational Perturbations

Gravitational Perturbations	Output	Input	units
Node precession rate - J2	Eq. 6-19 Pg 143	-4.976329661	**deg/day**
Node precession rate - Moon	Eq. 6-14 Pg. 142	-0.000150003	**deg/day**
Node precession rate - Sun	Eq. 6-15 Pg. 142	-6.07906E-05	**deg/day**
Total node precession rate		-4.976540454	**deg/day**
Node spacing		-23.27910782	**deg/rev**
Perigee rotation rate - J2	Eqn 6-20 Pg 143	3.818528725	**deg/day**
Perigee rotation rate - Moon	Eqn 6-16 Pg 143	9.94918E-05	**deg/day**
Perigee rotation rate - Sun	Eqn 6-17 Pg 143	4.53306E-05	**deg/day**
Total perigee rotation rate		3.818673548	**deg/day**

The rate of change due to J_2

$$\dot{\Omega} = -1.5nJ_2\left(\frac{R_E}{a}\right)^2(\cos i)\left(1-e^2\right)^{-2} \cong -2.06474\times10^{14}a^{-7/2}(\cos i)\left(1-e^2\right)^{-2}\left[\frac{\deg}{day}\right] \tag{35}$$

Where: n is mean motion in deg/day

R_E is Earth's equatorial radius

a is semi-major axis in km

e is eccentricity

i is inclination

Right ascension of the ascending node for the Moon and Sun:

$$\dot{\Omega}_{moon} = \frac{-0.00338(\cos i)}{n} \tag{36}$$

$$\dot{\Omega}_{sun} = \frac{-0.00154(\cos i)}{n} \tag{37}$$

The rate of change due to J_2

$$\begin{aligned}\dot{\omega} &= 0.75nJ_2\left(\frac{R_E}{a}\right)^2\left(4-5\sin^2 i\right)\left(1-e^2\right)^{-2}\\ \dot{\omega} &\cong 1.03237\times10^{14}a^{-7/2}\left(4-5\sin^2 i\right)\left(1-e^2\right)^{-2}\left[\frac{\deg}{day}\right]\end{aligned} \tag{38}$$

Argument of perigee for the Moon and Sun:

$$\dot{\omega}_{moon} = \frac{0.00169\left(4-5\sin^2 i\right)}{n} \tag{39}$$

$$\dot{\omega}_{sun} = \frac{0.00077\left(4-5\sin^2 i\right)}{n} \tag{40}$$

The ground station calculation can be seen in Table 5.9; the laser beam ground spot at perigee is 1.06 m and apogee is 1.09 m(Wertz, Orbit and Constellation Design, 2004).

Table 5.9: Ground Station

D =	0.8	M	H_perigee =	348000.00
λ =	1.00E-06	M	H_apogee =	358000.00

	input	output	units
Perigee d_spot	2.44(λ/D)h	1.06E+00	m
Apogee d_spot	2.44(λ/D)h	1.09E+00	m

Equation 41 is the Laser Beam Ground Spot

$$d_{spot} = 2\left(\frac{\lambda}{D}\right)H \tag{41}$$

Table 5.10 shows the cost vs. the space-rated cost for the major sub-systems of the SBSP demonstration. The total estimated cost is $12 million dollars for the demonstration (Wertz, Orbit and Constellation Design, 2004), (Chesley, Luts, & Brodsky, 2004).

Table 5.10: System Cost Analysis

Sub-systems		Company	Cost ($)	Space-Rated ($)
	International Space Station	NASA (FRAM)/JAXA (JEM)	0.00	0.00
	Launch Vehicle	SpaceX (Dragon Lab)	4,000,000.00	4,000,000.00
	Laser System	Southamton Photonic (YB)	120,000.00	1,200,000.00
	Acquisition Tracking Pointing System	Tesat (Laser Comm.)	1,000,000.00	1,000,000.00
	Instrument Bus	Saab (Spacecraft Manangement Unit)	3,000,000.00	2,000,000.00
	Safety and Control	Mirrors and Laser Optics	500,000.00	1,000,000.00
	Ground Station	solar cells, maintenance, and TTC (Telemetry, Tracking, and Communication)	1,000,000.00	1,000,000.00
Total			9,620,000.00	**10,200,000.00**
	Estimated Cost			12,000,000.00

The specification establishes the performance, design and development requirements of the Space Based Solar Power Demonstration. The instrument bus houses and maintains the instrument on the International Space Station. The main components of this system are: Power Management System, Command and Control, and Safety. The laser system is responsible for transferring the energy from SBSP plant to ground station in the form of amplified radiation. This system is to be an active closed loop control system, working together with the ATP. The controlling system will work with inputs of the ATP system. The major components of this system are as follows: laser, lasing material, pump source, optical cavity, laser curtain, pointing, and cooling system. The ATP system is responsible for acquiring and tracking the position of the target ground station and uses that information to point the SBSP system laser towards the target ground station. The system also needs to be an active, closed looped system, requiring information from the target ground station to help point the laser and assure that nothing is impeding the target ground station. The major components of this system are as follows: tracking, pointing, safety and control, and isolation and stabilization system. The safety and control system is responsible for assuring that nothing can impede the SBSP demonstration's path to the target ground station while the laser is operating. The major components of the system are as follows: laser curtain, ground-based modular laser, turn on and shut off system (Grady, 2008).

Table 5.11: Characteristics Specification

Acquisition, Tracking and Pointing (ATP) System (*assuming ISS payload not a free flyer)

Physical Characteristics	*Approximate values*
Mirror Aperture	75 cm
Mirror/Laser and on orbit System Mass	100 kg
Vibrations from the International Space Station	0.01-50 Hz *
Orbit Altitude	340.5 km *
Orbit Velocity	27,700 km/hr *
Orbit Inclination	51.64° (deg) *
Maximum Slant Range	500 nautical miles
Performance Characteristics	*Requirements*
Pointing Accuracy	10e-7 radian precision

Laser System

Physical Characteristics	*Approximate values*
Mirror Aperture	75 cm
Mirror/Laser and on orbit System Mass	100 Kg
Performance Characteristics	*Requirements*
Laser efficiency	at least 30%
Laser Power	800 Watts

Safety and Control System

Performance Characteristics	*Requirements*
Laser Curtain Spot Diameter	30 m
Laser Curtain Area	706.858 m^2

Instrument Bus *assuming a free flyer and not an ISS payload

Physical Characteristics	*Approximate values*
Solar Panel Area	10 m^2
Performance Characteristics	*Requirements*
Power*	4 kWatts*

The systems database was benchmarked with Werts & Larson, 2004. All the equations came directly from Space Mission Analysis and Design, Third Edition

Chapter 6: Conclusions

Photovoltaic cells convert solar energy directly into power for electricity grids. Solar radiation is directly proportional to the current source of a solar cell connected in parallel. Solar cells also consist of diodes which represent the p-n junction of a cell. There are over a dozen types of solar cell materials; this thesis focuses on thin film. Thin film solar cells are light weight and require fewer materials to produce; hence they are cheaper in cost per solar cell. Due to the high cost of space access and the massive quality of solar cells needed to collect solar power on orbits, thin-film solar cells are ideal for SBSP.

SBSP has been discussed and studied for over thirty years by people in related industries. The thesis describes a small-scaled space-based solar power demonstration using laser technology to beam down power from orbit. Laser beaming allows for practical sizing of components. Lower earth orbit (LEO) and the International Space Station are the perfect test beds for this technology to be demonstrated. However, for a full-scale space-based solar power system, geosynchronous orbit (GEO) is ideal. GEO period is equal to the Earth's rotational period with an orbital eccentricity of approximately zero.

Launch Vehicles (LV) are the main constraining variables of Space-Based Solar Power. Lowering cost to space access and support infrastructure is the first step to enable a viable system of SBSP. Systems engineering was performed to identify the SBSP system requirements and constraints. The N^2 Diagram identifies interactions between major factors of the system. The Functional Flow Block Diagrams indicate the sequential relationship of all functions that must be accomplished by a system. A complete set of project requirements includes the functional needs requirements, performance requirements, and interface requirements. The Product Breakdown Structures decomposed and allocated those requirements down to design elements. The components' physical characteristics and their approximated values are stated in Table 4.

The tradeoff study on launch vehicles cost is in Appendices I and II which compare the cost for all existing Expendable LV and Reusable LV. Reusable Launch Vehicles are concluded to be cheaper than Expendable launch vehicles. When comparing the Russian Proton and the Chinese Long Mach expendable launch vehicles to the United States, SpaceX Falcon 9 reusable launch vehicle, Falcon 9 launch cost is by far the cheapest at $8,200 per kg.

The demonstration in figure 13 shows the concept of Space-Based Solar Power. The components necessary for this demonstration are: Pointing Control Device – Gimbal or Spacecraft attitude control, Laser and Mirror Aperture, and Acquisition-Tracking-Pointing System. The demonstration experiment is to collect 2500W from the International Space Station, allocate 100W to support systems, and 2400W for laser beaming system. A prediction of 50% loss to the laser system, another 50% is estimated to be lost due to the atmosphere, and finally, a 50% loss to the Photovoltaic ground station. The laser power demonstration should beam 300W on the ground.

The Excel program should be used as a database of systems engineering for basic orbit dynamics calculation for the Space-Based Solar Power demonstration. The cost analysis estimated a total cost of the demonstration at $12 million dollars. Appendix III shows some structure analysis calculations. Appendix IV has the entire component specification document in details.

Additional future work that can be done related to this thesis is structure analysis for the Space-Based Solar Power systems design. This project also needs funding for demonstration.

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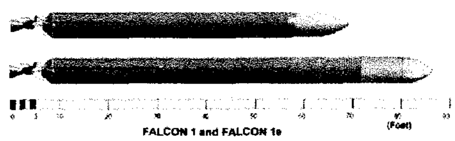
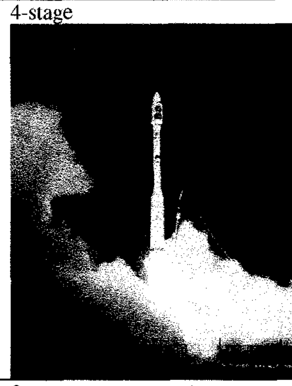
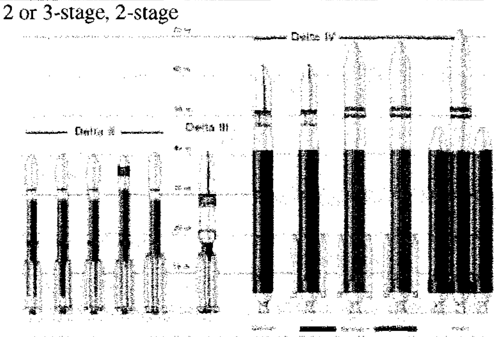
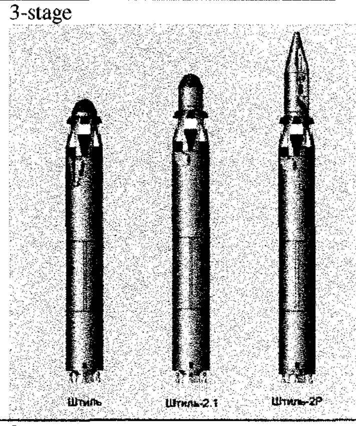
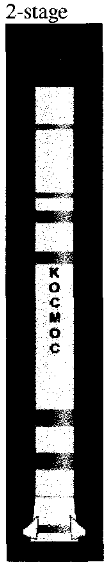
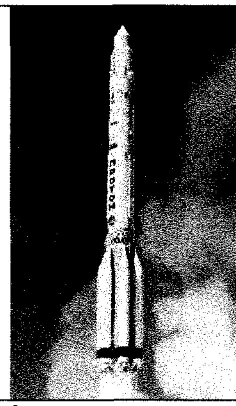
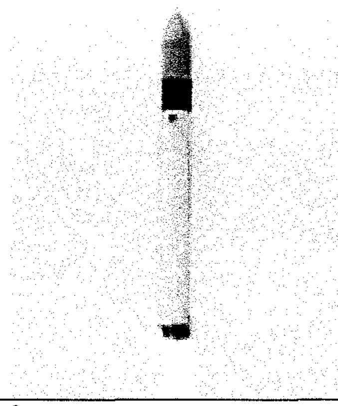
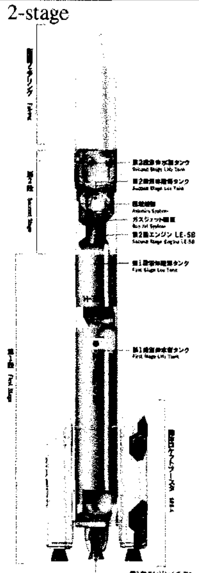
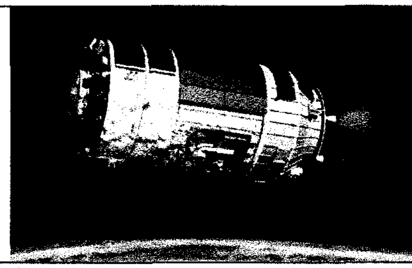
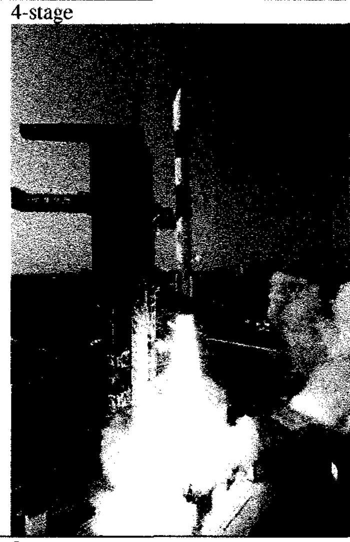
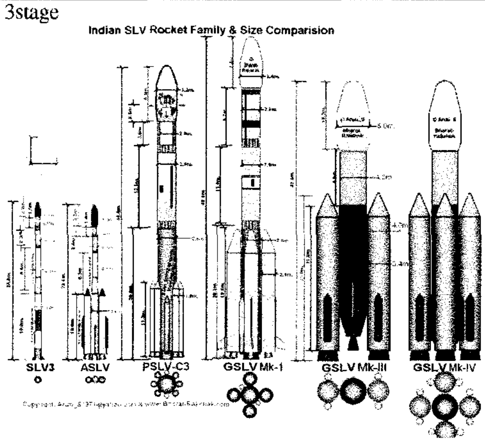
Appendices

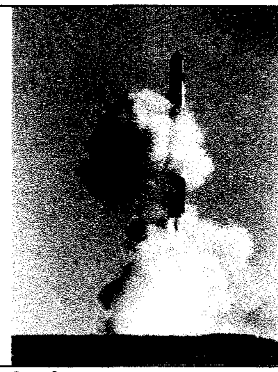
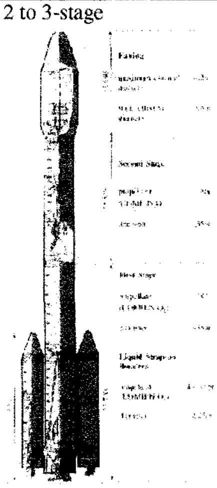
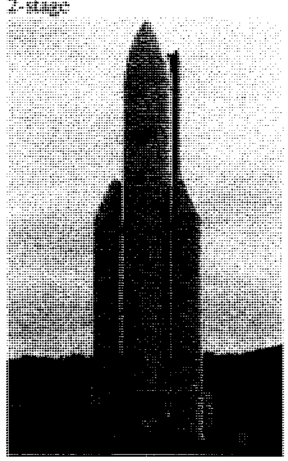
Appendix I: Expandable/Disposable, 2-stage, 3+stage

Country	Vehicle Name	Launch Type	Orbit	~ Cost
USA Space Exploration (SpaceX)	Falcon 1 Falcon 1e	Reusable, 2-stage Reusable, 2-stage 	LEO LEO	$8M / 400kg $9M / 1000kg
Orbital Sciences Corporation	**Taurus**	4-stage 	LEO	1,350 kg
Lockheed Martin-Boeing joint venture United Launch Alliance	**Atlas V**	2-stage 	LEO/GTO	10,300-20,050 kg 4,100-8,200 kg
United Launch Alliance	**Delta II & IV**	2 or 3-stage, 2-stage 	LEO/GTO	Cost per launch (1987) $36.7m 2,700 - 6,100 kg 900 - 2,170 kg 8,600 - 25,800 kg 3,900 - 10,843 kg
Orbital	**Minotaur**	4-stage	LEO/SSO	580 kg 331 kg

Sciences Corporat ion				
Russia State Rocket Center Makayev	**Volna/Shtil'**	3-stage 	LEO	$100K / 10kg
Russian Space Agency	**Cosmos-3M** (Космос-3M)	2-stage 	LEO/SSO	1,500 kg 775 kg
	Proton	3-stage (w/optional 4-stage)	LEO/GT	22,000 kg

Commercial and Russian government launches			O	6,000 kg
Russian Space Agency	**Rockot** (Ро́кот)	3-stage 	LEO	\$13M-\$15M/2000kg
Japan Made by Mitsubishi Heavy Industries (MHI) for JAXA	**H-IIA**	2-stage 	LEO/GTO	\$10K / 1kg – LEO 10,000 - 15,000kg 4,100 - 6,000kg
	HTV	H-II Transfer Vehicle to ISS	LEO	\$100M /

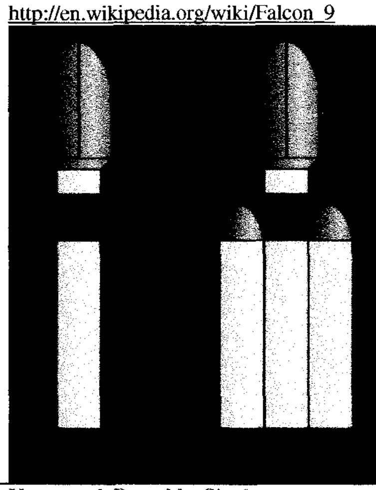
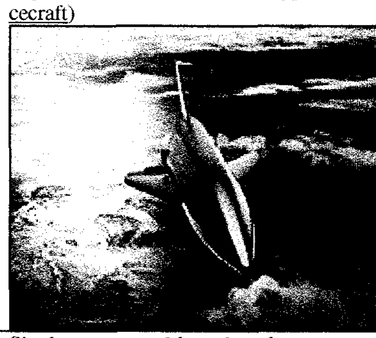
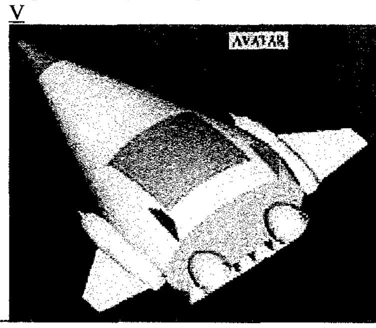
JAXA				6000kg
India Indian Space Research Organiza tion	**PSLV**	4-stage 	GEO ? LEO SSO GTO	$1.8M / 100kg 3,250 kg 1600 kg 1000 kg
Indian Space Research Organiza tion	**GSLV**	3stage  	LEO GTO	5,000 kg 2,500 kg $2M / 100kg
Ukraine	**Dnepr-1**	3-stage (4 or 5 with SpaceTug upper stages)	LEO	4,500 kg

Russian Space Agency			ISS TLI	3,200 kg 550 kg $10K / 1kg
China	**Long March**	2 to 3-stage 	LEO/GTO	$60 million cost; 5,200 kg to GTO, cost per kg is $11,500
Europe	**Ariane 5**	2-stage 	LEO/GEO	16,000 kg 6,800 kg
Israel	**Shavit**	3-stage	LEO	

Reference: http://en.wikipedia.org/wiki/Launch_vehicle

Appendix II: Reusable Launch System

Country	Vehicle Name	Launch Type	Orbit	~ Cost
USA Space Exploration (SpaceX)	Falcon 9 (Normal) Falcon 9 (Heavy) **2009**	Reusable, 2-stage (Manned) http://en.wikipedia.org/wiki/Falcon_9 	Low Earth Orbit (LEO) Geostationary Transfer Orbit (GTO)	**Falcon 9** – starts @ $35 million; able to boost 9,900 kg to LEO, 4,900 kg to GTO, minimum GTO cost per kg is $10,500 **Falcon 9 Heavy** – starts @ $78 million cost; able to boost 27,500 kg to LEO, 12,000 kg to GTO, minimum GTO cost per kg $8,200
EUROPE European Space Agency (**ESA**) and European Aeronautic Defence and Space Company EADS N.V. **(EADS)**	Hopper **2015**	Unmanned, Reusable, Signle-stage http://en.wikipedia.org/wiki/Hopper_(spacecraft) 	Low Earth Orbit (LEO) GEO (Geostationary Earth Orbit)	$15,000/kg
INDIA Defence Research and Development Organisation (DRDO) and Indian Space Research Organization (ISRO)	**AVATAR** (**A**erobic **V**ehicle for hypersonic **A**erospace **T**r**A**nspo**R**tation) **2015**	Single-stage reusable rocket planes http://en.wikipedia.org/wiki/Avatar_RLV 	Low Earth Orbit (LEO)	$67/kg / 500 kg to 1000 kg estimated vehicle life of 100 launches

Appendix III: Miscellaneous Excel Database

Structural Analysis

<u>*Material Properties*</u>: 7075 aluminum is chosen

Young's Modulus $E = 7.1\times10^{9}\ {}^{N}\!/_{m^2}$

Poisson's Ratio $\nu = 0.33$

Density $\rho = 2.8\times10^{3}\ {}^{kg}\!/_{m^3}$

Ultimate Tensile Strength $F_{tu} = 524\times10^{6}\ {}^{N}\!/_{m^2}$

Yield Tensile Strength $F_{ty} = 448\times10^{6}\ {}^{N}\!/_{m^2}$

Cylinder area moment of inertia, I

$$10 = 0.560\sqrt{\frac{EI}{m_B L^3}} = 0.56\sqrt{\frac{\left(71\times10^{9}\right)I}{(2{,}000)(10)^3}} \rightarrow I = 8.982\times10^{5}cm^4$$

The required thickness, $t = \frac{I}{\pi R^3}$

Type of Load	Weight (N)	Distance (m)	Load Factor	Limit Load
Axial	19,614	-	6.5	127,5000 (N)
Bending Moment	19,614	5	3.0	294,200 (N-m)

Equivalent axial load, $P_{eq} = P_{axial} + \frac{2M}{R}$

Ultimate Load = Limit load × Ultimate Factory of Safety

Sizing for Tensile Strength

Axial stress, $\sigma = \frac{P}{A}, A = 2\pi Rt$, solve for required thickness

Sizing for Stability (Compressive Strength)

Size the cylinder for stability, $\varphi = \frac{1}{16}\sqrt{\frac{R}{t}}$, $\gamma = 1.0 - 0.901\left(1.0 - e^{-\varphi}\right)$

Cylinder buckling stress, $\sigma_{cr} = 0.6\gamma\frac{Et}{R}$. Note that if σ_{cr} were greater than the material's proportional limit, we would use additional methods for inelastic buckling.

Critical buckling load, $P_{cr} = A\sigma_{cr}$

Margin of safety (MS) $MS = \frac{\text{Allowable Load or Stress}}{\text{Design Load or Stress}} - 1.0$ or

$$MS = \frac{\mathrm{P_{cr}} = \sigma_{cr}\times area}{\text{Design Load or Stress}} - 1.0$$

Calculating the Mass: Mass of the cylinder, $m = \rho 2\pi RtL$

Appendix IV: Components Specification Document

Space Based Solar Power Demo
Instrument Bus Specifications Overview

1.0 Scope

1.1 This specification established the performance, design and development requirements of the Space Based Solar Power (SBSP) Demonstration Instrument Bus.

2.0 Requirements

2.1 System Description

2.1.1 General Description The instrument bus houses and maintains the instrument in orbit or on the International Space Station.
The main components of this system are:

2.1.1.1 Power Management System This system manages the power to the LASER System, ATP System, and Safety and Controls System.

2.1.1.1.1 Solar Panels* If the Instrument is housed on a free flyer, the power for the instrument will need to be generated on-orbit using an array of solar panels

2.1.1.1.2 Battery Backup* If the instrument is housed on a free flyer, the bus will need to provide backup power for when the instrument's orbit is in eclipse.

2.1.1.2 Command and Control This system will manage the interaction between the different systems on board the SBSP demonstration.

2.1.1.2.1 Safety The Command and Control System will have a Safety system to prevent and electronic tampering with the SBSP demonstration.

2.1.1.3 Communications* If the instrument is housed on board a free flyer, the system will need to maintain its own operational communications with the ground.

2.2 Characteristics

2.2.1 Performance Characteristics		Requirements
2.2.1.1 Quantity		
2.2.1.1.1 Power*	4 kW*	

2.2.2 Physical Characteristics	Approximate values
2.2.2.1 Solar Panel Area*	10 m^2*

(*assuming a free flyer and not an ISS payload)

Space Based Solar Power Demo
LASER System Specifications Overview

3.0 Scope

1.1 This specification established the performance, design and development requirements of the LASER System for Space Based Solar Power (SBSP) Demonstrations.

4.0 Requirements

2.1 System Description

2.1.1 General Description The LASER system is responsible for transferring the energy from Space Based Solar Power Plant to the Ground station in the form of amplified radiation.
This system is to be an active closed loop control system, working together with the ATP. The controlling system will work with inputs of the ATP system.
The major components of this system are as follows:

2.1.1.1 LASER the LASER will convert the electrical output from the solar panel or the batteries into amplified radiations for transfer to Earth station.
The major components of LASER are as follows

2.1.1.1.1 Lasing material (crystal, gas, semiconductor, dye, etc...)

2.1.1.1.2 Pump source (adds energy to the lasing material, e.g. flash lamp, electrical current to cause electron collisions, radiation from a laser, etc.)

2.1.1.1.3 Optical cavity consisting of reflectors to act as the feedback mechanism for light amplification

2.1.1.1.3.1 LASER Curtain A LASER curtain is a low power beam, with a wider diameter, which would alert the ground station to send a signal shutting down the LASER if an object were to impede the LASER's path to the ground station

2.1.1.2 Pointing this is the optical system (consisting of lenses) which would be responsible for limiting the LASER to the desired spot on the ground station.

2.1.1.3 Cooling System LASER operation produces lot of heat energy; cooling system would regulate the temperature of the LASER.

2.2 Characteristics

2.2.1 Performance Characteristics	Requirements
2.2.1.1 Quality	
2.2.1.1.1 LASER efficiency	at least 30%
2.2.1.2 Quantity	
2.2.1.2.1 LASER Power	800W

2.2.2 Physical Characteristics	Approximate values
2.2.2.1 Mirror Aperture	75 cm
2.2.2.2 Mirror/Laser and on orbit System Mass	100 kg

Space Based Solar Power Demo
Acquisition, Tracking, and Pointing System Specifications Overview

5.0 Scope

1.1 This specification established the performance, design and development requirements of the Space Based Solar Power (SBSP) Demonstration Acquisition, Tracking and Pointing (ATP) System.

6.0 Requirements

2.1 System Description

2.1.1 General Description The ATP system is responsible for acquiring and tracking the position of the target ground station and uses that information to point the SBSP system laser towards the target ground station.

The system is to be an active, closed-looped system, requiring information from the target ground station to help point the laser and assure that nothing is impeding the target ground station.

The major components of this system are as follows:

2.1.1.1 Tracking The tracking mechanism will lock onto a signal sent from the target ground station and follow it as the SBSP Demo travels through its orbit. The period of acquisition will vary from a few minutes to about 10 to 15 minutes depending on pass elevation, altitude, etc.

2.1.1.2 Pointing The pointing mechanism will ensure that the SBSP system laser is positioned to accurately direct energy at the target ground receiver.

2.1.1.3 Safety and Control A signal sent from the target ground station will immediately shutdown the laser if any object impedes the safety laser curtain from the target ground station.

2.1.1.4 Isolation and Stabilization System The SBSP laser system must be isolated from outside vibrations in order to insure high accuracy pointing.

2.2 Characteristics

2.2.1 Performance Characteristics	Requirements
2.2.1.1 Quality	
2.2.1.1.1 Pointing Accuracy	at least 1×10^{-7} *radian* precision
2.2.2 Physical Characteristics	Approximate values
2.2.2.1 Mirror Aperture	75 cm
2.2.2.2 Mirror/Laser and on orbit System Mass	100 kg
2.2.2.3 Vibrations from the International Space Station	0.01-50 Hz*
2.2.2.4 Orbit Altitude	340.5 km *
2.2.2.5 Orbit Velocity	27,700 km/hr*
2.2.2.6 Orbit Inclination	51.64°*
2.2.2.7 Maximum Slant Range	500 nautical miles

(*assuming an ISS payload and not a free flyer)

Space Based Solar Power Demo
Safety and Control System Specifications Overview

7.0 Scope

1.1 This specification established the performance, design and development requirements of the Space Based Solar Power (SBSP) Demonstration Instrument Bus.

8.0 Requirements

2.1 System Description

2.1.1 General Description The Safety and Control System is responsible for assuring that nothing can impede the SBSP Demonstration's path to the target ground station while the LASER is operating

The system is to be an active, closed-looped system, requiring information from the target ground station to operate.

The major components of the system are as follows:

2.1.1.1 LASER Curtain The LASER curtain is a low power, wider spread beam with the purpose of providing an early warning sign in the event an object inadvertently passes through the path between the LASER and the Ground Station.

2.1.1.2 Ground-Based Modulatable LASER If the Ground Station receives the unobstructed signal from the LASER curtain, a modulatable LASER will send a signal to the instrument to activate the LASER system.

2.1.1.3 Turn on/Shutoff System This system will activate the SBSP LASER System if the SBSP instrument receives the signal from the modulatable LASER from the Ground Station confirming an unobstructed path between the LASER and the Ground Station receiver. At any time the laser loses contact with the modulatable LASER, this system will deactivate the LASER system.

2.2 Characteristics

2.2.1 Performance Characteristics	Requirements
2.2.1.1 Quantity	
2.2.1.1.1 Laser Curtain Spot Diameter	30 m
2.2.1.1.2 Laser Curtain Area	706.858 m^2