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DETAILED DESIGN OF A SPACE BASED SOLAR POWER SYSTEM

A Thesis

Presented to

The Faculty of the Department of Mechanical and Aerospace Engineering

San Jose State University

In Partial Fulfillment

of the Requirements for the Degree

Masters of Science

by

Sean Joseph Mobilia

May 2009

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DETAILED DESIGN OF A SPACE BASED SOLAR POWER SYSTEM

by
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ABSTRACT

DETAILED DESIGN OF A SPACE BASED SOLAR POWER SYSTEM

by Sean Joseph Mobilia

In 1968, Dr. Peter Glaser (Ledbetter, 2008) first developed the idea of Space Based Solar Power (SBSP), which involved collecting solar power in orbit and then transmitting that power to the surface. However, to this day, despite a lot of discussion about SBSP, there have been no SBSP watts transmitted to the Earth's surface. The International SBSP Initiative (ISI) demonstration is being developed by students at San Jose State University along with members of the NASA Space Portal to beam SBSP watts to the Earth's surface. The ISI demonstration is being developed based on previously designed technologies and concepts to beam power to the surface using a laser. This thesis discusses the detailed design of the ISI, examining the four main system components: the laser system, the instrument bus, the acquisition, tracking and pointing/safety system, and the ground station receiver. It also discusses the design tools that were developed and utilized for this design. From these design tools, the ISI is shown to need a 9.82 m diameter ground receiver for a laser beam generated with 75 cm diameter optics. The thesis also includes preliminary analysis of the ISI system, showing the critical design points for a potential ISI system.

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CHAPTER 1

INTRODUCTION

With the increasing cost and decreasing supply of conventional energy coupled with the concern over pollution and climate change, the need arises for clean, renewable alternative forms of energy. Presently, systems that make use of wind, geothermal, and terrestrial are being developed for implantation for large scale use. However, each of these systems is limited by geography as to where they can be implemented. This makes scaling those systems upwards to meet the needs of a growing world population difficult. For instance, the best places for locating terrestrial solar power are deserts, which are not typically the most populated places on the globe.

There is an alternative form of energy that could be scaled upward to meet the demands of Earth's growing population. Space-Based Solar Power (SBSP) involves the collection of solar power from satellites in orbit around Earth and utilizing wireless energy transfer to beam that energy down to Earth. This concept was first developed in 1968 by Dr. Peter Glaser (Ledbetter, 2008), who then went on to patent the concept. His initial concept involved transferring a large amount of power through a square kilometer microwave dish.

SBSP has been examined many times through the last couple of decades. NASA and the Department of Energy, with the aid of Boeing, collaborated on a study of SBSP in the late 1970's, however the technological and social conditions of the time made the

system infeasible. More recently, NASA later took a fresh look at SBSP in the late 1990's, and in 2001, The Aerospace Corporation conducted its own study into SBSP, applying modern technology to concepts of SBSP (Penn & Law, 2001). However, while there has been a lot of discussion about SBSP, still there has not been a single SBSP watt beamed down from Earth's orbit. A demonstration of SBSP would be vitally important to furthering its development as an alternative energy source. A demonstration will push the development of the components necessary to make SBSP and will provide a source of intellectual property from that development. Space-qualified power transmitters and pointing systems will need to be provided for a demonstration to run, and safety systems will be needed to assure that the solar power can beamed from orbit without causing any harm to other spacecraft, air traffic, or people. A SBSP demonstration will also provide more accurate measurements for atmospheric losses and system efficiencies. In this way, a demonstration can pave the way for future system development.

It was for these reasons that the International SBSP Initiative (ISI) is being developed. The ISI utilizes student and international support in developing a low cost SBSP demonstration. The ISI flight demonstration is being developed as either a payload deliverable to the International Space Station (ISS) or for use on a free flyer spacecraft in low Earth orbit (LEO).

This report discusses the Detailed Design of a Space-Based Solar Power System, specifically the ISI flight demonstrations. The report begins by explaining the scientific theory necessary for understanding the ISI design. The report then details the ISI

background and initial concept. It then gives a description of the Optics/Acquisition, Tracking and Pointing (ATP) Design Tool that was created to design to the system. The report later details the mechanical design of the ISI, describing the subsystems as well as displaying the model of the design. Finally, this report discusses the thermal model of the ISI system and provides some analysis of the system.

CHAPTER 2

SYSTEM DESIGN THEORY

This chapter details the scientific background necessary to understand the design of the ISI demonstration. This chapter begins discussing the background necessary to perform thermal analysis on a spacecraft, then goes into the theory behind the preliminary design of the laser optics, the background needed to determine a spacecraft slant range, and ends detailing the background needed for the pointing of optical systems.

2.1 Spacecraft Thermal Analysis

In order for a spacecraft to function properly on orbit, care must be taken to assure that the components making up these craft maintain their specified temperature limits. The flow of energy through heat transfer and the on orbit thermal environment will be important in verifying that these system components maintain their operational requirements.

2.1.1 Heat Transfer

The modes of heat transfer that take place as energy flows through a system are conduction, convection, and radiation. However, convection, which is the process of heat transfer enabled by a moving fluid (Lienhard & Lienhard, 2008), is not a significant portion of the SBSP system since its components are operating in a vacuum.

2.1.1.1 Conduction

Conduction involves the transfer of kinetic energy between particles. Steady unidirectional conduction was described by Fourier as

$$Q = -KA \frac{dT}{dx} \quad (2.1)$$

where Q is the rate of heat conduction [W] along path x, K is the thermal conductivity

$[\frac{W}{mK}]$ of the material, A is the cross sectional area $[m^2]$ along path x, and $\frac{dT}{dx}$ is the

temperature gradient $[\frac{K}{m}]$ along path x. This can also be expressed as

$$Q = \frac{K}{X} A(T_1 - T_2) \quad (2.2)$$

from Agrawal (1986), where Q is the heat conduction between points 1 and 2. This can be reduced to

$$q = \frac{Q}{A} = \frac{K}{X} (T_1 - T_2) \quad (2.3)$$

where q $[\frac{W}{m^2}]$ is the heat load acting on the system. In Agrawal (1986), it is shown that

Fourier's laws of conduction are similar to Ohm's law for electric currents. This

comparison gives the thermal resistance $[\frac{K}{W}]$ of a component, which can be written as

$$R_c = \frac{X}{KA} \quad (2.4)$$

Substituting Equation (2.4) into Equation (2.2), gives

$$Q = \frac{T_1 - T_2}{R_c} \quad (2.5)$$

In spacecraft systems, most heat transfer occurs as conduction through the solid components of the spacecraft, and is then radiated through the interior or out through the exterior into space.

2.1.1.2 Radiation

Radiation is mode of heat transfer carried out by the transmission of electromagnetic waves. This requires no heat transfer medium, and can occur across a vacuum (Agrawal, 1986). The rate that energy is radiated from a blackbody, a body that absorbs all radiation it comes into contact with, is described by Lienhard and Lienhard (2008) as

$$Q_{rad} = A_1 F_{1-2} \sigma (T_1^4 - T_2^4) \quad (2.6)$$

where Q_{rad} is the heat [W] radiated, A_1 is the area [m^2] of the object emitting the radiation, F_{1-2} is the shape factor between objects 1 and 2, σ is the Stefan-Boltzmann constant [$5.67 \times 10^{-8} \frac{W}{m^2 K^4}$], and T is the temperature [K] of black body 1 and 2. There are three things that can occur when radiation comes in contact with an object. The radiation can be absorbed, it can be reflected, or it can be transmitted. The processes are related by

$$\alpha + \rho + \tau = 1 \quad (2.7)$$

where α is the absorptivity, ρ is the reflectivity, and τ is the transmissivity. A material that absorbs almost all incoming radiation, a blackbody, is considered opaque and has a transmissivity of zero.

2.1.2 Spacecraft Thermal Environment

There are three main types of radiation which affect a craft in Earth orbit. These are direct solar flux, albedo or light reflected off Earth or the moon, and Earth infrared radiation (Clawson, et al., 2002). Direct solar flux is the heating caused by direct sunlight, and is the source of most spacecraft heating. Agrawal (1986) shows that solar flux is given by

$$Q_{solar} = S\mu_i A \quad (2.8)$$

where Q_{solar} [W] is the incident solar intensity, S [$\frac{W}{m^2}$] is the solar flux, A [m^2] is the total area of the surface, and μ_i is the ratio of the project surface over the total area. The ratio of the projected surface to the total surface is also equal to $\cos \theta$ for a plane surface where θ is the angle between the sun vector and the plane normal. This can be reduced to

$$q_{solar} = S\mu_i \quad (2.9)$$

where q_{solar} [$\frac{W}{m^2}$] is the solar heat load on the system. The solar flux varies throughout the course of the year, with a minimum value of 1322 W/m^2 and a maximum value of 1414 W/m^2 (Gilmore, Hardt, Prager, Grob, & Ousley, 2004).

Sunlight that is reflected off a celestial body such as the Earth is known as albedo. The amount of light reflected varies significantly depending on the surface conditions.

The heat load generated by albedo, $q_{\text{albedo}} \left[\frac{W}{m^2} \right]$, is described by

$$q_{\text{albedo}} = \alpha I_{\text{solar}} \rho_{\text{albedo}} F_{\text{albedo}} \quad (2.10)$$

where α is the component's absorptivity of the albedo, $I_{\text{solar}} \left[\frac{W}{m^2} \right]$ is the intensity of the solar fluxes, ρ_{albedo} is the Earth's albedo, and F_{albedo} is the geometric shape factor (Gilmore, Hardt, Prager, Grob, & Ousley, 2004). The albedo reaching the spacecraft varies over a course of an orbit, decreasing after it moves beyond the point where the sun and the Earth are at their zenith (Clawson, et al., 2002).

A portion of the sunlight not reflected by the Earth as albedo is absorbed and re-emitted as infrared radiation. The amount radiated can vary considerably for individual points in orbit, depending on the orbit inclination and the conditions on the Earth's surface. The Earth infrared heat loads, $q_{\text{EarthIR}} \left[\frac{W}{m^2} \right]$, is given by Gilmore (2004) as

$$q_{\text{EarthIR}} = \varepsilon I_{\text{EIR}} F_{\text{EIR}} \quad (2.11)$$

where ε is the emissivity of the component material in the infrared, $I_{\text{EIR}} \left[\frac{W}{m^2} \right]$ is the intensity of the Earth IR, and F_{EIR} is the geometrical shape factor for the component with respect to Earth. The IR radiation emitted by Earth is approximately the same wavelength as that of the spacecraft, making it difficult to reflect away. This can lead to

significant radiation affecting spacecraft with low-altitude orbits (Gilmore, Hardt, Prager, Grob, & Ousley, 2004).

2.2 Laser Optics

The optics for a space-based laser can be sized with Rayleigh's Criterion. While sizing optics for laser assisted solar sails, Taylor, Anding, Halford and Matloff (2003) give Rayleigh's Criterion as

$$\theta = \frac{D_{receiver}}{SEP_{las-receiver,max}} = \frac{2.44\lambda}{D_{optics}} \quad (2.12)$$

where $D_{receiver}$ is the diameter [m] of the receiver spot, $SEP_{las-receiver,max}$ is the maximum separation [m] between the laser power transmitter and the receiver, D_{optics} is the diameter [m] of the laser transmitter optics, and λ is the wavelength [m] of the transmitting laser. θ is the angle subtended by the receiver at a given separation. SBSP systems where the maximum separation does not occur directly overhead of the receiving ground station need to factor in the spacecraft elevation angle into the equation, shown as

$$\theta = \frac{D_{receiver} \sin(\epsilon)}{SEP_{las-receiver,max}} = \frac{2.44\lambda_{laser}}{D_{optics}} \quad (2.13)$$

where ϵ is the spacecraft elevation angle [$^{\circ}$] (Chesley, Lutz, & Brodsky, 2004). This has the effect of increasing the spot size or transmitter optics, depending on what is being design for.

2.3 Slant Range

Figure 2.1: Angular relationships between SBSP, GS, and Earth's center (Wertz, 2004).

The slant range of a spacecraft is the line-of-sight distance between the spacecraft and its ground receiver or target. In Figure 2.1, the relationships between the spacecraft, the target ground station and the center of the Earth are shown (Wertz, 2004). If the altitude of the spacecraft and either nadir angle, η , the spacecraft elevation angle, ϵ , or the Earth central angle, λ , the slant range, D , can then be determined (Wertz, 2004). The first step is calculating the angular radius of the Earth, ρ , is

$$\sin \rho = \cos \lambda_0 = \frac{R_E}{R_E + H} \quad (2.14)$$

Then, if Earth's central angle is known, the nadir angle can be calculated by the following equation.

$$\tan \eta = \frac{\sin \rho \sin \lambda}{1 - \sin \rho \cos \lambda} \quad (2.15)$$

If the nadir angle is known, the spacecraft elevation angle can then be calculated.

$$\cos \varepsilon = \frac{\sin \eta}{\sin \rho} \quad (2.16)$$

And if the spacecraft elevation angle is known, the nadir angle can then be calculated from Equation (2.17).

$$\sin \eta = \cos \varepsilon \sin \rho \quad (2.17)$$

With these two angles, the remaining angle can be determined from the following.

$$\eta + \lambda + \varepsilon = 90 \text{ deg} \quad (2.18)$$

The spacecraft slant range is then determined from Equation (2.19).

$$D = R_E \left(\frac{\sin \lambda}{\sin \eta} \right) \quad (2.19)$$

2.4 Optical Pointing

In order for a SBSP demonstration to be operated, it is important that the transmitter be properly aimed at the power receiver. One of the major sources of error for pointing systems is called boresight error, which arises from stress or noise acting on the system. Other major errors are reference frame errors, the inability to compensate for receiver movement, and error resulting from atmospheric turbulence. The pointing

system for the ISI will have to take into account the motion of the satellite relative to the ground. The point ahead angle, which takes this motion into account, is approximated by

$$\theta_p \approx \frac{2V}{c} \quad (2.20)$$

where θ_p is the point-ahead angle, and V is the velocity perpendicular to the line of sight of the craft. The speed of the light in a vacuum is given by c (Andrews and Phillips, 1998).

CHAPTER 3

ISI DEMONSTRATION BACKGROUND

This chapter discusses the ISI demonstration background. The chapter begins by discussion the primary and secondary objectives of the ISI. The chapter then goes on to discuss the preliminary ISI architecture and preliminary concept sizing. Finally, this chapter looks at how previous systems have shaped the current ISI concept.

3.1 ISI Demonstration Mission Objectives

The primary objective of the ISI demonstration is to beam SBSP watts down from orbit. The secondary objectives for the ISI demonstration include the global sourcing of parts, international participation, university involvement, system modularity, ISS compatibility, and to build off of existing technologies (Grady, 2008). The aim of these objectives is to reduce the cost of the final demonstration. The goal of system modularity could also increase the usefulness of the ISI demonstration. Instead of just one demonstration, the ISI could fulfill many potential design concepts. Multiple types of power transmitters could be tested. With designed system modularity, all that would be required would be to switch one transmitter with another and make some minor alternations.

3.2 Demonstration Architecture

The ISI demonstration is being developed to transmit power from lower Earth orbit through laser power transmission. It is designed to operate either as a payload on the ISS or on its own as a free flyer. The concept was sized to beam 200 W of electrical

energy to an Earth-based ground station. The ground station solar panels were estimated to have a 50% efficiency converting laser IR to electricity (Hoffert & Hoffert, 2008). The energy loss through the atmosphere was conservatively estimated to be 50%, and the laser efficiency was estimated to be 30%. This would require a 800 W laser powered by approximately 2.6 kW of electricity. The laser and the tracking system was estimated to weigh 50 kg, and require a mirror aperture of 75 cm (Grady, 2008).

The primary architecture breaks the ISI demonstration into the following systems: the Laser System, the ATP/Safety System, the Instrument Bus, and the Ground Station Receiver.

3.3 Other Influences on ISI Design

The ISI demonstration is not being designed from completely from scratch. One of the objectives of the demonstration is to make use of existing, developed technology while building and designing the demo. The design of the ISI demonstration intends to makes use of technological studies going back more than twenty years. This demonstration concept builds off of Strategic Defense Initiative (SDI) technology, studies done by The Aerospace Corporation, and laser communications technology.

Figure 3.1: A comparison of the ISI demonstration (Grady, 2008) to SDI concepts (Space Based Lasers [SBL], 2008).

3.3.1 SDI Technology

The ISI demonstration concept builds off of the work done on the SDI projects. Figure 3.1 shows the similarities between SDI (Space Based Lasers [SBL], 2008) and the SBSP demonstration (Grady, 2008). The red arrows point from the SBSP Demonstration to corresponding SDI components. Figure 3.2 shows another SDI concept, Talon Gold (Picture provided by Jim Grady, 2008). The Cassegrainian type optical systems that were used in these SDI systems are being built upon and used for the ISI demonstration.

Figure 3.2: Talon Gold (Picture Provided by Jim Grady, 2008).

The ISI system also has a second, smaller scope which is the tracking device for the demonstration. This is similar to the two SDI concepts shown.

3.3.2 The Aerospace Corporation Study on SBSP

The ISI demonstration also builds off work done by the Aerospace Corporation. As part of their report on space solar power (Penn & Law, 2001), The Aerospace Corporation examined Ytterbium fiber lasers as a possible transmitter that could be highly efficient, potentially obtaining conversion efficiencies close to 50%, but also scale up for a high power SBSP system. Figure 3.3 shows the theoretical efficiency of the Ytterbium fiber laser concept examined in the Aerospace Corporation Report.

Figure 3.3: Ytterbium Fiber Laser (Penn & Law, 2001).

The ISI demonstration makes use of fiber lasers for their system power transmitter, building off the Aerospace Corporation Report. For more information on the laser being used in the ISI demonstration design, see Chapter 5.

3.3.3 TESAT Laser Communications Technology

The ISI demonstration builds off laser communication technology, especially with regards to pointing. Both laser communications and the SBSP Demonstration require a high amount of pointing precision in order to operate at peak parameters. The pointing system of the demonstration was based on the design of the pointing system for TESAT's Laser Communication Terminal (LCT). TESAT is a German company that specializes in optical systems. In February of 2008, TESAT demonstrated its LCT terminals in an experiment sending 5.5 Gigabits per second across distances of 2,000 to 8,000 km (de Selding, 2008). Their experiment is shown in Figure 3.4

Figure 3.4: TESAT's Laser Communication Terminal (de Selding, 2008).

For the precision pointing, TESAT's LCT uses an array of fast-steering mirrors, shown in Figure 3.5 (Langenbach & Schmid, 2005).

Figure 3.5: The LCT Pointing System (Langenbach & Schmid, 2005).

The TESAT LCT Coarse Pointing Assembly (CPA) provides hemispherical coverage while the Fine Pointing Assembly (FPA) receives the incoming laser signal. The Point Ahead Assembly (PAA) adjusts the target for the outgoing signal with information it receives from the incoming signal. The point ahead assembly adjusts the outgoing beam so that it sends the signal to where the target will be, which is described in Equation (2.20). The TeLescope Assembly (TLA) bundles the incoming signal while expanding the outgoing signal (Barho & Schmid, 2003). Laser communication pointing systems also operate with feedback from their target system. The necessary pointing accuracy can be reduced if the system has some communication between transmitter and the target.

CHAPTER 4

OPTICS/ATP DESIGN TOOL

This chapter describes the design tool that was created to design the laser optics and the acquisition, tracking and pointing (ATP) system for an SBSP system as part of this project. This tool works by running through a system of equations based on the information input. The design tool was designed in Microsoft Excel. The design tool is divided into five sections and each of these sections is made up of five columns: the parameter description, the input (if applicable), the output, the parameter's units, and the parameter notes. The parameter notes column is used to show any assumptions used when calculating a parameter, any reference the parameter uses, and the code used to calculate each parameter. The five sections of the design tool are Constants, Slant Range Calculator, Preliminary Optics Sizer, Point Ahead Angle Calculator, and Safety Buffer Calculator. These five sections will be described in the following chapter.

4.1 Constants

The Constants section of the design tool is the section that displays the parameters that are constrained by the current system, such as the orbital parameters of the SBSP system and the information about the ground station location. There is one exception to this, the maximum slant range of the system, which is normally calculated by the Slant Range Calculator using the spacecraft elevation angle. However, this can be overwritten if a system is constrained by a particular slant range as opposed to the elevation angle. Each of the parameters in Constants is designed to allow for a user input, as they depend

on the system of the user, except for the speed of light, the Ground Station Radius, which is Earth's altitude plus the Ground Station Altitude, and the Orbital Circumference, which is calculated as the SBSP system's orbital velocity multiplied by its orbital period.

The defaults of the design tool were set for a SBSP system operating from the ISS. The ISS has an apogee of 361,000 m (Peat, 2009) and an orbital velocity of 7,706.6 m/s and an orbital period of 5,480.4 sec (International Space Station, 2009). Since the ISS has a small eccentricity, the angular velocity of the system orbit is approximated by assuming a circular orbit. For the default system, the maximum slant range is calculated from the Slant Range calculator. The ground station is assumed to be at sea level and the Earth's radius is assumed to be its equitorial radius. The values for the Earth's radius and its angular velocity are 6,378,136 m and 0.004178075 rad/s (Larson & Wertz, 2004).

	A	B	C	D	E
1					
2		input	output	unit	Parameter Notes
3	Constants				
4	SBSP Max Orbit Altitude		361000	m	(Peat, C. 3-19-2009), =IF(B4<>"",B4,361000)
5	SBSP Orbit Velocity		7706.6	m/sec	(ISS Wikipedia 3-19-2009), =IF(B5<>"",B5,7706.6)
6	Orbital Period		5480.4	sec	(ISS Wikipedia 3-19-2009), =IF(B6<>"",B6,5480.4)
7	Orbit Circumference		42235250.64	m	=IF(B7<>"",B7,C5*C6)
8	Orbit Angular Velocity		0.005688636	deg/sec	Since ISS orbit is almost circular, the orbital velocity is approximated by assuming the orbit is circular, V/Circum. = omega/360 deg. =IF(B8<>"",B8,C5*360/C7)
9	Max Slant Range		926000	m	Calculated from Slant Range Calculator. Default parameters determined from SBSP Demo System Architecture, =IF(B9<>"",B9,IF(B10<>"",B10*1852,'Slant Range Calculator'!C26))
10	Max Slant Range in nautical miles		500	na. mi.	1 na. mi. = 1852 m; =IF(B10<>"",B10,C9/1852)
11	Ground Station Altitude		0	m	Assumed Sea Level, =IF(B11<>"",B11,0)
12	Earth Radius		6378136.49	m	Earth Radius Assumed to be Earth Equitorial Radius, (Larson, W. J. and Wertz, J. R., 2004), =IF(B12<>"",B12,6378136.49)
13	Ground Station Radius		6378136.49	m	Effective Earth Radius is Earth Radius + Ground Station Altitude, =C12+Constants!C11 (Larson, W. J. and Wertz, J. R., 2004), =IF(B14<>"",B14,IF(B15<>"",2,7.2921158553*10^8*5*180/PI)))
14	Earth Angular Velocity		0.004178075	deg/sec	=IF(B15<>"",B15,C14*C12*2*PI/(360))
15	Earth Linear Velocity		465.1011023	m/sec	=IF(B16<>"",B16,C14*C13*2*PI/(360))
16	Ground Station Linear Velocity		465.1011023	m/sec	
17	Speed of Light = c		299792458	m/sec	(Larson, W. J. and Wertz, J. R., 2004), =299792458

Figure 4.1: A screenshot of the Constants section of the Optics/ATP Design Tool.

4.2 Slant Range Calculator

The Slant Range Calculator section of the design tool calculates the slant range of an SBSP system with a given altitude and spacecraft elevation angle. Using the input spacecraft altitude, the Slant Range Calculator uses Equation (2.14) to calculate the angular radius of the Earth. With the angular radius of the Earth, the program then calculates the nadir angle between the SBSP system and the ground station using Equation (2.17). Then the program uses Equation (2.18) to calculate Earth central angle from the nadir angle and the spacecraft elevation angle. Finally, the program uses these angles to calculate the slant range of the SBSP system from Equation (2.19).

The default spacecraft elevation angle was calculated by using the law of cosines to calculate the Earth central angle and the nadir angle for a system operating on the ISS with a maximum slant range of 500 nmi (926,000 m). Those angles were the input to Equation (2.18) to give the default spacecraft elevation angle of 19.165°.

	A	B	C	D	E
16	Slant Range Calculation	Input	Output	Units	Parameter Notes
17	epsilon		0.334492374	radians	Assuming Epsilon is limiting the slant range, so Epsilon_0=Epsilon =IF(B18<>"",PI()*B18/180,C13)
18			19.16500131	degrees	=IF(B18<>"",B18,C14)
19	pho		1.242001877	radians	Eq. 5-24, pg113; (Larson, W. J. and Wertz, J. R., 2004); =ASIN((Constants!C13)/(Constants!C12+Constants!C4))
20			71.16146572	degrees	=C19*180/PI()
21	eta		1.106145892	radians	Eq. 5-26b, pg 113; (Larson, W. J. and Wertz, J. R., 2004); =ASIN(COS(C17)*SIN(C19))
22			63.37749113	degrees	=180*C21/PI()
23	lambda		0.130158061	radians	Eq. 5-27, pg. 113; (Larson, W. J. and Wertz, J. R., 2004); =PI()/2-C21-C17
24			7.45750756	degrees	=90-C22-C14
25					
26	Slant Range			m	Eq 5-28, pg 113; (Larson, W. J. and Wertz, J. R., 2004) =Constants!C12*(SIN(C23)/SIN(C21))

Figure 4.2: A screenshot of the Slant Range Calculator section of the Optics/ATP Design Tool.

4.3 Preliminary Optics Sizer

The Preliminary Optics Sizer is the section of the design tool that calculates the size of the laser system optics or the corresponding ground spot size from a given set of optics. With the laser wavelength and either the diameter of the laser optics or the diameter of the receiving ground spot, the design tool will use Equation (2.12) to calculate the unconstrained variable. The distance for the maximum separation between the receiver and the laser is the maximum slant range, taken from either the Constants section or the Slant Range Calculator. The ground spot that comes from this, however, is perpendicular to the beam. If the SBSP system is not directly overhead of the ground station receivers at maximum separation, the design tool uses the spacecraft elevation angle along with Equation (2.13) to calculate the effect the angle has on the unconstrained variable.

The default for the laser transmission wavelength was taken from a Southampton Photonics Inc. fiber laser (SPI Lasers, 2008). The default for the diameter of the laser optics was based on the preliminary ISI system architecture (Grady, 2008) and the ISS JEM payload constraint (Chang, 2007).

	A	B	C	D	E
1	Preliminary Optics Sizing	input	output	unit	Parameter Notes
2	$2.44 \cdot \lambda_{\text{laser}} / D_{\text{las-tran}} = D_{\text{receiver}} / \text{SEP}_{\text{las-receiver, max}}$				(Taylor, T., Anding, R. C., Halford, D. and Matloff, G.L., 2003)
3	Laser wavelength, λ_{laser}		0.00000107	m	SPI Datasheet, =IF(B3<>"",B3,1070*10^-9)
4	Diameter of the Laser-transmitting optics, $D_{\text{las-tran}}$			m	SBSP System Architecture (Grady, J.R., 2008), =IF(B4<>"",B4,IF(AND(B4="","",B6<>""),2.44*C3^C5/C6,0.75))
5	Separation between the laser power station and receiver, $\text{SEP}_{\text{las-receiver, max}}$		926000	m	=Constants!C9
6	Receiver Diameter, D_{receiver}			m	=IF(B6<>"",B6,2.44*C3^C5/C4)
7					
8					To account for the angle of the beam, $D_{\text{receiver}} = D_{\text{receiver_actual}} \cdot \sin(e)$, (Chesley, Lutz, & Brodsky, 2004)
9	$D_{\text{las-tran_actual}}$			m	=IF(B4<>"",B4,IF(AND(B4="","",B6<>""),2.44*C3^C5/C6/SIN('Slant Range Calculator'!C17),0.75))
10	$D_{\text{receiver_actual}}$			m	=IF(B6<>"",B6,2.44*C3^C5/C4/SIN('Slant Range Calculator'!C17))

Figure 4.3 : A screenshot of the Preliminary Optics Sizer section of the Optics/ATP Design Tool.

4.4 Point Ahead Angle Calculator

The Point Ahead Calculator is the section of the design tool that calculated the point ahead angle for a ground station receiver and SBSP system. The design tool uses Equation (2.20) to estimate the necessary point ahead angles for the optical transmission.

The default angles are for an SBSP system operating on the ISS.

	A	B	C	D	E
1	Point Ahead Angle Calculator		output	units	Parameter Notes
2	Ground Station Point Ahead Angel				
3	$\theta_{\text{pa_G}} \approx 2 \cdot V/c$		5.14129E-05	radians	(Andrews, L.C., and Phillips, R.L., 1998), =2*Constants!C5/Constants!C17
4			0.0029571	degrees	=C3*180/PI()
5					
6	Spacecraft Point Ahead Angel				
7	$\theta_{\text{pa_S}} \approx 2 \cdot V/c$		3.10282E-06	radians	(Andrews, L.C., and Phillips, R.L., 1998), =2*Constants!C16/Constants!C17
8			0.0001773	degrees	=C7*180/PI()

Figure 4.4: A screenshot of the Point Ahead Angle Calculator section of the Optics/ATP Design Tool.

4.5 Safety Buffer Calculator

The last section of the design tool is the Safety Buffer Calculator. The Safety Buffer Calculator first calculates the amount of time needed for an optical signal to travel through the entire SBSP system. The design tool then calculates the relative angular velocity between the SBSP system orbit and its receiver station. The relative movement between the receiver and the SBSP system while the optical signal travels is then determined. The design tool then approximates the angular movement of the SBSP system from the ground station. Since the amount of time for the optical signal to move the SBSP system is small, this angle has been approximated by assuming the slant range does not change while the optical signal travels. With this assumption, the angle was calculated with the law of cosines. Then, with an input intercepting object, the design tool then estimates the transverse velocity of the beam at the altitude of the input object as the SBSP system moves through its orbit. Assuming the object is moving directly towards the beam along the path of the beam, the design tool then calculates how much the beam and the intercepting object move as the optical signal travels through the system. This buffer distance is then added to the receiver radius to give the total receiver size.

	A	B	C	D	E
1	Safety Buffer Calculator				
2			output	units	Parameter Notes
3	Signal Travel Time				
4	$t=x/c$		0.006177607	sec	$=2*Constants!C9/Constants!C17$
5					
6	Orbit Optics Safety Sizing (Receiver)	input	output	unit	
7	Ground Station Angular Velocity		0.004178075	deg/sec	(Larson, W. J. and Wertz, J. R., 2004), $=Constants!C14$
8	Relative Angular Velocity		0.061510561	deg/sec	$=ABS(C7-Constants!C8)$
9	Arc of SBSP system orbit		0.000379988	deg	$=C8*C4$
10	Distance SBSP system travels		44.69424333	m	Small arc, assuming distance to be linear, $=\{Constants!C12+Constants!C4\}^2*PI()/C9/360$
	Angle beam travels		0.002765434	deg	Assuming slant range change is insignificant, small distance traveled, angle found through law of cosines, $=ACOS((2*Constants!C9^2-C10^2)/2/Constants!C9^2)*180/PI()$
11					
12	Intercepting Object Altitude		15240	m	Default Object at 50,000 ft, $=IF(B12<>"",B12,15240)$
13	Intercepting Object Velocity		268.2235554	m/sec	Commercial Jet Cruising Speed, 600 mph, $=IF(B13<>"",B13,268.223555392634)$
14	Distance Laser Beam Travels at this Altitude		0.73557265	m	Similar Triangles, $=SQRT(2*(C12^2-C12^2*COS(C11*PI()/180)))$
15	Velocity Beam is moving at this Altitude		119.070806	m/sec	$=C14/C4$
16	Necessary Buffer Distance		2.392552375	m	$=\{C15+C13\}*C4$
17					
18	Buffer Plus D_receiver			m	$=\text{'Preliminary Optics Sizer'!C6}+2*C16$
19	Buffer Plus D_receiver_Actual			m	$=\text{'Preliminary Optics Sizer'!C10}+2*C16$

Figure 4.5: A screenshot of the Safety Buffer Calculator section of the Optics/ATP design tool.

4.6 Benchmarks

The sections of the Optics/ATP design tool are benchmarked, except for the Constants section, which is mainly inputs, and the Safety Buffer Calculator, which merely calculates the relative movement between the SBSP system laser and an intercepting object.

4.6.1 Slant Range Calculator Benchmark

In Shuch (1978), a geostationary satellite is calculated to have a slant range 20,215 nmi with a spacecraft elevation angle of 44.61°. A geostationary satellite has an altitude of 35786 km (Larson & Wertz, 2004). Inputting this elevation angle and geostationary orbit altitude into the Slant Range Calculator gives a slant range of 37,439,715.46 m, which is equal to 20,215.8 nmi.

16	Slant Range Calculation	Input	Output	Units	Parameter Notes
17	epsilon		0.778591379	radians	Assuming Epsilon is limiting the slant range, so Epsilon_0=Epsilon =IF(B18<>"",PI()*B18/180,C13)
18			44.61	degrees	=IF(B18<>"",B18,C14)
19	pho		0.151852153	radians	Eq. 5-24, pg 113; (Larson, W. J. and Wertz, J. R., 2004); =ASIN((Constants!C13)/(Constants!C12+Constants!C4))
20			8.700487493	degrees	=C19*180/PI()
21	eta		0.10789833	radians	Eq. 5-26b, pg 113; (Larson, W. J. and Wertz, J. R., 2004); =ASIN(COS(C17)*SIN(C19))
22			6.182118954	degrees	=180*C21/PI()
23	lambda		0.684306617	radians	Eq. 5-27, pg. 113; (Larson, W. J. and Wertz, J. R., 2004); =PI()/2-C21-C17
24			64.65287973	degrees	=90-C22-C14
25					
26	Slant Range			m	Eq 5-28, pg 113; (Larson, W. J. and Wertz, J. R., 2004) =Constants!C12*(SIN(C23)/SIN(C21))

Figure 4.6: A screenshot showing the Slant Range Calculator being benchmarked for a Geostationary Satellite with a Spacecraft Elevation Angle of 44.61°.

4.6.2 Preliminary Optics Sizer Benchmark

In Henderson and Gregory (1983), an SBSP system with a separation of 38,850 km, a laser aperture of 40 m in diameter, and a laser with a wavelength of 10.6 μm has a reception dish of 23.2 m in diameter. When these parameters are put into the Optics/ATP design tool, it calculates that the receiver diameter to be 23.18°.

	A	B	C	D	E
1	Preliminary Optics Sizing	input	output	unit	Parameter Notes
2	$2.44 \cdot \lambda_{\text{laser}} / D_{\text{optics}} = D_{\text{receiver}} / \text{SEP}_{\text{las-receiver, max}}$				(Taylor, T., Anding, R. C., Halford, D. and Matloff, G.L., 2003)
3	Laser wavelength, λ_{laser}		10.600000	m	SPI Datasheet, =IF(B3<>"",B3,1070*10^-9)
4	Diameter of the Laser-transmitting optics, D_{optics}			m	SBSP System Architecture (Grady, J.R., 2008), =IF(B4<>"",B4,IF(AND(B4="",B6<>""),2.44*C3*C5/C6,0.75))
5	Separation between the laser power station and receiver, $\text{SEP}_{\text{las-receiver, max}}$		35850000	m	=Constants!C9
6	Receiver Diameter, D_{receiver}			m	=IF(B6<>"",B6,2.44*C3*C5/C4)

Figure 4.7: A screenshot showing the Preliminary Optics Sizer being benchmarked for a SBSP system with a transmitter aperture of 40 m at an altitude of 35,850 km.

4.6.3 Point Ahead Angle Benchmark

In Andrews and Phillips (1998), a LEO satellite moving approximately 7 km/s has a point ahead angle of roughly 50 μrad . The Optics/ATP design tool calculates a point ahead angle of 47 μrad for a SBSP system moving at 7 km/s.

	A	B	C	D	E
1	Point Ahead Angle Calculator		output	units	Parameter Notes
2	Ground Station Point Ahead Angel				
3	$\theta_{pa_G} \approx 2 \cdot V/c$		4.6699E-05	radians	(Andrews, L.C., and Phillips, R.L., 1998), =2*Constants!C5/Constants!C17
4			0.00266	degrees	=C3*180/PI()

Figure 4.8: A screenshot showing the Point Ahead Angle Calculator being benchmarked for a system traveling at 7 km/s.

CHAPTER 5

ISI SYSTEM DESIGN

This chapter describes the mechanical design of the ISI demonstration. The mechanical design was based on the preliminary concept detailed in the ISI System Architecture (Grady, 2008) and discussed in Chapter 3. This chapter then discusses the possible ISI environment on orbit. The chapter then goes on to discuss the primary systems that make up the ISI, the Laser System, the Instrument Bus, the ATP/Safety System, and the Ground Station Receiver. Finally this chapter discusses the CAD model that was developed based on these systems.

5.1 ISI Orbit Environment

One of the mission objectives of the ISI demonstration is to be compatible with the ISS. Ideally, the ISI would make use of facilities provided by the ISS. If the ISI cannot operate on the ISS, it will operate as a free flyer on an equivalent orbit of the ISS. The ISI demonstration is being designed to take into account the external payload facilities available on the ISS, the orbit parameters of the ISS, and the thermal and power environment on the ISS.

5.1.1 ISS External Payload Facilities

There are four potential external payload facilities for the ISS, the Integrated Truss Structure (ITS) Starboard 3 (S3) module, the ITS Port 3 (P3) module, the Columbia module and Japanese Experiment Module (JEM) Exposed Facility (EF). These external facilities are shown on the ISS in Figure 5.1 (Cook, 2007).

Figure 5.1: ISS External Payload Facilities (Cook, 2007).

The payloads for the ITS modules and the Columbia are constrained by the Flight Releasable Attachment Mechanism (FRAM) Express Logistics Carrier (ELC) Payload Envelope, shown in Figure 5.2 (NASA, 2008).

Figure 5.2: FRAM Express Logistics Carrier (ELC) Payload Envelope, in inches (millimeters) (NASA, 2008).

The payloads for the JEM-EF are constrained by a different platform, shown in Figure 5.3 (Chang, 2007).

Figure 5.3: JEM-EF External Payload Envelope (Chang, 2007).

5.1.2 ISS Orbit Parameters

The ISS operates in LEO and has a nearly circular orbit. The apogee is 361 km and the perigee is 348 km, giving it an eccentricity of 0.000954, an orbital inclination of 51.64°, a Right Ascension of the Ascending Node of 323.7°, and an argument of the perigee of 196.3° (Peat, 2009). The average orbital speed of the ISS is 7.7 km/s and it has an orbital period 91.34 minutes (International Space Station, 2009).

5.1.3 ISS Thermal and Power Environment

In order for the ISI demonstration to operate on the ISS, care must be taken to assure that all components can function under the conditions that occur there. Table 3.1

shows the external radiation conditions for payloads at the external facilities of the ISS (Cook, 2007).

Table 5.1: External radiation conditions for an ISS External Payload (Cook, 2007).

Case	Solar Constant (W/m ²)	Earth Albedo	Earth Outgoing Long Wave Radiation (W/m ²)
Cold	1321	0.2	206
Hot	1423	0.4	286

While on the JEM-EF, a payload such as the ISI demonstration needs to be able to withstand temperature variation between -49°F and +149°F (JAXA, 2000). A payload on the FRAM ELC needs to be conditioned for temperatures between -135°F and 260°F (NASA, 2008).

The payloads on the US FRAM ELC sites have a shared available power of 2.5 kW. The Columbia FRAM ELC sites also have a shared 2.5 kW available for external payloads. The JEM-EF sites have 3 kW shared among its payloads. The ISS also has 2.5 kW available for payload operations. So an ISI demonstration should have access to at least 2.5 kW if operated on the ISS (NASA, 2000). However, assuming 100 W are used to support payload capabilities, this leaves only 2.4 kW for laser power. If a 30% efficient laser were used for transmission, this would slightly reduce the amount of power beamed to the surface, from 200 W to 180 W.

5.2 Laser System

The ISI demonstration is being designed to wireless transmit power from orbit using a laser. As shown in Chapter 3, previous research into SBSP lead to the use of a

fiber laser in the ISI design. The ISI is currently being designed to be used with a fiber laser from Southampton Photonics Inc.

5.2.1 Southampton Photonics Inc. (SPI) Fiber Laser

The SPI fiber laser is designed in 400 W modules. However, after a conversation with the General Manager of SPI's US Division, Ken Dzurko, it was learned that these modules can be coupled together in order to provide greater power. It was also revealed that this laser has an efficiency of 30%, which meets the limit set in the ISI systems architecture. Table 5.1 shows some characteristics of the SPI fiber laser. This laser has not yet been space or man rated for use on orbit or the ISS.

Table 5.2: SPI fiber laser characteristics (SPI Lasers, 2008).

Optical Characteristics for 400 W Module

Central Emission Wavelength	1070 $\pm$ 10 nm
Mode of Operation	CW and Modulated
Output Power Variation	< $\pm$ 0.5% (typical)
Targeting Laser	Standard

Pulse Characteristics for 400 W Module

Maximum Modulation Rate	100 kHz
Pulse to Pulse Energy Variation	< 0.5%
Minimum Pulse Width	< 10 μ S

5.2.2 Laser Optics

The preliminary optics sizing for the ISI demonstration laser system is done using the Optics/ATP design tool described in Chapter 4. For an ISI demonstration operating on an orbit equivalent to that of the ISS and using an SPI laser, the optics would need to

be sized to be 73.6 cm in diameter to transmit to a 10 m diameter ground station with a spacecraft elevation angle of 19.2° between the ISI demonstration and the ground station.

5.3 Instrument Bus

The ISI demonstration Instrument Bus will provide the main structure for the free flyer concept. This instrument bus will house the solar panels, the command and control module for the ISI, and house fuel for the propulsion system. On the ISS, the main structure will be provided by the external payload platform, the power will be supplied by ISS solar panels, and orbital maintenance will not be needed. The command and control module will still be needed for the ISS concept.

The ISI is being designed to be used with the Saab Spacecraft Management Unit (SMU) for the command and control functions. The SMU is designed to provide housekeeping and payload control for the system, such as command decoding and validation, communication with the payload and other systems, and assuring that each system operates within their proper parameters. The typical SMU has dimensions of 400 mm in length, 200 mm in width, and 276 mm in height. The size and functions of the SMU can be scaled, however, to meet varying mission needs. Functions can be removed if they are not needed on the ISS, and the system can be resized to fit on the external payload platform provided (Saab International, 2005).

For the free flyer concept, the ISI is being designed to be used with Micrel Solar Concentrator Cells to collect power on orbit. During a presentation by Micrel Vice President Mark Lunsford (2008), it was learned that these cells use a lens to magnify the sunlight acting on the solar cells. The Micrel solar concentrator cells have an efficiency

of 27.5%, and the lens has an efficiency of 90%. Using a modified concentrator cell modeler, originally provided by Lunsford (2008), the ISI will need 7.88 m^2 (lens and solar cells) to gather the power necessary to beam 200 W of power to the ground. The free flyer system, with an estimated need of 4 kW, would require a total of 11.82 m^2 of concentrators to operate the spacecraft. More information of this Solar Concentrator Sizer is shown in the Appendix.

5.4 ATP/Safety System

The ATP/Safety system for the ISI demonstration is based on technology used in laser communications. The system is being designed to lock onto a signal sent from the ground station receiver to assist with system pointing. This signal will also act as part of the safety system, as the laser will not be allowed to activate unless it receives confirmation from the ground that it is pointed at the receiver ground station.

In order to assure that the path between the ISI demonstration and ground station remains clear once the laser is activated, a safety buffer around the beam will have to be constantly monitored while the beam is activated. The needed safety buffer for the ISI demonstration was sized by the Safety Buffer Calculator portion of the Optics/ATP system to be 2.39 m. Anything entering this safety buffer will result in the beam deactivating.

5.5 Ground Station Receiver

The ground station receiver is being designed to collect power from solar cells tuned to collect power in the IR frequencies, giving them an efficiency of about 50%. The minimum spot size needed for the ground station receiver is determined by

Rayleigh's Criterion. By using the Preliminary Optics Sizer with an input of 75 cm for the diameter of the laser optics, the maximum optics size that could be used on all ISS or free flyer configurations of the ISI, the minimum receiver diameter size needed for a 926 km slant range and a 19.2° spacecraft elevation angle would be 9.82 m. Figure 5.4 shows the relationship between the size of the laser optics, and the size of the receiver ground station. This shows that the spot size and the laser optics are inversely related.

Figure 5.4: Change in ground spot size as the diameter of laser optics is increased

If necessary, the size of the optics could be reduced slightly and only small increases to the spot size would occur. However, for optics sizes below 50 cm, the diameter of the ground spot starts to become too large for a manageable ground station for the ISI demonstration. The spacecraft elevation angle also has an effect on the receiver diameter size. As mentioned in Chapter 4 when discussing the Preliminary Optics Sizer,

the laser spot is perpendicular to the beam, and the Preliminary Optics Sizer takes this angle into account. Figure 5.5 shows the relationship between the spot size and the elevation angle.

Figure 5.5: Ground Spot Size as ISI Spacecraft Elevation Angle Increases

As the elevation angle increases, the effect on the spot size increases. If the elevation angle were increased to 30°, this would reduce the spot size diameter by 1/3rd. This reduction in spacecraft elevation angle would also reduce the maximum separation between the transmitter and the ground station, although this effect is not shown in Figure 5.5. Reducing the elevation angle would also reduce the time of the overhead pass of the ISI system, however, and that may not be desirable.

5.6 Mechanical Design Model

The CAD models in the FRAM and JEM-EF Configuration were drawn based on the preliminary architecture design, using Dassault Systèmes SolidWorks Corp's

Solidworks Student Edition. The configuration for the JEM-EF is shown below. The lasing device is shown in red, the tracking device is shown in blue, the instrument bus is shown in brown, the mirror aperture is shown in grey and the ISS interfaces are shown in black. The dimensions derived from the ISI System Architecture (Grady, 2008). The primary mirror has a diameter of 75 cm, with a 10 cm hole at the center for the laser beam. The beam strikes the smaller secondary and is reflected back to the primary mirror. This focuses the beam and transmits it to the target.

Figure 5.6: CAD drawing of ISI demonstration configured for use on the JEM-EF.

The mirror aperture has a thickness of 1.5 cm, giving the SBSP demonstration a clearance of 1 cm from the payload envelope, on each side. The lasing device has a height of 30 cm and the mirror aperture has a height of 65 cm, which gives a clearance of 5 cm

from the end of the payload envelope. The tracking device has a diameter of 10 cm and a height of 20 cm.

Other CAD models, including a model of the ISI configured for the FRAM ELC, will be shown in Appendix A.

CHAPTER 6

THERMAL MODEL AND ANALYSIS

This chapter describes the thermal model that was built to analyze the ISI demonstration. This thermal model was based on the preliminary mechanical design discussed in Chapter 5. This chapter describes the how the thermal model was built, details the case that was run to analyze the thermal system, and displays and analyzes the results of that preliminary case.

6.1 Thermal Model

The thermal model was designed to represent and ISI demonstration placed on the ISS JEM-EF. The preliminary thermal model, created using Thermal Desktop from Cullimore and Ring Technologies, Inc., used five rectangular plates to simulate the JEM-EF external payload platform and is shown in Figure 6.1.

Figure 6.1: The Thermal Desktop Model of the ISI Demonstration on the JEM-EF.

Two disks, and a cylinder, shown in red, were used to simulate the lasing device. A lone cylinder, shown in green, was used to simulate the laser mirror aperture. These objects were modeled as two dimensional wireframes and then given a thickness to make them three dimensional. The sizes of the objects for the thermal model were based on the sizes derived from ISI Systems Architecture, discussed in Chapter 5 and shown in Figure 5.4.

Once the objects were shaped, they were given physical properties. All of the objects for this model were designed using a dummy material, in order to find the critical areas of the system. This dummy material was defined as having a Conductivity of 50 W/m/K, a Density of 1000 kg/m³, and a Specific Heat of 100 J/kg/K. Once the physical properties of these objects were entered, nodes were generated for each object. Since the purpose of this model was to determine the critical points affecting the system, it was designed as a coarse, uniform model. Each object was given five nodes in the x-direction of the object and five nodes in the y-direction of the object.

To simulate the laser, a small 10 cm diameter disk was created at the center of the lasing device and a heat load of 2.4 kW was applied to it. This value was chosen because it was estimated that an ISS payload would need 100 W for supporting systems. Given the amount of power the ISS can provide a payload as discussed previously in Chapter 5, 2.5 kW, this leaves 2.4 kW available for use with the ISI laser. Since a disk was used to approximate the much more complex mechanism inside the lasing device, contactors were added from this disk to the surrounding lasing device.

Finally, in order to complete the thermal model, an orbit for the system had to be created. The ISI orbit was modeled as a Keplerian orbit, and entered the perigee and the

apogee of the ISS, 348 km and 361 km, as the main parameters for this orbit. Thermal Desktop then was able to shape the orbit from these two parameters. The orbital inclination of the ISS, 51.64° was then input into the model. The orbit was set so that the orientation of the system was NADIR pointing, so the ISI demonstration was modeled always point towards the ground, and the planet was set as Earth.

6.2 Test Case Run

Once the model was specified, the test case on the system could be run. Thermal Desktop interfaced with Sinda/Fluint to perform the thermal analysis of the system. The test case was run as a transient case of 900 seconds, the amount of time estimated for an ISS pass over a ground station receiver, was specified. The case was set to calculate radks, Thermal Desktop radiation analysis, and heating rates, analysis about the external heating environment such as albedo Earth IR and solar flux affecting the system. This 900 second run was done twice, once with no heat load applied to the lasing device, and once with the 2.4 kW heat load applied, so the a pass over a target ground station could be compared to the ambient state of the model. Once these tests were run temperature profiles of the results were displayed.

6.3 Results

The results of these thermal tests are displayed in Figure 6.2 and 6.3 respectively. Figure 6.2 shows the ambient run for the ISI demonstration and Figure 6.3 shows the run with the laser activated for the system.

Figure 6.2: Thermal Analysis of Ambient System.

Figure 6.3: Temperature Profile of System with Laser Activated for 900 Seconds

For the ambient case, the temperatures are between 150 and 270 K. When the heat load of 2.4 kW is applied to the system, the temperatures of the system fall between 170 and

600 K. The hotter temperatures occur around the lasing device, with the portion of the payload envelope surrounding the lasing device reaching temperatures between 366 K and 300 K. This heat will need to be cooled through the system or bled off through radiators.

CHAPTER 7

CONTINUED EFFORT FOR ISI DEMONSTRATION

7.1 Program Refinement

Both the thermal models and the Optics Design tool can be improved for future analysis. The thermal can replace the simple lasing device approximation with something more realistic. Instead of using a simplistic approximation for the laser system, details such as laser diodes and cooling systems must be developed and analyzed to determine their system performance. More realistic materials can also be used when doing analysis. Instead of using a generic dummy material, spacecraft parts can be modeled using commonly used spacecraft material. This material can be treated with optical paints and analyzed to see how effectively the material transfers/traps heat.

The optics design tool can also be improved by allowing for more complex trajectories of the SBSP system and the intercepting objects. The orbit parameter of the SBSP system can be added to create a model that calculates the beam location as the demonstration moves along its orbit.

7.2 Final Design and Fabrication

Further design work needs to be performed on the ISI demonstration. Radiation Analysis can be performed to determine if energetic particles in the Earth's magnetosphere would have any effect on the ISI demonstration. Structural analysis can be done to determine that the demonstration could survive launch to orbit. Thermal

analysis needs to be done to assure that either the ISS or the ISI can safely get rid of the excess heat generated by powering a 30% efficient, 800 W laser.

Once this is finished, work can be done assembling the craft in preparation for launch. System verification and validation must be done to assure that each part provided will do what it was designed or purchased to do.

CHAPTER 8

CONCLUSIONS

This thesis has discussed the Detailed Design of the ISI demonstration. A thermal model of the ISI demonstration was built, and the critical thermal designs points of the systems were determined to be around the lasing device. Something must be done to dissipate heat and cool the laser once it has been activated. The systems that make up the ISI demonstration were detailed in this thesis. The laser system was designed to be used with a fiber laser from SPI. The laser optics system was sized to be between 73.6 and 75 cm. The Saab SMU was selected to carry out system maintenance and run the command and control systems for the demonstration. If the ISI operates as a free-flyer, the spacecraft will require 11.82 m² of Micrel, Inc., solar concentrator cells. The ground station receiver spot will be between 9.82 and 10 m in diameter. The safety system of the ISI will need to maintain a 2.39 m buffer around the beam to assure that no stray objects can enter the path of the beam without automatically deactivating it. This means that the ground station receiver has to be between 14.60 and 14.79 m in diameter in order to give enough time for the safety systems to deactivate the laser should anything be endangered by the beam.

The Excel program, the Optics/ATP design tool, was created for this thesis for the purpose of designing optical systems of general SBSP systems. Several benchmarks have been given for the Optics/ATP design tool, comparing the results of the design tool to known satellite systems and SPSB designs.

More work has to be done before the ISI demonstration can be considered flight ready. The Optics/ATP design tool has to be refined to allow for more detailed orbits and more accurate ground spot sizes. The thermal model needs to be designed with more detail, especially of the lasing device. The entire ISI demonstration system has to go through fabrication design so it can be assembled and prepared for launch.

Once this is done, Dr. Peter Glaser's vision of SBSP can finally be realized.

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APPENDIX A:

DETAILED DESIGN

Appendix A shows other configurations of the ISI demonstration as well as component parts of the system. Like the figure shown in Chapter 5, these were developed using Dassault Systèmes SolidWorks Corp's Solidworks Student Edition and were based off of the ISI preliminary Mission Architecture.

Figure A.1: ISI Demonstration configured for the FRAM ERC.

Figure A.2: Model of ISI demonstration configured as a free flyer.

Figure A.3: Model of ISI demonstration mirror aperture.

APPENDIX B:

ISS EXTERNAL PAYLOAD SITES

The following images are close-ups of the US FRAM ERC attachment sites and the JEM-EF.

Figure B.1: S3 and P3 Truss Sites (Cook, 2007).

Figure B.2: Japanese Experiment Module - Exposed Facility (JAXA, 2000).

UCM-P (Umbilical Connect Mechanism-Payload Side)
 TS-P (Trunnion Structure-Payload side)

Figure B.3: JEM Payload Attachment Mechanism (PAM) Payload Unit (PU) (Cook, 2007).

APPENDIX C:

MODIFIED LUNSFORD SOLAR CONCENTRATOR PROGRAM

In August of 2008, Mark Lunsford, a Vice President at Micrel Inc., came to the NASA Space Portal to talk about his companies new concentrator solar cells. Along with his presentation, he passed along a program he created to model the Micrel Inc. Solar Concentrators for the ISI demonstration. This program works by starting with the targeted amount of power transmitted to the ground, and works backwards though the system to size it.

200 Watts on the Wire	400 Watts delivered to the cells on the Ground
400 Watts to the Ground cells means	800 Watts delivered from the laser
800 Watts from the laser means	2,667 Watts delivered to laser from the Space Cells
2,667 Watts from the Space Cells means	9,697 Watts delivered to the Space Cells
9,697 Watts to the Space Cells means	10,774 Watts delivered to the Collection Lens
10,774 Watts to the Space Lens means	7.88 Square meters of Space Sun
10,774 Watts in space means	0.0269 Square meters of Space Cells
So we need	
10 Square cm of solar cells on the ground	
269.36 Square cm of solar cells in Space	

Figure C.1: A screenshot from the modified Mark Lunsford solar cell modeler.

APPENDIX D:

CONCENTRATOR SIZING FOR GROUND STATION REQUIREMENTS

Table D.1 shows the relationship between the amount of watts collected on the ground and the surface area of Micrel Solar Concentrator Cells needed to collect enough power to transmit the required watts to the surface.

Table D.1: Solar concentrator area needed to supply watts.

Watts to the Ground	100	150	200	250	300	350	400	450
Watts on Orbit	5387	8081	10774	13468	16162	18855	21549	24242
Concentrator Area Needed (W/m ²)	3.94	5.91	7.88	9.85	11.82	13.79	15.76	17.73

Watts to the Ground	500	550	600	650	700	750	800	850
Watts on Orbit	26936	29630	32323	35017	37710	40404	43098	45791
Concentrator Area Needed (W/m ²)	19.70	21.67	23.65	25.62	27.59	29.56	31.53	33.50

Watts to the Ground	900	950	1000	1050	1100	1150	1200	1250
Watts on Orbit	48485	51178	53872	56566	59259	61953	64646	67340
Concentrator Area Needed (W/m ²)	35.47	37.44	39.41	41.38	43.35	45.32	47.29	49.26

Watts to the Ground	1300	1350	1400	1450	1500	1550	1600	1650
Watts on Orbit	70034	72727	75421	78114	80808	83502	86195	88889
Concentrator Area Needed (W/m ²)	51.23	53.20	55.17	57.14	59.11	61.08	63.05	65.02

Watts to the Ground	1700	1750	1800	1850	1900	1950	2000
Watts on Orbit	91582	94276	96970	99663	102357	105051	107744
Concentrator Area Needed (W/m ²)	67.00	68.97	70.94	72.91	74.88	76.85	78.82

APPENDIX E:

GROUND SPOT VARIATION DUE TO ORBIT ALTITUDE

The following table shows the relationship between the ground spot size and the orbit altitude. These spot sizes were generated from lasers directly overhead of their targets. These calculations were made using the Optics/ATP design tool. The orbits chosen were circular orbits, except for the Molinya orbit. For that orbit, the laser was generated at the apogee of the orbit. The laser used generating the table was as an SPI laser with a wavelength of 1070 nm. The optical system was assumed to be 75 cm in diameter for each case.

Table E.1: Ground spot variation due to orbit altitude.

	Unit							GEO	Molinya
Orbit Altitude	m	100000	200000	300000	400000	500000	600000	35786000	39750000
Ground Spot	m	0.348107	0.696213	1.04432	1.392427	1.740533	2.08864	124.5735	138.3724