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The Application of Systems Engineering to a Space-based Solar Power Technology Demonstration Mission

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The Application of Systems Engineering to a Space-based Solar Power Technology Demonstration Mission

by

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Dedicated to my parents for their eternal support and unwavering confidence in me.

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The Application of Systems Engineering to a Space-based Solar Power Technology Demonstration Mission

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The University of Texas at Austin, 2012

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This thesis presents an end-to-end example of systems engineering through the development of a Space-based Solar Power Satellite (SSPS) technology demonstration mission. As part of a higher education effort by NASA to promote systems engineering in the undergraduate classroom, the purpose of this thesis is to provide an educational resource for faculty and students. NASA systems engineering processes are tailored and applied to the development of a conceptual mission in order to demonstrate the role of systems engineering in the definition of an aerospace mission.

The motivation for choosing the SSPS concept is two fold. First, as a renewable energy concept, space-based solar power is a relevant topic in today's world. Second, previous SSPS studies have been largely focused on developing full-scale concepts and lack a formalized systems engineering approach. The development of an SSPS technology demonstration mission allows for an emphasis on determining mission, and overall concept, feasibility in terms of technical needs and risks. These are assessed through a formalized systems engineering approach that is defined as an early concept or feasibility study, typical of Pre-Phase A activities. An architecture is developed from a mission scope, involving the following trade studies: power beam type, power beam frequency, transmitter type, solar array, and satellite orbit. Then, a system

hierarchy, interfaces, and requirements are constructed, and cost and risk analysis are performed.

The results indicate that the SSPS concept is still technologically immature and further concept studies and analyses are required before it can be implemented even at the technology demonstration level. This effort should be largely focused on raising the technological maturity of some key systems, including structure, deployment mechanisms, power management and distribution, and thermal systems. These results, and the process of reaching them, thus demonstrate the importance and value of systems engineering in determining mission feasibility early on in the project lifecycle.

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Glossary

FoM - Figure of Merit. A metric, or attribute, by which a stakeholder's expectations will be judged in assessing satisfaction with a product or system. Used in trade studies to compare and evaluate design alternatives.

MW - Microwave. Refers to microwave radiation.

PV - Photovoltaic. Refers to the photovoltaic cell used in solar panels to generate electricity from incident solar radiation via the photoelectric effect.

RF - Radio Frequency. Refers to RF power, i.e., radiation in the radio frequency range of 3 kHz to 300 GHz.

SDS - Solar Dynamic Systems. Refers to systems that generate electricity from incident solar radiation via heat transfer.

SSP - Space-based solar power. Refers to the idea of generating large-scale solar power in space to then be used on the ground.

SSPS - Space-based Solar Power Satellite. A satellite that uses solar energy to generate large amounts of power for use by external systems. The key features are large solar arrays and the transmitter portion of the WPT element.

SSPS system/concept - Space-based Solar Power Satellite system/concept. Includes both the space and ground segment. The space segment is the SSPS. The ground segment consists of the receiver portion of the WPT element and the power grid interface.

SSPS-TD - Space-based Solar Power Satellite - Technology Demonstration. Mission to demonstrate and validate key SSPS system concepts and technologies. Generally refers to the space segment unless the phrase *SSPS-TD mission* or something similar is used.

TRL - Technology Readiness Level. A metric by which to measure the state of development of a technology in terms of its readiness for mission implementation.

WPT - Wireless Power Transmission. The method of transmitting power through electromagnetic waves.

WPT Element - Wireless Power Transmission Element. The physical system that performs WPT. This involves 1) the conversion of electricity into electromagnetic waves via RF or laser generators, 2) the transmission of these waves via a transmitter (antenna), 3) the collection and conversion of these waves into electricity via a receiver.

Chapter 1

Introduction

1.1 Motivation

This thesis, sponsored by a NASA grant through the Texas Space Grant Consortium (TSGC)[1], provides an end-to-end application of systems engineering for the undergraduate classroom. The motivation for this presentation comes from an undergraduate course entitled Space Systems Engineering that is offered at the University of Texas at Austin (UT) and developed by NASA engineer Lisa A. Guerra. The course covers the full range of NASA's systems engineering processes with the goal of teaching the "fundamentals of systems engineering such that future practicing engineers are familiar with the concepts and processes to be exercised further in the work environment."[2] It does not seek to create systems engineers but rather give engineering students a systems perspective.

With this incentive, it is worth developing an end-to-end example of a conceptual mission that provides a concise treatment of systems engineering. In this way, it can be used by students and faculty as an educational resource in understanding the practices of systems engineering. Furthermore, the course at UT is a prerequisite for the senior capstone design course in the Department of Aerospace Engineering, so this thesis is helpful for seniors who will have to apply classroom systems engineering to an aerospace capstone design project. The motivation is furthered by the fact that

[1] http://www.tsgc.utexas.edu/
[2] http://spacese.spacegrant.org/

the concept of systems engineering is often difficult to grasp. Systems engineering has no single role or definition in the development of a project, and it is not a linear process but rather an iterative one. The best way then to understand systems engineering is to see it in practice. This thesis therefore demonstrates key systems engineering processes by applying them to the development of a conceptual mission, and thus highlights the role of systems engineering in mission design and reveals how these processes come together to create a mission.

1.2 Background

An excellent definition of systems engineering appears in a paper entitled "The Art and Science of Systems Engineering":[3]

> Systems engineering is the art and science of developing an operable system that meets requirements within imposed constraints. Systems engineering is holistic and integrative. Systems engineering is first and foremost about getting the right design - and then about maintaining and enhancing its technical integrity, as well as managing complexity with good processes to get the design right.

Systems engineering therefore offers a systematic approach to overcoming the challenge of design, which can be a complex problem with many variables and no obvious solution. It is a broad field with a range of methodologies and there exist a number of definitive texts on the discipline, like the *NASA Systems Engineering Handbook*[4] and *Systems Engineering Analysis*, by B.S. Blanchard and W. J. Fabrycky.[5] As a foundation, this thesis uses the NASA systems engineering baseline.

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[3] [34]
[4] [9]
[5] [12]

Part of systems engineering is describing and interacting with all of the phases of the design and development that define the project lifecycle. The project lifecycle, as defined in the systems engineering discipline, is shown in Figure 1.1. Because this thesis focuses on the development of a *conceptual* mission, the following systems engineering analyses are associated with Pre-Phase A. The purpose of this phase is to produce a range of ideas or design alternatives for a mission, from which a new project can be selected. Studies in this phase are thus typically referred to as concept or feasibility studies.

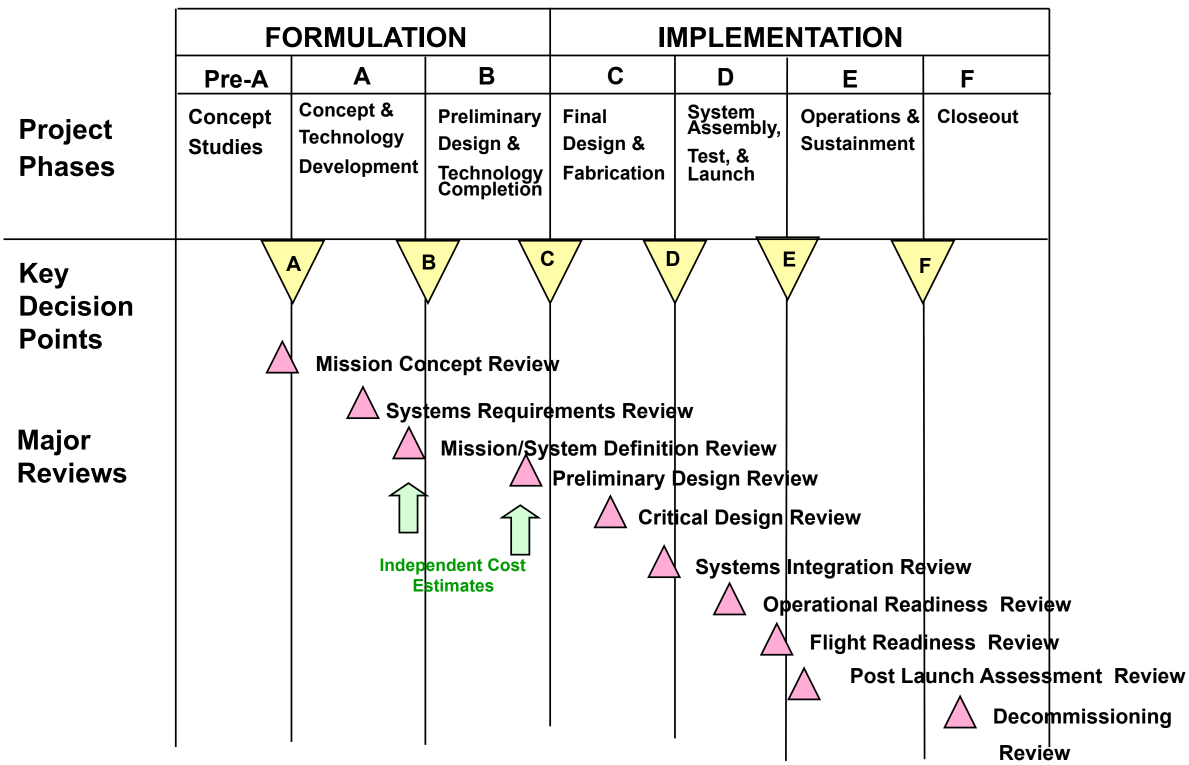

Figure 1.1: Project Lifecycle with key decision points and major reviews

The standard NASA project lifecycle is presented here, displaying the major phases, decisions points, and reviews. This thesis will take place in Pre-Phase A as a concept study, with the goal of reaching the Mission Concept Review and beginning Phase A development.

The activities that characterize Pre-Phase A are:

- Define mission needs, goals & objectives (e.g. mission scoping)
- Develop a concept of operations
- Perform studies of a broad range of mission concepts that fulfill the goals and objectives
- Develop draft project-level requirements
- Identify potential technology needs, mission hurdles, and risks
- Demonstrate at least one mission concept that is feasible

The goal, according to Figure 1.1, is to reach the Mission Concept Review, where a feasible concept is selected to be developed as a functional baseline in Phase A.

The mission concept that is chosen for this thesis is the Space-based Solar Power Satellite (SSPS). The idea is to utilize solar energy collected in space and transmit it wirelessly to the ground (power beaming), where it can be linked to the power grid. As a renewable energy concept it is a relevant topic and an interesting design problem due to its complex and unique nature.

This is not a new idea, as the U.S. and Japan have considered it in paper studies since the 1970's. However, developing an SSPS operational scenario is an extremely challenging task. The concept involves a variety of technologies, many of which are cutting-edge and technologically immature, and there exist stringent requirements and constraints on the system that influence fundamental mission features like orbit and satellite size. There is therefore a need for a technology demonstration mission, and the development of such a mission, here named the SSPS-TD mission, is the focus of this thesis.

As shown in Figure 1.2, the technology demonstration mission is a necessary and major step in the realization of the SSPS concept; all of the SSPS-relevant technologies must be demonstrated in a relevant environment, and as an integrated system before a full-scale SSPS implementation. Furthermore, it is vital to develop such a mission within the context of a Pre-Phase A concept study in order to better understand mission feasibility and the effort required to implement the SSPS system. The idea of mission feasibility and its relationship to technology maturity and risk are thus central to this thesis, and demonstrate the value of systems engineering in the early part of the project lifecycle.

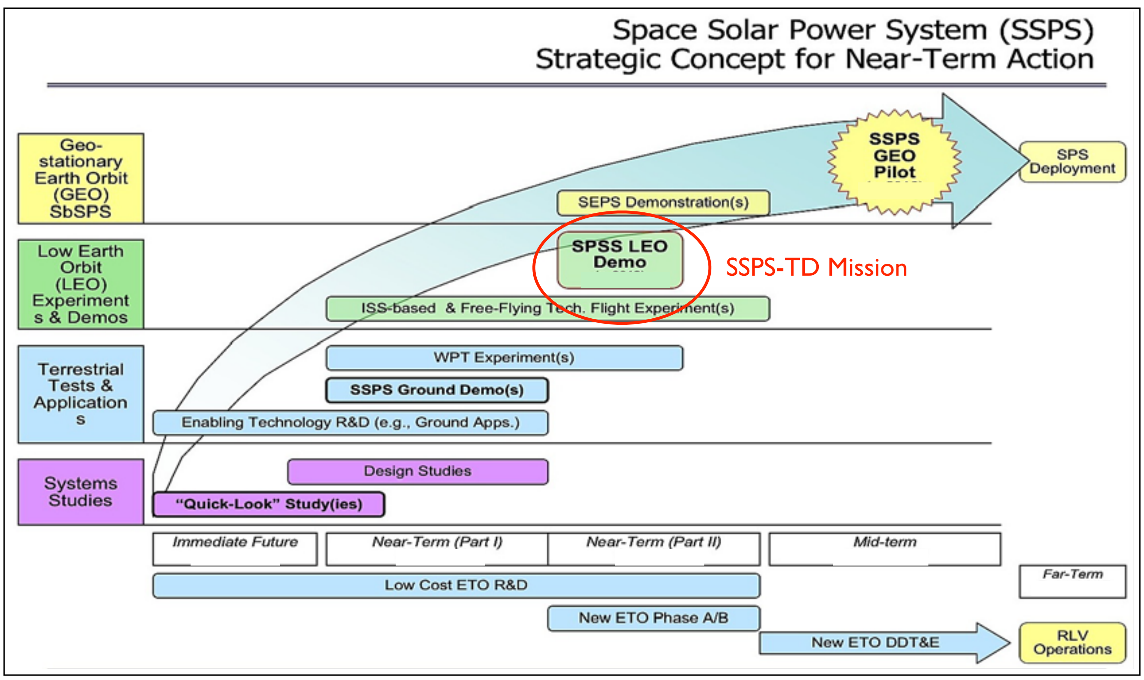

Figure 1.2: Pathway to SSPS operational implementation

This figure reveals the role that the technology demonstration mission in this thesis (SSPS-TD) plays in the ultimate development and implementation of a full-scale SSPS concept. Source: [8]

1.3 Thesis Organization

The thesis has a total of ten chapters, throughout which the range of systems engineering processes are performed. As mentioned earlier, systems engineering is an iterative process, making it difficult to present. Instead, a top-down approach is used that follows the organization of the Space Systems Engineering course. The concept of iterations is discussed throughout in order to better reveal its role and appreciate its impact and necessity.

Chapter 2 develops the mission scope and concept of operations. Included is more information on the concept and history of SSPS, the motivation behind choosing it as a potential renewable energy source, and the rationale for developing a technology demonstration mission.

Chapter 3 develops the architecture and performs the trade studies necessary for its formulation. The principles of power beaming are discussed within the context of designing a technology demonstration mission, and this paves the way for trade studies on the power beam, receiver, solar arrays, and satellite orbit.

With the architecture defined, Chapter 4 focuses on the system hierarchy and presents the Product Breakdown Structure with a preliminary description of necessary subsystem specifications and configurations.

Chapter 5 pulls together the results from Chapters 3 and 4 to present the satellite design summary, including mass, power, and efficiency budgets.

Chapters 6,7 and 8 continue the systems engineering analysis by discussing interfaces, generating requirements, and performing cost analysis, respectively.

Chapter 9 identifies the technology needs and hurdles and maps these to risks that affect overall mission feasibility and success.

Finally, Chapter 10 concludes the thesis with some final results and discussions on the mission design, and the role that systems engineering played through its development.

Chapter 2

Scope & Concept of Operations

The project scope and concept of operations (ConOps) is formalized in this section. The scope is the broadest formalization of the project, where the need, goals, and objectives are identified within the context of stakeholder expectations. This is the first step in establishing the mission architecture and provides the basis for the concept of operations. The concept of operations then describes how the mission will be operated in order to meet the need, goals, and objectives defined in the scope.

As the first step in the systems engineering process, the scope and ConOps are restricted to a top-level analysis, or "first iteration". Further mission details, including specific design-related assumptions and constraints, can be found in later chapters where they are developed and derived.

2.1 Scope

The needs, goals and objectives of the project are discussed here. From these, a mission description is constructed and the authorities and responsibilities involved, mission assumptions, and mission constraints are stated.

2.1.1 Need

To investigate renewable energy sources or methods of energy generation and distribution that can supplement the current electrical power grid.

This need comes primarily from two issues that together raise concern over the long-term sustainability of modern life: global energy demands, and environmental changes. In order to fully understand these issues, a brief discussion of current energy supply and demand is presented.

Today's society is extremely power hungry. In developed countries, everyday life hinges on a multitude of power-consuming technologies. This power consumption is divided into the following sectors:

- transportation
- industrial (manufacturing, etc.)
- commercial (public, services, etc.)
- residential

Energy utilized in any of these sectors comes from one of the following sources:

- Fossil Fuels: Organic matter that is formed under high temperatures and pressures in the Earth over long periods of time. The energy in these fossil fuels is harnessed through burning. Fossil fuels include oil (petroleum), coal, and gas.
- Nuclear: Self-sustained atomic fission is used to produce heat which is converted to electricity.
- Renewables: Energy derived from forces of nature and converted to electricity. Renewable energy sources include wind, solar, biomass, biofuels, wood, geothermal, and hydropower.

It is important to note that these energy sources have distinct physical and operational characteristics. They are therefore not readily interchangeable or adaptable to existing power system interfaces. The most obvious example is automobile transportation and its support infrastructure (refueling stations), which make nearly sole use of oil. The introduction of new energy sources within those sectors listed above thus requires fundamental changes to global infrastructures. Cost then becomes the leading issue, and it is clear that without political or social pressure, the types of energy sources and their usage are determined entirely by fiscal economics.

Figure 2.1 displays the breakdown of energy consumption in the U.S. in 2010. The essentially irreplaceable fossil fuels are the dominant energy source with the leading transportation and industrial sectors using 94% oil, and 40/41% oil and natural gas, respectively (not shown in figure).

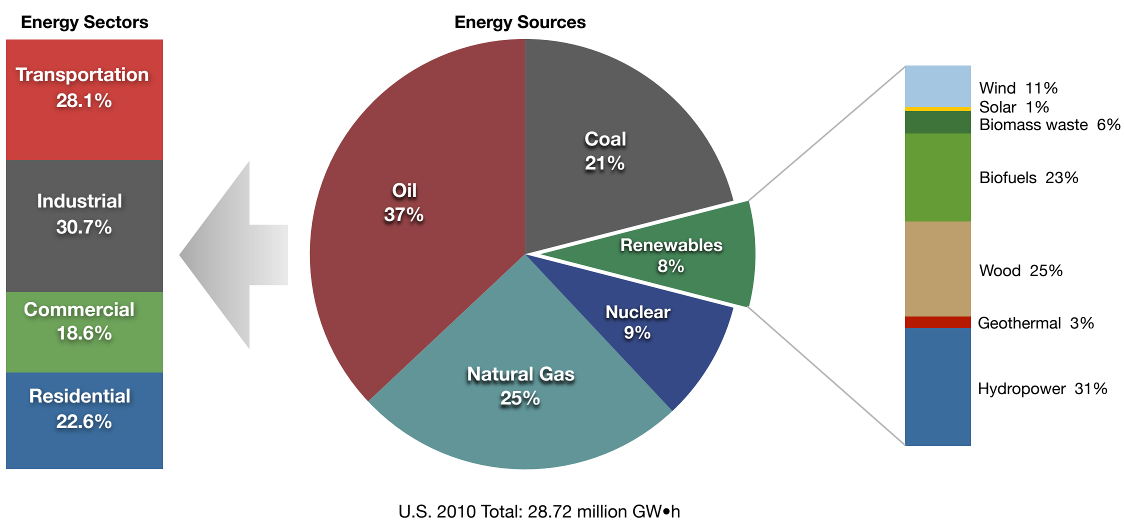

Figure 2.1: U.S. energy consumption in 2010

On the left is the usage breakdown by sector, on the right is the usage breakdown by energy source. Data taken from the U.S. Energy Information Administration.

Figure 2.2 displays the breakdown of electricity generation capacity in the U.S. in 2010. Here, we see the dominance of coal and natural gas. Reliance on such a monopolized power grid poses serious risks for future world development. These will now be discussed.

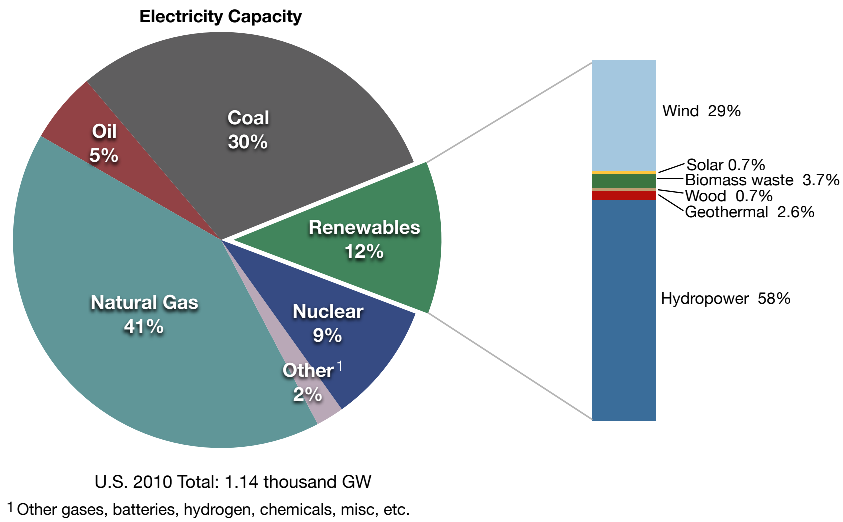

Figure 2.2: U.S. electricity generation capacity in 2010

This data is based on currently existing generators/power facilities in the U.S. Data taken from the U.S. Energy Information Administration.

Global Energy Demands

As shown in Figure 2.3, world energy consumption (i.e., demand) has increased by nearly 50% since 1990. Over 70% of future growth will come from developing countries, like China and India, and over 80% of the total energy demanded will be provided by fossil fuels.[1] This growth is due to the combined effects of population growth, and the rapid advancement and spread of power-consuming technologies in both developed and underdeveloped countries.

Affordable and accessible electricity is essential to economic growth, particularly in developing countries. Global prosperity is therefore dependent on an increase in energy supply. But as energy production rises to meet the increasing demands,

[1]World Energy Council, 2007 Survey

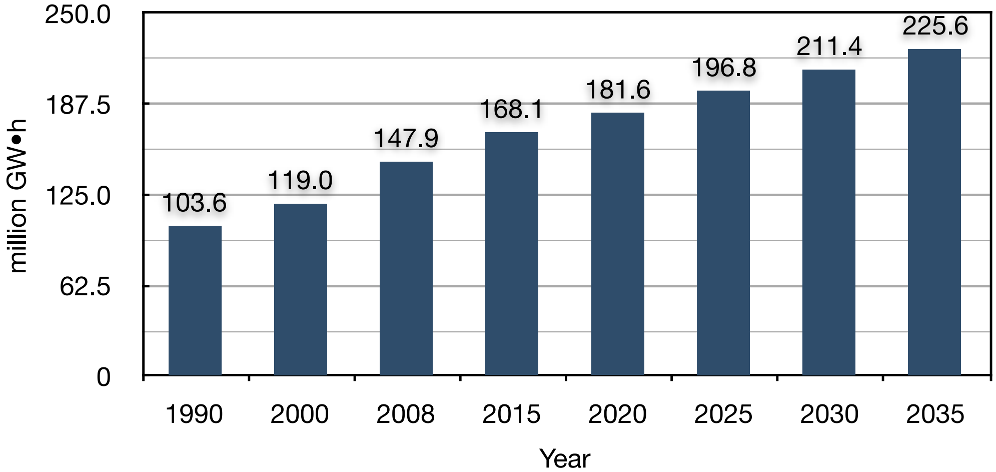

Figure 2.3: Global energy consumption with forecast

Data taken from the U.S. Energy Information Administration (Report DOE/EIA-0484, 2011).

irreplaceable fossil fuels are more quickly consumed. Current trends suggest that the conventional means of electric power generation (namely fossil fuels) will be unable to keep up with the forecasted demand, making it all the more urgent that the existing energy production infrastructure be adapted to new sources of energy on a global scale.

Environmental Concerns

There are a variety of environmental effects produced by modern energy sources. These are categorized in Table 2.1.

The severity of these effects varies, but fossil fuels are by far the most environmentally damaging. The most immediate and pervasive effects are atmospheric pollution and global warming. A transition away from conventional fossil fuels and towards cleaner and safer sources of energy is therefore of great importance. Consequently, more than half of the states have their own renewable electricity mandates or

Table 2.1: Environmental Effects of Energy Sources

Effect	Type	Energy Source	Impact	Description
Greenhouse gas emissions	Atmospheric pollution	Fossil Fuels	Global	Disruption of the natural Carbon cycle through the increase in greenhouse gases leads to global warming and climate change.
Acid rain	Atmospheric pollution	Fossil Fuels (mainly coal)	Global	Harmful to plants, soil, aquatic life and infrastructure
Smog/CO emissions	Atmospheric pollution	Fossil Fuels (e.g., vehicle emissions, industrial fumes, and coal burning), Wood burning	Local	Reduces air quality, causing serious harm to human health (mainly pulmonary effects)
Nuclear Waste	Groundwater/soil pollution	Nuclear Power (storage of waste materials)	Local	Radionuclides leak out of underground storage into surrounding soil and groundwater where it is ingested by local plants, animals and people.
Oil spills	Surface water pollution	Oil	Local	Accidental oil spills during sea transportation poison surrounding life
Waste heat	Thermal pollution	Solar Energy, Geothermal, Coal Plants	Local and Global	Heating of local atmosphere likely creates an urban heat island, among other minor effects.
Land Use/Alteration	Structural pollution	Solar Energy, Hydropower, Windpower	Local	Intensive use of land area disrupts local fauna and flora.

energy saving policies, though no federal mandate currently exists.[2] These typically specify a target percentage of electricity that must be renewable by a certain year.

The Case for Space-based Solar Power

The increased global power demand points toward the need for more primary energy sources, while environmental concerns points toward the need for cleaner and safer

[2] [3]

energy sources. The decision-making process to implement new energy sources is thus forced away from an exclusively economic basis. As Mankins[3] points out, energy R&D acts as an "insurance" against the economic and environmental uncertainties described above.

Renewable energy sources are an obvious solution to these two issues, with the added benefit of combatting high oil prices. Harnessing the energy of naturally occurring forces on Earth creates an essentially unlimited power source insofar as long-term supply is concerned - hence the term "renewable". These energy sources are generally environmentally friendly, with the relatively minor exceptions listed in Table 2.1.

Of particular interest is solar power. The average extraterrestrial solar irradiance[4] is 1361 W/m^2.[5] On a clear day, about 76% of this sunlight, or 1034 W/m^2, reaches the Earth's surface and can be harnessed on the ground as electrical power.[6]

This energy conversion is done either indirectly using Solar Dynamic Systems (SDS)[7], or directly via photovoltaics (PV). SDS uses lenses or mirrors to focus incoming sunlight into a concentrated beam that heats a fluid, which in turn generates power. Current typical end-to-end[8] efficiencies range between 15-30%. PV utilizes a solar cell, or photovoltaic cell, that generates electricity via the photoelectric effect. Current typical end-to-end efficiencies range between 8-20%.[9,10]

In the past, SDS was far more efficient than PV cells and so was the preferred

[3] [40]
[4] The amount of sunlight that reaches the Earth before passing through the atmosphere.
[5] [33]
[6] Data from UO Solar Radiation Monitoring Laboratory.
[7] Also referred to as Concentrated Solar Power (CSP) or Solar Thermal.
[8] From incoming sunlight to DC output.
[9] With the exclusion of Concentrator Silicon cells that operate at $\sim 38\%$.
[10] U.S. Energy Information Administration, data from 2009.

method for large scale power production. In the last decade, however, major advances in PV technology have made it competitive with SDS efficiencies and costs. Today, the higher mass, cost and complexity of SDS make PV systems the dominant choice for solar power production. As they are also the only solar power technology that is space qualified, from here on out, any references to solar generated power refers exclusively to PV systems unless otherwise stated.

The main advantages of terrestrial (i.e. ground-based) solar power are:

- Renewable: As discussed above, solar power is a renewable energy source, utilizing the Sun's natural radiation.
- Reduce d Dependence on Fossil Fuels: The direct generation of electricity from solar energy requires no fossil fuels.[11]
- Environmental Advantages: Solar power generation is far more benign than conventional power systems from an environmental standpoint. The production process is emission-free with the exception of waste heat.
- Scalability and Modularity: Solar power is generated via PV cells which are inherently modular. Solar power output is therefore easily scalable.
- Local Location Flexibility: A direct consequence of solar power scalability is the flexibility of solar power facility locations. Solar power production can be installed in small scale at the end user's site (e.g. home) or in remote locations.
- Longevity/Reliability: Due to the relative simplicity of PV technology, solar power systems are highly reliable with lifetimes around 10-20 years due to grad-

[11] The manufacturing of relevant infrastructure is excluded.

ual performance degradation. Combined with the low risk nature of PVs this means that support infrastructure and any oversight or maintenance is minimal.

The main disadvantages are:

- Environmental Disadvantages: Substantial solar power generation requires extensive and dedicated land use and emits thermal pollution (see Table 2.1 above). Solar cell manufacturing also generates its own atmospheric pollution.

- Variable Availability: Since solar power generation is inherently dependent on incoming sunlight, it does not provide uninterrupted power. Weather conditions, like clouds and precipitation, and nighttime render solar power systems useless.

- Global Location Inflexibility: As mentioned above, solar power systems require sunlight and land use. Facilities must therefore be placed on clear and accessible land and preferably in areas of the world where poor weather is minimal.

- Cost: Though solar energy costs continue to decrease, it has still not reached grid parity (i.e. comparable to current utility prices). This is further discussed below.

Of these disadvantages, the variable availability is the chief limiting factor of solar power use. But with such tremendous advantages, solar power is one of the most appealing energy sources. Only in the last decade, however, have technological advancements and increased interest in renewable energies made it possible for solar power to become sufficiently cost-effective for widespread use, and it is now heralded as one of the most promising new sources of energy. In fact, the U.S. solar industry has grown by 670% since 2006 (69% in the last year alone), making it the fastest growing energy sector and one of the fastest growing sectors in the economy.[12] Total solar energy (SDS and PV) consumed in the U.S. in 2010 was 32 TW·h, a 60% increase from 2006.[13]

Due to increasing PV cell efficiencies, major manufacturing technology improvements, and economies of scale, solar panel prices have dropped by 30% in the last year and solar power is projected to be the largest source of new electric power in the U.S. by 2014.[14] Table 2.2 lists the estimated utility costs for several different energy sources in 2010.

Table 2.2: Utility Costs of Electricity in the U.S. in 2010

Energy Source	Lifetime Cost ¢/kW·h [1]
Hydropower[2]	5.0
Natural Gas	6.4
Wind Turbine	7.5
Conventional Coal	8.0
Nuclear	10.0
Solar PV	**15.0**
Solar Thermal	16.0

[1] Subsidies not included.
[2] Hooverdam only.
Source: Zweibel, 2010.

The momentum of the solar energy market ensures the continued spread of solar power infrastructure and the advancement of relevant technologies. Potential applications of solar energy are thus made attractive by a firm base of economic, infrastructural, and technological support.

One such conceptual application is space-based solar power (SSP), commonly

[12]Solarbuzz 2010, *Marketbuzz*
[13]U.S. Energy Information Administration (EIA), Monthly Energy Review (MER) March 2011, DOE/EIA-0035 (2011/03) (Washington, DC, March 2011)
[14]Source: SEIA

referred to as the Space-based Solar Power Satellite (SSPS). In essence, it is a solar power facility situated in Earth orbit: sunlight is collected in space and used to form an electromagnetic beam, or *power beam*, that is transmitted to a ground station. The collected energy is then converted to electricity and fed into the local power grid for customer use (see Section 2.2). Due to a number of constraints (see Sections 2.1.7 and 3.2), the power beam is generally either an infrared laser beam or ordinary electromagnetic waves situated in the microwave region (300 MHz-300 GHz). The SSPS system thus consists of the space segment - the satellite, which houses the solar arrays and the transmitter - and the ground segment - the receiver (called a *rectenna* for the case of a microwave beam), linked to the power grid. The core features of this system are the use of renewable extraterrestrial solar energy (space-based solar power), and the wireless transfer of this energy.

PV cells are ideal for space use because they have long lives, high reliability, and are operationally simple and safe (e.g., no moving parts and no vibrations).[15] And space-based PV solar arrays are preferable over terrestrial PV arrays for the following reasons:

- Increased Solar Power: As discussed previously, there is up to 32% more solar energy per unit area in space than on the ground.
- Environmental Advantages: Since the SSPS is in space, no land is used and waste heat can be vented with no environmental consequences. Additionally, no dust or dirt is accumulated on the arrays.
- Availability: 1) Solar power can be generated regardless of atmospheric coverage (e.g. poor weather), and 2) By choosing an appropriate orbit, the solar arrays

[15] [5]

can be exposed to sunlight longer than ground-based systems; sun-synchronous orbits, for instance, allow for 24 hour exposure, and hence continuous solar power generation.

Wireless energy transfer, referred to as beamed power transmission for the case of SSPS, is technologically accomplished using *Wireless Power Transmission* (WPT). WPT offers many advantages over traditional means of point-to-point energy transfer:[16]

- Low Mass: This is a wireless system so there is no mass (e.g. wires) or transport vehicles required between the energy source and receiver.
- High Transmission Speed: Energy is transferred at the speed of light rather than conventional electron current speeds.
- Low Transmission Losses: Loss of energy during transfer is null in space vacuum and minimal in Earth's atmosphere if an appropriate transmission frequency and a suitable receiver site are chosen.
- Flexible/Global Power Availability: 1) Direction of energy transfer can be quickly changed, and 2) Energy transfer can be done between different environments or terrain, as it is independent of differences in gravitation potential between end points. Together, these two features mean that power can potentially be delivered anywhere on Earth where the SSPS transmitter has line-of-sight and there exists a receiver.

[16]Brown, 1992

Together, space-generated solar power and beamed power transmission form the core of the SSPS system. As an integrated system, however, there are some disadvantages:

- Complexity: The SSPS system requires the interfacing of many technically complex elements. This introduces reliability concerns and necessitates a high degree of operational oversight.
- Inaccessibility: Since the SSPS is situated in orbit, access by crew or machine is difficult. This makes the SSPS a high risk system and reliability must be high.
- Environmental Risks: There are several environmental risks related to the power beam that take the form of design constraints. High irradiation levels (i.e. power densities) and particular frequencies can have negative impacts on the atmosphere or local biota. These are further discussed in Section 2.1.7. Also note that terrestrial pollution associated with parts manufacturing is unavoidable.
- Cost: Though only theoretical cost estimates exist, it is clear that that any implementation of space-based solar power will require a considerable initial investment to construct the space and ground infrastructures and a establish an efficient energy link between the receiver and the power grid (i.e. a new smart grid and proper energy storage). It remains to be seen whether or not the system is profitable in the long-run and competitive with the lifetime costs of existing power systems.

Despite these drawbacks, the SSPS system has the potential for low cost, low mass, high power delivery over immense distances with minimal energy loss.

There are a number of markets for such a system, some of which are listed below:

- Remote Locations: Power can be delivered to ground sites in harsh, remote locations, like research centers near the poles or habitations on small islands. These are areas where conventional transmission lines cannot be practically installed and available power is generally limited to diesel fueled generators. Small receivers at these sites could provide enough power for basic amenities like drinking water, heat and telecommunication, thus encouraging local development.

- Peak Power Supplementation: At a mid scale level of operation, space-based solar power can be used to supplement the existing power grid during peak loading times in the day (around 7-10 am and 5-9 pm). The SSPS can be designed to deliver scheduled on-demand power to any specific location(s), making it an ideal choice for such a use.

- Large Scale Power Generation: Due to its modularity, the SSPS system could ultimately be used to generate power on the order of giga-watts. This is enough to power small cities or large towns, and with multiple satellites and receivers this capacity could be increased even more. Due to limited transmission availability and reliability, however, space-based solar power could never be the exclusive source of energy for these communities.

- Space-Space: The SSPS could also be used to transmit power to other satellites in orbit. This would increase satellite lifetimes or mission durations and would be especially beneficial for small satellites that don't have large arrays for solar arrays or internal power systems. This application would likely require a laser power beam in order to provide high power densities to the relatively small receivers onboard target satellites. This has its own risks and technological difficulties, however, as discussed in Section 3.2.

The potential of space-based solar power was recognized by Dr. Peter Glaser in 1968, when he proposed the first concept of the SSPS. It wasn't until the late 70's, however, that a major concept study was performed under the joint effort of the DOE and NASA. The study culminated with the publication of the Solar Power Satellite (SPS) Reference System Report[17] in 1979. The design was centered around 60 enormous satellites in geostationary Earth orbit (GEO), each capable of delivering 5 GW of power to the US power grid (Figure 2.4).[18] This exhaustive study concluded that, using existing methods and technologies, the "cost-to-first-power" would be more than \$370 billion (FY 2011). It was therefore recommended that the SSPS concept be re-evaluated in a decade, when relevant technologies would hopefully be further developed. This, however, did not happen until 1995.

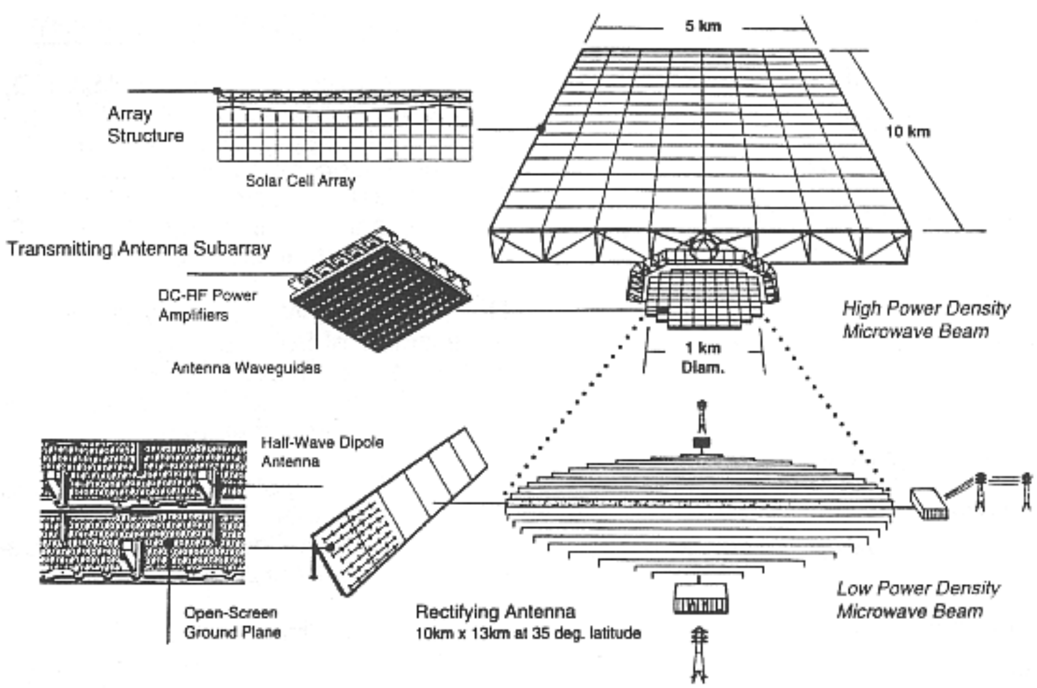

Figure 2.4: 1979 Reference SSPS concept by NASA ([4]): 5 GW output, GEO

[17][4]

[18]Each satellite had 55 km^2 of solar arrays and a 1 km diameter transmitter.

By the end of the 1980's, there was little active interest in the SPS concept. Research and development in the U.S. was focused mainly on solar power systems, with relatively minor studies on SSPS potential applications through the early 1990's. There was, however, some international interest; Japan conducted significant SSPS research, including WPT flight experiments[19] and there was some activity in Europe and Canada.[20]

Then, in 1995, NASA carried out the "Fresh Look Study". The goal of this effort was to determine the viability of space-based solar power given the technological advancements since the 1970's. A number of new SSPS configurations were examined and found to not only be promising candidates, but also more technically feasible than the original 1979 concept. Still though, the scope of the concept was daunting.

NASA followed up this study with the SSP Concept Definition Study in 1998. The principle goals of this effort were to validate the findings in 1995 and produce "strategic road maps for the possible development of SSP [space-based solar power] technologies."[21]

The SSP Exploratory Research and Technology (SERT) Program was conducted from 1999 to 2000, and sought to further define systems concepts and key challenges while initiating R&D corresponding to the strategic road maps identified in the Concept Definition Study. This program created a number of projects that involved a wide range of participants, including multiple NASA centers, universities, laboratories and international organizations.

Immediately following this, NASA lead the SSP Concepts & Technology Maturation Program (SCTM) until 2002. Key conceptual and technological features of the

[19] See [5] for instance.
[20] [40]
[21] [40]

SSPS were further developed and a number of high-risk research studies associated with SSP challenges were performed.

Despite its long history of development, the SSPS remains a futuristic concept. This is due to 1) a lack of political and programmatic motivation, and 2) a need to improve and demonstrate key technologies and their integrated functionality in the SSPS.

The nature and scale of the SSPS concept demands financial and programmatic support on a level only achievable by government involvement, and hence, strong political support. In light of public awareness of climate concerns, renewable energy research has been firmly backed in the political arena. However, the radical nature and exceptionally high developmental costs of the SSPS concept make it a risky endeavor and it is not widely seen as a practical solution to energy-related concerns. This makes the SSPS concept politically unappealing. This extends to the programmatic level; the large-scale ground and space infrastructure required to implement a SSP system of practical use necessitates what is generally seen as a prohibitive level of planning and resource allocation.

As discussed above, it is not the technical issues that most challenge the SSPS concept. In fact, the SSPS system is technically feasible and has been for several decades. However, like all space projects that involve ground-breaking technologies, the SSPS concept needs to be proved out with a *technology demonstration mission* before being developed and implemented at full scale. Deep Space 1, for example, was launched in 1998 as the first in-flight demonstration of ion propulsion technology. The mission's success demonstrated the capabilities of these engines and set the standard for future asteroid mission designs.

The drivers for a technology demonstration mission are 1) to demonstrate and advance the *technology readiness levels* (TRLs) of key technologies and subsystems, and 2) to demonstrate or assess the practicality and viability of the concept.

An end-to-end[22] technology demonstration mission is required for the SSPS concept. In particular, the *gamechanging* technology[23] of Wireless Power Transmission is the least proven SSPS-related technology and must be successfully demonstrated both independently and as an integrated subsystem with the solar panels and satellite structure. The measured performances and costs of the individual subsystems and of the integrated SSPS can then be used to evaluate the viability of the system. Proving out individual technologies and integrated subsystems is done using the concept of a TRL scale that identifies the current maturity of the technology or subsystem and defines the criteria for its advancement to a mission-ready level. The TRL concept and scale is discussed in Section 2.1.4.

Demonstrating the key technologies of the SSPS concept through a technology demonstration mission helps to overcome the many strategic hurdles associated with its viability and eventual synthesis. Though these hurdles are primarily technical, a successful technology demonstration mission would also greatly boost political and programmatic motivation. Together, these issues drive the need for a technology demonstration mission.

The strategic technical hurdles are discussed in detail in Section 9.1 along with their associated technologies.

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[22] The end-to-end process begins with the solar energy collected and ends with the power outputted from the receiver (before modifications for grid use).

[23] "Gamechanging" refers to the central role of an innovative technology in the pioneering of a new concept.

Stakeholders Investing in energy technologies is often the government's purview, as is evidenced by the many energy-related policies and mandates in the U.S. As discussed previously, the SSPS concept, like most potential renewable energy solutions, is a high risk endeavor that requires a massive level of long-term financial, material, and programmatic responsibility. This situation is highly undesirable to the commercial sector as profit is uncertain this early on. The federal government, however, has the capabilities and motivations to carry out this undertaking, and hence it is the initial stakeholder. Political stakeholders exist within the government as well. In the face of climate concerns and diminishing resources, renewable energy research is a positive step for the future and it is the role of these politicians to have the necessary concern and foresight to support this.

This stakeholder structure is analogous to the development of the commercial satellite industry. After the launch of Sputnik I in 1957, NASA and the Department of Defence (DoD) were directed by Congress to begin developing and experimenting with communications satellites. This initial stage of government investment was motivated by the "benefits, profits, and prestige associated with satellite communications,"[24] and supported by the underlying politics of the early Cold War. This oversight also enabled direct access to existing government infrastructures and technologies critical to the project, namely launch sites and rockets developed by NASA and the DoD.

By 1964, six communications satellites had been successfully operated in space, two by AT&T, and four by companies that had been contracted by NASA. Then in April of 1965, COMSAT launched its first satellite and so began the commercial takeover of communications satellites and the beginning of a multi-billion dollar industry.

[24] http://history.nasa.gov/satcomhistory.html

Similarly, future stakeholders of the SSPS concept would be in the energy industry of the commercial sector.

The need described above, and the subsequent goals and objectives are all presented from these stakeholders' perspectives.

2.1.2 Goals

Based on the need for alternative energy sources and the identification of the SSPS concept as a potential solution, the fundamental goal of the project can be stated as:

Determine the viability of the Space-based Solar Power Satellite (SSPS) as an energy source.

The viability of the SSPS concept is ideally determined by the quantitative evaluation of its advantages and disadvantages; for example, one could ask whether the cost per unit power delivered to the grid is competitive with existing energy systems, given the system's end-to-end efficiency, power availability, cost, risks, etc. Of course, all of these factors are dependent upon the system architecture and design, and hence the need for rigorous concept development and systems engineering.

This fundamental goal can be broken down into the following three specific goals:

1. ***Investigate and demonstrate gamechanging technologies for the SSPS***
2. ***Develop and operate an integrated end-to-end technology demonstration mission for the SSPS concept (SSPS-TD).***

3. ***Evaluate the SSPS as a potential power system.***

This study will be primarily focused on meeting Goal 2 through a top-level design process that concentrates mainly on the space-based mission segments.

2.1.3 Objectives

The following top-level objectives have been identified as meeting the mission goals.

1. Demonstrate and validate in-orbit Wireless Power Transmission (WPT) of $>$ 100 kW.

   This objective contains the following tasks that seek to validate and refine theoretical predictions for in-orbit WPT performance:

   - Measure and track power delivered over seasonal and atmospheric variations, taking into account orbital characteristics.
   - Investigate and confirm safety and environmental constraints and effects through the measurement of incident irradiation levels in the atmosphere and on the ground and any observed consequences.

   The WPT element is the central feature of the SSPS that allows collected solar energy to be converted and transmitted to Earth for grid use via electromagnetic waves. Currently, WPT has only been demonstrated at small scales and over relatively small distances ($<$100 km). Even as a technology demonstration mission, this mission must demonstrate the capability to transmit practical amounts of power from orbit, and do so safely and efficiently.

   Derivation: Goal 1

2. Evaluate and implement available gamechanging technologies into the SSPS design.

   Gamechanging technologies, like WPT and ultra lightweight solar cells, are integral to the functionality and viability of the SSPS system. By considering these technologies in the SSPS-TD design, projections of future full-scale operations are made possible (i.e. performance characteristics and viability), while allowing for comparisons with older SSP designs and conventional power systems (further viability analysis).

   Derivation: Goals 1 and 3

3. Demonstrate and evaluate the functionality and performance characteristics of the SSPS-TD system across key interfaces.

   As a technology demonstration mission, the SSPS-TD must demonstrate how the enabling SSP technologies interface. Though this mission will not be sized to generate the power required for full scale implementation, it must be fully functional and consist of the essential, properly interfaced SSPS elements so that a full scale system can be evaluated. These key interfaces will be defined in Chapter 6.

   Derivation: Goals 2 and 3

4. Evaluate the SSPS end-to-end efficiency.

   This measurement characterizes the performance of the SSPS design and is defined by the ratio of the output power at the grid to the input power from the solar arrays. This is critical to the evaluation of the SSPS as a viable power system and in future designs. Note that by combining the efficiency with a reliable cost estimate allows for predictions on the potential costs of a larger-scale SSPS system and the market viability of the concept as an energy source. This "business-case" analysis is beyond the scope of the study, but an independent cost analysis is performed.

Derivation: Goal 3

2.1.4 Mission Description

The SSPS-TD mission is a technology demonstration mission for space-based solar power (SSP). Per the mission goals (Section 2.1.2), it is a flight quality prototype (end-to-end SSPS system) that will promote a comprehensive understanding of SSP in order to assess its viability as an energy source. The mission's primary purpose in this regard is to prove out a number of technologies crucial to the SSPS concept, foremost among them being wireless power transmission (WPT), and evaluate the performance of these technologies across the system interfaces (Section 2.1.3).

The technical goal of SSP is to convert in-space solar radiation into grid-compatible electricity on the ground, i.e., perform a series of conversions between different energy types. In order to do this, SSPS-TD consists of a space segment and a ground segment. The space segment is the orbiting satellite which contains the solar arrays and microwave transmitter (as well as the other standard subsystems). The ground segment is the receiver which is connected to the power grid. See Chapter 3 for a further discussion of the architecture and Chapter 4 for a Product Breakdown Structure (PBS).

Nominal operation can be summarized as a 5-step process: 1) Sunlight (solar energy) is absorbed by the PV cells in the solar arrays and converted to electricity (DC), 2) This electrical energy is fed into the transmitter where it is first converted into electromagnetic waves via the appropriate generators (laser or microwave), and then guided into a directed power beam at the transmitter aperture (i.e., antenna), 3) The power beam travels through the Earth's atmosphere and then strikes the receiver, 4) Elements in the receiver subsystem convert the incident microwaves into electricity (DC) whereupon it can be, 5) modified for grid compatibility (e.g., DC to AC and voltage step-up) and fed into the local power grid. Note that solar power production and power beam transmission are dependent on solar and receiver line-of-sight, respectively, and though the solar arrays can function independently of the transmitter, the reverse is not true as the transmitter requires a continuous supply of energy when online. For more details on the operational stages and relevant component functions see Section 2.2.

The functions performed by the SSPS-TD mission ensure that proper technology demonstration is achieved. As discussed in Section 2.1.1, the purpose of this technology demonstration mission is to advance the maturity of technologies vital to SSP. The current state of maturity of these technologies exists somewhere on a technology ladder that begins with an initial idea and ends with full-scale implementation; the goal is then to move up this ladder.

This progression does not necessarily occur in a linear or predictable manner; there exist identifiable milestones along the way. Technology readiness levels (TRLs) represent these milestones and act as the ladder rungs, tracking and describing the increasing levels of a technology's maturity.

The SSPS-TD mission will use the TRL scale commonly used by NASA and displayed in Table 2.3. The accompanying descriptions indicate the achievement of that level and thus serve as "graduation" criteria.[25]

In order to meet these criteria, there may exist sets of sub-criteria that are

[25] [7]

Table 2.3: Technology Readiness Levels

<table>
<tr><th>TRL</th><th>NASA Standard Mission[1]</th><th>Terminology[2]</th></tr>
<tr><td>1</td><td>Basic principles observed and reported</td><td rowspan="9">Breadboard - bench-top implementation in which all key mechanical and electrical interfaces are simulated but form, fit, and scale are not considered.

Prototype - initial implementation having the correct form, fit, function and scale but not necessarily flight quality.

Environment - The spectrum of operating conditions, interfaces, and design conditions to which the technology adanve will be exposed both during testing and during flight operations.

Relevant environment - Subset of ”environments” defined to be that environment(s), operation condition(s), or combinations thereof that most stress the technology advance and is consistent with that expected in the spectrum of likely initial applications.</td></tr>
<tr><td>2</td><td>Technology concept and/or application formulated</td></tr>
<tr><td>3</td><td>Analytical and experimental critical function and/or characteristic proof-of-concept</td></tr>
<tr><td>4</td><td>Component and/or breadboard validation in laboratory environment</td></tr>
<tr><td>5</td><td>Component and/or breadboard validation in relevant environment</td></tr>
<tr><td>6</td><td>System/subsystem model or prototype demonstration in a relevant environment (ground or space)</td></tr>
<tr><td>7</td><td>System prototype demonstration in a target/space environment</td></tr>
<tr><td>8</td><td>Actual system completed and ”flight qualified” through test and demonstration (ground or flight)</td></tr>
<tr><td>9</td><td>Actual system ”flight proven” through successful mission operations</td></tr>
</table>

= High Risk, = Low Risk (if implemented at this TRL)

[1] NASA Systems Engineering Handbook, 2010

[2] New Millennium Program (NMP) Technology Readiness Descriptions, May 2003

unique to the type of mission. In particular, technology demonstration missions differ from standard science missions in that some sort of breadboard or prototype is the end product to be flown. This changes what the ”actual system” is and pushes the TRL scale back in a sense. At TRL 5, for instance, a performance predictable model (i.e., simulation) built from experience gained on the ground may be considered a sufficient demonstration of operation in a ”relevant environment”. It is therefore important to develop clearly stated definitions of what constitute each TRL achievement with responsible Project Management at the beginning of the project lifecycle.

The New Millennium Program (NMP) TRL definitions are used as a reference for this mission. NMP is NASA's low-cost, experimental spacecraft program, whose primary goal is to validate new technologies for future missions, thus providing "a critical bridge from initial concept to exploration-mission use."[26] Since NMP was created specifically to perform technology demonstration missions, its TRL definitions are perfectly suited for the SSPS-TD mission.

TRLs are applied independently at both a system and sub-system[27] (e.g. specific component or technology) level, as can be seen by the terminology in the TRL definitions. The TRL scale is interpreted differently at each of these hierarchical levels. At the system level, TRLs describe the state of technological maturity of the overall mission concept, which in this case is the application of SSP. It can begin as early as TRL 1 but the system is not physically evaluated until all subsystems and components (e.g. solar cells, transmitter generators, receiver sub-elements) have achieved sufficient TRLs (i.e., TRL 5) to allow for a system-wide implementation (beginning at TRL 6). Though the system is an integrated application of technologies that are at higher (and potentially different) TRLs, the system TRL is independent of these individual TRLs and must be assessed all over again according to the TRL scale.

For the SSPS-TD mission, the goal is to reach a system TRL 7. This particular achievement assures that "system engineering is adequate, that trans-interface interactions are adequately modeled and understood, and that in-space operation at

[26]nmp.nasa.gov

[27]*Sub-system* refers to the products in the mission found below the *subsystem* level in the system hierarchy (Chapter 4).

the appropriate scale is both as expected and as predicted."[28] TRL 7 is especially important for technology demonstration missions like this one where many of the technologies and subsystems are both "mission critical and high risk". The SSPS-TD can therefore be viewed as a prototype for the SSPS concept.[29]

The actual launch and demonstration of this prototype, however, cannot occur until at least TRL 5 is reached for all subsystems and components. TRL 5 marks the point at which actual implementation can begin with a technology. For SSPS-TD, this means that all critical technologies at the sub-system level, like the solar arrays and WPT element, have been integrated appropriately so that total application, i.e., SSP, is ready to be tested in its relevant environment. At the system level, this occurs in TRL 6, but since the mission takes place in the target environment (space) and involves a system prototype, this level will be met concurrently with TRL 7, and TRL 6 is effectively skipped over. However, though the system jumps TRLs, the sub-systems do not. They are tested independently on the ground (satisfying TRL 6) or may have been tested onboard past missions (satisfying higher TRLs). Certain types of PV cells, for instance, have long been in use onboard space missions and are confirmed at TRL 9. This means that the system involves the integrated application of higher TRL technologies with lower ones (which, as mentioned earlier, has no bearing on the system TRL).

SSPS-TD is responsible for precisely this system level of integrated testing. Confirmation of TRL 7 occurs with the successful achievement of the mission objectives (Section 2.1.3) that, together, prove out the SSPS concept and its technologies.

[28] NMP Technology Readiness Descriptions, May 2003

[29] Technically, the SSPS-TD is not designed to full SSPS scale and so, by strict definition, not a prototype. However, the major technical components (e.g., solar arrays, microwave generators, and transmitters) are inherently modular and so the design allows for relatively simple extensibility to full scale, thus allowing SSPS-TD to act as a valid prototype.

The specific TRLs of SSPS-TD mission technologies is discussed in Chapter 3.

2.1.5 Authority & Responsibility

As discussed in Section 2.1.1, SSP is a high risk space energy technology and the primary stakeholder is the government. The SSPS-TD mission therefore falls under the authority of the federal government, namely NASA and the Department of Energy (DOE).

The project will follow a cooperative and interdependent organizational structure similar to the NASA-NOAA[30] joint arrangement for the development and operation of the current U.S. weather satellite fleet.[31] NASA will develop the architecture, and construct and launch the satellite, while the DOE will provide top-level requirements and oversee nominal operations. In particular, the DOE is responsible for generating requirements associated with the power output at the receiver-grid interface (e.g. quantity, demand schedule, electrical current properties). Requirements flowdown is then performed by NASA, enabling the comprehensive design and construction of the SSPS system. Actual subsystem or component development and fabrication may be contracted out as well. During nominal operations, the DOE will control the flow of power across both the satellite-receiver and receiver-grid interfaces, while NASA will monitor the more technical aspects of the system, maintain the satellite subsystems, and can perform any necessary maneuvers or alignments.

Data analysis pertaining to the future viability of the SSPS system can be performed by DOE labs and third-party organizations (e.g., commercial stakeholders or outside contractors).

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[30]National Oceanic and Atmospheric Administration
[31]http://www.nesdis.noaa.gov/SatInformation.html

2.1.6 Assumptions

Top-level assumptions that shape the mission's baseline architecture and general concept of operations are listed below. Specific design or subsystem related assumptions are stated in later sections. Note that these assumptions may be revisited during the project lifecycle due to the iterative nature of systems engineering and project development.

1. The SSPS-TD will launch within the next 15 years.

   This mission is motivated by current energy needs and designed within current technological capabilities that might otherwise be obsolete in the far future. In other words, energy solutions or technologies available for SSP in the far future cannot be predicted and so designing a mission of this nature that won't be launched for a long time is of little value. 15 years is considered an appropriate window of time to avoid this problem and maintain mission relevancy.

2. SSPS-TD will launch from Cape Canaveral Air Force Station (CCAFS) or Vandenberg Air Force Base (VAFB).

   This assumption exists because 1) SSPS-TD is a U.S. mission, and 2) it helps to refine the baseline architecture and simplify future trade studies. The specific launch site is stated in Section 3.7, after the mission orbit profile is determined. Some relevant launch site information is given in Table 2.4.

Table 2.4: Launch Site Information

	Lat	Long	Inclinations Served
CCAFS	28.47° N	80.56° W	28° - 57°
VAFB	34.77° N	120.60° W	51° - 145°

Source: astronautix.com

3. The mission will have a baseline operational lifetime of 1 year.

   Technology demonstration missions are typically designed to operate in the range of 6 months to a couple years. Similarly, SSPS-TD must perform sufficiently long to attempt the completion of its mission objectives. In particular, it must operate long enough to prove out the concept, i.e., test and demonstrate the SPS technologies to the appropriate TRL. This assumption is also necessary in order to guide the design of various subsystems and components in light of their associated reliability and performance characteristics.

4. The receiver will be located in the continental U.S.

   This assumption facilitates the design, construction, and operation of the receiver. In particular, resource allocation, supply chain management and operational oversight are all made easier by this centralized location. It also satisfies the need to connect to an existing power grid, something not so easily done in a globally remote location. Furthermore, architecture trade studies are simplified by reducing the number of receiver site options. See the results in Section 3.4 for the specific location of the receiver.

5. The SSPS-TD mission will use an existing communication network for all space-ground data transfers.

   As a technology demonstration mission, it is imperative that ground control have 24 hour communication access to SSPS-TD. And since a specific orbit has yet to be determined, such accessibility may require multiple ground stations and auxiliary satellites. In order to avoid the need to design and build this new infrastructure, the mission will use an established communication network like TDRSS (Tracking and Data Relay Satellite System), which can offer near global coverage by relaying data through a satellite network connected to a ground terminal (see Section 2.2 for more details).[32] This ensures the reliability of such a critical element of the mission and allows all design efforts to focus directly on meeting the mission objectives. Furthermore, mission costs are reduced and schedule constraints (e.g., Assumption 1) are more easily met.

2.1.7 Constraints

Top-level constraints are listed below:

1. Type: Regulatory Constraint

   For standard electromagnetic radiation (e.g., incoherent, uncollimated), the WPT power beam is restricted by international regulations to operate in the ISM band, i.e. narrow frequency intervals located between 6.765 MHz and 246 GHz (wavelengths 44.3 m and 1.2 mm, respectively).

   Standard electromagnetic WPT at large distances involves a high power-density beam with a wide diffraction pattern so it is important that it does not interfere with communications, GPS or other bands in use. The ISM band is internationally reserved for industrial, scientific, and medical (ISM) use unrelated to communication, and is therefore appropriate for SSP WPT.

2. Type: Environmental/Technological Constraint

   The WPT power beam is restricted to the microwave or infrared bandwidth due to 1) atmospheric and meteorology-related transmittance, and 2) current WPT technologies.

[32] https://www.spacecomm.nasa.gov/spacecomm/programs/tdrss/default.cfm

The WPT power beam must operate at a frequency that allows it to pass through the atmosphere relatively unattenuated (see Figure 2.5). Frequency choices are furthermore restricted by available technologies. Microwaves (300 MHz to 300 GHz) suffer minimally from atmospheric absorption, even with cloud cover, and have long been in use for space-ground transmission (e.g. radar, GPS, radio astronomy). Several bandwidths in the infrared are also suitable for trans-atmospheric WPT, and are associated with laser-based WPT (due to available technologies).

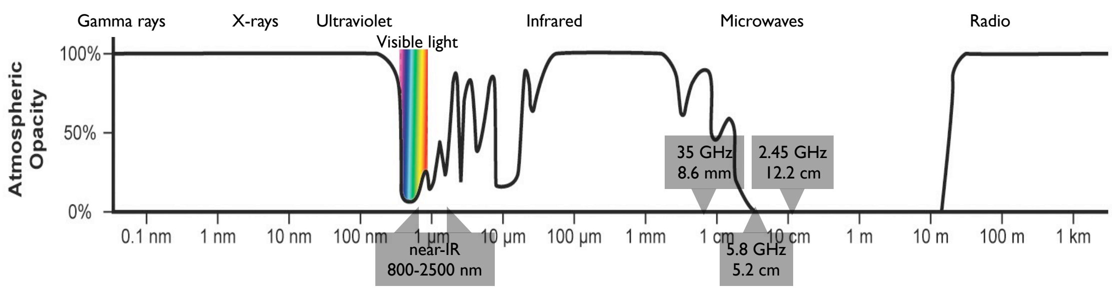


Figure 2.5: Atmospheric electromagnetic opacity. Adapted from NASA.

Due to a number of complications and disadvantages to space-ground laser transmission (see Section 3.2), WPT technology development has focused mainly on microwave generators and receivers for the transmitter and rectenna, respectively. Components for the SSPS WPT element are thus most readily available in this bandwidth, reducing developmental costs and risks and helping to maintain a launch date within 15 years. The most common frequencies of interest to microwave WPT are 2.45, 5.8 and 35 GHz (wavelengths: 12.2, 5.2, and 0.86 cm, respectively).[33]

[33] Mankins (2002), DOE/NASA (1978) for instance

The most common frequencies of interest to laser WPT are in the near-Infrared spectrum (800-2500 nm), where several windows of relatively low atmospheric and meteorolgy-related absorption exist. A particularly suitable window for both atmospheric transmission and existing technology is between 840-845 nm.[34]. Two other popular frequencies often considered for SSP applications are 305.9 THz (wavelength: 980 nm)[35] and 371 THz (wavelength: 808 nm).[36]

Atmospheric effects on WPT are further discussed in Section 3.2. WPT technology is further discussed in Chapter 3.

3. Type: Environmental/Safety Constraint

   In order to avoid potentially harmful environmental impacts, a WPT power beam operating in the RF range (300 kHz to 300 GHz) is not to exceed an ionospheric power flux threshold (W/m^2) given by (Duncan, 1981):

$$I_{TSI} = (0.43)(10^{12}/N_e)(T_e/1000)^4(f)^3$$

   where N_e is the electron density (m^{-3}) and T_e is the electron temperature (K), in the ionosphere, and f is the frequency (GHz) of the WPT power beam.

   This power density threshold is associated with the triggering of thermal self-focussing instabilities (TSI) in the ionosphere, the most important atmospheric effect of SPS WPT. This nonlinear process is a consequence of collisional heating and generates positive feedback, leading to further instabilities that change ionospheric plasma densities and the local chemical content. This generates potentially harmful atmospheric effects and significant local telecommunication disruptions.

[34] [15]
[35] [41]
[36] [32]

This threshold value is computed in Section 3.3.3 when specific frequency choices are studied.

Environmental effects due to WPT are discussed in Sections 3.2.

4. Type: Regulatory/Safety Constraint

The ground-incident WPT power beam is not to exceed a power density of 100 W/m^2 outside of or near the edge of the receiver for ordinary electromagnetic radiation between 10 MHz and 100 GHz. Preliminary recommendations exist for WPT laser beaming.

Legally-defined radiation exposure upper limits are listed in Table 2.5.[37]

Table 2.5: U.S. Radiation Exposure Upper Limits

Limit Type	**Microwave**	**Laser**[1]
General Population/Cautionary Safety Limit	$< 1 - 10$ W/m^2	n/a[2]
Employee Safety Limits/Incidental Exposure or Viewing	< 100 W/m^2	< 10 W/m^2
Typical Eye Safety Limit for Long Durations (> 10 min)	n/a	< 25 W/m^2
Typical Eye Safety Limit for Short Durations (> 10 s)	n/a	< 50 W/m^2

[1] Specific safety limits vary largely depending on the laser frequency
[2] No defined limit, see discussion below.

U.S. Occupational Safety and Health Standard 1910.97 [38] dictates that for normal environmental conditions and nonionizing, intermittent or continuous incident electromagnetic radiation between 10 MHz and 100 GHz, the radiation protection guide is 10 mW/cm^2, or 100 W/m^2. This limit becomes a constraint on the power density of a microwave WPT beam near the edge and outside

[37] [41]
[38] www.osha.gov/

of the receiver (as computed from the diffraction pattern), where the risk of exposure to facility personnel is greatest. For the general population, an order of magnitude less is recommended.

For a laser power beam, the safety limits are far too low to avoid. In light of this, [5] recommends that the power density not exceed 1000 W/m^2 so long as personnel entry in the receiver site is heavily monitored and controlled, and animal eye injury (including humans) can be neglected. In this case, however, there may be substantial local ecological changes. If there is a risk of laser irradiation on animals (including humans) then the power threshold density should be less than 0.01 W/m^2 to protect their eyes against retinal injury. In a sterilized environment with no living organisms near the beam, the power density is limited only by environmental effects (which are poorly understood). These constraints define safety protocol and potential safety zone sizes around the receiver.

Safety issues are further discussed in Section 3.2.

5. Type: Form & Fit/Operational Constraint

   SSPS-TD must be capable of launching onboard a U.S. launch vehicle.

   From assumptions (1) and (2), SSPS-TD must launch on an existing U.S. launch vehicle because the mission is sponsored by the U.S. government (Section 2.1.5). In anticipation of considerable system mass and size, the largest and heaviest payload-capable launch vehicles are the most likely choices.

   Some relevant launch vehicle specifications for the two likeliest choices are given below in Table 2.6.

Table 2.6: Launch Vehicle Performance Summary

	Delta IV Heavy	Atlas V 500
Max Payload		
LEO[1]	23,975 kg	20,520 kg
SSO[2]	19,265 kg	n/a
GEO	6,276 kg	3,890 kg
Max Fairing Dimensions	22.4 m x 5 m	23.4 m x 5.4 m
Launch Sites	CCAFS, VAFB	CCAFS, (VAFB)[3]

Source: [30]
[1] Circular 185 km, 28.7°
[2] Sun-synchronous 833 km, 98.7°
[3] No present capability. Future missions can launch from VAFB with appropriate infrastructure additions.

6. Type: Form & Fit/Design Constraint

   SSPS-TD satellite deployment must utilize only one launch.

   Because this mission is a technology demonstration, it is necessarily a small-scale operation and both cost and complexity need to be minimized. As evidenced by the construction of the International Space Station (ISS), multiple launches and in-orbit construction demand immense resources, finances, and coordination, not to mention the design challenges associated with such a system. This is clearly inappropriate for a technology demonstration mission and it is important to establish this constraint given the large sizes of previously researched full-scale SSPS concepts.

   This constraint is the most important determinant of the mission feasibility in terms of the system size, and is the major driver for this design feasibility study, as discussed in Sections 3.6 and 3.7.

2.2 Concept of Operations

The SSPS concept is highly complex and developing the SSPS-TD mission profile involves many fundamental decisions that define the baseline architecture. The concept of operations (ConOps) is thus limited to the most general formulation of the mission that has been generated up to this point. This is valid since the current mission formulation is based on the mission scope, and so satisfies the mission objectives, assumptions and constraints.

Since the mission architecture has yet to be finalized, some of the details in the ConOps represent *possible* design solutions in order to provide a reference "big picture" of the mission, and are noted as such. In fact, the conceptual and somewhat hypothetical nature of the ConOps is what allows the architectural trade studies in Chapter 3 to be identified. Subsequent iterations of the systems engineering process (in the design phases of the project lifecycle) will update the ConOps with these trade study results and any other design choices or changes. The ConOps is thus a living document that matures throughout the project lifecycle and ultimately informs the detailed operations plan of the mission.

2.2.1 Concept of Nominal Operations

A *conceptual* top-level pictorial representation of the nominal operations for the SSPS-TD mission is shown in Figure 2.6. This graphical interpretation is perhaps the most helpful tool in communicating the ConOps during the early design phases, helping to generate requirements, define interfaces, and initiate functional analysis of the system, among other systems engineering practices. The word *conceptual* is used to remind the reader that the configuration (e.g. geographic locations, satellite shape, orbit, etc.) of the mission elements shown in the picture do not necessarily

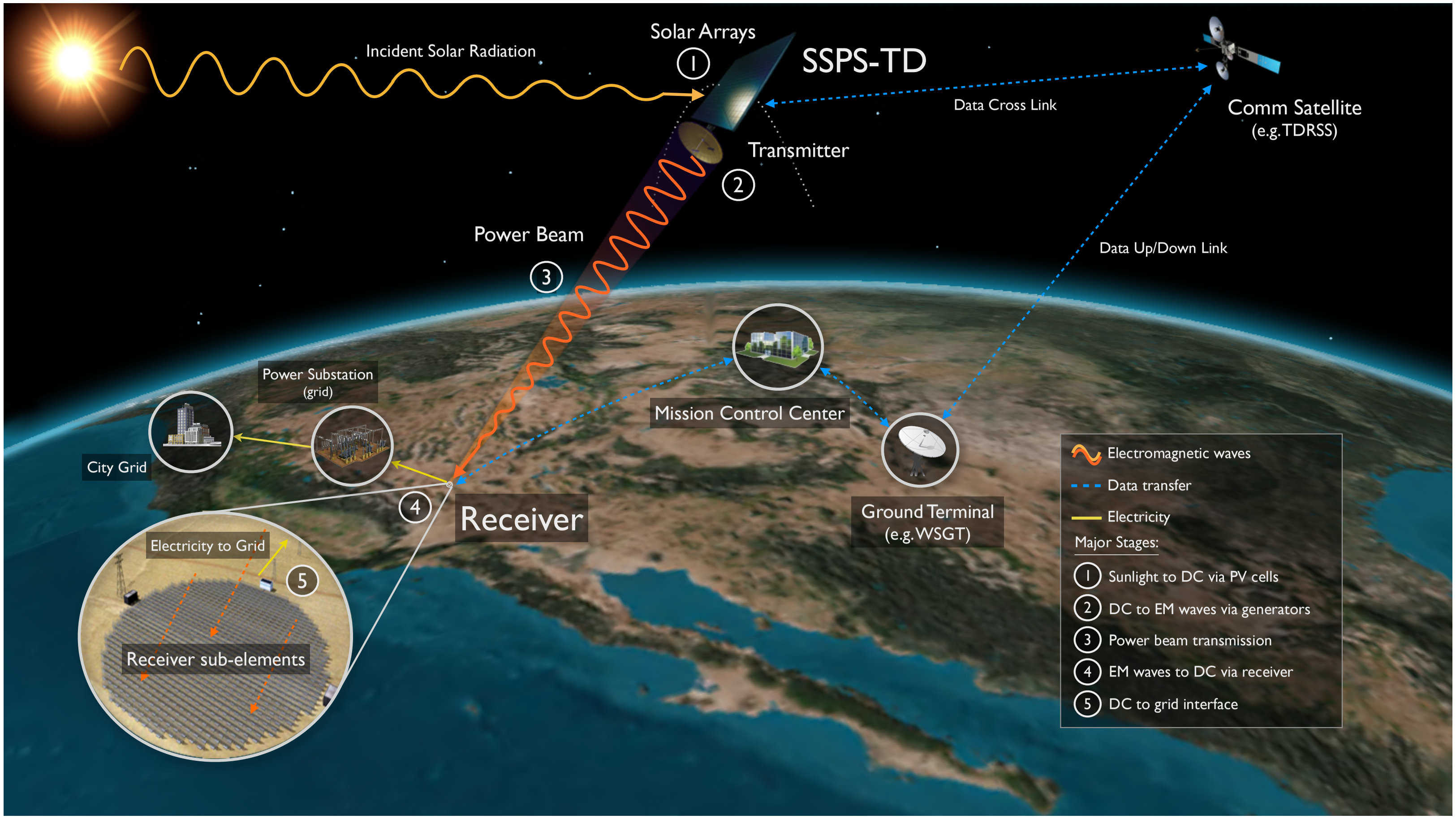


Figure 2.6: Concept of Nominal Operations for the SSPS-TD mission (not to scale).

represent the actual or final mission configuration but are chosen to best illustrate the *concept* of operations.

Though this ConOps graphic only depicts the nominal operations phase of the mission, it includes all segments pertinent to nominal operations and their primary assets (marked in italics):

Space Segment - *SSPS-TD (solar arrays, transmitter with microwave/laser generators)*

The space segment is responsible for generating the power beam from incoming sunlight (i.e. the first phase of SSP). It is fully robotic (i.e. unmanned), though not entirely autonomous since it receives commands from the manned ground segment.

Ground Segment - *Receiver, local power grid interface, Mission Control & Analysis Facilities*

The ground segment is responsible for converting the power beam into electricity and transferring it to the local power grid of the receiver and the local power grid interface which (i.e. the second phase of SSP). It also includes all ground-based facilities associated with mission control, data analysis, and launch. Note that the launch facilities are not shown in Figure 2.6 since they are not part of nominal operations.

Communication Segment - *Communication Satellites, Ground Terminals* The communications segment refers of the Communication Network or Strategy responsible for all data transfers between the ground and space segments. The Communication Network consists of its own space and ground segment, with multiple communication satellites and ground terminals positioned strategically on Earth to maximize communication coverage during the mission. The satellites relay data from SSPS-TD to their nearest ground terminal, which in turn relays the data to Mission Control.

Nominal operations begin with the SSPS-TD in a geocentric orbit. In the active mode, the satellite is within range and line-of-sight (LOS) of the receiver, in a so-called *coverage window*, and the 5-step SSP process discussed in the Mission Description (Section 2.1.4) is performed. To summarize, the on-board solar arrays convert incoming sunlight to DC, which is fed into the transmitter to be converted to electromagnetic (EM) waves and then directed into a laser or microwave power beam targeted at the receiver. The power beam then propagates through the atmosphere and is collected by the receiver and re-converted into DC. In this ConOps picture, the receiver has been placed in a dry and remote location to minimize both atmospheric attenuation and environmental or biological risk (see Sections 3.2 and 3.4). Finally, this electricity is modified for power grid compatibility and transferred to the local grid via power substations. Each of these steps is parameterized by an efficiency that takes into account any conversions and losses, and when put together, form the end-to-end efficiency, or link budget, of the system (see Chapter 5).

Note that the SSP process is essentially continuous during these nominal operations, i.e., once initialized, all of these steps are performed concurrently to create a continual supply of power to the grid. Interruptions or delays, however, may occur at the receiver-grid interface due to checkout procedures.

The inactive mode is defined by a loss-of-lock (LOL) between SSPS-TD and the receiver, i.e., the satellite is out of range or out of line-of-sight with the receiver. In this mode, there is no power beam formation or WPT and solar power generation is reduced to support additional systems only, or deactivated entirely if possible, and batteries will power the additional subsystems. Allowable time windows for active and inactive modes are determined primarily by the satellite orbit and receiver location (Sections 3.4 and 3.7). But the active mode will not necessarily be activated whenever LOS and range with the rectenna is confirmed; WPT performance and associated environmental effects are not yet fully understood and studying these may require varying power beam durations in order to mitigate risks and generate desired test results. It is however, a better test of the WPT hardware for the active mode to be engaged continuously. This is both an operational and design decision related to environmental or safety risks and the satellite orbit, which determined the availability and duration of coverage windows.

During both active and inactive modes of nominal operation, the mission is continuously controlled and supervised by Mission Control. In particular, control commands (e.g., satellite or solar array attitude adjustments, and WPT function scripts) are uploaded from Mission Control to the satellite and receiver, while satellite sensor and performance data (e.g., transmitter temperature, and solar array efficiency) are downloaded from these systems to Mission Control. Data transfer between the receiver and Mission Control can occur directly and it is likely that a smaller control and monitoring center will exist at the receiver location. Data transfer between the satellite and Mission control uses an external communications network that relays the data and ensures continuous communication coverage.

A possible communications strategy shown in this ConOps is to use the TDRSS network. This network consists of 9 communications satellites distributed in geosynchronous orbit.[39] The current Generation 2 TDRSS satellites offer multiple access data transfers in the S-band, and can also operate in the Ku and Ka bands with data

[39] 100% global coverage is only ensured for satellites in low Earth orbit (LEO).

rate capabilities of up to 800 Mbps. The first two Generation 3 satellites are expected to join the TDRSS network in the next two years, further increasing its capabilities.[40]

The primary TDRSS ground segment is the White Sands Ground Terminal (WSGT) located in Las Cruces, New Mexico. Data from the Mission Control Center is forwarded to WSGT, uplinked to a TDRSS satellite and sent to SSPS-TD.[41] Similarly, return data is relayed via the TDRSS satellite network to WSGT and then on to Mission Control.[42]

The SSPS concept only makes practical sense as a viable energy source if a large-scale system is constructed with high power output. The full-scale SSPS architecture would consist of a fleet of SSP satellites with multiple receivers to maximize active mode operation times. But as discussed previously, a technology demonstration mission is the necessary precursor for such an endeavor, whose objective is not to deliver maximum power but to investigate and increase the TRLs of SSP technologies. This is the rationale for the smaller scale of the SSPS-TD mission architecture which consists of only one satellite and one receiver.

2.2.2 Operational Phases

All of the major phases of the SSPS-TD mission are now discussed in chronological order. These phases match the system functions derived in the mission objectives (Section 2.1.3) to mission operations. These operations are then mapped and decomposed to sub-system functions and requirements that allow the system architecture to be formulated (Sections 3). This functional and requirements analysis is

[40] http://www.boeing.com/defense-space/space/bss/factsheets/601/tdrs_kl/tdrs_kl.html
[41] The data may be relayed between several TDRSS satellites depending upon LOS with SSPS-TD.
[42] https://www.spacecomm.nasa.gov/spacecomm/programs/tdrss/default.cfm

performed iteratively as the ConOps evolves and the project matures.

Launch and Deployment

The SSPS-TD will launch from Cape Canaveral or Vandenberg Air Force Base onboard a Heavy-class U.S. launch vehicle. Upon successful launch and orbit achievement, all subsystems are initialized by Mission Control and tested for positive response. Once SSPS-TD reaches its nominal orbit, deployable subsystems are setup (e.g. solar arrays unfolded) and any necessary in-orbit construction is performed. An extensive Deployment Checkout procedure is executed throughout this timeframe, concurrent with ground segment tests in preparation for nominal operations. In these tests, subsystems and components are put through their full range of motion or other capabilities.

Integrated System Checkout and Configuration

Upon successful deployment and subsystems checkout, the Pre-Nominal Operations Checkout begins. This procedure tests all integrated processes associated with the various mission modes. In particular, a trial SSP process is activated to confirm that the system is ready for steady state operations.

Nominal Operations

The SSPS-TD now enters the routine operations discussed in the previous section. In the active mode, energy flows into the local power grid after a series of power "quality" checkout procedures. In the inactive mode, there is no energy flow from the satellite.

Mission Analysis

Both during and after nominal operations, the performance of the SSPS-TD is evaluated in order to satisfy the mission goal. Onboard data is continuously transmitted from the satellite to Mission Control via the communication network, and then sent to any data analysis teams (e.g. DOE's National Renewable Energy Laboratory). Other measurements, like irradiance levels and effects, are measured from the ground or from airborne instruments. Together, these data provides many metrics that can be used to assess the viability of the SSPS concept in the context of this particular architecture. For instance, seasonal and atmospheric effects on power transmission and their relation to orbital characteristics and receiver location are critical to this analysis.

Failure/Maintenance Modes

Since this is a technology demonstration mission, it is a heavily monitored and operated system, but there are still significant risks and uncertainties. There are thus various Safe Hold Operations and Anomaly Resolution modes that exist in the event of a failure or indication of off-nominal operation. In these modes, certain mission control measures are activated depending on the failure. In general, all non-essential subsystems are powered down and tests are be performed to identify, assess and possibly repair subsystems or components.

Maintenance operations may be necessary for general station-keeping or repairs due to failures, and require a Maintenance mode as well.

End of Life

The mission has been designed to operate for 1 year. After this, there exist two end of life paths: 1) Disposal, where the satellite is boosted to a disposal orbit to be discarded, deorbited into the Pacific, and 2) Continued operations, where SSPS-TD either continues to operate as a technology demonstration mission with possible refits and updates, or it is converted and expanded into a full-scale system. Due to the anticipated high costs of the project, the second pathway is a more likely choice. The modularity of the system also encourages an upwards rescaling.

As discussed earlier, this is the most general formulation of a ConOps that will mature iteratively over the design lifecycle. A fully mature ConOps greatly expands upon these descriptions with the ultimate goal of containing sufficient detail for subsequent definition documents like the operations handbook, and for planning activities like facility staffing and network scheduling. It includes subsystem and component specifications (including support subsystems like structure, C&DH architecture, thermal subsystems, PMAD, and ADCS)[43], an end-to-end communications strategy, integrated logistics support, an operational timeline, design reference missions (DRMs), and all operational facilities, equipment and critical events associated with each mission phase.[44]

[43] C&DH - command and data handling, PMAD - power management and distribution, ADCS - attitude determinations and control system

[44] [9]

Chapter 3

Architecture and Trade Studies

3.1 Introduction

Needs analysis and project scoping enable an abstract operational description of the SSPS-TD system in the form of the concept of operations, from which materializes the system-level functional requirements. The system architecture is then constructed by linking these two elements to a specific and overall design or structure of the system, and thus providing the first real description of the SSPS-TD mission, including hardware and sub-system interfaces, processes, constraints, and behaviors.

The architecture represents an engineered solution to the problem statement defined during project scoping (i.e., the mission needs, goals, and objectives). But such a solution is generally not unique and is instead made up of a number of design choices that define a fundamental and specific mission architecture. This necessitates top-level decisions early on that often need to be treated as trade studies. And like the ConOps, the architecture matures over multiple iterations of the design as functional and performance requirements are modified to create a better match between the problem statement and the candidate solution.

Within this set of design choices, there exist fundamental decisions whose outcomes drive all subsequent product architecting and design, and define the *baseline architecture*. These decision outcomes are in turn driven by feasibility constraints and goals, making the architecture definition the center of the SSPS-TD mission feasibility study. It is therefore essential to prioritize these decisions and first establish this baseline. Due to the interdependence of the SSPS-TD baseline features with some specific subsystems, however, the architecture cannot be developed independently from subsystem design. This highlights the actual nonlinearity of the systems engineering process and forces this study to consider both architecture and subsystem-level trade studies concurrently.

So far, the SSPS-TD system consists of a single solar-powered (PV cells[1]), Earth-orbiting satellite, that wirelessly beams power down to a single Earth-based receiver that then converts it into electricity. In order to define the baseline architecture, this description must be concretized with a number of fundamental and specific mission features. A number of critical trade studies and design choices have been identified as driving the mission implementation and feasibility, and their results will complete the architecture definition (and several key subsystem designs):

- **Power Beam Type & WPT Hardware**

  The power beam *type* refers to the nature of the electromagnetic radiation that makes up the WPT power beam. This is defined by its classification, or spectral region, and its form, i.e., ordinary electromagnetic radiation or laser light. The power beam type constrains the specific frequency that will be chosen, and defines both the WPT hardware and the beam physics and characteristics. This decision is thus necessary for sub-system design to begin and has a widespread effect on many systems engineering metrics, like mass, size, cost, risk, and performance.

- **Receiver Location**

  Choosing a receiver location is a prerequisite for further refinement of the concept of operations. This critical decision must take into account a number of

[1] This decision was made in the needs analysis (Section 2.1.1)

environmental and operational constraints, many of which cannot be precisely stated until the power beam type is chosen. Furthermore, a specific receiver location is necessary to constraining orbit parameters and defining coverage times with the satellite segment.

- **Solar Array Design**

  Though technically a component or part-level design choice, and hence not part of the architecture description, the solar array design must be determined in order to accurately size the SSPS-TD system. This includes both the PV cell type and the array design and structure. Together with the WPT hardware, this design choice allows the mission to be sized and weighed, and thus plays a large role in mission feasibility.

- **Satellite Orbit**

  Once the nature of the power beam has been established, the SSPS-TD orbit is perhaps the most defining feature of the system, and the core of the feasibility study. It is the most complex decision due to the orbit's nearly independent role in determining system size, mass, operational modes and timing (e.g. launch logistics, and receiver coverage times defined by the active and inactive modes), and a multitude of performance characteristics associated with power densities, thermal effects, and structural design, among other factors. And these factors all have an extensive influence on sub-system design and systems engineering metrics.

Mission scoping generated a number of objectives, assumptions, constraints, and operational concepts that greatly influence the decision making process for these architecture features. Consequently, they are translated into requirements and metrics that are used to evaluate the decision options for each of these features.

For the cases of the power beam type and satellite orbit, this evaluation process is performed through formalized and rigorous trade studies. Upon choosing the power beam type, the power beam frequency, WPT hardware, and solar array design are selected using more straightforward comparison methods. These latter decisions are not fundamental to the baseline architecture, but are part of the general architecture which must be described, and integral to the orbit trade study.

The receiver location is chosen based after a careful evaluation of the operational needs and constraints, as discussed above. This decision must occur after choosing the power beam type, but before the orbit trade study.

Note that this study will focus primarily on the design of the space segment (i.e., satellite), within the context of the larger SSPS system concept.

3.2 Trade Study: Power Beam Type

In this first trade study, the type of the power beam is determined using AHP (Analytical Hierarchy Process). As discussed, trans-atmospheric WPT is best performed using either a microwave beam or an infrared laser beam. Consequently, these two beam types define the trade space. The trade study is formalized below:

Objective

Select the power beam type for the WPT of SSPS-TD system. The type is defined above as the classification and form of the electromagnetic radiation that makes up the power beam.

The beam must allow the WPT element to meet the following top-level functional requirements:

- Long range transfer: The SSPS-TD WPT element shall, by definition, transmit power from the satellite's orbital position to a receiver on the ground.

- High transmission efficiency: The WPT power beam shall be designed to both minimize atmospheric attenuation and allow for high energy densities, in order to optimize SSPS-TD capabilities.

- High directivity, low divergence, and high pointing accuracy: The WPT power beam shall be highly directional[2] and controllable. Furthermore, the less divergent the beam, the less it will spread due to diffraction, allowing for longer transmission distances and smaller receiver sizes or higher collection efficiencies. High pointing accuracy is required to control the beam's path and ensure reliable and functional WPT while meeting safety and environmental constraints.

Rationale

This decision is fundamental to the mission architecture for two reasons:

1. Different beam types use different means of generating and receiving the power beam that correspond to distinct sets of hardware for the WPT system. A proper system hierarchy cannot be created until a method has been chosen, thereby allowing the sub-system design process to begin. As such, this choice greatly affects the mass, cost, and performance characteristics of the system.

2. The nature of the beam defines the physics of the power beam's transmission through a medium and thus describes the beam's profile and characteristics through the atmosphere and on the ground. This is integral to conducting

[2]Referring here to the mathematical concept of peak directivity.

system sizing and performance analyses, understanding environmental effects, and identifying safety concerns and risk factors.

Assumptions and Constraints

1. The SSPS-TD will launch within the next 15 years (Section 2.1.6, Assumption 1). The WPT element is therefore restricted to power beam technology that can reach at least TRL 5 within this timeframe.

2. The power beam is restricted to the microwave or infrared bandwidth (Section 2.1.7, Constraint 2).

3. The power beam types are evaluated under the assumption that the power beam is designed to meet Constraints 3 and 4 (Section 2.1.7), relating to safe power density limits.

Alternative Designs

There are only two types of electromagnetic radiation that can presently perform trans-atmospheric WPT:

1. Microwaves: A microwave power beam consists of ordinary electromagnetic radiation transmitted at a frequency located between 300 MHz and 300 GHz (1 m down to 1 mm wavelengths). A microwave-based WPT element utilizes arrays of microwave generators to form the power beam in the transmitter aperture. The microwave power beam is then incident upon a specialized receiver, the rectenna, which is a specialized antenna designed to collect incident microwaves and convert them to electricity. For the purpose of this trade study, only power

beams at 2.45 and 5.8 GHz are considered, as discussed in Constraint 2. The 35 GHz frequency is dismissed due to very low TRLs for associated WPT technologies, and a consequent lack in reliable data. This is further discussed in Section 3.3.3.

2. Infrared Lasers: An infrared laser power beam consists of coherent, collimated light at a frequency located between 300 GHz to 405 THz (1 mm down to 740 nm wavelengths). A laser-based WPT element converts a transmitted laser beam at the receiver site into electricity via PV cells specially designed with peak responsivity at the power beam wavelength, or first into heat, via heat exchangers, which can be converted to mechanical and then electrical energy. For the purpose of this trade study, the laser power beam will operate in the near-Infrared, as discussed in Constraint 2, and the receiver will utilize PV cells due to their higher efficiencies and proven capabilities for power beaming. Furthermore, the laser technology is restricted to solid-state, diode-pumped lasers due to their high power density capabilities, low mass, and high efficiencies.

The principle advantages to laser beaming over microwave beaming are a smaller receiving site due to very small beam divergence (i.e., diffraction) since the light is collimated), negligible radiation outside of the beam, and the side lobes do not interfere with other electromagnetic radiation (e.g. no radio-frequency interference). Disadvantages include low system efficiency, high mass, and both significant and unstable atmospheric losses.[3]

[3] [5]

Method of Trade Study

The power beam type will be decided using the Analytical Hierarchy Process (AHP). This requires a set of evaluation criteria or metrics, called *Figures of Merit* (FoMs), that are applied to each design alternative in such a way as to determine the best choice through a series of pair-wise comparisons. AHP has been deemed the most appropriate for this decision due to its ability to rank the FoMs in order of importance (i.e. assign weights to each FoM) and employ both quantitative and qualitative FoMs in a common framework.

Evaluation Criteria

The following FoMs have been identified as the most important criteria in choosing a power beam type. They are not ranked in any particular order since this is as yet unknown and will be determined through AHP.

Note that though it is desired to maximize or minimize a given FoM, this may be at the expense of another FoM, hence creating a multivariable optimization problem that is solved by AHP. Optimizing a FoM in this sense refers to its actual value (e.g. the mass in kg) and not the FoM rating assigned to each design alternative (e.g. a 1-9 rating scale). A higher rating indicates a more preferable choice.

1. *Space-based WPT Mass.* The mass of the WPT components in the space segment, namely the power beam generator/transmission subsystem. The mass is the most useful metric associated with cost, launch sizing, and deployment logistics. This is especially important given the constraint of only one launch vehicle (Constraint 6). A higher rating for this FoM indicates a smaller mass, and hence a cheaper and more feasible mission.

2. *Efficiency.* A combination of three different efficiencies: 1) the efficiency of the beam propagation due to atmospheric attenuation, 2) the efficiency of the hardware in both the space and ground segment, namely the power beam generators (DC-EM conversion) and receiver components (EM-DC conversion), and 3) an indirect efficiency associated with the divergence properties of the power beam and the consequent transmitter aperture and receiver size necessary to collect a given amount of power (collection efficiency). These efficiencies make up one segment of the system's extensive end-to-end link budget (Chapter 5), and contribute directly to the SSPS-TD system performance. A higher rating for this FoM indicates higher efficiencies for these three factors, resulting in higher grid power output, smaller system size, and lower thermal waste that must be managed.

3. *TRL/R&D^3*. A combination of the current TRL and the R&D degree of difficulty (R&D^3). The latter is a measure of the R&D effort needed to take the WPT technology associated with a power beam type to TRL 5 (e.g., ready for integration into the system). A rating scale for this metric was developed by NASA in the mid-1990s,[4] and shown in Table 3.1. It expresses the difficulty and probability of R&D success on a 1-5 scale. For this mission, a chosen rating will mainly reflect the associated cost and schedule to reach TRL 5, and to a lesser degree, the general technical feasibility. The evaluation of these two concepts acts as a form of technology risk assessment for each power beam type, based on the probability of technology development success. A higher rating for this FoM indicates a cheaper, higher reliability, and more feasible mission given top-level development constraints like schedule.

[4] [37]

Table 3.1: Research and Development Degree of Difficulty Scale

R&D[3]	Definition[1]	Explanation[2]	Probability of Success[3]
1	Very low degree of difficulty anticipated in achieving R&D objectives for this technology	Only 1 or 2 short-duration technological approaches needed to be assured of a high probability of success in achieving technical objectives in later systems applications.	99%
2	Moderate degree of difficulty anticipated in achieving R&D objectives for this technology	2-3 technological approaches needed; conducted early to allow an alternative to be pursued to assure of a high probability of success in achieving technical objectives in later systems applications.	90%
3	High degree of difficulty anticipated in achieving R&D objectives for this technology	3-4 technological approaches needed; conducted early to allow an alternative subsystem approach to be pursued	80%
4	Very high degree of difficulty anticipated in achieving R&D objectives for this technology	4+ technological approaches needed; conducted early to allow an alternate system concept to be pursued	50%
5	Degree of difficulty anticipated in achieving R&D objectives for this technology is so high that a fundamental breakthrough in science is needed	Basic research in key areas of physics, chemistry, etc. needed before feasible system concepts can be refined	10%-20%

[1] [37]
[2] [41]
[3] Assuming "normal" R&D effort

4. *Safety.* A measure of the hazardous effects of the power beam and associated WPT components to biota (humans, plants, and animals) and the environment (atmosphere and climate). Risk in the form of specific failure events is not included (see Risk Analysis in Chapter 9) but is accounted for to some degree by evaluating all forseeable power beam effects, some of which would occur during a failure. A higher rating for this FoM indicates a safer mission design.

5. *Extensibility.* The ability of the SSPS-TD system to be utilized after its operational lifetime, i.e., continued technology demonstrations or converted into a full-scale system. In this sense, extensibility is a measure of a system's adaptability or flexibility, and is especially dependent on its modularity. A higher rating for this FoM indicates a better ability to do so.

Note that the metrics of cost and complexity are not listed as they are implicit and inferable from the more fundamental and measurable mass and TRL FoMs. Reliability is not included since there is insufficient data available, but a primitive measure can be obtained from the TRL metric.

Analysis

The trade study has now been fully defined and the AHP computational procedure can begin. This process consists of three phases. In the first phase, the FoMs are ranked via pair-wise comparisons in the AHP prioritization matrix, generating respective weights. In the second phase, a score is generated via pair-wise comparisons for each design alternative for a given FoM (i.e. one score per design alternative per FoM). In the third phase, the final score for each design alternative is calculated using the FoM weights found in the first phase.

In the first two phases the 1-9 AHP weighting scale is used, where 1 represents "neutral" and 9 represents "extremely prefer". Note that pair-wise FoM comparisons require a pair-wise hierarchy that is determined by the system engineer. However, this in no way predetermines their final rankings since the FoMs have only been ranked in pairs, and the relative degree of their importances (i.e. their weights) are as yet unknown.

Phase I: FoM Prioritization Matrix

The pair-wise comparisons in the prioritization matrix are evaluated on the basis of several hierarchical decisions and a sensitivity analysis method.

First, it is established that TRL/R&D^3 and safety or the two most important FoMs, with mass following in third. The schedule assumption acts as a hard time constraint on technology development, while cost and general mission feasibility are major factors in TRL advancement, all of which necessitate a power beam type with low R&D^3. Safety is also critical to the mission, as it always is, due to the inherent risk and uncertainty in a technology demonstration mission. This is especially important for the SSPS-TD mission since the power beam directly affects the local environment and interacts with a staffed ground segment. Finally, mass is always an important metric as it relates to cost and general mission feasibility. Efficiency and extensibility do have mission significance as well, but relative to the previous three criteria, they represent more desirable features than mission critical ones.

With this basic hierarchical setup, a "binary" pair-wise comparison can be made, i.e., safety is more important than efficiency. But this does not specify how much more important one FoM is over the other, as expressed by the AHP weighting scale. To help the decision process of assigning a specific weight during this pair-wise comparison, a form of sensitivity analysis is performed on two opposing extreme scenarios. This is best illustrated in an example:

Consider the pair-wise comparison of the safety and efficiency FoMs in the AHP prioritization matrix, and further assume that these are the only two FoMs for this trade study and an infinite variety of design alternatives exist. As discussed above, it is known that the importance of safety supercedes that of efficiency, but not by how much (i.e. safety will have some rating greater than 1 when compared against efficiency).

Now, consider a design scenario in which a power beam type offers 100% efficiency but is extremely dangerous; given the mission priorities, this is unacceptable and this design alternative would not be chosen. Then, imagine the opposite extreme of this scenario in which the power beam type offers a 1% efficiency but is completely safe (no dangers or harmful effects); because this efficiency is so low, it can be assumed that such an SSPS system is not viable or practical as a power generation system, and hence this design alternative will not be chosen either. If this latter scenario had been acceptable, however, then efficiency would be deemed irrelevant in the face of safety, and safety would be assigned a 9 (and efficiency would receive 1/9). Since this is not the case here, nor is it realistic in general, a form of sensitivity testing can be performed on these two extreme scenarios. Consider a power beam type that offers 75% efficiency but is significantly dangerous (but less so than before); still, this is unacceptable given the importance of safety. But imagine a power beam type that offers 25% efficiency but is generally safe with only minor risks or harmful effects; now, the viability of this system is unknown but seems to be reasonable, and most importantly, essentially safe, therefore offering an acceptable design choice. This reveals an imbalance in the sensitivity of the decision acceptance to variations in the FoMs.

The degree of this imbalance can be better resolved by creating further scenarios in which the FoMs approach each other more closely. In the end, a measure of qualitative judgement or "ball-parking" is required by the systems engineer in order to pinpoint the exact weight. In this example, an acceptable design choice was significantly more dependent upon safety than it was on efficiency, and so a rating of 7 is given.

Two major difficulties presented themselves when populating the prioritization matrix using this method. The first is that quantitative and qualitative FoMs are not so easily compared during sensitivity testing because it is impossible to vary the FoMs on an equal scale (e.g. efficiency can be measured with percentages while extensibility is largely measures qualitatively or using several different metrics). Then, the systems engineer must simply use his or her best judgement. The second difficulty arises from the hard constraints imposed by mission scoping and general design logistics. For instance, any design alternative involving technology that requires over 15 years to be developed to TRL 5 and integrated into the mission is immediately discarded. Efficiency must also be "reasonable" as discussed in the example above. This situation creates conditional FoM scenarios that may complicate AHP rating decisions and must be kept in mind (e.g. so long as the mass is under X kg, it is unimportant relative to the other FoMs). Where possible, this problem is solved by only choosing design alternatives that do not violate these constraints.

Using the concepts discussed above, the prioritization matrix is populated and the resulting scores are computed, as shown in Figure 3.1.

According to the results, the FoMs are ranked and weighted as follows:

1. TRL/R&D^3 - 41.50%
2. Safety - 35.70%
3. Space-based WPT Mass - 12.55%
4. Efficiency - 6.92%
5. Extensibility - 3.33%

Prioritization Matrix
Power Beam Type

	Space-based WPT Mass	Efficiency	TRL/R&D3	Safety	Extensibility
Space-based WPT Mass	1	4	0.167	0.125	6
Efficiency	0.250	1	0.167	0.143	4
TRL/R&D3	6	6	1	2	7
Safety	8	7	0.5000	1	9
Extensibility	0.167	0.25	0.143	0.111	1
Column Sum:	15.42	18.25	1.98	3.38	27.00

Normalized Values

						Row Average	**FoM Weights**
Space-based WPT Mass	0.0649	0.2192	0.0843	0.0370	0.2222		**0.1255**
Efficiency	0.0162	0.0548	0.0843	0.0423	0.1481		**0.0692**
TRL/R&D3	0.3892	0.3288	0.5060	0.5919	0.2593		**0.4150**
Safety	0.5189	0.3836	0.2530	0.2959	0.3333		**0.3570**
Extensibility	0.0108	0.0137	0.0723	0.0329	0.037		**0.0333**

Figure 3.1: AHP prioritization matrix for Power Beam Type Trade Study.

A number greater than 1 indicates a preference towards the FoM in the row over the FoM in the column.

As predicted, TRL/R&D^3 and safety are the dominant FoMs, with the space-based WPT mass coming in third. It might be expected that safety would outrank or be closer to TRL/R&D^3, however, the purpose of a technology demonstration like SSPS-TD is to better identify and understand the risks and uncertainties in the technology, and therefore raise its TRL to the point where it is considered safe. In other words, the mission cannot be designed with absolute safety first because the knowledge of how to do so is lacking, and thus necessitates a technology demonstration mission.

Also, remember that though the space-based WPT mass and efficiency FoMs have relatively low weights, there are boundaries (i.e., semi-quantitative extremes assessed using general reason) associated with each of them (e.g. mass should not exceed 100,000 kg, or efficiency should not be under 15%[5]). This means that so long as the design alternatives are within these boundaries, then the FoMs carry the weights found above.

Phase 2: Design Alternative Matrices

The design alternatives are now investigated within the context of each of the five FoMs in order to formulate a comparative rating on the 1-9 scale. Because there are only two design alternatives, only one rating is required in each of the five design alternative matrices. Choosing a rating, however, is made difficult by the fact that there exist many forms of both microwave or laser-based SSPS concepts. These concepts are distinguished by different design characteristics (e.g. modularity), hardware (e.g. converter type), and configurations (e.g. structural), that have significant consequences on their FoM values. And in some cases, choosing a concept will restrict other subsystem choices that are unrelated to the power beam type (e.g. PMAD or structure), and that may have undesirable features (e.g. low TRL). Though no perfect solution to this problem exists,[6] an attempt is made to either choose the "best" form for each power beam type, or take an average value where appropriate; where relevant, this will be noted in the discussion.

1. *TRL/R&D*[3]

   The current TRL of each power beam type is assessed by considering both the achievements of previous relevant WPT demonstrations and the individual

[5]These are examples that "sound reasonable" and should only be used if there exist design alternatives well within these thresholds.

[6]Hence, the use of an iterative design loop in these early stages of development.

TRLs of its key components, within the context of the SSPS-TD application. Similarly, R&D[3] is determined by considering a power beam type's R&D history on both a system and sub-system level, within the context of the SSPS-TD application. The resulting two ratings are compared for each power beam type and an AHP rating is extrapolated via a semi-quantitative "best" estimate or fit.

The current state of microwave WPT is best appreciated by reviewing the key tests and demonstrations that involved this technology:[7]

Table 3.2: Key Microwave WPT Tests

Year	Test	Achievement(s)
1964	Beamed Power RC Helicopter	First field demonstration of WPT by Raytheon/William Brown
1975	High Power WPT	$\sim$ 34 kW WPT demonstration (highest power to date) by NASA JPL and Raytheon
1983	MINIX	First test of microwave WPT though the ionosphere from one section of a sounding rocket to another) by Kyoto University
1987	SHARP	First demonstration of a microwave WPT powered, unpiloted aircraft in Canada
1992	METS	First measurement of non-linear ionosphere interactions due to microwave WPT by Japan
1995	Power Transmission to Airship	5 kW-class microwave WPT to an airship by Japan
2003	Integrated Sandwich Module Demo	Kobe University established a physical baseline for the sandwich SSPS configuration at 2.45 GHz, including PV, structure and RF elements
2006	Furoshiki Sounding Rocket Experiment	First test in space by University of Tokyo, Kobe University, and University of Vienna, of an end-to-end SSPS deployment concept (with limited WPT elements and capability)- very low TRL
2008	Solar-Powered Microwave WPT at Long Range	Longest ranged (ground-to-ground) solar powered demonstration of WPT at 2.45 GHz (only 20 W after collection, but at a distance of 148 km), sponsored by Discovery Communications
2009	Advanced Technology Retrodirective Phased Array Test	Test of microwave WPT with high-efficiency solid-state amplifiers and active beam steering to a moving vehicle, by Kobe University

Laser WPT demonstrations have focused mainly on either small-scale power-beaming over short ranges, or weaponized applications for air-to-air combat.

[7] [41]

The latter involves higher power systems that are more in line with those that would be used in the SSPS-TD mission and future full-scale applications, but it is still in early phases of testing and utilizes pulsed bursts rather than continuous beaming, which is necessary for WPT. A central issue is that most research has been on laser-generating technology itself, without consideration for the important technological features necessary for laser-based WPT, like high power density, high efficiency, and low mass.

Some key relevant laser-based WPT tests are summarized in Table 3.3.

Table 3.3: Key Laser WPT Tests

Year	Test	Achievement(s)
2006	Commercial Laser Powerbeaming	PowerBeam develops laser powerbeaming for consumer environments (e.g. lights and handheld devices), with range capability in the tens of meters and power capability in the tens of watts
2008	JHPSSL Laser Weapon Test	Northrop demonstrates a moveable 100 kW solid-state diode-pumped weapon system (but the size of a truck trailer)- the longest continuous firing was 10 minutes
2009	Laser powerbeaming to operate a lift	LaserMotive demonstrates laser powerbeaming over 1 km, winning NASA's 2009 Power Beaming Challenge
2010	Airborne Military Laser Test	100 kW-class laser onboard a Boeing-747 shoots down scud missiles in a test by the DoD and Northrop

It is evident that laser-based WPT technology is still in its infancy as far as becoming a space-worthy system for SSP. This is due in large part to lack of funding and the inherent complexity associated with generating high power lasers and reducing system mass. Meanwhile, microwave WPT in an SSPS concept is seen to be technically feasible, with research focused mainly on improving hardware efficiencies, power management and specific mass or power (i.e., power to weight ratio or power over area), among other technical issues.

These sub-system issues are reflected by TRLs and R&D^3 ratings for the individual technology areas needed for SSPS-based WPT, which together will provide an approximate system-level rating for each power beam type. A comprehensive 2011 study performed by the IAA (International Academy of Astronautics)[8] identifies these key WPT technological areas for three of the most popular SSPS concepts, classified as Type I, II, and III, and provides a preliminary TRL and R&D^3 assessment for each. Type I is an updated version of the original microwave-based concept developed by NASA in 1979, Type II is a solid-state, diode-pumped laser WPT system, and Type III is a modern microwave-based system using the sandwich module developed in 2003. The principle WPT difference between Types I and III are the use of RF tubes as microwave generators in the first, and solid-state amplifiers in the second (see Section 3.3.4 for more information).

A WPT system-level TRL and R&D^3 rating is extrapolated from this study by combining the sub-system ratings associated with the identified technologies of each SSPS Type. Specifically, these are the microwave or laser generators (e.g. microwave tube or solid-state amplifiers, or diode laser arrays), TMS (thermal management system), PMAD, and beam control (these are further discussed in Section 9.1). The final ratings are obtained by an approximate average of these technologies, but taking into account that the values in the 2011 study are for a full-scale system, not a technology demonstration. The results are shown in Table 3.4.

These values are now assessed in the context of the SSPS-TD mission and after consideration of the power beam type's history of demonstrations, or heritage. The final TRL and R&D^3 ratings are determined in Table 3.5, along with the resulting AHP rating. In general, the ratings represent the more conservative

[8] [41]

Table 3.4: Power Beam Type - Individual Technologies Assessment

SSPS Type	**TRL**	**R&D^3**
Microwave Power Beam (RF tubes, Type I)	4-5	3
Microwave Power Beam (Solid-state amplifiers, Type III)	4-5	2
Laser Power Beam (Type II)	3-4	3-4

choice and take into account the technology's readiness for actual integration into the SSPS-TD mission, given the scope of past demonstrations.

Table 3.5: Power Beam Type - TRL/R&D^3 FoM

Power Beam Type	**TRL**	**R&D^3**	**AHP Weight**
Microwave Power Beam	4-5	2-3	**6**
Laser Power Beam	4	4	**1/6**

The AHP rating reflects several shortcomings of laser power beaming. Most importantly, it is highly unlikely that a practical laser-based WPT system would be ready for demonstration within 15 years (Assumption 1). Advancement from TRL 4 to TRL 5 requires an immense technological leap in terms of lightweight, efficient, high power output. Most importantly, high power laser transmission has only been developed for relatively short ranges compared to the orbital distances involved in SSP systems. Thermal management is also central to large-scale laser WPT since heating issues currently constrain run times. Laser power beaming thus has a very high risk associated with its technology development. Conversely, microwave WPT tests have proven out the basic concepts involved and the hardware necessary for full SSP technology demonstrations like SSPS-TD. There are thus fewer foreseeable problems with microwave power beaming and related R&D, which makes it a far cheaper and more reliable option to laser power beaming.

2. *Safety*

   The major environmental and safety issues associated with each power beam type are essentially independent of design form and are listed and categorized by locale:

   (a) Atmosphere

   - Microwave Power Beam:

     ⋄ A microwave beam will raise the local electron temperature in the ionosphere due to collisional absorption. This has the following safety-related effects:[9]

     – Plasma density variations that cause small changes in local chemical content (including ozone). These effects are theoretically negligible (<1%) so long as the TSI power density threshold in Constraint 3 is respected.

     – Significant interference of trans-ionospheric radio signals passing through this heated region (e.g. over 10 dB loss for HF frequency of 6 MHz).

     ⋄ There is no evidence of direct meteorological effects but detailed studies are recommended.

   - Infrared Laser Beam:

     ⋄ According to [46], selected infrared wavelengths can cause local climate changes, and turbulences may be produced that could be dan-

[9] [48]

gerous to aircraft crossing the beam.

◇ Ionospheric plasma perturbations are believed to be insignificant, but further studies are recommended.

◇ There is also no evidence of direct meteorological effects but detailed studies are recommended. ◇ No radio interference in the infrared.

The effects discussed above are based almost entirely on theoretical studies and small scale experiments or observations. Multiple sources[10] emphasize the need for further tests involving the higher power densities and radiation exposure times associated with SSPS power beaming. This level of uncertainty for both beam types translates to high risk, and both types are seen as having nearly equal safety ratings in this particular locale. Laser power beaming may be slightly preferably here due to the absence of interference effects.

(b) Ground (Receiver Site)

- Microwave Power Beam:

  ◇ Power leakage at the rectenna site takes the form of waste heat given off by the conversion process from RF to DC. Large rectennas (≥ 120 km^2) can radiate heat comparable to that of large cities. This energy emission has the following safety-related effects:

  - Local climate change is highly probable even for a smaller rectenna that would be used in a technology demonstration mission.
  - Ecological damage to the surrounding area depending on the amount of waste heat. Changes can range from no apparent effect on biota to complete death or sterilization of the local ecolosystem. This

[10] [5] and [48] for instance.

effect can be mitigated by choosing an appropriate rectenna site (e.g. remote desert) and imposing constraints on the design.

◇ Microwaves can have significant effects on animals (including humans), most of which occur at extremely high power densities (e.g. thermally induced pain at the skin surface) and are easily avoided by adhering to regulatory constraints (e.g. Constraint 4). The following are effects that must still be considered:[11]

  – Thermal swelling of internal organs under radiations as low as 1 W/m^2.

  – Retinal and tissue damage for exposure times on the order of minutes for power densities around 100 W/m^2 at 3 GHz

These effects pose a serious risk to microwave beaming and necessitate a strict set of conditions on animal and human presence in the rectenna.

◇ There is very little data concerning short and long term microwave exposure on plants.

- Infrared Laser Beam:

  ◇ Power leakage at the receiver site is also a problem for a laser power beam. Though the receiver is much smaller than that for a microwave beam, the power density is much higher and the conversion efficiency is much lower, resulting in higher waste heat dissipation that is comparable to conventional large thermal power plants.[12] Ecological change may, however, be more localized due to the smaller receiver size.

[11] [5]
[12] [5]

◇ Infrared light is absorbed at the skin surface level and converted to heat. The threshold radiation density for human skin is 1700 W/m^2, above which thermal effects are produced.[13] Protection suits for exposed personnel are recommended.

◇ Retinal damage can occur at laser power densities as low as 10^{-2} W/m^2, far below the expected operating power density of an SSPS laser power beam. This is the greatest risk to animals and on-site personnel as even scattered light can cause significant loss of vision.

◇ Significant harm to plants and insects exposed to the power beam.

Though it is difficult to determine whether local ecological effects are more severe with laser than with microwave beams, the greater risk of a laser beam to humans at the receiver site, especially in the event of a misdirected beam, make the microwave beam a safer choice in this particular locale.

In examining these effects it is seen that the microwave power beam is slightly safer than the infrared laser power beam. Still, both beam types carry significant risk resulting from the large degree of uncertainty in safety-related effects in both locales. But, as discussed in the TRL/R&D^3 section above, microwaves have a far longer heritage than lasers, especially for trans-atmospheric applications, making them a lower risk, higher reliability solution. Therefore, the microwave power beam is given an AHP rating of **4** in safety, corresponding to a laser power beam rating of **1/4**.

3. *Space-based WPT Mass*

The space-based WPT mass consists primarily of the power beam generators

[13] [5]

(i.e., the DC-EM converters) which form the transmitting aperture (e.g. antenna), the PMAD and TMS subsystems, support structure for the generator arrays, and any other hardware associated with forming the power beam across the aperture (e.g. waveguides). Together, these subsystems constitute the transmitter element (see Section 3.3.4). Since the system has not yet been sized (and cannot be until the architecture is defined), the most useful mass parameters are the transmitter subsystem specific mass (kg/m^2 or kg/m^3) and the transmitter specific power (W/kg). The transmitter mass of a 105 kW system with each power beam type is also used for comparison to account for differing power densities (see discussion below).

Several key issues arise when determining these parameters:

i. There exist a multitude of hardware options and configurations for each power beam type that differ in these two parameters.

   Fortunately, this issue can be largely ignored. The most popular microwave WPT systems all have similar specific masses and powers so that a reliable average can be taken. Furthermore, the only promising laser technology for SSP applications is a solid-state, diode-pumped laser, as mentioned earlier.

ii. The mass parameters are determined from reference systems or experiments that may not scale linearly with size or power. This is particularly true for laser systems and greatly affects the applicability of these parameter values when examined in the context of the SSPS-TD mission.

   This problem can be mitigated by choosing the most relevant reference systems and assuming that the comparative order of magnitude of the mass parameters between each power beam type remains roughly the same.

iii. Microwave generators are essentially 2-dimensional in physical configuration, while lasers are distinctly 3-dimensional, making it impossible to compare specific masses on an equal footing.

   In the absence of a direct solution to this problem, both types of specific masses are determined in order to appreciate the numbers, but more comparative consideration is given to the specific power which can be compared equivalently. This is also the reason for comparing the total transmitter mass for a reference system, as discussed below.

It would seem equally important to consider the corresponding aperture power density (W/m^2) since this will define the final transmitter aperture size and hence affect the final mass. However, in an actual SSPS system, this parameter is not simply the product between the two previous mass parameters. Thermal constraints, safety limits, and other power-related thresholds (including those for the receiver) all restrict this number and vary according to frequency, orbit and a number of other design features that cannot be specified until the baseline architecture is established. More importantly, this parameter cannot be determined comparatively for each beam type because the laser WPT system is 3-dimensional and hence the aperture size does not reflect the total system size or mass; a microwave WPT system, on the other hand, is essentially one large 2-dimensional aperture. This is important to acknowledge because laser WPT operates a much higher power densities.

Since a direct power density comparison cannot be reliably performed, the transmitter mass of a 105 kW system is used for comparison, ignoring SSPS-related constraints. 105 kW is chosen based on the available data for laser systems, and the maximum microwave transmitter power density is taken to be 26 kW/m^2

(see Section 3.3.4).

Microwave Power Beam

The average of three 5.8 GHz state-of-the-art microwave generator elements (klystron, magnetron, and solid-state) is used to determine the mass parameters associated with a microwave power beam. [39] provides this data and includes the PMAD and thermal components in the mass parameters. Since these are well-established technologies whose SSPS-related issues are largely independent of mass, the associated mass parameters are not expected to improve very much within the allowed timeframe of 15 years. It is assumed that similar mass parameters exist for 2.45 GHz transmitters.
Specific mass: 35.4 kg/m^2, Specific power: 700 W/kg, 105 kW transmitter mass: $\sim$150 kg

Laser Power Beam

The laser power beam mass parameters are found by considering three reference laser diode systems:

1. JHPSSL Firestrike: A 1.06 μm (283 THz), 105 kW laser, developed by Northrum Grumman. It consists of seven 15-kW amplifier modules, fit together in individual boxes measuring 0.305 m x 0.584 m x 1.016 m, for a total mass of about 1750 kg. This is a theoretically scalable system, and the best mass reference system for laser power beaming.[14]
   Specific mass: 1381 kg/m^3, Specific power: 60 W/kg

2. HELLADS (High Energy Liquid Laser Area Defence System): Project still in development, with the ambitious goals of producing a 150 kW laser

[14]http://spectrum.ieee.org/semiconductors/optoelectronics/ray-guns-get-real/3

that has a maximum weight of 750 kg and maximum envelope of 3 m^3. Since this system is still in the early stages of development, these values cannot be reliably used, but they do reflect potential future capabilities and relatively rapid improvement.[15]

Specific mass: 250 kg/m^3, Specific power: 200 W/kg

3. LaserMotive Power Beam: LaserMotive won the NASA Power Beaming Challenge in 2009 with a 1 kW, 2 kg, 808 nm diode laser. Since then, LaserMotive has developed a 800 W/kg system that can continuously power a UAV at altitudes up to 1.6 km and ranges up to 16 km. These systems have a limited range (<10 km for the 2009 system), however, so they are not entirely applicable to the SSPS concept.[16]

   Specific mass: unavailable (possibly 11,000 kg/m^3 with the 1 kW diode laser by itself on the order of 2×10^{-4} m^3), Specific power: 800 W/kg

Given these three reference systems, and with a strong bias toward the first, a reasonable specific mass and power are 1000 kg/m^3 and 150 W/kg, respectively, with an expected 105 kW transmitter mass of about 1500 kg.

Table 3.6 summarizes these results and indicates the extrapolated AHP rating. The results clearly indicate that, from a purely mass-centric point of view, microwave power beaming is the preferable choice.

4. *Efficiency*

The three types of efficiencies measured here are 1) the hardware efficiencies of the front and back-end power beam conversion process, 2) the power beam transmission efficiency due to the atmosphere and meteorological effects, and 3)

[15] http://www.darpa.mil
[16] [43]

Table 3.6: Power Beam Type - Space-based WPT Mass

	Microwave Power Beam[1]	Laser Power Beam
Specific Mass	35.4 kg/m^2	∼1000 kg/m^3
Specific Power	700 W/kg	∼150 W/kg
105-kW Transmitter Mass	150 kg	∼1500 kg
AHP Weight	**8**	**1/8**

'∼' = approximate or expected value
[1] Average of 5.8 GHz state-of-the-art klystron, magnetron, and solid-state microwave transmitters. Includes PMAD and thermal components. Source: [39]

the divergence of the beam. As mentioned previously, the beam divergence is not strictly a measure of efficiency, but it determines the size of the receiver and transmitter aperture, and thus represents a measure of the collection efficiency and affects sizing, mass, cost, and overall design.

These efficiencies are summarized in the WPT efficiency link budget shown in Figure 3.2. Note that geographical and seasonal effects are ignored, as well as circuit losses and small transmission losses associated with random failures, beam pointing, and phase matching.

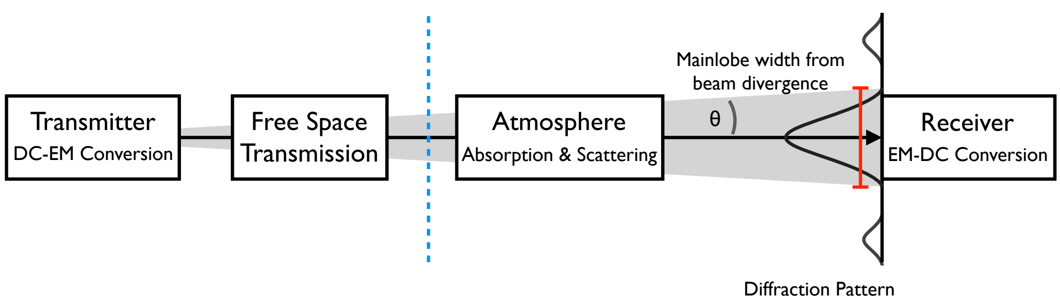


Figure 3.2: WPT Efficiency Link Budget (not to scale)

Sources of inefficiency for the WPT process only (no solar energy conversion or PMAD losses)

The corresponding efficiencies for each power beam type are presented in Table

3.7, along with the AHP rating. The following discussion provides the background and rationale behind these values.

Table 3.7: Power Beam Type - Efficiency

	Microwave Power Beaming		Laser Power Beaming
Source of Inefficiency	2.45 GHz	5.8 GHz	near-Infrared[1]
Transmitter (DC-EM)	~86%	86%[2]	~25%
Atmosphere (Space-Ground) (Transmittance)			
Clear Sky	98.9%	98.6%	45-70%[3]
Moderate Cloud Cover[4]	98.6%	98.5%	<1%
Light Rain[5]	97.6%	92.7%	<1%
Receiver (EM-DC)[6]	87%	82%	$\sim$ 60%
Divergence ($D_l << D_m$)	$2\theta = 0.244/D_m$		$2\theta = 3.18 \times 10^{-10}/D_l$
AHP Weight	**7**		**1/7**

Key: '–' = negligible effect (same as clear sky), '~' = approximate or expected value
[1] Atmospheric transmission windows only, no cloud boring
[2] Average of state-of-the-art klystron, magnetron, and solid-state microwave transmitters
[3] Mid-latitude, summer, rural, 5 km visibility.
[4] Cloud 1.5 km thickness, 0.3 g/m^3 water content
[5] ~4 mm/hr, 3 km cloud height, 4 km cloud thickness, 2.3 g/m^3 water content
[6] Assumes 100 W/m^2 incident on receiver for microwave power beam (irrelevant for laser)

1. WPT Hardware:

   The WPT hardware efficiency is primarily defined by the conversion efficiencies of the transmitter and the receiver. The conversion between electromagnetic waves and electricity is not 100% efficient and substantial heat is generated in this process. Furthermore, these conversion efficiencies are dependent on the power beam frequency, the converter technology, and, for microwave receivers, the incident power density.

   The hardware efficiencies for a microwave beam at either 2.45 GHz or 5.8 GHz are provided by [39]. An average of the three most common microwave generators is used to compute a transmitter efficiency at 5.8 GHz, and it is assumed that they operate similarly at 2.45 GHz. The peak observed efficiency at each frequency is used for the rectenna, assuming 100 W/m^2 irradiation (where the rectenna consists of an array of dipole antennas connected to Schottky barrier diode circuits, see Section 3.3.5).

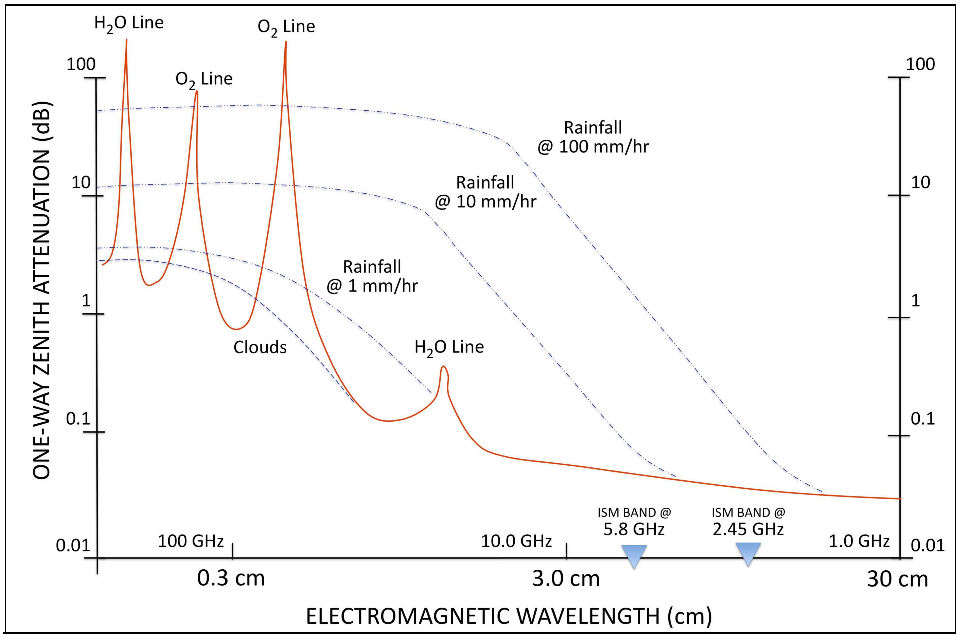
The hardware efficiencies for a laser beam are extrapolated from several sources. According to the DoD, current laser beam generators in the near-IR operate at 20%, and the goal is to improve to 30%.[17] JHPSSL operates at 19.3% efficiency[18] and [41] therefore assumes 25% efficiency for near-term SSP applications. Note that laser generators are highly dependent on temperature, and efficiencies of up to 85% (in the near-IR) have been demonstrated with systems cooled to -50 C°.[19] This technology and level of thermal management, however, is considered unrealistic for near-term SSP applications.

The laser-based WPT receiver is an array of PV cells whose peak receptivity is tuned to the power beam's wavelength to maximize efficiency. PV cells are also much more efficient with monochromatic light, as is the case for a laser beam. The Type II SSPS concept from[41] utilizes tailored bandgap PV cells with expected efficiencies of 60%.

2. Atmosphere:

   Atmospheric losses are due to either absorption or scattering from airborne molecules, and are largely dependent on the local water content (e.g. cloud cover and precipitation). The atmospheric attenuation in the microwave and near-IR regions is shown in Figures 3.3 and 3.4, respectively.

[17] www.irconnect.com/noc/press/pages/news_releases.html?d=202483
[18] www.laserfocusworld.com
[19] [23]



Credit: Figure Provided by Artemis Innovation Management Solutions, 2010, after NASA Ref. Pub. 1082(04), Feb. 1989

Figure 3.3: Atmospheric Attenuation at RF Wavelengths

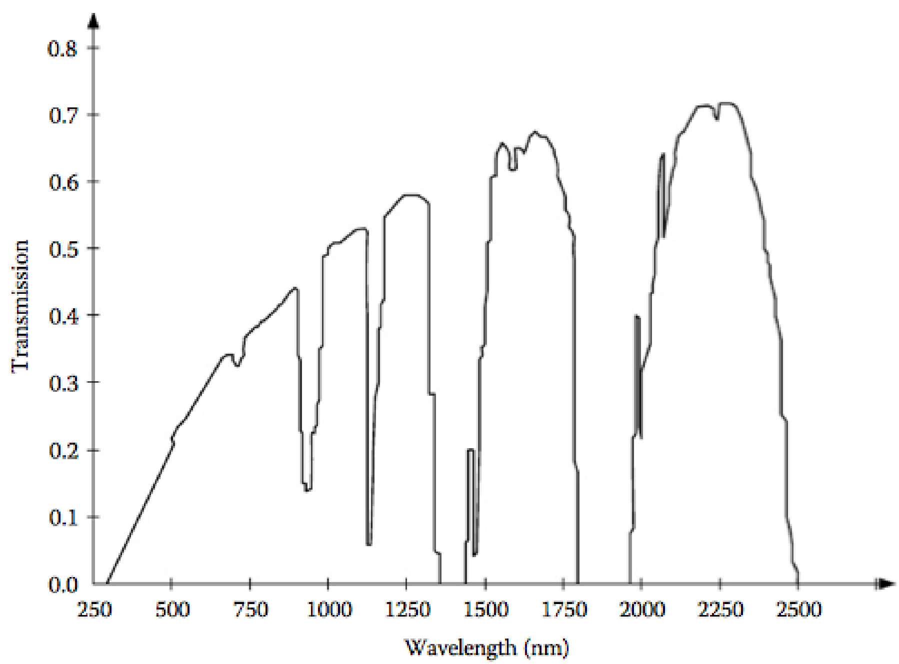


Figure 3.4: Atmospheric Transmittance in the Near-Infrared. Adapted from [54] (mid latitude, summer, rural, 5 km visibility).

Accurate atmosphere transmission efficiencies are computed from [54] and [10], under the assumptions listed in Table 3.7.

It is important to note that the majority of near-IR laser beam attenuation on a clear day occurs below altitudes of 0.5 km, so choosing an elevated receiver location can improve transmission efficiency. Furthermore, at high enough power densities it may be possible to bore through cloud cover, though these levels of densities far exceed current testing capabilities and safety thresholds.

3. Divergence:

The divergence of light is a result of diffraction and is measured as the angular spreading of light waves as they propagate through space. The divergence of each power beam type will be defined as the angular width of the beam mainlobe at the receiver site, referenced from the transmitter (Figure 3.2). It is assumed that all apertures are uniformly illuminated, circular, perfect lenses (aberration free), and the receiver is located in the far-field and lies parallel to the aperture (normal to the optical axis).

Under these assumptions, the angular mainlobe width of a microwave beam is given by the Airy Disk formula:

$$2\theta = 2.44\frac{\lambda}{D_m} \tag{3.1}$$

where θ is the angle (in radians) to the first minimum in the diffraction pattern (relative to the optical axis), λ is the wavelength, and D_m is the transmitter aperture diameter. For a microwave transmitter, the aperture width is the same as the transmitter width.

The laser power beam is a coherent, collimated light beam so its angular mainlobe width is given by:

$$2\theta = \frac{\lambda}{\pi D_l} \tag{3.2}$$

where technically D_l is the beam waist width as it emerges from the laser, but is equivalent to the laser aperture size for the purpose of this comparison study. Unlike the microwave transmitter, the laser aperture is a small section of the laser transmitter. In fact, the beam waist of a WPT laser beam is an order of magnitude smaller than the aperture of a microwave WPT system; and the larger the aperture, the smaller the divergence. The Firestrike laser module, for instance, outputs 15 kW from a circular aperture several centimeters across, while an equivalent 15 kW microwave aperture would measure about 1 m across. This is still not enough, however, for the microwave beam divergence to even approach that of the laser beam.

The divergence angles are compared in Table 3.7 with λ on the order of 10^{-1} m for the microwave beam, and 10^{-8} m for the near-IR laser beam.

It is clear from Table 3.7 that microwave WPT is far more efficient at converting and transmitting the power beam. In particular, microwaves are hardly affected by atmospheric features like clouds and rain, hence their widespread use in space-ground communications and radar. Conversely, the near-IR bandwidth is especially sensitive to these features, with nearly 100% transmission loss under even moderately inclement weather. While weather effects can be partially negated by choosing a proper receiver location, even on a clear day the laser beam suffers substantial losses.

The one important advantage to laser power beaming is the extremely low divergence due to both the smaller wavelength and the light collimation. And the greater the transmission distance, the more this feature becomes valuable; at orbital distances, laser-based SSPS systems can have transmitters and receivers on the order of meters rather than kilometers in diameter (as would be the case for microwave beaming), for power levels in the hundreds of megawatts. Aside from reducing the receiver's land-area requirements, the size reduction of the space-based transmitter greatly decreases the launch payload volume. This has a significant impact on cost and launch logistics (i.e., entire system could be launched on one vehicle). In addition, the smaller aperture of the laser beam reflects its ability transmit power at much higher power densities than microwave systems (safety notwithstanding).

In the end, however, the space-to-ground transmission efficiency of a near-IR laser beam is too low and too sensitive to atmospheric conditions (i.e., unreliable) for practical SSPS systems, even given the low beam divergence. This is evidenced by the majority of laser WPT research focused on short range applications. Space-based WPT is therefore almost constrained to use microwave beaming for significant and reliable power delivery. The microwave power beam is given an efficiency AHP rating of **7**, corresponding to a laser power beam rating of **1/7**.

5. *Extensibility* The extensibility of a microwave or laser-based SSPS concept is determined by evaluating 1) its potential for upgrades or conversions, and 2) its degree of modularity, especially within the context of scalability.

   Without specifying the nature of the upgrades and conversions, it is assumed that these changes principally require hardware changes that necessitate the same effort and type of work, regardless of the power beam type. Then, the system extensibility is measured primarily by the degree of system modularity.

In the context of the power beam type, the modularity of the WPT element (both the space and ground segment) is of greatest concern; the other critical element being the space-based solar arrays that are independent of the power beam type and inherently modular.

The degree of modularity of both microwave and laser-based SSPS concepts is largely dependent on the architecture and physical configuration or design of the system. Generalizations can be made, however, and the analysis is made easier by considering the three SSPS Types discussed previously from [41].

The microwave-based sandwich-type SSPS systems (Type III) are designed with scalability in mind, so called "hyper-modular" because they exhibit an extremely high degree of modularity across all components (not just the WPT element). Both the transmitter and rectenna are made up of vast arrays of individually operating modules that interface with correspondingly modular PMAD networks and structures. While the original 1979 SSPS concept (Type I) utilizes a similar modular approach with the transmitter and rectenna, support subsystems like PMAD, TMS and structure, were not designed with the same degree of modularity in mind, hence this concept has a relatively low degree of modularity.

Using the Type II SSPS concept from [41] as the basis for the laser-related analysis, a medium to high degree of modularity is observed. Like the Type III microwave concept, this laser-based system is designed to be modular by forming a "semi-independent collection of incoherently combined laser WPT transmissions on a distributed field of bandgap tailored PV arrays" ([41]). However, though the Type II and Type III concepts share several similar architecture features, the microwave-based design incorporates much greater levels of modularity. Furthermore, while the Type II concept is found to be technically feasible, substantial R&D is still required, so there remains a high degree of uncertainty in the actual modularity of such a system.

Given the preceding discussion, the Type III microwave concept is the most modular, followed by the Type II laser concept, and finally the Type I microwave concept. [41] confirms this ranking and further considers each concept to be separated by an order of magnitude in degree of modularity. Favoring the more modern approach of the Type III concept over the Type I, microwave power beaming becomes the preferable choice in extensibility over laser power beaming. The microwave power beam is thus given an AHP rating of **4** in extensibility, corresponding to rating of **1/4** for the laser power beam.

Phase 3: Final Score

With the comparative AHP ratings for each FoM assigned, the AHP analysis can be completed and the final score is computed, as shown in Figure 3.5.

Results

As there were only two design alternatives and microwave power beaming was more preferable in every FoM, the results were as expected: the microwave power beam scored an **84%**, with the laser at **16%**. While the microwave power beam could be predicted to be the more favorable of the two beam types, the actual degree or measure of preference of the microwave beam over the laser beam could only be assessed via the previous analysis. This analysis is also crucial to understanding the challenges associated with microwave power beaming, despite its comparative advantages.

In sum, the technological immaturity (low TRL and high R&D[3]) of laser WPT makes it a technically difficult design solution for the SSPS concept. Consequently,

TRL/R&D3

	Microwave	Laser
Microwave	1	6
Laser	0.167	1
Column Sum:	1.17	7.00

Normalized Values

			Score
Microwave	0.8571	0.8571	**0.8571**
Laser	0.1429	0.1429	**0.1429**

Space-based WPT Mass

	Microwave	Laser
Microwave	1	8
Laser	0.125	1
Column Sum:	1.13	9.00

Normalized Values

			Score
Microwave	0.8889	0.8889	**0.8889**
Laser	0.1111	0.1111	**0.1111**

Safety

	Microwave	Laser
Microwave	1	4
Laser	0.250	1
Column Sum:	1.25	5.00

Normalized Values

			Score
Microwave	0.8000	0.8000	**0.8000**
Laser	0.2000	0.2000	**0.2000**

Efficiency

	Microwave	Laser
Microwave	1	7
Laser	0.143	1
Column Sum:	1.14	8.00

Normalized Values

			Score
Microwave	0.8750	0.8750	**0.8750**
Laser	0.1250	0.1250	**0.1250**

Extensibility

	Microwave	Laser
Microwave	1	4
Laser	0.250	1
Column Sum:	1.25	5.00

Normalized Values

			Score
Microwave	0.8000	0.8000	**0.8000**
Laser	0.2000	0.2000	**0.2000**

Power Beam Type Final Score

Microwave	**0.8401**
Laser	**0.1599**

Figure 3.5: AHP comparative FoM matrices for Power Beam Type Trade Study

A number greater than 1 indicates a preference towards the FoM in the row over the FoM in the column.

adequate technology development for laser WPT in the SSPS-TD mission is not foreseen as feasible within the next decade and a half. Furthermore, the combination of its extremely low atmospheric transmission efficiency and high mass make the laser power beam impractical for space-based, long range and high power WPT. Lastly, safety is a large concern, and one that would generate significant political opposition in the face of unknown environmental, biological, and military[20] risks.

In contrast, the microwave power beam is a technologically feasible, and potentially efficient method of performing space-based WPT that has been investigated ever since the first concepts of WPT and SSPS systems were thought up. As such, the SSPS-TD mission will utilize a microwave-based WPT element.

3.3 Microwave SSPS

Now that the microwave power beam type has been chosen, the physics and principles of the SSPS-TD power beam and WPT hardware can be discussed. This reveals important design considerations, like subsystem sizing and power density distributions, that are used together with the previous analyses to determine the beam frequency and the transmitter and rectenna types.

3.3.1 Principles of Power Beaming

The general physics and principles of the microwave beam are summarized here. For full derivations see Appendix A.

[20] e.g. Military treaties like the 1972 US-Soviet anti-ballistic missile treaty prohibit space-based defenses that have the ability to intercept long-range ballistic missiles.

Assumptions

All subsequent analysis of the WPT element and power beam will be made under the following assumptions:

- All apertures are circular (transmitter and rectenna)

  This simplifies many of the computations later on and is a standard assumption in the literature.

- The transmitter (antenna) acts as a perfect lens (aberration free)

  Lens aberration is beyond the scope of this project, and as an inherent property of the lens, it can be neglected this early on in the design process.

- Unless otherwise stated, the image or observation plane (rectenna plane) is parallel to the transmitter plane, i.e. normal to the optical axis (the axis that runs from the center of the antenna to the observation plane).

- The observation plane is always located in the far-field, or Fraunhofer region

  The far-field region is the region where the radiation pattern is independent of the distance from the transmitting aperture. If an antenna with diameter D transmits at wavelength λ, then a point at a distance R is in the far-field region if all of the following three conditions are met:

$$\begin{aligned} R &> 2D^2/\lambda \\ R &>> D \\ R &>> \lambda \end{aligned}$$

  For any SSPS system, R is the orbit altitude, and hence, extremely large. Therefore, the last two conditions are easily met. Furthermore, since λ will operate in the microwave range between $1-15$ cm, and D will not exceed 1 km. Then, the first condition is also met, and the far-field assumption is valid. Note also that most antenna feeds have well-behaved radiation patterns so that the far-field distance is not absolutely critical.[21]

Notation: In the following analysis, the subscript t denotes physical quantities at the transmitter site and the subscript r denotes physical quantities in the image plane, i.e., rectenna site.

Radiation Pattern and Encircled Power

The propagation of the microwave power beam is like that of any other ordinary electromagnetic radiation emitted from an antenna, and it is characterized by its radiation pattern. This is measured in terms of the irradiance or power density distribution (W/m^2) at some distance R from the transmitter; this is the radiant flux (energy per unit time) per unit area. The radiation pattern is used to determine the power density at any point on the ground or in the atmosphere, and can be integrated to find the encircled power, i.e., total power incident on a specific area (e.g. the rectenna).

Figures 3.6 and 3.7 show the schematic setup that will be used in the following analysis. The antenna is the microwave transmitter, and it acts as the transmission aperture or exit pupil of the beam. The rectenna acts as the receiver aperture and is located in the image or observation plane. The transmitter has radius a and area A_t, and it radiates a peak power density I_{t_0} at wavelength λ. Let (ρ, ϕ) be the polar coordinates in the transmitter plane, where ρ is normalized to a. Let $I_t(\rho, \phi)$ be the

[21] [44]

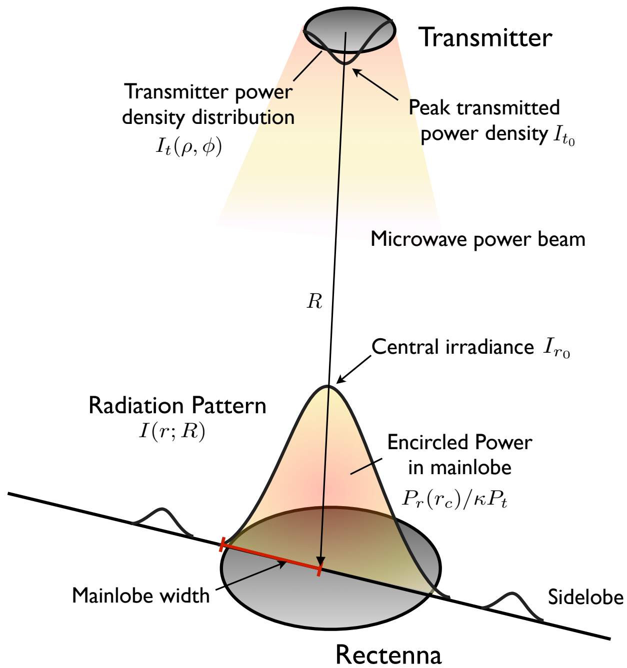


Figure 3.6: Power beaming system set up

Note that the mainlobe width is defined as half of the total width. This figure is not to scale.

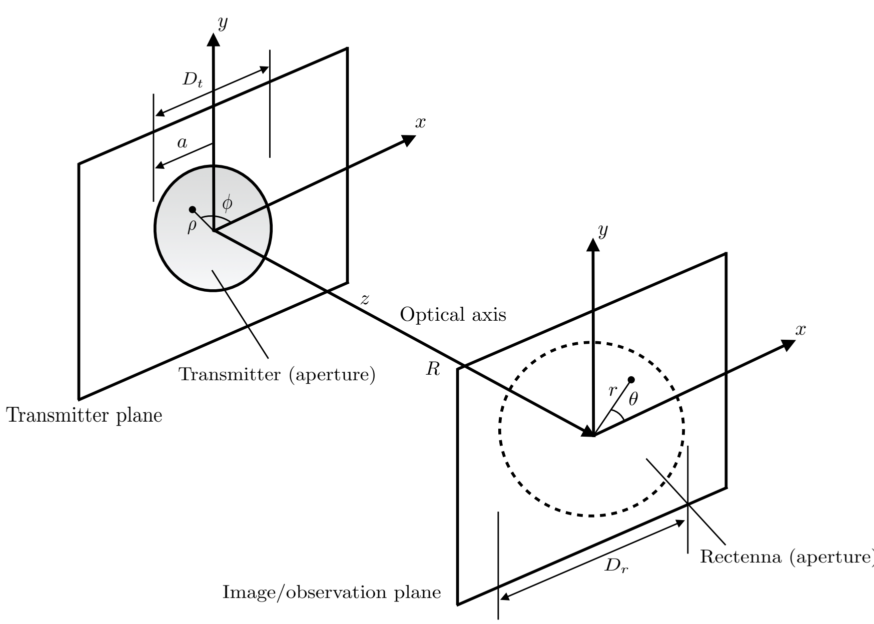


Figure 3.7: Power beaming optical set up and definitions (not to scale)

power density distribution across the transmission aperture, normalized to its peak value. Let R be the distance between the transmitter plane and the image plane. Let (r, θ) be the polar coordinates in the image plane, referenced from the point of intersection of the image plane and the optical axis. r is normalized by the factor $\lambda R/2a$.

Now, the radiation pattern of the microwave power beam in the image plane is given by its far-field diffraction pattern (Fraunhofer diffraction). This pattern is characterized by a mainlobe containing the majority of the transmitted power and a series of ever-decreasing sidelobes. The diffraction pattern is computed from the point-spread function (PSF) but it is first necessary to know the power density distribution across the transmitter, or $I_t(\rho, \phi)$.

Because the transmitter is made up of an array of microwave generators, each generator can be configured to output a different power so that a power density profile can be constructed across the transmitter (the total power transmitted is the integral of this distribution function). The most common profiles considered for SSPS concepts are uniform and Gaussian distributions, as shown in Figure 3.8.[22] For a uniformly illuminated transmitter, the diffraction pattern and encircled power correspond to that of the *Airy disk*, with well known analytical solutions. For a Gaussian illumination, the solution is obtained numerically and depends on the taper or truncation of the Gaussian profile. The taper is expressed in decibels as the power density at the transmitter edge, relative to the peak central power density I_{t_0}. At the microwave frequencies considered for the SSPS-TD missions, a 10 dB taper is recommended.[23] In practice, a Gaussian profile is approximated by a certain number

[22]Note that these two distributions are rotationally symmetric about the optical axis, which greatly reduces the computations.
[23][11], [39]

of discrete steps (e.g. 10-step 10 dB taper), as shown in Figure 3.8.

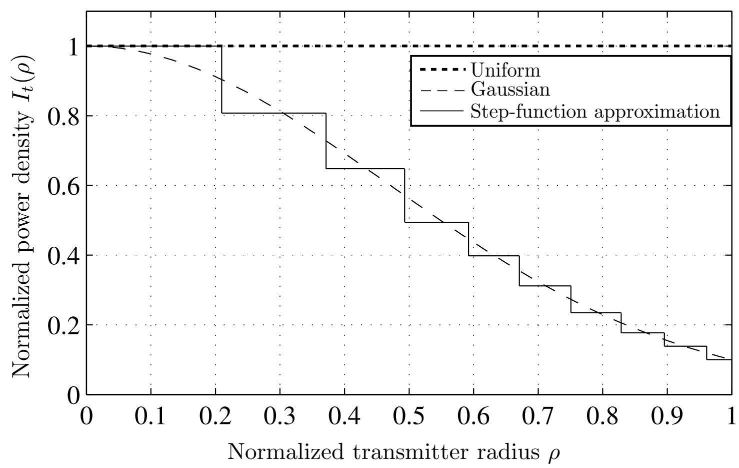


Figure 3.8: Normalized uniform and 10 dB Gaussian power density profiles across the transmitter.

The encircled power distribution is defined as the fraction of the total power P_t in the image plane contained in a circle of radius r_c (in units of $\lambda R/2a$), centered at $r = 0$. By setting r_c to the radius of the rectenna, the encircled power represents the amount of power collected by the rectenna. The atmospheric losses discussed in Section 3.2 are accounted for by assuming that the transmitted power, P_t, is actually κP_t, where κ represents the atmospheric transmittance (κ=1 for no losses). The encircled power is therefore $P_r(r_c)/\kappa P_t$.

The radiation pattern (irradiance function), encircled power, and several other important features for each transmitter power density profile are listed in Table 3.8, based on [36]. For full derivations see Appendix A. The radiation pattern and encircled power for each is plotted in Figure 3.9 for equal transmitted power.[24]

[24]Note that in reality P_t will be different for each illumination type but this only changes the amplitudes of the irradiance, not the shape, lobe widths or encircled power distribution.

Table 3.8: Microwave Power Beam Properties

	Transmitter Power Density Profile	
	Uniform	10 dB Gaussian Taper
Illumination Function, $I_t(\rho, \phi)$	I_{t_0}	$I_{t_0} e^{-2.303\rho^2}$
Transmitted Power[1] P_t	$I_{t_0} A_t$	$0.391 I_{t_0} A_t$
Irradiance Function, $I(r; R)$	$I_{r_0}\left[\frac{2J_1(\pi r)}{\pi r}\right]^2$	$\frac{\kappa I_{t_0} A_t^2}{\pi^2 \lambda^2 R^2}\left[\int_0^1 2\pi e^{-1.152\rho^2} J_0(\rho\pi r)\rho d\rho\right]^2$
Central Irradiance, I_{r_0}	$\kappa P_t A_t / \lambda^2 R^2$	$0.353\kappa \frac{I_{t_0} A_t^2}{\lambda^2 R^2}$
Peak Irradiance in 1st Sidelobe	$0.0175 I_{r_0}$	$0.0037 I_{r_0}$
Mainlobe width (in units of $\lambda R/2a$)	1.22	1.47
Encircled Power, $P_r(r_c)/\kappa P_t$	$1 - J_0^2(\pi r_c) - J_1^2(\pi r_c)$	$\frac{\int_0^{r_c} 2\pi I(r;R) r dr}{0.391\kappa I_{t_0} A_t}$
Encircled Power in Mainlobe[2]	83.8%	96.5%
Encircled Power to 1st Sidelobe	86.7%	97.0%

[1] Before atmospheric losses.
[2] Expressed as a % of the total power transmitted.
Note: J_x is the xth-order Bessel function of the first kind.

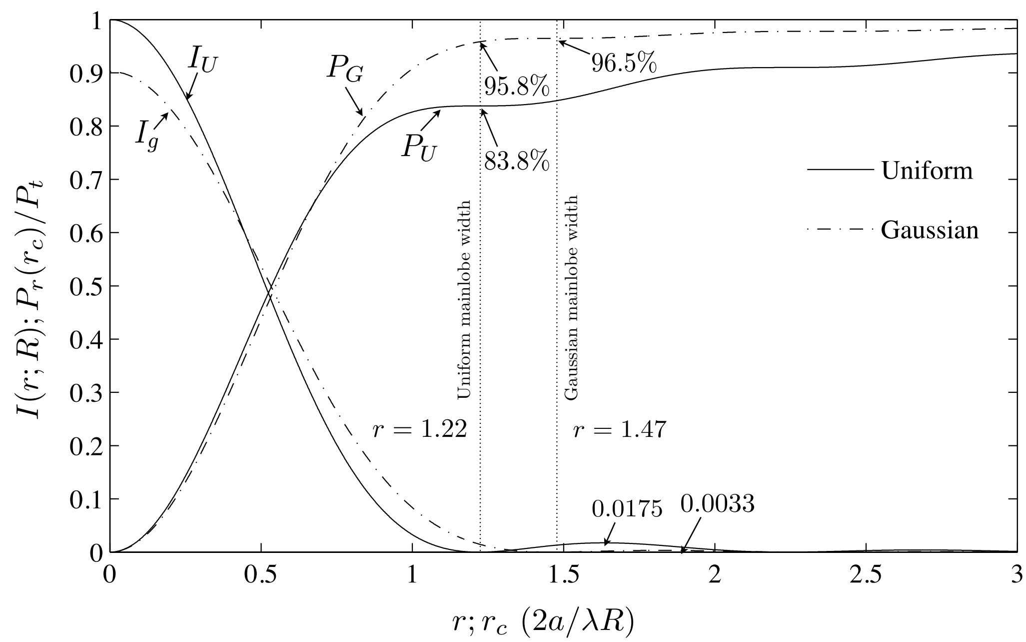


Figure 3.9: Transmitter Radiation Patterns and Encircled Power

Normalized radiation pattern and encircled power distribution for uniform and 10 dB Gaussian transmitter distributions, for equal transmitted power P_t ($\kappa = 1$). The subscripts $_U$ and $_G$ denote uniform and Gaussian distributions, respectively. The irradiance function is symmetric about the y-axis. Note that the transmitter for the Gaussian beam is 2.6 times larger than that for the uniform beam in order for equal power to be transmitted.

The equations in Table 3.8 reveal the following important features of the radiation pattern:

- Increasing R widens the mainlobe
- Increasing the transmitter size narrows the mainlobe
- Increasing λ narrows the mainlobe
- Changes in I_t or P_t only change the amplitude of the resulting radiation pattern at each point r

These features have important implications for WPT sizing, as will be seen further on.

It is immediately obvious that the effect of a Gaussian taper is to broaden the main lobe and lower the side lobe levels. These results offer two very important advantages:

1. *Increased power collection over the same area*: As shown in Figure 3.9, a rectenna sized to the mainlobe width of a uniformly illuminated transmitter ($r = 1.22$) would collect 12% more power with a 10 dB Gaussian tapered transmitter. In general, the main lobe broadening means that beyond a certain r, the encircled power is greater for the Gaussian profile than for the uniform, allowing more power to be collected over the same area.

2. *Safe side lobe levels*: As indicated in Table 3.8, a Gaussian tapered power beam results in a maximum sidelobe irradiation level an order of magnitude smaller than for a uniform beam. This is important in meeting radiation safety constraints outside of the rectenna site.

The main disadvantage to using a Gaussian power beam is that the total transmitted power is much less than that of a uniform beam for the same transmitter, as seen in the second equation of Table 3.8. For a 10 dB taper this is a nearly 40$ loss, which means that the transmitter for a Gaussian tapered beam would require an area 2.6 times larger than a transmitter for a uniform beam, just in order to transmit the same power. And even at this equivalent power, irradiance levels near the center are lower.

Now, because the SSPS-TD mission is a technology demonstration mission, it will be small in scale (resulting in small irradiance levels such that safety is essentially guaranteed) and maximizing the collected power is not important. With some foresight, it is also known that strict minimum incident power density requirements exist for the rectenna, (see Section 3.3.5), such that the transmitted power should be maximized for as small a transmitter as possible. Therefore, the SSPS-TD transmitter will generate a uniform power density distribution and a resulting **uniform power beam**.

3.3.2 WPT Sizing

The analysis in the previous section shows that the power beam characteristics are defined by: 1) a distance or orbit altitude R,[25] 2) a transmitter radius a, 3) the power beam wavelength, λ, and 4) a transmitter power density profile $I_t(\rho, \phi)$. Knowing these parameters (which is the goal of the rest of Chapter 3) then provides a means of sizing the WPT element, and ultimately the entire SSPS-TD system.

The transmitter size is defined by its radius, and must take into account a multitude of considerations and constraints because of both its central role in determining the ground irradiation pattern, and its impact on the satellite size and mass. This analysis is done in Section 3.7.

[25]Note that for non-geostationary orbits, $R = R(t)$

The exact rectenna size will not be specified in this study since the focus is on the satellite, and the choice of size is largely determined by the desired power to be collected (along with cost and construction logistics). Instead the general approach to rectenna sizing is discussed along with some rules and considerations.

The rectenna is sized based on the computed ground irradiation pattern and resulting encircled power distribution. From Figure 3.9, it is seen that the most cost-efficient rectenna size is near the mainlobe width. With the majority of the collectable transmitted power within the mainlobe, and the extreme diminishing returns on collected power past this point, it makes no economic sense to build the rectenna past the mainlobe. The mainlobe width thus represents an upper bound for the SSPS-TD rectenna size, is given by:[26]

$$D_r = \frac{1.22R\lambda}{a} \tag{3.3}$$

From Table 3.8 and Figure 3.9, a rectenna this size will collect 83.8% of the incident power (κP_t), with a zero power density at the rectenna edge.

Another reason this is an upper bound is that the rectenna has minimum power density thresholds for which it will operate, thus restricting the power density at the rectenna edge to these limits, as explained in Section 3.3.5. As a result, the rectenna must be designed and sized so that these thresholds are met across the entire rectenna. Since the ground irradiation pattern is dependent on the transmitter size and orbit altitude, rectenna sizing cannot be performed until after the orbit has been determined.

[26]This does not include the safety zones that should exist around the rectenna to reduce the risk of human exposure.

Given the impact of the four parameters listed above on the SSPS-TD system sizing, they are the most important factors in determining the feasibility of the SSPS-TD mission. And because they are restricted by environmental, design, and operational constraints, they cannot simply be chosen based on mission performance objectives. This poses an especially difficult challenge due to tradeoffs that exist between them.

The most important constraint is that of using one launch vehicle (Constraint 6), which restricts the space-based mass of a system that inherently needs to be large. The design feasibility study is thus centered on the determination of the power beam parameters in light of this issue. And due to their interdependence, all but the wavelength are determined together in the orbit trade study (Section 3.7), once the WPT hardware constraints and properties are all properly defined.

3.3.3 Frequency

The power beam frequency can now be chosen based on the options discussed previously (Section 2.1.7): 2.45 GHz, 5.8 GHz, and 35 GHz. Fortunately this decision does not necessitate an extensive trade study, but instead is more obvious, and can be determined using qualitative assessment based on the available data and literature.

As was done in the power beam type trade study, the 35 GHz power beam option is dismissed due to its low associated TRL, lack of available data, and its much lower transmission efficiency as indicated in Figure 3.3. Then, the 5.8 GHz option can be compared with the 2.45 GHz option for several key criteria. Due to the nature of this comparison, it can be conducted largely qualitatively, and the results are shown in Table 3.9.

It is evident that a 5.8 GHz power beam is preferable to a 2.45 GHz beam. The most important advantages are the smaller transmitter size, which reduces space-

Table 3.9: 5.8 GHz vs 2.45 Ghz: "A 5.8 GHz power beam has..."

Criteria	5.8 GHz compared to 2.45 GHz Power Beam
Transmitter/Rectenna Size	Smaller transmitter size for same mainlobe width, or smaller mainlobe width for same transmitter size (Eq. 3.3)
MW Generator Efficiency	Slightly higher DC-RF conversion efficiency but higher waste heat output that must be dealt with
Rectenna Efficiency[1]	Slightly lower RF-DC conversion efficiency ($\sim$ 5% difference at 100 W/m^2)
WPT Hardware TRL	Equal
Atmospheric Effects[2]	- Slightly lower transmittance in poor weather conditions - Order of magnitude greater threshold for ionosphere power density constraint - Scattering and scintillation effects significantly less (up to 10 dB amplitude variations expected at 2.45 GHz during a severe geomagnetic storm)

[1] [39]
[2] [48]

based mass (and hence cost), and the significantly lower atmospheric effects (ignoring the small penalty in transmittance). Therefore, SSPS-TD will utilize a **5.8 GHz** power beam.

With the frequency chosen, the ionosphere power density limit (Constraint 3) is I_{TSI} = 425 W/m^2. The transmittance factor κ will be between 0.927 and 0.986 for mild to moderate weather conditions, according to Table 3.7, and will be more precisely defined once the rectenna site location has been determined (Section 3.4).

3.3.4 Trade Study: Transmitter

The transmitter's role is to convert incoming DC into RF (microwave radiation), and transmit this power in a controlled manner with minimal losses. For SSPS-scale WPT application, this is achieved by using a phased-array antenna as the transmitter.

A phased-array antenna is necessary in order to distribute the RF power across the transmitter aperture and allows the power beam to be electronically steered. Electronic steering is achieved by varying the relative phases of the radiating elements (the MW generators) so that a constructive/destructive interference pattern is created in the desired direction. This removed the need for any mechanical steering to point the power beam, which is an important advantage for the large-scale structure of the SSPS concept, and also allows the beam to jump from one target to the next without sweeping. The disadvantages are that the antenna structure is quite complex, and due to the physics of phased-array technology, the maximum achievable Field-of-View (FOV) is 120°. This becomes an important constraint to the satellite-rectenna LOS and resulting active mode coverage.

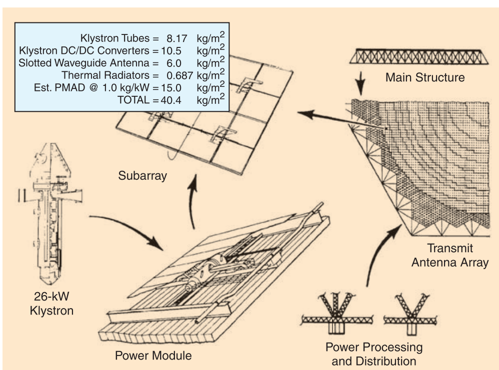
The phased-array antenna is comprised of arrays of microwave converters that are organized into modules supported by independent PMAD, thermal, and structural components. As part of the PMAD system, input DC arrives from the solar arrays where some DC-DC conversion is needed to supply the required voltage for the microwave generators; the circuitry for this DC-DC conversion is highly efficient, and can easily reach 98%.[27] Next, the MW generators execute the DC-RF conversion and output 5.8 GHz radiation that is configured via phase controlling to create the desired power beam. The power (W) of this radiation is dependent on the generator device and the thermal management subsystem (TMS). Associated with the MW generators is the DC-RF conversion efficiency, where inefficiencies are generated as waste heat that must be managed by TMS.

In general, the output power capabilities of the MW generators are greater than the TMS capabilities, so the latter drives the transmitter's maximum output.

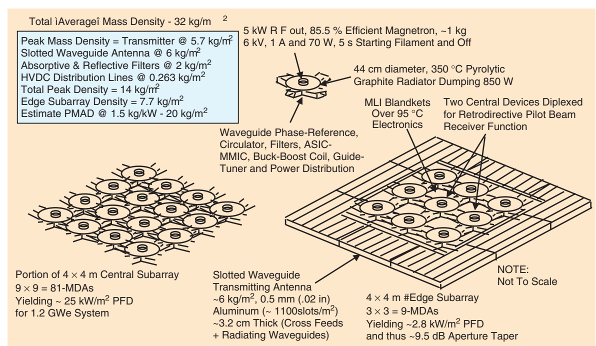
[27] [45]

This translates into a maximum allowable power density for the transmitter. In the original 1979 NASA study, a conservative value of 23 kW/m^2 is used, which [39] updates to 26 kW/m^2. Due to the lack of further data, 26 kW/m^2 will be used for the SSPS-TD peak transmitter power density, but it should be noted that this is a highly conservative estimate.

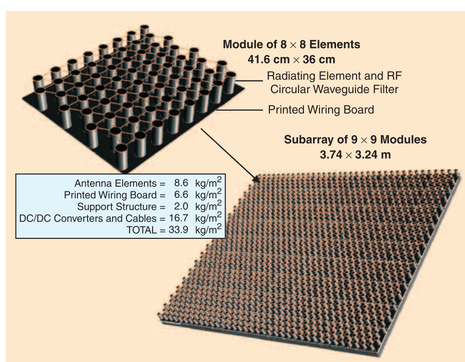
With the basic principles of the microwave transmitter outlined, the three most common MW converters for WPT applications are examined. SSPS-based setups for each converter type are shown schematically in Figure 3.10 and Table 3.10 compares their performances parameters.



(a) Klystron transmitter



(b) Magnetron directional amplifier transmitter



(c) Solid-state transmitter

Figure 3.10: Transmitter Types

Adapted from [39]

Table 3.10: 5.8 GHz Transmitter Comparisons

	Klystron	**Magnetron**	**Solid-State**[1]
Max converter power output (W)	26,000	5,000	59
Converter RF-DC efficiency	83%	85.5%	90%
Converter mass (kg)	14.15	1	0.001
Transmitter specific mass (kg/m^2)	40.4	32	33.9
Max transmitter specific power[2] (kW/m^2)	26	26	26
Operating temperatures	300°C on tube body 500°C on collectors	350°C on radiator	300°C at junction
Converter operating voltage (V_{DC})	28,000	6,000	80
Lifetime[3]	25 years	50 years	400 years (MTBF)[4]

[1] GaN-based alloy.
[2] Limited by thermal constraint.
[3] Based on ground tests.
[4] [49]
Source: [39]

- **Microwave Tubes**

  Conventional microwave power production is performed using tube technology that dates back many decades. This technology operates using thermionic emission: A voltage is used to accelerate free electrons through a vacuum tube where part of their energy is transferred to an RF field carried by a microwave structure. There are many ways to generate and carry this RF field, and each corresponds to a different microwave tube type. The two most common microwave tubes considered for WPT applications are examined:

  1. *Klystron* - The klystron is a linear microwave beam tube and the most common microwave power source in use. The electron beam passes through a series of resonant cavities separated by narrow drift tubes that cause the electrons to accelerate and decelerate, releasing electromagnetic energy into

the cavities that matches the frequency of an initial RF drive signal. This energy is contained in the cavities where it is extracted by a waveguide, and fed into the power beam. The waveguide is a structure that is geometrically designed to confine wave propagation at a specific frequency and carry it to a desired location. For WPT applications, a slotted waveguide array is used, in which each klystron tube feeds many radiating slots.

2. *Magnetron* - A magnetron is a circular diode with a magnetic field parallel to its axis. Resonant cavities are placed along the outer edge and oscillations occur when free electrons are given specific values of angular velocity. Like the klystron, the resulting RF energy is carried away using a slotted waveguide array. Filters are used to suppress noise.

- **Solid-State Microwave Devices**

  Solid-state amplifier circuits can be used to produce microwaves with high efficiency, and represent the latest state-of-the-art technology. A solid-state transmitter places a 5.8 GHz power amplifier and phase shifter behind every radiating element and microwave filters are needed to suppress carrier noise and harmonics generated by the amplifier.

  For WPT applications a GaN-based alloy is most suited for the power amplification. However, the TRL of a GaN-based solid-state phased-array transmitter is still low, but according to [39], the necessary R&D is achievable within the SSPS-TD time frame.

  The biggest advantages to solid-state devices are a much higher component reliability than tube technology, and, since the solid-state array is essentially "thin-film" technology, it allows for far more flexible SSPS design. Of great interest, for instance, is the concept of an integrated transmitter and solar array which would result in significant mass savings.

Like the frequency choice made previously, the transmitter type can be selected directly. Given the discussion above and the parameters in Table 3.10, a **solid-state microwave transmitter** is the obvious preferred choice for the SSPS-TD design. With this choice, the full transmitter link budget is shown in Table 3.11.

Table 3.11: 5.8 GHz Transmitter Efficiency Linkbudget

Sources of Inefficiency	**Efficiency**	**Notes**
DC-DC conversion	0.980	
DC-RF conversion	0.900	GaN solid-state amplifier
Subarray random electronic failures	0.980	Estimated 1% failures
Amplitude error	0.996	± 1 dB amplitude deviation
Phase error	0.978	$\pm 15°$ phase deviation
Phase quantization	0.997	5-b phase shifter
Taper quantization	0.989	10 steps
Aperture efficiency	0.980	Conductive losses in aperture
Transmitter efficiency	**81.4%**	(propagation losses are next)

Source: [39]. Assumes no transmitter scan loss, no mismatch loss, and negligible meteorite hits.

As a final note, in order to generate at least 100 kW of power the transmitter must be over 3.8 m^2. ADCS and station keeping considerations must account for solar radiation pressure and the reaction force of the power beam on the this surface area.

3.3.5 Rectenna

The role of the WPT receiver is to collect incident power and convert it to DC power. To do this, it must be able to handle high power densities, be unaffected by radio interference, have high efficiency, and be able to passively radiate waste heat, in addition to being reliable, light weight and cost-effective. The rectenna concept meets all of these criteria and consists of an array of individual rectenna elements linked via PMAD systems to output DC power. Though the overall design and size of the rectenna is beyond the scope of this study, it is important to understand the principles behind its operation in order to better formulate the concept of operations, requirements, interfaces, and mission performance .

A rectenna element is shown schematically in Figure 3.11. Microwaves are first collected by an antenna, that is typically either a dipole or patch antenna. These waves then pass through a Schottky-barrier diode that acts as the rectifier, performing the RF-DC conversion. The HF filter ensures impedance match between the rectifier and the antenna for optimal power transfer and the output DC filter smooths the output DC voltage and current by attenuating noise from harmonics generated by the RF signal and the nonlinear rectification process. The final stage is the DC-grid conversion, which is treated as a separate phase outside of the rectenna system.

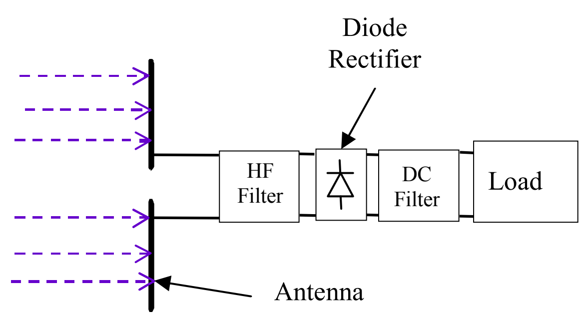


Figure 3.11: Schematic of rectenna element. Adapted from [42]

The efficiency of the rectenna is measured by the RF-DC conversion, or rectification, efficiency. The diode is the main source of inefficiency, and because of its presence, the conversion efficiency is dependent on the incident power level. Furthermore, due to the voltage drop in the forward direction of the diode, the efficiency is a nonlinear function that drops off quickly at low incident power densities. Measured performance data is available and shown in Figure 3.12. The most important consequence of this feature is that rectenna elements near the edge of the rectenna may have low or even 0% operating efficiencies unless this is considered.

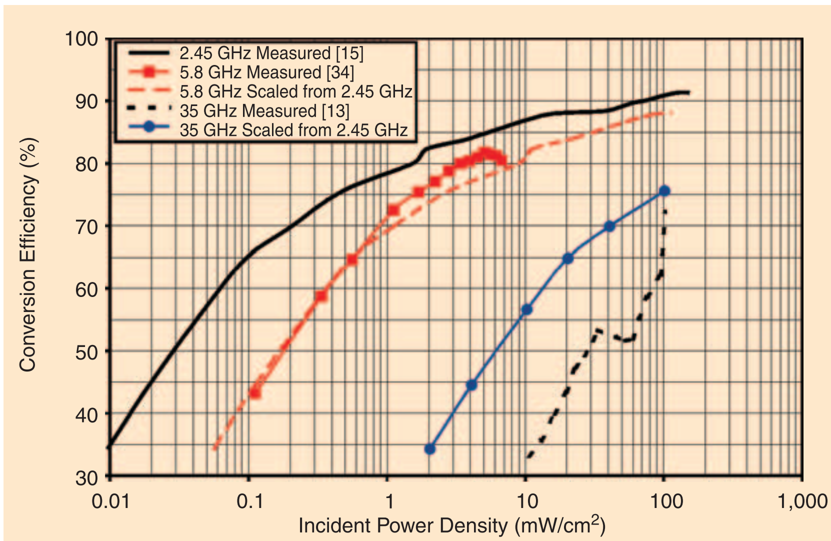


Figure 3.12: Rectenna efficiencies as a function of incident power density. Source: [39]

This effect suggests that there exists a power density threshold under which the rectifier will not be activated, and no energy conversion will occur. This diode activation minimum is given by:[28]

$$I_{d_{min}} = \frac{V_0^2 4\pi}{R_L \lambda^2 G} \tag{3.4}$$

where I_d is the incident power density on the rectifiers (W/m^2), V_0 is the voltage drop across the diode (the minimum voltage to turn it on), R_L is the antenna impedence, λ is the RF wavelength, and G is the antenna gain.

[28] [5]

Another limitation exists because the parasitic capacitance of the diode degrades the conversion efficiency. This translates into another diode activation minimum:[29]

$$I_{d_{min}} = \frac{C_d V_0^2 f^3 4\pi}{c^2 G} \tag{3.5}$$

where C_d is the capacitance of the diode, f is the RF frequency, and c is the speed of light.

Two constraints on the minimum incident power density therefore exist, and the greater of the two values must be taken as the limit. However, at this limit, the diode is turned on but it will not yet deliver any power to the system, so the actual threshold should be about an order of magnitude greater than the values in Eq. 3.4 and 3.5, according to [5]. Together, these thresholds are one of the most important constraints for small-scale SSPS demonstrations like SSPS-TD since the transmitter must be sized to generate a radiation pattern at high enough power densities across all of the rectenna. It is therefore essential to choose a rectenna design with parameters that maximize $I_{d_{min}}$.

These constraint equations, however, assume that each antenna feeds its own diode. In practice, many antennae can be linked to feed a common diode and thereby increase the collection surface of each diode. Furthermore, parabolic dishes can track the orbiting transmitter and concentrate the incoming power flux onto the rectifying elements. The appropriate rectenna design can therefore negate these constraints and allow each rectifier to be fed the required power for optimal conversion efficiency. Then, the only requirement is that the power incident on each rectifier is higher than the background noise (e.g. thermal noise). From [5], this threshold is assumed to

[29] [5]

be 100 mW, and given the design considerations above, a minimum incident power density constraint of $I_{r_{min}} = 25$ mW/m^2 is assumed to be reasonable.

There are many types of rectennas and they mainly differ in antenna type and array configuration. However, the rectenna design is beyond the scope of this study, so a feasible 5.8 GHz capable rectenna is generated for the SSPS-TD design, using data from [39] and [42]. The result is a printed dipole rectenna using zero bias Schottky diodes. At 5.8 GHz, the dipole length (both sides) needs to be 0.025 m, which allows for dozens of dipoles per m^2.

It will be further assumed that the rectenna will be divided into subarrays and they or the parabolic concentrators will be attached to mechanically steered structures that move with the transmitter. They will track the transmitter such that the effective rectenna plane (really a series of sub-planes all facing the same direction) can always be treated as parallel to the transmitter plane, thus minimizing power collection loss. Note that this mission configuration results in discrete rectenna elements that are at different distances from the transmitter array. This however, is nearly negligible due to the extremely large satellite-rectenna distance, and further irrelevant in light of the parabolic concentrators which ensure minimum incident power densities.

A pilot beam will also be emitted from the rectenna to the satellite in order to calibrate the transmitter for power beaming. This has the added benefit of acting as a fail-safe for a misdirected power beam; if the transmitter does not receive the pilot beam, it will not activate.

The relevant rectenna properties are displayed in Table 3.12 and the full rectenna link budget in Table 3.13. The physical design of the rectenna is beyond the scope of this study, but it is assumed that it will be designed to perform near the peak measured efficiency, i.e. ∼80%. The mass properties are also unknown, but are not really a concern since the rectenna is ground-based. For completeness, the

Table 3.12: Rectenna Description

Rectenna Type	Measured Peak Conversion Efficiency	Peak Output Power per Element (W dc)	C_d (pF)	R_L (Ω)	V_0 (V)	G
Printed dipole, zero bias Schottky diode	82.7%	0.052	0.18	50	0.150	1.5

Sources: [39], [42]

Table 3.13: 5.8 GHz Rectenna Efficiency Linkbudget

Sources of Inefficiency	Efficiency	Notes
Rectenna reflection loss[1]	0.980	
Rectenna random failures	0.990	Estimated 1% failures
RF filter insertion loss (IL)	0.891	Estimated IL= 0.5 dB
RF-DC conversion	0.800	Assumes optimal design (multiple antenna per diode and parabolic concentrators)
Rectenna efficiency	69.2%	

Note: Rectenna mechanically steered to track transmitter so that incident radiation perpendicular to rectenna plane.

[1] [4]

Source: [39]. Assumes no rectenna scan loss and no mismatch loss.

minimum power density thresholds can be computed from Eqs. 3.4 and 3.5 (with an added order of magnitude):

$$\text{Diode activation min due to voltage drop: } I_{d_{min}} = 14.1 \text{ W/m}^2$$

$$\text{Diode activation min due to parasitic capacitance: } I_{d_{min}} = 0.74 \text{ W/m}^2$$

The voltage drop effect is the dominant effect at 5.8 GHz, and therefore $I_d \geq 14.1$ W/m^2 in order for the rectifiers to operate. As discussed, a 25 mW/m^2 minimum incident power density is established, that must then be collected to meet this diode activation minimum. Note that this incident power density is well under the general population safety limit of 10 W/m^2 stated in Constraint 4, Table 2.5.

As a final note, recall in Section 3.3.2 that a rectenna sized to the airy disk mainlobe width would receive 2% ($I_{r_{edge}}/I_{r_0} = 0.02$) of the central irradiance at its edge. Because of this, the rectenna needs to be carefully designed so that the rectifiers at the rectenna edge are still supplied with sufficient power to efficiently operate.

3.4 Rectenna Location

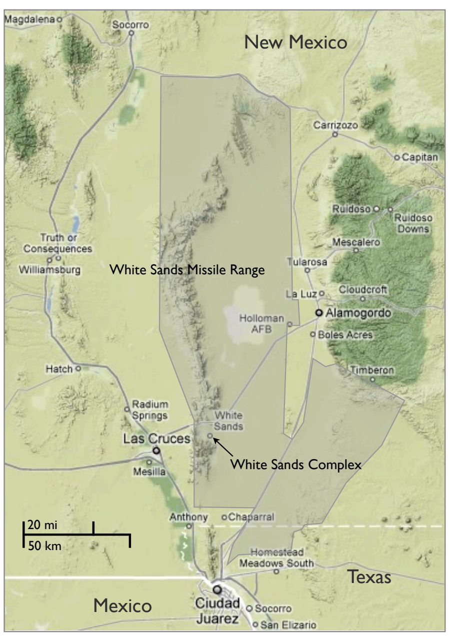
The rectenna location is an important feature of the SSPS-TD architecture definition because its latitude constrains the choice of SSPS-TD orbit inclinations and altitudes. This decision is also necessary in order to specify local environmental, atmospheric, and safety parameters, and further develop the concept of operations.

The following requirements are imposed on the rectenna location:

- Continental U.S. - From Assumption 4 in mission scoping.
- Large, flat, plain ground - The rectenna is a potentially large system (the 1979 NASA concept was a 5 GW system in GEO with a 10 km rectenna diameter), and in addition, diffractive spreading at the ground of up to several kilometers exists due to linear scattering and scintillation. This effect necessitates a safety zone around the rectenna, further increasing its footprint. Furthermore, construction of the rectenna must be feasible, hence the need for flat, plain ground.
- Low population density - Minimizes the risk of beam exposure and interference. This includes the need for minimal local infrastructure, like high ways.
- Low biota - Minimizes environmental impact of the rectenna (e.g. no endangered species).

- Accessible - The rectenna must be accessible to ground personnel and have LOS to the satellite. No hills, mountains, valleys, forests, or waterways should exist as obstacles.
- Benign climate/environment - Though a 5.8 GHz power beam is relatively unaffected by weather conditions (Section 3.2), harsh weather like snowstorms, windstorms, and heavy rain, can degrade the rectenna hardware. Regions of seismic activity should be avoided as well.
- Power grid proximity - The rectenna should be relatively close to the local power grid so that it can more easily feed it, without the need for extensive transmission line construction.

One particularly suitable rectenna location is White Sands, New Mexico, shown in Figure 3.13. Relevant information is listed in Table 3.14.



(a) White Sands Territory (Adapted from Google Maps)

(b) White Sands Complex (NASA)

Figure 3.13: White Sands, New Mexico

Table 3.14: Rectenna Location: White Sands, NM

Name	White Sands
Lat, Long	32.38° N, 106.50° W
Area	160x65 km
Elevation	1.783 km
Authority	U.S. government
Primary Functions	Missile range, rocket testing, WSGT (TDRSS ground terminal), NASA testing facility
Terrain Type	Desert (dunes, minimal plant life)
Avg Annual Rainfall	8.4 inches
Avg Annual Snowfall	2.1 inches
Avg Annual Precipitation Days	42
Avg Annual Sunny Days	291
Avg Annual Temperature[1]	58.2° F
Population	982 (outside of restricted area)
Grid Proximity	Immediate local grid connection
Avg Atmospheric Transmittance at 5.8 GHz[2]	97.9%

[1] Average of max and min temperatures from each month.
[2] Weighted average based on the values in Table 3.7. Assumes 291 clear sky days, 42 light rainy days, and 32 moderately cloudy days.
Average values obtained from bestplaces.net (updated in 2011)

The data confirms the validity of this choice. As a government-owned testing ground, White Sands is well-suited for the SSPS-TD mission since it is a technology demonstration mission operated under federal authorities (Section 2.1.5). Furthermore, the climate is nearly ideal and the land is already devoted to government testing, satisfying environmental and safety concerns. The only drawback to this location are sandstorms, which though intermittent, can be quite strong and potentially damage or degrade the rectenna. Despite this, it is considered one of the best location options and the SSPS-TD design will therefore assume that the rectenna site will be constructed and operated in White Sands, New Mexico.

3.5 Trade Study: Solar Array Design

The SSPS concept relies on solar energy to provide sufficient power to the transmitter for nominal WPT operations. In addition, solar energy is also expected to provide the majority of the power for the other onboard subsystems during both active and inactive modes. As such, the SSPS-TD satellite will utilize arrays of photovoltaic (PV) cells[30] laid onto one or more solar panels or subarrays that are connected to the main satellite bus. Together with the associated PMAD components, the entire system is called the *solar array*.

The PV cells themselves are technologically independent from the panels, which act as the support structure for both the cells and the PMAD components (including the current collectors). In this way, solar cell types may be first investigated for potential SSPS use before the entire solar array is designed.

The goal of this design decision is to determine both the type of the PV cells and the material of the panels such that the fundamental parameters associated with the solar array mass, size, and performance are known. These parameters are necessary in order to size the entire SSPS-TD satellite given the multitude of constraints, as will be seen in Section 3.7.

The two most important metrics for the solar array design are the specific power (W/kg) and the volume of the packaged array when it is launched. These are critical to mission feasibility as they are the limiting factors to launch logistics in terms payload capabilities, especially given the constraint to fit on one launch vehicle (Constraint 6).

[30]For the purposes of this study, the terms *photovoltaic cell* and *solar cell* will be used interchangeably.

3.5.1 Solar Cell

A photovoltaic cell typically consists of a glass cover, an antireflective coating, the cell itself, and a backplate or substrate. The cell is primarily characterized by its individual EM-DC efficiency, size, and mass, which together provide the specific power, specific mass, and power density. Degradation is another important factor, but space-rated solar cells are generally designed not to degrade more than a few percent over several years, which is longer than the SSPS-TD mission lifetime. The cell technology TRL is also critical to this mission because the solar array is integral to the operation of the WPT element; it poses too much risk to utilize low TRL photovoltaics with unknown reliability.

The solar cell types considered for the SSPS-TD mission are displayed in Table 3.15. The performance parameters are given for the bare cell only. The act of integrating the solar cell into the solar array structure will worsen these values, but it largely depends on the panel or structure material. Note that degradation rates were not available, so ultimately this metric should be determined and compared across solar cell types before in later studies.

Despite their lower efficiencies, thin-film PV cells are the obvious choice due to their incredibly high specific powers and thin, flexible nature which allows for compact payload packaging. They are also far cheaper to produce, and, as will be seen in the next section, they are ideally suited to perform on ultra-lightweight structures. Due to the lower specific power and TRL level of CIGS cells[31], the SSPS-TD mission will utilize **thin-film a-Si:H photovoltaic cells**.

[31] The lower specific power is due to the fact that CIGS cells are usually deposited on 30 μm thick metal foil which is heavier than that used for a-Si cells.

Table 3.15: Comparison of State-of-the-Art Solar Cell Types

				Thin-film	
	High-efficiency Si	**Triple Junction (GaAs)**	**IMM**[1]	**a-Si:H**[2]	**CIGS**[3]
Features	rigid	rigid	thin, flexible	ultra-thin, flexible	
TRL	10 in common use	10 in common use	6 flown in SLATE	6 Flown in ISS	5 Lab cells tested
Efficiency[4]	10-12.5%	22-30%	24-30%	9-12.5%	12-20%
Specific Power (W/kg)	575	320	350 (>700 feasible)	4300	3000
Specific Mass (kg/m^2)	0.23	0.85	0.84	0.03	0.068
Power Density (W/m^2)	143	275	294	129	205
Scalability	Limited by launch volume	Limited by launch volume	Easy	Very easy	Limited by array design

All data given at EOL (end-of-life) after 1 year for individual bare solar cell.
Sources: [52], [47], [26], [55]
[1] Inverted Metamorphic
[2] Amorphous Silicon
[3] Polycrystalline Cu(Ga,In)Se2
[2] Measured at AM0 (in space).

3.5.2 Solar Array

The importance of the specific power metric and the choice of thin-film photovoltaics naturally leads to the selection of a solar array structure that is as light as possible. Fortunately, recent advances in PV technology has allowed for solar cells to be placed on thin-film substrates that act as the panel structure. These structures are non-rigid (e.g. rollable and foldable), allowing the solar array to be packaged extremely compactly in the launch vehicle, and are relatively cheap to produce. The most problematic issue for these materials, however, is erosion due to radiation and atomic oxygen.

Japan was the first to demonstrate this technology in space with the IKAROS mission in 2010. A 14 m x 14 m solar sail made of 7.5 icron thick polymide film was impregnated with thin-film solar cells and successfully deployed in space using a spinning motion that generated a centrifugal force.

One of the most promising solar array designs was first developed by the Neuchatel partners who developed a method to deposit a-Si cells onto 6 μm thick CP1 polymer film, referred to as CP1/a-Si:H arrays.[32] CP1 polyimide was developed by SRS technologies under a NASA contract and is space-rated for 10 years in GEO. Using this technology, Kayser-Threde and the German Aerospace Center (DLR) have developed a deployable, ultra-lightweight space-rated solar array, or solar sail. The deployment structure uses booms made of carbon-fiber reinforced polymers (CFRP) to unfold the array in a slow, linear manner that has been verified by NASA under vacuum conditions. A ground test of a full-scale array was successfully demonstrated on the ground by the European Space Agency (ESA) for a 20 m square array, as shown in Figure 3.14. The array had a 12 μm thick aluminized Myler and 7.5 μm

[32] [47]

Kapton polyimide. At this scale and with this configuration, the solar array delivers 68 kW while weighing only 32 kg, with a specific power of 2,125 W/kg and a specific mass of 0.08 kg/m^2. The payload volume would be the "size of a suitcase".[33] A larger 50 m square solar array based on this design should deliver 425 kW, weigh 75 kg, and have a specific power of 5,670 W/kg and a specific mass of 0.03 kg/m^2.

Figure 3.14: Deployed CP1/a-Si:H solar array with CFRP booms

Fully deployed 20 m square array at ESA/DLR European Astronaut Center in Cologne. The booms run across the diagonals and are deployed before the sail segments. Source: [47]

Though all together this solar array technology is only at TRL 5-6, the PV cells and polymer film are individually at much higher TRLs, and given the rapid development of the solar energy technology, this technology is considered feasible for the SSPS-TD mission. And while the actual physical configuration and deployment design of the solar array is beyond the scope of this study, these performance parameters provide the necessary basis for the SSPS-TD solar array design. The 20 m and 50 m CP1/a-Si:H solar array will therefore be used as modular subarrays for the satellite solar power (e.g. multiple subarrays of each to provide the necessary power). It is further assumed that the solar array will be mechanically steered to track the

[33] [47]

sun and provide maximum solar energy generation. The added weight of the PMAD hardware is assumed to be offset by a reduction in the polymer film thickness as the technology matures, but the mass of the steering mechanism is not included. This design choice is summarized in Table 3.16.

Table 3.16: SSPS-TD Solar Array

Solar Array Type	CP1/a-Si:H with CFRP booms	
Solar Array Efficiency (EOL)	12.5%	
Solar Cell Type	a-Si:H	
Sizes (square)	20 m	50 m
Specific Power (W/kg)	2,125	5,670
Specific Mass (kg/m^2)	0.08	0.03
Power Density (W/m^2)	170	170
Power Output (kW)	68	425
Total Mass (kg)	32	75

Projected performance from [47].

3.6 Satellite Size and Mass

For the purposes of this study, the SSPS-TD satellite can be divided into three segments: 1) the transmitter, 2) the solar array, and 3) all additional subsystems (TT&C, C&DH, GN&C, ADCS, etc.). With some foresight, it is known that the majority of the satellite surface area consists of the transmitter and the solar array, so that the satellite size can be approximated by these two subsystem sizes. The satellite dry mass is simply the sum of the masses of these three segments (with any added margins and reserves). Both of these values are critical to constructing a feasible mission under the existing constraints, foremost among these being the payload and deployment capabilities (i.e. how much mass can be launched and how big of a system can be deployed). More specifically, since the payload mass capability varies with orbit, and the satellite mass is directly related to its size, these values play a central role in determining a feasible orbit.

With the transmitter and solar array designs complete, the expressions for their respective masses and sizes can be generated. The masses of the additional subsystems will be computed in Chapter 4 after the PBS and subsystem specifications are defined. But, with some foresight it is known that the minimum transmitter size will still correspond to a mass far greater than these subsystems. This is a similar situation to the James Webb Space Telescope (JWST), where the payload is about 70% of the satellite dry mass. An upper bound for the SSPS-TD dry mass is thus obtained by requiring that the transmitter (i.e., the payload) not weigh more than 70% of the dry mass, including margins and reserves.

The conceptual method for computing satellite size and mass is discussed first, and then presented in mathematical form in Table 3.17.

- **Transmitter** - The transmitter size is given by its radius a (determined in Section 3.7), which gives the total area A_t. This is then multiplied by the transmitter specific mass $\rho_t = 33.9$ kg/m^2 to give the transmitter mass M_t. This mass does not include the support structure or deployment mechanism.

- **Solar Array** - The solar array is sized to provide the power required by the transmitter plus that needed by other onboard subsystems like telecommunications and C&DH. The power required by the transmitter is defined after all conversion losses so the 81.4% transmitter efficiency needs to be accounted for by supplying $1.23P_t$. Then, since the transmitter will require significantly more power (at least an order of magnitude greater) than the rest of the satellite, a margin of 10% is added to this to cover the power demands of the other subsystems, and any margins, reserves, and losses, namely due to power distribution and failures. Therefore, the solar array is sized based on the minimum number of 50 m and 20 m solar subarrays necessary to generate $1.35P_t$ Watts, with preference toward the 50 m subarrays (since they have higher specific power). The solar array mass M_s is then found by adding up the masses of the subarrays. This mass includes the structure and deployment mechanism, but not the steering mechanism.

- **Additional Subsystems** - As discussed, the mass of the additional subsystems, $M_{add.subs}$, is computed in Chapter 4. In general, they are expected to be very small relative to the transmitter mass, with the exception of the structure. Note that the majority of the power subsystem is accounted for by the solar array, but batteries and PMAD hardware must still be considered.

- **Margins** - Margins[34] are extremely important in mission design. They account for the large uncertainty that is inherent in a design, and are higher earlier in the project lifecycle. As the design matures, the mass estimate becomes more refined, and the margins can be reduced and distributed at across the system hierarchy. Margins are therefore critical to concept designs early in the project lifecycle, and are especially important for complex designs with low TRL technologies, like the SSPS-TD mission.

  A 10% power margin has already been added, and a 25% mass margin is added to the total mass to account for the immaturity of the design, as discussed in [35]. This mass margin is the highest suggested value, and used because the SSPS-TD mission design is treated at an early conceptual level.

- **Total Satellite Mass and Size** - By adding up the transmitter mass, solar array mass, and the extra 15%, an approximate total satellite dry mass, M_{dry} is

[34] *Margins* will refer to margins, contingencies, and reserves since there is no distinction this early on in the lifecycle.

found. The approximate size is computed as the surface area of the combined transmitter and solar array.

As evidenced by the equations in Table 3.17, the transmitter size is the key variable that allows the rest of the satellite to be sized. And because the transmitter has a much higher specific mass than the solar arrays (33.9 kg/m^2 vs 0.03-0.08 kg/m^2), minimizing the transmitter size effectively minimizes the satellite mass. Setting this variable is one of the most complex problems of the SSPS-TD mission design since it determines the performance of the WPT element, which in turn is bounded by several constraints, and depends heavily on the orbit altitude. The transmitter size is therefore determined in the orbit trade study.

It is, however, important to first establish constraints on the size and mass of the satellite. Though the actual physical configuration and deployment design of the satellite is beyond the scope of this study, and mass becomes the primary constraint for launch feasibility, an upper bound on sizing should still be determined.

The mass constraint is derived from Constraint 6 and is simply determined by the maximum payload mass capability of the launch vehicle to the desired orbit. Some values were given in Table 2.6.

A size constraint can be extrapolated from the mass constraint, but this would value would not take into consideration packaging and deployment considerations, as well as the context of a technology demonstration (e.g. small, low cost). Instead, a maximum size is derived as follows.

Since the solar array is sized based on the transmitter size, the focus of the satellite size is on a feasible transmitter size. Then, the following four factors are considered:

Table 3.17: SSPS-TD Size and Mass Equations

Subsystem	**Size** (m^2)	**Mass** (kg)	**Comments**
Transmitter	$A_t = \pi a^2$	$M_t = \rho_t A_t$	$\rho_t = 33.9$ kg/m^2. $M_t \leq 0.7 M_{dry}$ (including reserves). Support structure not included.
Solar Array	$A_s = 400n_{20} + 2500n_{50}$	$M_s = 32n_{20} + 75n_{50}$	$n_{50} = \lfloor 1.35P_t/425000 \rfloor$ (power needed divided by 50 m subarray output and rounded down) $n_{20} = \lceil (1.35P_t \mod 425000)/68000 \rceil$ (remainder of power needed divided by 20 m subarray output and rounded up) Includes main structure but not steering mechanism. Includes 10% power margin and accounts for transmitter losses.
Margins & Reserves	n/a	$M_{mar} = 0.25 M_{dry}$	25% margin/reserve.
Satellite Dry Mass	$A_{sat} \approx A_t + A_s$	$M_{dry} = M_t + M_s + M_{mar} + M_{add.subs}$	Includes margins & reserves.

1. Size and complexity should be minimized for a technology demonstration mission (thereby reducing cost as well).
2. The launch vehicle has maximum payload fairing dimensions of approximately 5 m x 23 m.
3. The transmitter is a modular, uniformly distributed structure, allowing for compact packaging and modular deployment.
4. The largest single-launch scientific satellite designed to date is the James Webb Space Telescope, which has a 6.5 m diameter mirror and a sun shade the size of a tennis court (about 12 x 24 m).

Given these factors, a feasible maximum transmitter diameter is assumed to be **15 m**. At this maximum, the transmitter weighs 6,000 kg and requires 4.6 MW. The solar array must then produce 6.21 MW, resulting in a 1178 kg array with an area of 36,600 m^2 (fourteen 50 m subarrays and four 20 m subarray, the equivalent of a 191 m square, or about 2 football fields).

The satellite size and mass are determined following the orbit trade study, and given in Chapter 5.

3.7 Trade Study: Satellite Orbit

The SSPS-TD mission has now been sufficiently defined for the satellite orbit trade study to be performed. This trade study will incorporate many of the preceding results from Chapter 3, and is driven by the question of mission feasibility. The result will complete the architecture baseline definition and determine some of the most important operational features of the mission.

Objective

The primary objective is the selection of the orbit altitude and inclination. There exist a variety of important operational features that are functions of, or connected with these two parameters, and must be defined:

- The Keplerian orbital elements of the satellite orbit, and relevant periods and velocities
- The transmitter and the resulting space-based WPT mass (the rectenna size will not be specified, but will instead be discussed qualitatively)
- Coverage times, i.e. the access or active mode times when the satellite can beam power to the rectenna
- Tracking information during the coverage times, i.e., angular tracking rate of rectenna to transmitter LOS
- Stationkeeping considerations based on orbital perturbations (e.g., J2 effects)
- The rectenna location has already been specified (Section 3.4), but it is listed here as a reminder of its interconnectedness with the orbit.

Rationale

Two key mission requirements are linked to the objective and become the chief drivers for the trade study:

1. The mission must adequately demonstrate SSPS technology in a manner that is relevant to future SSPS applications.
2. The system size or mass must be consistent with the scale of technology demonstration missions and adhere to launch vehicle constraints.

These requirements may be viewed as the basis for the SSPS-TD mission feasibility. In other words, the mission is only feasible if the system size and mass is viable (in terms of constraints, cost, and other technical practicalities), and if the transmitter beams power of a sufficient amount, over a sufficient time period (i.e. coverage times). Both of these issues are directly related to the SSPS-TD orbit, as both the orbit altitude and inclination are fundamental parameters that determine the system size, mass, and coverage times. When combined with a multitude of constraints (see below), only a narrow window of mission feasibility exists, whose identification is the focus of the trade analysis.

Assumptions

Includes assumptions that both directly and indirectly define or restrict the orbit. Previously stated assumptions are listed without their rationales.

- SSPS-TD satellite orbit is circular

  This makes the analysis much simpler and more feasible, given the nearly infinite orbits to choose from. In addition, given the nature of the technology demonstration mission and the principles of power beaming, elliptical orbits do not offer any immediate advantages over circular orbits.

- Rectenna located in White Sands, NM: 32.38° N, 106.50° W

- 5.8 GHz power beam (Section 3.3.3)

- Transmitter and rectenna are circular in shape (Section 3.3.1)

- The transmitter (antenna) acts as a perfect lens (Section 3.3.1)

- Mechanically steered rectenna maintains the rectenna plane parallel to the transmitter plane (Section 3.3.1)

- Rectenna located in the far-field (Section 3.3.1)
- The transmitter will generate a uniformly distributed power beam (Section 3.3.1)
- The atmospheric transmittance κ is 0.979 (Section 3.4)
- The satellite is sized according to Section 3.6

Constraints

Includes constraints that both directly and indirectly define or restrict the orbit. Previously stated constraints are listed without their rationales.

- The SSPS-TD satellite orbit is restricted to 28°-145° inclinations due to the available launch sites (Assumption 2)
- SSPS-TD satellite must launch onboard a U.S. launch vehicle (Constraint 5)
- SSPS-TD satellite must utilize only one launch (Constraint 6)
  This becomes a mass constraint that depends on the launch vehicle and orbit.
- The transmitter subsystem cannot exceed 70% of the satellite dry mass.
  It will be further assumed that due to the relatively large structural (i.e. non-consumable) mass of the SSPS-TD satellite, the dry mass (with margins and reserves) is a good approximation of the total satellite mass. Together with the previous constraint, this places an upper bound on the transmitter mass depending on the launch vehicle payload capability for a given orbit.
- A 25% margin/reserve is placed on the satellite mass (Section 3.6)
- The transmitter is not to exceed 15 m in diameter (Section 3.6)

- SSPS-TD satellite must have LOS with the Sun during coverage times (e.g. active mode)

  Active WPT can only be performed when there is adequate power to form the power beam at the transmitter. Since no significant onboard power storage will be performed, the solar arrays must supply this power during the active mode, and hence must have line-of-sight with the Sun.

- Transmitter has a 120° FOV, resulting in an approximately 30° elevation minimum from the rectenna for power beaming (affects the coverage times) (Section 3.3.4)

- Ionosphere power density limit (Constraint 3): $I_{ionosphere} < I_{TSI} = 425$ W/m^2 (Ionosphere border considered at 80 km altitude)

- Incident power density maximum at rectenna edge (Constraint 4, safety related): $I_{r_{edge}} \leq 100$ W/m^2

- Incident power density minimum (irradiance minimum): $I_r \geq I_{r_{min}} = 25$ mW/m^2 (Section 3.3.5)

- Transmitter peak power density (due to thermal constraints): $I_{t_0} \leq 26,000$ W/m^2 (Section 3.3.4)

Alternative Designs

The potential solution set is all circular orbits of any altitude and with any inclination that allows for a U.S. continental launch and simultaneous Sun and rectenna coverage. This reduces the available inclinations to 28°-62.38°, given the rectenna location, available launch sites, and 30° elevation angle constraint. However, since the rectenna is located at 32.38° N, coverage times and WPT efficiency are maximized when the satellite passes directly overhead. The orbit inclination is therefore set equal to the rectenna latitude so that the majority of orbit passes occur near the rectenna zenith. This means that the satellite must launch from CCAFS.

The available orbits are therefore solely dependent on the altitude, and are categorized as low Earth orbits (LEO), medium Earth orbits (MEO), geostationary Earth orbit (GEO), high Earth orbits (HEO), and Sun-synchronous orbits (SSO). GEO offers continuous coverage with the rectenna and is the most common orbit considered in the literature for a full-scale SSPS concept. SSO offers continuous sunlight to the solar arrays while LEO offers the highest potential payload and no need for an upper stage. For this reason, many scientific and large-scale space missions occur in LEO, like the Hubble Space Telescope and the International Space Station (ISS). Most GPS and communications networks are located in MEO, where longer periods and more coverage is possible.

Because this is a technology demonstration mission and the WPT must be carefully controlled and monitored, it is also highly desirable to have regular daily coverage times. This is best achieved by choosing an orbit with a repeating ground track.

Orbit solution set: *Circular, repeating groundtrack, 32.38° inclination*

Method of Trade Study

The trade study is completed through quantitative analysis and comparison of the potential orbits identified above. Due to the complexity of the problem and the scope of the solution set, rigorous orbit optimization is not possible and so no definitively optimal orbit solution will be found. Instead, the most desirable orbit is selected from a limited set of orbits that is refined through progressive comparisons of the transmitter size, consequent satellite mass, and coverage times, under the listed constraints. Sensitivity analysis is applied throughout to construct the best possible solution set.

Two tools are used to perform the required analysis:

1. STK 9 by Agi[35] is a mission modeling and analysis software for space, defence and intelligence systems. It is used here to construct circular, repeating ground track orbits and determine the coverage times given the rectenna location. Multiple LOS and elevation constraints are easily incorporated and all data analysis can be outputted graphically. The orbital computations are generated using the built-in J2Perturbation propagator.

   - Inputs: rectenna coordinates and elevation, rectenna LOS constraint, Sun LOS constraint, elevation angle constraint, orbit inclination, RAAN, approximate orbit altitude or number of cycles to repeat

   - Outputs: Orbit trajectory and ground track, orbital elements and parameters (e.g. period), coverage times and associated parameters (e.g., elevation angle, range)

2. MATLAB by MathWorks[36] is a programming environment that can be used for algorithm development, numerical computation, data analysis and visualization. Two MATLAB models are constructed for this trade study, using the equations in Table 3.8.

[35] www.agi.com
[36] www.mathworks.com

The first model computes the Airy mainlobe width, peak transmitted power density I_{t_0}, central irradiance I_{r_0}, irradiance in the ionosphere $I_{ionosphere}$, and total power transmitted P_t, given a transmitter size and altitude, and under the listed constraints. A basic optimization algorithm is used to maximize the transmitted power, given the constraints, and output the corresponding aforementioned WPT performance parameters. The transmitted power is maximized in order to minimize the transmitter size, as discussed further on. $I_{ionosphere}$ is found by computing I_{r_0} at an altitude of $R - 80$ km, where R is the satellite's altitude. This model is used in conjunction with STK to refine the orbit set through the comparison of the transmitter size with the WPT performance parameters for different orbits.

The second model numerically integrates the uniform beam equations in Table 3.8 to generate the ground irradiation pattern, given an altitude R, I_{t_0}, and the transmitter size. This model is used to determine the rectenna size and the resulting collected power.

- Inputs: all maximum and minimum power density constraints, power beam frequency and power density distribution (uniform), transmitter radius, orbit altitude
- Outputs: Airy mainlobe width, I_{t_0}, I_{r_0}, $I_{ionosphere}$, P_t, ground irradiation pattern

Together, these tools seek to identify an optimal orbit that allows for a feasible mission in terms of the requirements discussed above.

Evaluation Criteria

The Figures of Merit (FoMs) for the orbit trade study are derived directly from the two mission requirements of the trade rationale:

1. *Transmitter Size.* This is the most important metric for the size, mass, and cost of the SSPS-TD system. The transmitter size determines the transmitted power, which defines the solar array size necessary to supply the needed power, and the rectenna size based on the resulting irradiation pattern (for a given orbit altitude). The sizes are then extrapolated to masses as discussed in Section 3.6, which in turn become costs associated with the launch payload and construction.

   The transmitter size is characterized by its radius a, and because it defines the irradiation pattern for a given altitude, it is bound by the multiple power density constraints listed above. Recall the following properties of the WPT element from Section 3.3.1:

   - Increasing the orbit altitude R widens the mainlobe and reduces the power density distribution I_r on the ground
   - Increasing the transmitter size narrows the mainlobe and raises the power density distribution I_r on the ground
   - Increasing the transmitted peak power density I_t or the total transmitted power P_t increases the power density distribution I_r on the ground

   The orbit altitude and transmitter size must therefore be adjusted together to meet the power density constraints and without exceeding the mass and size constraints. This relationship is shown further on in the trade analysis.

   Now, because SSPS-TD is a technology demonstration mission, the system should be as small as possible in order to reduce launch costs and improve operational feasibility in terms of design complexity and risk (e.g. construction and deployment). This is reflected in the 15 m diameter limit of the transmitter size. At this small scale, the minimum incident power density is the limiting factor and the upper bounds on the power density - the ionosphere and safety constraints - are not an issue. This means that in order to minimize the transmitter size, it needs to operate at the peak power density and at the lowest possible orbit altitude. As discussed in Section 3.6, this minimizes the satellite mass as well, resulting in reduced cost and greater flexibility in terms of physical design and orbit options.

2. *Coverage Time*. This is defined as the duration of possible power transmission per orbit revolution. These windows exist when the satellite has LOS with both the rectenna and the Sun, and is at least 30° above the horizon from the rectenna. In practice, the averages of the coverage times, number of coverage times, and total coverage time per 24 hours is considered.

   For the SSPS-TD mission the coverage times must be long enough to properly demonstrate and analyze the WPT performance. There is no fixed minimum as this is a decision based on availability and desired performance, but longer coverage times are obviously more desirable since they provide more opportunities for testing and analysis, and should be at least on the order of minutes. Note that continuous or long periods of power transmission will be avoided because the WPT technology must be demonstrated carefully and intermittently, especially given uncertainties and risks associated with atmospheric and ground effects. Therefore, continuous coverage time is not absolutely necessary.

   The orbit trajectory is directly related to the duration of the coverage times. Higher orbits correspond to longer periods, and thus longer individual windows of coverage. Lower orbits correspond to shorter windows of coverage but more passes. The total coverage time per 24 hours can be more or less, depending on the relative orbit altitudes.

These two FoMs are unfortunately inversely related: higher orbits correspond to longer coverage times but larger transmitters, while lower orbits correspond to smaller transmitters but shorter coverage times. This tradeoff is therefore the root of the orbit trade study and the choice of values here is the primary determinant of mission feasibility.

The transmitter size is prioritized given the associated costs and logistics issues, with the coverage time treated as a variable constraint depending on the available times. Sensitivity analysis can then be applied to optimize both FoMs and determine a "best" orbit.

This process of determining the mission feasibility in terms of the orbit is shown as a flowchart in Figure 3.15, for the goal of minimizing the transmitter size, given by radius a. This schematic represents the methodology behind the trade analysis and explains the roles of the computational models discussed above. It also reveals the central role of the transmitter size in sizing the whole SSPS-TD system and determining its performance. Note that in this design there are no restrictions on the rectenna size. If such constraints existed, then a feedback loop would exist between the the rectenna and transmitter sizes.

Note that because this is a technology demonstration mission, maximizing the end-to-end efficiency and total delivered power is not a mission goal. While a bigger SSPS system is more cost-effective, and a full-scale system would certainly seek to maximize efficiency and delivered power, this is not important to the SSPS-TD mission.

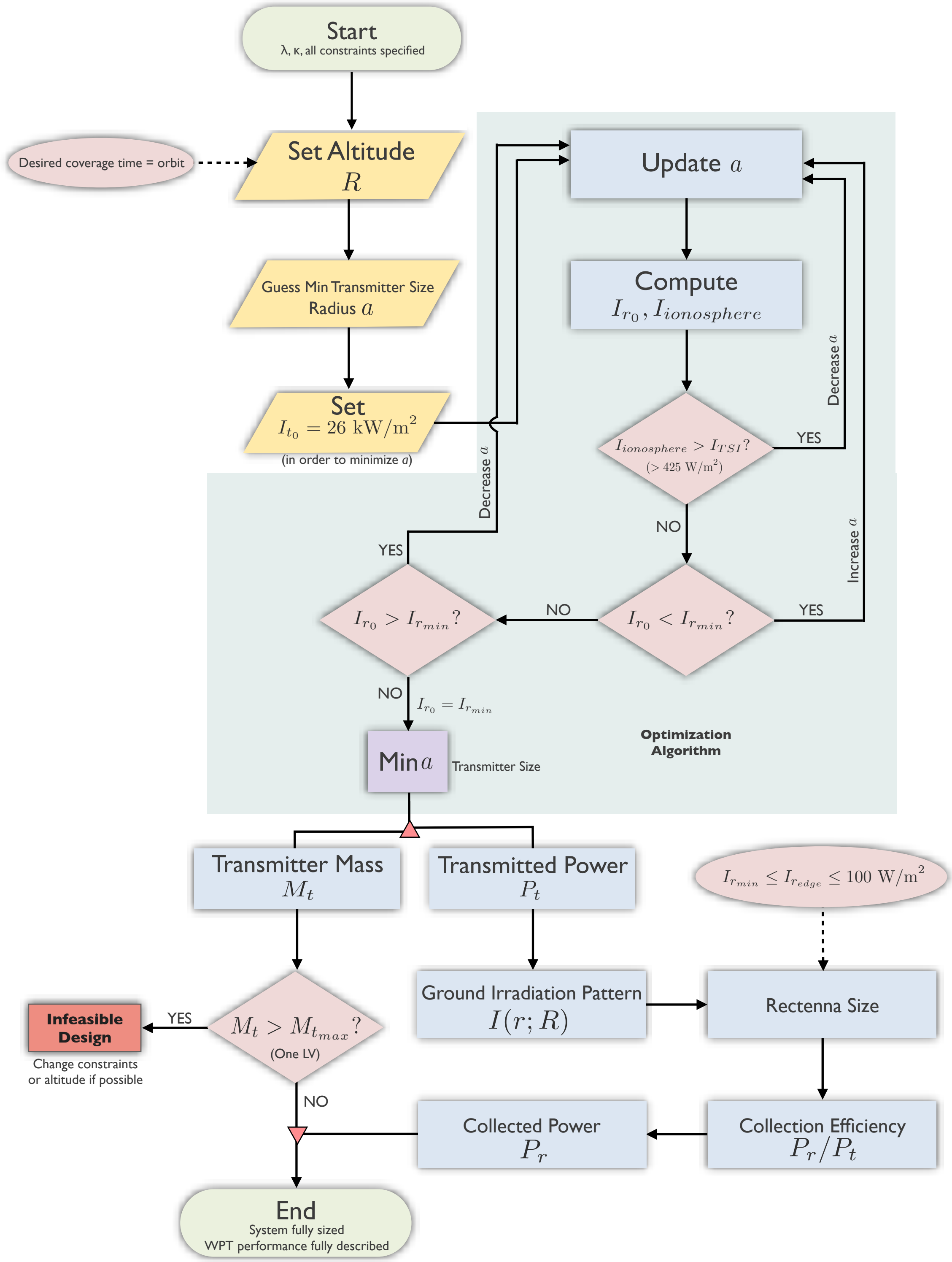


Figure 3.15: Computational flowchart for determining feasible orbit with minimum transmitter size

Analysis

With the inclination already determined, it remains to investigate the orbit altitude and its impact on the transmitter size and coverage times as they relate to mission feasibility.

First, since the satellite can transmit at elevation angles of up to 30°, the satellite-rectenna range will be up to twice the orbit altitude R. According to the equations in Table 3.8, this maximum range will result in the reduction of the irradiance distribution by a factor of four. For any non-GEO, the following orbital analysis must therefore account for this effect by choosing a central irradiance at least four times greater than the stated minimum incident power density so that the rectenna is activated during all coverage times. This so called full coverage minimum is then ≥ 0.1 W/m^2 and it becomes the new minimum threshold for $I_{r_{min}}$, and is computed at the satellite zenith (elevation 90°).

Now, the relationship between the transmitter size (radius a) and the orbit altitude R is explored in order to refine the orbit solution set before considering coverage times. The goal here is to locate a feasibility region where the areas of feasible size, mass, and irradiance all intersect. The size feasibility is based on the transmitter size that cannot exceed 15 m in diameter, while the irradiance feasibility is defined for power densities between 0.1 W/m^2 and 425 W/m^2. The mass feasibility is considered by assuming a Delta IV Heavy launch vehicle and restricting the transmitter mass to no more than 70% of the satellite launch mass. This launch vehicle is chosen because it has the highest payload capability. A 400 kg payload adaptor is assumed.[37]

[37] [13]

Figure 3.16 shows the results of this preliminary analysis for a wide range of orbit altitudes. The central irradiance is plotted as a function of transmitter diameter for each orbit altitude, and the corresponding transmitter mass is displayed. The transmitter size is displayed well above the 15 m limit for demonstrative purposes.

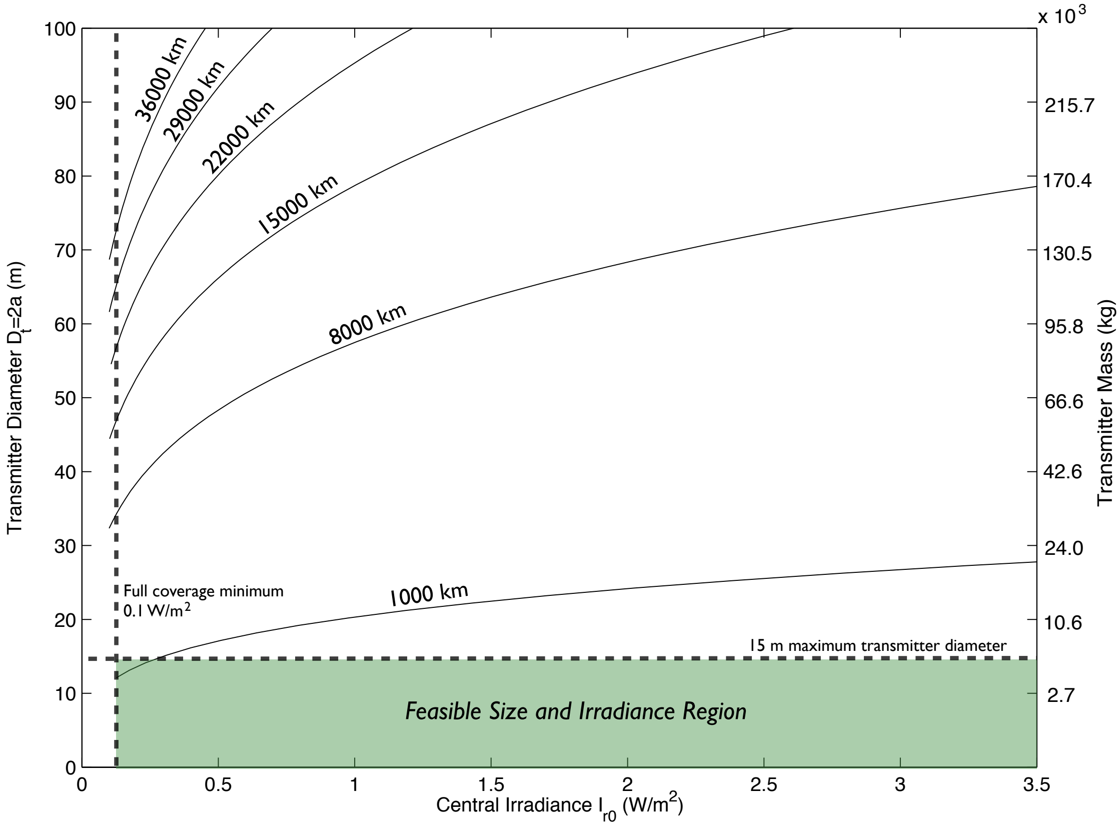


Figure 3.16: Orbit Feasibility: Transmitter Size & Mass vs Central Irradiance for Different Orbit Altitudes

Central irradiance as a function of the transmitter diameter for various orbit altitudes (generated with MATLAB). The size and irradiance constraints are marked. The feasible size and irradiance region is constructed within these constraints. All points shown here are at the maximum transmittable power density of 26 kW/m^2. Note that the mass function is not linear and is computed for each tick only.

It is immediately obvious from this plot that the satellite must be located in **LEO**. The limiting factors are the minimum irradiance and the maximum transmittable power density. Because the transmitter cannot transmit at higher than 26 kW/m^2, it must be made large enough to generate a ground-incident power density greater than the minimum irradiance limit. And the greater the orbit altitude, the larger the transmitter needs to be. Since the irradiance minimum is based on background noise, the thermal-based transmitter peak power density is really the most important parameter, as discussed in Section 9.1.

Because of the 15 m size limit on the transmitter, the satellite restriction to LEO can actually be made without considering mass. However, from a mass viewpoint, LEO can also be derived as the necessary orbit region. Because the mass feasibility depends on the orbit, it cannot be easily identified in FIgure 3.16 . But it can be computed for a given altitude from the *Delta IV Payload Planners Guide*.[38] The payloads are given for circular orbits and the values for 45° inclination orbits are used as this is the closest inclination to the SSPS-TD orbit (the actual payload will be slightly higher). Then, at 1,000 km, the payload capability is 22,300 kg (after subtracting the payload adaptor), resulting in a maximum allowable transmitter mass of 15,600 kg; this is well above the 6,000 kg for a 15 m transmitter. But at GEO (∼36,000 km), for instance, the payload capability is only 6,276 kg, resulting in a maximum transmitter mass of 4,400 kg. And since the transmitter must be 70 m in diameter just to meet the irradiance minimum, not only is the size constraint violated, but the mass would be 130,500 kg, orders of magnitude greater than this limit.

The advantages to LEO are that the highest payload capabilities are available and the satellite is located on the outskirts of the Van Allen radiation belt, thereby reducing potential hardware degradation and shielding requirements. The first advantage means that the mass feasibility is no longer a concern, and the focus is instead

[38] [13]

on the size and irradiance feasibility.

However, the major disadvantage of LEO is the very low coverage times that result from such low orbit altitudes. The coverage times are low enough that it is necessary to enforce a minimum coverage during per pass of approximately 10 minutes. This minimum is established after making preliminary examinations of typically available coverage times for ground-repeating orbits in the LEO range. This is, unfortunately, far lower than would be considered desirable (hours rather than minutes), and it is an important result that reveals the challenges that must be overcome. Mission feasibility may be compromised on a cost-benefit level, highlighting the need for technology development and design changes that allow for a higher orbit while retaining a small transmitter, as discussed in Section 9.1. Identifying these issues is critical during these early design stages, and the rationale behind concept feasibility studies.

So now, coverage feasibility must be incorporated into the orbit analysis. Then the four feasibility areas of size, mass, irradiance, and coverage can be assessed together to determine an overall mission feasible orbit.

In order to maximize the satellite coverage time while remaining within the mass, size, and irradiance constraints, the transmitter must be near the maximum 15 m in order to allow for the highest possible orbit altitude. But as shown in Figure 3.16, the altitudes must be constrained to the LEO range. The plot shows that a 1,000 km altitude is feasible from a mass, size, and irradiance viewpoint, so orbits near that altitude are investigated to find the maximum coverage time allowable within those constraints. Using STK, a series of circular, repeating groundtrack orbits are generated, and their average coverage windows are computed. The averages are calculated for a 1 month period beginning March 31st, 2012, and maximized by optimizing the RAAN. Recall that there must be LOS with the Sun and rectenna simultaneously for a valid coverage window, and the satellite cannot transmit below 30○ elevations. The result is shown in Figure 3.17.

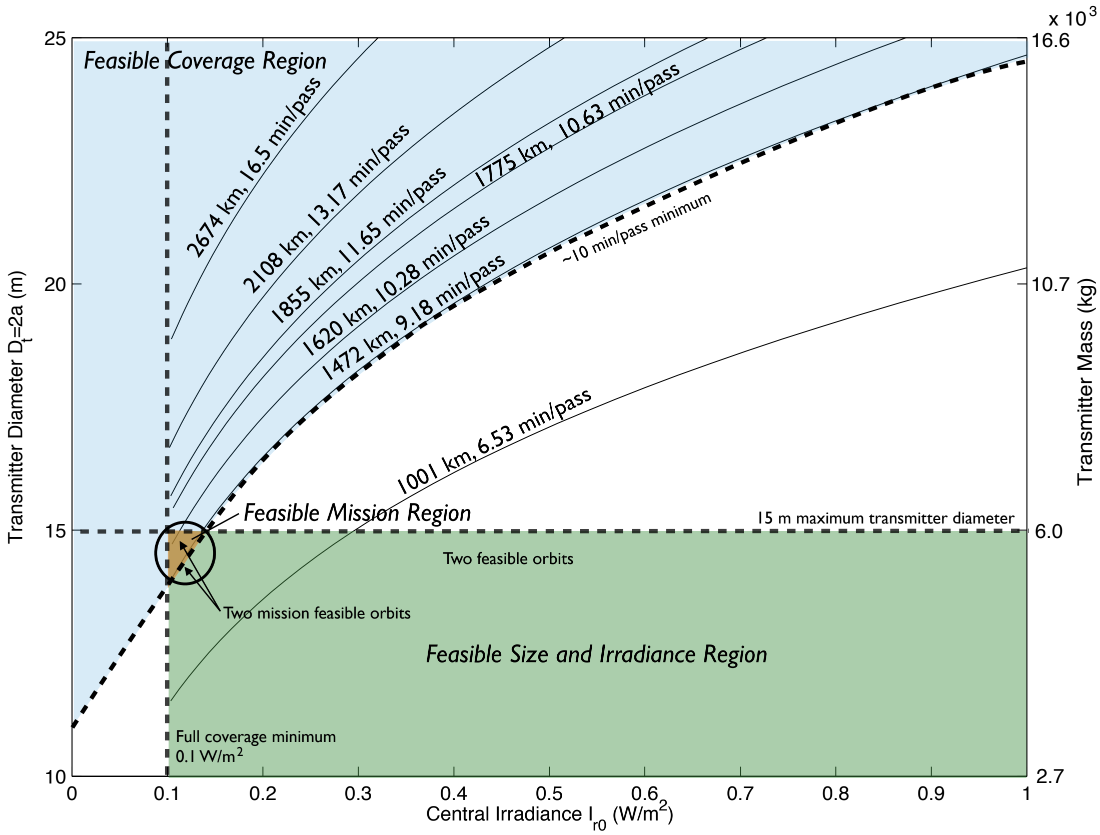


Figure 3.17: Orbit Feasibility in LEO with Coverage Times

The same type of plot as before is generated for a set of specific orbit altitudes that correspond to circular repeating groundtrack orbits in LEO. The average coverage times are listed so that coverage feasibility region can be identified. The feasible size and irradiance region is marked as well. The mission feasible region is the intersection of these two regions (mass feasibility is guaranteed). All points shown here are at the maximum transmittable power density of 26 kW/m^2. Note that the mass function is not linear and is computed for each tick only.

The feasible mission region is identified in the plot as the intersection between the feasible coverage region and the feasible size and irradiance region (mass feasibility is guaranteed in LEO for this size transmitter). As predicted, this window of feasibility is quite small due to the numerous constraints, and only two orbits fall within it. These two candidates are described below in Table 3.18.

Table 3.18: Final Orbit Comparisons

	Orbit 1	**Orbit 2**
Altitude[1]	1472 km	1620 km
Inclination	32.38°	32.38°
Period	1.923 hrs	1.977 hrs
RAAN[2]	41.69°	41.69°
Central Irradiance[3]	0.137 W/m^2	0.113 W/m^2
# of revolutions/repeat cycle	37	12
Average coverage/pass	9.18 min	10.28 min
Average passes/day	3.87	3.87
Total coverage/month	18.38 hrs	20.58 hrs
Payload Capability[4]	21,400 kg	21,200 kg

Coverage windows computed for a 1 month period beginning March 31st, 2012.
[1] Circular, repeating groundtrack
[2] Optimized for max coverage time
[3] 15-m transmitter diameter, zenith
[4] Delta IV Heavy, 45° inclination, 400 kg payload adaptor subtracted

Though the two solutions are nearly identical, the lower number of revolutions per repeat cycle (easier to operate) and the slightly higher coverage window make **Orbit 2** the best choice.

As an aside, Figure 3.17 reveals the usefulness of sensitivity analysis. For instance, if the transmitter was set to 16 m, this would allow for an orbit with a little over 1 minute extra coverage per pass; at 20 m, the coverage time would approach 20 minutes per pass. This type of analysis can therefore optimize the mission feasibility by identifying the best cost-benefit design, right before the point of diminishing returns. However, sensitivity analysis is generally applied later on in the lifecycle, when the design has matured beyond conceptual studies and a more exact solution is required.

Results

The SSPS-TD mission orbit has been selected to be a circular, 32.38°, 1620 km orbit, with about 40 minutes of coverage each day. The results of this choice allow the architecture definition to be finalized in terms of a number of mission features that will now be discussed.

To reach this orbit, a Delta IV Heavy will launch from CCAFS with a payload capability of about 21,200 kg. At this orbit altitude a 15 m, 6,000 kg transmitter operating at 26 kW/m^2, resulting in 4.6 MW of transmitted power (before atmospheric losses). Then, according to Section 3.6, the solar array produces 6.22 MW with an area of 36,600 m^2, and weighs 1178 kg (fourteen 50 m subarrays and four 20 m subarray). Together, these two subsystems make up 33.9% of the payload capability, leaving a maximum of approximately 14,022 kg for additional subsystems and margins, or 8,722 kg after the 25% margin is removed.

The orbit groundtrack is shown in Figure 3.18. The orbit is posigrade but the track moves west over each revolution, returning to the indicated location after 12 revolutions. Because of this, are several revolutions where the satellite is never within LOS of the rectenna. This reveals the disadvantage to SSPS concepts in low-altitude orbits, and explains why full-scale SSPS designs often place the satellite in GEO or use multiple rectennas.

The coverage window pattern generated by STK is shown in Figure 3.19 in order to demonstrate the regularity of the windows (due to the repeating groundtrack orbit) and the separation between the active mode passes. 120 coverage windows are

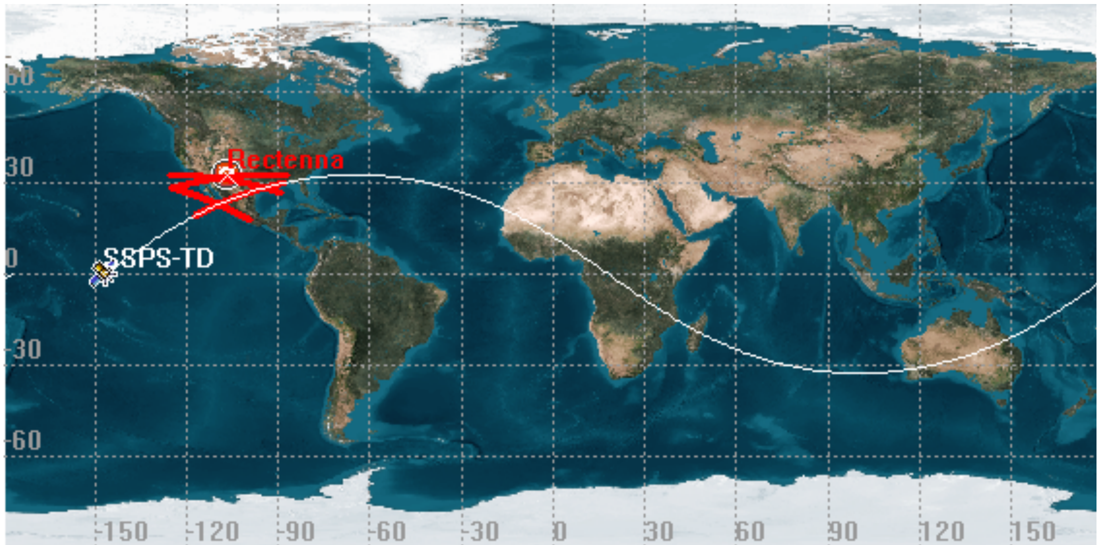


Figure 3.18: SSPS-TD Orbit Groundtrack

32.38° inclination, 1620 km altitude, 41.69° RAAN. The groundtrack shifts west on every revolution, and repeats after 12 revolutions. The red lines mark the feasible coverage windows and correspond to the coverage times in Figure 3.19.

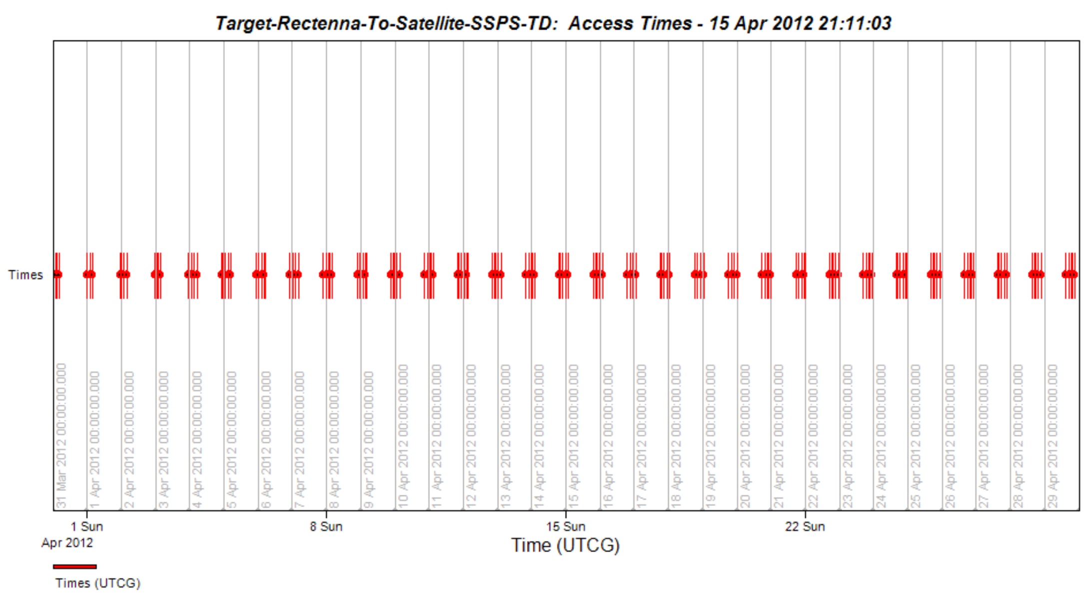


Figure 3.19: SSPS-TD Coverage Windows

Coverage window pattern for all times over one month period when the satellite has LOS with the Sun and rectenna, and is over 30° elevations.

found, with an average satellite-rectenna range of 2131 km, which corresponds to a mean elevation of 46°.

Using this mean range, the radiation pattern (i.e. irradiance) on the ground is computed and shown in Figure 3.20. By plotting the encircled power over this

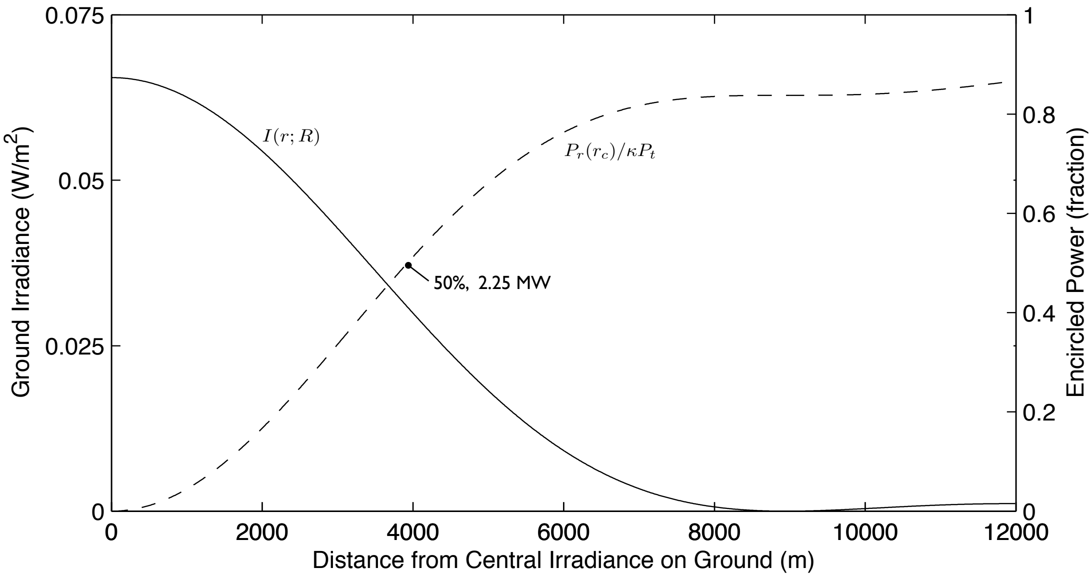


Figure 3.20: SSPS-TD Ground Radiation Pattern and Encircled Power

Average irradiance distribution (solid line) and encircled power distribution (dotted line) on the ground for the SSPS-TD orbit and 15 m transmitter (4.6 MW total transmitted power). The average is computed with the mean satellite-rectenna range of $R = 2131km$, corresponding to an elevation of 46°. Atmospheric transmittance is accounted for with $\kappa = 0.979$. Note that the peak irradiance would be 0.113 W/m^2 for 90° elevation.

distribution, the rectenna can be chosen according to the desired power collection. As seen in this figure, the small transmitter size and large distance R results in a very wide radiation pattern, with a mainlobe width of over 8 km (16 km total). But because this is a technology demonstration, the rectenna should be much smaller than this, and is primarily limited by cost and construction logistics. The 25 mW/m^2 minimum introduced in Section 3.3.5 should also be considered. Technically, this constraint applies to the power incident on each rectifier, and concentrators will be used to ensure this value. However, the incident power density should not be much lower than this or design complications will likely arise. At this cutoff power density, the rectenna would still need to be 4,400 km in radius and would collect about 58% of the transmitted power (after atmospheric losses). Note that the sidelobes are extremely low and well within the safety limits (Section 2.1.7).

The purpose of the discussion above is to emphasize that the size is somewhat flexible and depends largely on the operational goals and design constraints. Because of this, and since the design focus is on the space segment, the exact rectenna size is not specified here. This only determines the collection efficiency η_c, so the end-to-end efficiency is straightforward to compute for any given rectenna size and corresponding collection efficiency (all other efficiencies, including the rectenna conversion efficiency, are known).

Finally, environmental conditions and perturbation effects are examined for this orbit:

- *Van Allen Belts* - The energized free electrons in ions trapped in the Earth's magnetic field can pose serious hazards to satellite subsystems. Though the inner Van Allen Belts are centered around 3,300 km altitudes, it still has a presence down to about 400 km, especially over the South Atlantic Ocean. This radiation must therefore be accounted for by hardening the SSPS-TD satellite subsystems. The solar array is particularly susceptible to degradation under these conditions and cannot be entirely avoided. As a reference, the ISS solar arrays, at about 400 km, degrade at 0.2-0.5% per year.[39] Though the solar cell

[39] [31]

efficiency stated in Section 3.5.1 is an EOL efficiency, the 10% power margin is included in part to account for any unexpected increase in solar cell degradation.

- *Shadow/Thermal Effects* - Due to this orbit trajectory, the satellite will not always be in the Sun, so onboard energy storage or batteries must be present to power the non-WPT subsystems during these shadow periods. The satellite will also be passing in and out of the Sun frequently, and different components will experience different thermal effects. The resulting temperature fluctuations and gradients across the satellite subsystems must be accounted for in the thermal design of the satellite.

- *Gravitational Perturbations* - Due to the large size and low altitude of the SSPS-TD satellite, a gravity gradient may be present that can perturb the satellite attitude. J2 perturbations must also be considered, though the secular perturbations will only affect the longitude of the ascending node Ω and the mean anomaly M (not the semi-major axis or the inclination). Together the effects of these perturbations on the satellite orbit trajectory and attitude must be accounted for with proper ADCS and GN&C designs (see Section 4). This is especially important in light of the strict pointing requirements for the solar array and the transmitter. Given the 1 year operational lifetime, however, extensive stationkeeping and a dedicated propulsion system is not likely to be necessary.

With the orbit trade complete, the major systems of the SSPS-TD satellite are fully described, and a significant portion of the architecture definition is complete. The top-level SSPS-TD baseline is considered finalized, and defined by the power beam type, rectenna location, satellite orbit, and launch logistics. The results from this section are recapped in the summary of the SSPS-TD design in Chapter 5.

Now, that many of the operational conditions and top-level requirements have been determined, the systems engineering process can proceed with further subsystem design and the identification of requirements and interfaces.

Chapter 4

System Hierarchy

The system hierarchy or product breakdown structure (PBS) defines all of the products that are integrated into the final system. The development of the scope, concept of operations and architecture provide the first steps in constructing the PBS by identifying the major systems and technologies utilized in the mission. This also reveals operational requirements that may need to be met by additional systems. This top-level PBS can then be broken down into hierarchical levels from system all the way down to its constituent parts.

The PBS useful in creating a product-oriented map or reference of the system and understanding all of the pieces that go into it. It drives the system design beyond a top-level description and reveals interfaces which give way to new requirements, constraints, and functional analysis. In addition, the PBS helps manage resource allocation in terms of mass, power, cost, and workforce. In particular, the PBS plays a central role in team organization and task assignments or responsibilities. The result of this work allocation takes the form of the work breakdown structure (WBS).

The PBS for the SSPS-TD mission is shown in Figure 4.1 for all of the products discussed in this study. Only the key elements of the mission are expanded down to the parts level, though all subsystems should be done so in a full PBS, including the additional subsystems that are abbreviated in the figure.

These additional subsystems are important, however, in providing crucial mission capabilities and establishing a full SSPS-TD mission design. Their specific design

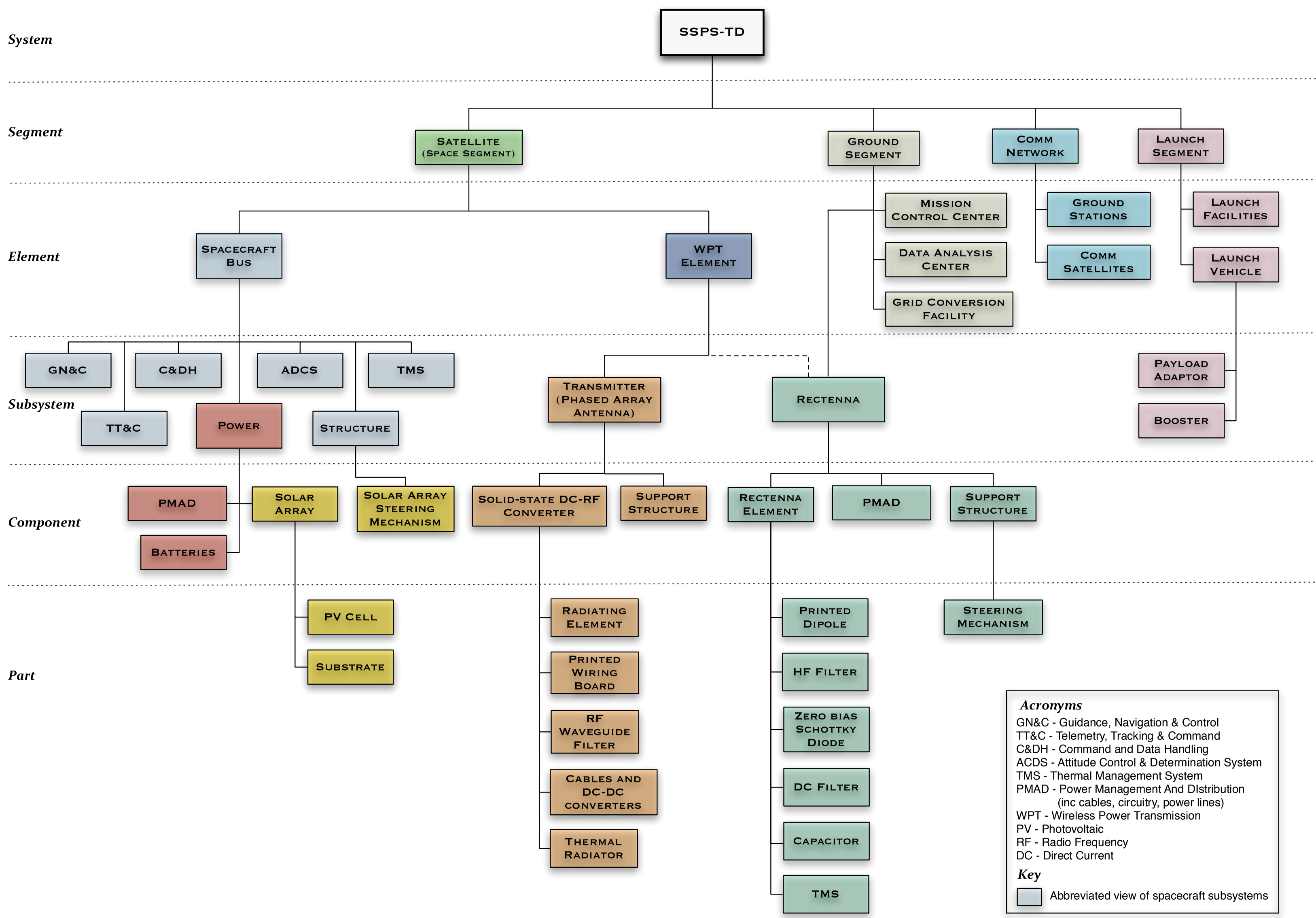


Figure 4.1: SSPS-TD Product Breakdown Structure

is beyond the scope of this study, but a top-level discussion is provided, along with a conservative mass and power estimate (including some redundancy). *Space Mission Analysis & Design* ([35]) provides mass and power estimates for specific subsystem designs, but some of these may not be valid for a mission of this scale and nature. In these cases, Japan's reference SSPS technology demonstration ([5]), the original NASA 1979 concept ([4]), and the JWST mass budget ([51]) are more applicable references and will be used together to generate best estimates. Recall that a 566 kW power margin (10% of 1.23×4.6 MW) is dedicated to these subsystems and any margins/reserves, but this full amount is only available during Sun line-of-sight (LOS). A maximum of 8,722 kg (after margins) is allowed.

1. **ADCS** - The Attitude Determination and Control Subsystem is primarily responsible for maintaining the SSPS-TD transmitter orientation and steering the solar array to counteratct environmental torques (e.g., gravity gradients and solar radiation) and the natural satellite motion. The transmitter is a nadir-pointed, body-fixed payload so that 2 axes of body attitude control are necessary. It should be pointed with an accuracy <0.1 deg/s. The Sun-oriented solar array is not body-fixed and is a planar array so it too should have its own 2 axes of control.

   In order to meet the transmitter requirements, the spacecraft should be 3-axis stabilized, with separate actuators for the solar array (e.g., magneto torque rods and reactions wheels). A propulsion system may be necessary as well to provide sufficient attitude control. A suite of sensors will also be required as part of this ADCS system, including sun sensors, Earth sensors, and gyroscopes. Due to the uniqueness of the satellite design, the 1979 reference concept is used to obtain the mass and power fractions, instead of the more standard estimates provided by [35].

Mass: 1000 kg (including $\sim$200 kg propellant and tanks for ADCS and GN&C orbit control-see below)

Power: 20 KW (majority required only during active power beaming, includes GN&C orbit control)

2. **GN&C** - The Guidance, Navigation, and Control Subsystem is responsible for determining the satellite position and velocity, and make any necessary orbit adjustments due primarily to gravitational perturbations. The first task is important for establishing LOS vectors with the rectenna and the communications network (e.g. TDRSS), and is thus vital to the operation of the WPT element. Orbit control can be obtained by using thrusters, which have been included in the ADCS design, and will be under ground-based control rather than autonomous. Navigation systems like GPS, landmark tracking and communications satellite tracking are all necessary as well. The mass and power are given for these systems only, since guidance and control systems are included in the ADCS. These are then standard systems and [35] is used to provide the estimates.

   Mass: 50 kg

   Power: 125 W

3. **TT&C** - The Telemetry, Tracking, and Command Subsystem is responsible for 1) carrier tracking, 2) transmitting and receiving data to and from the ground and other SSPS-TD satellite subsystems, and 3) ranging. The satellite will likely use the TDRSS network for communications, which operates in the S, Ka, and Ku-band. From a TT&C perspective, the SSPS-TD satellite can be treated as a standard communication satellite, and will thus have about 2000

I/O points, divided between telemetry and command. The mass and power estimates are taken from [5] and cross-checked with [35].

Mass: 60 kg

Power: 150 W

4. **C&DH** - The Command and Data Handling Subsystem is the flight computer and processing center for SSPS-TD commands and mission data. In particular, it is responsible for sending commands to the satellite subsystems and tracking the satellite health. Since this is a technology demonstration mission, the C&DH subsystem is treated as a "complex"-class system according to [35], and corresponding mass and power values can be estimated from this reference.

   Mass: 50 kg

   Power: 100 W

5. **Power** - The Power Subsystem is responsible for generating, storing, and distributing the satellite's electrical power. The previously sized solar array is the primary power source, but batteries will be used for storage and power distribution during shadow periods and as a backup for the additional subsystems. In this way, the critical control, command, and data subsystems can remain online in the case of a major emergency like a solar array deployment failure. The batteries therefore only need to meet the relatively low power requirements of the additional subsystems during inactive modes, and with specific energy densities on the order of several hundred W·hr/kg, the batteries are estimated at only 10 kg ([35]).

   The PMAD and wiring systems are both complex and extensive due to the vast amounts of power being generated, and the strict requirements on its regulation and distribution. The mass of these systems is therefore significant, and power will be lost through transmission. This power loss is treated as a percentage of the power generated by the solar array, and it is taken out of the 10% margin specified in Section 3.6. The mass and power fractions for the NASA 1979 concept are used as a reference since no other sources treat power distribution on the scale utilized by SSPS-TD.

Mass: 1010 kg (includes batteries)

Power: 5% (leaves 5% power margin)

6. **TMS** - The Thermal Management Subsystem is responsible for maintaining optimal temperatures across the various satellite components. This is especially challenging due to 1) the intermittent shadow and solar illumination that the satellite experiences throughout its orbit trajectory, 2) the large planar structures of the satellite (solar array and transmitter), and 3) the high operating temperature of the transmitter. These effects will create extreme temperature gradients across the satellite that must be mitigated. In addition, waste heat must be managed, namely the 2.6 kW/m^2 released by the transmitter during the DC-RF conversion process.

   To minimize design complexity, the SSPS-TD satellite will rely primarily on passive thermal systems like multi-layer insulation and radiator panels. One idea implemented into the "Sandwich SPS" concept is to use large reflectors to redirect incoming sunlight onto an integrated solar, PMAD, and WPT structure in order to maintain high temperatures; there is no system for waste heat management, however.

   Since the transmitter and solar arrays are the biggest issues for the TMS, its mass is assumed to be 5% of their added masses (from [35] and [4], which both agree).

Mass: 350 kg

Power: 0 W

7. **Structure** - The structure is responsible for supporting all of the satellite subsystems, and includes all of the physical and mechanical interfaces. The solar array structure has already been accounted for in its design, but not the support structure for its steering mechanism or its interface with the rest of the satellite. The support/deployment structure for the transmitter and the rest of the satellite bus must also be included, as well as all wiring and mechanism harnesses. Due to the large structure size, advanced lightweight materials should be considered, but configurations can likely draw heritage from the ISS.

   From [35], the structure dry mass is assumed to be 15% of the dry mass, before the 25% margin/reserves. This agrees with the mass fraction from [4].

   Mass: 15% of the dry mass (before margins/reserves)

   Power: 0 W

The mass and power budgets for the entire SSPS-TD satellite are presented in Chapter 5.

Chapter 5

SSPS-TD Design Summary

The SSPS-TD mission can now be summarized in terms of all of the the technical design choices that were made in the last two chapters, and the resulting performance parameters. The end result is a mission whose traceability back to the needs, goals, and objectives is readily apparent. Then, with the top-level design loop closed, further subsystem and operational considerations can be investigated with formal interface and requirements generations. And most importantly for this concept and feasibility study, systems engineering practices can be used to analyze the potential of this mission's design in terms of cost, risk and overall viability.

A summary of the SSPS-TD mission is given below, including all decisions that have been made during the architecture development and trade studies. Following this description is the mass and power budget, shown in Table 5.1, and the end-to-end efficiency linkbudget in Table 5.2.

Recall that the individual mass and power parameters for the satellite subsystems were given in Chapter 4. The mass and power margin philosophy was discussed in Section 3.6. Note that due to careful designing and adequate constraints, the mass is well within the allowable maximum of 21,200 kg (due to the one launch constraint), even with margins, and the power consumed is as expected with appropriate margins.

The efficiency linkbudget is constructed by identifying all of the sources of inefficiency in the WPT process. These were given in Sections 3.3.4 and 3.3.5 for the transmitter and rectenna subsystems, respectively. The beam coupling losses are associated with the atmospheric transmittance determined in Section 3.4, and the rectenna size, which remains undetermined, as discussed in Section 3.7. Putting these individual linkbudgets gives an end-to-end efficiency for the SSPS-TD WPT performance, i.e., the ratio of the output power to the grid over the input power to the solar arrays (the collected solar radiation). As mentioned in the mission objectives (Section 2.1.3), this performance parameter is one of the defining characteristics of an SSPS concept design. It provides a comparison metric for other SSPS reference systems, and most importantly, allows for future economic analyses regarding the "business-case" for SSPS, and thus the viability of SSP as a marketable energy source.

- **Initial Operational Capability:**
  - Orbit: LEO, 1620 km, 32.38°, circular, repeating groundtrack (12 revs)
  - Average Coverage: 3.87×10.18 min/day
  - Rectenna Location: White Sands, NM
  - Power Beam Type: Uniform microwave beam
  - Power Beam Frequency: 5.8 GHz
  - Total Satellite Surface Area: 36,780 m^2
  - Total Satellite Mass (without adaptor): 13,854 kg
  - Peak Incident Power Density: 0.113 W/m^2 (well below the safety limit of 10 W/m^2)
  - Collectable Power on Ground: 4.504 MW
  - End-to-End Efficiency (Collected Solar/Output Power to Grid): $0.0642\eta_c$ (see Table 5.2)
  - Mission Lifetime: 1 year

- **Satellite:**
  - Transmitter: 15-m diameter, GaN solid-state phased array, 4.6 MW transmitted
  - Solar Array: 6.22 MW, 36,600 m^2 (fourteen 50-m, four 20-m subarrays)
    - ◇ Solar Cells: thin-film a-Si:H photovoltaics
    - ◇ Subarrays: CP1/a-Si:H with CFRP structure/deployment booms
  - Additional Subsystems:
    - ◇ ADCS: 3-axis stabilized, solar array actuators, control thrusters, sensor suite
    - ◇ GN&C: ground-based control, navigation sensor suite, control thrusters in ADCS
    - ◇ TT&C: S, Ka, or Ku-band (compatible with TDRSS communications network), ∼2000 I/O points
    - ◇ C&DH: flight computer, "complex"-class (as specified by [35])
    - ◇ Power: solar array, batteries for additional subsystems
    - ◇ TMS: primarily passive systems (multi-layer insulation, radiator panels)
    - ◇ Structure: solar array steering mechanism, transmitter support/deployment structure, spacecraft bus structure, wiring and mechanism harnesses
- **Launch Segment:**
  - Launch Vehicle: Delta IV Heavy
  - Payload Capability: 21,200 kg, 22.4 m × 5 m fairing (based on 45° orbit)
  - Launch Site: CCAFS

- **Ground Segment:**
  - Rectenna: printed dipole array with zero bias Schottky diode, mechanically steered structure with parabolic concentrators
  - Mission Control Center, Data Analysis Center, Grid Conversion Facility

Table 5.1: SSPS-TD Satellite Mass & Power Budget

Subsystem	**Mass** (kg)	**Power** (kW)	**Notes**
Transmitter (payload)	6000	4600	
Spacecraft Bus (dry)	4923	310.9	Margin not included
Power	2188	310.5	
Solar Array	1178	0	Generates 6.21 MW
PMAD	1000	311	5% power loss
Batteries	10	0	Supplies ≥395 W (bus requirements)
ADCS	800	0.020	
GN&C	50	0.125	
TT&C	60	0.150	
C&DH	50	0.100	
TMS	350	0	Assumes passive thermal systems
Structure	1425	0	15% of dry mass before margin
Margin	2731	283	25% mass margin, 5% power margin[1]
Satellite Dry Mass/Power	13654	5194	
Propellant	200	0	
Loaded Mass/Power	13854	5194	Same as injected mass (no kickstage)
Launch Vehicle Adapter	400	0	
Total Launch Mass/Power	**14254**	**5194**	Equivalent to boosted mass.

Power listed is power consumed, not produced.
Grey rows denote totals.
[1] 5% of $1.23P_t$, as discussed in Section 3.6

Table 5.2: End-to-End Efficiency Linkbudget

Sources of Inefficiency	**Efficiency**	**Power (MW)**	**Notes**
Incident Solar Energy		51.582	Ignores solar variations.
Solar Array - 12.3%			
Solar Array (EM-DC conversion)	0.124	6.220	CP1/a-Si:H. Includes cell and array efficiencies. Includes 5% power margin.
Random failures	0.990	6.158	Estimated 1% failures
Circuitry - 95%			
PMAD/Wiring	0.950	5.850	e.g. IR^2
Transmitter - 81.4%			
Maximum Input		5.655	Final transmitted power must be 4.6 MW to satisfy constraints. Difference between this row and previous is left over margin.
DC-DC conversion	0.980	5.542	
DC-RF conversion	0.900	4.988	GaN solid-state amplifier
Subarray random electronic failures	0.980	4.888	Estimated 1% failures
Amplitude error	0.996	4.868	± 1 dB amplitude deviation
Phase error	0.978	4.761	$\pm 15°$ phase deviation
Phase quantization	0.997	4.747	5-b phase shifter
Taper quantization	0.989	4.695	10 steps
Aperture efficiency	0.980	4.601	Conductive losses in aperture
Beam Coupling - $97.9\eta_c$%			
Atmospheric losses	0.979	4.504	Based on rectenna location. Includes seasonal variations.
Collection efficiency	η_c	$4.504\eta_c$	Depends on rectenna size.
Polarization loss	1.000	$4.504\eta_c$	Assumes near perfect alignment.
Rectenna - 69.2%			
Rectenna reflection loss	0.980	$4.414\eta_c$	
Rectenna random failures	0.990	$4.370\eta_c$	Estimated 1% failures
RF filter insertion loss (IL)	0.891	$3.894\eta_c$	Estimated IL= 0.5 dB
RF-DC conversion	0.800	$3.115\eta_c$	Assumes optimal design (multiple antenna per diode and parabolic concentrators)
End-to-End Efficiency	$\mathbf{0.0642\eta_c}$	$\mathbf{3.115\eta_c}$	
DC-DC Efficiency	$0.523\eta_c$		Circuitry to rectenna output

Chapter 6

Interfaces

Interfaces are defined as the boundaries between two or more functions, and arise naturally from the PBS and mission operational baseline. Functional analysis of these features identifies which systems perform what functions, and often two or more systems must work together. In this case, interfaces must exist between them, and due to the hierarchical nature of the PBS, they exist at many different levels.

As discussed, tasks are allocated based on the PBS, and the work is often executed simultaneously. But since the system products interact, is imperative that all relevant interfaces are defined beforehand to ensure successful system integration and operation. Interfaces are explicitly defined through requirements, in an Interface Requirements Document (IRD). The interface type must also be distinguished, typically as either physical/mechanical, electrical, or data transfer. In this way, identifying interfaces is a necessary step in generating requirements, and thus drives the design process. As with most systems engineering processes, this is an iterative procedure, as interface solutions may require new functions, systems, or requirements.

A useful way of representing interfaces is through an N^2 diagram. System components or functions are placed on the diagonal, and interfaces are defined by lines connecting them; outputs are horizontal, inputs are vertical. Figure 6.1 shows an N^2 diagram for the SSPS-TD system, highlighting the WPT function (i.e., the process of generating and transmitting the power beam) unique to this mission. Where the lines depart or enter a box corresponds to an interface, whose purpose, or function, is labelled. Note that the power subsystem includes the solar array.

Of particular importance are the data loops required to establish successful power beaming. The satellite and ground must be in constant communication in order to check LOS vectors and subsystem status and settings, and thus determine whether power beaming should be activated and what commands are necessary. The satellite flight computer in the C&DH subsystem requests and processes all of the satellite data and sends it ground control via the telemetry subsystem. Ground control processes this data along with the rectenna status and uploads commands back to the satellite. The C&DH subsystem sends these commands to the necessary subsystems, which in turn, report back their updated status and settings. This constant, chained feedback loop implies a series of data input and output interfaces across many of the SSPS-TD subsystems.

This N^2 diagram depicts only the subsystem interfaces relevant to the WPT function; N^2 diagrams can be generated within each subsystem all the way down to the component. A complete N^2 diagram of the system would include all interfaces, including those for more generic functions and the subsystems that were excluded (e.g. TMS).

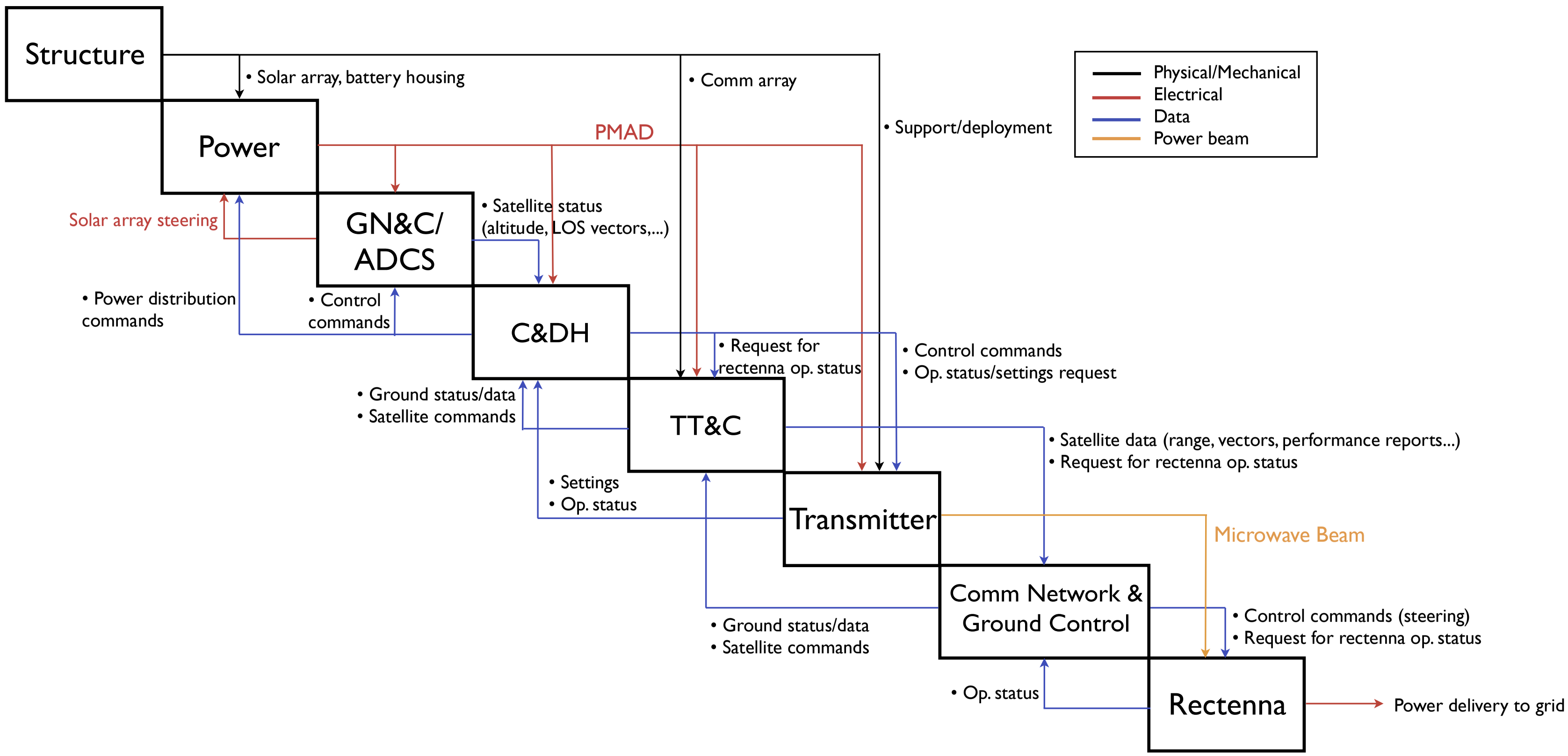


Figure 6.1: N^2 diagram for the Wireless Power Transmission function.

Chapter 7

Requirements

Requirements development is the most important step in synthesizing a design solution from the initial scoping and stakeholder expectations.[1] Top-level requirements are directly generated from the mission scope and ConOps, and are then distributed into the architecture all the way down to the parts level. In this way, requirements are decomposed hierarchically, thus making them traceable and ensuring that the project meets the mission objectives without any superfluous activities. For two successive levels of requirements, the first level will bound the scope of a problem that needs to be solved, and the second level will provide a more specific bound to the problem, that brings the engineer one step closer to identifying and designing the actual solution. At the lowest level, requirements act as specific manufacturing instructions for products that will provide this solution. Requirements are therefore independent of the design solution, instead describing the problem that must be solved and allowing the engineers to develop the optimal solution.

Because requirements are generated from systems engineering activities that are inherently iterative, developing requirements is also an iterative process. As the development lifecycle of the project matures, design choices or changes will introduce new requirements or change existing ones. And due to requirements traceability, this can have a significant impact on the overall design. In a real mission, requirements are locked down by the end of Phase A, corresponding to the System Requirements

[1] [29] offers an excellent treatment of requirements.

Review (SRR). This control gate into Phase B reviews the requirements in order to establish a baseline for the final concept, and thus begin actual preliminary designing.[2]

Developing requirements starts with identifying and defining functional requirements, i.e., requirements that describe "what" a system or product will do. These are first generated for the mission or top-level systems, and come primarily from the mission scope and ConOps. Since these processes help to formulate the architecture, PBS, and interface definitions that follow, functional analysis at these new levels gives rise to more specific functional requirements that can be traced to parent requirements in the scope and ConOps.

The other major type of requirement are performance requirements that state "how well" a system or product must perform its function. These are thus quantitative requirements that come primarily from design decisions made in trade studies.

Other types of requirements exist as well, like interface and verification requirements. Interface requirements are those related to all interfaces between subsystems and with the external world (e.g. command and control, computer to computer, electrical, thermal, data). Verification requirements describe how confidence will be established such that the system will perform in its intended environment (i.e. requirements related to testing and qualifying).

Constraints, however, are defined differently from requirements, despite their similar influence on the mission design. The distinction is that constraints are generally beyond the control of the design, regardless of changes made to it. Constraints are therefore often the drivers behind many requirements, as is the case for the SSPS-TD mission (see below).

[2] [9]

Now, the full requirements document for a mission like SSPS-TD would be longer than this study. Instead, a sample of requirements will developed for demonstrative purposes, that represents the scope of this mission and highlights some of its more unique features. The requirement type, rationale, and parent(s) are given in abbreviated form, as well as an identification tag that makes it easier to identify the hierarchy below the subsystem level. An attempt is made to demonstrate requirements of all types, including interface requirements that correspond to the different interface types identified in Chapter 6.

The first requirements are derived directly from the mission objectives, and thus represent the most top-level set of requirements. They are given in Table 7.1. The rationales are omitted since they come from the mission objectives discussed in Chapter 2.

Table 7.1: Mission Requirements

ID	**Mission Objective** "The SSPS-TD mission shall..."
MO-1	Demonstrate in-orbit wireless power transmission of > 100 kW with space-based solar power.
MO-2	Demonstrate the functionality of key SSPS concept interfaces.
MO-3	Evaluate the performance of all key SSPS concept hardware and interfaces.
MO-4	Measure the end-to-end efficiency of the system.

From these requirements, a set of system-level requirements can be generated that begin to describe the architecture that was chosen. Due to the focus of this study on the space segment, the satellite is treated as the system-level, and a sample of corresponding requirements are shown in Table 7.2.

Table 7.3 presents some of the unique SSPS-TD subsystem requirements that are derived or allocated from the preceding requirements. These are further decomposed to the component or part level in Table 7.4, thus showing the full hierarchical traceability involved in requirements development.

Table 7.2: Satellite Requirements

ID	Requirement	Type	Rationale	Parent
SAT-1	The satellite shall be in an Earth orbit.	Functional	This is the most practical orbit for the SSPS concept where power is delivered to the ground. Also necessary for a technology demonstration mission so that it is accessible and more feasible.	MO-1
SAT-1.1	The satellite shall be in a 1620 km, 32.38° orbit.	Performance	Required for a feasible mission given several constraints (Section 3.7).	SAT-1
SAT-2	The satellite shall maintain continuous contact with the ground-based Mission Control.	Functional	To mitigate mission risk, ensure successful WPT operation, and collect performance data.	MO-3,4
SAT-3	The satellite shall determine its altitude, position, and range to the rectenna.	Functional	For general satellite tracking and determining WPT mode (active vs inactive).	MO-1,3
SAT-4	The satellite shall record the performance and health of all subsystems and interfaces.	Functional	To meet the mission objectives associated with evaluating and measuring the mission performance.	MO-3,4
SAT-5	The satellite shall transmit all measured and recorded data to the ground-based Mission Control.	Functional	To analyze the satellite status and performance, and make command/control decisions.	MO-2,3,4
SAT-6	The satellite shall maintain Sun and rectenna line-of-sight during power beaming.	Functional	To maintain beam power with the solar arrays, and transmit the beam to the rectenna.	MO-1
SAT-7	The satellite mass (with payload adaptor) shall not exceed 21,200 kg at launch.	Interface (mechanical)	Launch vehicle constraint based on orbit.	MO-1, SAT-1.1
SAT-8	The satellite size shall not exceed 22.4 m × 5 m at launch.	Interface (mechanical)	Launch vehicle payload fairing size (does not include adaptor size).	MO-1, SAT-1.1
SAT-9	The satellite shall deploy the transmitter and solar array upon orbit arrival.	Functional	Given the system size and the launch vehicle constraints, a deployment method is required once the orbit has been reached.	MO-1, SAT-9

Table 7.3: Subsystem Requirements

ID	Requirement	Type	Rationale	Parent
PWR-1	The power subsystem shall provide continuous power to all satellite subsystems over the operational lifetime.	Functional	Every subsystem with electronics needs power to ensure successful operation.	MO-1,2
XMTR-1	The transmitter shall generate a 5.8 GHz power beam.	Performance	To successfully perform WPT given atmospheric and technological constraints (Section 3.3.3).	MO-1
XMTR-2	The transmitter shall be body-fixed and electronically steered.	Functional	To ensure the successful operation of the solid-state transmitter.	MO-2
XMTR-3	The transmitter shall be at least 15 m in diameter.	Performance	To meet the minimum incident power density constraint for the satellite orbit (Section 3.7).	SAT-1.1
ADCS-1	The ADCS shall maintain the solar array perpendicular to the Sun during power beaming to within TBD°.[1]	Performance	To provide the transmitter with sufficient energy to form the power beam.	SAT-6, PWR-1
ADCS-2	The ADCS shall maintain the body-fixed transmitter in the Earth-nadir direction.	Functional	To meet optimal power beaming line-of-sight requirements.	SAT-6
TMS-1	The TMS shall maintain the transmitter junction at 300° C.	Interface (thermal)	To ensure that the transmitter operates at nominal performance.	MO-2
REC-1	The rectenna shall measure the input and output power.	Functional	To meet the mission objectives associated with evaluating and measuring the mission performance.	MO-3,4
REC-2	The rectenna shall convert RF to DC.	Functional	To successfully perform a key stage of WPT.	MO-1,2
REC-3	The rectenna shall deliver AC power to the local power grid.	Functional	To successfully perform a key stage of WPT.	MO-1,2
REC-4	The rectenna shall maintain its effective plane parallel to the transmitter during power beaming.	Functional	To satisfy the minimum incident power density constraint and optimize the WPT performance.	SAT-6

[1] TBD left intentionally, as is often done during early requirements generation. This reveals how requirements generation helps to identify necessary parameters that need to be determined.

Table 7.4: Component Requirements

ID	Requirement	Type	Rationale	Parent
PWR-1.1 (SA-1)	The solar array shall provide power to the transmitter and additional subsystems during power beaming.	Functional	To fulfill the role of solar energy in the SSPS concept and provide the power necessary for WPT (including the support roles of the additional subsystems).	MO-1, PWR-1
PMAD-1	The PMAD component shall distribute DC power from the solar array to the transmitter during power beaming.	Functional	To provide the necessary power to the transmitter.	PWR-1
PMAD-1.1	The PMAD component shall provide 80 V DC and 59 W to each converter element.	Interface (electrical)	To meet the power requirements of the solid-state transmitter elements.	PMAD-1
XMTR-1.1	The transmitter converters shall convert DC to RF (EM).	Functional	To successfully perform a key stage of WPT (i.e., construct the power beam).	XMTR-1
XMTR-2.1	The transmitter shall point the power beam to an accuracy <0.1 deg/s.	Performance	To ensure that the power beam is incident upon the rectenna, and with minimal power losses and safety/environmental risks.	XMTR-2

Chapter 8

Satellite Cost

The mission cost is one of the key metrics for determining the viability of the SSPS concept. This is especially important for the SSPS-TD mission since it is acting a technology demonstration for a potential new energy source that would have its own market. Then, treating the SSPS concept as a business makes cost a primary driver for any investment in SSPS technologies and the creation of a space-based solar power market. Cost analysis is therefore concerned with estimating development and operational costs that can be used with the satellite performance metrics to arrive at a quantifiable cost-benefit conclusion, like a utility cost (e.g. \$/kW).

This level of cost analysis, however, requires an in-depth knowledge of the system design, which is beyond the scope of this early concept and feasibility study. Instead, cost analysis can be performed at a top-level to arrive at an estimate that will bound the cost order of magnitude and identify areas where cost is the highest. A reliable cost analysis at this level must then include all costs associated with design, development, testing, assembly, integration, production, and launch. Nominal operation costs are desired as well, but generally too difficult to estimate this early on.

The major difficulty with costing the SSPS-TD mission is that it has very little heritage and no similar missions have been performed to provide cost comparisons. The best estimate is then made using *Cost Estimated Relationships* (CERs) that take into account individual subsystem specifications and TRLs, and overall integration and development complexity. These mathematical equations are the most common approach to performing cost analysis during Pre-Phase A studies.[1]

A good CER-based model is the NASA/Air Force Cost Model (NAFCOM). This government cost model uses a large set of historical data from previous missions and allows for extensive subsystem specifications. It is a software model that is continuously updated and access must be applied for. Total cost estimates for each system or subsystem are broken down into several categories:

- *D&D* - Design and Development cost.
- *STH* - System Test Hardware cost, accounting for the level of testing required.
- *Flight Unit* - Cost of producing the first flight unit, from the start of production to the delivery of the unit.
- *DDT&E* - Cost associated with the developmental effort from the beginning of Phase C/D through to factory checkout of the first flight unit. It does not include costs of the flight unit, but does include labor, material, special test equipment, tooling, and other expenses incurred by the prime contractor.
- *Production* - Cost of the flight unit multiplied by the quantity of flight units to be produced. At the subsystem level treatment of the cost model, all quantities are one for the SSPS-TD mission.

The total cost is then the sum of the DDT&E and Production costs.

The complexity and effort required for several critical processes of the project lifecycle can be specified for each subsystem according to a set of unique rating scales. These are labeled as Common Multi-Variable Inputs and are as follows:

[1] [6]

- *Manufacturing Methods* - 1-5 scale that describes the level of use of advanced manufacturing techniques (1=limited use, 5=maximum use)
- *Engineering Management* - 1-5 scale that describes the level of design changes (1=minimal, 5=major requirements changes)
- *New Design* - 1-8 scale that describes the amount of new design vs heritage design that the subsystem is expected to use (1=flight proven design requiring no modifications, 8=new design, components validated in a lab environment or relevant environment)
- *Funding Availability* - 1-3 scale that reflects the anticipated funding availability (1=funding assured, 3=funding constrained)
- *Test Approach* - 1-3 scale that describes the level of testing required for qualification (1=minimum testing using simulation and analysis, 3=maximum testing at the component level)
- *Integration Complexity* - 1-3 scale that reflects the expected number of interfaces involving multiple contractors and/or centers (1=minimal major interfaces, 3=extensive major interfaces)
- *Pre-Development Study* - 1-3 scale that reflects the magnitude of the study efforts that were conducted prior to the start of design and development (1=2 or more study contractors in Phase A&B, greater than 9 months of study, 2=one study contract with between 9-18 months of study, 3=less than 9 months of Pre-Phase C/D study)

In addition, each subsystems is rated according to a modified TRL scale called the Technology Maturity Index (TMI), that takes into account the experience with the technology, flight experience, test experience, and the application of the technology. This is a 1-12 scale defined by the NAFCOM manual as:

1. Technology research has begun to be translated into applied research and development.
2. Technology is in the conceptual or application formulation phase.
3. Technology has been subjected to extensive analysis, experimentation, and/or a characteristic proof of concept, but has no flight experience.
4. Technology has been validated in a lab/test environment, but has no flight experience.
5. Technology has experience, but not in a space environment.
6. Technology has flight experience, but not recent flight experience.
7. Technology has recent flight experience (< 5 years) and the application of technology is at the edge of experience.
8. Technology has recent flight experience (< 5 years) and the application of technology within realm of experience.
9. Technology is approaching maturity (5-10 years) of flight experience encompassing at least 3 missions and the application of technology is at the edge of experience.
10. Technology is approaching maturity (5-10 years) of flight experience encompassing at least 3 missions and the application of technology within realm of experience.
11. Technology is mature (> 10 years) of flight experience encompassing at least 5 missions and the application of technology is at the edge of experience.
12. Technology is mature (> 10 years) of flight experience encompassing at least 5 missions and the application of technology within realm of experience.

Together, these ratings are extremely useful for a technology demonstration mission like SSPS-TD, which incorporates many innovative and low TRL technologies that play a large role in increasing costs. This cost model is therefore ideal for the SSPS-TD cost analysis, and so the NAFCOM11 Model (version 2011) will be used to generate a cost estimate for the SSPS-TD space segment (i.e., satellite), at a subsystem level (according to the PBS in Chapter 4). Costing the rectenna is not possible with this model and beyond the scope of this study, though it is important to remember that it will add a significant cost.

8.1 Assumptions & Groundrules

The following assumptions and groundrules are made with the NAFCOM model:

- The Uncrewed Earth Orbiting Spacecraft template is used.
- All costs are computed for FY2011.
- Margins are not included since they were not distributed at a subsystem level, and no system-level margin input is available.
- All subsystem specifications are taken from Chapters 3 and 4. Masses are summarized in Table 5.1.
- All subsystems rated at Engineering Management (2), Funding Availability (3), Test Approach (3), Integration Complexity (3), and Pre-Development Study (2) since this is an early design with large uncertainty, little heritage, low technology TRLs, and a high level of complexity.
- Transmitter assumptions/inputs:

- Entered as a communication subsystem
- Mass: 6000 kg
- Manufacturing Methods (1), New Design (7)
- No TMI input available

- Solar Array assumptions/inputs:
  - Mass: 1178 kg, Area: 36,600 m^2
  - Default settings of 4 transmitters, 1 frequency band, partially redundant
  - Manufacturing Methods (1), New Design (6)
  - No TMI input available

- PMAD/Batteries assumptions/inputs:
  - Mass: 1010 kg
  - Default setting of 12 month design life
  - Manufacturing Methods (3), New Design (6)
  - No TMI input available

- ADCS assumptions/inputs:
  - 3-axis stabilized, full sensor suite, fully redundant
  - Pointing accuracy: $< 0.1°$
  - Mass: 800 kg
  - Manufacturing Methods (2), New Design (5)
  - TMI 7 since ADCS methods well-established as individual technologies and can draw on ISS heritage

- GN&C assumptions/inputs:

  - 3-axis stabilized, partially redundant
  - Pointing accuracy: $< 0.1°$
  - Mass: 50 kg
  - Default setting of 12 month design life
  - Manufacturing Methods (2), New Design (4)
  - TMI 7 (same reason as ADCS rating)

- CC&DH assumptions/inputs:
  - Includes the C&DH and TT&C.
  - Mass: 110 kg
  - Default settings of 4 transmitters, 1 frequency band, partially redundant
  - Manufacturing Methods (3), New Design (3)
  - Well-established technology so TMI 11

- TMS assumptions/inputs:
  - 24 month design life, louvers/no heaters, special materials/configurations
  - Mass: 350 kg
  - Manufacturing Methods (1), New Design (7)
  - No TMI input available

- Structures & Mechanisms assumptions/inputs:
  - Large inert structure, significant deployables
  - Mass: 1425 kg
  - Manufacturing Methods (1), New Design (7)
  - No TMI input available

- The following values are used:
  - Program Support is 10%
  - Vehicle Level Integration is 15%

  The last two are above the suggested values since this is an early design concept and the payload packaging is considered one of the most difficult problems for this mission.

8.2 Results

The NAFCOM CERs are not publicly available, but the results are presented in Figure 8.1.

The SSPS-TD satellite cost is estimated at $4.49B (FY2011). As a technology demonstration mission, this is extremely high, but as a general space mission it is comparable to other innovative, state-of-the-art missions. The Hubble Space Telescope, for instance, cost $3.87B (adjusted to FY2011)[2], and JWST is estimated at $8.7B.[3] Given this high cost, it is therefore of utmost important to assess the viability of the SSPS concept before committing to an end-to-end, in orbit technology demonstration.

However, since this cost analysis is occurring very early during concept development, the cost estimate is a rough estimate that reflects the rough design. Adding contingency is therefore of little value to the analysis at this stage. But as the design matures, this cost estimate becomes more refined, and ultimately a monte carlo analysis would be used to develop a cost-risk profile for the SSPS-TD mission. From

[2]http://www.astrophys-assist.com/educate/hubble/hubble.htm (cost to first operational capability

[3]www.space.com/

this, reserves and contingencies and be derived and allocated. A reliable rectenna cost would also be necessary, and require dedicated cost analysts.

WBS Element	D&D (FY2011 $M)	STH (FY2011 $M)	Flight Unit (FY2011 $M)	DDT&E (FY2011 $M)	Production (FY2011 $M)	Total (FY2011 $M)
1.0 Satellite	-	-	134.9	2,969.4	134.9	**3,104.3**
1.1 Satellite Subsystems	-	-	116.4	1,926.5	116.4	**2,042.9**
Transmitter	946.2	13.9	10.7	960.1	10.7	**970.8**
Spacecraft Bus	-	-	105.6	966.4	105.6	**1,072.0**
Power	636.0	57.4	44.1	693.4	44.1	**737.5**
Solar Array	570.4	19.5	15.0	589.9	15.0	**604.9**
PMAD/Batteries	65.6	37.9	29.1	103.5	29.1	**132.6**
ADCS	121.4	53.8	41.4	175.2	41.4	**216.6**
GN&C	3.0	1.5	1.1	4.5	1.1	**5.6**
CC&DH	9.2	3.4	2.6	12.6	2.6	**15.2**
TMS	24.9	5.8	4.5	30.7	4.5	**35.2**
Structures & Mechanisms	34.6	15.4	11.9	50.0	11.9	**61.9**
1.2 System Integration	-	-	18.5	1,042.9	18.5	**1,061.4**
Integration, Assembly and Checkout (IACO)*	-	-	-	-	-	**0**
Systems Test Operations (STO)	-	-	-	158.0	-	**158.0**
Ground Support Equipment (GSE)	-	-	-	454.8	-	**454.8**
System Engineering & Integration (SE&I)	-	-	11.4	251.1	11.4	**262.5**
Program Management (PM)	-	-	7.1	179.0	7.1	**186.1**
Launch & Orbital Operations Support (LOOS)*	-	-	-	-	-	**0**
2.0 Program Support			14.8	326.6	14.8	**341.4**
3.0 Vehicle Level Integration			30.6	673.7	30.6	**704.3**
4.0 Launch Services**	-	-	-	-	-	**335.7**
Total Space Segment Cost	-	-	-	$3,969.7	$180.3	**$4,485.7**

*Calculated as 0 for unknown reasons.

**Delta IV Heavy (From IPAO ELV Database)

Figure 8.1: SSPS-TD Satellite Cost using NAFCOM model

Chapter 9

Strategic Risks

The development of the architecture revealed several key areas of technology that are paramount to the feasibility and success of the SSPS-TD mission, and consequently to the overall SSPS concept. But due to the innovation and technological immaturity of the SSPS concept, many of these technologies have low TRLs and carry with them significant risk. Furthermore, overarching programmatic issues exist as well that can have a serious impact on mission feasibility and performance, and thus represent more source of risk.

One of the primary goals of early concept studies is to investigate all of these potential challenges to the SSPS-TD mission, and their corresponding risks. First, the major technical challenges to the mission are examined. Then, in the next section, they are mapped into risks along with programmatic issues.

9.1 Strategic Hurdles

To borrow the term from John C. Mankins,[1], the technical challenges to the SSPS-TD mission are referred to as *strategic hurdles*. It is important to identify these hurdles for each of these critical technology areas because they are the challenges that should be overcome before the actual development of the SSPS-TD mission. Then, the technologies associated with these hurdles are seen as "game changers",

[1] [40]

and they represent areas of R&D that should be focused on, and thus where initial financial investments should be made. Identifying these strategic hurdles is therefore an integral part of the systems engineering process, and one of the principle goals in Pre-Phase A concept and feasibility studies. The feasibility of overcoming each of these hurdles can then be assessed in order to evaluate the overall mission feasibility, and the eventual viability of the SSPS concept.

For the SSPS concept, and the SSPS-TD mission, most of these hurdles arise due to the large scale of the system in terms of power and structure, and the space-related issues associated with WPT hardware operation. The following key strategic hurdles have been identified for the SSPS-TD mission and should be aggressively investigated:

- *Large-scale Structure, Assembly, Integration, and Deployment*

  Despite being a technology demonstration mission sized to the absolute minimum (given the design choices and constraints), the SSPS-TD satellite is extremely large, rivaling the scale of the ISS. Though the mass remains feasible in LEO, the sheer size of the transmitter and solar array pose a serious challenge to payload packaging and deployment, and is perhaps the most critical issue for mission feasibility or practicality.

  Though the deployment of an individual solar subarray has been demonstrated (on the ground), the SSPS-TD satellite involves a complex solar array design whose deployment is distinctly different: 18 subarrays must all deploy while they are connected to the spacecraft bus and transmitter. The 15 m transmitter must also be designed to fit within the payload fairing and then deploy into a circular structure. This is far larger than any major structure that has been deployed in a single launch, with the design of the JWST 8 m telescope the only system that approaches this scale.

It is therefore evident that major advances in structure design and packaging/deployment configurations are necessary in order to meet the SSPS-TD mission design. Constraints, like the one launch vehicle restriction, may need to be reevaluated as well. Most importantly, efforts should be made to further minimize the size of the satellite. As discussed, the satellite size is primarily driven by the transmitter size and resulting power required, so new strategic hurdles related to the transmitter performance arise from this issue. These are addressed in the next discussions.

Note mass does become an issue if higher orbits are considered. Recall that this is because higher orbits require larger transmitters, and hence larger and heavier systems. After a certain point, a single-launch mission will no longer be possible just from a mass feasibility perspective.

- *Thermal Management*

  There are two challenges related to the thermal management of the SSPS-TD satellite. The first is related to the operating temperatures of the solar cells and the transmitter. To operate efficiently, the transmitter converters must be maintained at a very high temperature (300° C), which is extremely difficult in the space environment. Conversely, solar cell performance is limited to low temperatures. Then, recall that the large surface area of the satellite will be under significant solar irradiation when in the Sun, and in complete shadow otherwise, creating large temperature variations. PMAD efficiency must also be considered in light of thermal effects.

  The second challenge, and perhaps the most important, is that of waste heat dissipation at the transmitter. As seen in the orbit trade study, the thermal constraint (I_t) on the transmittable power density is the most limiting factor to the transmitter size, such that the minimum power density constraints on the ground are met. Raising the value of this thermal constraint is therefore one of the most critical steps in improving mission feasibility and performance. And because this is a design constraint rather than environmental or safety-related, it can be improved with sufficient R&D. The value of 26 kW/m^2 used in this study is a conservative estimate (Section 3.3.4), but a practical value should still be several times larger. However, the unfortunate side effect of increasing this thermal limit is that, while the transmitter size can be reduced, the total transmitted power increases in order to compensate, thereby requiring an even larger solar array.

Ultimately, these two challenges must be solved by investigating thermal management techniques that allow for better waste heat disposal, and better device efficiencies at different temperatures.

- *Large-scale PMAD*

  The SSPS concept relies on an extensive PMAD subsystem that carries an immense amount of power. Due to the space environment, power cannot be transported at very high voltages, like it is in ground-based transmission lines, so mass and efficiency becomes an issue. Viable PMAD designs must be investigated and the heritage from the ISS may provide some solutions. The ISS generates 84 kW of power, which while small compared to the needs of the SSPS-TD mission, is far greater than any other space missions to date, and classified as large-scale, like this mission. Furthermore, the ISS continuously generates this much power, thus requiring a highly reliable PMAD design.

- *WPT Hardware*

  The relatively low TRL of the WPT hardware corresponds to substantial reliability issues and thus calls for accelerated R&D in terms of space-qualifying the transmitter components and proving out the general WPT performance for an endeavor of this scale. Unlike the solar cell industry, WPT technology does not have a large market so technological advances in this area must be specifically targeted and prioritized. In particular, WPT hardware that operates at higher frequencies should be investigated, as this is perhaps the easiest way to drastically reduce the system size.

  The other issue to consider are risks associated with microwave power beaming, namely radio interference and environmental impacts. These are further discussed below, but relate directly to the hardware in terms of testing and qualifying.

Given that the satellite size is the primary driver behind these strategic hurdles, the two most important R&D goals for the SSPS-TD mission should be to increase the thermal constraint on the transmitter, and utilize WPT technology at a higher frequency.

In addition to these major strategic hurdles, a number of general hurdles exist for all SSPS-TD technologies:

- All components must be space-qualified
- All technologies should be robust; they should meet all reliability requirements through proper testing, qualification, and verification
- Conversion efficiencies for the solar cells, PMAD, transmitter converters, and rectifiers should be as high as possible to increase mission performance, with

the added benefit of reducing satellite size.

9.2 Risk Identification & Assessment

As a technology demonstration, the central role of the SSPS-TD mission is to prove out a number of technologies in such a way as to identify and mitigate any risks, and ultimately get the mission approved. Risk analysis is therefore one of the most important steps that needs to be performed, and must include both technical and programmatic risks.

There are many ways to perform risk analysis, like FMEA (Failure Modes Effects Analysis) and PRA (Probabilistic Risk Analysis), but these require an in-depth knowledge of of the subsystems, components, or parts involved in the mission. Since this study is focused more on the conceptual design of the SSPS-TD mission, risk analysis at this level instead involves the identification of risks related to the front-end mission drivers. These drivers include all of the key technologies involved in the mission, and all of the programmatic features that affect mission development and feasibility. The strategic hurdles identified in Section 9.1 can be directly mapped into technical risks, and it remains to determine the major programmatic risks.

The best way to communicate risk at this level is with a risk matrix, where the probability of failure is plotted on the y-axis, and the impact of failure on the x-axis. In this way, all of the risks are plotted together and categorized according to a priority scale. This method of risk analysis will be used to assess the current state of risk of the SSPS-TD mission. In other words, a snapshot will be taken of the risks that are inherent in the current design, before any technologies have been moved up the TRL ladder.

The risk matrix is shown in Figure 9.1 and the risks are identified and explained below. Note that because this risk matrix is assessed qualitatively at a top-level, the exact placement of the risks in the matrix is not absolutely necessary (nor possible); they are relative estimates or approximations.

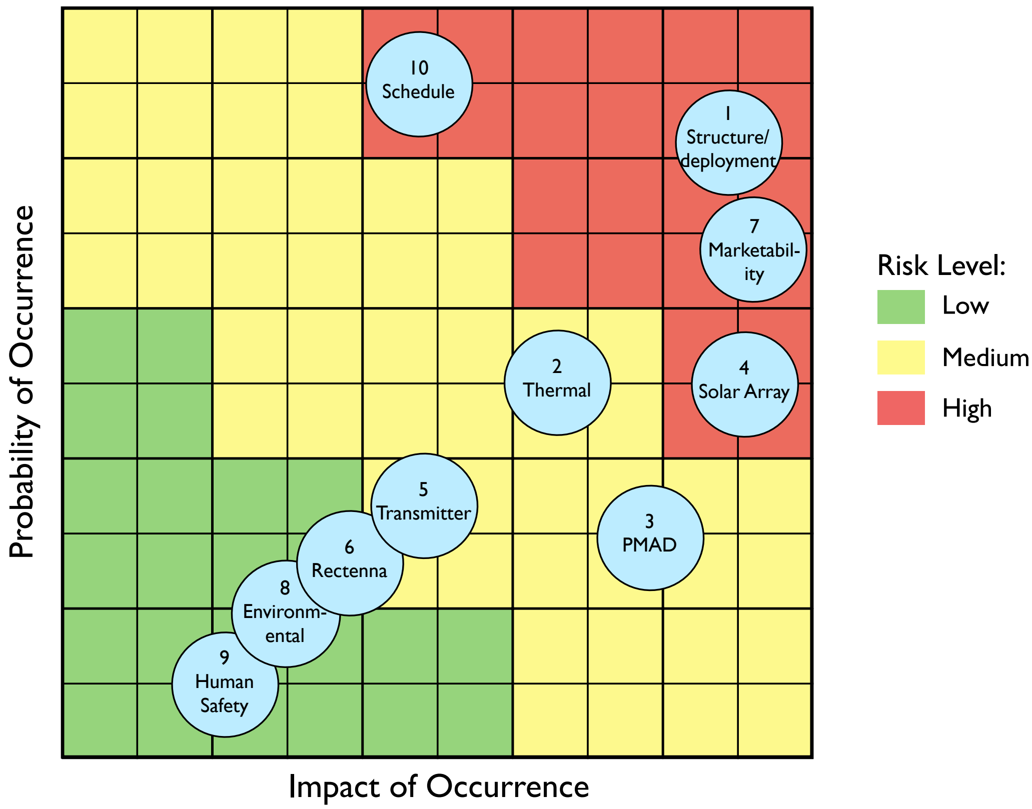


Figure 9.1: Risk Matrix: SSPS-TD Drivers

Relative/qualitative assessment of the major SSPS-TD risks associated with front-end mission drivers.

1. *Structure/Deployment* - The packaging, deployment, and ultimate structure configuration for a system as large as the SSPS-TD is a major design challenge, especially since the only comparably sized system is the ISS, which utilized many launches and in-orbit construction. The complexity and magnitude of this issue therefore corresponds to a very low TRL for the structure subsystem design.

The biggest risks for this subsystem are deployment failure or even the inability to package a system of this size at all, both of which will result in total mission failure. Given its low TRL, the current probability of occurrence for a failure of this magnitude is high.

2. *Thermal Management* - The WPT performance of the SSPS-TD satellite is largely dependent on thermal management subsystems. Risks related to thermal management are associated with component and performance failures, which would drastically reduce the efficiency of the transmitter, and if severe enough, could disable converter elements.

   Under the current design, the passive thermal systems utilize well-established technologies, however, the unique and strict thermal requirements and complexity of the satellite design translate into considerable risk.

3. *PMAD* - The unique and extensive power requirements of the transmitter calls for an innovative PMAD subsystem design, and thus corresponds to a low TRL. But given possible heritage from the ISS and the well-understood design of power transmission systems on Earth, it is likely that a reliable solution will be found. While this lowers the probability of a PMAD error or failure, the impact of such an occurrence ranges from partial to total mission failure, since power is required by the spacecraft bus, and most importantly, the transmitter.

4. *Solar Array* - The risk associated with the solar array centers around the deployment and integration of the multiple subarrays. Individually, each subarray would be expected to deploy and perform reliability, but integrating 18 of them together, packaging them with the rest of the satellite into one launch vehicle, and ensuring successful deployment is a far riskier task.

The largest risk is a deployment failure, which would result in total mission failure, since no power would be available to the transmitter. The probability of occurrence is assumed to be medium since the subarray deployment has been proven on the ground.

5. *Transmitter* - The risk associated with the transmitter is directly related to the low TRL of the transmitter hardware. The solid-state converters have yet to be tested in a relevant environment or as an integrated phased array antenna, so their performance and reliability onboard the SSPS-TD satellite is unknown.

   Risk with the transmitter therefore corresponds to component malfunctions. But since the transmitter is modular, isolated failures are acceptable (and included in the efficiency linkbudget) and do not largely affect the satellite performance. For the same reason, wide-spread failure is unlikely, though it would result in partial to total mission failure, depending on the extent of the failure.

   Note that the risk associated with the transmitter structure and deployment is included in Risk 1.

6. *Rectenna* - Like the transmitter, the risk associated with rectenna is due to its TRL. However, because the rectenna is located on the ground and there have been several small-scale tests near this frequency, it is a much more mature technology. Furthermore, because it is inherently modular, the results from these small-scale tests can be reliably extrapolated to a larger system.

   The real risks with the rectenna are performance errors and malfunctions related to the necessarily complex design. The dipole arrays must be linked to rectifier elements, which must in turn be linked together by an extensive PMAD system in order to be outputted to the local grid. Furthermore, the rectenna must be designed to track the satellite and ensure that the minimum power density constraint is met for each rectifier (Section 3.3.5). The latter point is especially important as any design flaws in power distribution across the rectenna will result in drastic performance degradation.

Since rectenna's have been built and tested before, much of its complexity is assumed to be accounted for in the heritage, and the probability of a failure is considered relatively low. And because the rectenna is located on the ground, it will be far easier to build, test, and maintain, thereby increasing its reliability and reducing the impact of a failure.

7. *Marketability* - The marketability refers to the viability of an SSPS business model based on the SSPS-TD design (recall the mission goals in Section 2.1.2). If a business case cannot be made on the basis of the SSPS-TD mission concept, then there is a risk that the mission will be cancelled or funding will be drastically cut. The impact of this is obviously severe, and the probability of occurrence is high due to the ever-changing market environment, and the "rough" nature of this mission design which is not refined enough for accurate economic analyses.

   Note that this is not a risk for most space missions because they are typically concerned with performing science or gathering data, and so a cost-benefit analysis is not applicable.

8. *Environmental Effects* - Any mission with a ground segment requires an Environmental Impact Study (EIS). And the results from this study will drive the status of the mission approval.

   As discussed in Section 3.2, there are several environmental effects of microwave power beaming, namely atmospheric disturbances and waste heat generated at the rectenna site. However, the SSPS-TD power beam is of an extremely low power density and is not active for more than $\sim$10 minutes at a time, so atmospheric or ecological effects due to the power beam itself are not a concern. Furthermore, the rectenna pilot beam acts as a fail-safe ensuring that the power beam is never activated unless it is correctly pointed at the rectenna, thus avoiding any stray irradiation.

Heat waste at the rectenna site due to the conversion process may have an impact on the local ecosystem. But this should be minimal due to the rectenna location (e.g. desert climate), though it does depend on the rectenna size and the resulting waste heat generated.

The last potential risk is ecological damage due to the construction of the rectenna. Again, the impact depends on the size of the rectenna, and should be mitigated by the remote rectenna location. Furthermore, the rectenna is located on part of a missile test range which means that some environmental impact was likely accounted for when the White Sands complex was first established.

9. *Human Safety* - Like the environmental risks, human-related risks associated with the power beam are nearly negligible due to the intermittent, low power density beam (< 1 W/m^2, well under any safety limits). And since the rectenna site is isolated and strict safety regulations will be in effect, the risk of human exposure is even lower (even accounting for side lobes which are extremely small).

   The only other human risk is exposure to the heat waste, which again, is minimized by safety regulations and rectenna safe zones. Furthermore, this heat would have no sustained effect on a human.

10. *Schedule* - The project schedule is one of the most difficult things to predict, and more often than not, it is underestimated. Given the low TRLs of the SSPS-TD technologies and the uniqueness of its design, the 15 year launch assumption (Assumption 1) is highly improbable. The impact of a schedule delay does not directly affect the actual mission performance, especially since there is no constraint on the launch date. But, violating schedule constraints will lead to higher costs and could result in lower component reliability or faulty design if different stages of the development are accelerated in order to make up for lost time (e.g. less testing). Higher costs could also result in budget cuts in other areas, thereby increasing the chance of failures. Both of these effects could have serious consequences on the mission reliability or even feasibility.

The results from the risk matrix confirm that the satellite size is the biggest technical risk, since it involves the large-scale structure and deployment mechanisms (including the solar array). As expected, the programmatic issues of schedule, environmental impact, marketability, and safety are also classified as "high risk", since they drive the mission development and form the basis for the mission needs, goals, and objectives.

Note that cost is a risk, but is really an undesirable result of the aforementioned risks. In order to ensure that cost is not a risk, it is therefore necessary to examine the risks that feed into it, and invest in reducing them early on.

Chapter 10

Conclusion

The development of the SSPS-TD mission presented in this study has demonstrated a variety of formalized systems engineering processes. These were presented in a top-down fashion within the context of an early concept or feasibility study on the space-based solar power concept. A single need was identified at the beginning which was transformed into a complete mission architecture that represents a closed design solution. From this design, a number of further systems engineering activities are performed, including requirements generation, interface identifications, cost analysis, and risk assessment.

The results of this study reveal that despite the long history of the SSPS concept, the synthesis of this idea into a feasible and viable application still requires a tremendous effort in terms of technology investment and analysis. Critical to this study was the identification of SSPS technologies with low TRLs, thus driving the need for a technology demonstration mission that would further prove them out. There remains, however, a lot of ground work, testing, and analyses that are needed to get a technology mission approved. This study thus represents only a shallow region of a much deeper analysis effort that is required to bring the SSPS-TD mission to implementation.

These conclusions reveal why systems engineering is so important, especially in the early design phases. First, it is able to answer the two fundamental feasibility questions: why hasn't this idea been done before, and what would it take to achieve it? The answers to these questions form the basis for the problem solving that is design, and they lie in the recognition and identification of strategic hurdles, programmatic issues, and risks. These are areas on which R&D and program management should focus in an effort to improve mission feasibility, mitigate risk, and ultimately provide for a better design solution. Systems engineering is therefore concerned with performing technical analyses and understanding programmatic issues, and bridging these two areas within the context of mission design and feasibility.

Secondly, and perhaps most importantly, this study reveals that the mission development process is all about tradeoffs and compromises that can only arise through an iterative systems engineering design process. The nonlinearity of the systems engineering process and the value of iterating is thus emphasized. Though an attempt here was made to show a "first" iteration, this was impossible, and the final design in this study is a product of several iterations. These are not always full cycle iterations, but rather arise naturally at various points in the development stages due to the interconnectivity of different subsystems and design metrics.

Iterations are, of course, are a good thing, and as the design solution continues to be iterated upon, the mission gets more and more defined, trade studies get more in depth, and requirements get more specific. As shown in this study, the first milestone that falls out of this process is the preliminary design, and it is the first confirmation of mission feasibility. The systems engineering discipline therefore allows the engineer to understand the mission feasibility and the effort required to achieve a feasible solution. From this conceptual design, the viability or future potential of the mission and its technologies can be evaluated. For a mission like SSPS-TD, this is critical to assessing the business case for the technology, with the ultimate goal of determining whether or not the mission should be carried out.

In the end, the role of the systems engineering process that has been demonstrated here is best summarized by previous NASA Administrator Michael Griffin:[1]

> System engineering is the link which has evolved between the art and science of engineering. The system engineer designs little or nothing of the finished product; rather, he seeks a balanced design in the face of opposing interests and interlocking constraints. The system engineer is not an analyst; rather, he focuses analytical resources upon those assessments deemed to be particularly important, from among the universe of possible analyses which might be performed, but whose completion would not necessarily best inform the final design. There is an art to knowing where to probe and what to pass by, and every system engineer knows it.

[1] Griffin, M., "System Engineering and the 'Two Cultures' of Engineering", Boeing Lecture, Purdue University, March 2007.

Appendices

Appendix A

Power Beaming Physics and Derivations for Free Space Transmission

This appendix derives the results stated in Section 3.3.1 and further explains the physics involved in power beaming.

A.1 Assumptions

All subsequent analysis will be made under the following assumptions (same as those stated in the thesis):

- All apertures are circular (transmitter and rectenna)

  This simplifies many of the computations later on and is a standard assumption in the literature.

- The transmitter (antenna) acts as a perfect lens (aberration free)

  Lens aberration is beyond the scope of this project, and as an inherent property of the lens, it can be neglected this early on in the design process.

- Unless otherwise stated, the image or observation plane (rectenna plane) is parallel to the transmitter plane, i.e. normal to the optical axis (the axis that runs from the center of the antenna to the observation plane).

- The observation plane is always located in the far-field, or Fraunhofer region

The far-field region is the region where the radiation pattern is independent of the distance from the transmitting aperture. If an antenna with diameter D transmits at wavelength λ, then a point at a distance R is in the far-field region if all of the following three conditions are met:

$$
\begin{aligned}
R &> 2D^2/\lambda \\
R &>> D \\
R &>> \lambda
\end{aligned}
$$

For any SSPS system, R is the orbit altitude, and hence, extremely large. Therefore, the last two conditions are easily met. Furthermore, since λ will operate in the microwave range between $1-15$ cm, and D will not exceed 1 km. Then, the first condition is also met, and the far-field assumption is valid. Note also that most antenna feeds have well-behaved radiation patterns so that the far-field distance is not absolutely critical.[1]

Notation: In the following analysis, the subscript t denotes physical quantities at the transmitter site and the subscript r denotes physical quantities in the image plane, i.e., rectenna site.

A.2 Free Space Transmission

The following sections analyze the propagation of electromagnetic radiation through free space , i.e. no losses through the transfer medium, and subject to the assumptions made in Section 2.1.6. Then, the only "loss" is the free-space path loss due to 1) the natural spreading of the electromagnetic energy according to an

[1] [44]

inverse square law, and 2) the ineherent ability of an antenna to receive power from an incoming electromagnetic wave:

$$L_s = \left(\frac{\lambda}{4\pi R}\right)^2 \tag{A.1}$$

where λ is the wavelength and R is the distance from the point of transmission. Since this is a natural property of electromagnetic radiation it is implicit in all of the following calculations.

Notation: Unless otherwise stated, the terms aperture and antenna are used interchangeably to refer to the transmitting element in the WPT element. The subscript t denotes physical quantities at the aperture site and the subscript r denotes physical quantities in the image plane or receiving site (the rectenna in this case).

A.2.1 Point-Spread Function

The distribution of light in the image of a point source is described by the diffraction point-spread function (PSF), or radiation equation. It can be modified for arbitrarily shaped apertures, making it central to the analysis of a WPT element. By computing the irradiance distribution of the antenna, the intensity is known at any point along the image plane and the power delivered over any specified area can be computed.

To begin, refer to Figure A.1. Consider a circular exit pupil, the transmitting aperture, of radius a and area A_t, radiating a peak power density I_{t_0} at wavelength λ. Let (ρ, ϕ) be the polar coordinates in the aperture plane, where ρ is normalized to a. Let $E_t(\rho, \phi)$ be the electric field amplitude distribution across the transmitting aperture, normalized to its peak value. Let R be the distance between the aperture plane and the observation, or image, plane. Let (r, θ) be the polar coordinates in

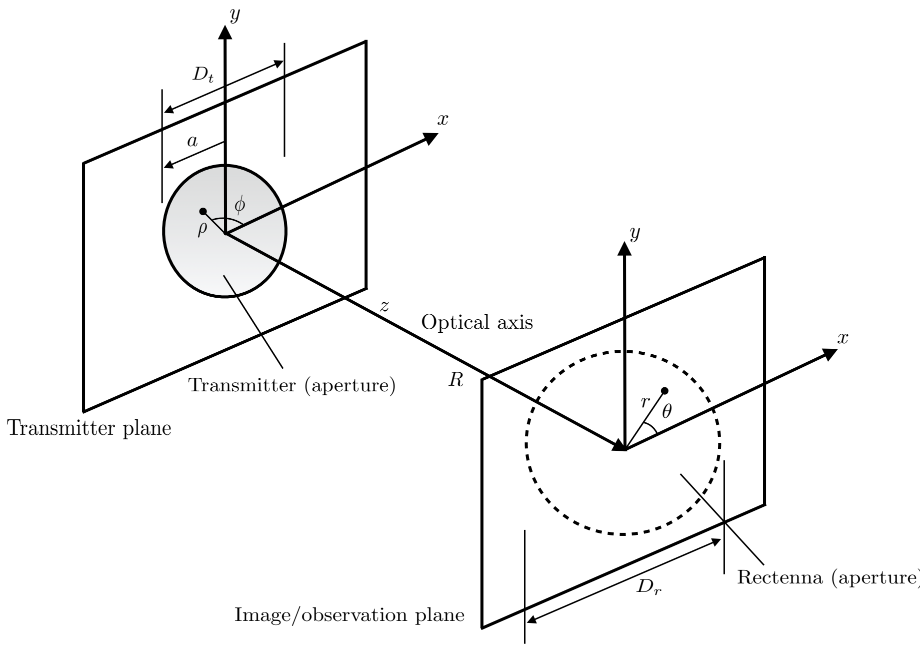


Figure A.1: Power beaming setup and definitions (not to scale)

the image plane, referenced from the point of intersection of the image plane and the optical axis. r is normalized by the factor $2a/\lambda R$. Then, repeating the established assumptions, the far-field diffraction pattern of an aberration-free lens in a plane normal to the optical axis at a distance R may be written as:[2]

$$I(r,\theta;R) = \frac{I_{t_0} A_t^2}{\pi^2 \lambda^2 R^2} \left[\int_0^{2\pi} \int_0^1 E_t(\rho,\phi) \exp\left\{-i\pi r\rho\cos(\phi-\theta)\right\} \rho d\rho d\phi \right]^2 \tag{A.2}$$

The PSF gives the irradiance as a power density distribution in units of $\mathrm{W/m^2}$. This is the radiant flux (energy per unit time) per unit area. In a WPT element this represents the amount of collectible power per unit area incident on the rectenna site.

Frequently, the a perture distribution is rotationally symmetric about the optical axis, i.e. independent of the azimuthal coordinate ϕ, so that $E_t(\rho,\phi) = E_t(\rho)$.

[2]Mahajan, 1991

Then Eq. A.2 becomes:

$$I(r;R) = \frac{I_{t_0} A_t^2}{\pi^2 \lambda^2 R^2} \left[\int_0^1 2\pi E_t(\rho) J_0(\rho\pi r) \rho d\rho \right]^2 \tag{A.3}$$

where J_0 is the zeroth-order Bessel function of the first kind. As expected, the irradiance distribution is also rotationally symmetric about the optical axis at any distance R.

The PSF can always be evaluated numerically, though analytical solutions exist for certain aperture distributions (See Section A.2.3).

A.2.2 Encircled Power Distribution

The encircled power distribution is defined as the fraction of the total power in the image plane contained in a circle of radius r_c (in units of $2a/\lambda R$), centered at $r = 0$. The total power in the image is some fraction κ of the total power transmitted, P_t, depending on transmission losses ($\kappa = 1$ for no losses).

Mathematically, the encircled power distribution is obtained by integrating the PSF:

$$\frac{P_r(r_c)}{\kappa P_t} = \frac{\int_0^{2\pi} \int_0^{r_c} I(r,\theta;R) r dr d\theta}{\int_0^{2\pi} \int_0^{\infty} I(r,\theta;R) r dr d\theta} \tag{A.4}$$

Alternatively, P_t can be written in terms of the power density distribution at the aperture:

$$P_t = \int_0^{2\pi} \int_0^{a} I(\rho,\phi) \rho d\rho d\phi \tag{A.5}$$

For the case of a rotationally symmetric aperture distribution, the encircled power distribution is given by:

$$\frac{P_r(r_c)}{\kappa P_t} = \frac{\int_0^{r_c} 2\pi I(r;R) r dr}{\int_0^{\infty} 2\pi I(r;R) r dr} \tag{A.6}$$

In a WPT element, this represents the power incident at the rectenna site. Multiplying the right-hand side by P_t gives the actual incident power in Watts.

A.2.3 Uniform Illumination

Consider a uniformly illuminated circular aperture, i.e, constant power density I_{t_0} across the aperture. Such an aperture is knows as an *Airy disk*. Then $E_t = 1$ and $P_t = I_{t_0} A_t$. The irradiance distribution can be computed analytically from Eq. A.3:

$$I(r; R) = I_{r_0} \left[\frac{2J_1(\pi r)}{\pi r} \right]^2 \tag{A.7}$$

where $I_{r_0} = \kappa P_t A_t / \lambda^2 R^2$ and is the irradiance at the center of the diffraction pattern, which in our case is along the boresight of the antenna ($r = 0$). J_1 is the 1st-order Bessel function of the first kind. This is known as the *Airy pattern* and is shown in Figure A.2.

Using Eq. A.6, the encircled power distribution is given by:

$$\frac{P_r(r_c)}{\kappa P_t} = 1 - J_0^2(\pi r_c) - J_1^2(\pi r_c) \tag{A.8}$$

This is shown in Figure A.2 as well. The first minimum marks the mainlobe width and occurs at $r = 1.22$, where $I = 0$ and $P_r(r_c) = 0.838$. In other words, 83.8% of the total power is contained in the mainlobe. The first sidelobe peak occurs at $r = 1.64$, where $I = 0.0175$ and $P_r(r_c) = 0.867$. As $r_c \rightarrow \infty$, $P_r \rightarrow P_t$.

A.2.4 Gaussian Tapered Illumination

Of particular interest to the SSPS is a gaussian illuminated aperture, whose normalized amplitude distribution may be written as

$$E_t(\rho) = e^{-\frac{\gamma}{2}\rho^2} \tag{A.9}$$

where γ defines the taper or truncation of the gaussian profile. Knowing that the irradiance is proportional to the square of the electric field, the power density distribution

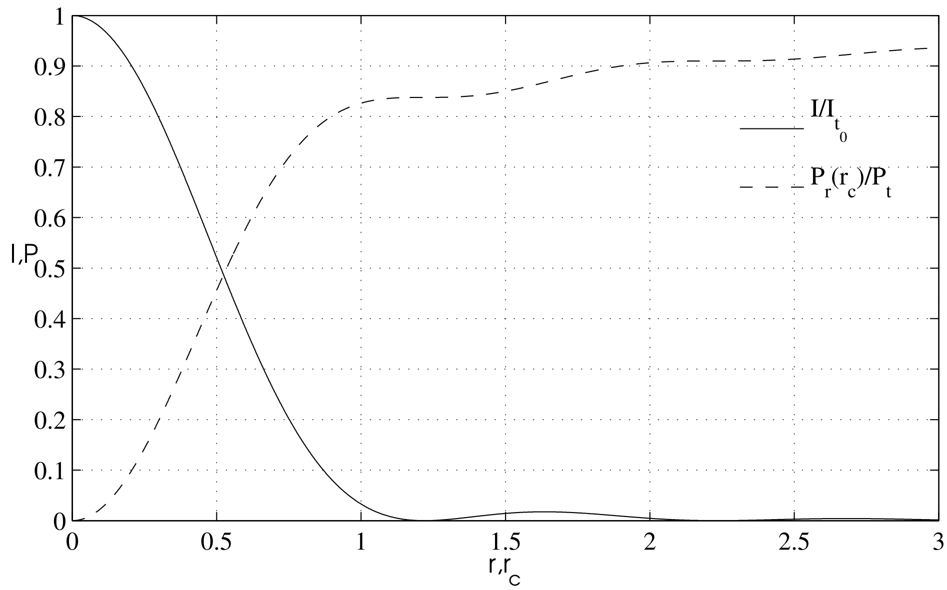


Figure A.2: Normalized irradiance and power encircled distribution for a uniformly illuminated aperture (Airy pattern) with $\kappa = 1$

across the aperture can be written as

$$I_t(\rho) = I_{t_0} e^{-\gamma \rho^2} \tag{A.10}$$

where I_{t_0} is the peak central power density at the aperture. The power transmitted at a point ρ^* can then be written as $P_t(\rho^*) = I_t(\rho^*) dA_t^*$, where dA_t^* is a differential area element on the aperture centered at ρ^*.

The total power transmitted by the Gaussian truncated aperture is obtained by integrating Eq. A.10 over the aperture area A_t:

$$P_t = \int_0^{2\pi} \int_0^a I_{t_0} e^{-\gamma \rho^2} \rho d\rho d\phi = \frac{I_{t_0} \pi}{\gamma} (1 - e^{-\gamma a^2}) \tag{A.11}$$

The taper is often expressed in decibels as the power density at the antenna edge, so that

$$I_t(a) = 10^{-dB/10} I_{t_0} \tag{A.12}$$

where dB is the taper in decibels and a is the antenna radius. For example, a 10dB gaussian tapered antenna transmits a tenth of the central power density at its edge, and 6dB taper transmits about a fourth of the central power density at its edge.

For a specified dB taper, γ is obtained by combining Eqs. A.10 and A.12 for $\rho = a$:

$$\gamma = \frac{dB}{a^2}\frac{\ln 10}{10} \approx \frac{dB}{a^2}0.2303 \tag{A.13}$$

Inserting Eq. A.13 into Eq. A.11, the total power transmitted as a function of the dB taper is:

$$P_t = \frac{I_{t_0} A_t}{dB\frac{\ln 10}{10}}(1 - 10^{-dB/10}) \tag{A.14}$$

In practice, it is impossible to build an antenna with a continuous Gaussian taper (Eq. A.10). Instead, the Gaussian profile is approximated by a certain number of discrete steps. A 10-step 10dB Gaussian taper is shown in Figure A.3. A well-fitted step-function will have an area-under-the-curve essentially equal to that of the continuous distribution so that the power transmitted is the same and the previously derived equations apply.

Now that the aperture distribution and power transmitted is known, the irradiance distribution in the image plane-in this case the rectenna site-is the solution to the following point-spread function:

$$I(r;R) = \frac{I_{t_0} A_t^2}{\pi^2 \lambda^2 R^2}\left[\int_0^1 2\pi e^{-\frac{\gamma}{2}\rho^2} J_0(\rho\pi r)\rho d\rho\right]^2 \tag{A.15}$$

where $E_t(\rho)$ from Eq. A.9 has been inserted into the point-spread function (Eq. A.3), and γ can be found from Eq. A.13. While this integral must be solved numerically, an analytical solution exists for the central irradiance, $r = 0$:

$$I_{r_0} = \frac{I_{t_0} A_t^2}{\lambda^2 R^2}\left[\frac{2\left(1 - e^{\gamma/2}\right)}{\gamma}\right]^2 = \frac{I_{t_0} A_t^2}{\lambda^2 R^2}\left[\frac{1 - 10^{-dB/20}}{dB\frac{\ln 10}{20}}\right]^2 \tag{A.16}$$

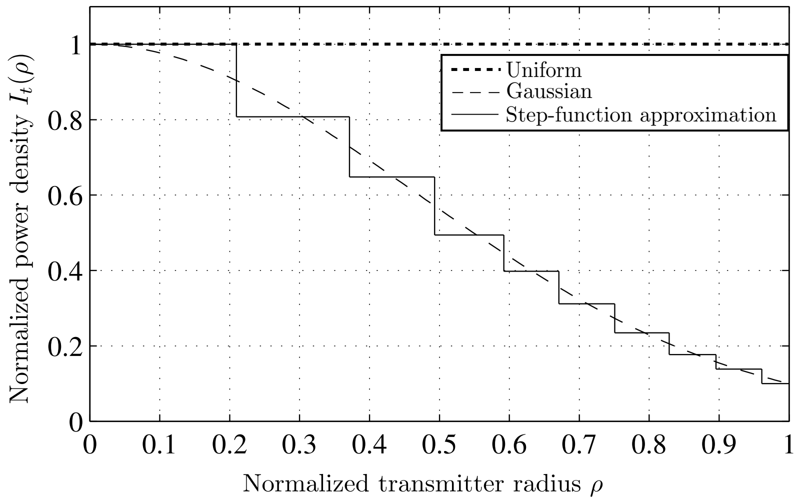


Figure A.3: Normalized 10 dB Gaussian power density profiles across the transmitter

Shown for comparison are the normalized uniform distribution function and a 10-step step function that approximates the Gaussian curve. The total power transmitted is the area under the curve.

where γ is given in Eq. A.13 (and $a = 1$ from the normalization in the above equation). This peak power density at the rectenna boresight is a key parameter in the WPT element design as will be seen later on.

The numerically solved irradiance pattern is shown in Figure A.4 for several taper values, with the uniform density case (Airy disk) as a reference. For comparison, the uniform distribution has the same total power in the aperture as the Gaussian. The plots are divided in order to remind the reader that the uniform power density in the aperture is different for each taper value in order to equate the total power transmitted (so P_t is different for each case as well). In other words, if I_{r_G} is the Gaussian irradiance distribution at the receiving site, andP_{t_G} is the total power transmitted by a Gaussian tapered aperture (Eq. A.14), then a uniformly irradiated aperture (of the same area) transmitting the same power P_{t_G} will do so at a constant power density given by

$$I_{t_U} = \frac{P_{t_G}}{A_t} \tag{A.17}$$

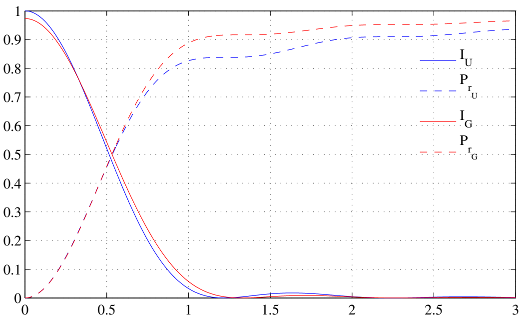
A comparison can then be made by normalizing I_{r_G} to I_{t_U}.

The effect of a Gaussian taper is to broaden the main lobe and lower the side lobe levels. These results offer two very important advantages:

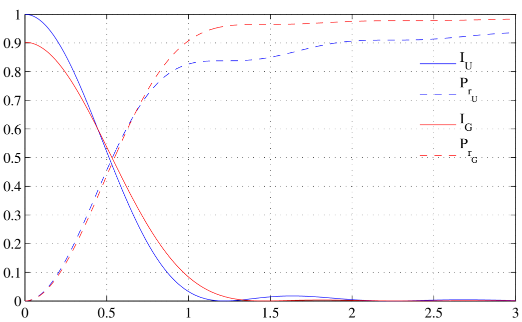
1. *Increased power reception over the same area*: The main lobe broadening means that for a given taper there exists an r_c where the encircled power P_r is greater for the Gaussian profile than for the uniform, allowing more power to be received for the same sized rectenna.

2. *Safe side lobe levels*: The ability to lower side lobe levels is necessary to maintain safe levels of irradiance outside of the receiving area and plays a large role in the design of the WPT configuration.

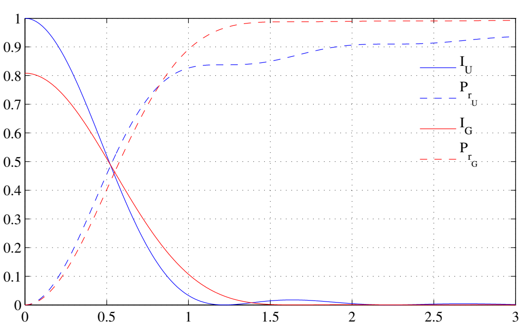
These features are further discussed in Section 3.3.1.



(a) dB=5



(b) dB=10



(c) dB=15

Figure A.4: Irradiance and encircled power distributions for various Gaussian tapers

Distributions are normalized to a uniform distribution with the same total transmitted power. The subscripts $_U$ and $_G$ denote uniform and gaussian distributions, respectively.

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Vita

Julien Chemouni Bach grew up in San Francisco, California, spending most of his summers in France due to a French-American background. He graduated from The International High School with an International Baccalaureate Diploma, and moved to Boston, MA to attend Tufts University. There, he pursued a dual B.S. degree in Physics and Astrophysics. After graduating in 2009, Julien continued with his passion for space and science and went straight to the University of Texas at Austin (UT) to pursue a Masters degree in Aerospace Engineering. Despite a focus in orbital mechanics, it was here that he found his calling in Systems Engineering. He was a teaching assistant for the Space Systems Engineering course for five semesters and the lead systems engineer in the graduate design course.

Upon receiving his Masters of Science in Engineering at UT, Julien plans to seek out job opportunities that involve both systems engineering and renewable energy research.

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