”Energy Orbit “ Wirelessly Powering Satellites using Small Space Solar Power Satellite Constellation

アディティア, バラスカ

出版情報：Kyushu University, 2021, 博士（工学）, 課程博士
バージョン：
権利関係：

“Energy Orbit”

Wirelessly Powering Satellites using Small Space Solar Power Satellite Constellation

A dissertation Submitted in partial fulfillment of the requirements for the degree of Doctor of Engineering

By

ADITYA BARASKAR

Under the Supervision of

Professor Toshiya Hanada

九州大学
KYUSHU UNIVERSITY

Space System Dynamic Laboratory
Department of Aeronautics and Astronautics Engineering
Graduate School of Engineering, Kyushu University
744 Motooka, Nishi-ku, Fukuoka 819-0395,
Japan

(2021)

CERTIFICATE

This is to certify that the thesis work entitled **"Energy Orbit" – Wirelessly Powering Satellites using Small Space Solar Power Satellite Constellation** submitted by **Aditya Baraskar** the partial fulfillment of Doctor of Engineering in Aeronautics and Astronautics Engineering has been satisfactorily carried out under guidance as per the requirement of Graduate School of Engineering, Kyushu University, Japan.

Supervisor: **Professor Toshiya Hanada**

Place: Fukuoka

Date:

Space System Dynamic Laboratory
Department of Aeronautics and Astronautics Engineering
Graduate School of Engineering, Kyushu University
744 Motooka, Nishi-ku, Fukuoka 819-0395,
Japan

(2021)

DEDICATION

This Thesis is dedicated to my parents
Chandararekha and Chandrabhan Baraskar
Who introduced me to the joy of reading from birth,
Enabling such a study to take place today.

This work is also dedicated to
Astronaut Koichi Wakata (D.Eng.),
an inspirational personality and my ideal,
who has always guided me during my space science studies.

Thesis Summary

Wireless Power Transmission (WPT) technology using a satellite-to-satellite system represents a valuable and convenient technology for transferring power wirelessly among Space Solar Power Satellites (SSPS) to Satellite and potential upcoming interplanetary missions. This direct transmission offers a possible solution to deliver continuous, convenient, and unlimited energy supply to satellites help replace traditional power storage and reduce the weight and ultimately the costs of launching satellites. Satellite industries traditionally use photovoltaic cells and nuclear generators to satisfy the needed electricity needed by spacecraft. Current power generation and effective management systems occupy up to 10-25% of the satellite's mass.

The concept of laser-based WPT from Energy Satellite (E-Sat) can overcome substantial problems. The current design of SSPS primarily focuses on designing a massive satellite to generate and transmit gigawatts of energy to an Earth-based ground receiving antenna. This consistent idea can be adopted for spacecraft by developing a constellation of E-Sat called Energy Orbit (E-Orbit) to supply sufficient power to spacecraft within range. It will increase the impressive performance and operational lifetime, especially for small and cube satellites using microwaves and laser-based power transmission. By developing a small scale SSPS, E-Sat for WPT application in space, followed by a practical demonstration of mother-daughter satellite and accurately evaluating the possibility for subsequent implementation. In addition, creating 1600 E-Sats constellations to fulfill the power demand in low earth orbit.

Nevertheless, another potential avenue where E-Orbit supports a possible application is its utility towards rovers and habitat. For instance, the rovers find it difficult to investigate in the far side, crater, and polar region of the Moon, where sunlight is unavailable for a few

days. This challenge can be suitably overcome by employing an E-Sats, which can be used for WPT, independent of its location. Such techniques demonstrate possible applications towards power transmission for unmanned aerial vehicles for faster mapping purposes. As such, the dependence of those aerial vehicles towards fixed energy storage becomes alleviated. Simultaneously the future habitat on Mars and the Moon will receive continuous power by developing a perfect Mars and Moon E-Orbit system for interplanetary and solar-system investigation mission satellites to achieve continuous power.

The theoretical modeling allows the analysis of power conversion or transmission for each unit in terms of laser impacts, transmission efficiency, and photovoltaic-cell thermal property. Maximum power transmission efficiency is calculated based on a linear approximation of power conversation between electricity-to-laser and laser-to-electricity validated by numerical simulation. This efficiency variation depends on the selection of Laser, transmitter, transmission distance, and photovoltaic cells, the same as increasing the maximum transmission efficiency of information in a wireless communication network. Consequently, this thesis gives insight into wireless power transmission in general and adequate guidelines of the satellite to satellite power transmission system design in practice. The development and demonstration of this technology can help fulfill Space Solar Power Satellites' idea to transfer gigawatts of renewable energy to Earth.

Chapter 1: Introduction: This chapter contains the literature review for the energy crisis on Earth concerning the ever-growing population. The production of affordable electricity from renewable and nonrenewable energy and associate problem. Implement a new renewable power system and economic production using space-based SSPS and unique challenges to design it. The current power management systems for space and associated technologies efficiently utilize to sufficiently support diverse missions around Earth and interplanetary mission. Creating E-Sats and a constellation design in low Earth orbit, E- Orbit.

Chapter 2: Literature Review and E-Orbit Refence Mission Designing: This Chapter includes the necessary information on and overviews of SSPS designing and the historical development of WPT, and remarkable experiments efficiently conducted from all over the world. The numerical analysis and exercise of microwave and laser systems for wireless power transmission. The current policies and key challenges around space solar power and wireless power transmission system designing. Discussed the reference parameters for E-Sat and E-Orbit.

Chapter 3: Energy Satellite: The system analysis and system engineering to develop a small-scale SSPS properly as E-Sat. The essential components, high accuracy attitude, orbital control subsystem, propulsion subsystem, active sensors, power generation, and management unit, including laser power transmission subsystem as payload for E-Sat. The power transmission efficiency and power density at receiving object to satisfy user requirements. The novelty of E-Sat compares to the historical SSPS design. The necessary orbital variables and their operational performance inside the proper orbit.

Chapter 4: Energy Orbit: The practical importance of Energy Orbit and the formation flying of E-Sats in low Earth orbit. Orbital characteristics to properly transfer continuous non-disruptive laser power to customer satellite and maintaining power across Energy Orbit. The orbital characteristics and continuous interaction with the customer satellite with the specific decided range of power transmission. The system modeling and necessary phase to phase timeline for progressively developing Energy Orbit across low Earth orbit. Numerical analysis and the other significant scenario can tackle the debris removal, laser propulsion, safety, deorbiting, orbital maneuver—the driven economy and potential revenue within the 20 successful years of launching Energy Orbit in space. In addition, this key section suggests and recommended few subsequent policies to properly consider for the successful development of SSPS and WPT in the context of Energy Orbit.

Chapter 5: Interplanetary Energy Orbit: The designing and power support system for Moon and Mars as Moon Energy Orbit and Mars E-Orbit, respectively. The Constellation designing to transfer power to habitat, Rover, Orbiter, and Rover using E-Orbit. Support for deep-space mission and robotic mission using E-Orbit.

Chapter 6: Conclusion: Summarized the findings of the studies by general conclusions and future scope of Energy orbit.

Table of Contents

List of Figure

List of Table

List of Symbol

α	Incident Angle
∇	Vector Gradient
ω	Argument of Perigee
Ω	Right Ascension of the Ascending Node
a	Semi-Major Axis
η	Efficiency
$\eta_{DC\ output}$	DC Output Efficiency
η_{el}	Electricity to Laser Conversion Efficiency
η_{PV}	PV Efficiency
µ	Earth Gravitational Parameter (398,600.4415 km3 /s2)
σ	Constant
χ	Constant
κ	Visibility
ρ	Size Distribution of the Scattering Particle
A	Area
Ac	Constant
AlGaAs	Aluminum Gallium Arsenide
Amp	Ampere(s)
Ar	Argon
A_r	Aperture Area of Receiving Antenna
A_t	Aperture Area of Transmitting Antenna
B	Earth’s Magnetic Field
bn	Billion(s)
cm	Centimeter(s)
cm3	Centimeter Cubed
CO	Carbon Monoxide
CO2	Carbon Dioxide
$CuInSe_2$	Copper Indium Gallium Selenide
d	Distance
D	Dimeter
db	Decibel(s)

e	Eccentricity
E	Energy
f	True Anomaly
G	Gravitation Constant (Earth: 9.80665 m/s^2)
GaAs	Gallium Arsenide
GaAlAs	Gallium Aluminum Arsenide
GaSb	Gallium Antimonide
GHz	Gigahertz(s)
G_r	Gain of Receiving Antenna
G_t	Gain of Transmitting Antenna
GW	Gigawatt(s)
h	Altitude of Satellite or Hight of Satellite from Sea Level
H_2	Hydrogen Gas
H_2O	Water
HCl	Hydrogen Chloride Gas
HO_x	Hydrogen Oxide Radicals
i	Inclination
I	Current
InGaAs	Indium Gallium Arsenide
InGaP	Indium gallium phosphide
I_s	Solar Irradiance (1367 W/m^2)
I_S	Saturation current
I_{SC}	PV short circuit current
J2	Earth's Oblateness effect (1.0826269×10^{-3})
J_d	Given Integer Number
J_n	Given Integer Parameter
k	Boltzmann Constant (1.380649×10−23 J·K^{-1})
kg	Kilogram(s)
km	Kilometer(s)
kV	Kilovolts(s)
kW	Kilowatt(s)
kWh	Kilowatt-Hour
LiF	Lithium Fluoride

Li-CF_x	Lithium Carbon Monofluoride
Li-SO_2	Lithium-Sulfur Dioxide
LNG	Liquid Natural Gas
LOx	Liquid Oxygen
m	Meter(s)
M	Million(s)
MHz	Megahertz(s)
MM	Magnet Moment
m^3	Meter Cubed
Me	Geomagnetic Strength of the Dipole Vector
MJ	Megajoule(s)
mW	Milliwatt(s)
MW	Megawatt(s)
n	Normal Vector
N	Nitrogen
N_2	Nitrogen Gas
N_2O_4	Dinitrogen Tetroxide
nc	Number of Coil Winding
NM	Control Torque
nm	Nanometer(s)
NH_3	Ammonia
NH4ClO4	Ammonium Perchlorate
NOx	Nitrous Oxide Radicals
P	Phosphorus
P_{DC}	Input DC Power
P_i	Injected Power
P_l	input Power to Laser
P_O	Output Power
PM10	Particulate Matter (under 10 micrometers in diameter)
P_{mp}	Maximum Power
P_r	Received Power
PR	Performance Ratio
P_t	Transmitted Power
q	Electron Charge Constant ($1.602176634 \times 10^{-19}$ coulomb)

Q_E	External Quality (magnetron)
r	Solar Panel Yield or Efficiency
R	Unit Geocentric Position Vector
R_E	Radius of Earth (6378.1363 Km)
R_S	Length of the Geocentric Position Vector
s	Sun Vector
Sb	Antimony
Si	Silicon
SO_2	Sulphur Dioxide
t	Time
T	Temperature
T_Ω	Orbit Nodal Period
TiO_2	Titanium Dioxide
TWh	Trillion Watts Hour
U-235	Uranium-235
UDMH	Unsymmetrical Dimethylhydrazine
V_m	Thermal Voltage
V_o	Output Voltage
W	Watt(s)
Wh	Watt hour

Abbreviation

AC	Alternating Current
AMO	Airmass Zero
APU	Auxiliary Power Unit
BOL	Beginning of Life
CAD	Computer Aided Design
CAST	China Academy of Space Technology
CMG	Control Moment Gyro
CRM	Crew and Resupply Module
CW	Continuous Wave
DDT&E	Design, Development, Test and Evaluation
DRM	Design Reference Mission
EMI	Electromagnetic Interference
EOL	End of Life
FET	Field Effect Transistor
ACS	Attitude Control System
ACSS	Attitude Control and Station keeping Subsystem
AOCS	Attitude and Orbit Control System
Amp	Ampere(s)
ATA	Automatic Threshold Adjust
ATCS	Active Thermal Control System
C&DH	Command and Data Handling
C4MJ	Upright 3J Metamorphic Cell
CAST	China Academy of Space Technology
CCD	Charge-Coupled Device
CMOS	Complementary Metal–Oxide–Semiconductor
CNSA	China National Space Administration
CR	Concentration Ratio
DC	Direct Current
DoE	Department of Energy
EADS-ST	EADS-Space Transportation
E-Orbit	Energy Orbit
EPS	Electric Power System

E-Sat	Energy Satellite
ESA	European Space Agency
ESPOS	Energy Storable Orbital Power Station
GEO	Geosynchronous Earth Orbit
GFRTP	Graphite Fiber Reinforced Thermoplastic
GNC	Guidance, Navigation and Control
GPS	Global Positioning System
HEO	High Elliptical Orbit
IoT	Internet of Things
IR	Infrared
ISAS	Institute of Space and Astronautical Science
ISM	Industrial, Scientific, Medical
ISRO	Indian Space Research Organisation
ISS	International Space Station
ISY-METS	International Space Year Microwave Energy Transmission in Space
JAXA	Japan Aerospace Exploration Agency
Laser	Light Amplification by Stimulated Emission of Radiation
LCRODD	Lunar Crater Observation and Sensing Satellite
LEND	Lunar Exploration Neutron Detector
LEO	Low Earth Orbit
LGA	Lunar Gravitational Attraction
LiDAR	Light Detection and Ranging.
LLNL	Lawrence Livermore National Laboratory
LNA	Low Noise Amplifier
LNG	Liquefied natural gas
LPT	Laser Power Transmission
LRO	Lunar Reconnaissance Orbiter
M^3	Moon Mineralogy Mapper
MINIX	Microwave Ionosphere Nonlinear Interaction eXperiment
MMRTG	Multi-Mission RTG
MPT	Microwave Power Transmission
MSFC	Marshall Space Flight Center
NASA	National Aeronautical and Space Administration
NASDA	National Space Development Agency of Japan

Nd:YAG	Neodymium-Doped Yttrium Aluminum Garnet
NEDO	New Energy and Industrial Technology Development Organization
NIAC	NASA's Innovative Advanced Concepts
OBDH	Onboard Data Handling
OECD	Organisation for Economic Co-operation and Development
OPEC	Organization of the Petroleum Exporting Countries
PTCS	Passive Thermal Control System
PV	Photovoltaic
QPSK	Quadrature Phase-Shift-Keying
RCAST	Research Center for Advanced Science and Technology
RF	Radio Frequency
RLV	Reusable Launch Vehicle
RTG	Radioisotope Thermoelectric Generator
SC	Spacecraft
SDG	Sustainable Development Goal
SERT	Scientific Exploration Research Technology
SGA	Solar Gravitational Attraction
SPS	Solar Power Satellite
SRP	Solar Radiation Pressure
SS	Satellite-to- Satellite
SSPA	Solid-State Power Amplifier
SSPS	Space Solar Power Satellite / Station
TWTA	Traveling Wave Tube Amplifier
UAV	Unmanned Aerial Vehicle
USA	United State of America
USEF	Institute for Unmanned Space Experiment Free Flyer
USSR	Union of Soviet Socialist Republics / Soviet Union
WPT	Wireless Power Transmission

Part - I

Chapter 1

Introduction

Chapter 1

Introduction

1.1. Energy Problem on Earth

Since the Industrial Revolution, electricity becomes a necessity for humans on Earth. Much of the technological developments within the last century show just how much humans have become dependent on electricity, which has become an industry essential to the global economy and sustainability. Electricity is used in various ways, including heating, cooling, processing computers, manufacturing in industries, and provides convenience and connectivity in daily lives. Sources of electricity can be categorized into renewable energy and nonrenewable energy sources.

Renewable energy is defined as natural sources that can be regenerated and replaced over a short period of time, such as solar, hydro, wind, biomass, and geothermal. Nonrenewable energy is characterized by its difficulty replacing after consumption and takes more extended periods to produce sources such as nuclear isotopes and fossil fuels, including coal, petroleum, and gas. Nuclear isotopes are limited on Earth; cannot be produced again after consumption. Fossil fuels are traditional energy sources for electric power plants, automobiles, and industrial plants. The nonrenewable energy source produces electricity by burning fossil fuel which is more reliable and efficient than a renewable energy source. However, they release pollutants and toxic gases into the atmosphere, which causes harsher weather conditions, increased Earth's temperature, all symptoms of global warming. Burning fossil fuels produce large amounts of CO_2, CO, and other toxic gases, which degrade the ozone layer atmosphere around Earth. While recently technology for carbon capture and storage system to store and capture to utilize and convert CO_2 back to its fossil fuel form is in development, it is only in its early stages and not yet capable of restoring the damage done.

However, power comes at a cost. The highest incurred costs are economical due to their unsustainability. Almost 60 % of global primary energy comes from petroleum and gas. Members of OPEC now make up around 81 % of the entire proven global oil supply, besides the nations of the OECD own just around 4 % of the reserves, although using roughly 52 % of the world's total quantity. OPEC accounts for just around 4 % of the reserves.

Other consequences are also present. As the developing nations come online to enhance their living standards dramatically, the nature of energy usage is fast changing. Within the climate change impact framework, energy consumption is expected to climb fast in summer to refresh in the 21st century. Furthermore, the thermal energy efficiency that presently accounts for around 80 % of worldwide electricity will diminish as temperatures increase. As Earth's meager resources become depleted, demands are increasing.

The capacity of a nation to lead successfully in the external politics arena or ideological priorities to defend foreign gas and petrol supplies may be undermined by a financial burden. In addition to issues with safety, carbon-based energy also involves health, and environmental expenses, including air emissions, water discharge, contaminated soil, mining soil degradation, black lung cancer, global warming, and acid rain. Such a nonrenewable fossil-based system is not only costly even so inefficient. Oil fields, coal, and natural gas farms are far from the most productive centers, incurring additional travel costs and/or electrical transmission expenses and frequently leading to loss of power during transit. The archaic distribution infrastructure is incompatible with or overlaps with at least one renewable green alternative to fossil-driven energy.

Utilizing the natural resource that can be regenerated quickly is a renewable energy source, as discussed above. Renewable energy sources do not release toxic greenhouse gases. In comparison, this technology does have high initial costs; even so, it has an economic advantage of sustainability. However, few renewable energy sources, such as wind turbines, are a concern among the ornithologist and ecologists because renewable energy could endanger wildlife, particularly birds and their migration around the Earth. Currently, advancement in the solar panel industry shows the highest potential for energy production at social, commerce, and industrial levels. The problem associated with renewable energies is high initial costs, careful planning, vast land utilization, and unpredictable weather conditions. Renewable energy is still facing many challenges and is yet to become the primary source of energy for global needs.

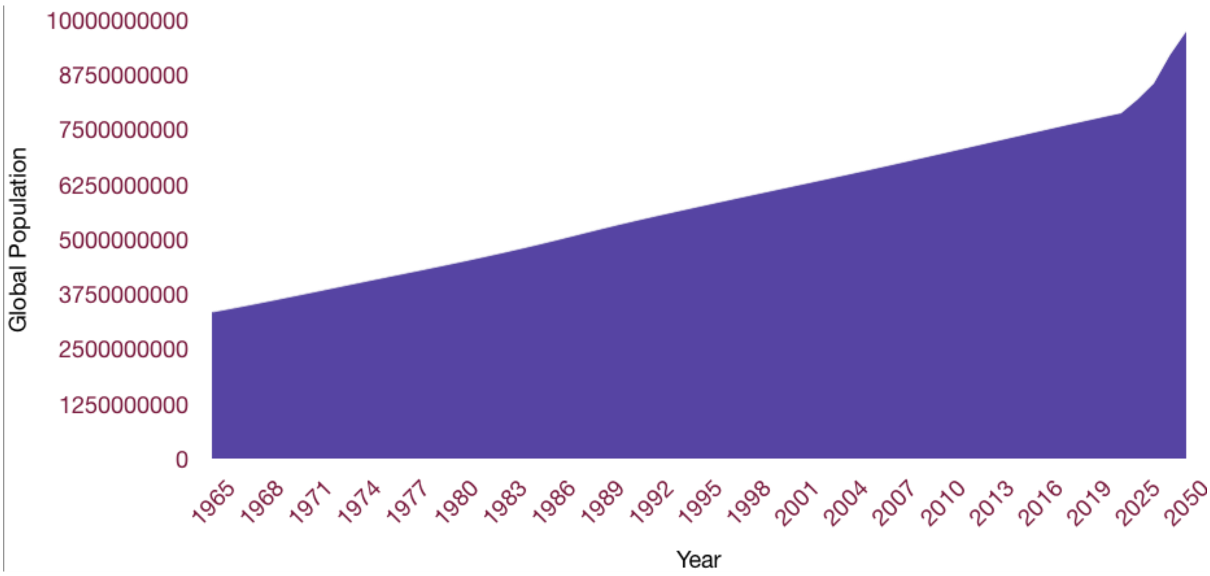


Figure 1.1 World Population from 1965 – 2050

In contrast to the limited energy sources, the demand for energy is skyrocketing. According to the reports, the population on Earth is 7.79 Billion [1] and generates approximately 26,907 TWh of electricity [2], and the global total electricity consumption in 2020 approximates to 23,177 TWh [3], which is significantly increasing year by year. Global electricity demand is projected to double from 2015 to 2050, at 42,000 TWh [4], with the population nearing ten billion, as predicted in Figure 1.1.

To satisfy global energy demands, a new perspective is necessary to build new power plants that include nuclear power plants, coal power plants, hydroelectric power plants, solar power stations, wind power plants, among others. Nevertheless, energy producers will need to look into the perspective of global climate change concerning energy production as such power plants are harming the environment due to greenhouse gases. Two-thirds of global greenhouse gas emissions are burning fossil fuels to produce energy for heating, electricity, transport, and industry. The global energy consumption from the variable energy source is shown in Figure 1.2.

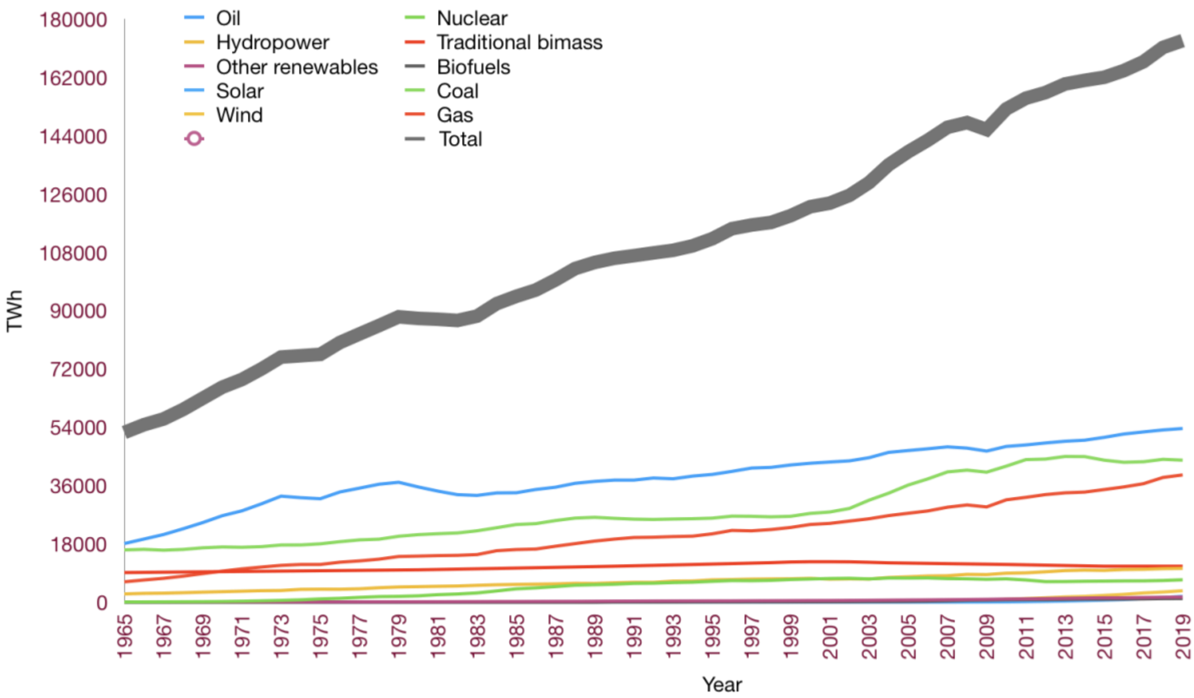


Figure 1.2 Global energy consumption in TWh

In Japan, electricity is costly compare to the other parts of the world. After the 2011 nuclear disaster, only five nuclear power plants out of 22 generators are currently active. Japan's total energy production is approximately 2,592.08 bn kWh, and actual consumption is about 989.3 bn kWh from all sources of electricity; Table 1.1 shows the energy production data with per capita requirement of electricity.

Table 1.1 Japan total energy production

Energy Source [5]	Total in Japan (bn kWh)	Percentage in Japan	Per Capita in Japan (kWh)
Fossil Fuels	1,840.38	71,0 %	14,575.54
Nuclear Power	25.92	1,0 %	205.29
Water Power	207.37	8,0 %	1,642.31
Renewable Energy	518.42	20,0 %	4,105.79
Total Production Capacity	2,592.08	100,0 %	20,528.93
Actual total Production	989.30	38.2 %	7,835.11

1.2. Space Solar Power Satellite and Power Beaming

In the 21st century, electricity has become a necessity for daily life. As an alternative to low carbon generation, a modern-style nuclear power plant offers the lowest electricity cost [6] to satisfy the increasing number of populations versus the electricity demand. Unquestionably, the world's energy needs will dramatically rise in the future, and the quest for alternative sources of energy will increase, as shown in above Figure 1.1. Currently, the world is moving towards the PV-based energy solution by placing substantial solar farms on a broad landmass. Solar farms can provide a significant amount of energy without any ecological impact. However, the cost of manufacturing, storage, and the instability of light sources make it challenging to rely on such farms. For example, a 1 GW coal power plant at 4 km^2 can be replaced by 24 km^2 (for daytime power generation) and six-seven units of power storage facility within three billion USD as shown in Figure 1.3.

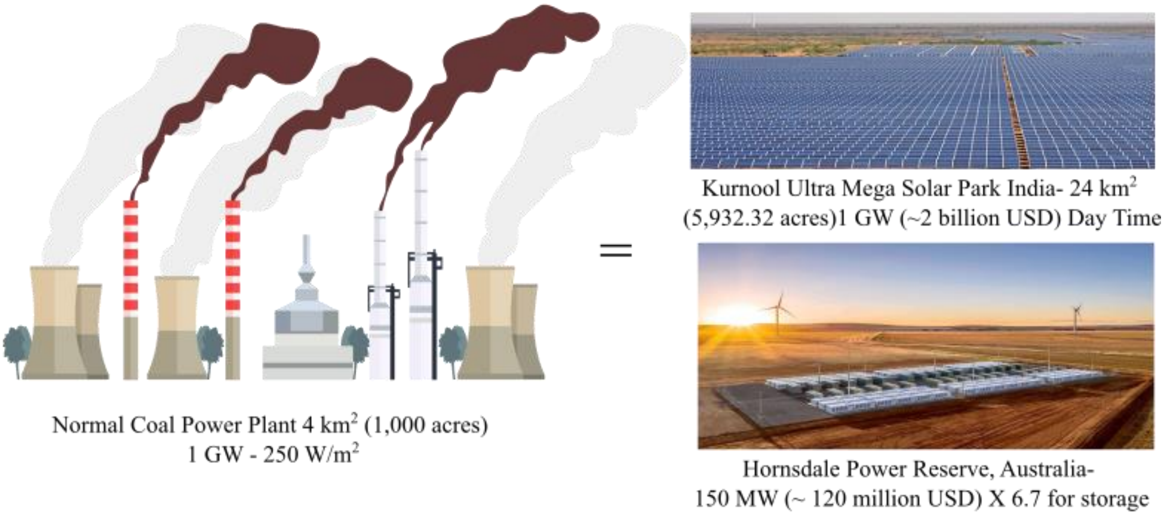


Figure 1.3 Comparison of Coal Power Plant Vs. PV Solar Farm and Storage Facility

Peter Glaser suggests that SSPS will be an alternate source for fulfilling and find a novel concept of transmitting energy from space to Earth. In the era of nuclear energy, highly developed new generation systems provide safety and security to any nation. However, Peter Glaser quoted that in [7] "There are at present serious doubts that nuclear energy can be relied upon to fill a continuously larger percentage of energy requirements since there are no known ways as yet to provide complete insurance against environmental pollution by nuclear energy plants." The recent Fukushima Daiichi nuclear power plant accident leads to the search for an alternative safe and renewable energy source for continuous power supply [8-10]. The concept of SSPS comes into light only after this accident. Meanwhile, the space community has already proposed and investigated several designs long before this accident.

The first design was prosed and patented by Peter Glaser in the late 1960s [11,12]. The design was to collect and convert solar radiation energy to electric energy using GEO based satellite. The produced electricity is converted and transmitted using MPT to a receiving site on Earth. It has advantages over the traditional PV solar panels on ground-based generation units of constant, 24X7, and renewable power supply to any location using WPT. The WPT nowadays is getting popular with the advancement of technology. Cellphones are coming with wireless charging, and IoT devices use short-distance WPT functionality for productivity. The WPT concept was proposed and examined in the 19th century. Nicola Tesla designed the Tesla tower for long-distance WPT [13]. In the experiment, Tesla uses a minor wave frequency for transmission. Since then, researchers have enhanced the capability of WPT using microwaves and lasers, with the characteristics of WPT that include transmission through the atmosphere, simplicity in design with directional emission, and ease to convert back into another energy form. The two types of WPT, MPT and LPT will be extensively discussed in the following chapters.

The dangers of SSPS system operators cannot be ignored. Chapter 2 in this thesis describes the following security hazards associated with SSPS towards the Earth damage and evaluates their capability. Issues considered explicitly under international treaties include licenses, electronic transfer, and liability management methods, including insurance, responsibility waives, and dispute settlement procedures. Demand for electricity nearby Earth orbit for SC, interplanetary mission, and future sustainability will certainly influence business strategies in space.

Advancing the use of wireless communications brought about substantial changes worldwide, and potential innovation to provide value and real-time power supply for connected networks is the subsequent significant advance in wireless transmission of power. While radiative WPT shares many common properties with wireless information or transmission, the performance requirements, transmitter, receiver designs, and hardware limitations also differ dramatically and have been widely investigated.

1.3. Space Solar Power for Space Application

WPT technology using a satellite-to-satellite system represents a valuable and convenient technology for transferring power wirelessly among SSPS to Satellite and potential future interplanetary missions. This direct transmission can help replace traditional power storage and reduce the weight and ultimately the costs of launching satellites. This thesis discusses the WPT between one small-scale SSPS to another operational satellite, followed by a demonstration of small-scale SSPS and evaluates the possibility for future implementation. SSPS will increase the performance and operational lifetime, especially for small and cube satellites using microwaves and laser-based power transmission. The development and demonstration of this technology can help fulfill SSPS idea to transfer gigawatts of renewable energy to Earth. This thesis will summarize several WPT methods, examine the history of radioactive WPT technology, and discuss the significant obstacles facing the conception of modern WPT systems and conclude with describing the new ways of telecommunication and communication systems to face these issues with power transmission. Topics presented include energy harvesting, WPT power transmission, channel acquisition, WPT Smart Energy Area Characterization, linear- and semi-energy-receiver model wave forming, WPT security and health problems, WPT-based SSPS management wireless charging.

The current design and technology enhanced the original ideas of Glaser. Deployment collectors can be situated in the GEO, MEO, or moon surface while visualizing arrays in the GEO. More recent plans have even allowed SSPS to be placed at LEO with much less energy to go to ground stations. Glaser foresees photovoltaic panels as collectors to collect the solar radiation in a receiver that absorbs radiant energy. PV film designs are becoming thinner with technological development. Although overall efficiency has increased over time, new materials with longer lifespans are expected to appear in the market shortly.

Other novel technologies include both MPT or LPT (coherent visible or infrared light). While the efficiency of the microwave has grown considerably, laser transfer can become more comparable. The WPT would be gathered in broad areas antenna (or rectennas) from one to ten km across the ground. The antennae would be rectified. One of its many advantages is that the energy created does not pollute, and no toxic waste needs to be disposed. Furthermore, the soil below the rectennas may be utilized for other industries, such as agricultural or aquatic factory farms. The differences are somewhere between these regions and those needed for coal and nuclear power plants. The cost and performance success of SSPS systems is mainly determined by favorable political and legal conditions as its technological capabilities are evident.

One such variable is the safety hazards and potential liabilities of the SPS system operators. Developers must address complications of the culpability of harm and damage under international treaties. This way, satellite to satellite power transfer can be performed using small-scale SSPS, Energy Satellite or so-called E-Sat. A constellation of 1600 E-Sats will create an orbit across Earth to electrify the LEO. It will be called Energy Orbit or E-Orbit. Will be able to provide energy to customer satellites via laser and microwave to fulfill their energy demands for performing operations. So, clients will be able to send their satellites without mounting any power generation components on satellites, decreasing the weight and increasing the satellite's lifetime. Hence, the launch cost will be reduced. Initially, E-Sat's altitude will be near 900 KM, where the density of active satellites is minimum. However, by 2026, the number of satellites will be increased in 400 – 1400 km attitude. E-Sat will provide energy to customer satellites in the 500 km radius, even in eclipse mode.

Using the 900 km altitude of E-Sats, almost more than 60 % of satellites in LEO will be in the range of the constellation of 1600 E-Sats. Customer satellites near the E-Sats will receive energy via microwave or laser, and the remaining customer satellite in a 500 km radius will receive energy via laser. However, with the progress of this project, the investigated a basic functional block diagram throughout the LPT system explained in Figure 1.4 where the PMS is working for generating and converting power from PV cells convert into DC form. Afterward, the DC will be converted to a laser using an optical antenna for power transmission to the objective destination with the confirmed transmission. Power generation and power storage devices will no longer be required thanks to constant LEO to LEO energy supply.

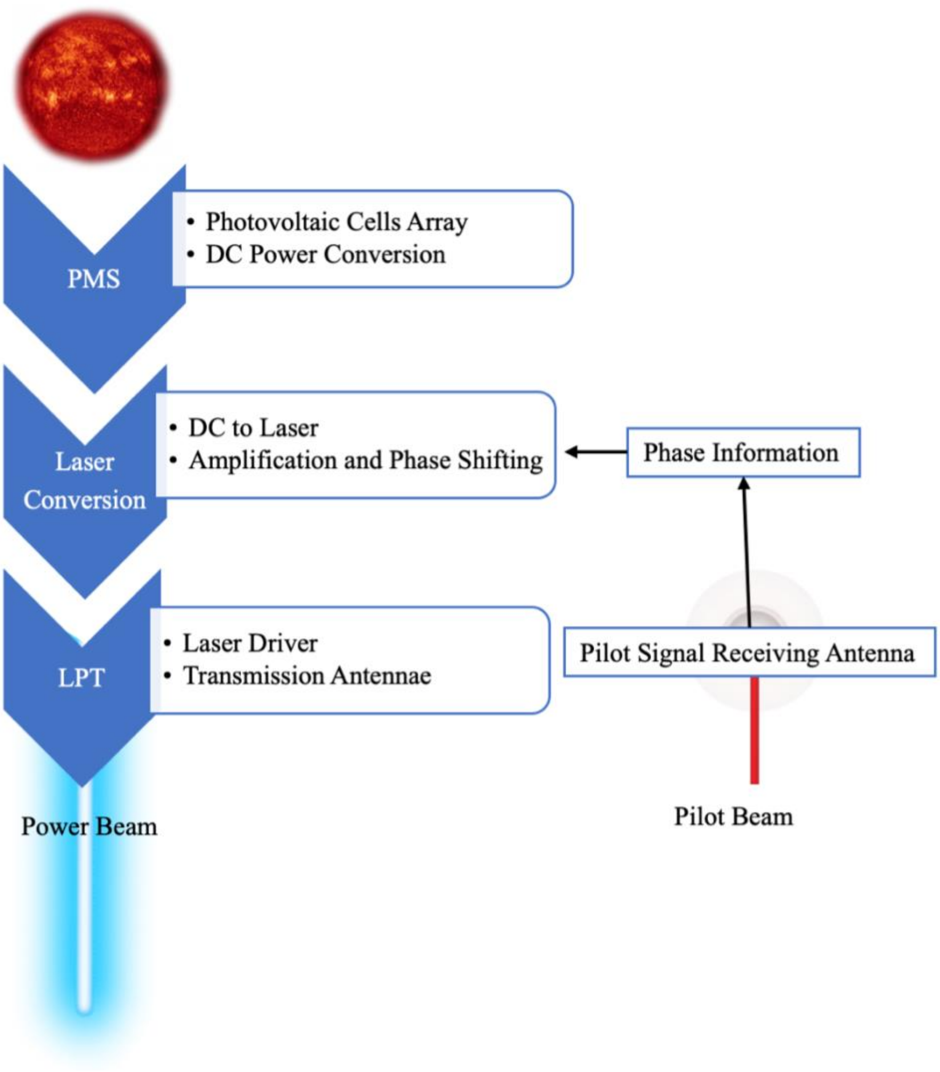


Figure 1.4 Generic Space Solar Power Satellite and Power Beaming functional block diagram

Solid-state lasers like Nd-YAG ranging of 10 kW will be used for transmitting energy via laser beams. 200 W laser source from a small group of E-Sats will help some customer satellites for orbit transfer called laser propulsion. This propulsion will be achieved by three-axis movements of laser source and highly accurate rectenna on customer satellites. Hence rectennas will replace solar panels and batteries on operational satellites. The weight of the propulsion system of the customer satellite will also be drastically reduced, and the life of such a satellite will increase. The technical demonstration mission is planned to launch to validate technical aspects of E-Orbit. It contains two satellites where one of them will be an E-sat called a mother satellite, and another one will be a daughter satellite which will receive energy from the mother satellite. It will launch in an orbit of inclination 90 degrees to the Sun.

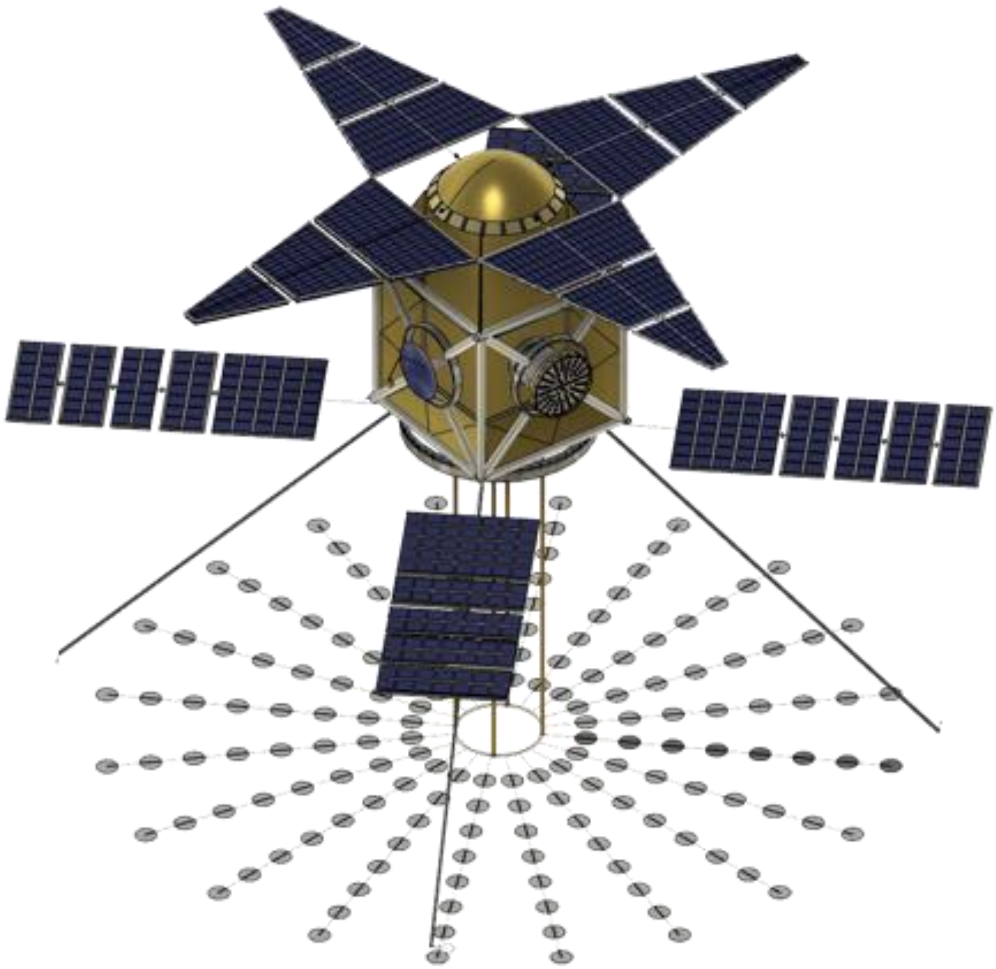


Figure 1.5 The conceptual design of Energy Satellite (E-Sat)

Currently, designing, development, validation, and analysis of E-sat are in the last steps. The demonstration model of the E-sat will be the small-scale SSSP satellite. It will send laser energy to one daughter satellite from lasers of 10 KW with efficiency ranging from 8 - 12 %. Mother satellite will generate energy from both sides of solar panels and thermoelectric generators. Thin foiled and thermal white reflectors will reflect sunlight on the backside of solar panels. Mother satellite will be mounted on the bus of 3×1×1 m. During the Earth reentry mission, the Sun-facing side will have a conical shape to protect the system for reusability, as shown in Figure 1.5. A timeline of the entire project is shown in Figure 1.6, as the project of E-Orbit is started in October 2018 with the idea of creating an SSPS system in compact and with current technology to make it appropriate for business point of view.

1.4. Power Management System in Spacecraft

Space exploration and spacecraft have become more advanced and require more computational power, necessitating cutting-edge power management systems. Power products meet the restrictive voltage specifications of microprocessors, microcontrollers, PV, sensors, payload, and other harsh-environment space applications. Typically, PMS includes PV array, batteries, wire, heater, cooler, heating pipe, and subsystem for operational uses.

The Sun is a massive power hub, luminating a tremendous amount of energy of approximately 3.827 × 1026 W/second. Nearby the Earth's orbit, the solar irradiance is about 1360 W/m^2 with slight changes according to the distance due to elliptical orbit. Most of the satellite uses PV array for power generation in space using the energy illuminated from Sun to produce power. Recently PV industry progressed tremendously in the research of solar cell efficiency. Space agencies use nearly 30 % of their budget for solar cells, whose maximum conversion of energy to power efficiency only adds up to 47.1 %.

Batteries also play a vital role in the PMS system, storing energy for eclipse time to serve the required power at payload and onboard subsystem of satellites. Recently, Lithium-ion batteries are mostly used in space industries, for example, Li-SO_2 and Li-CF_x with specific energy densities of 238 Wh kg^{-1} and 414 Wh kg^{-1}, respectively work between - 40 and 70 °C [14]. Previously, most space missions used NiCd, NiH and AgZn with a specific energy of 30, 60, and 100 in Wh kg^{-1}. The hydrogen fuel cells are being used for the Space Shuttle mission to generate 12 kW with a specific power of 275 W kg^{-1}. However, batteries have disadvantages, such as having a finite life cycle that defines the operational life of any satellite [15].

The connecting wires, transmission wires, sensor system, payload subsystem, motors, and onboard data handling system are also factors considered in PMS for regulating and controlling the temperature in order to satellites properly in every condition. The PMS system occupies approximately 30 % of the satellite mass, in which the PV and batteries occupy an area of more than 80 % depending on mission requirements. Using E-Orbit, the PMS system of a satellite can be reduced up to 25 % of its mass. This is defined in the section below following the descriptions of E-Orbit, E-Sat, and their comparison to the traditional power management methods in chapters 3 and 4. The E-Orbit will play a significant role in interplanetary missions to generate continuous power 24×7 on the Moon even when it is on the dark side. The constellation is designed to produce power and transmit energy from one satellite [16] to the other even when they are not direct sunlight.

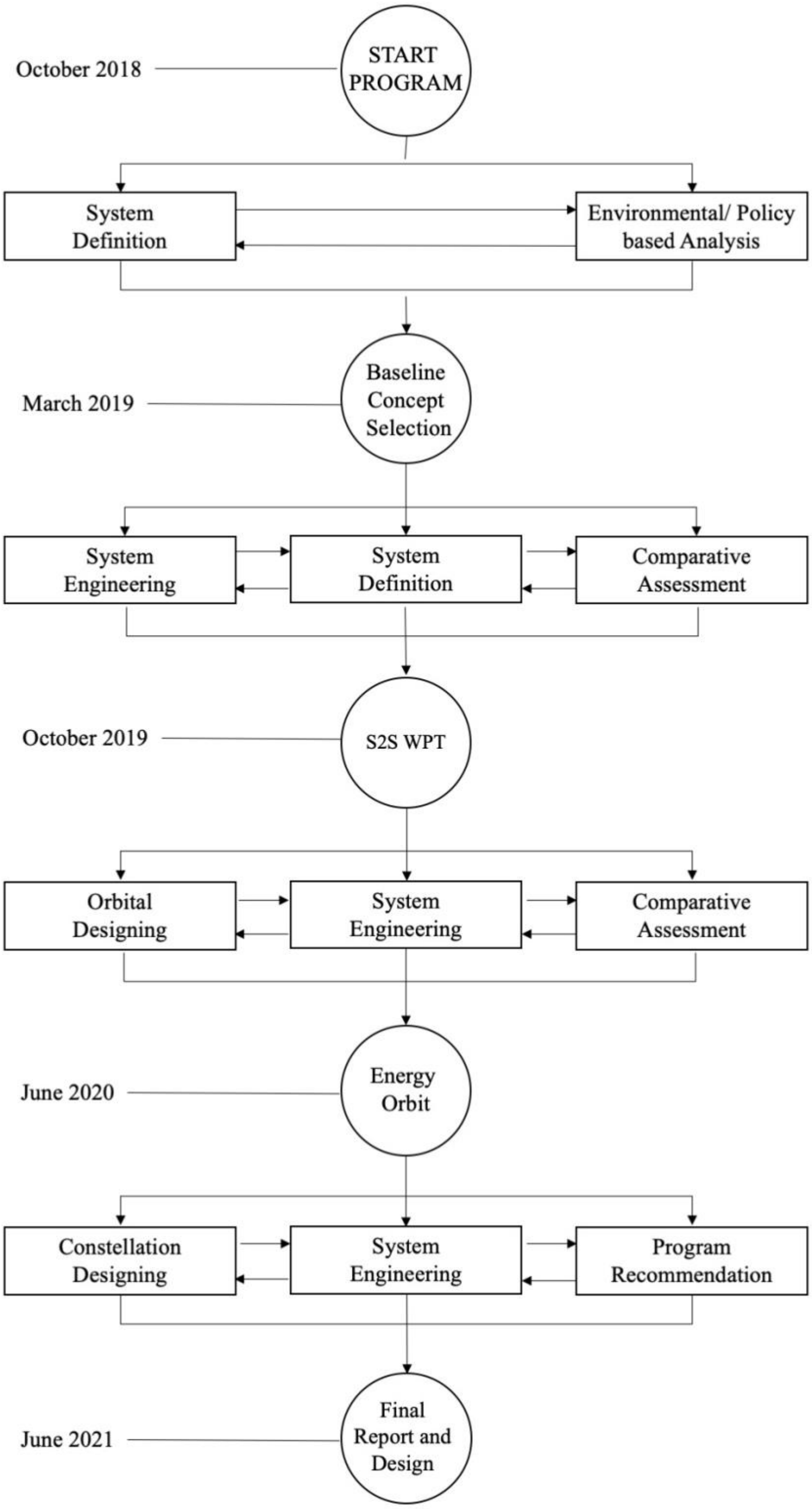


Figure 1.6 A timeline of E-Orbit study and design Evolution Model

Chapter 2

Literature Review and E-Orbit Reference Mission Designing

Chapter 2

Literature Review and E-Orbit Reference Mission Designing

2.1. Space Solar Power Satellite

Space Solar Power Satellite/ Station or Space Solar Power as SSPS represents a method of generating renewable energy from the Sun by collecting photonic energy on solar PV array in the form of DC and converting it to electricity to transmit via the microwave, millimeter-wave or laser to Earth or other applicable missions. SSPS was first introduced by Dr. Peter Glaser [17] in 1968, who was concerned with increasing energy demand and industrial consumption with increasing population. He proposed GEO-based SSPS to harvest MW of solar energy and transmit through microwave to a ground base rectifying antenna or rectenna. The rectenna converts microwave energy into a desirable DC power to feed into the on-site electric grid and dispatch it back to the consumer site. He imagined that solar power would be a cleaner energy source.

After recognizing the need to replace the current energy source to clean and renewable energy, most agencies and countries started surveying researching the SSPS system. NASA conducted the first SSPS study, Dr. Peter Glaser and others with Arthur D. Little, Inc. [18] in 1974. The team introduced a brief description of the SSPS system with key advantages, environmental effects, and estimated costs. In 1976 possible SSPS conversions and delivery system study was conducted by NASA [19] Grumman Aerospace Corporation. In 1978 [20], a collaborative work between the US DoE and NASA on concept development of about a massive scale 60 SSPS systems to transmit 5 GW power to rectenna [21] and later till 1980, several comparative studies were conducted, including for MPT, antenna designing cost-effective SSPS system [22].

From 1980 to the 1990s, several international space agencies and leading countries are considered SSPS as crucial technology and started investigating. Japan, Europe, Russia, and Canada started the experiment with Kyoto University leading the first satellite-based experiment to scientifically investigate the nonlinear interaction of strong microwaves with the ionosphere (In august 29, 1983.) called Microwave Ionosphere Nonlinear Interaction Experiment -MINIX [23, 24].

ISAS and Toshiba proposed a 10 MW SSPS system called ESPOS which in LEO which transmits potential energy every 114 minutes after a revolution to transmit for 35 minutes, and the energy will be stored as heat fluid, with LiF used as the thermal Transfer fluid for the Brayton Cycle Engine [19, 20]. In 1992 International Space University conducted a Space Solar Power Program at Kitakyushu, Japan, collaborating with NASA and other space agencies, to present the final report on the SSPS system. That year, they proposed a reusable transportation system economic-based design concept [20] and in 1993, Japan completed another experiment the ISY-METS sounding rocket experiment [21]. During this experiment, leading researchers scientifically investigated the SSPS rectenna system based on Japan's experiments on the mother-daughter MPT system to evaluate the nonlinear plasma effect on high-power microwave beams in space.

A space solar power working group proposed SPS2000 (1993) from ISAS Japan to place a satellite at LEO with a unique prism shape to transmit 10 MW of power by MPT 2.45 GHz frequency by slots antenna. The NEDO Japan proposed Sunshine Project in 1994 to transmit 1 GW by MPT at 2.45 GHz using the dipole antenna with a diameter of 1 km [24].

From 1995-1997 NASA scientifically re-investigated the futuristic look of the SSPS system as a "Fresh Look Study" for a large-scale SSPS system and improved the SSPS abstract by defining a road map for future SSPS technological promotion [25]. In 2000 NASA's MSFC start the SSPS SERT program, which published the modern prospective technological demonstrative concept and experiment to advance SSPS system articles in which many scholars joined with their exploratory work and comprehensive research on SSPS. Furthermore, Mr. J. Mankins successfully demonstrated the MPT experiment in Hawaii (2008) in collaboration with researchers from Texas A&M University and the University of Kobe to transmit energy at a distance of 148 km; in the successful experiment, they received 0.00001 W. In 2001, ESA proposed multi-SSPS sails called Sail tower SPS with 60 pairs of 150 m^2 to efficiently produce 3700 kW by a total of 450 MW in GEO using MPT 2.45 GHz.

JAXA proposed a new SSPS JAXA model in 2004 utilizing a mirror-based SSPS system using MPT 5.8 GHz to transmit 1 GW of power. Mitsubishi proposed Distributed SSPS Solar Bird Idea in 2005 to transmit 1 GW using MPT 5.8 GHz using multiple satellites. Jspacesystem and JAXA proposed the SSPS USEF Model and JAXA L-SSPS model, with 1 GW from MPT and LPT. In 2011 the attractive eye pointing idea was proposed by Mr. J. Mankins and NASA, SPS-ALPHA [21]. It consists of a large earth-oriented array, directional mirror work as heliostat that adjusts the direction of the mirror according to the Sun to concentrate power with a different type of SPS – ALPHA. China has arisen with a modern idea of SSPS- OMEGA with mirror-based high concentrated power generation [26].

Placing a large disk of solar panels in GEO to collect solar energy in space and utilize WPT to transmit the energy in the form of microwaves to the receiving station on Earth [7,17]. This design demonstrates several advantages in the terrestrial-based solar power generation method, as this avoids the impediments of inclement weather and nighttime outages during dark periods when no solar energy can be collected. The satellite can transmit continuous, renewable, clean, affordable, and safe power supply to any location on Earth, except during short periods when it will go under an eclipse.

The NASA, JAXA, DoE, Naval Research Laboratory, USSR, Republic of India, and CSNA investigated and proposed more than 27 significant variant designs that generate hundreds of kW to GW of continuous power and transmit using microwave and laser-based WPT, which include, e.g., NASA/DoE Microwave Sandwich Concept, 1980, Japanese SPS2000, 1995, SolarDisc, 1997, SunTower1997, Solaren2010, SPS-ALPHA2013, CAST multi rotary Joint SPS, 2015, SPS OMEGA 2014, SPS-Alpha MKII 2016, CASSIOPeiA 2017 which are all advanced SSPS designs as shown in Table 2.1.

Table 2.1 Conceptual design of various SSPS design

SSPS	**Shape**	**Orbit**	**Dimension**	**WPT (MW)**
PETER GLASER, 1973 [7]	Disk Type	GEO	64.749 km^2	10,000
DoE/ NASA, 1979 [27]	Rectangular	LEO/GEO	I0 × 5 × 0.5 km	5,000
JAXA SSP2000, 1993 [28]	Prism	LEO	336 × 336 × 303 m	9.8
NEDO Sunshine Project [29]	rectangular	GEO	3.2 × 2 km	1,000
Sun Tower 1997-1999 [30]	Multiple Disk	GEO	200 × 300 m	1,200
SERT, NASA 1999-2000 [31]	Multiple reflectors	GEO	4 × 7.2 km	0.1 ~ 10,000
Sail Tower SPS 2001 [32]	Multiple Square	GEO	60 pair × 150 m	450
SSPS JAXA Model 2004 [33]	Pair reflector	GEO	2.5 × 3.5 × 2 km	1,000

Table 2.1 Conceptual design of various SSPS design (continue)

SSPS	Shape	Orbit	Dimension	WPT (MW)
SOLARBIRD SPS Mitsubishi Electric 2004 [34]	Pair Reflector	LEO	1000 kg	1,000
SSPS USEF Model 2006 [35]	Square	GEO	2.6 × 2.4 km	1,000
JAXA L-SSPS 2006 [36]	Reflector	GEO	0.4 × 0.2 × 12 km	1,000
Solar Power Beaming, LLNL 2009 [37]	Concentrator and Reflector	LEO	9125 kg	1
Aerospace Corp Laser Concept 2009 [18]	Multiple PV array	GEO	29.7 ton	1200
Solaren SSP 2011 [18]	Reflector	GEO		250-2,250
SSPS ALPHA, 2011 [38]	Multiple reflectors	GEO	25,260 metric tons	2,000
Multi-Rotary Joints SSP 2015 [39]	Multiple PV	GEO	10,000 ton	1300
Sunflower Thermal Power Satellite 2015 [40]	Concentrator and heat engine	GEO	29,500 ton	5,000
SSPS OMEGA 2015 [41]	Multiple reflectors	GEO	3.4 ~ 4.5 km	2,000
CASSIOPeiA [42]	Rotating Multiple reflectors	GEO	866 tons	688
Energy Satellite [43]	Multiple Reflector	LEO	56 m^2	0.01

Table 2.1 describes the SSPS concept from Peter Glaser, 1973 idea to the space System Dynamic Laboratory 2019 Idea Energy Satellite or E-Sat in the LEO. with the shape, orbit,

dimension, and desired transmission power from various SSPS projects. However, Most of the SSPS is MPT-based. Only a few are LPT-based.

2.1.1. United States of America

2.1.1.1. Peter Glaser' SSPS, 1973

Peter E Glaser Proposed the first-time design of SSPS to collect and convert solar radiation to microwave energy. The Microwave energy is then transmitted to an earth-based ground receiving antenna or rectenna and converted to electric power for distrusting for commercial and private use. The president of the USA's message to congress for the demand for electricity and necessity was addressed on June 4, 1971. With this, a collaborative study group is formed for creating the SSPS study program.

In his design of SSPS, he proposed two GEO satellites at 35888.371 km above the Earth parallel to Earth's equatorial plane and perpendicular to Sun in SSO. The Satellites were moving from East to the west with stationary to earth receiving point in orbit. The satellite is placed about 21° out of phase (12713.84 Km) apart with a direct line of sight. Using the partition and gape will provide a continuous power supply to Earth even in eclipse time a year. The satellites are designed to convert solar energy using PV cells with the conjunction of the Seebeck effect to convert the radiated heat across the satellite by high vacuum and plasma diode and incorporated with the high optical concentrators to focus the solar radiation PV cells. To maintain the temperature, liquid nitrogen and helium would be used to treat PV cells at -258.15° C degrees, in addition to thermal insulation, and multistage cryogenic refrigeration systems to absorb heat leaks. This design would generate and covert energy using 40.23 km^2 PV cells structure to transmit 10 GW (20 kV at 5×10^5 Amp) using a combination of 10,000 amplitrons, each with a capability of 1000 kW, as shown in Figure 2.1. The phased-array planner antenna would be 100 rows with 100 panels in each row. The antenna size is different on the complete array, and the antenna would be about 500 x 500 m while it varies from center to edge, respectively 1.6 × 1.6 m to 20 × 20 m. phase synchronized with each other. The proposed frequency was from 1.498 GHz to 2.997 GHz to transmit 99.9 % generated microwave energy to 57.94 km^2 earth-based rectenna using 8 % overall efficiency. The rectenna is elevated slightly above the ground so the research facility and required operation building can be built under the structure [7].

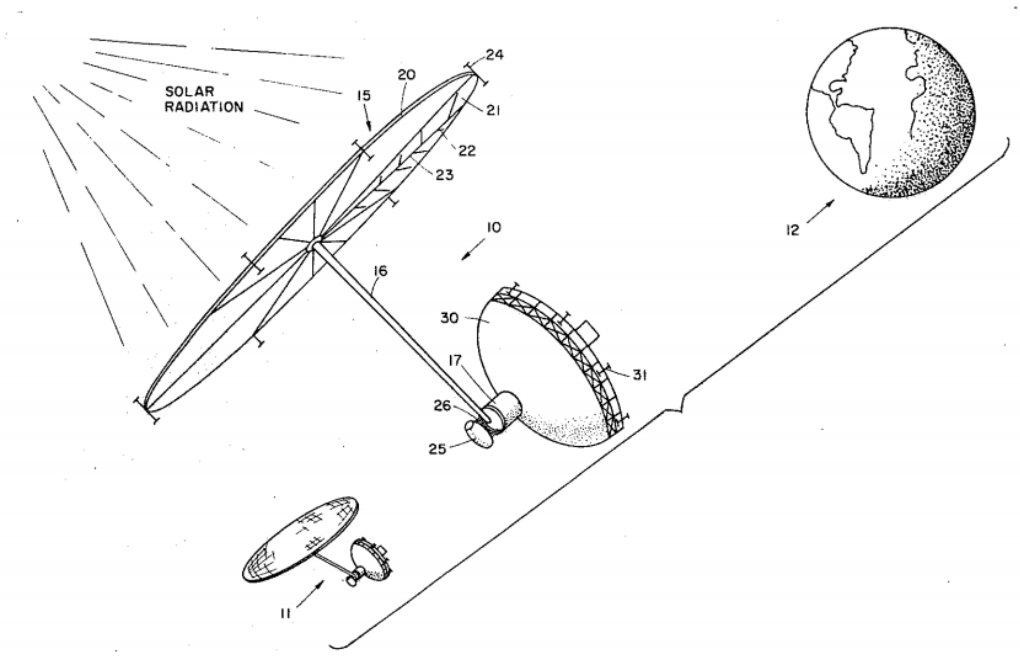


Figure 2.1 Peter Glaser's conceptual SSPS design, 1973

2.1.1.2. NASA DoE Reference Model, 1978

After Peter Glasser's proposal, the US DoE and NASA developed a conceptual design of SSPS for the year 2000 in 1978, shown in Figure 2.2. The design consists of a planar solar array with a 10.4 km × 5.2 km blanket area of 52.34 km^2 and planeform area of 54.08 km^2 on silicon base without concentrator and GaAlAs based CR system 10.6 km × 5.25 km with a blanket area of 26.52 km^2 and reflector area including planform area of 55.13 km^2 with GFRTP structure. A one km diameter phased array microwave transmission antenna with the frequency of 2.45 GHz - 6.72 GW was placed on another side. The transmission consists of a klystrons-based power amplifier with slotted waveguides elements, as shown in Figure 2.2 and detailed system information in Table 2.2.

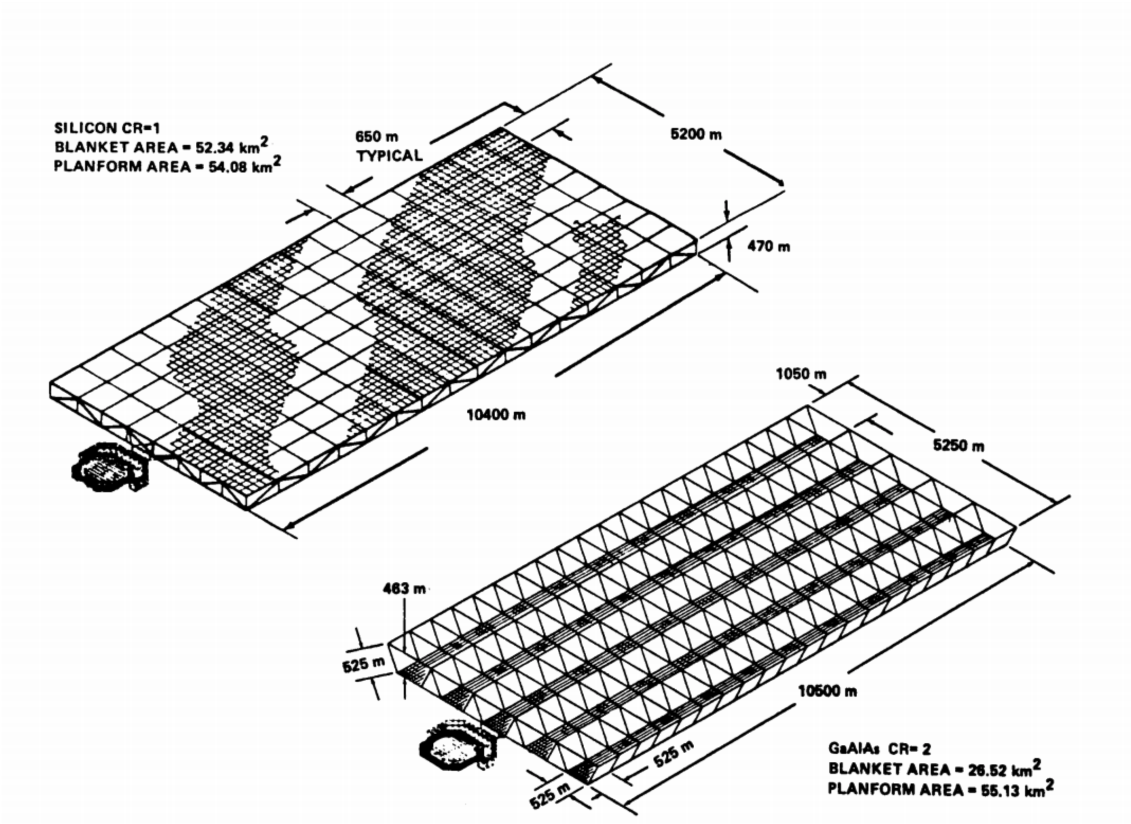


Figure 2.2 NASA DoE reference model, 1978

The reference system is designed for commercial output of 5 GW per year, and adding 2 SSPS per year will provide a 300 GW total capacity system with 2030 and the technology availability based on 1990. The total overall efficiency for the reference system is 7 %, with an EOL of 30 Years. The power density on the ground was set to a maximum of 23 mW/cm^2 at the center and one mW/cm^2 at the edge, which is thought not to cause nonlinear interactions in the ionosphere. The area of the power receiving rectenna is 10 km in diameter in the equator with an area of 78.5 km^2 [27].

Table 2.2. NASA DoE reference model constraints [27]

SSPS Design

PV – solar radiation conversion	Silicon CR=1	GaAlAs CR=2
SSPS generation capability	5 GW	5 GW
PV efficiency at 28°C	17.3 %	20 %
Dimension (km)	10.4×5.2 Blanket area = 52.34 km^2 Planform area = 54.08 km^2	10.6×5.25 Blanket area = 26.52 km^2 Reflector area = 53.04 km^2 Planform area = 55.13 km^2
SSPS mass (kg)	51×10^6	34×10^6
Structural material	GFRTP	
Orbit	GEO	
MPT	Yes	
No of antenna	1	
DC-RF	Klystron	
Frequency	2.45 GHz	
MPT conversation efficiency	63%	
Rectenna dimension (km)	10×13	
Rectenna power density (MW/cm^2)	Center	23
	Edge	1
Transportation system		
Earth-LEO	Cargo (payload)	Vertical take-off, winged 2-stage (424,000 Kg)
	Personnel (Number)	Modified shuttle (75)
LEO-GEO	Cargo	Dedicated elect. OTV
	Personnel (Number)	2-stage LOX/LN 2 (75)

2.1.1.3. Fresh Look Study 1997-1999

The NASA project group studied a new approach to investigate and create a new SSPS system design. The report presents two unique designs with the 1990s technology; the Sun Tower and the Solar Disc.

2.1.1.3.1. Sun Tower

Sun Tower is an innovative design to lower the development and lifecycle cost. The design was first deployed in LEO and then transferred using elliptical Earth orbit. The design is capable of transmitting 100 - 400 MW using a single SSPS module. The Satellite constellation of MPT is characterized by its GHz frequency-based system, medium size, gravity gradient stabilizer, sunflower concept. The leaves are PV cells, and the Earth point face is the transmitter with a diameter of 26 m. A single transmitter cell is a hexagonal structure with a 5 cm diameter.

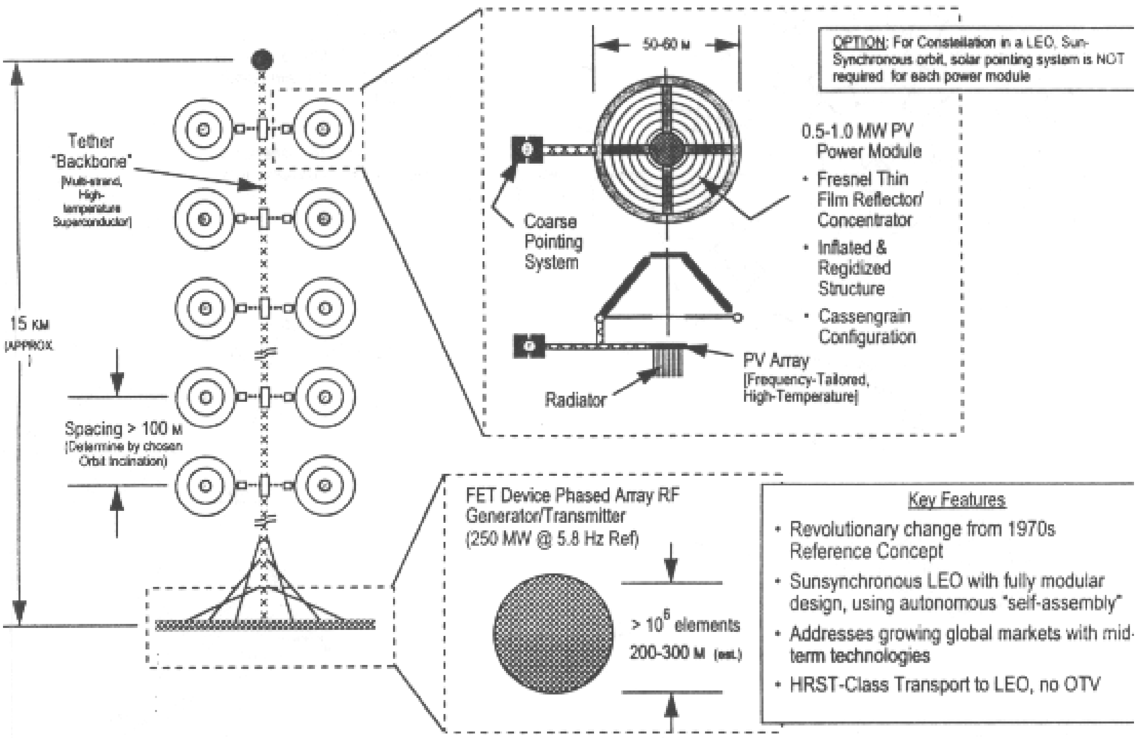


Figure 2.3 Sun Tower design by NASA

At 1000 km, SSO can transmit 200 MW with beam steering of 60° shown in Figure 2.3 and CAD in Figure 2.4, which is received by a ground receiving rectenna that is 4 km in diameter. The constellation design can transmit 3.5 - 4 GW with a 40-year lifetime, and will have18-24 launches every ten years to stabilize the satellite power system in orbit. The commercial cost of the first power unit is 6-8 bn USD, and the total constellation cost is 35 - 40 bn USD, with the average power cost versus price is 0.04 USD /kW-hr to 0.21 USD /kW-hr, respectively [30].

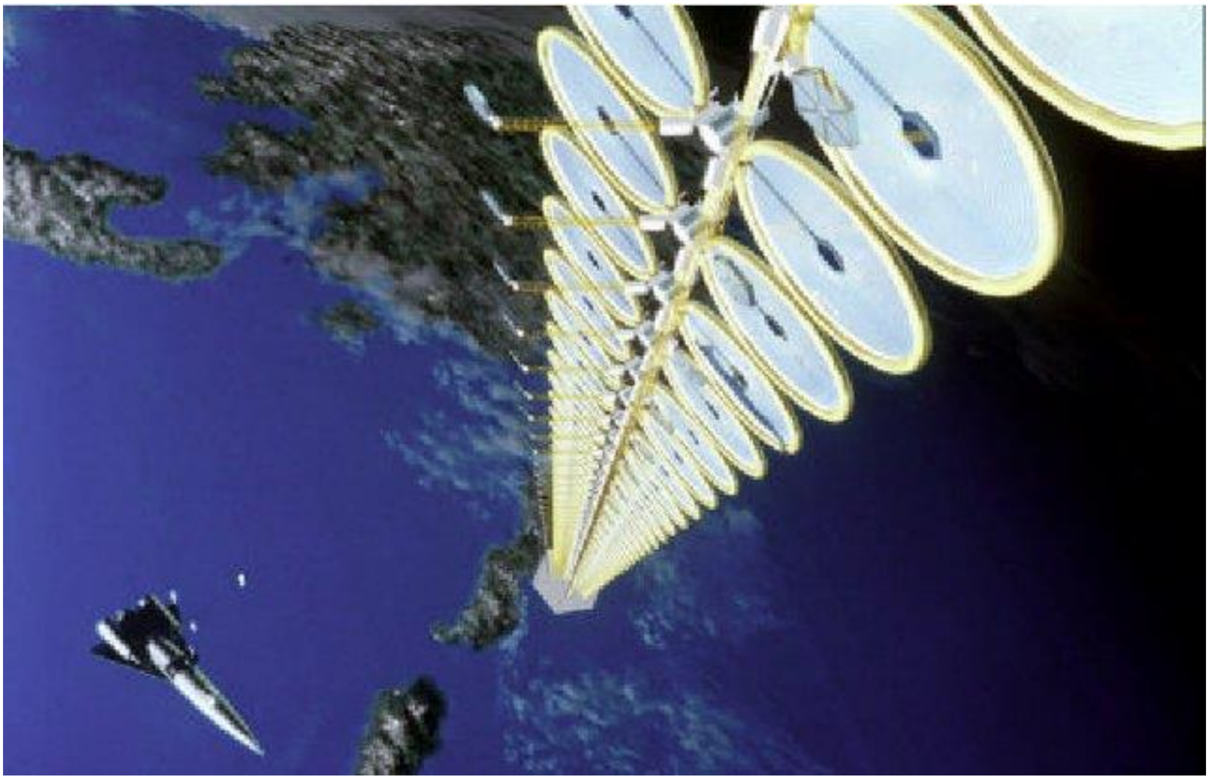


Figure 2.4 Conceptual design of Sun Tower

2.1.1.3.2. SolarDisc

This GEO-based SSPS system is a single-designed satellite to fulfill the energy demand that uses a frequency of 5.8 GHz for WPT to transmit 1 -10 GW from space to Earth. This disk design is 3-6 km in diameter, perpendicular to the sun site for maximum efficiency from Sun. The transmitter is 1 km diameter placed Earth pointing phased array. The beam steering capability is 10 degrees and the ground segment are approximately 5-6 km diameter in size, shown in Figure 2.5. [30]

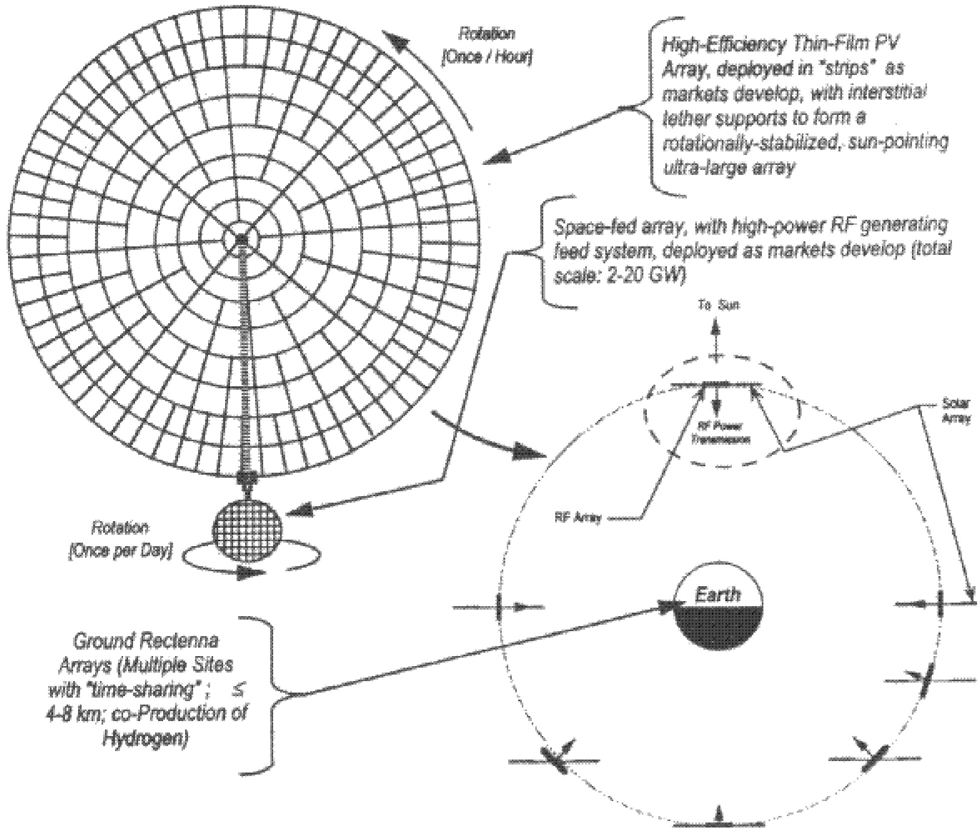


Figure 2.5 Conceptual design of GEO based SolarDisc

2.1.1.4. SERT SSPS Design 1999-2000

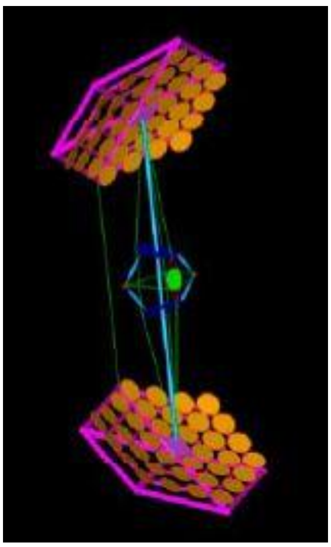


Figure 2.6 SERT SSPS design

The sunlight is collected in two "shells" and the central solar cell. The mast is in the average direction of the orbit. There are 24 mirror types with 2 × focusing and 36 mirror types with 4× focusing. The minimum weight is a 36-mirror type with 470 kg, which uses a magnetron and quantum dot solar cells. The diameter of the "seashell" is about 4 km, and the mast is 7.2 km. The initial weight in LEO is 31,500 tons, shown in the conceptual design in Figure 2.6 [31].

2.1.1.5. Solar Power Beaming Concept (2009)

The LLNL proposed as in the Figure 2.7, the Laser-based SSPS concept for 1 MW power transmission using a 795 nm diode-pumped laser. With 50 % electric to Laser conversion efficiency and 70 % light to light transmission efficiency, within a total packed volume of 2 m^2 × 4 m tall structure, this prototype would be easy to transport in space by Falcon Heavy. The total weight of the structure is 9125 kg. The system utilizes the reflector to concentrate solar power on foldable solar panels and a hydrogen generator, using a diode-pumped Laser to transmit energy from space to Earth [37].

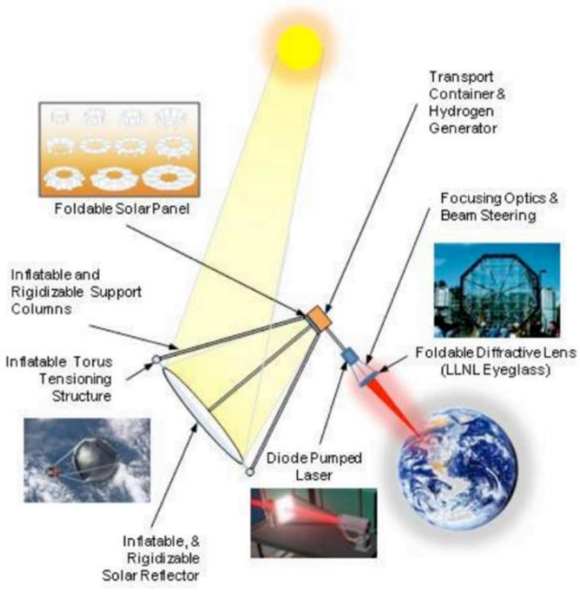


Figure 2.7 Solar Power Beaming Concept proposed by LLNL

2.1.1.6 SPS-ALPHA, 2011

Under the NIAC program, John Mankins designed SSPS which consists of a sizeable earth-oriented array, a sun-shielding reflector system that acts like a heliostat that follows the movement of the Sun and adjusts the orientation of the mirror, and a truss that connects them at GEO. The power generation unit is a fixed axisymmetric gravity tilt stable satellite, and the satellite does not incorporate 3-axis movements. The extensive array is a retro directive phased array with a coherent RF transmitter, as shown in Figure 2.8. The satellite has multiple joints and connectors, and every component is designed to be below 100 – 300 kg. The structure of SPS-Alpha is comparable to a HexBus, which has a diameter of 4m capable of wireless communication between systems and an average weight of 25 kg with more than 200,000 structures used for designing the system. A > 900,000 interconnects equal to 1 kg for connecting the system.

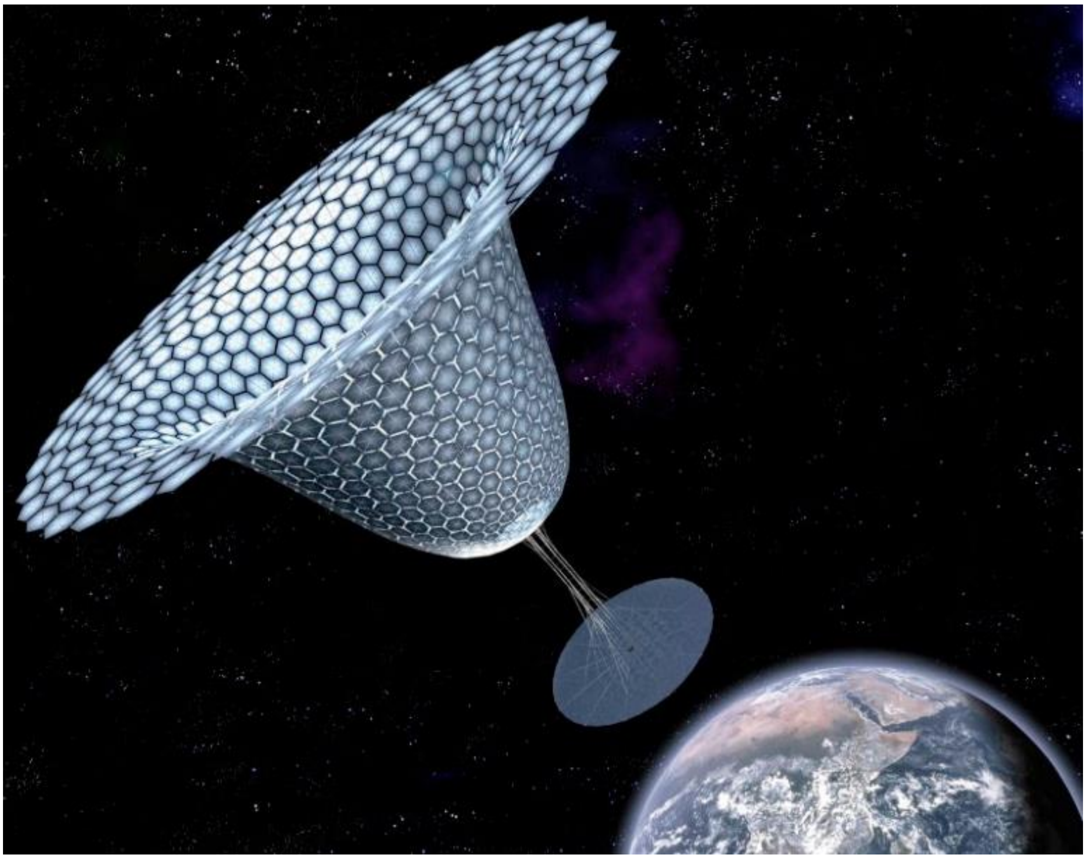


Figure 2.8 Conceptual design of SPS-ALPHA

HexFrame structural module is subsystem is deployable and expandable beam to provide the base for reflector and a connection system between reflector, power and transmitter system with 5,000 elements with 50 kg. The reflector subsystem is made from thin-film by Al on Kapton, with 4,000 – 5,000 elements weight around 75 - 100 kg. The WPT is combined with the HuxBus unit. HexBus is 200,000 – 300,000 elements with 50 kg each. The space assembly is constructed by rotating robotic arms with an actuation system with <5,000 elements with a weight of 10 kg. The propulsion, AOCS, and GNC system are between the weight of 50 – 200 kg, with each weight of 50 -500 kg depending on the propulsion mass [38]. The team investigated five different SPS-Alpha DRM as DRM-1, DRM-2, DRM-3, DRM-4, DRM-5. Table 2.3 shows all DRM modules with specification, cost, and efficiency models.

Table 2.3 Specification of SPS-ALPHA DRM system

Parameters	**DRM-1**	**DRM-2**	**DRM-3**	**DRM-4**	**DRM-5**
Orbit	LEO	LEO	GEO	GEO	GEO
Altitude	SSO 900 km	SSO 900 km	35,800 km	35,800 km	35,800 km
Objective	Technology flight demonstration	Moderate Power technology demonstration	Pilot plant	Full-scale plant	Large scale power plant
Total Platform Hardware Mass (kg)	12,108	16,768	232,610 - 1,045,252	11,795,271	34,813,882
Power (MW)		.200	2 - 18	500	2000
Launching					
Frequency (GHz)			2.45 GHz	2.45 GHz	2.45 GHz

Table 2.3 Specification of SPS-ALPHA DRM system (continue)

Parameters	DRM-1	DRM-2	DRM-3	DRM-4	DRM-5
WPT Efficiency DC-RF			70%	70%	80%
Cost to First Power (USD)			~ 4.5 Bn (~250 per W)	~12.2 B (~24 per W)	31 B (~16 per W)
Lifetime (Years)			10	>30	45>>30
LCOE (USD per kW-hr)			~ 3.26	~ 0.15	~ 0.09

2.1.2. Japan

2.1.2.1. Japan's SPS 2000 Project

Japanese space engineers and scientists started designing SSPS systems as early as the 1980s. In 1990, the Japanese government reformed its New Earth 2s objective to solve the world energy problem to respond to global climate change. The ministry started to investigate an alternative method for energy production with nuclear energy progress. They initiated a project group dedicated to SSPS systems with 13 sub-group members, 80 engineers, research scientists, and academicians from ISAS and Japanese universities. In 1994 a fully functional paper was published as SPS 2000 [45].

A study report conducted by RCAST of Tokyo University and Keio University (1996-1997) compares the current energy production using coal, oil, nuclear with SSPS to showcase the tackle down of global climate. According to the report that factors in SSPS to the total energy generation, the total CO^2 (grams relished per unit of one Wh) by the power plant will drop to 1/30, with individual values being coal- 1225, petroleum – 846, LNG- 631, nuclear – 22, SSPS – 17 [46]. Building the SPS 2000 will cut down the cost of power production and transmission from Japan, a global exporter of power transmission, as mentioned in a report by

RCAST of Tokyo University. By deploying rectenna over developing countries will slow down the production of global greenhouse gases through this shared technology increasing Japanese diplomacy and leadership worldwide [45].

A nuclear power plant has a 34 - 40 % thermal efficiency, while MPT-based SSPS has 80-90 % conversion efficiency at rectenna, and LPT-based SSPS has 40-60 %. The transmission of power does generate a heating effect, which translates to transmission losses [45]. The SPS 2000 pilot project is created for the output of 10 MW using 2.45 GHz from 1100 km at LEO shown in Figure 2.9. To supply 200 – 300 kW of electricity to small villages in developing countries located on the equator.

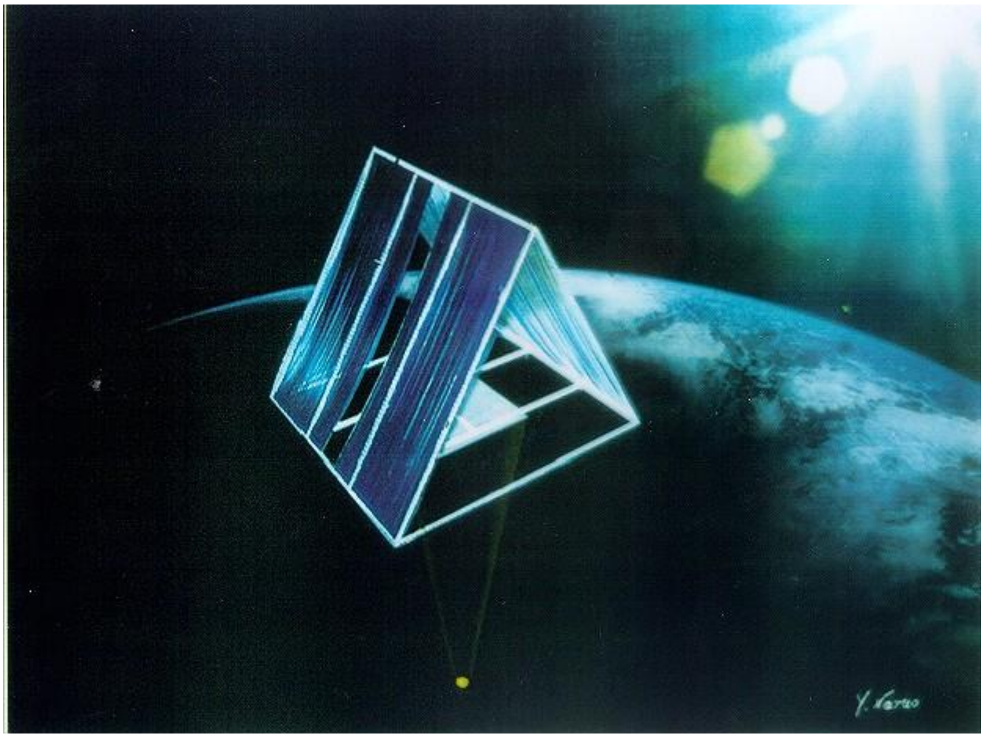


Figure 2.9 A conceptual design of SPS2000 by Japan

Researchers developed the SPS 2000 mainly and investigated across six countries independently (Papua New Guinea, Brazil, Ecuador, Tanzania, Maldives, and Indonesia) [45]. The full-scale model is for a 5 GW power transmission with a 50,000 tons SC. In the power generation, thin-film Si-PV cells with 15 % were used for transportability in space. The specific power is 1 kW/kg, and the bus voltage of 1 kV. The system comprises 110 subarrays composed of 12 solar cells on each with 180 A at 1 kV. A 45-array module with an approximate weight of each is 270 kg attached to each wing of prism structure. The structure size is 336 × 336 × 303 m with an antenna of 132 × 132 m in space and a ground-based rectenna of a diameter of 2 km [45, 47].

2.1.2.2. The Japanese Version of SPS (NEDO Sunshine Project), 1994

NEDO proposed its SSPS design in 1994 with the NASA reference model. The project uses MPT of 2.45 GHz frequency for 1 GW from a GEO-based Satellite. The satellite uses a two 3.2 × 2 km solar panels to generate electricity and transmit using a 1 km dipole antenna with a reflector weighing 26000 - 27000 tons, including 7400 tons. A conceptual design is shown in Figure 2.10. [29, 48].

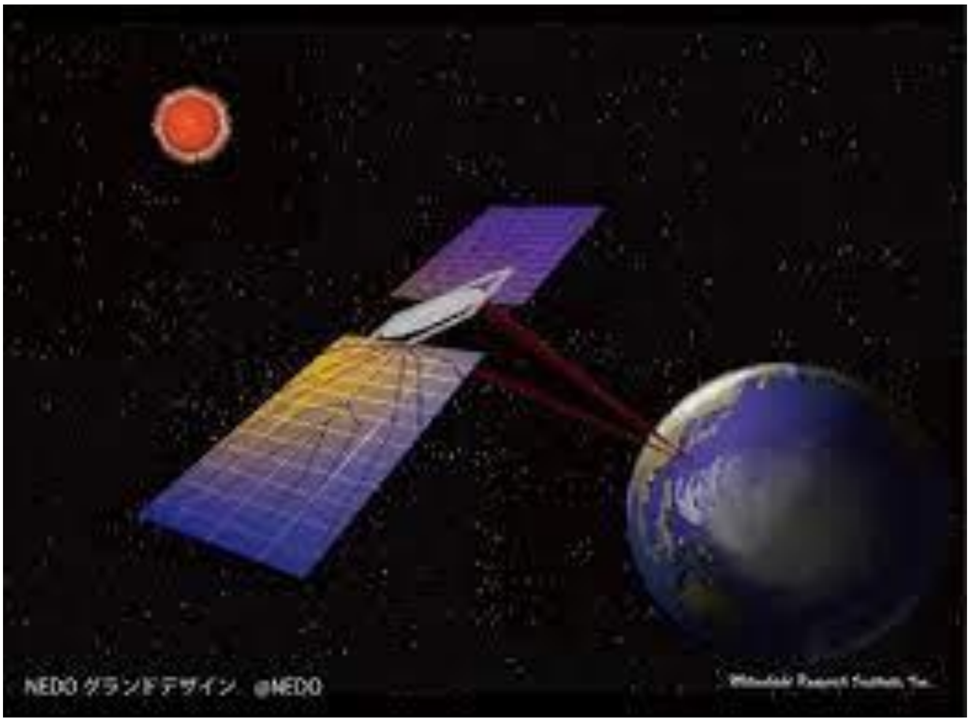


Figure 2.10. Conceptual design of NEDO Sunshine project

2.1.2.3. SSPS JAXA Model, 2004

The SSPS Review Committee makes the SSPS JAXA model of NASDA (later JAXA) during the period 1998 - 2008. A 2.5 km × 3.5 km primary mirror collects sunlight four times, and a solar cell with a diameter of 1.25 km generates electricity. It transmits 1 GW at the frequency of 5.8 GHz from a phased array with a diameter of 1.8 km, shown in Figure 2.11. The power receiving site, rectenna with a diameter of 2.74 km. The power generation unit and the power transmission unit are separated from securing a heat exhaust surface. The primary mirror is characterized by flying in formation independently of the power transmission department [33].

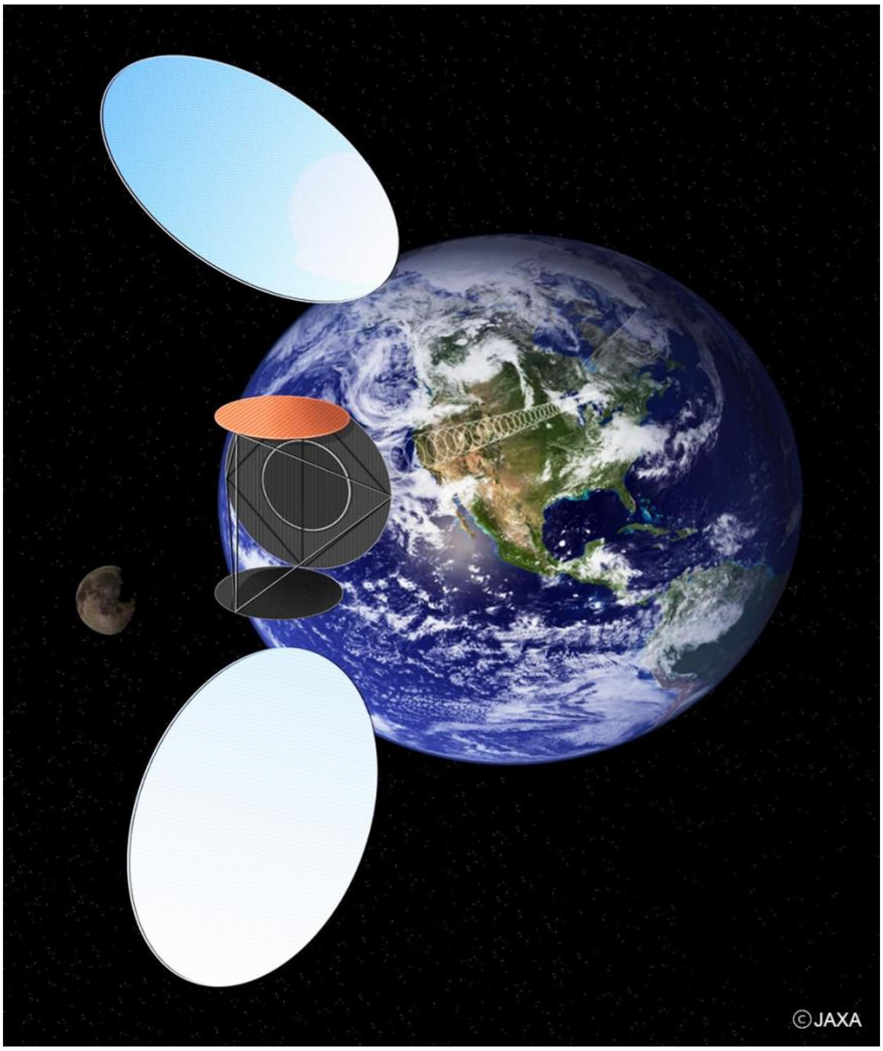


Figure 2.11 SSPS JAXA model

2.1.2.4. SSPS USEF Model, 2006

The Japan Space Systems Research and Development Organization proposed a new SSPS model created by the expert committee of 2001-03 as part of the research on SSPS conducted under the commission of the Ministry of Economy, Trade, and Industry. A gravity inclination posture stabilization method connects a substantially flat panel with integrated power transmission and power transmission and a distant bus part above it with a tether [49]. In watts class power transmission, the power transmission panel measures 2.6 km × 2.4 km, and the upper surface consists of solar cells, and the lower surface consists of phased array antennas and solar cells. It is hung from the bus section by multiple tethers of about 10 km and consists of sub-panels with a thickness of 10 cm and 100 m square shown in Figure 2.12. The diameter of the ground rectenna is designated as 3.5 km [35].

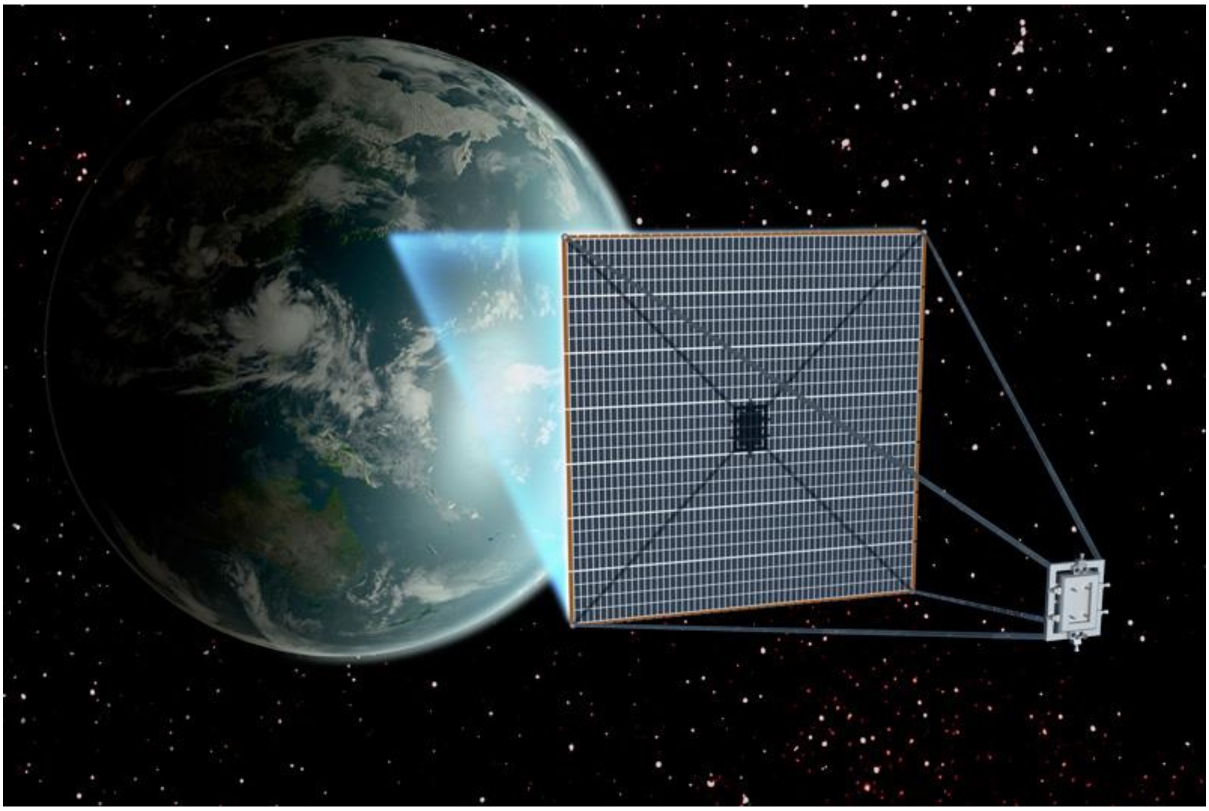


Figure 2.12 SSPS USEF model

2.1.2.5. JAXA L-SSPS Model, 2006

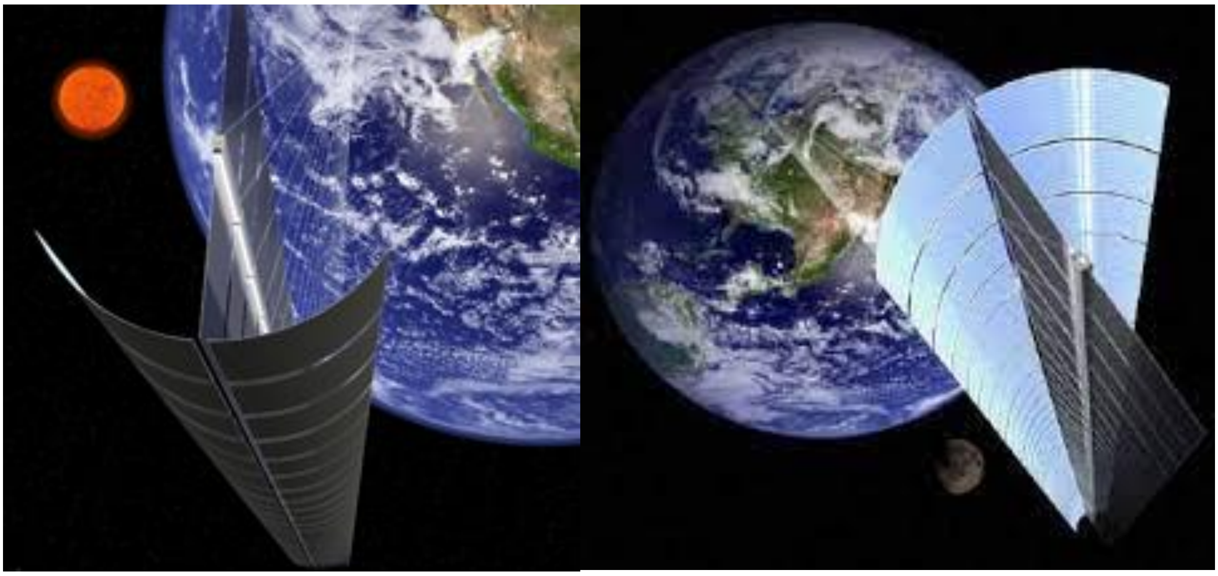


Figure 2.13 JAXA L-SSPS model

This model has a laser output of 1GW. It connects 100 lasers vertically with 10 MW output lasers with a width of 400 m, a depth of 300 m, and a length of 120 m by Nd-YAG & Cr-YAG laser [36] shown in Figure 2.13.

2.1.3. European Union, Sail Tower SPS, 2001

ESA proposed an SSPS concept like sun tower with new technology as 60 pairs of "sails" of 150 m^2 are solar cells with an output of 3,700 kW, totaling 450 MW. It is a geostationary satellite that uses gravity tilt stability, shown in Figure 2.14. It generates microwaves with 400,000 magnetrons for microwave ovens with a frequency of 2.45 GHz and an output of about 1 kW, transmits electricity from an antenna with a diameter of 1 km, and receives energy with a rectenna with a diameter of 10 km [32].

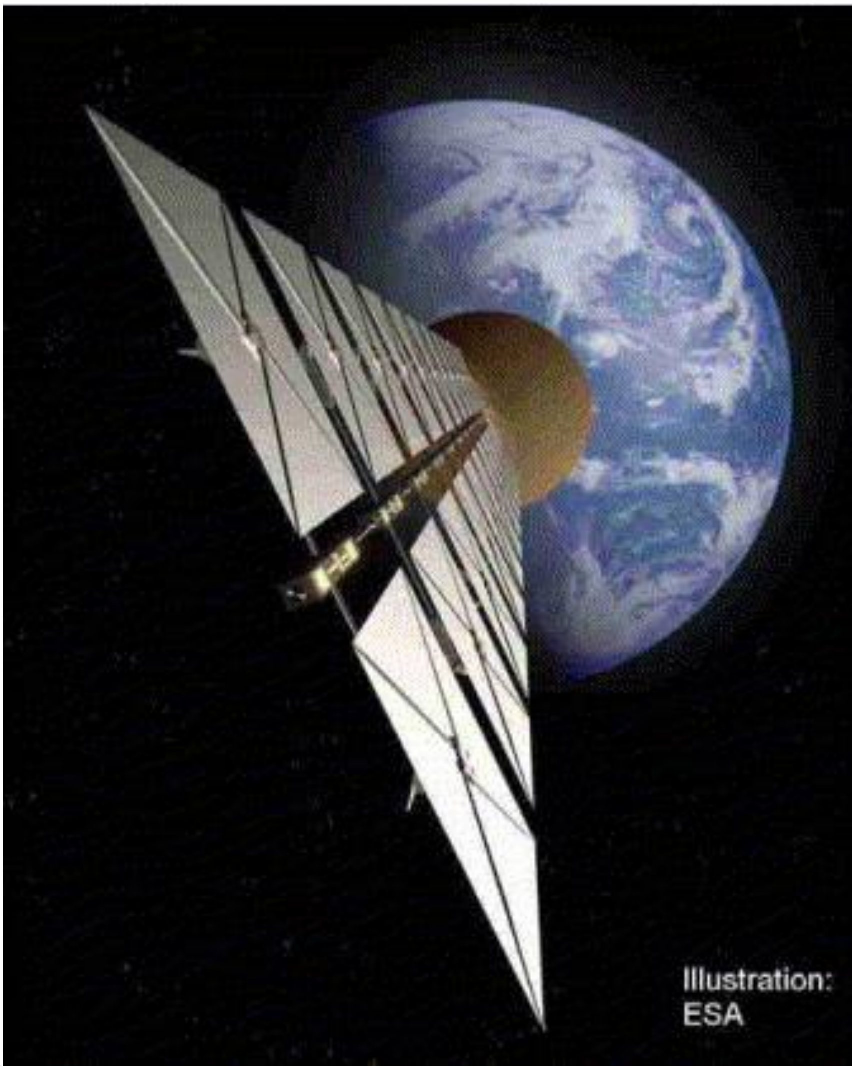


Figure 2.14 Sail Tower SPS

2.1.4. China

2.1.4.1. Multi-Rotary Joints SSP (2015)

China Academy of Space Technology proposed an MPT-based SSPS in 2015 to deliver 1.2 GW of power from GEO based system. The architecture consists of multiple PV in two directions to transmit electricity with a 5.8 GHz frequency using a microstrip antenna. The thin-fil, GaAs-based PV cells with 40 % efficiency will be used for power generation, and the total power output will be 2.4 GW. The MPT system efficiency is estimated at 54 % with a 1 km diameter antenna, consisting of 128,000 antenna module elements capable of transmitting

12.5 kW per module. The ground base receiving system is approximately 5 km in diameter, as shown in Figure 2.15. The total efficiency of the system is 13 % with 30 years lifetime. The 1 N thrust will maintain the AOCS for orbit keeping [39].

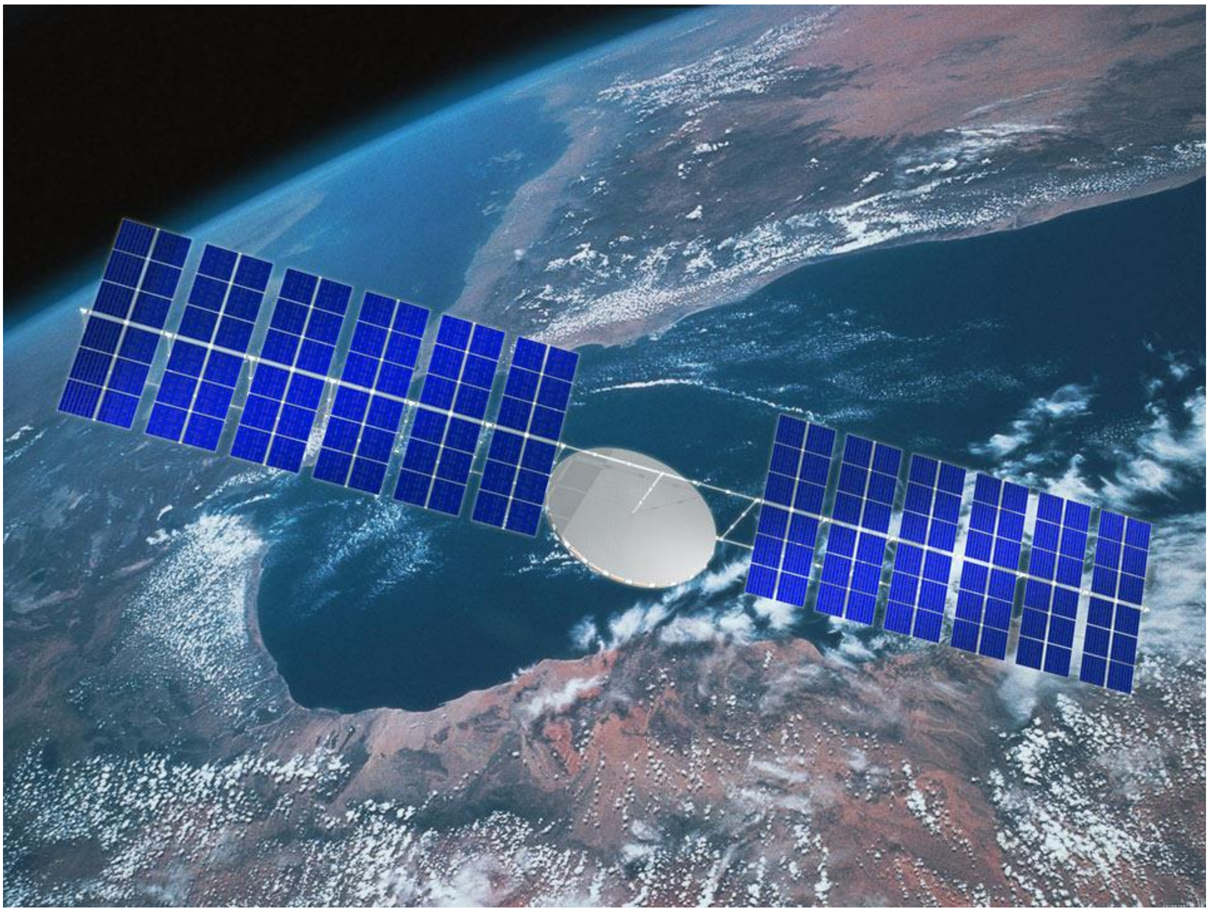


Figure 2.15 Proposed design by CAST: Multi-Rotary Joints SSP

2.1.4.2. SSPS OMEGA [41]

The Xidian University of China proposed a Space Solar Power Station via Orb-shape Membrane Energy Gathering Array for a 2 GW, 5.8 GHz MPT-based spherical solar power collector SSPS design. The PV cells are arranged in a hyperboloid structure, and the reflector can adjust to concentrate the sunlight inside the sphere for a high concentration of power. A one km diameter transmitting antenna is used for power transmission from GEO to Earth-based receiving rectenna, as shown in Figure 2.16.

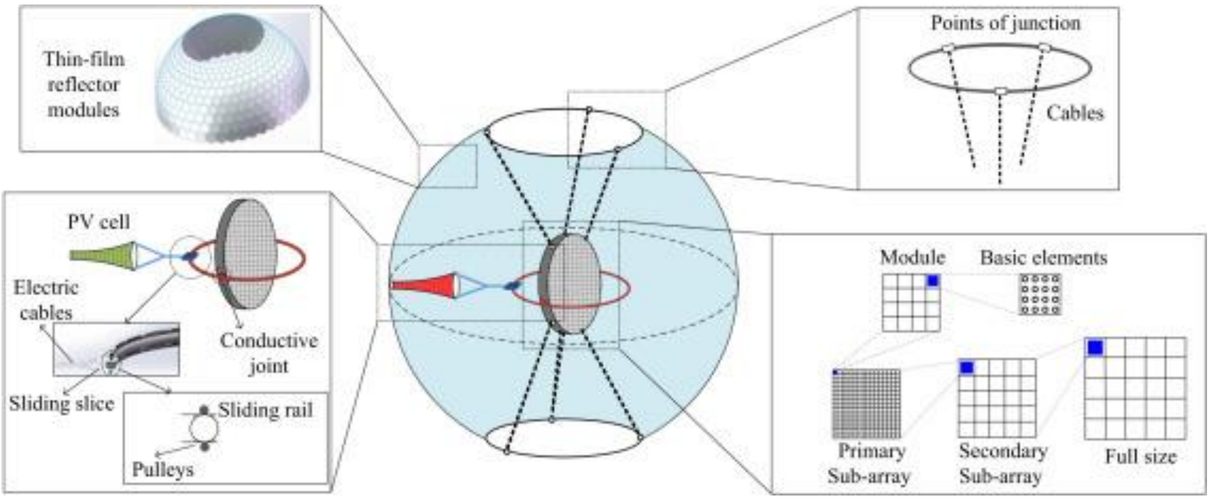


Figure 2.16 A conceptual Design of SSPS OMEGA

2.1.5. United Kingdom, CASSIOPeiA, 2017

This is a design by Ian Cash, with MPT and three different variants with 688, 974, 1310 MW respectively weight of 866, 1020, 1280 tons. It uses an SSO elliptical orbit at an inclination of 116.6 °, the design in Figure 2.17.

Figure 2.17 CASSIOPeiA design

The concept can manage cosine losses of solar energy collection, and the concept features no moving parts. The PV cells are located on the helical-shaped structure and provide a continuous power system in orbit with continuous power towards Earth using an MPT-based system [42].

2.2. Wireless Power Transmission

WPT technology has undergone several enhancements in recent history. Nikola Tesla designed the Wardenclyffe Tower, also known as the Tesla Tower, one of the earliest experiments to prove potential WPT [12, 50]. In 1973, Dr. Peter Edward Glaser coined the idea of an SSPS capable of transmitting power via microwaves from a GEO to a ground microwave collector station on Earth [7, 17]. Microwave energy transmission can be conducted at all frequencies above 1 GHz. An optical lower band is preferred because of its efficiency to transmit through the atmosphere, rain, and a gray area. [51].

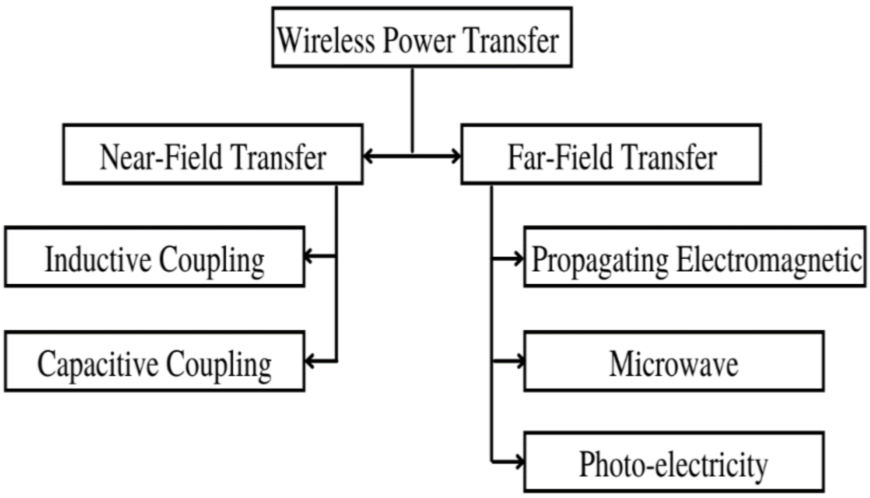


Figure 2.18 WPT method via distance base

The effective range of different methods varies from a few centimeters to a few meters, as shown in Table 2.4, which compares technology's range, efficiency, size, and cost to develop.

This WPT technology is currently used in mobile charging. In addition, the investigation and research on the far-field transmission, long-distance SSPS system, including MPT, LPT, electromagnetic waves, are underway as shown in Table 2.4.

Table 2.4 Comparison of different WPT techniques [54, 55]

Parameters	**Inductive Coupling**	**Microwave**	**Laser**
Distance	Few millimeters	Up to 100s km	Few meters can be increased by highly intensive beam
Transmitted Power	~ few W	~100s MW.	~100s MW.
Efficiency	Low	High	Medium
Penetration (clouds/ fog/ rain)		Very High	No
Aperture size (transmitter and receiving antenna)	Small	Large	Small
Cost	Economical	Expansive	Economical
Safety (Biological point of view)	Safe	Dangerous due to radiation but controllable	Dangerous

Most of the SSPS studies conducted by space agencies and private entities are based on MPT. It can transmit energy in cloudy or remote regions with higher transmission efficiency, an effect of extensive research in this area compared to LPT. In such a system, the PV cells generated power in DC and converted it into microwaves using magnetron and klystron with high beam efficiency.

The Laser of an LPT can be generated from the solid-state laser diode and solar pumping laser generator. LPT undoubtedly shows the highest prospects for long-distance, slight beam divergence with high-energy transmission [56]. The receiving and transmission antenna of an LPT system is comparatively smaller than the traditional MPT system. Therefore, the LPT system can deliver energy longer distances than any other WPTS [57-59].

The most straightforward and popular design among the SSPS community is the sandwich architecture, where one side is a power generation unit, in center electronics and conversion unit, and another side is a transmission unit. The Caltech Space Solar Power Project is one example [60]. The SSPS avionics and structure design define the size of the structure. The use of ultra-light material can reduce the weight and cost of the system [50]. Using advanced PV and CMOS integrated circuits provides a perfect DC to RF conversion and phased control for better transmission [61]. The SSPS structure should be designed with the sustaining force, torques, and vibration frequency below 0.1 Hz [62]. The transmission system is a significant component in SSPS design. Researchers are focusing on two main types of transmission systems, MPT and LPT, which will be extensively discussed throughout chapters 2 and 3.

2.2.1. Microwave Power Transmission

The MPT system is an RF-based system, using an ISM band and much research has been done to confirm its safety. Most MPT proposed systems are with 2.45 GHz - 5.8 GHz frequency. The structure of the transmission antenna and rectenna is also dependent on the frequency. The MPT system utilizing higher frequency gives advantages by increasing efficiency and reducing the size of the transmitting antenna. The transmission system combines several antennae with a controlled phase to achieve a narrow beamwidth. The Friis Equation is used to calculate the efficiency of WPT between transmission and rectenna [62]. From a system standpoint, the critical derived parameter for the satellite is the specific power, S. S is the RF power radiated at the satellite per kilogram of mass in orbit (W/kg). The larger S is, the less mass is required to obtain a fixed amount of radiated power. Our goal is to maximize S while minimizing weight and managing risk and cost. S is estimated using Equation (2.1). The link budget for the SSPS system is modeled using a modification of the Friis formula [63,64].

$$S = \frac{\eta_{PV}\, \eta_{DCRF}\, \eta_{Tx}}{m_{SV}} A_{pV}\ (AM0) \tag{2.1}$$

$$P_r = [\eta_{PV}\, \eta_{DCRF}\, \eta_{Tx}\, \eta_{dif}\, A_{PV}\ (AM0)] \left(\frac{f}{cd}\right)^2 A_{PV}\ A_r \tag{2.2}$$

The receiving power will be shown as power with the cost-effective system using Equation (2.2). Two more efficiencies are used to calculate the DC to RF and RF to DC and if the AC instrument or payload will utilize DC to AC conversation efficiency. Recently, Naval Research Laboratory demonstrated using Photovoltaic Radiofrequency Antenna Module Flight Experiment (PRAM-FX) experiment in orbit as a sandwich model space solar architecture. The experiment generated 8.4 W RF power with 37.1 % DC to RF conversion efficiency, and the total module efficacy was 8 % [54].

The electric energy coming from the PV array is managed correctly and rectified with a power management system and hardware section. Successful conversion of DC-to-RF will be achieved and transmitted to the daughter satellite using MPT for phase 1 of E-Orbit, the schematic diagram of the MPT transmission system. The power density received at the center of the rectenna can be defined by Brown and Eve's Equation (2.3).

$$\mathrm{P_d} = \frac{A_t\, P_t}{(2\, \lambda\, D)^2} \tag{2.3}$$

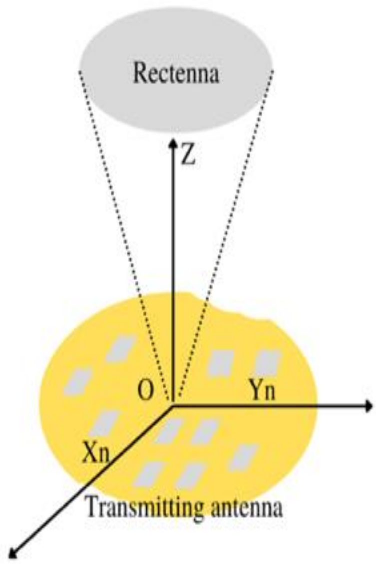


Figure 2.19 The geometry of transmitting antenna and rectenna (65)

P_d is power density at the center of the rectenna; P_t is the total radiated power from the transmitter (all power in kW). The transmitting antenna has a high beam efficiency, as shown in the geometry of both antennas in Figure 2.19, obtained from the BCE Equation (2.4). Therefore, the BTE can be calculated by Equation (2.5). On the other hand, the total MPT of a system can be calculated using both equations [65, 66].

$$BCE = \frac{P_{\Psi}}{P_{\Omega}} \tag{2.4}$$

$$BTE = \frac{P_{\Omega}}{P_{\Psi}} \tag{2.5}$$

In the above equations, P_{Ψ} is the power radiating over the angular region, and P_{Ω} is the total power transmitted over the visible region. The antenna can generate various types of errors due to vibration, high-temperature change, radiation. These errors can be minimized during launch and by maintaining the orbit.

2.2.2. Laser Power Transmission

Research for LPT is concentrated on fueling the interplanetary, habitat, and orbiter systems. In any WPT system, frequency denies the transmission's efficiency, size, and reception aperture. By this definition, mm-wave, which uses 35 GHz frequency, has higher transmission efficiency than an MPT system, which uses 2.45 and 5.8 GHz [19]. On the other hand, LPT has reduced efficiency, despite using an even higher frequency than mm-wave and MPT. Thereby, LPT is principally used to avoid the drawbacks of MPT, which consist of side lobes/spikes, difficulty controlling during failure, and the high mass and size requirements of transmission elements [18].

The LPT system is optical-based and has more remarkable ability to deliver long-distance and large amounts of power to a small aperture than the MPT system [59, 68, 69]. There are different types of lasers classified by design, component, and power. For example, CO_2 gas-dynamic (λ = 10.6 um), HF/DF Laser (λ = 2.41 – 3.38 um) gives 2 MW output using Middle Infrared Advanced Chemical Laser (MIRACL using Deuterium Fluoride). Chemical Oxygen Iodine Laser (Coil: λ = 1.315 um) is a type of chemical gas dynamic laser, also called Solid-state laser (Nd:YAG, Ti:sapphire, Er:Fiber (erbium-doped optical fiber)). Laser diode (HAMAMATSU: 1200 W CW) has lower quality compared to solid-state Laser. However, the system is with high heat load and lower efficiency compared to the laser diode. [68]

In 2002, EADS-SPACE Transportation (EADS-ST) demonstrated a 532 nm wavelength with an output power of 5 W Nd:YAG laser to a rover within 280 m [56], JAXA demonstrate LPT using a 200m high tower to transmit a 500 W with an accuracy of 1 rad. NASA, Russia, China, Lockheed Martine, and LaserMotive, Inc. conducted a series of experiments for long-distance transmission to consider high output efficiency. The available laser technology within the range of 780 and 1100 nm is the best choice for long-distance with sufficient power [70]. The diode-pumped solid-state Laser as fiber and disk laser have high advantages with high power and beam quality [71-73]. Moreover, compared to the diode-pumped solid-state laser, the laser diode is less expensive, has higher efficiency, and is more suitably compact for short distances because it produces less bright beams. Equation (2.6) shows the equation used to select available Laser for practical purposes with a high bright enough beam to transmit power for long-range

$$\phi = \frac{R_{source}\, A_{source}\, n_{trans}}{d^2} \tag{2.6}$$

Laser communication is already extensively investigated, researched, and currently in use for inter-satellite linking. The low latency of Lasers allows the system a continuous fast medium for transmissions. Likewise, LPT reduces the size of any transmission and reception system. The most efficient DC-to-laser converters are solid-state laser diodes commercially employed in fiber optic and free-space laser communication, as demonstrated in the following examples [74, 75].

- F. Steinsiek and his team at space Power infrastructure of EADS-ST demonstrated a laser operated rover by using Nd: YAG laser to pinpoint pilot connection for continuous power supply of a few W within 280 m, as shown in Figure 2.20.

Figure 2.20 EADC-ST demonstration of Laser power rover (source: EADS-ST)

- JAXA-based SSPS team demonstrated with a series of experiments in 2012 - 2013 the high accuracy beaming controlled by atmospheric disturbance performed at 500 m using the horizontal LPT test with accuracy of 1 μrad as in Figure 2.21. In another test, JAXA performs a 200 m high tower to mimic the space and Earth base station as the L-SSPS. Furthermore, they are expected to achieve a main laser output power of 500 W with the target of an accuracy of 1 μrad.

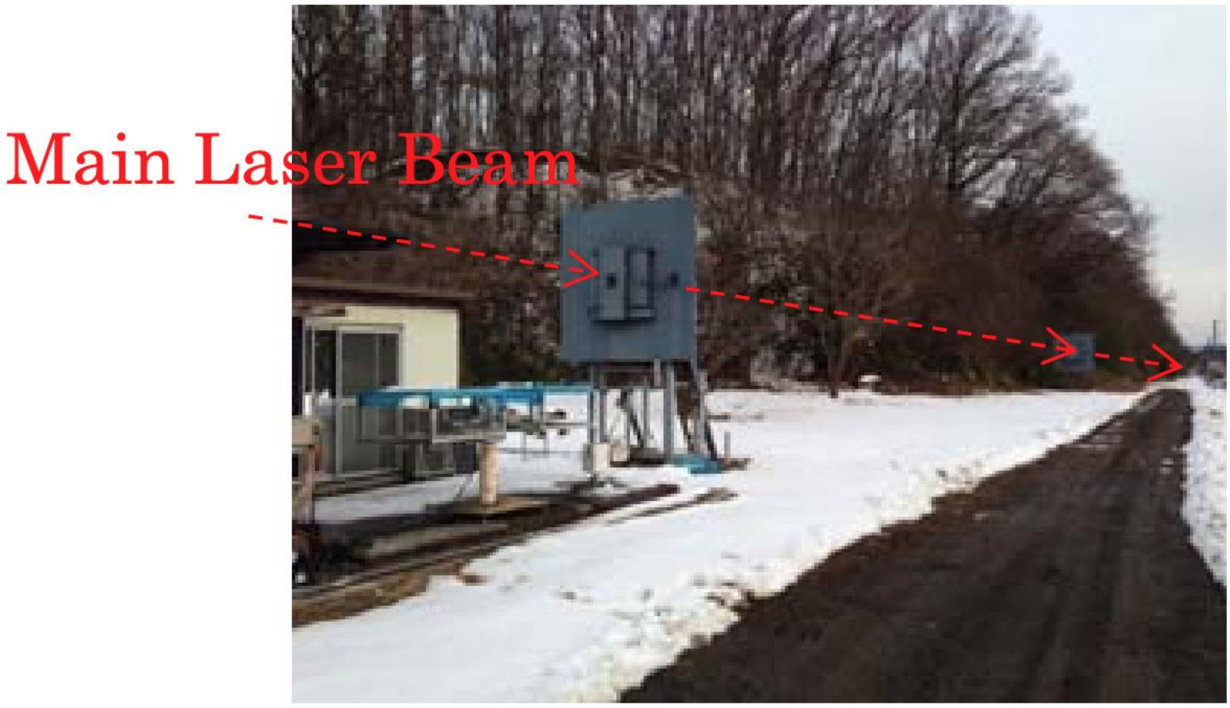


Figure 2.21 A 500 m horizontal laser transmission test at JAXA Kakuda Space Center (source JAXA)

- The United States Naval research laboratory demonstrated a free-space power beaming system using two 13-foot-high towers, one being a 2-kW laser transmitter, as in Figure 2.22, and the other, a receiver of specially designed photovoltaics. The laser beam of 400 W of power can travel across 325 m [59,76,77].

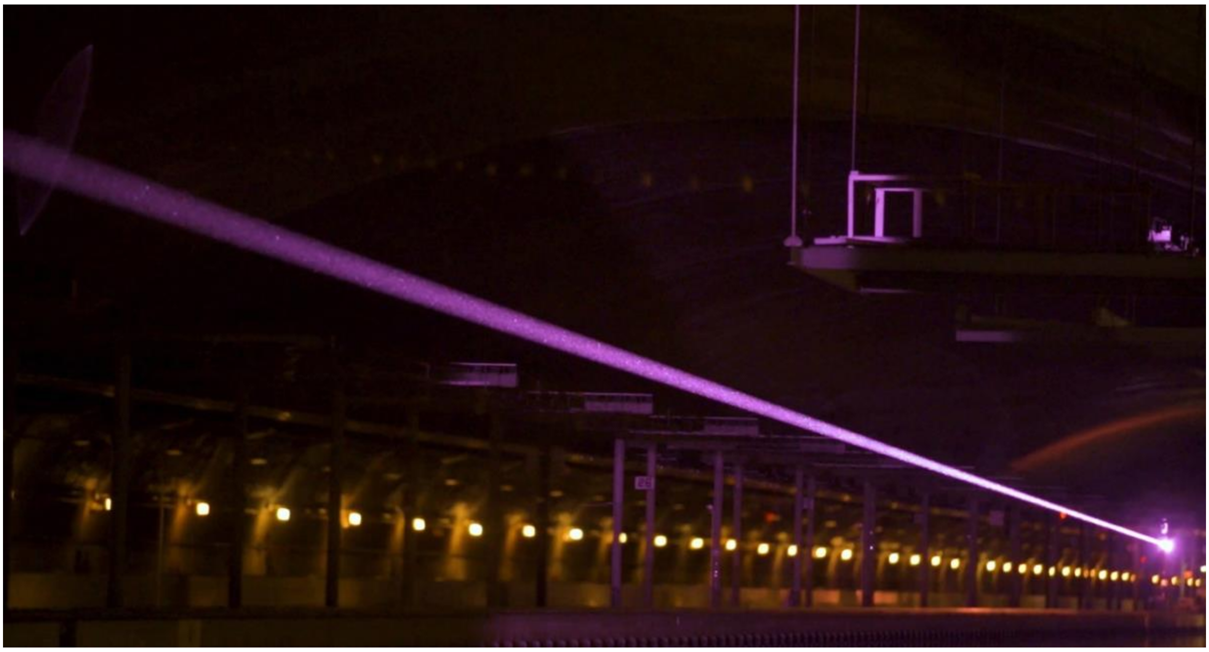


Figure 2.22 2 kW Laser Power Transmission demonstration by USNRL

For LPT, laser selection is significant as it should be small and low in weight while allowing a high-power output with low dispersion of heat and temperature concerning low mass and small size and with a high-quality laser beam to achieve at small receiving surfaces and control of phase [78]. While using a Laser, safety is prioritized. According to the International Electrotechnical Commission, laser types can be classified into seven distinct classes, including the subcategories from class 1 to 4. The class four Laser is deemed hazardous, with a high risk of eye damage. However, in a highly controlled system, it can transmit hundreds of W from a single pointer. The Sandia National Laboratories USA researcher analyzed the LPT system to acquire an output power of 370 W, and power at the load is equal to 35.06 W/cm^2 [79]. This indicates that a lower divergence beam is preferred for optimate laser transmission performance. The divergence of laser beams calculated by the Gaussian Beams is as shown in Equation (2.7).

The angle of divergence of the beam

$$\theta = 2\lambda \, \pi \, R \tag{2.7}$$

The final divergence (D) of the Laser beam can be calculated as in Equation (2.8).

$$D = 2 \, d \, tan\left(\frac{\theta}{2}\right) \tag{2.8}$$

Using the above calculation, the power density at any load surface can be calculated using Equation (2.9)

$$Power\ Density = \frac{P}{\left(\pi\left(\frac{d}{2}\right)^2\right)} \tag{2.9}$$

Here R is beam radius, is a wavelength, d is the beam diameter of Laser [79]. The selection of the Laser is essential with the output power and the wavelength because there are different types of lasers using different wavelengths, as shown in Figure 2.23. The laser selection for effective transmission uses 0.8 um 1.64 um lasers.

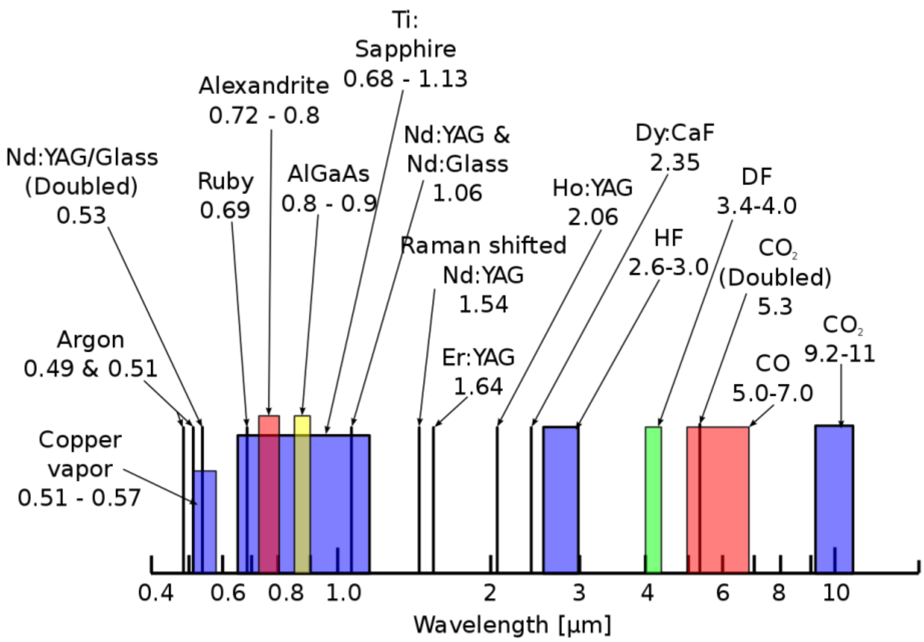


Figure 2.23 Spectral output of different Laser (Source: [79])

The most commonly used Laser being Nd: YAG laser, consisting of a diode with a plugin efficiency of up to 80 % at 795 - 850 nm. A combination of thousands of laser diodes allows more power transmission accuracy and efficiency in more extensive power transmission.

The Laser based SSPS Research Team of JAXA has been researching and investing in developing technology to perform high-efficiency LPT with the controlled direction of a laser beam with an accuracy of 1 μrad (5.7×10^{-5} °) to limit the divergence to a few centimeters when transmitting for several hundreds of km. The high-power laser beam sends electric power to the daughter satellite. Commercially, LD offers a high efficiency of about 40 % ~ 60 % with around 1 kW laser power. As the power will be a few 100 W, the dissipated heat is high, due to which the cooling system will be more complicated. Their collection is achieved with PV cells' help on daughter satellites [81].

2.3. The E-Orbit Reference Mission Designing

The previously proposed SSPS design has not yet been developed. However, to analyze technical infrastructure development, evaluate environmental effects, a societal concern, and performance assessment-based guidelines were established. Using these constraints, the thesis is focused on designing and evaluating the system analysis with following reference system with the current technology is discussed throughout the thesis in Part II. During concept study, a range of technological alternatives were identified. A combination of power generation and transmission system evaluated would have been beneficial for minimizing the size, mass and cost of the satellites. With this stand, the primary and secondary objectives of E-Sat and E-orbit are,

Primary Objective:

To generate, convert and transmit electricity in space using laser-based or microwave system with the concept of SSPS by developing small satellites and creating a constellation of the E-Sat to create an E-Orbit in space.

Secondary Objective:

To demonstrate the capability of high-power generation in a compact design using a concentrator and bifacial PV solar array. The utilization of an LPT system for laser thrust and propulsion generation for space sustainability and power distribution for debris removal and orbital maneuvering.

Studying the architectural and technical requirements for the SSPS projects leads the study to investigate a scenario of E-Orbit with the following required millstone.

E-Sat: The previously presented SSPS designs are huge in structure with concentrator and non-concentrator to generate bus power, distributed power, or laser direct excitation system as briefly described in section 2.1. With this,

- The examination and structural optimization are required with desired orbital parameters for a single hub station or designing a constellation of multiple E-Sat systems.
- A compact bus and easy structural system to the rapid construction of E-Sat.
- A perfection of sun-tracking and AOCS system for maximum solar irradiance using current technology.
- Power distribution and allocation (inhouse E-Sat subsystem, E-Sat-to-satellite, and E-Sat to E-Sat).

- Clear roadmap to realization, demonstration prototype, and creating market.
- Public acceptance with the legal framework, space policy and energy policy.
- Easy to launch and integrated into space using economical space launch vehicle.

Power Generation: The selection of power generation unit is the essential factor for the high efficiency and sustainability of SSPS in orbit; it includes

- Selection of PV Solar cells with economical to develop the E-Sat.
- Solar light collection system (concentrator or non-concentrator system)
- The satellite power management with large power distribution, interruption, decentralization, and superconductivity
- Protection against the radiation

WPT: The SSPS design is defined on the basis of the method of the power transmission system; theory is discussed in section 2.2. The selection of WPT system is defined with the power generation and transmission integration as,

- Selection of desired operational frequency radio wave, lightwave, with spatial power density, beam shape, modulation, and frequency band.
- In the scenario of MPT
    - The selection of microwave oscillator (microwave, millimeter-wave generator, and electron tube)
    - Selection and designing of MPT beamforming, phased array, splitter/phaser, sidelobe suppression devices
    - Designing of RF circuit, antenna and rectenna system.

- In the scenario of LPT
    - The selection of LD, wavelength and material with laser oscillation.
    - Laser beam formation, transmission
    - Laser power receiving device
    - Beam orientated control system
- Propagation of radio waves and light waves in troposphere and ionosphere
- Selection of wavelength with neglecting and avoiding inference to the ecosystem, atmospheric, and operational communication system.

Economics and Business:

- Economic considerations and cost requirements for the development of SSPS to the end-users service,
- Feasibility and marketability evaluation for distribution and integration to the end side grid
- Sociology and environmental studies for a sustainable society, global warming prevention, green energy and fossil fuel alternative energy source.

With above mission designing constrain and milestone to achieve a sustainable PMS system using SSPS technology. The broad objective of E-Sat system designing is based on initial sets of numerical requirements for space power transmission missions. To create the mission with current trending technology, need functional, and operational requirements. A constraint to control the over-budget and environment scenario to utilize the system capacity efficiently with the scheduled implementations in Table 2.5.

Table 2.5 Reference system mission requirements

Sr. No.	Requirement	The factor which Typically Impacts the Requirement	E-Orbit
FUNCTIONAL			
1.	**Performance**	The primary objective	Transmission of power (min 10 kW)
		Primary power generation system	PV Solar cells (54 m^2 -112 m^2)
		Secondary power generation system	Thermoelectric Generator
		Payload	LPT /MPT subsystem
		Orbit	LEO
		Pointing	Multi 900 km
		Weight	250 kg
2.	**Coverage**	Orbit,	500 km radius
		Number of satellites	1- 1600
		Power	10 kW (minimum)
3.	**Responsiveness**	Communication architecture, processing delays, operation	Real-time data transceiving system`
4.	**Secondary Mission**	For mother-daughter experimental satellite	Debris removal, laser propulsion and thrust generation on daughter satellite
		For full constellation	Debris removal, laser propulsion and thrust generation, orbital maneuver
OPERATIONAL			
1.	**Duration**	Operations life, level of redundancy	Mission operational at least 20 years
2.	**Availability**	Level of redundancy, and power transmission	available for 24 × 7
3.	**Survivability**	Orbit, hardening, electronics	Natural environment only

Table 2.5 Reference system mission requirements (Continue)

Sr. No.	Requirement	The factor which Typically Impacts the Requirement	E-Orbit
4.	**Data Distribution**	Communication architecture	Real-time communication between satellite to satellite using Laser and ground stations by RF
5.	**Data Content, Form, and Format**	User needs, level, and place of processing, payload	Across energy orbit up to 10 kW of power
CONSTRAINTS			
1	**Efficiency**	PV conversion efficiency	30 % (min)
		DC to RF or	80 %
		DC to Laser	40 - 60 %
		Beam collection at rectenna	90 %
		RF – DC conversion at receiving site	80
		Laser to DC power conversion at receiving site, or PV system	59 %, 30 %
2.	**Pointing accuracy**	LD divergence	9 mrad
3.	**Interlinking**	The readiness of E-Sat at any point of time for power transmission	Minimum 3 satellites will always communicate for continuous power supply at any time
4.	**Cost**	Designing cost of single E-Sat	10 M USD
5.	**Technology**	Available technology	2020 years
6.	**Schedule**	Technical readiness, program size	Initial operating capability by 2024, the final operational E-Orbit by 2035
7.	**Regulation**	Law and policy	LPT and MPT regulation
8.	**Political**	Sponsor, government	On public demand

Table 2.5 Reference system mission requirements (Continue)

Sr. No.	Requirement	The factor which Typically Impacts the Requirement	E-Orbit
9.	**Environmental**	Orbit, Lifetime	Natural
10.	**Interfaces**	Level of the user and operator infrastructures	To the sponsor and distributed between satellite users who will use E-Orbit
11.	**Development Constraints**	Sponsoring organization	Launching by the international collaboration of space agencies to creating a unified E-Orbit.
12.	**Data Handling Substation**	The command and control for preciously operation	2 Earth station

2.4. Space Law

The space law policies are formulated and proposed by COPUOS in the 1960s. The treaty discusses the peaceful, responsible use of outer space, including the Moon and celestials' bodies. Currently, 111 countries adhere to the space law, and 23 signed treaties but have not provided the ratification. The COPUOS has formulated the main five multilateral treaties and resolutions are as below.

- 1967: Principles Governing the Activities of States in the Exploration and Use of Outer Space, including the Moon and Other Celestial Bodies.
- 1968: Agreement on the Rescue of Astronauts, the Return of Astronauts, and the Return of Objects Launched into Outer Space.
- 1972: Convention on International Liability for Damage Caused by Space Objects.
- 1975: Convention on Registration of Objects Launched into Outer Space.

- 1979: Agreement Governing the Activities of States on the Moon and Other Celestial Bodies

The five main resolutions are:

- The 1963 Declaration of Legal Principles Governing the Activities of States in the Exploration and Use of Outer Space (resolution 1962 (XVIII))
- The 1982 Principles Governing the Use by States of Artificial Earth Satellites for International Direct Television Broadcasting (resolution 37/92)
- The 1986 Principles Relating to Remote Sensing of the Earth from Outer Space (resolution 41/65);
- The 1992 Principles Relevant to the Use of Nuclear Power Sources in Outer Space (resolution 47/68);
- The 1996 Declaration on International Cooperation in the Exploration and Use of Outer Space for the Benefit and in the Interest of All States, Taking into Particular Account the Needs of Developing Countries (resolution 51/122).

2.4.1. International Collaboration for Feasibility of Space Solar Power

Because of the emergence of SSPS technological progress and the associated business enterprise, both components mainly depend on one another. SSSP must at minimum be in the areas of economic feasibility to encourage commercial space players to invest, exhibiting market to increase the number of end-users with controllable and profitable costs. SSSP operates as an accessible renewable energy source; the cost of placing the system is to use it.

In particular, the starting cost drives the SSSP to the sky. This is an issue that all space activities face.

Public-private collaboration between participants in both industries can distribute some outrageous cost tags and build the essential infrastructure. Their risk and responsibility are shared. The idea of space security is one of the significant challenges in the feasibility of SSSP. Space security has been described as the protection of human and/or spacecraft through all phases of a space mission, whether it is 'man' or 'unknown.' The space security concept includes: (a) all elements ranging from pre-start, launch, orbital or sub-orbital, re-entry and landing, (b) to safeguard ground and flight infrastructures and their surroundings in the vicinity of the launch sites and (c) to secure space-based services, the structure, and the autonomous satellites.

SSPS's task is to strike a balance between 'constantly transferring energy to Earth at a level high enough to be efficient yet low enough to cause no harm.' At the early phases of SSPS, many believed that security threats flow considerably above the more usual radio use, from the energy level sent to the Earth and the atmospheric dangers posed by the large number of heavy-lifting lifts required for the required equipping in space. Indeed, SSPS shares the safety hazards associated with successful launches that failed in all other space activities.

Similarly, SSPS can produce space waste, particularly given the enormous areas of easier-to-hit designs with more moving components to damage. Another danger of safety shared by all space activities is a collision. However, the primary security problem highlighted in the framework of SSPS is Earth's WPT and its potential for detrimental effects on biota and the environment. The most important aspects are the shape and intensity of the energy exposed to life and its environment, particularly the climate. The WPT problem may be classified as environmental consequences, including RFIs and biosphere hazards, and path of beam that could create dangerous situations. While much of the literature talks on microwave transmission, the laser has the same restrictions to provide acceptable levels of safety. The main difference between laser and microwave is its armament potential.

Initial worries about the level of power returned to Earth were alleviated by the increasing usage of microwave power in communications, medical, radar, and industrial applications as well as the common usage of microwave ovens in households over decades. The beam falls below the harmful threshold even in the most intense area and under continuous exposure to the laser beam. Non-ionizing radio frequency power is conceivable; thermal effects should be modest but only approximately 1/4 the intensity of the transmitted beam. The mid-range WPT bioeffects (5.8 GHz) investigations of NASA showed no significant risk of exposure. There are few interactions between the atmosphere and the power beam, and the possibility of harm is unknown.

However, there are still specific concerns. Both microwaves and lasers, radiating beyond the visual region (e.g., IR lasers), are undetectable to the naked human eye. They are insidiously exposed to persons working inside and living around terrain stations, airplanes, and other aircraft such as ultra-lights and balloons and, presumably, to bird species. The "startle factor," or momentary blindness or blinding of airline pilots, is problematic for visible light-lasers. Damage to birds and mitigation techniques may be incorporated in the environmental safety requirements in the broader environmental context. Harmonizing environmental security requirements among nations might effectively set an acceptable minimum threshold for WPT emissions, as found in other parts of the spatial activity and electromagnetic energy systems.

Frequency and microwave safety are correlated. Higher frequencies make antenna dimensions and gains more attractive but at the sacrifice of the lowness of power. In addition, the higher the frequency, the denser and the stronger the beam, the more harmful. Consequently, the challenge of allocating frequencies and orbital slots for SSPS extends beyond political and legal matters. The most suitable frequency for the transmission of the microwave was initially 2.45GHz. However, given that it is now widely used, this is no longer a feasible choice. NASA and other organizations have proven 5.8GHz as a viable option, and this has been discussed as an ISM band for SSPS microwave transmission. As the environmental danger of WPT is radio interference, international telecommunications would be an essential factor.

2.4.2. Liability

As noted above, several SPS-related designs are expected to emerge, and each of these models will have its specific risks and liabilities. Some of them are briefly summarized as follows:

- Flaws in the technical design and its ability to produce power at projected levels and cost-efficiency. This might give rise to liability (including product liability) claims by those who funded the project;
- Flaws in rectenna design or problems with its location; e, g. reflected energy could harm aircraft, nearby homes or industries, or even satellites that need a low noise environment to operate. Rectenna could have an adverse ecological effect on fish or ocean life, or local flora or fauna if on land, etc.;
- Transmission lines to cities would have to get regulatory approval and, depending on the level of transmission power, could have an impact on housing or industry that are along the transmission pathway;
- Transmission via laser, millimeter-wave, or microwave and translation from electrical power to radiofrequency and back has several issues involving interference to other satellite systems, medical protective systems, and others. This

could result in operation from LEO (that is quite congested) and would be very difficult indeed;

- Malfunction of concentrators so that they focused destructive power and light on other satellites or even high-altitude aircraft would be a concern and would call for a "failsafe design" in this regard;
- If the SPS unit is designed for upgrade or for retrofit of PV cells or to take sections apart, this would have implications in terms of investor claims, end of life disposal, and others.; and
- Suppose there was a pointing accuracy problem in the power transmission unit from the SPS back to Earth, especially if the pointing system malfunctioned and started beaming power or radio frequency emissions at military, communications, remote sensing, navigation and time, weather satellites, and many more so that they could be disabled. In that case, this could trigger a multi-billion USD liability.

2.5. Adverse Effects on Human Health and Property

There are some concerns regarding the adverse impact on humans who work or live close to ground-based rectennae that receive microwaves carrying power transmissions from an SPS. There is currently no data fully proving such risk, mainly due to current indecision about the type of radiofrequency to be used and the intensity of the power to be transmitted. Still, the URSI has expressed its concern that "above the center of the rectenna, the SPS power flux density will be considerably higher than the currently permissible safety levels for human beings."

The URSI warned that human access around the rectennae "would need to be carefully controlled to ensure environmental safety and health standards are maintained." The URSI noted that the International Commission on Non-Ionising Radiation Protection and Japan both apply more stringent limits. However, there are no legally binding international safety standards to ensure human health safety from exposure to microwaves carrying power transmissions from an SSSPS. Any damage in the form of adverse effects on human health (i.e., 'impairment of health') caused by SSPS electric transmissions would become a basis for imposing responsibility and/ or liability on the SSPS operating States and, consequently, for a claim for compensation under international space law.

Frequency bands, satellites to be used, power and density levels, and the diameter of the antenna, An SSPS system involving any international cooperation would therefore have to satisfy all of these requirements, creating a lengthy bureaucratic procedure before it could become functional. Because States are internationally liable for damage caused by their space objects, whether the responsible entity is public or private, they will tend to put domestically applicable liability regimes in place in order to limit their liability vis-á-vis entities involved in an SSPS operation and to be able to recover possible international claims made against them. SSPS operations will therefore need to be conducted with consideration of the relevant liability regimes.

2.6. Managing Risks

In order to make SSPS operations more attractive and economically viable, it would be prudent to consider ways of managing the risks associated. This can be done in several ways, including risk allocation, public-private partnerships, transparent regimes of dispute resolution, and improved risk management regulation through domestic legislation. Each of these will be discussed herein turn. Traditionally, as in commercial undertakings in a wide variety of scenarios, exposure to liability in an SSPS system could be handled by allocating a particular risk to the party or parties best suited to manage it at a minimal cost. This would apply in ventures whether they are private or public, or some combination thereof. Often, the risk is allocated by the procurement of insurance coverage. Insurance can be obtained by the satellite owner (the exception), launch suppliers (the rule), or the satellite operator.

The risk allocated has:

- Been made fully aware of the risks they are taking,
- The most significant capacity [expertise and authority] to manage the risk effectively and efficiently (and thus charge the lowest risk premium),
- The capability and resources to cope with the risk eventuating,
- The necessary risk appetite to want to take the risk and
- Been given a chance to charge an appropriate premium for taking it.

Thesis Summary

Wireless Power Transmission (WPT) technology using a satellite-to-satellite system represents a valuable and convenient technology for transferring power wirelessly among Space Solar Power Satellites (SSPS) to Satellite and potential upcoming interplanetary missions. This direct transmission offers a possible solution to deliver continuous, convenient, and unlimited energy supply to satellites help replace traditional power storage and reduce the weight and ultimately the costs of launching satellites. Satellite industries traditionally use photovoltaic cells and nuclear generators to satisfy the needed electricity needed by spacecraft. Current power generation and effective management systems occupy up to 10-25% of the satellite's mass.

The concept of laser-based WPT from Energy Satellite (E-Sat) can overcome substantial problems. The current design of SSPS primarily focuses on designing a massive satellite to generate and transmit gigawatts of energy to an Earth-based ground receiving antenna. This consistent idea can be adopted for spacecraft by developing a constellation of E-Sat called Energy Orbit (E-Orbit) to supply sufficient power to spacecraft within range. It will increase the impressive performance and operational lifetime, especially for small and cube satellites using microwaves and laser-based power transmission. By developing a small scale SSPS, E-Sat for WPT application in space, followed by a practical demonstration of mother-daughter satellite and accurately evaluating the possibility for subsequent implementation. In addition, creating 1600 E-Sats constellations to fulfill the power demand in low earth orbit.

Nevertheless, another potential avenue where E-Orbit supports a possible application is its utility towards rovers and habitat. For instance, the rovers find it difficult to investigate in the far side, crater, and polar region of the Moon, where sunlight is unavailable for a few

days. This challenge can be suitably overcome by employing an E-Sats, which can be used for WPT, independent of its location. Such techniques demonstrate possible applications towards power transmission for unmanned aerial vehicles for faster mapping purposes. As such, the dependence of those aerial vehicles towards fixed energy storage becomes alleviated. Simultaneously the future habitat on Mars and the Moon will receive continuous power by developing a perfect Mars and Moon E-Orbit system for interplanetary and solar-system investigation mission satellites to achieve continuous power.

The theoretical modeling allows the analysis of power conversion or transmission for each unit in terms of laser impacts, transmission efficiency, and photovoltaic-cell thermal property. Maximum power transmission efficiency is calculated based on a linear approximation of power conversation between electricity-to-laser and laser-to-electricity validated by numerical simulation. This efficiency variation depends on the selection of Laser, transmitter, transmission distance, and photovoltaic cells, the same as increasing the maximum transmission efficiency of information in a wireless communication network. Consequently, this thesis gives insight into wireless power transmission in general and adequate guidelines of the satellite to satellite power transmission system design in practice. The development and demonstration of this technology can help fulfill Space Solar Power Satellites' idea to transfer gigawatts of renewable energy to Earth.

Chapter 1: Introduction: This chapter contains the literature review for the energy crisis on Earth concerning the ever-growing population. The production of affordable electricity from renewable and nonrenewable energy and associate problem. Implement a new renewable power system and economic production using space-based SSPS and unique challenges to design it. The current power management systems for space and associated technologies efficiently utilize to sufficiently support diverse missions around Earth and interplanetary mission. Creating E-Sats and a constellation design in low Earth orbit, E- Orbit.

Chapter 2: Literature Review and E-Orbit Refence Mission Designing: This Chapter includes the necessary information on and overviews of SSPS designing and the historical development of WPT, and remarkable experiments efficiently conducted from all over the world. The numerical analysis and exercise of microwave and laser systems for wireless power transmission. The current policies and key challenges around space solar power and wireless power transmission system designing. Discussed the reference parameters for E-Sat and E-Orbit.

Chapter 3: Energy Satellite: The system analysis and system engineering to develop a small-scale SSPS properly as E-Sat. The essential components, high accuracy attitude, orbital control subsystem, propulsion subsystem, active sensors, power generation, and management unit, including laser power transmission subsystem as payload for E-Sat. The power transmission efficiency and power density at receiving object to satisfy user requirements. The novelty of E-Sat compares to the historical SSPS design. The necessary orbital variables and their operational performance inside the proper orbit.

Chapter 4: Energy Orbit: The practical importance of Energy Orbit and the formation flying of E-Sats in low Earth orbit. Orbital characteristics to properly transfer continuous non-disruptive laser power to customer satellite and maintaining power across Energy Orbit. The orbital characteristics and continuous interaction with the customer satellite with the specific decided range of power transmission. The system modeling and necessary phase to phase timeline for progressively developing Energy Orbit across low Earth orbit. Numerical analysis and the other significant scenario can tackle the debris removal, laser propulsion, safety, deorbiting, orbital maneuver—the driven economy and potential revenue within the 20 successful years of launching Energy Orbit in space. In addition, this key section suggests and recommended few subsequent policies to properly consider for the successful development of SSPS and WPT in the context of Energy Orbit.

Chapter 5: Interplanetary Energy Orbit: The designing and power support system for Moon and Mars as Moon Energy Orbit and Mars E-Orbit, respectively. The Constellation designing to transfer power to habitat, Rover, Orbiter, and Rover using E-Orbit. Support for deep-space mission and robotic mission using E-Orbit.

Chapter 6: Conclusion: Summarized the findings of the studies by general conclusions and future scope of Energy orbit.

Table of Contents

List of Figure

List of Table

List of Symbol

α	Incident Angle
∇	Vector Gradient
ω	Argument of Perigee
Ω	Right Ascension of the Ascending Node
a	Semi-Major Axis
η	Efficiency
$\eta_{DC\ output}$	DC Output Efficiency
η_{el}	Electricity to Laser Conversion Efficiency
η_{PV}	PV Efficiency
μ	Earth Gravitational Parameter (398,600.4415 km3 /s2)
σ	Constant
χ	Constant
κ	Visibility
ρ	Size Distribution of the Scattering Particle
A	Area
Ac	Constant
AlGaAs	Aluminum Gallium Arsenide
Amp	Ampere(s)
Ar	Argon
A_r	Aperture Area of Receiving Antenna
A_t	Aperture Area of Transmitting Antenna
B	Earth's Magnetic Field
bn	Billion(s)
cm	Centimeter(s)
cm3	Centimeter Cubed
CO	Carbon Monoxide
CO2	Carbon Dioxide
$CuInSe_2$	Copper Indium Gallium Selenide
d	Distance
D	Dimeter
db	Decibel(s)

e	Eccentricity
E	Energy
f	True Anomaly
G	Gravitation Constant (Earth: 9.80665 m/s^2)
GaAs	Gallium Arsenide
GaAlAs	Gallium Aluminum Arsenide
GaSb	Gallium Antimonide
GHz	Gigahertz(s)
G_r	Gain of Receiving Antenna
G_t	Gain of Transmitting Antenna
GW	Gigawatt(s)
h	Altitude of Satellite or Hight of Satellite from Sea Level
H_2	Hydrogen Gas
H_2O	Water
HCl	Hydrogen Chloride Gas
HO_x	Hydrogen Oxide Radicals
i	Inclination
I	Current
InGaAs	Indium Gallium Arsenide
InGaP	Indium gallium phosphide
I_s	Solar Irradiance (1367 W/m^2)
I_S	Saturation current
I_{SC}	PV short circuit current
J2	Earth's Oblateness effect (1.0826269×10^{-3})
J_d	Given Integer Number
J_n	Given Integer Parameter
k	Boltzmann Constant (1.380649×10−23 J·K^{-1})
kg	Kilogram(s)
km	Kilometer(s)
kV	Kilovolts(s)
kW	Kilowatt(s)
kWh	Kilowatt-Hour
LiF	Lithium Fluoride

Li-CF_x	Lithium Carbon Monofluoride
Li-SO_2	Lithium-Sulfur Dioxide
LNG	Liquid Natural Gas
LOx	Liquid Oxygen
m	Meter(s)
M	Million(s)
MHz	Megahertz(s)
MM	Magnet Moment
m^3	Meter Cubed
Me	Geomagnetic Strength of the Dipole Vector
MJ	Megajoule(s)
mW	Milliwatt(s)
MW	Megawatt(s)
n	Normal Vector
N	Nitrogen
N_2	Nitrogen Gas
N_2O_4	Dinitrogen Tetroxide
nc	Number of Coil Winding
NM	Control Torque
nm	Nanometer(s)
NH_3	Ammonia
NH4ClO4	Ammonium Perchlorate
NOx	Nitrous Oxide Radicals
P	Phosphorus
P_{DC}	Input DC Power
P_i	Injected Power
P_l	input Power to Laser
P_O	Output Power
PM10	Particulate Matter (under 10 micrometers in diameter)
P_{mp}	Maximum Power
P_r	Received Power
PR	Performance Ratio
P_t	Transmitted Power
q	Electron Charge Constant ($1.602176634 \times 10^{-19}$ coulomb)

Q_E	External Quality (magnetron)
r	Solar Panel Yield or Efficiency
R	Unit Geocentric Position Vector
R_E	Radius of Earth (6378.1363 Km)
R_S	Length of the Geocentric Position Vector
s	Sun Vector
Sb	Antimony
Si	Silicon
SO_2	Sulphur Dioxide
t	Time
T	Temperature
T_Ω	Orbit Nodal Period
TiO_2	Titanium Dioxide
TWh	Trillion Watts Hour
U-235	Uranium-235
UDMH	Unsymmetrical Dimethylhydrazine
V_m	Thermal Voltage
V_o	Output Voltage
W	Watt(s)
Wh	Watt hour

Abbreviation

AC	Alternating Current
AMO	Airmass Zero
APU	Auxiliary Power Unit
BOL	Beginning of Life
CAD	Computer Aided Design
CAST	China Academy of Space Technology
CMG	Control Moment Gyro
CRM	Crew and Resupply Module
CW	Continuous Wave
DDT&E	Design, Development, Test and Evaluation
DRM	Design Reference Mission
EMI	Electromagnetic Interference
EOL	End of Life
FET	Field Effect Transistor
ACS	Attitude Control System
ACSS	Attitude Control and Station keeping Subsystem
AOCS	Attitude and Orbit Control System
Amp	Ampere(s)
ATA	Automatic Threshold Adjust
ATCS	Active Thermal Control System
C&DH	Command and Data Handling
C4MJ	Upright 3J Metamorphic Cell
CAST	China Academy of Space Technology
CCD	Charge-Coupled Device
CMOS	Complementary Metal–Oxide–Semiconductor
CNSA	China National Space Administration
CR	Concentration Ratio
DC	Direct Current
DoE	Department of Energy
EADS-ST	EADS-Space Transportation
E-Orbit	Energy Orbit
EPS	Electric Power System

E-Sat	Energy Satellite
ESA	European Space Agency
ESPOS	Energy Storable Orbital Power Station
GEO	Geosynchronous Earth Orbit
GFRTP	Graphite Fiber Reinforced Thermoplastic
GNC	Guidance, Navigation and Control
GPS	Global Positioning System
HEO	High Elliptical Orbit
IoT	Internet of Things
IR	Infrared
ISAS	Institute of Space and Astronautical Science
ISM	Industrial, Scientific, Medical
ISRO	Indian Space Research Organisation
ISS	International Space Station
ISY-METS	International Space Year Microwave Energy Transmission in Space
JAXA	Japan Aerospace Exploration Agency
Laser	Light Amplification by Stimulated Emission of Radiation
LCRODD	Lunar Crater Observation and Sensing Satellite
LEND	Lunar Exploration Neutron Detector
LEO	Low Earth Orbit
LGA	Lunar Gravitational Attraction
LiDAR	Light Detection and Ranging.
LLNL	Lawrence Livermore National Laboratory
LNA	Low Noise Amplifier
LNG	Liquefied natural gas
LPT	Laser Power Transmission
LRO	Lunar Reconnaissance Orbiter
M^3	Moon Mineralogy Mapper
MINIX	Microwave Ionosphere Nonlinear Interaction eXperiment
MMRTG	Multi-Mission RTG
MPT	Microwave Power Transmission
MSFC	Marshall Space Flight Center
NASA	National Aeronautical and Space Administration
NASDA	National Space Development Agency of Japan

Nd:YAG	Neodymium-Doped Yttrium Aluminum Garnet
NEDO	New Energy and Industrial Technology Development Organization
NIAC	NASA's Innovative Advanced Concepts
OBDH	Onboard Data Handling
OECD	Organisation for Economic Co-operation and Development
OPEC	Organization of the Petroleum Exporting Countries
PTCS	Passive Thermal Control System
PV	Photovoltaic
QPSK	Quadrature Phase-Shift-Keying
RCAST	Research Center for Advanced Science and Technology
RF	Radio Frequency
RLV	Reusable Launch Vehicle
RTG	Radioisotope Thermoelectric Generator
SC	Spacecraft
SDG	Sustainable Development Goal
SERT	Scientific Exploration Research Technology
SGA	Solar Gravitational Attraction
SPS	Solar Power Satellite
SRP	Solar Radiation Pressure
SS	Satellite-to- Satellite
SSPA	Solid-State Power Amplifier
SSPS	Space Solar Power Satellite / Station
TWTA	Traveling Wave Tube Amplifier
UAV	Unmanned Aerial Vehicle
USA	United State of America
USEF	Institute for Unmanned Space Experiment Free Flyer
USSR	Union of Soviet Socialist Republics / Soviet Union
WPT	Wireless Power Transmission

Part - I

Chapter 1

Introduction

Chapter 1

Introduction

1.1. Energy Problem on Earth

Since the Industrial Revolution, electricity becomes a necessity for humans on Earth. Much of the technological developments within the last century show just how much humans have become dependent on electricity, which has become an industry essential to the global economy and sustainability. Electricity is used in various ways, including heating, cooling, processing computers, manufacturing in industries, and provides convenience and connectivity in daily lives. Sources of electricity can be categorized into renewable energy and nonrenewable energy sources.

Renewable energy is defined as natural sources that can be regenerated and replaced over a short period of time, such as solar, hydro, wind, biomass, and geothermal. Nonrenewable energy is characterized by its difficulty replacing after consumption and takes more extended periods to produce sources such as nuclear isotopes and fossil fuels, including coal, petroleum, and gas. Nuclear isotopes are limited on Earth; cannot be produced again after consumption. Fossil fuels are traditional energy sources for electric power plants, automobiles, and industrial plants. The nonrenewable energy source produces electricity by burning fossil fuel which is more reliable and efficient than a renewable energy source. However, they release pollutants and toxic gases into the atmosphere, which causes harsher weather conditions, increased Earth's temperature, all symptoms of global warming. Burning fossil fuels produce large amounts of CO_2, CO, and other toxic gases, which degrade the ozone layer atmosphere around Earth. While recently technology for carbon capture and storage system to store and capture to utilize and convert CO_2 back to its fossil fuel form is in development, it is only in its early stages and not yet capable of restoring the damage done.

However, power comes at a cost. The highest incurred costs are economical due to their unsustainability. Almost 60 % of global primary energy comes from petroleum and gas. Members of OPEC now make up around 81 % of the entire proven global oil supply, besides the nations of the OECD own just around 4 % of the reserves, although using roughly 52 % of the world's total quantity. OPEC accounts for just around 4 % of the reserves.

Other consequences are also present. As the developing nations come online to enhance their living standards dramatically, the nature of energy usage is fast changing. Within the climate change impact framework, energy consumption is expected to climb fast in summer to refresh in the 21st century. Furthermore, the thermal energy efficiency that presently accounts for around 80 % of worldwide electricity will diminish as temperatures increase. As Earth's meager resources become depleted, demands are increasing.

The capacity of a nation to lead successfully in the external politics arena or ideological priorities to defend foreign gas and petrol supplies may be undermined by a financial burden. In addition to issues with safety, carbon-based energy also involves health, and environmental expenses, including air emissions, water discharge, contaminated soil, mining soil degradation, black lung cancer, global warming, and acid rain. Such a nonrenewable fossil-based system is not only costly even so inefficient. Oil fields, coal, and natural gas farms are far from the most productive centers, incurring additional travel costs and/or electrical transmission expenses and frequently leading to loss of power during transit. The archaic distribution infrastructure is incompatible with or overlaps with at least one renewable green alternative to fossil-driven energy.

Utilizing the natural resource that can be regenerated quickly is a renewable energy source, as discussed above. Renewable energy sources do not release toxic greenhouse gases. In comparison, this technology does have high initial costs; even so, it has an economic advantage of sustainability. However, few renewable energy sources, such as wind turbines, are a concern among the ornithologist and ecologists because renewable energy could endanger wildlife, particularly birds and their migration around the Earth. Currently, advancement in the solar panel industry shows the highest potential for energy production at social, commerce, and industrial levels. The problem associated with renewable energies is high initial costs, careful planning, vast land utilization, and unpredictable weather conditions. Renewable energy is still facing many challenges and is yet to become the primary source of energy for global needs.

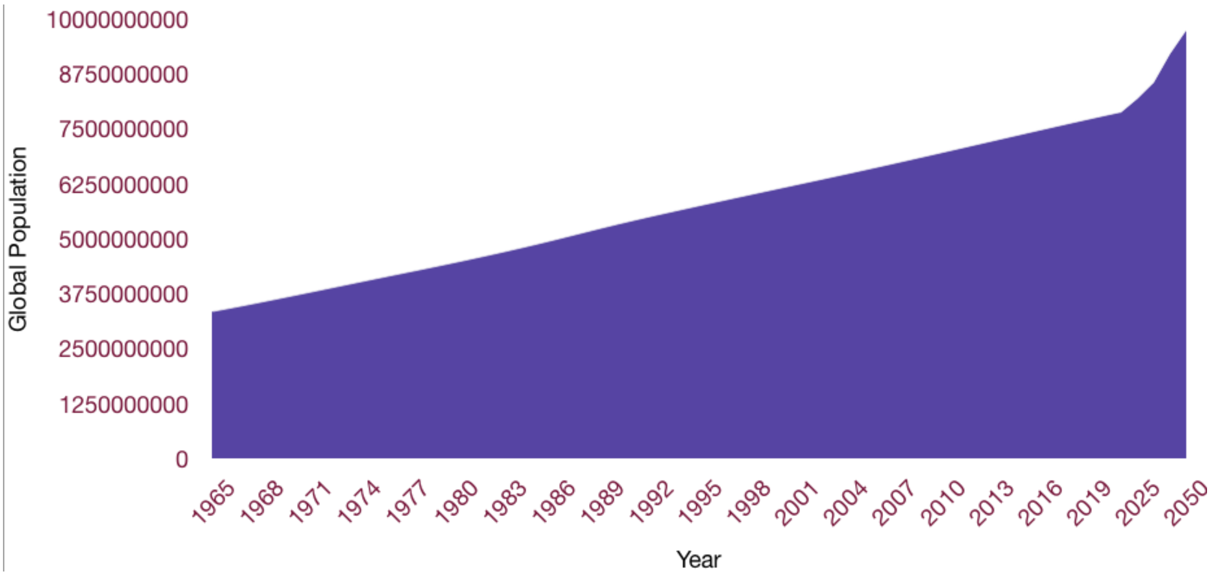


Figure 1.1 World Population from 1965 – 2050

In contrast to the limited energy sources, the demand for energy is skyrocketing. According to the reports, the population on Earth is 7.79 Billion [1] and generates approximately 26,907 TWh of electricity [2], and the global total electricity consumption in 2020 approximates to 23,177 TWh [3], which is significantly increasing year by year. Global electricity demand is projected to double from 2015 to 2050, at 42,000 TWh [4], with the population nearing ten billion, as predicted in Figure 1.1.

To satisfy global energy demands, a new perspective is necessary to build new power plants that include nuclear power plants, coal power plants, hydroelectric power plants, solar power stations, wind power plants, among others. Nevertheless, energy producers will need to look into the perspective of global climate change concerning energy production as such power plants are harming the environment due to greenhouse gases. Two-thirds of global greenhouse gas emissions are burning fossil fuels to produce energy for heating, electricity, transport, and industry. The global energy consumption from the variable energy source is shown in Figure 1.2.

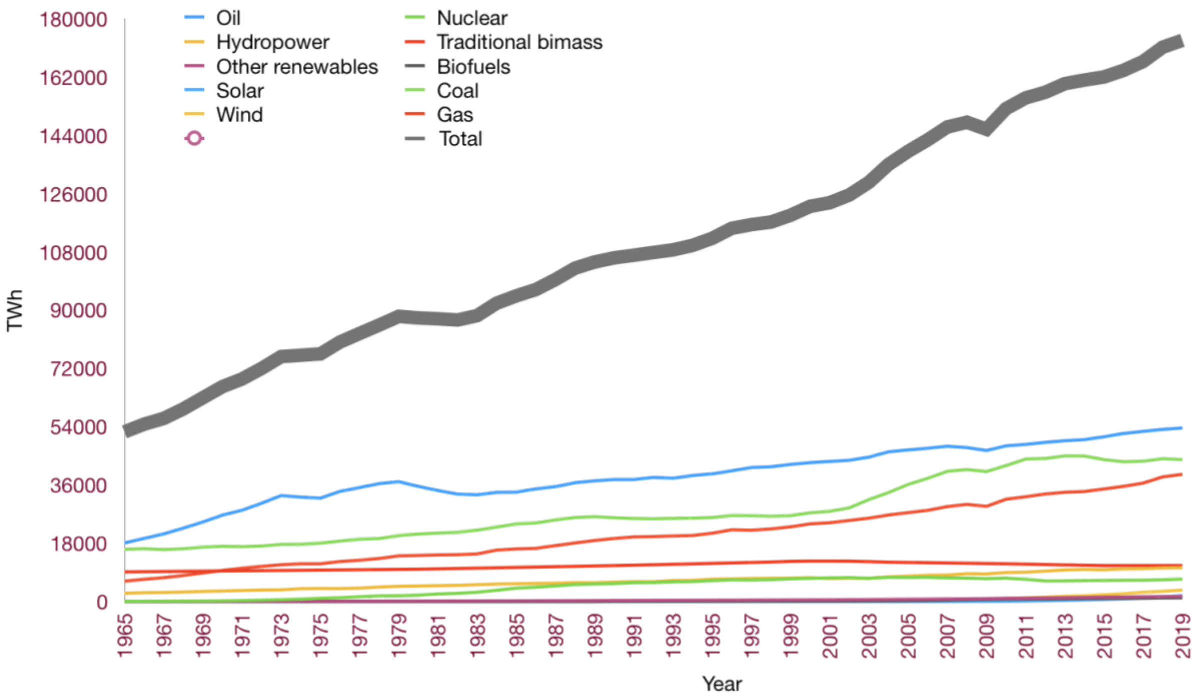


Figure 1.2 Global energy consumption in TWh

In Japan, electricity is costly compare to the other parts of the world. After the 2011 nuclear disaster, only five nuclear power plants out of 22 generators are currently active. Japan's total energy production is approximately 2,592.08 bn kWh, and actual consumption is about 989.3 bn kWh from all sources of electricity; Table 1.1 shows the energy production data with per capita requirement of electricity.

Table 1.1 Japan total energy production

Energy Source [5]	Total in Japan (bn kWh)	Percentage in Japan	Per Capita in Japan (kWh)
Fossil Fuels	1,840.38	71,0 %	14,575.54
Nuclear Power	25.92	1,0 %	205.29
Water Power	207.37	8,0 %	1,642.31
Renewable Energy	518.42	20,0 %	4,105.79
Total Production Capacity	2,592.08	100,0 %	20,528.93
Actual total Production	989.30	38.2 %	7,835.11

1.2. Space Solar Power Satellite and Power Beaming

In the 21st century, electricity has become a necessity for daily life. As an alternative to low carbon generation, a modern-style nuclear power plant offers the lowest electricity cost [6] to satisfy the increasing number of populations versus the electricity demand. Unquestionably, the world's energy needs will dramatically rise in the future, and the quest for alternative sources of energy will increase, as shown in above Figure 1.1. Currently, the world is moving towards the PV-based energy solution by placing substantial solar farms on a broad landmass. Solar farms can provide a significant amount of energy without any ecological impact. However, the cost of manufacturing, storage, and the instability of light sources make it challenging to rely on such farms. For example, a 1 GW coal power plant at 4 km^2 can be replaced by 24 km^2 (for daytime power generation) and six-seven units of power storage facility within three billion USD as shown in Figure 1.3.

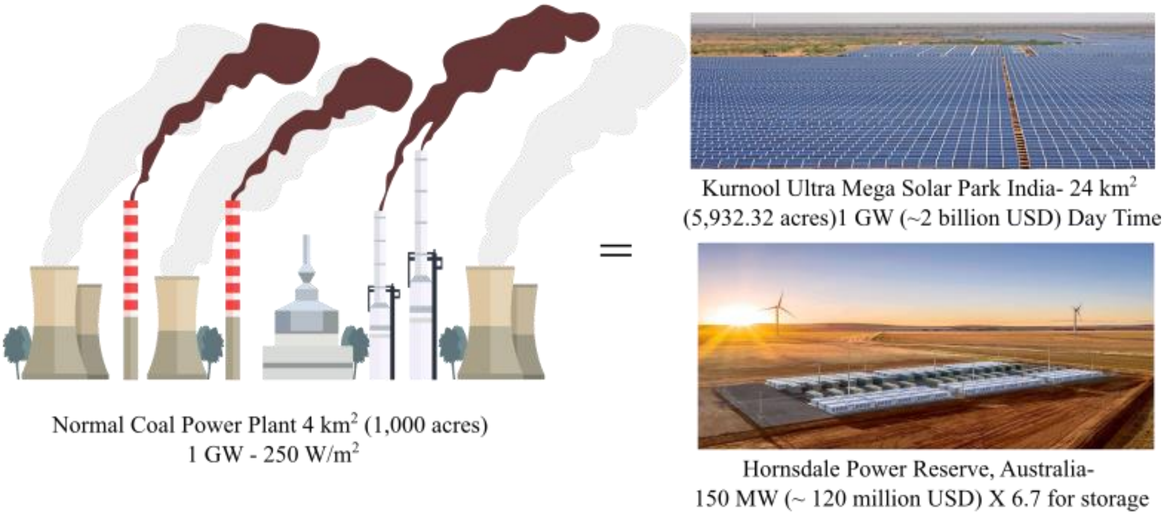


Figure 1.3 Comparison of Coal Power Plant Vs. PV Solar Farm and Storage Facility

Peter Glaser suggests that SSPS will be an alternate source for fulfilling and find a novel concept of transmitting energy from space to Earth. In the era of nuclear energy, highly developed new generation systems provide safety and security to any nation. However, Peter Glaser quoted that in [7] "There are at present serious doubts that nuclear energy can be relied upon to fill a continuously larger percentage of energy requirements since there are no known ways as yet to provide complete insurance against environmental pollution by nuclear energy plants." The recent Fukushima Daiichi nuclear power plant accident leads to the search for an alternative safe and renewable energy source for continuous power supply [8-10]. The concept of SSPS comes into light only after this accident. Meanwhile, the space community has already proposed and investigated several designs long before this accident.

The first design was prosed and patented by Peter Glaser in the late 1960s [11,12]. The design was to collect and convert solar radiation energy to electric energy using GEO based satellite. The produced electricity is converted and transmitted using MPT to a receiving site on Earth. It has advantages over the traditional PV solar panels on ground-based generation units of constant, 24X7, and renewable power supply to any location using WPT. The WPT nowadays is getting popular with the advancement of technology. Cellphones are coming with wireless charging, and IoT devices use short-distance WPT functionality for productivity. The WPT concept was proposed and examined in the 19th century. Nicola Tesla designed the Tesla tower for long-distance WPT [13]. In the experiment, Tesla uses a minor wave frequency for transmission. Since then, researchers have enhanced the capability of WPT using microwaves and lasers, with the characteristics of WPT that include transmission through the atmosphere, simplicity in design with directional emission, and ease to convert back into another energy form. The two types of WPT, MPT and LPT will be extensively discussed in the following chapters.

The dangers of SSPS system operators cannot be ignored. Chapter 2 in this thesis describes the following security hazards associated with SSPS towards the Earth damage and evaluates their capability. Issues considered explicitly under international treaties include licenses, electronic transfer, and liability management methods, including insurance, responsibility waives, and dispute settlement procedures. Demand for electricity nearby Earth orbit for SC, interplanetary mission, and future sustainability will certainly influence business strategies in space.

Advancing the use of wireless communications brought about substantial changes worldwide, and potential innovation to provide value and real-time power supply for connected networks is the subsequent significant advance in wireless transmission of power. While radiative WPT shares many common properties with wireless information or transmission, the performance requirements, transmitter, receiver designs, and hardware limitations also differ dramatically and have been widely investigated.

1.3. Space Solar Power for Space Application

WPT technology using a satellite-to-satellite system represents a valuable and convenient technology for transferring power wirelessly among SSPS to Satellite and potential future interplanetary missions. This direct transmission can help replace traditional power storage and reduce the weight and ultimately the costs of launching satellites. This thesis discusses the WPT between one small-scale SSPS to another operational satellite, followed by a demonstration of small-scale SSPS and evaluates the possibility for future implementation. SSPS will increase the performance and operational lifetime, especially for small and cube satellites using microwaves and laser-based power transmission. The development and demonstration of this technology can help fulfill SSPS idea to transfer gigawatts of renewable energy to Earth. This thesis will summarize several WPT methods, examine the history of radioactive WPT technology, and discuss the significant obstacles facing the conception of modern WPT systems and conclude with describing the new ways of telecommunication and communication systems to face these issues with power transmission. Topics presented include energy harvesting, WPT power transmission, channel acquisition, WPT Smart Energy Area Characterization, linear- and semi-energy-receiver model wave forming, WPT security and health problems, WPT-based SSPS management wireless charging.

The current design and technology enhanced the original ideas of Glaser. Deployment collectors can be situated in the GEO, MEO, or moon surface while visualizing arrays in the GEO. More recent plans have even allowed SSPS to be placed at LEO with much less energy to go to ground stations. Glaser foresees photovoltaic panels as collectors to collect the solar radiation in a receiver that absorbs radiant energy. PV film designs are becoming thinner with technological development. Although overall efficiency has increased over time, new materials with longer lifespans are expected to appear in the market shortly.

Other novel technologies include both MPT or LPT (coherent visible or infrared light). While the efficiency of the microwave has grown considerably, laser transfer can become more comparable. The WPT would be gathered in broad areas antenna (or rectennas) from one to ten km across the ground. The antennae would be rectified. One of its many advantages is that the energy created does not pollute, and no toxic waste needs to be disposed. Furthermore, the soil below the rectennas may be utilized for other industries, such as agricultural or aquatic factory farms. The differences are somewhere between these regions and those needed for coal and nuclear power plants. The cost and performance success of SSPS systems is mainly determined by favorable political and legal conditions as its technological capabilities are evident.

One such variable is the safety hazards and potential liabilities of the SPS system operators. Developers must address complications of the culpability of harm and damage under international treaties. This way, satellite to satellite power transfer can be performed using small-scale SSPS, Energy Satellite or so-called E-Sat. A constellation of 1600 E-Sats will create an orbit across Earth to electrify the LEO. It will be called Energy Orbit or E-Orbit. Will be able to provide energy to customer satellites via laser and microwave to fulfill their energy demands for performing operations. So, clients will be able to send their satellites without mounting any power generation components on satellites, decreasing the weight and increasing the satellite's lifetime. Hence, the launch cost will be reduced. Initially, E-Sat's altitude will be near 900 KM, where the density of active satellites is minimum. However, by 2026, the number of satellites will be increased in 400 – 1400 km attitude. E-Sat will provide energy to customer satellites in the 500 km radius, even in eclipse mode.

Using the 900 km altitude of E-Sats, almost more than 60 % of satellites in LEO will be in the range of the constellation of 1600 E-Sats. Customer satellites near the E-Sats will receive energy via microwave or laser, and the remaining customer satellite in a 500 km radius will receive energy via laser. However, with the progress of this project, the investigated a basic functional block diagram throughout the LPT system explained in Figure 1.4 where the PMS is working for generating and converting power from PV cells convert into DC form. Afterward, the DC will be converted to a laser using an optical antenna for power transmission to the objective destination with the confirmed transmission. Power generation and power storage devices will no longer be required thanks to constant LEO to LEO energy supply.

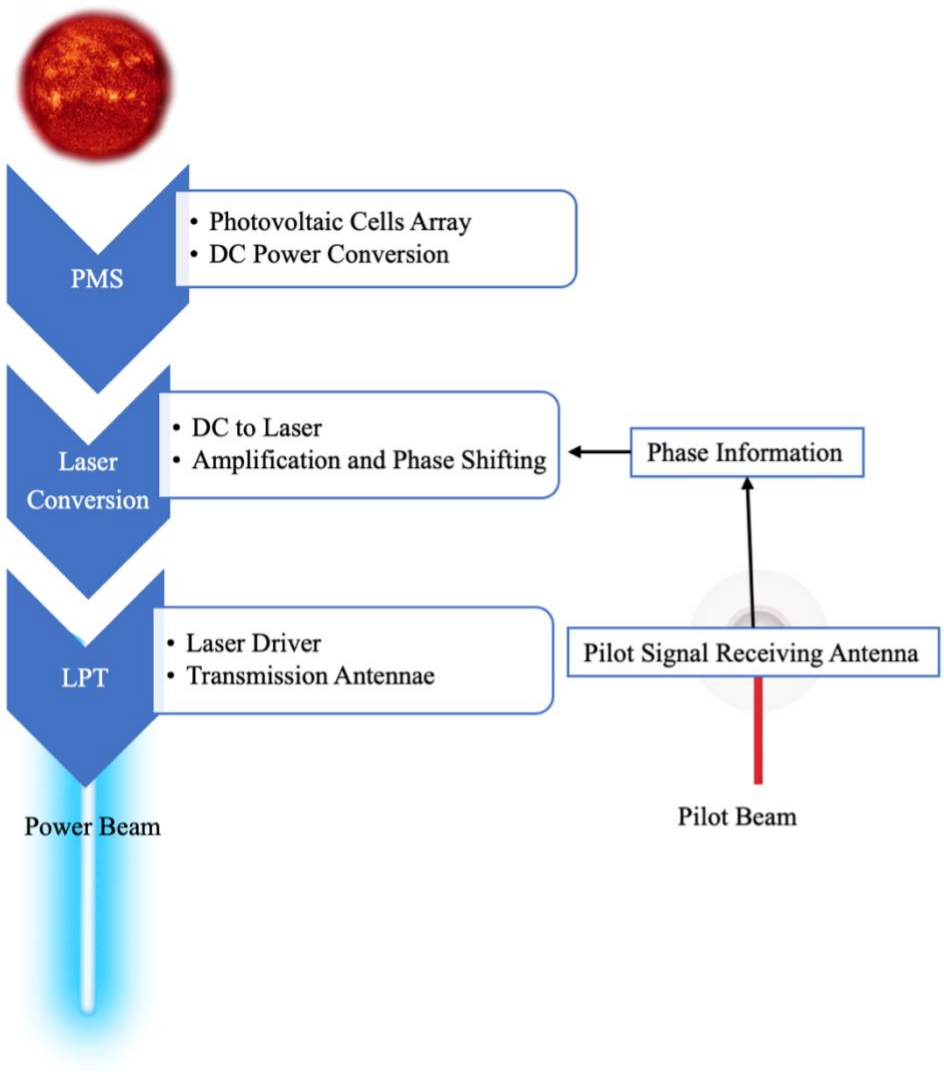


Figure 1.4 Generic Space Solar Power Satellite and Power Beaming functional block diagram

Solid-state lasers like Nd-YAG ranging of 10 kW will be used for transmitting energy via laser beams. 200 W laser source from a small group of E-Sats will help some customer satellites for orbit transfer called laser propulsion. This propulsion will be achieved by three-axis movements of laser source and highly accurate rectenna on customer satellites. Hence rectennas will replace solar panels and batteries on operational satellites. The weight of the propulsion system of the customer satellite will also be drastically reduced, and the life of such a satellite will increase. The technical demonstration mission is planned to launch to validate technical aspects of E-Orbit. It contains two satellites where one of them will be an E-sat called a mother satellite, and another one will be a daughter satellite which will receive energy from the mother satellite. It will launch in an orbit of inclination 90 degrees to the Sun.

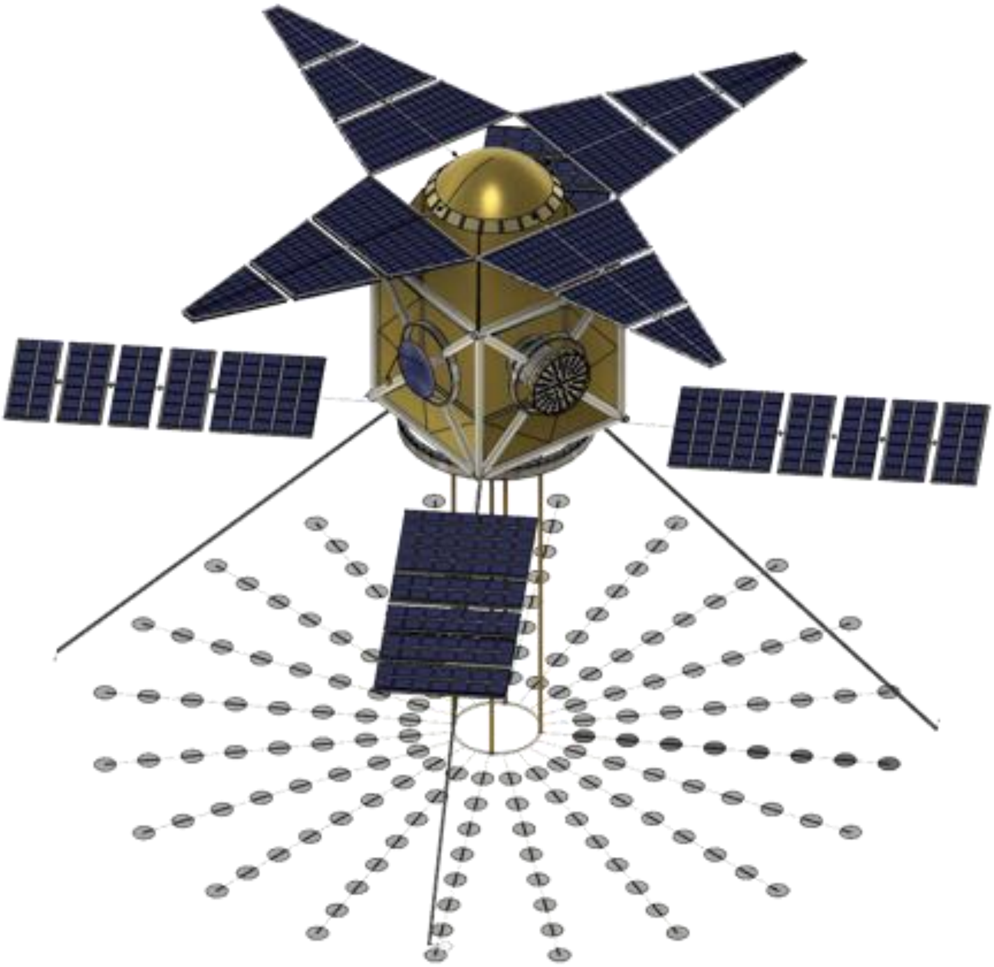


Figure 1.5 The conceptual design of Energy Satellite (E-Sat)

Currently, designing, development, validation, and analysis of E-sat are in the last steps. The demonstration model of the E-sat will be the small-scale SSSP satellite. It will send laser energy to one daughter satellite from lasers of 10 KW with efficiency ranging from 8 - 12 %. Mother satellite will generate energy from both sides of solar panels and thermoelectric generators. Thin foiled and thermal white reflectors will reflect sunlight on the backside of solar panels. Mother satellite will be mounted on the bus of 3×1×1 m. During the Earth reentry mission, the Sun-facing side will have a conical shape to protect the system for reusability, as shown in Figure 1.5. A timeline of the entire project is shown in Figure 1.6, as the project of E-Orbit is started in October 2018 with the idea of creating an SSPS system in compact and with current technology to make it appropriate for business point of view.

1.4. Power Management System in Spacecraft

Space exploration and spacecraft have become more advanced and require more computational power, necessitating cutting-edge power management systems. Power products meet the restrictive voltage specifications of microprocessors, microcontrollers, PV, sensors, payload, and other harsh-environment space applications. Typically, PMS includes PV array, batteries, wire, heater, cooler, heating pipe, and subsystem for operational uses.

The Sun is a massive power hub, luminating a tremendous amount of energy of approximately 3.827 × 1026 W/second. Nearby the Earth's orbit, the solar irradiance is about 1360 W/m^2 with slight changes according to the distance due to elliptical orbit. Most of the satellite uses PV array for power generation in space using the energy illuminated from Sun to produce power. Recently PV industry progressed tremendously in the research of solar cell efficiency. Space agencies use nearly 30 % of their budget for solar cells, whose maximum conversion of energy to power efficiency only adds up to 47.1 %.

Batteries also play a vital role in the PMS system, storing energy for eclipse time to serve the required power at payload and onboard subsystem of satellites. Recently, Lithium-ion batteries are mostly used in space industries, for example, Li-SO_2 and Li-CF_x with specific energy densities of 238 Wh kg^{-1} and 414 Wh kg^{-1}, respectively work between - 40 and 70 °C [14]. Previously, most space missions used NiCd, NiH and AgZn with a specific energy of 30, 60, and 100 in Wh kg^{-1}. The hydrogen fuel cells are being used for the Space Shuttle mission to generate 12 kW with a specific power of 275 W kg^{-1}. However, batteries have disadvantages, such as having a finite life cycle that defines the operational life of any satellite [15].

The connecting wires, transmission wires, sensor system, payload subsystem, motors, and onboard data handling system are also factors considered in PMS for regulating and controlling the temperature in order to satellites properly in every condition. The PMS system occupies approximately 30 % of the satellite mass, in which the PV and batteries occupy an area of more than 80 % depending on mission requirements. Using E-Orbit, the PMS system of a satellite can be reduced up to 25 % of its mass. This is defined in the section below following the descriptions of E-Orbit, E-Sat, and their comparison to the traditional power management methods in chapters 3 and 4. The E-Orbit will play a significant role in interplanetary missions to generate continuous power 24×7 on the Moon even when it is on the dark side. The constellation is designed to produce power and transmit energy from one satellite [16] to the other even when they are not direct sunlight.

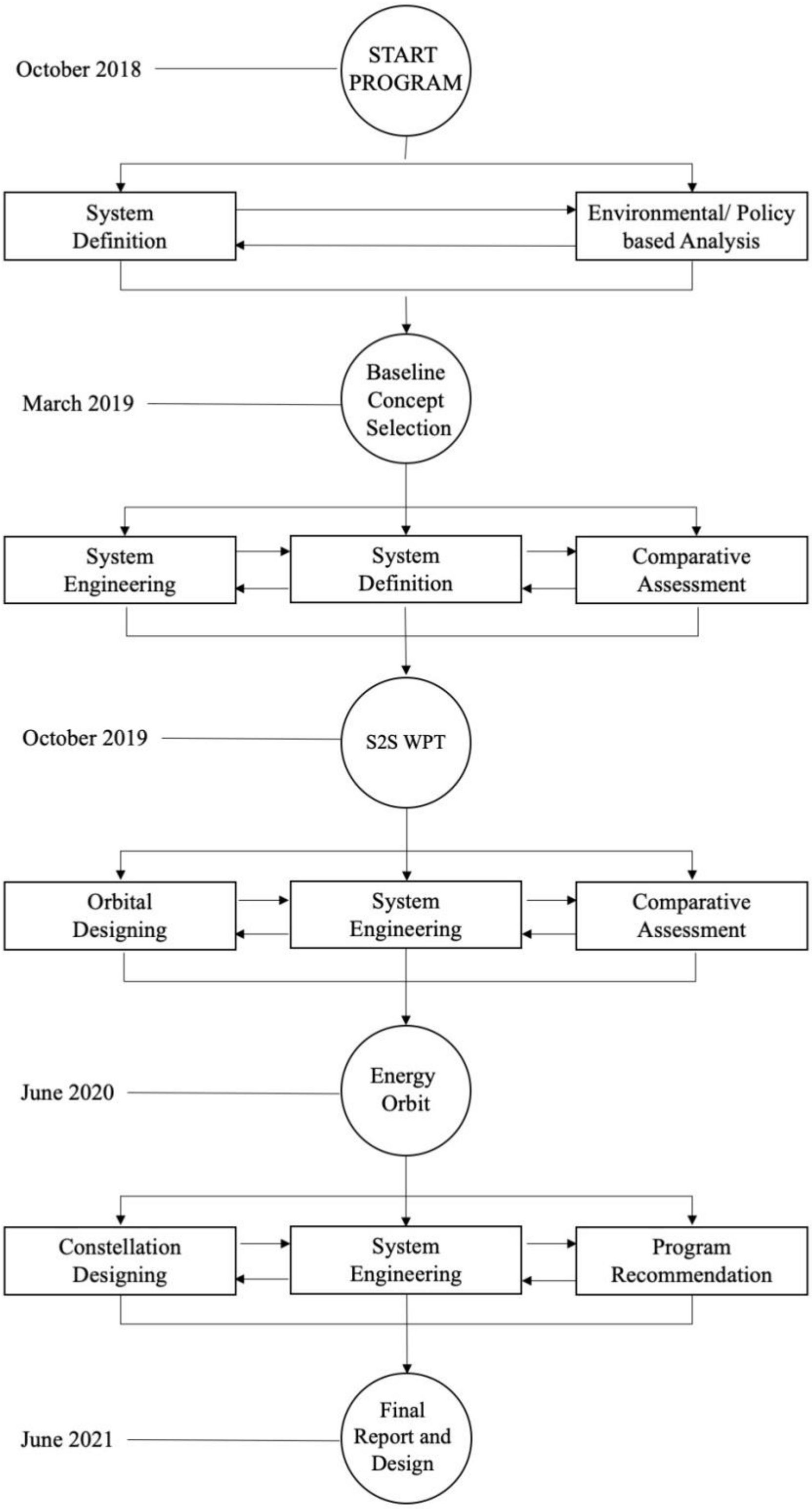


Figure 1.6 A timeline of E-Orbit study and design Evolution Model

Chapter 2

Literature Review and E-Orbit Reference Mission Designing

Chapter 2

Literature Review and E-Orbit Reference Mission Designing

2.1. Space Solar Power Satellite

Space Solar Power Satellite/ Station or Space Solar Power as SSPS represents a method of generating renewable energy from the Sun by collecting photonic energy on solar PV array in the form of DC and converting it to electricity to transmit via the microwave, millimeter-wave or laser to Earth or other applicable missions. SSPS was first introduced by Dr. Peter Glaser [17] in 1968, who was concerned with increasing energy demand and industrial consumption with increasing population. He proposed GEO-based SSPS to harvest MW of solar energy and transmit through microwave to a ground base rectifying antenna or rectenna. The rectenna converts microwave energy into a desirable DC power to feed into the on-site electric grid and dispatch it back to the consumer site. He imagined that solar power would be a cleaner energy source.

After recognizing the need to replace the current energy source to clean and renewable energy, most agencies and countries started surveying researching the SSPS system. NASA conducted the first SSPS study, Dr. Peter Glaser and others with Arthur D. Little, Inc. [18] in 1974. The team introduced a brief description of the SSPS system with key advantages, environmental effects, and estimated costs. In 1976 possible SSPS conversions and delivery system study was conducted by NASA [19] Grumman Aerospace Corporation. In 1978 [20], a collaborative work between the US DoE and NASA on concept development of about a massive scale 60 SSPS systems to transmit 5 GW power to rectenna [21] and later till 1980, several comparative studies were conducted, including for MPT, antenna designing cost-effective SSPS system [22].

From 1980 to the 1990s, several international space agencies and leading countries are considered SSPS as crucial technology and started investigating. Japan, Europe, Russia, and Canada started the experiment with Kyoto University leading the first satellite-based experiment to scientifically investigate the nonlinear interaction of strong microwaves with the ionosphere (In august 29, 1983.) called Microwave Ionosphere Nonlinear Interaction Experiment -MINIX [23, 24].

ISAS and Toshiba proposed a 10 MW SSPS system called ESPOS which in LEO which transmits potential energy every 114 minutes after a revolution to transmit for 35 minutes, and the energy will be stored as heat fluid, with LiF used as the thermal Transfer fluid for the Brayton Cycle Engine [19, 20]. In 1992 International Space University conducted a Space Solar Power Program at Kitakyushu, Japan, collaborating with NASA and other space agencies, to present the final report on the SSPS system. That year, they proposed a reusable transportation system economic-based design concept [20] and in 1993, Japan completed another experiment the ISY-METS sounding rocket experiment [21]. During this experiment, leading researchers scientifically investigated the SSPS rectenna system based on Japan's experiments on the mother-daughter MPT system to evaluate the nonlinear plasma effect on high-power microwave beams in space.

A space solar power working group proposed SPS2000 (1993) from ISAS Japan to place a satellite at LEO with a unique prism shape to transmit 10 MW of power by MPT 2.45 GHz frequency by slots antenna. The NEDO Japan proposed Sunshine Project in 1994 to transmit 1 GW by MPT at 2.45 GHz using the dipole antenna with a diameter of 1 km [24].

From 1995-1997 NASA scientifically re-investigated the futuristic look of the SSPS system as a "Fresh Look Study" for a large-scale SSPS system and improved the SSPS abstract by defining a road map for future SSPS technological promotion [25]. In 2000 NASA's MSFC start the SSPS SERT program, which published the modern prospective technological demonstrative concept and experiment to advance SSPS system articles in which many scholars joined with their exploratory work and comprehensive research on SSPS. Furthermore, Mr. J. Mankins successfully demonstrated the MPT experiment in Hawaii (2008) in collaboration with researchers from Texas A&M University and the University of Kobe to transmit energy at a distance of 148 km; in the successful experiment, they received 0.00001 W. In 2001, ESA proposed multi-SSPS sails called Sail tower SPS with 60 pairs of 150 m^2 to efficiently produce 3700 kW by a total of 450 MW in GEO using MPT 2.45 GHz.

JAXA proposed a new SSPS JAXA model in 2004 utilizing a mirror-based SSPS system using MPT 5.8 GHz to transmit 1 GW of power. Mitsubishi proposed Distributed SSPS Solar Bird Idea in 2005 to transmit 1 GW using MPT 5.8 GHz using multiple satellites. Jspacesystem and JAXA proposed the SSPS USEF Model and JAXA L-SSPS model, with 1 GW from MPT and LPT. In 2011 the attractive eye pointing idea was proposed by Mr. J. Mankins and NASA, SPS-ALPHA [21]. It consists of a large earth-oriented array, directional mirror work as heliostat that adjusts the direction of the mirror according to the Sun to concentrate power with a different type of SPS – ALPHA. China has arisen with a modern idea of SSPS- OMEGA with mirror-based high concentrated power generation [26].

Placing a large disk of solar panels in GEO to collect solar energy in space and utilize WPT to transmit the energy in the form of microwaves to the receiving station on Earth [7,17]. This design demonstrates several advantages in the terrestrial-based solar power generation method, as this avoids the impediments of inclement weather and nighttime outages during dark periods when no solar energy can be collected. The satellite can transmit continuous, renewable, clean, affordable, and safe power supply to any location on Earth, except during short periods when it will go under an eclipse.

The NASA, JAXA, DoE, Naval Research Laboratory, USSR, Republic of India, and CSNA investigated and proposed more than 27 significant variant designs that generate hundreds of kW to GW of continuous power and transmit using microwave and laser-based WPT, which include, e.g., NASA/DoE Microwave Sandwich Concept, 1980, Japanese SPS2000, 1995, SolarDisc, 1997, SunTower1997, Solaren2010, SPS-ALPHA2013, CAST multi rotary Joint SPS, 2015, SPS OMEGA 2014, SPS-Alpha MKII 2016, CASSIOPeiA 2017 which are all advanced SSPS designs as shown in Table 2.1.

Table 2.1 Conceptual design of various SSPS design

SSPS	**Shape**	**Orbit**	**Dimension**	**WPT (MW)**
PETER GLASER, 1973 [7]	Disk Type	GEO	64.749 km^2	10,000
DoE/ NASA, 1979 [27]	Rectangular	LEO/GEO	I0 × 5 × 0.5 km	5,000
JAXA SSP2000, 1993 [28]	Prism	LEO	336 × 336 × 303 m	9.8
NEDO Sunshine Project [29]	rectangular	GEO	3.2 × 2 km	1,000
Sun Tower 1997-1999 [30]	Multiple Disk	GEO	200 × 300 m	1,200
SERT, NASA 1999-2000 [31]	Multiple reflectors	GEO	4 × 7.2 km	0.1 ~ 10,000
Sail Tower SPS 2001 [32]	Multiple Square	GEO	60 pair × 150 m	450
SSPS JAXA Model 2004 [33]	Pair reflector	GEO	2.5 × 3.5 × 2 km	1,000

Table 2.1 Conceptual design of various SSPS design (continue)

SSPS	Shape	Orbit	Dimension	WPT (MW)
SOLARBIRD SPS Mitsubishi Electric 2004 [34]	Pair Reflector	LEO	1000 kg	1,000
SSPS USEF Model 2006 [35]	Square	GEO	2.6 × 2.4 km	1,000
JAXA L-SSPS 2006 [36]	Reflector	GEO	0.4 × 0.2 × 12 km	1,000
Solar Power Beaming, LLNL 2009 [37]	Concentrator and Reflector	LEO	9125 kg	1
Aerospace Corp Laser Concept 2009 [18]	Multiple PV array	GEO	29.7 ton	1200
Solaren SSP 2011 [18]	Reflector	GEO		250-2,250
SSPS ALPHA, 2011 [38]	Multiple reflectors	GEO	25,260 metric tons	2,000
Multi-Rotary Joints SSP 2015 [39]	Multiple PV	GEO	10,000 ton	1300
Sunflower Thermal Power Satellite 2015 [40]	Concentrator and heat engine	GEO	29,500 ton	5,000
SSPS OMEGA 2015 [41]	Multiple reflectors	GEO	3.4 ~ 4.5 km	2,000
CASSIOPeiA [42]	Rotating Multiple reflectors	GEO	866 tons	688
Energy Satellite [43]	Multiple Reflector	LEO	56 m^2	0.01

Table 2.1 describes the SSPS concept from Peter Glaser, 1973 idea to the space System Dynamic Laboratory 2019 Idea Energy Satellite or E-Sat in the LEO. with the shape, orbit,

dimension, and desired transmission power from various SSPS projects. However, Most of the SSPS is MPT-based. Only a few are LPT-based.

2.1.1. United States of America

2.1.1.1. Peter Glaser' SSPS, 1973

Peter E Glaser Proposed the first-time design of SSPS to collect and convert solar radiation to microwave energy. The Microwave energy is then transmitted to an earth-based ground receiving antenna or rectenna and converted to electric power for distrusting for commercial and private use. The president of the USA's message to congress for the demand for electricity and necessity was addressed on June 4, 1971. With this, a collaborative study group is formed for creating the SSPS study program.

In his design of SSPS, he proposed two GEO satellites at 35888.371 km above the Earth parallel to Earth's equatorial plane and perpendicular to Sun in SSO. The Satellites were moving from East to the west with stationary to earth receiving point in orbit. The satellite is placed about 21° out of phase (12713.84 Km) apart with a direct line of sight. Using the partition and gape will provide a continuous power supply to Earth even in eclipse time a year. The satellites are designed to convert solar energy using PV cells with the conjunction of the Seebeck effect to convert the radiated heat across the satellite by high vacuum and plasma diode and incorporated with the high optical concentrators to focus the solar radiation PV cells. To maintain the temperature, liquid nitrogen and helium would be used to treat PV cells at -258.15° C degrees, in addition to thermal insulation, and multistage cryogenic refrigeration systems to absorb heat leaks. This design would generate and covert energy using 40.23 km^2 PV cells structure to transmit 10 GW (20 kV at 5×10^5 Amp) using a combination of 10,000 amplitrons, each with a capability of 1000 kW, as shown in Figure 2.1. The phased-array planner antenna would be 100 rows with 100 panels in each row. The antenna size is different on the complete array, and the antenna would be about 500 x 500 m while it varies from center to edge, respectively 1.6 × 1.6 m to 20 × 20 m. phase synchronized with each other. The proposed frequency was from 1.498 GHz to 2.997 GHz to transmit 99.9 % generated microwave energy to 57.94 km^2 earth-based rectenna using 8 % overall efficiency. The rectenna is elevated slightly above the ground so the research facility and required operation building can be built under the structure [7].

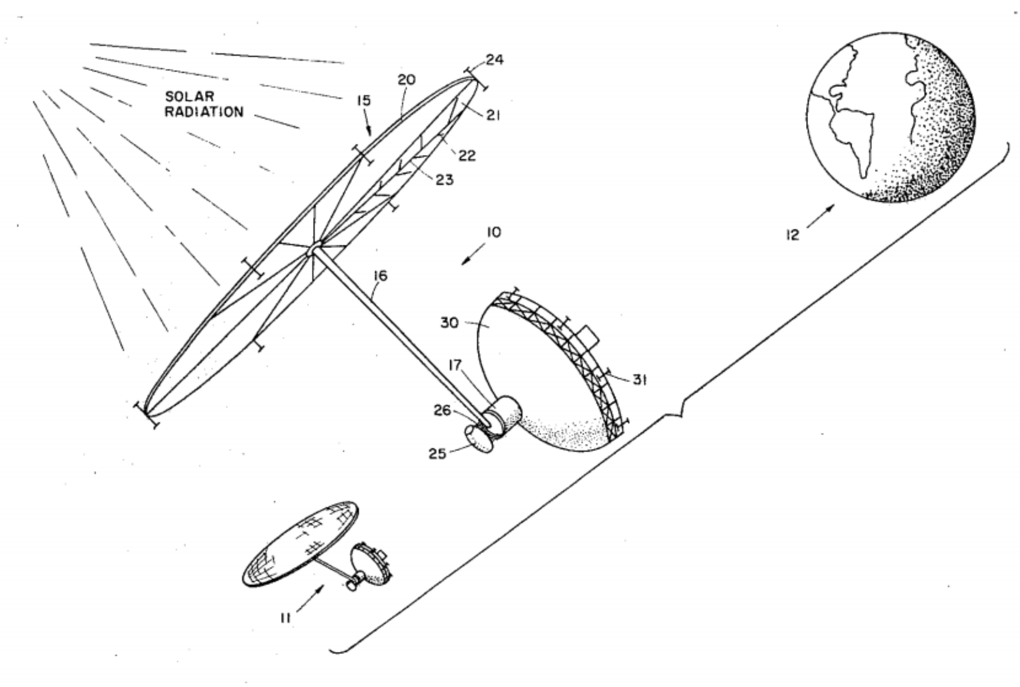


Figure 2.1 Peter Glaser's conceptual SSPS design, 1973

2.1.1.2. NASA DoE Reference Model, 1978

After Peter Glasser's proposal, the US DoE and NASA developed a conceptual design of SSPS for the year 2000 in 1978, shown in Figure 2.2. The design consists of a planar solar array with a 10.4 km × 5.2 km blanket area of 52.34 km^2 and planeform area of 54.08 km^2 on silicon base without concentrator and GaAlAs based CR system 10.6 km × 5.25 km with a blanket area of 26.52 km² and reflector area including planform area of 55.13 km² with GFRTP structure. A one km diameter phased array microwave transmission antenna with the frequency of 2.45 GHz - 6.72 GW was placed on another side. The transmission consists of a klystrons-based power amplifier with slotted waveguides elements, as shown in Figure 2.2 and detailed system information in Table 2.2.

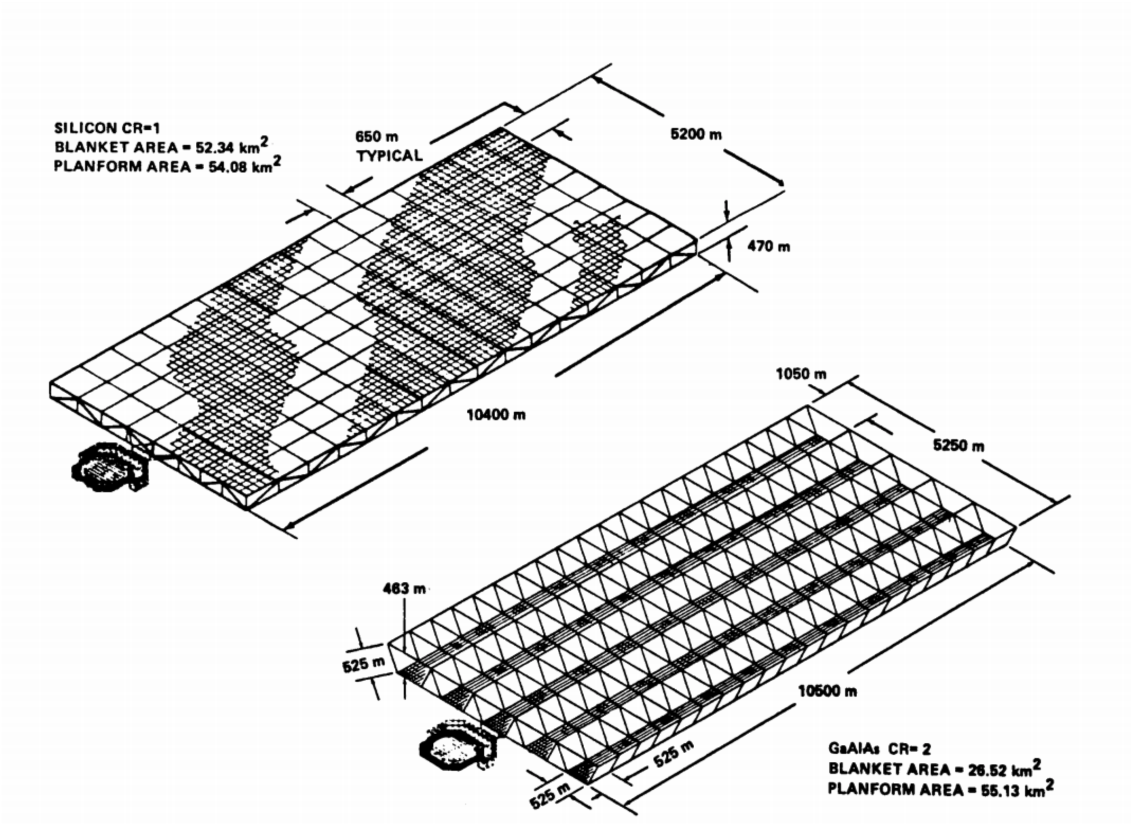


Figure 2.2 NASA DoE reference model, 1978

The reference system is designed for commercial output of 5 GW per year, and adding 2 SSPS per year will provide a 300 GW total capacity system with 2030 and the technology availability based on 1990. The total overall efficiency for the reference system is 7 %, with an EOL of 30 Years. The power density on the ground was set to a maximum of 23 mW/cm² at the center and one mW/cm² at the edge, which is thought not to cause nonlinear interactions in the ionosphere. The area of the power receiving rectenna is 10 km in diameter in the equator with an area of 78.5 km² [27].

Table 2.2. NASA DoE reference model constraints [27]

SSPS Design		
PV – solar radiation conversion	Silicon CR=1	GaAlAs CR=2
SSPS generation capability	5 GW	5 GW
PV efficiency at 28°C	17.3 %	20 %
Dimension (km)	10.4×5.2 Blanket area = 52.34 km^2 Planform area = 54.08 km^2	10.6×5.25 Blanket area = 26.52 km^2 Reflector area = 53.04 km^2 Planform area = 55.13 km^2
SSPS mass (kg)	51×10^6	34×10^6
Structural material	GFRTP	
Orbit	GEO	
MPT	Yes	
No of antenna	1	
DC-RF	Klystron	
Frequency	2.45 GHz	
MPT conversation efficiency	63%	
Rectenna dimension (km)	10×13	
Rectenna power density (MW/cm^2)	Center	23
	Edge	1
Transportation system		
Earth-LEO	Cargo (payload)	Vertical take-off, winged 2-stage (424,000 Kg)
	Personnel (Number)	Modified shuttle (75)
LEO-GEO	Cargo	Dedicated elect. OTV
	Personnel (Number)	2-stage LOX/LN 2 (75)

2.1.1.3. Fresh Look Study 1997-1999

The NASA project group studied a new approach to investigate and create a new SSPS system design. The report presents two unique designs with the 1990s technology; the Sun Tower and the Solar Disc.

2.1.1.3.1. Sun Tower

Sun Tower is an innovative design to lower the development and lifecycle cost. The design was first deployed in LEO and then transferred using elliptical Earth orbit. The design is capable of transmitting 100 - 400 MW using a single SSPS module. The Satellite constellation of MPT is characterized by its GHz frequency-based system, medium size, gravity gradient stabilizer, sunflower concept. The leaves are PV cells, and the Earth point face is the transmitter with a diameter of 26 m. A single transmitter cell is a hexagonal structure with a 5 cm diameter.

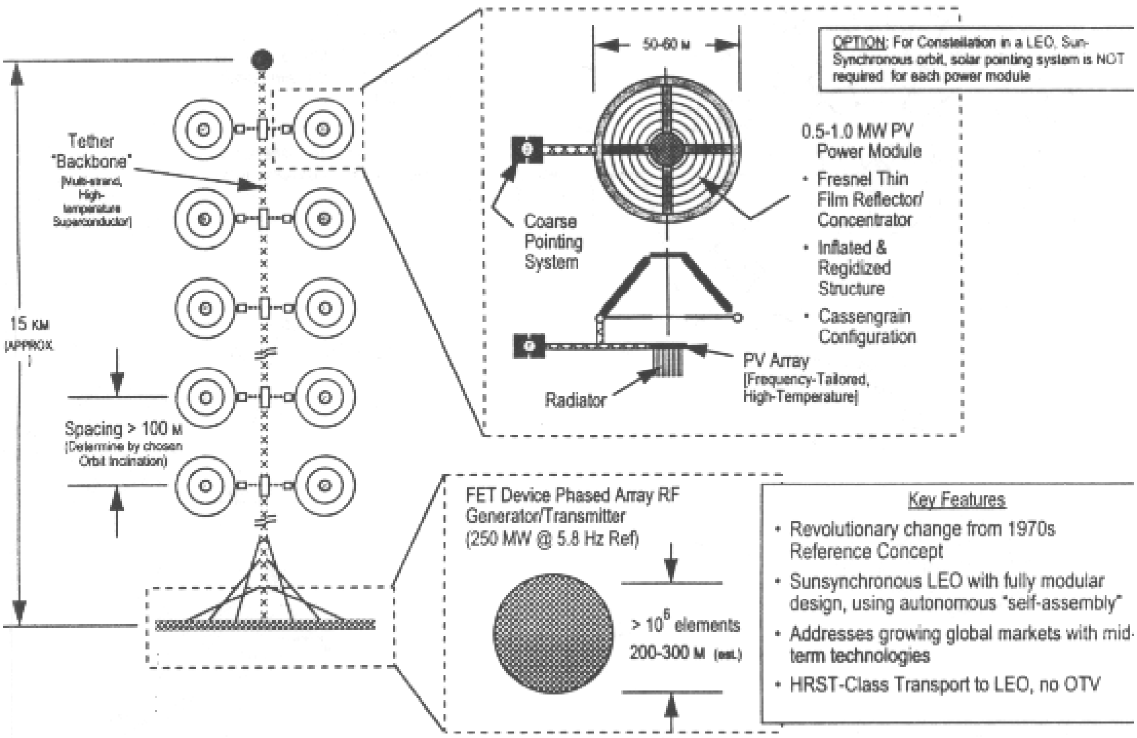


Figure 2.3 Sun Tower design by NASA

At 1000 km, SSO can transmit 200 MW with beam steering of 60° shown in Figure 2.3 and CAD in Figure 2.4, which is received by a ground receiving rectenna that is 4 km in diameter. The constellation design can transmit 3.5 - 4 GW with a 40-year lifetime, and will have18-24 launches every ten years to stabilize the satellite power system in orbit. The commercial cost of the first power unit is 6-8 bn USD, and the total constellation cost is 35 - 40 bn USD, with the average power cost versus price is 0.04 USD /kW-hr to 0.21 USD /kW-hr, respectively [30].

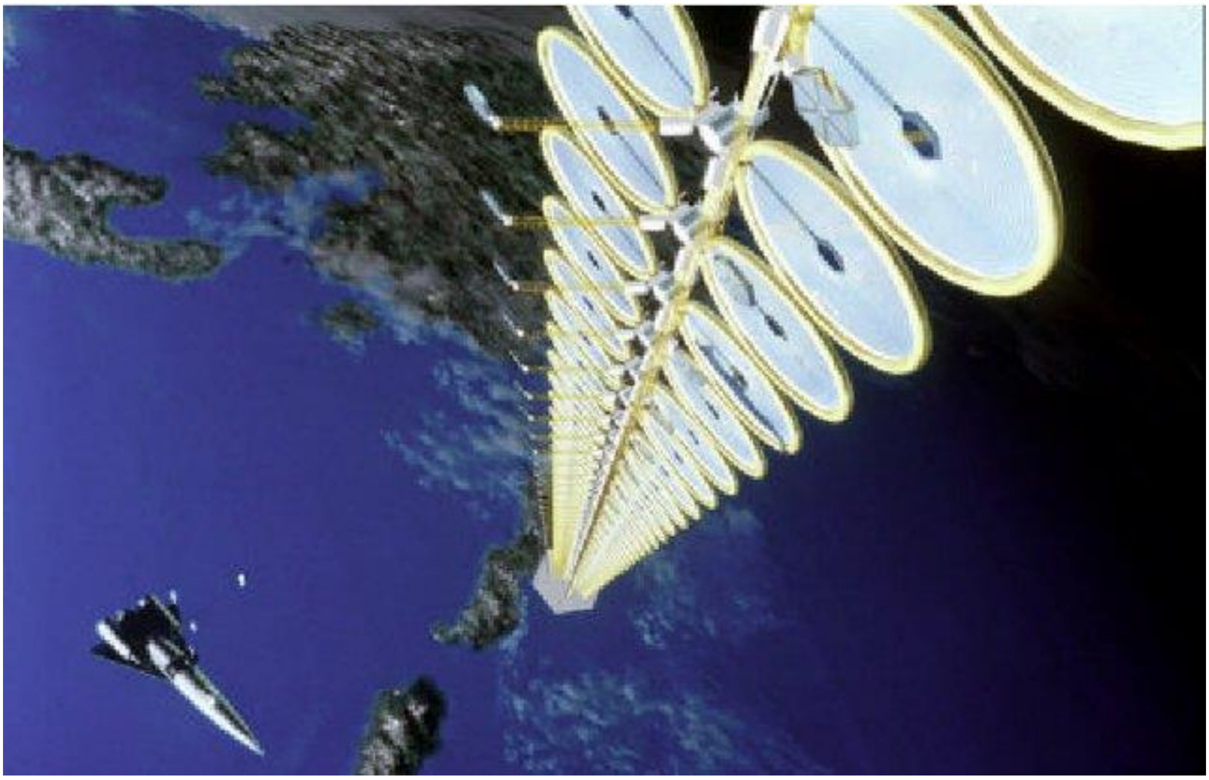


Figure 2.4 Conceptual design of Sun Tower

2.1.1.3.2. SolarDisc

This GEO-based SSPS system is a single-designed satellite to fulfill the energy demand that uses a frequency of 5.8 GHz for WPT to transmit 1 -10 GW from space to Earth. This disk design is 3-6 km in diameter, perpendicular to the sun site for maximum efficiency from Sun. The transmitter is 1 km diameter placed Earth pointing phased array. The beam steering capability is 10 degrees and the ground segment are approximately 5-6 km diameter in size, shown in Figure 2.5. [30]

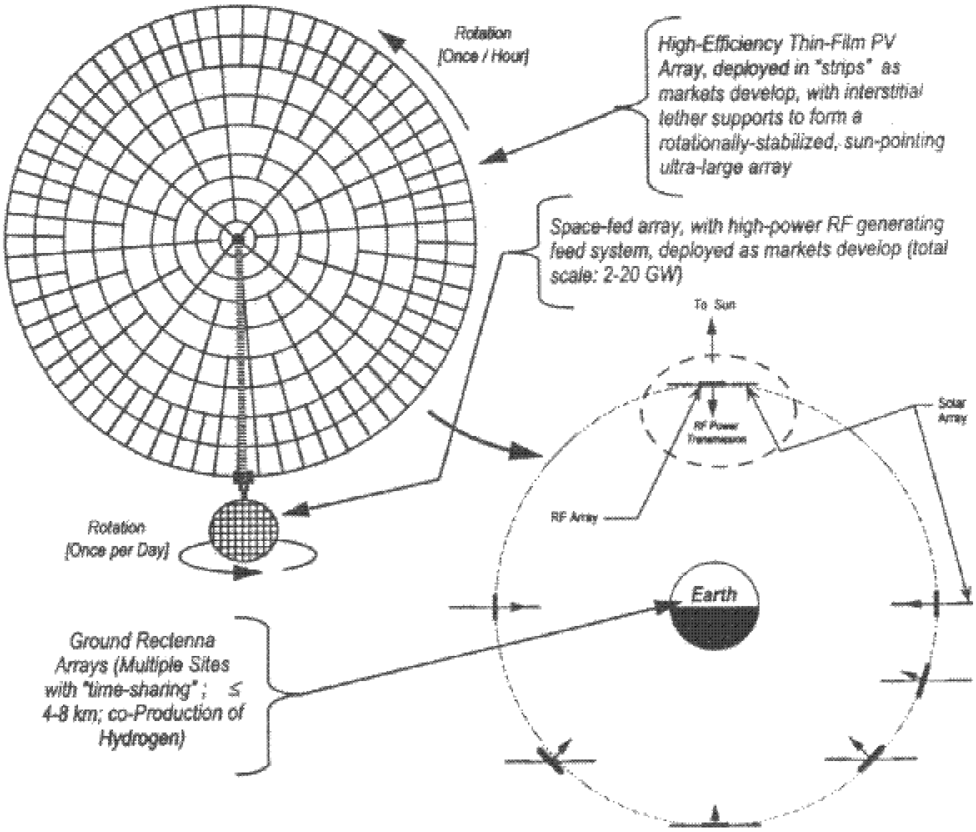


Figure 2.5 Conceptual design of GEO based SolarDisc

2.1.1.4. SERT SSPS Design 1999-2000

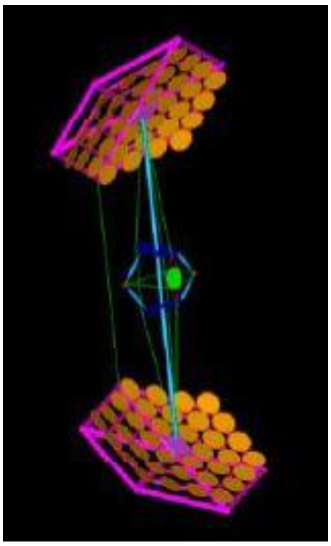


Figure 2.6 SERT SSPS design

The sunlight is collected in two "shells" and the central solar cell. The mast is in the average direction of the orbit. There are 24 mirror types with 2 × focusing and 36 mirror types with 4× focusing. The minimum weight is a 36-mirror type with 470 kg, which uses a magnetron and quantum dot solar cells. The diameter of the "seashell" is about 4 km, and the mast is 7.2 km. The initial weight in LEO is 31,500 tons, shown in the conceptual design in Figure 2.6 [31].

2.1.1.5. Solar Power Beaming Concept (2009)

The LLNL proposed as in the Figure 2.7, the Laser-based SSPS concept for 1 MW power transmission using a 795 nm diode-pumped laser. With 50 % electric to Laser conversion efficiency and 70 % light to light transmission efficiency, within a total packed volume of 2 m^2 × 4 m tall structure, this prototype would be easy to transport in space by Falcon Heavy. The total weight of the structure is 9125 kg. The system utilizes the reflector to concentrate solar power on foldable solar panels and a hydrogen generator, using a diode-pumped Laser to transmit energy from space to Earth [37].

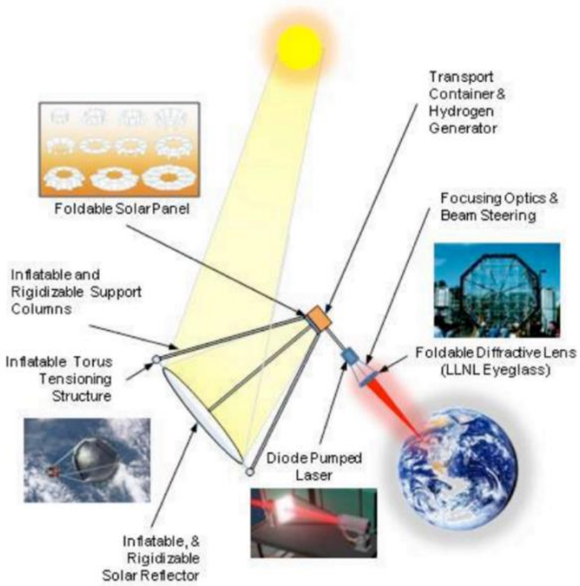


Figure 2.7 Solar Power Beaming Concept proposed by LLNL

2.1.1.6 SPS-ALPHA, 2011

Under the NIAC program, John Mankins designed SSPS which consists of a sizeable earth-oriented array, a sun-shielding reflector system that acts like a heliostat that follows the movement of the Sun and adjusts the orientation of the mirror, and a truss that connects them at GEO. The power generation unit is a fixed axisymmetric gravity tilt stable satellite, and the satellite does not incorporate 3-axis movements. The extensive array is a retro directive phased array with a coherent RF transmitter, as shown in Figure 2.8. The satellite has multiple joints and connectors, and every component is designed to be below 100 – 300 kg. The structure of SPS-Alpha is comparable to a HexBus, which has a diameter of 4m capable of wireless communication between systems and an average weight of 25 kg with more than 200,000 structures used for designing the system. A > 900,000 interconnects equal to 1 kg for connecting the system.

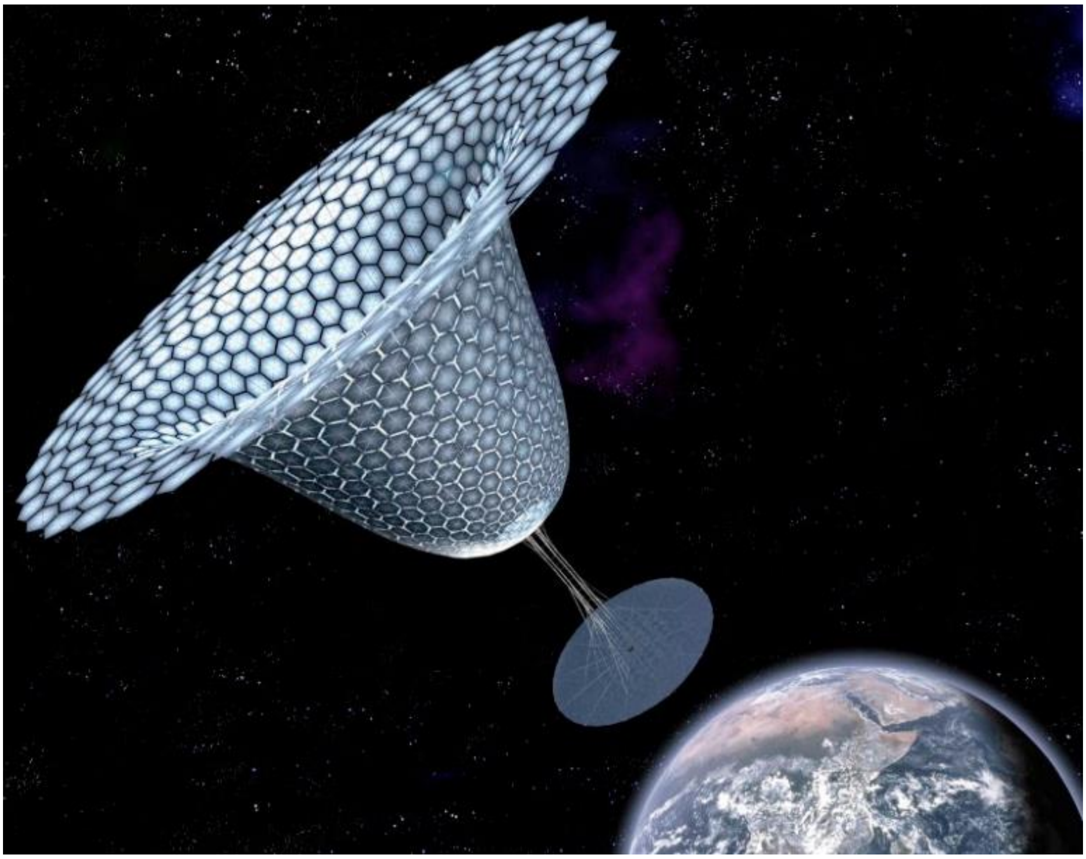


Figure 2.8 Conceptual design of SPS-ALPHA

HexFrame structural module is subsystem is deployable and expandable beam to provide the base for reflector and a connection system between reflector, power and transmitter system with 5,000 elements with 50 kg. The reflector subsystem is made from thin-film by Al on Kapton, with 4,000 – 5,000 elements weight around 75 - 100 kg. The WPT is combined with the HuxBus unit. HexBus is 200,000 – 300,000 elements with 50 kg each. The space assembly is constructed by rotating robotic arms with an actuation system with <5,000 elements with a weight of 10 kg. The propulsion, AOCS, and GNC system are between the weight of 50 – 200 kg, with each weight of 50 -500 kg depending on the propulsion mass [38]. The team investigated five different SPS-Alpha DRM as DRM-1, DRM-2, DRM-3, DRM-4, DRM-5. Table 2.3 shows all DRM modules with specification, cost, and efficiency models.

Table 2.3 Specification of SPS-ALPHA DRM system

Parameters	**DRM-1**	**DRM-2**	**DRM-3**	**DRM-4**	**DRM-5**
Orbit	LEO	LEO	GEO	GEO	GEO
Altitude	SSO 900 km	SSO 900 km	35,800 km	35,800 km	35,800 km
Objective	Technology flight demonstration	Moderate Power technology demonstration	Pilot plant	Full-scale plant	Large scale power plant
Total Platform Hardware Mass (kg)	12,108	16,768	232,610 - 1,045,252	11,795,271	34,813,882
Power (MW)		.200	2 - 18	500	2000
Launching					
Frequency (GHz)			2.45 GHz	2.45 GHz	2.45 GHz

Table 2.3 Specification of SPS-ALPHA DRM system (continue)

Parameters	DRM-1	DRM-2	DRM-3	DRM-4	DRM-5
WPT Efficiency DC-RF			70%	70%	80%
Cost to First Power (USD)			~ 4.5 Bn (~250 per W)	~12.2 B (~24 per W)	31 B (~16 per W)
Lifetime (Years)			10	>30	45>>30
LCOE (USD per kW-hr)			~ 3.26	~ 0.15	~ 0.09

2.1.2. Japan

2.1.2.1. Japan's SPS 2000 Project

Japanese space engineers and scientists started designing SSPS systems as early as the 1980s. In 1990, the Japanese government reformed its New Earth 2s objective to solve the world energy problem to respond to global climate change. The ministry started to investigate an alternative method for energy production with nuclear energy progress. They initiated a project group dedicated to SSPS systems with 13 sub-group members, 80 engineers, research scientists, and academicians from ISAS and Japanese universities. In 1994 a fully functional paper was published as SPS 2000 [45].

A study report conducted by RCAST of Tokyo University and Keio University (1996-1997) compares the current energy production using coal, oil, nuclear with SSPS to showcase the tackle down of global climate. According to the report that factors in SSPS to the total energy generation, the total CO^2 (grams relished per unit of one Wh) by the power plant will drop to 1/30, with individual values being coal- 1225, petroleum – 846, LNG- 631, nuclear – 22, SSPS – 17 [46]. Building the SPS 2000 will cut down the cost of power production and transmission from Japan, a global exporter of power transmission, as mentioned in a report by

RCAST of Tokyo University. By deploying rectenna over developing countries will slow down the production of global greenhouse gases through this shared technology increasing Japanese diplomacy and leadership worldwide [45].

A nuclear power plant has a 34 - 40 % thermal efficiency, while MPT-based SSPS has 80-90 % conversion efficiency at rectenna, and LPT-based SSPS has 40-60 %. The transmission of power does generate a heating effect, which translates to transmission losses [45]. The SPS 2000 pilot project is created for the output of 10 MW using 2.45 GHz from 1100 km at LEO shown in Figure 2.9. To supply 200 – 300 kW of electricity to small villages in developing countries located on the equator.

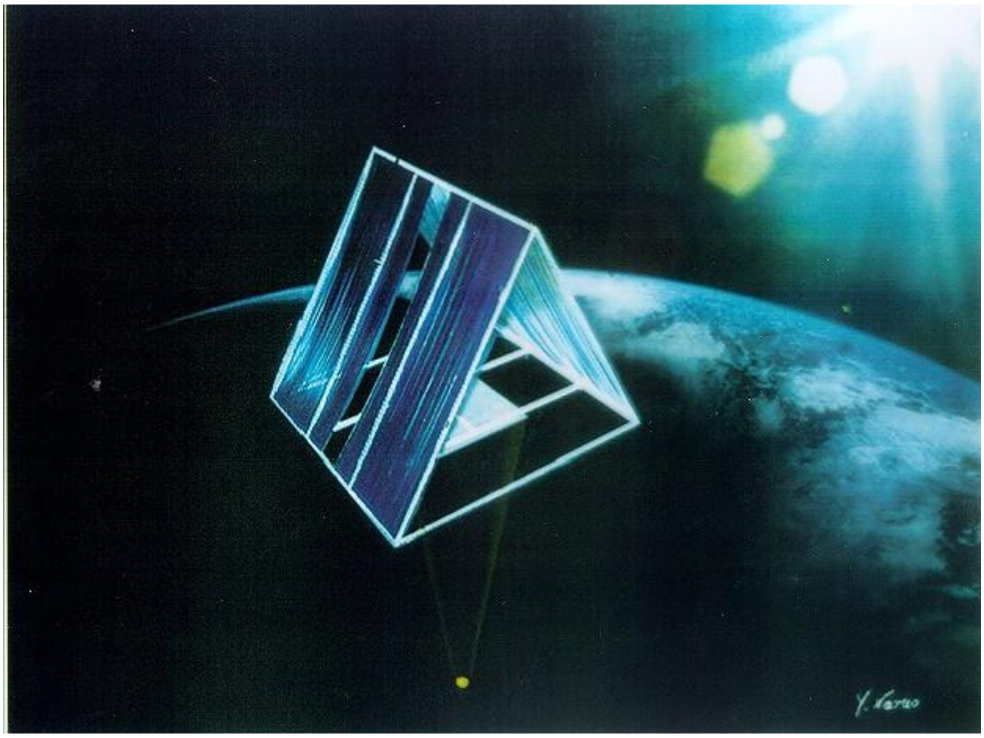


Figure 2.9 A conceptual design of SPS2000 by Japan

Researchers developed the SPS 2000 mainly and investigated across six countries independently (Papua New Guinea, Brazil, Ecuador, Tanzania, Maldives, and Indonesia) [45]. The full-scale model is for a 5 GW power transmission with a 50,000 tons SC. In the power generation, thin-film Si-PV cells with 15 % were used for transportability in space. The specific power is 1 kW/kg, and the bus voltage of 1 kV. The system comprises 110 subarrays composed of 12 solar cells on each with 180 A at 1 kV. A 45-array module with an approximate weight of each is 270 kg attached to each wing of prism structure. The structure size is 336 × 336 × 303 m with an antenna of 132 × 132 m in space and a ground-based rectenna of a diameter of 2 km [45, 47].

2.1.2.2. The Japanese Version of SPS (NEDO Sunshine Project), 1994

NEDO proposed its SSPS design in 1994 with the NASA reference model. The project uses MPT of 2.45 GHz frequency for 1 GW from a GEO-based Satellite. The satellite uses a two 3.2 × 2 km solar panels to generate electricity and transmit using a 1 km dipole antenna with a reflector weighing 26000 - 27000 tons, including 7400 tons. A conceptual design is shown in Figure 2.10. [29, 48].

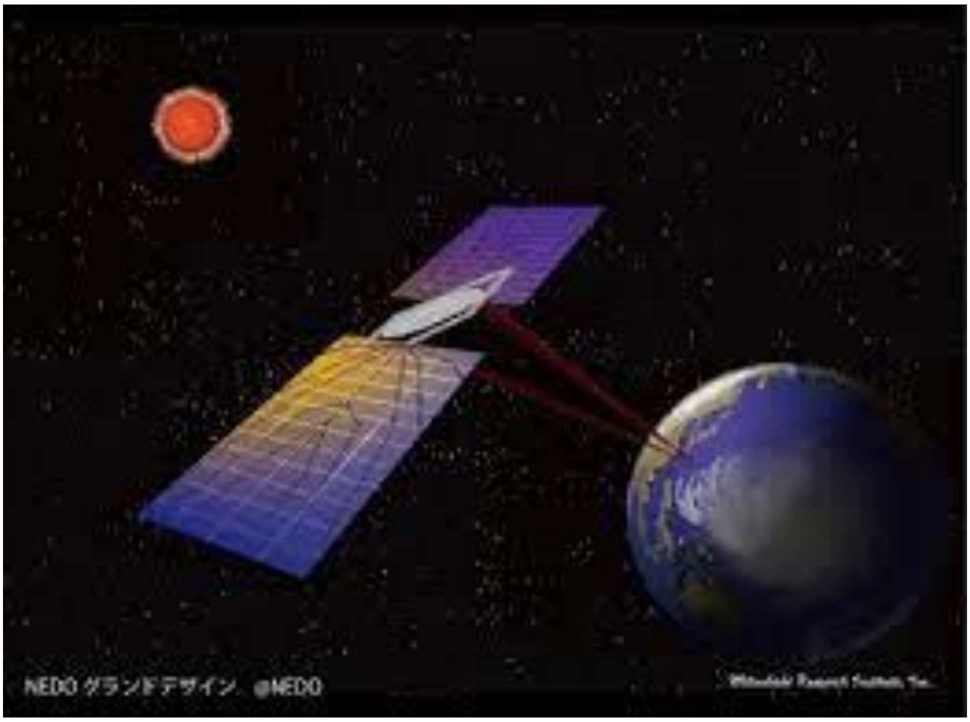


Figure 2.10. Conceptual design of NEDO Sunshine project

2.1.2.3. SSPS JAXA Model, 2004

The SSPS Review Committee makes the SSPS JAXA model of NASDA (later JAXA) during the period 1998 - 2008. A 2.5 km × 3.5 km primary mirror collects sunlight four times, and a solar cell with a diameter of 1.25 km generates electricity. It transmits 1 GW at the frequency of 5.8 GHz from a phased array with a diameter of 1.8 km, shown in Figure 2.11. The power receiving site, rectenna with a diameter of 2.74 km. The power generation unit and the power transmission unit are separated from securing a heat exhaust surface. The primary mirror is characterized by flying in formation independently of the power transmission department [33].

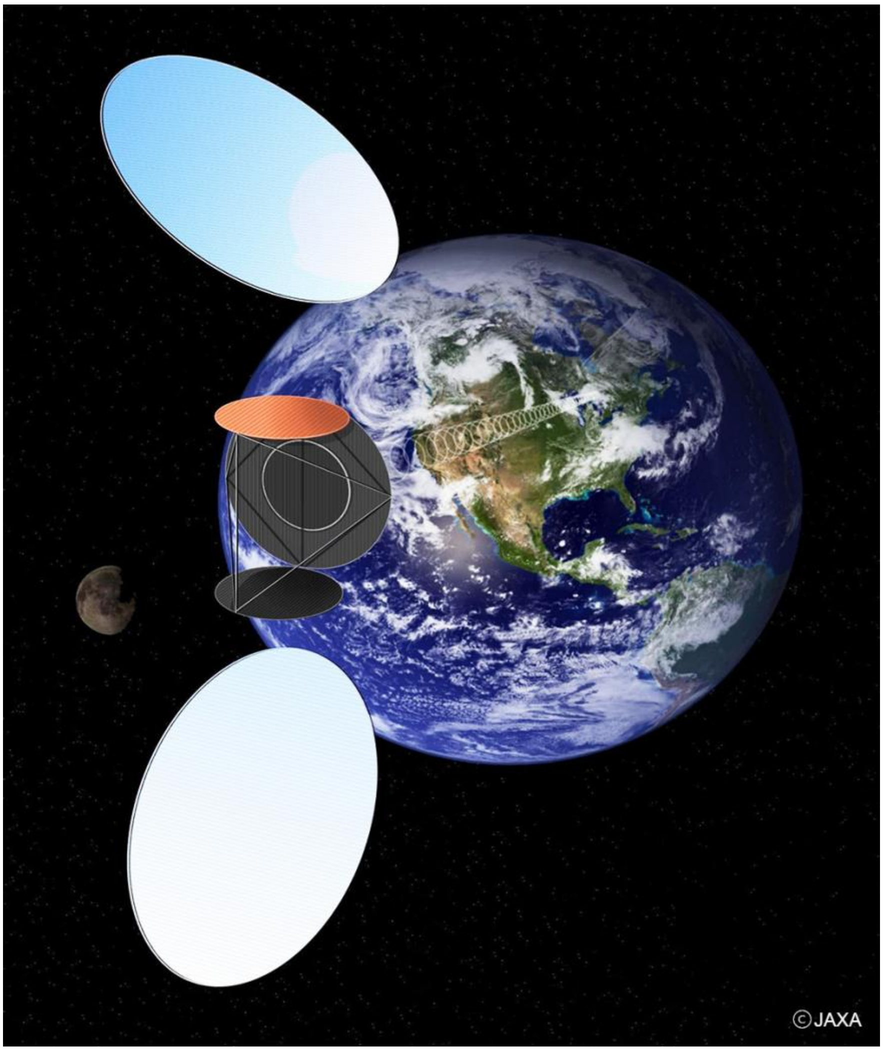


Figure 2.11 SSPS JAXA model

2.1.2.4. SSPS USEF Model, 2006

The Japan Space Systems Research and Development Organization proposed a new SSPS model created by the expert committee of 2001-03 as part of the research on SSPS conducted under the commission of the Ministry of Economy, Trade, and Industry. A gravity inclination posture stabilization method connects a substantially flat panel with integrated power transmission and power transmission and a distant bus part above it with a tether [49]. In watts class power transmission, the power transmission panel measures 2.6 km × 2.4 km, and the upper surface consists of solar cells, and the lower surface consists of phased array antennas and solar cells. It is hung from the bus section by multiple tethers of about 10 km and consists of sub-panels with a thickness of 10 cm and 100 m square shown in Figure 2.12. The diameter of the ground rectenna is designated as 3.5 km [35].

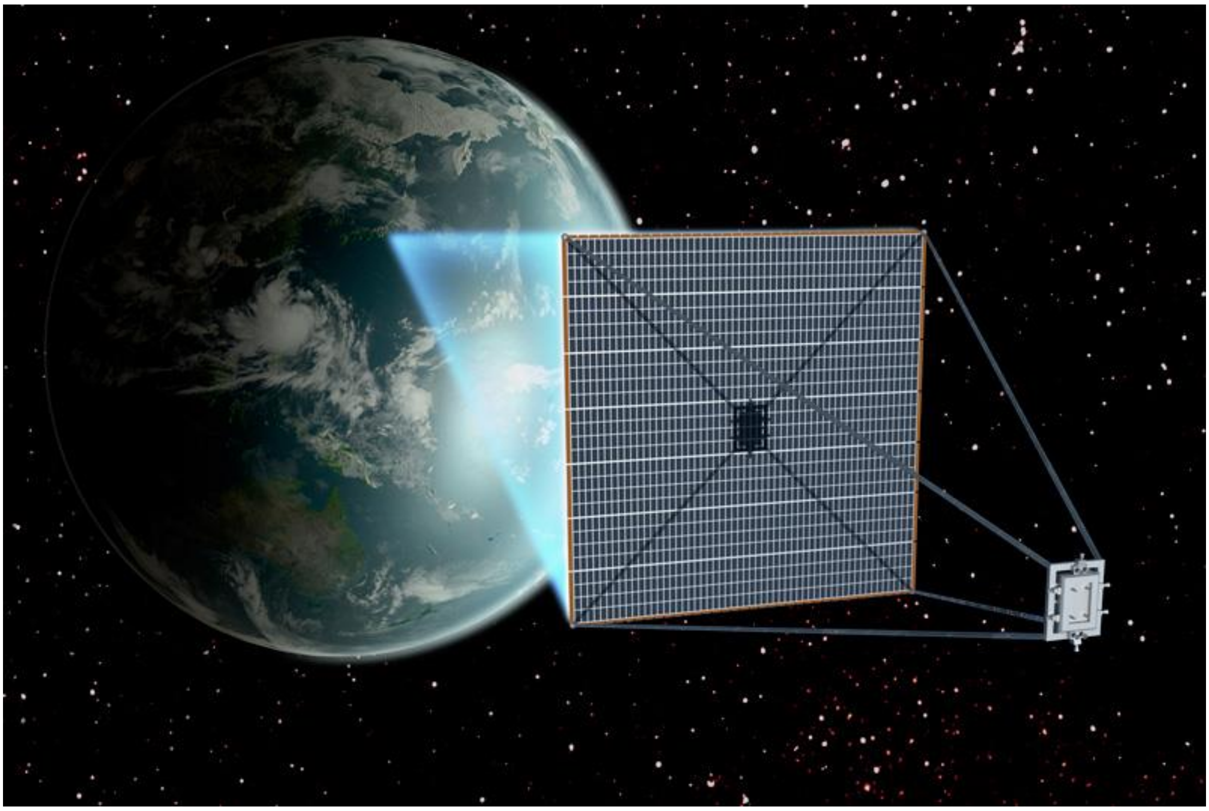


Figure 2.12 SSPS USEF model

2.1.2.5. JAXA L-SSPS Model, 2006

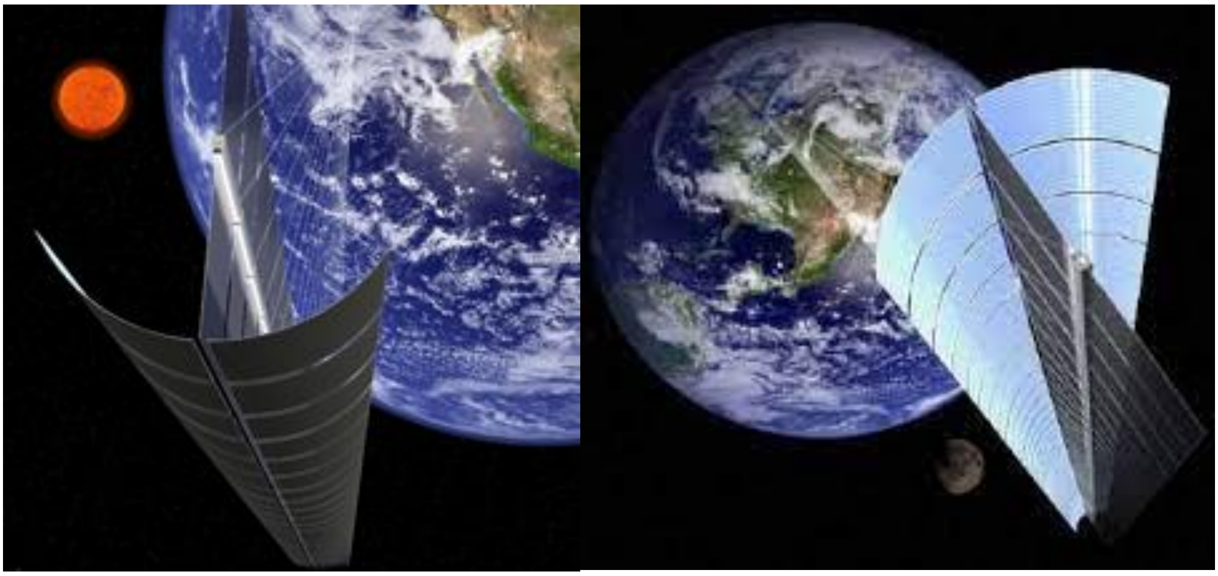


Figure 2.13 JAXA L-SSPS model

This model has a laser output of 1GW. It connects 100 lasers vertically with 10 MW output lasers with a width of 400 m, a depth of 300 m, and a length of 120 m by Nd-YAG & Cr-YAG laser [36] shown in Figure 2.13.

2.1.3. European Union, Sail Tower SPS, 2001

ESA proposed an SSPS concept like sun tower with new technology as 60 pairs of "sails" of 150 m^2 are solar cells with an output of 3,700 kW, totaling 450 MW. It is a geostationary satellite that uses gravity tilt stability, shown in Figure 2.14. It generates microwaves with 400,000 magnetrons for microwave ovens with a frequency of 2.45 GHz and an output of about 1 kW, transmits electricity from an antenna with a diameter of 1 km, and receives energy with a rectenna with a diameter of 10 km [32].

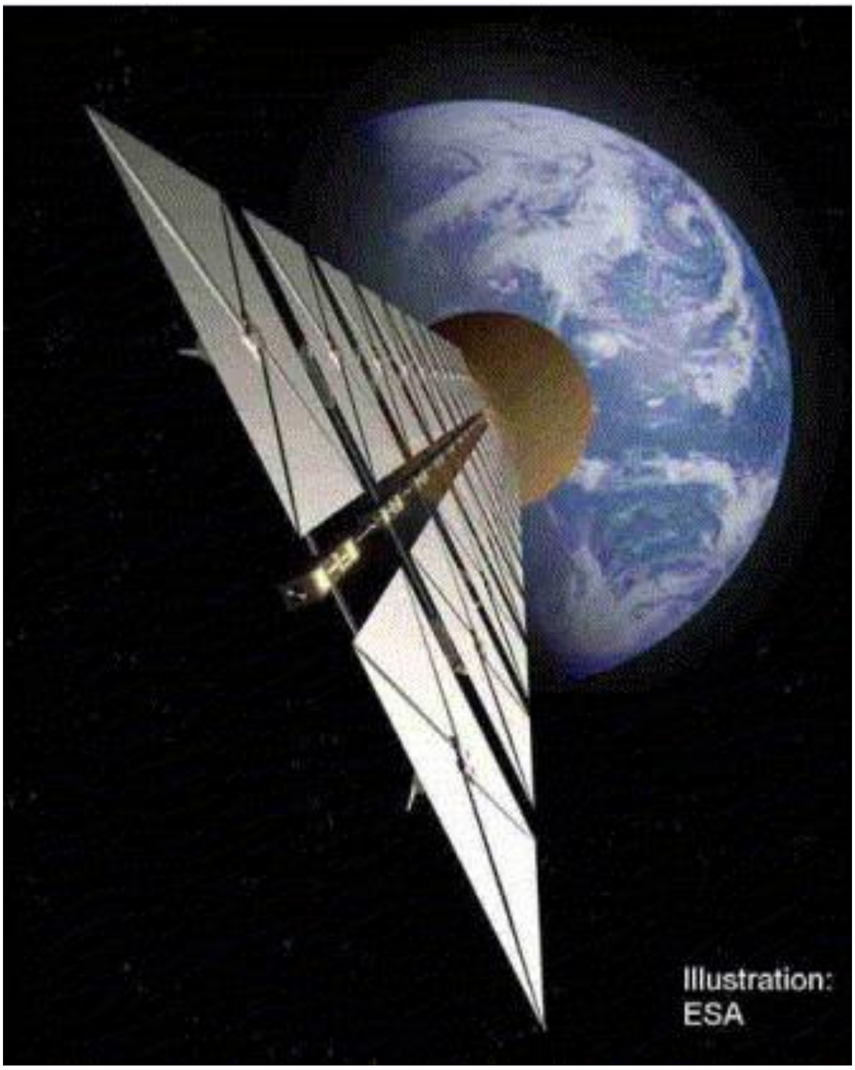


Figure 2.14 Sail Tower SPS

2.1.4. China

2.1.4.1. Multi-Rotary Joints SSP (2015)

China Academy of Space Technology proposed an MPT-based SSPS in 2015 to deliver 1.2 GW of power from GEO based system. The architecture consists of multiple PV in two directions to transmit electricity with a 5.8 GHz frequency using a microstrip antenna. The thin-fil, GaAs-based PV cells with 40 % efficiency will be used for power generation, and the total power output will be 2.4 GW. The MPT system efficiency is estimated at 54 % with a 1 km diameter antenna, consisting of 128,000 antenna module elements capable of transmitting

12.5 kW per module. The ground base receiving system is approximately 5 km in diameter, as shown in Figure 2.15. The total efficiency of the system is 13 % with 30 years lifetime. The 1 N thrust will maintain the AOCS for orbit keeping [39].

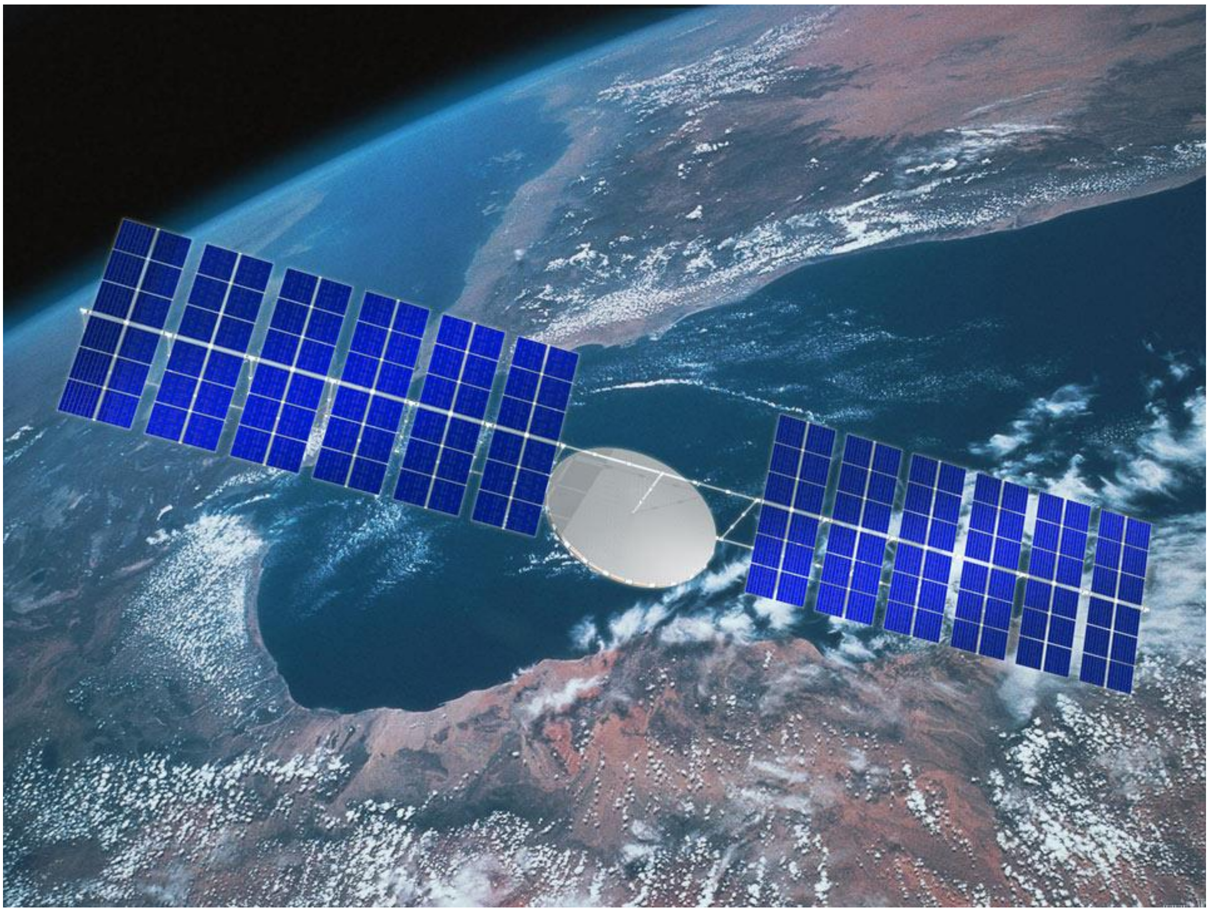


Figure 2.15 Proposed design by CAST: Multi-Rotary Joints SSP

2.1.4.2. SSPS OMEGA [41]

The Xidian University of China proposed a Space Solar Power Station via Orb-shape Membrane Energy Gathering Array for a 2 GW, 5.8 GHz MPT-based spherical solar power collector SSPS design. The PV cells are arranged in a hyperboloid structure, and the reflector can adjust to concentrate the sunlight inside the sphere for a high concentration of power. A one km diameter transmitting antenna is used for power transmission from GEO to Earth-based receiving rectenna, as shown in Figure 2.16.

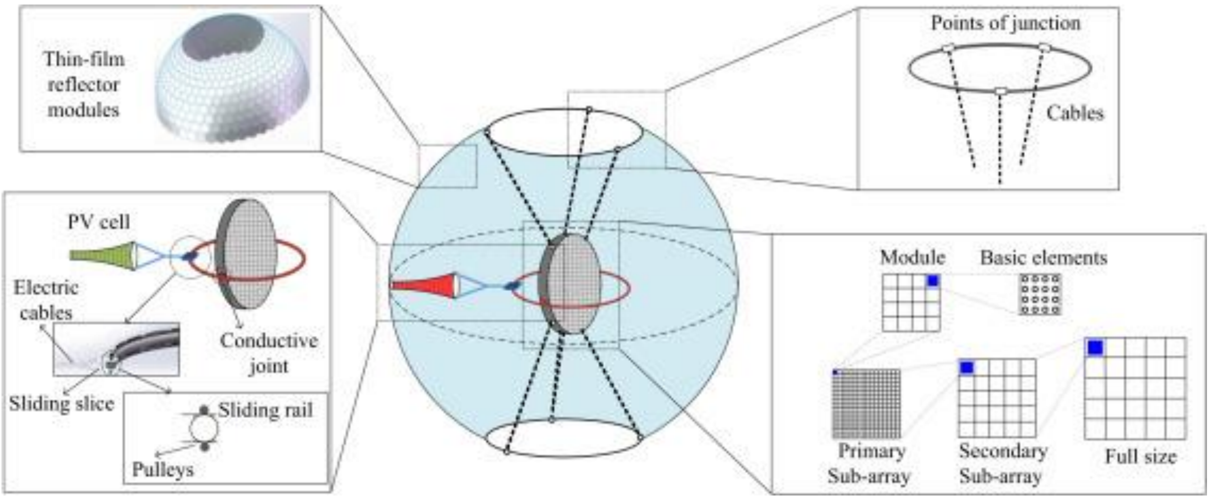


Figure 2.16 A conceptual Design of SSPS OMEGA

2.1.5. United Kingdom, CASSIOPeiA, 2017

This is a design by Ian Cash, with MPT and three different variants with 688, 974, 1310 MW respectively weight of 866, 1020, 1280 tons. It uses an SSO elliptical orbit at an inclination of 116.6 °, the design in Figure 2.17.

Figure 2.17 CASSIOPeiA design

The concept can manage cosine losses of solar energy collection, and the concept features no moving parts. The PV cells are located on the helical-shaped structure and provide a continuous power system in orbit with continuous power towards Earth using an MPT-based system [42].

2.2. Wireless Power Transmission

WPT technology has undergone several enhancements in recent history. Nikola Tesla designed the Wardenclyffe Tower, also known as the Tesla Tower, one of the earliest experiments to prove potential WPT [12, 50]. In 1973, Dr. Peter Edward Glaser coined the idea of an SSPS capable of transmitting power via microwaves from a GEO to a ground microwave collector station on Earth [7, 17]. Microwave energy transmission can be conducted at all frequencies above 1 GHz. An optical lower band is preferred because of its efficiency to transmit through the atmosphere, rain, and a gray area. [51].

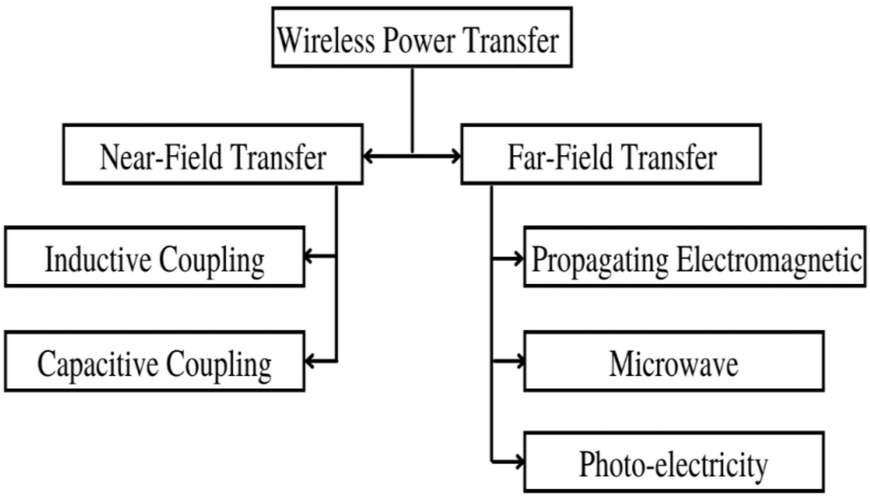


Figure 2.18 WPT method via distance base

The effective range of different methods varies from a few centimeters to a few meters, as shown in Table 2.4, which compares technology's range, efficiency, size, and cost to develop.

This WPT technology is currently used in mobile charging. In addition, the investigation and research on the far-field transmission, long-distance SSPS system, including MPT, LPT, electromagnetic waves, are underway as shown in Table 2.4.

Table 2.4 Comparison of different WPT techniques [54, 55]

Parameters	**Inductive Coupling**	**Microwave**	**Laser**
Distance	Few millimeters	Up to 100s km	Few meters can be increased by highly intensive beam
Transmitted Power	~ few W	~100s MW.	~100s MW.
Efficiency	Low	High	Medium
Penetration (clouds/ fog/ rain)		Very High	No
Aperture size (transmitter and receiving antenna)	Small	Large	Small
Cost	Economical	Expansive	Economical
Safety (Biological point of view)	Safe	Dangerous due to radiation but controllable	Dangerous

Most of the SSPS studies conducted by space agencies and private entities are based on MPT. It can transmit energy in cloudy or remote regions with higher transmission efficiency, an effect of extensive research in this area compared to LPT. In such a system, the PV cells generated power in DC and converted it into microwaves using magnetron and klystron with high beam efficiency.

The Laser of an LPT can be generated from the solid-state laser diode and solar pumping laser generator. LPT undoubtedly shows the highest prospects for long-distance, slight beam divergence with high-energy transmission [56]. The receiving and transmission antenna of an LPT system is comparatively smaller than the traditional MPT system. Therefore, the LPT system can deliver energy longer distances than any other WPTS [57-59].

The most straightforward and popular design among the SSPS community is the sandwich architecture, where one side is a power generation unit, in center electronics and conversion unit, and another side is a transmission unit. The Caltech Space Solar Power Project is one example [60]. The SSPS avionics and structure design define the size of the structure. The use of ultra-light material can reduce the weight and cost of the system [50]. Using advanced PV and CMOS integrated circuits provides a perfect DC to RF conversion and phased control for better transmission [61]. The SSPS structure should be designed with the sustaining force, torques, and vibration frequency below 0.1 Hz [62]. The transmission system is a significant component in SSPS design. Researchers are focusing on two main types of transmission systems, MPT and LPT, which will be extensively discussed throughout chapters 2 and 3.

2.2.1. Microwave Power Transmission

The MPT system is an RF-based system, using an ISM band and much research has been done to confirm its safety. Most MPT proposed systems are with 2.45 GHz - 5.8 GHz frequency. The structure of the transmission antenna and rectenna is also dependent on the frequency. The MPT system utilizing higher frequency gives advantages by increasing efficiency and reducing the size of the transmitting antenna. The transmission system combines several antennae with a controlled phase to achieve a narrow beamwidth. The Friis Equation is used to calculate the efficiency of WPT between transmission and rectenna [62]. From a system standpoint, the critical derived parameter for the satellite is the specific power, S. S is the RF power radiated at the satellite per kilogram of mass in orbit (W/kg). The larger S is, the less mass is required to obtain a fixed amount of radiated power. Our goal is to maximize S while minimizing weight and managing risk and cost. S is estimated using Equation (2.1). The link budget for the SSPS system is modeled using a modification of the Friis formula [63,64].

$$S = \frac{\eta_{PV}\,\eta_{DCRF}\,\eta_{Tx}}{m_{SV}} A_{pV}\;(AM0) \tag{2.1}$$

$$P_r = [\eta_{PV}\,\eta_{DCRF}\,\eta_{Tx}\,\eta_{dif}\,A_{PV}\;(AM0)]\left(\frac{f}{cd}\right)^2 A_{PV}\;A_r \tag{2.2}$$

The receiving power will be shown as power with the cost-effective system using Equation (2.2). Two more efficiencies are used to calculate the DC to RF and RF to DC and if the AC instrument or payload will utilize DC to AC conversation efficiency. Recently, Naval Research Laboratory demonstrated using Photovoltaic Radiofrequency Antenna Module Flight Experiment (PRAM-FX) experiment in orbit as a sandwich model space solar architecture. The experiment generated 8.4 W RF power with 37.1 % DC to RF conversion efficiency, and the total module efficacy was 8 % [54].

The electric energy coming from the PV array is managed correctly and rectified with a power management system and hardware section. Successful conversion of DC-to-RF will be achieved and transmitted to the daughter satellite using MPT for phase 1 of E-Orbit, the schematic diagram of the MPT transmission system. The power density received at the center of the rectenna can be defined by Brown and Eve's Equation (2.3).

$$\mathrm{P_d} = \frac{A_t\,P_t}{(2\,\lambda\,D)^2} \tag{2.3}$$

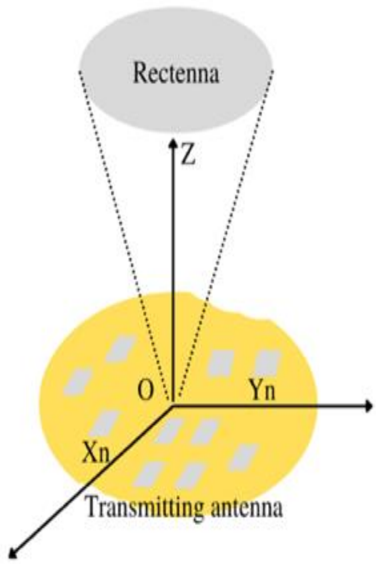


Figure 2.19 The geometry of transmitting antenna and rectenna (65)

P_d is power density at the center of the rectenna; P_t is the total radiated power from the transmitter (all power in kW). The transmitting antenna has a high beam efficiency, as shown in the geometry of both antennas in Figure 2.19, obtained from the BCE Equation (2.4). Therefore, the BTE can be calculated by Equation (2.5). On the other hand, the total MPT of a system can be calculated using both equations [65, 66].

$$BCE = \frac{P_{\Psi}}{P_{\Omega}} \tag{2.4}$$

$$BTE = \frac{P_{\Omega}}{P_{\Psi}} \tag{2.5}$$

In the above equations, P_{Ψ} is the power radiating over the angular region, and P_{Ω} is the total power transmitted over the visible region. The antenna can generate various types of errors due to vibration, high-temperature change, radiation. These errors can be minimized during launch and by maintaining the orbit.

2.2.2. Laser Power Transmission

Research for LPT is concentrated on fueling the interplanetary, habitat, and orbiter systems. In any WPT system, frequency denies the transmission's efficiency, size, and reception aperture. By this definition, mm-wave, which uses 35 GHz frequency, has higher transmission efficiency than an MPT system, which uses 2.45 and 5.8 GHz [19]. On the other hand, LPT has reduced efficiency, despite using an even higher frequency than mm-wave and MPT. Thereby, LPT is principally used to avoid the drawbacks of MPT, which consist of side lobes/spikes, difficulty controlling during failure, and the high mass and size requirements of transmission elements [18].

The LPT system is optical-based and has more remarkable ability to deliver long-distance and large amounts of power to a small aperture than the MPT system [59, 68, 69]. There are different types of lasers classified by design, component, and power. For example, CO_2 gas-dynamic (λ = 10.6 um), HF/DF Laser (λ = 2.41 – 3.38 um) gives 2 MW output using Middle Infrared Advanced Chemical Laser (MIRACL using Deuterium Fluoride). Chemical Oxygen Iodine Laser (Coil: λ = 1.315 um) is a type of chemical gas dynamic laser, also called Solid-state laser (Nd:YAG, Ti:sapphire, Er:Fiber (erbium-doped optical fiber)). Laser diode (HAMAMATSU: 1200 W CW) has lower quality compared to solid-state Laser. However, the system is with high heat load and lower efficiency compared to the laser diode. [68]

In 2002, EADS-SPACE Transportation (EADS-ST) demonstrated a 532 nm wavelength with an output power of 5 W Nd:YAG laser to a rover within 280 m [56], JAXA demonstrate LPT using a 200m high tower to transmit a 500 W with an accuracy of 1 rad. NASA, Russia, China, Lockheed Martine, and LaserMotive, Inc. conducted a series of experiments for long-distance transmission to consider high output efficiency. The available laser technology within the range of 780 and 1100 nm is the best choice for long-distance with sufficient power [70]. The diode-pumped solid-state Laser as fiber and disk laser have high advantages with high power and beam quality [71-73]. Moreover, compared to the diode-pumped solid-state laser, the laser diode is less expensive, has higher efficiency, and is more suitably compact for short distances because it produces less bright beams. Equation (2.6) shows the equation used to select available Laser for practical purposes with a high bright enough beam to transmit power for long-range

$$\phi = \frac{R_{source}\, A_{source}\, n_{trans}}{d^2} \tag{2.6}$$

Laser communication is already extensively investigated, researched, and currently in use for inter-satellite linking. The low latency of Lasers allows the system a continuous fast medium for transmissions. Likewise, LPT reduces the size of any transmission and reception system. The most efficient DC-to-laser converters are solid-state laser diodes commercially employed in fiber optic and free-space laser communication, as demonstrated in the following examples [74, 75].

- F. Steinsiek and his team at space Power infrastructure of EADS-ST demonstrated a laser operated rover by using Nd: YAG laser to pinpoint pilot connection for continuous power supply of a few W within 280 m, as shown in Figure 2.20.

Figure 2.20 EADC-ST demonstration of Laser power rover (source: EADS-ST)

- JAXA-based SSPS team demonstrated with a series of experiments in 2012 - 2013 the high accuracy beaming controlled by atmospheric disturbance performed at 500 m using the horizontal LPT test with accuracy of 1 μrad as in Figure 2.21. In another test, JAXA performs a 200 m high tower to mimic the space and Earth base station as the L-SSPS. Furthermore, they are expected to achieve a main laser output power of 500 W with the target of an accuracy of 1 μrad.

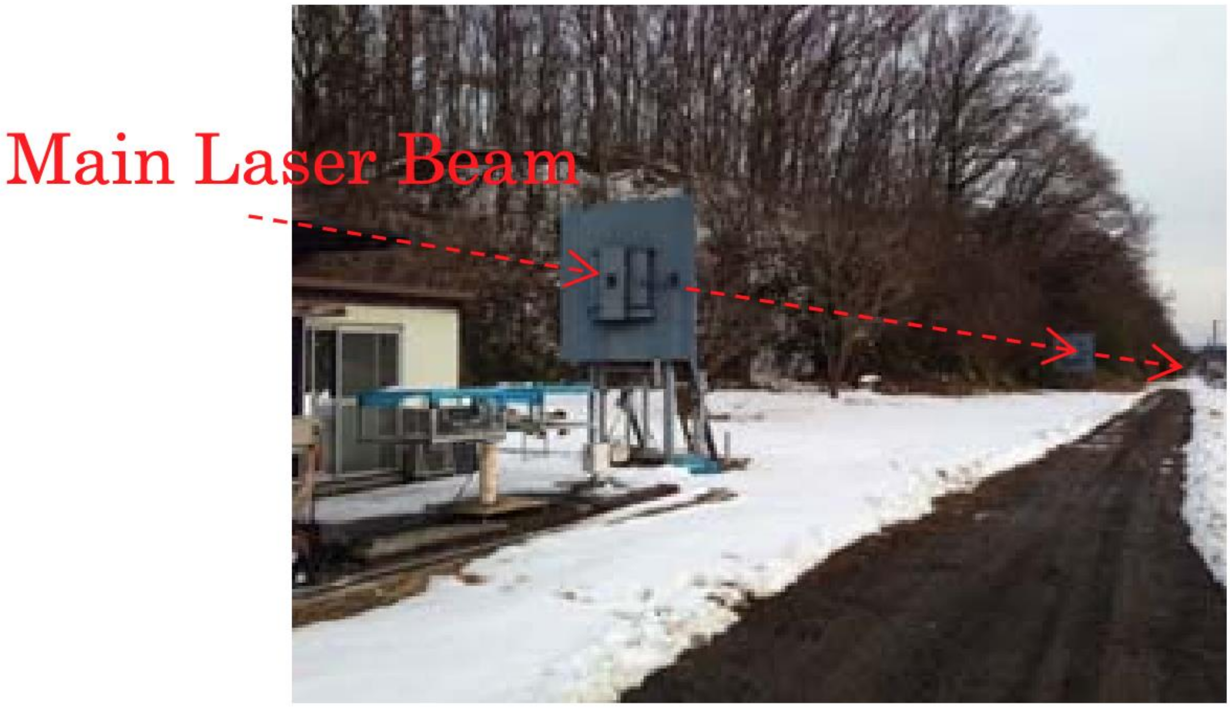


Figure 2.21 A 500 m horizontal laser transmission test at JAXA Kakuda Space Center (source JAXA)

- The United States Naval research laboratory demonstrated a free-space power beaming system using two 13-foot-high towers, one being a 2-kW laser transmitter, as in Figure 2.22, and the other, a receiver of specially designed photovoltaics. The laser beam of 400 W of power can travel across 325 m [59,76,77].

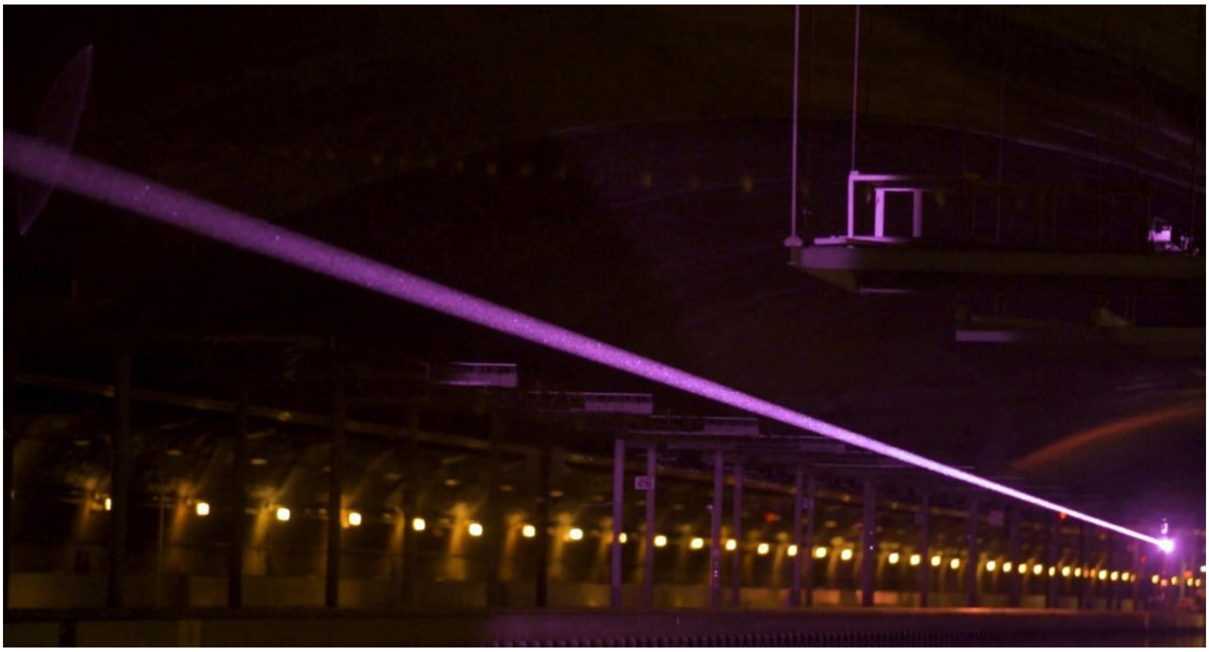


Figure 2.22 2 kW Laser Power Transmission demonstration by USNRL

For LPT, laser selection is significant as it should be small and low in weight while allowing a high-power output with low dispersion of heat and temperature concerning low mass and small size and with a high-quality laser beam to achieve at small receiving surfaces and control of phase [78]. While using a Laser, safety is prioritized. According to the International Electrotechnical Commission, laser types can be classified into seven distinct classes, including the subcategories from class 1 to 4. The class four Laser is deemed hazardous, with a high risk of eye damage. However, in a highly controlled system, it can transmit hundreds of W from a single pointer. The Sandia National Laboratories USA researcher analyzed the LPT system to acquire an output power of 370 W, and power at the load is equal to 35.06 W/cm^2 [79]. This indicates that a lower divergence beam is preferred for optimate laser transmission performance. The divergence of laser beams calculated by the Gaussian Beams is as shown in Equation (2.7).

The angle of divergence of the beam

$$\theta = 2\lambda \, \pi \, R \tag{2.7}$$

The final divergence (D) of the Laser beam can be calculated as in Equation (2.8).

$$D = 2 \, d \, tan\left(\frac{\theta}{2}\right) \tag{2.8}$$

Using the above calculation, the power density at any load surface can be calculated using Equation (2.9)

$$Power\ Density = \frac{P}{\left(\pi \left(\frac{d}{2}\right)^2\right)} \tag{2.9}$$

Here R is beam radius, is a wavelength, d is the beam diameter of Laser [79]. The selection of the Laser is essential with the output power and the wavelength because there are different types of lasers using different wavelengths, as shown in Figure 2.23. The laser selection for effective transmission uses 0.8 um 1.64 um lasers.

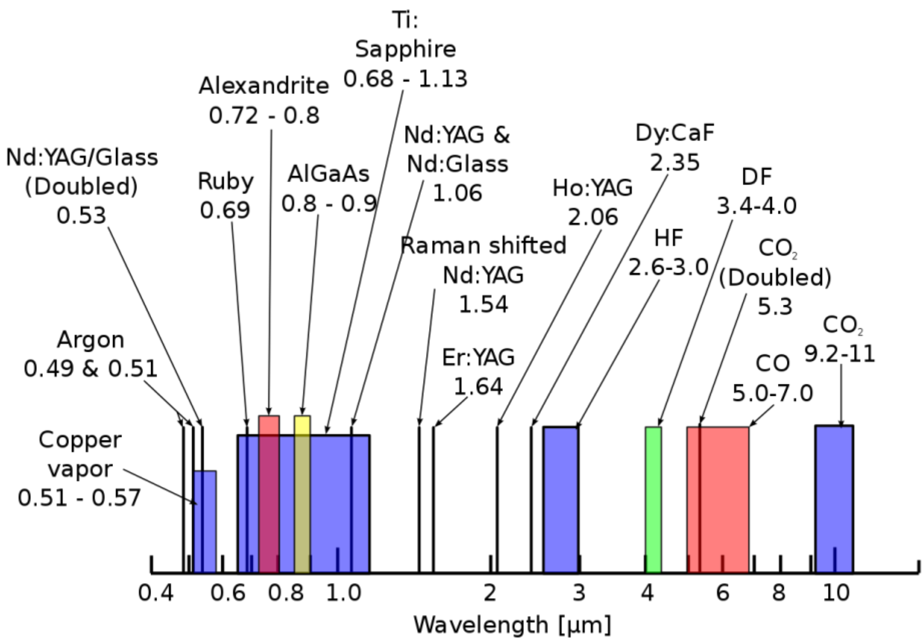


Figure 2.23 Spectral output of different Laser (Source: [79])

The most commonly used Laser being Nd: YAG laser, consisting of a diode with a plugin efficiency of up to 80 % at 795 - 850 nm. A combination of thousands of laser diodes allows more power transmission accuracy and efficiency in more extensive power transmission.

The Laser based SSPS Research Team of JAXA has been researching and investing in developing technology to perform high-efficiency LPT with the controlled direction of a laser beam with an accuracy of 1 μrad (5.7×10^{-5} °) to limit the divergence to a few centimeters when transmitting for several hundreds of km. The high-power laser beam sends electric power to the daughter satellite. Commercially, LD offers a high efficiency of about 40 % ~ 60 % with around 1 kW laser power. As the power will be a few 100 W, the dissipated heat is high, due to which the cooling system will be more complicated. Their collection is achieved with PV cells' help on daughter satellites [81].

2.3. The E-Orbit Reference Mission Designing

The previously proposed SSPS design has not yet been developed. However, to analyze technical infrastructure development, evaluate environmental effects, a societal concern, and performance assessment-based guidelines were established. Using these constraints, the thesis is focused on designing and evaluating the system analysis with following reference system with the current technology is discussed throughout the thesis in Part II. During concept study, a range of technological alternatives were identified. A combination of power generation and transmission system evaluated would have been beneficial for minimizing the size, mass and cost of the satellites. With this stand, the primary and secondary objectives of E-Sat and E-orbit are,

Primary Objective:

To generate, convert and transmit electricity in space using laser-based or microwave system with the concept of SSPS by developing small satellites and creating a constellation of the E-Sat to create an E-Orbit in space.

Secondary Objective:

To demonstrate the capability of high-power generation in a compact design using a concentrator and bifacial PV solar array. The utilization of an LPT system for laser thrust and propulsion generation for space sustainability and power distribution for debris removal and orbital maneuvering.

Studying the architectural and technical requirements for the SSPS projects leads the study to investigate a scenario of E-Orbit with the following required millstone.

E-Sat: The previously presented SSPS designs are huge in structure with concentrator and non-concentrator to generate bus power, distributed power, or laser direct excitation system as briefly described in section 2.1. With this,

- The examination and structural optimization are required with desired orbital parameters for a single hub station or designing a constellation of multiple E-Sat systems.
- A compact bus and easy structural system to the rapid construction of E-Sat.
- A perfection of sun-tracking and AOCS system for maximum solar irradiance using current technology.
- Power distribution and allocation (inhouse E-Sat subsystem, E-Sat-to-satellite, and E-Sat to E-Sat).

- Clear roadmap to realization, demonstration prototype, and creating market.
- Public acceptance with the legal framework, space policy and energy policy.
- Easy to launch and integrated into space using economical space launch vehicle.

Power Generation: The selection of power generation unit is the essential factor for the high efficiency and sustainability of SSPS in orbit; it includes

- Selection of PV Solar cells with economical to develop the E-Sat.
- Solar light collection system (concentrator or non-concentrator system)
- The satellite power management with large power distribution, interruption, decentralization, and superconductivity
- Protection against the radiation

WPT: The SSPS design is defined on the basis of the method of the power transmission system; theory is discussed in section 2.2. The selection of WPT system is defined with the power generation and transmission integration as,

- Selection of desired operational frequency radio wave, lightwave, with spatial power density, beam shape, modulation, and frequency band.
- In the scenario of MPT
    - The selection of microwave oscillator (microwave, millimeter-wave generator, and electron tube)
    - Selection and designing of MPT beamforming, phased array, splitter/phaser, sidelobe suppression devices
    - Designing of RF circuit, antenna and rectenna system.

- In the scenario of LPT
    - The selection of LD, wavelength and material with laser oscillation.
    - Laser beam formation, transmission
    - Laser power receiving device
    - Beam orientated control system
- Propagation of radio waves and light waves in troposphere and ionosphere
- Selection of wavelength with neglecting and avoiding inference to the ecosystem, atmospheric, and operational communication system.

Economics and Business:

- Economic considerations and cost requirements for the development of SSPS to the end-users service,
- Feasibility and marketability evaluation for distribution and integration to the end side grid
- Sociology and environmental studies for a sustainable society, global warming prevention, green energy and fossil fuel alternative energy source.

With above mission designing constrain and milestone to achieve a sustainable PMS system using SSPS technology. The broad objective of E-Sat system designing is based on initial sets of numerical requirements for space power transmission missions. To create the mission with current trending technology, need functional, and operational requirements. A constraint to control the over-budget and environment scenario to utilize the system capacity efficiently with the scheduled implementations in Table 2.5.

Table 2.5 Reference system mission requirements

Sr. No.	Requirement	The factor which Typically Impacts the Requirement	E-Orbit
FUNCTIONAL			
1.	**Performance**	The primary objective	Transmission of power (min 10 kW)
		Primary power generation system	PV Solar cells (54 m^2 -112 m^2)
		Secondary power generation system	Thermoelectric Generator
		Payload	LPT /MPT subsystem
		Orbit	LEO
		Pointing	Multi 900 km
		Weight	250 kg
2.	**Coverage**	Orbit,	500 km radius
		Number of satellites	1- 1600
		Power	10 kW (minimum)
3.	**Responsiveness**	Communication architecture, processing delays, operation	Real-time data transceiving system`
4.	**Secondary Mission**	For mother-daughter experimental satellite	Debris removal, laser propulsion and thrust generation on daughter satellite
		For full constellation	Debris removal, laser propulsion and thrust generation, orbital maneuver
OPERATIONAL			
1.	**Duration**	Operations life, level of redundancy	Mission operational at least 20 years
2.	**Availability**	Level of redundancy, and power transmission	available for 24 × 7
3.	**Survivability**	Orbit, hardening, electronics	Natural environment only

Table 2.5 Reference system mission requirements (Continue)

Sr. No.	Requirement	The factor which Typically Impacts the Requirement	E-Orbit
4.	**Data Distribution**	Communication architecture	Real-time communication between satellite to satellite using Laser and ground stations by RF
5.	**Data Content, Form, and Format**	User needs, level, and place of processing, payload	Across energy orbit up to 10 kW of power
		CONSTRAINTS	
1	**Efficiency**	PV conversion efficiency	30 % (min)
		DC to RF or	80 %
		DC to Laser	40 - 60 %
		Beam collection at rectenna	90 %
		RF – DC conversion at receiving site	80
		Laser to DC power conversion at receiving site, or PV system	59 %, 30 %
2.	**Pointing accuracy**	LD divergence	9 mrad
3.	**Interlinking**	The readiness of E-Sat at any point of time for power transmission	Minimum 3 satellites will always communicate for continuous power supply at any time
4.	**Cost**	Designing cost of single E-Sat	10 M USD
5.	**Technology**	Available technology	2020 years
6.	**Schedule**	Technical readiness, program size	Initial operating capability by 2024, the final operational E-Orbit by 2035
7.	**Regulation**	Law and policy	LPT and MPT regulation
8.	**Political**	Sponsor, government	On public demand

Table 2.5 Reference system mission requirements (Continue)

Sr. No.	Requirement	The factor which Typically Impacts the Requirement	E-Orbit
9.	**Environmental**	Orbit, Lifetime	Natural
10.	**Interfaces**	Level of the user and operator infrastructures	To the sponsor and distributed between satellite users who will use E-Orbit
11.	**Development Constraints**	Sponsoring organization	Launching by the international collaboration of space agencies to creating a unified E-Orbit.
12.	**Data Handling Substation**	The command and control for preciously operation	2 Earth station

2.4. Space Law

The space law policies are formulated and proposed by COPUOS in the 1960s. The treaty discusses the peaceful, responsible use of outer space, including the Moon and celestials' bodies. Currently, 111 countries adhere to the space law, and 23 signed treaties but have not provided the ratification. The COPUOS has formulated the main five multilateral treaties and resolutions are as below.

- 1967: Principles Governing the Activities of States in the Exploration and Use of Outer Space, including the Moon and Other Celestial Bodies.
- 1968: Agreement on the Rescue of Astronauts, the Return of Astronauts, and the Return of Objects Launched into Outer Space.
- 1972: Convention on International Liability for Damage Caused by Space Objects.
- 1975: Convention on Registration of Objects Launched into Outer Space.

- 1979: Agreement Governing the Activities of States on the Moon and Other Celestial Bodies

The five main resolutions are:

- The 1963 Declaration of Legal Principles Governing the Activities of States in the Exploration and Use of Outer Space (resolution 1962 (XVIII))
- The 1982 Principles Governing the Use by States of Artificial Earth Satellites for International Direct Television Broadcasting (resolution 37/92)
- The 1986 Principles Relating to Remote Sensing of the Earth from Outer Space (resolution 41/65);
- The 1992 Principles Relevant to the Use of Nuclear Power Sources in Outer Space (resolution 47/68);
- The 1996 Declaration on International Cooperation in the Exploration and Use of Outer Space for the Benefit and in the Interest of All States, Taking into Particular Account the Needs of Developing Countries (resolution 51/122).

2.4.1. International Collaboration for Feasibility of Space Solar Power

Because of the emergence of SSPS technological progress and the associated business enterprise, both components mainly depend on one another. SSSP must at minimum be in the areas of economic feasibility to encourage commercial space players to invest, exhibiting market to increase the number of end-users with controllable and profitable costs. SSSP operates as an accessible renewable energy source; the cost of placing the system is to use it.

In particular, the starting cost drives the SSSP to the sky. This is an issue that all space activities face.

Public-private collaboration between participants in both industries can distribute some outrageous cost tags and build the essential infrastructure. Their risk and responsibility are shared. The idea of space security is one of the significant challenges in the feasibility of SSSP. Space security has been described as the protection of human and/or spacecraft through all phases of a space mission, whether it is 'man' or 'unknown.' The space security concept includes: (a) all elements ranging from pre-start, launch, orbital or sub-orbital, re-entry and landing, (b) to safeguard ground and flight infrastructures and their surroundings in the vicinity of the launch sites and (c) to secure space-based services, the structure, and the autonomous satellites.

SSPS's task is to strike a balance between 'constantly transferring energy to Earth at a level high enough to be efficient yet low enough to cause no harm.' At the early phases of SSPS, many believed that security threats flow considerably above the more usual radio use, from the energy level sent to the Earth and the atmospheric dangers posed by the large number of heavy-lifting lifts required for the required equipping in space. Indeed, SSPS shares the safety hazards associated with successful launches that failed in all other space activities.

Similarly, SSPS can produce space waste, particularly given the enormous areas of easier-to-hit designs with more moving components to damage. Another danger of safety shared by all space activities is a collision. However, the primary security problem highlighted in the framework of SSPS is Earth's WPT and its potential for detrimental effects on biota and the environment. The most important aspects are the shape and intensity of the energy exposed to life and its environment, particularly the climate. The WPT problem may be classified as environmental consequences, including RFIs and biosphere hazards, and path of beam that could create dangerous situations. While much of the literature talks on microwave transmission, the laser has the same restrictions to provide acceptable levels of safety. The main difference between laser and microwave is its armament potential.

Initial worries about the level of power returned to Earth were alleviated by the increasing usage of microwave power in communications, medical, radar, and industrial applications as well as the common usage of microwave ovens in households over decades. The beam falls below the harmful threshold even in the most intense area and under continuous exposure to the laser beam. Non-ionizing radio frequency power is conceivable; thermal effects should be modest but only approximately 1/4 the intensity of the transmitted beam. The mid-range WPT bioeffects (5.8 GHz) investigations of NASA showed no significant risk of exposure. There are few interactions between the atmosphere and the power beam, and the possibility of harm is unknown.

However, there are still specific concerns. Both microwaves and lasers, radiating beyond the visual region (e.g., IR lasers), are undetectable to the naked human eye. They are insidiously exposed to persons working inside and living around terrain stations, airplanes, and other aircraft such as ultra-lights and balloons and, presumably, to bird species. The "startle factor," or momentary blindness or blinding of airline pilots, is problematic for visible light-lasers. Damage to birds and mitigation techniques may be incorporated in the environmental safety requirements in the broader environmental context. Harmonizing environmental security requirements among nations might effectively set an acceptable minimum threshold for WPT emissions, as found in other parts of the spatial activity and electromagnetic energy systems.

Frequency and microwave safety are correlated. Higher frequencies make antenna dimensions and gains more attractive but at the sacrifice of the lowness of power. In addition, the higher the frequency, the denser and the stronger the beam, the more harmful. Consequently, the challenge of allocating frequencies and orbital slots for SSPS extends beyond political and legal matters. The most suitable frequency for the transmission of the microwave was initially 2.45GHz. However, given that it is now widely used, this is no longer a feasible choice. NASA and other organizations have proven 5.8GHz as a viable option, and this has been discussed as an ISM band for SSPS microwave transmission. As the environmental danger of WPT is radio interference, international telecommunications would be an essential factor.

2.4.2. Liability

As noted above, several SPS-related designs are expected to emerge, and each of these models will have its specific risks and liabilities. Some of them are briefly summarized as follows:

- Flaws in the technical design and its ability to produce power at projected levels and cost-efficiency. This might give rise to liability (including product liability) claims by those who funded the project;
- Flaws in rectenna design or problems with its location; e, g. reflected energy could harm aircraft, nearby homes or industries, or even satellites that need a low noise environment to operate. Rectenna could have an adverse ecological effect on fish or ocean life, or local flora or fauna if on land, etc.;
- Transmission lines to cities would have to get regulatory approval and, depending on the level of transmission power, could have an impact on housing or industry that are along the transmission pathway;
- Transmission via laser, millimeter-wave, or microwave and translation from electrical power to radiofrequency and back has several issues involving interference to other satellite systems, medical protective systems, and others. This

could result in operation from LEO (that is quite congested) and would be very difficult indeed;

- Malfunction of concentrators so that they focused destructive power and light on other satellites or even high-altitude aircraft would be a concern and would call for a "failsafe design" in this regard;
- If the SPS unit is designed for upgrade or for retrofit of PV cells or to take sections apart, this would have implications in terms of investor claims, end of life disposal, and others.; and
- Suppose there was a pointing accuracy problem in the power transmission unit from the SPS back to Earth, especially if the pointing system malfunctioned and started beaming power or radio frequency emissions at military, communications, remote sensing, navigation and time, weather satellites, and many more so that they could be disabled. In that case, this could trigger a multi-billion USD liability.

2.5. Adverse Effects on Human Health and Property

There are some concerns regarding the adverse impact on humans who work or live close to ground-based rectennae that receive microwaves carrying power transmissions from an SPS. There is currently no data fully proving such risk, mainly due to current indecision about the type of radiofrequency to be used and the intensity of the power to be transmitted. Still, the URSI has expressed its concern that "above the center of the rectenna, the SPS power flux density will be considerably higher than the currently permissible safety levels for human beings."

The URSI warned that human access around the rectennae "would need to be carefully controlled to ensure environmental safety and health standards are maintained." The URSI noted that the International Commission on Non-Ionising Radiation Protection and Japan both apply more stringent limits. However, there are no legally binding international safety standards to ensure human health safety from exposure to microwaves carrying power transmissions from an SSSPS. Any damage in the form of adverse effects on human health (i.e., 'impairment of health') caused by SSPS electric transmissions would become a basis for imposing responsibility and/ or liability on the SSPS operating States and, consequently, for a claim for compensation under international space law.

Frequency bands, satellites to be used, power and density levels, and the diameter of the antenna, An SSPS system involving any international cooperation would therefore have to satisfy all of these requirements, creating a lengthy bureaucratic procedure before it could become functional. Because States are internationally liable for damage caused by their space objects, whether the responsible entity is public or private, they will tend to put domestically applicable liability regimes in place in order to limit their liability vis-á-vis entities involved in an SSPS operation and to be able to recover possible international claims made against them. SSPS operations will therefore need to be conducted with consideration of the relevant liability regimes.

2.6. Managing Risks

In order to make SSPS operations more attractive and economically viable, it would be prudent to consider ways of managing the risks associated. This can be done in several ways, including risk allocation, public-private partnerships, transparent regimes of dispute resolution, and improved risk management regulation through domestic legislation. Each of these will be discussed herein turn. Traditionally, as in commercial undertakings in a wide variety of scenarios, exposure to liability in an SSPS system could be handled by allocating a particular risk to the party or parties best suited to manage it at a minimal cost. This would apply in ventures whether they are private or public, or some combination thereof. Often, the risk is allocated by the procurement of insurance coverage. Insurance can be obtained by the satellite owner (the exception), launch suppliers (the rule), or the satellite operator.

The risk allocated has:

- Been made fully aware of the risks they are taking,
- The most significant capacity [expertise and authority] to manage the risk effectively and efficiently (and thus charge the lowest risk premium),
- The capability and resources to cope with the risk eventuating,
- The necessary risk appetite to want to take the risk and
- Been given a chance to charge an appropriate premium for taking it.

Part II

Chapter 3
Energy Satellite

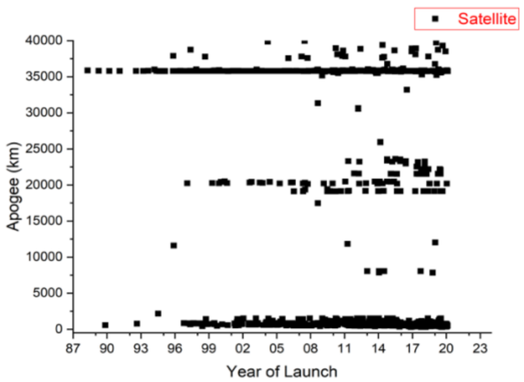
Chapter 3

Energy Satellite

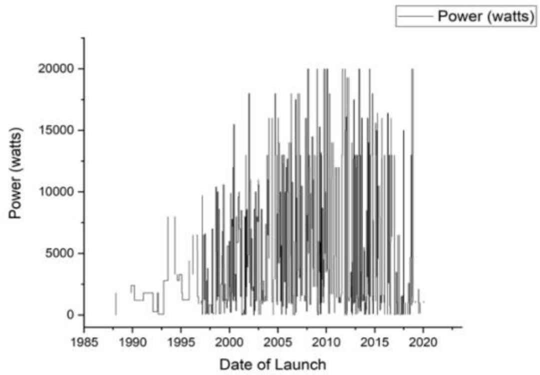
3.1. Introduction

Today, more than 4,048 satellites from the first artificial satellite Sputnik 1 (Спутник-1), 1957 are currently orbiting in space. At present, more than 3,328 satellites are operating in LEO [84]. This satellite industry covers the global space economy between 271 bn USD of 366 bn USD [85]. Such satellites, primarily located in LEO and GEO are used for telecommunication, remote sensing, navigation, research, science experiments, and military application. Figure 3.1 (a) shows that satellite increments have increased exponentially in space in the last three decades. Figure 3.1 (b) shows the LEO-based satellite population. In the previous two decades, most satellites were launched and placed between 400 – 900 km. The constellation designs for fast communication are established by SpaceX and OneWeb company. The satellite industry is growing with technological advancement; by 2026, the total small satellite market in LEO is projected to be worth 15.69 bn USD with more than 9000 active satellites. While the technology for fast satellite communication has been improved, the PMS of the satellite is the same since the very first satellite Sputnik. More than 99 % of space missions rely on the same PV array, battery or radioisotope thermoelectric generators. Every decade, agencies spend millions of USD maintaining solar panels, new batteries, and power management systems of the international space station. Researchers are trying to create a reasonable and long-term prospect of renewable energy sources for space application. The new perspective of power generation and management is required with low cost and high-power capacity to satisfy the mission demand.

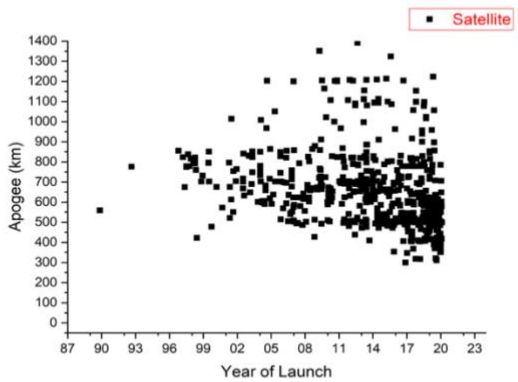
A satellite requires a continuous power source for the onboard sensor, maintaining altitude, telecommunication, and payload. The required power varies with size, orbital planes, altitude, functionality, and payload. For example, the international space station uses 75 to 90 kW while generating 84 to 120 kW using four PV array sets. Figure 3.2 (a) illustrates the power generated and utilized by satellites in the last three decades. Figure 3.2 (b) shows the power generated by satellites in the last five years. Figure 3 indicates that most satellites operate below 5 kW with a load of onboard operation and payload functionality. Therefore, most of the time-space industry relies on PV arrays for power supply. However, it is expansive, having multiple losses due to wiring, blocking diode, mismatch, calibration, cover glassing, UV, micro-meteorites, and temperature change [86]. The demonstrative mission of WPT will address the issues stated above, as discussed in the following [87].



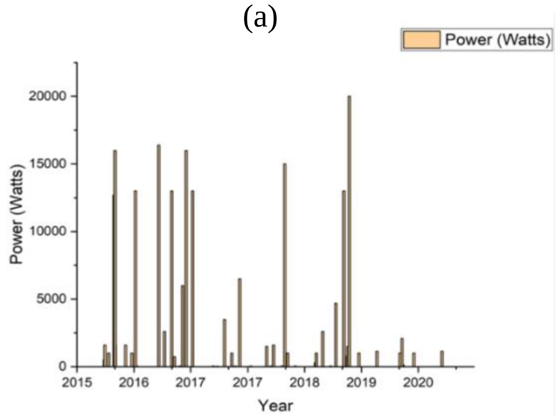
(a)



(a)



(b)



(b)

Figure 3.1. Satellites launched in space (a) nearby earth orbit (b) satellites in LEO

Figure 3.2 Power generated by satellite (a) in last thirty years (b) in last five years

3.2. Energy Satellite

This research activity primarily aims to develop, validate, and apply mission designing and analysis designing for a small-scale SSPS named the Energy Satellite or E-Sat. The design of E-Sat is based on 2020 technology that can be improvised with the idea of creating a system of compact SSPS to generate efficient energy using mirrors and a two-sided PV cell array: the PV cell array attached on the backside will absorb light reflected from the mirror to maximize energy generation. The thermal temperature of both PV sides will be maintained by a thermoelectric generator which will produce more than 2 % of electricity on one side and simultaneously cool down the other side of the PV cells array and vice versa using a sandwich structure. The small reflector will be placed precisely, focused on each back PV cell array to produce twice the amount of electricity in DC form (as shown in Figure 1.5)

LPT generally uses the power transmission with 1064 nm, [78] point of contact. One E-Sat weighs approximately 250 kg, and a fully functional E-Sat can generate electricity using PV cell arrays, thermoelectric generators, and reflectors. The satellite is designed for the LEO-based operation and to fulfill the mission energy requirements. The following chapter will provide a complete explanation that describes the system design, functionality, structure, and avionics system. The guideline and assumption are as follow in Table 3.1:

Table 3.1 Mission assumption

Parameters	
Mission Architecture	Small SSPS
Mission Orbit	LEO
Altitude	900 km
System Lifetime	20 years
Technology Availability	2020
Payload	LPT
PV Efficiency	30 %

A functional block diagram explains the E-Sat and E-Orbit in Figure 3.3, where the E-Sat subsystem and power conversion and transmission are explained in the architecture flow. With the combination of AOCS, GNC and OBDH system for operation of E-Orbit in LEO, with communication and Earth to orbit launch system.

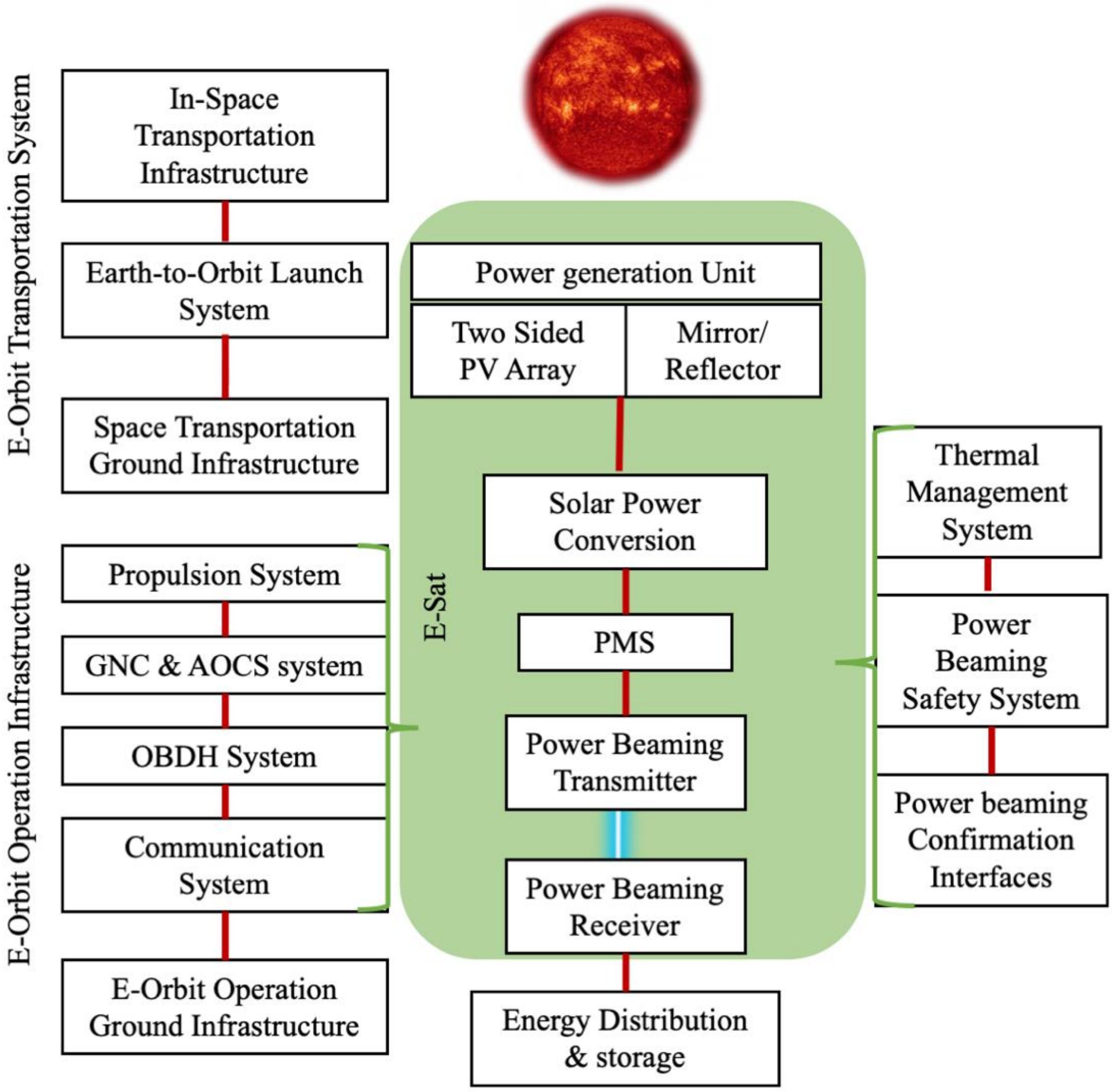


Figure 3.3 E-Sat and E-Orbit architecture function block diagram

3.3. Novelty of E-Sat

The previously designed SSPS systems are complex and more extensive in size, as mentioned briefly, with a notable example in chapter 2. The proposed complex SSPS design needs a continuous launching and in-space assembly unit, currently expansive and needs a futuristic robotics system. The space agencies adequately demonstrate the operational capability of WPT in space with various space missions, for example, the MINIX and PRAM. These missions are research-oriented with exorbitant costs; however, they remain critical architectural units for the futuristic GW SSPS project. At this rate of considerable progress in the active SSPS community, the GW mission design will be appropriately completed by 2050. Commercialization and economical solution are needed to progressively increase the progress rate of strategically designing SSPS. With this fundamental concept, a small-scale SSPS system, easy to launch in space with present technology and economical solutions, invented the E-Sat system. The delightful novelty of E-Sat remains identified as follows.

- A microsatellite with three cubes structures closely arranged for compact design.
- Bifacial PV cells array in a sandwich configuration for small designing.
- A thermoelectric generator efficiently generates a minimum of 2 % electricity while adequately maintaining the appropriate temperature of the PV cells in the sandwich configuration.
- Multiple concave reflectors for rear side PV cells.
- Payload is a laser-based WPT system that uses 50 LD of 200 W each to accurately provide constant 10 kW power.
- Modern technology is utilizing, and subsystem is strategically designed based on the year 2020.

- The specific structure is incorporated with the concept of reusability at EOL.
- The E-Sat is designed for multitasking as WPT, laser propulsion, and providing thrust to companied satellites.
- The orbital accuracy of E-Sat and carefully maintain for WPT using machine learning techniques.

3.4. E-Sat Designs:

An SC has two central parts, the SC bus and payload. SC bus controls and maintains orbit to support the desired operation by payload system. It consists of a communication unit, thruster, AOCS, OBDH, interlinking, and others. The E-Sat System design is a simple structure made with a small multiple one-meter cubes designed to quickly assemble multiple satellites to satisfy energy requirements. The structure consists of three and four cube structures per mission requirements. The main functionality of E-Sat is to generate, convert and transmit energy to any customer satellites. The reference payload system is equivalent to 10 kW DC power output into a conventional power grid and works as a transformer. The final E-Sat has an LPT payload at two opposite sides of the cube end-mounted optical antenna transmits to the rectenna of the customer satellite. Previously the E-Sat project started using both WPT systems, first with the microwave and then later with Laser. The system design and the structural designs are explained below [43].

The configuration of all E-Sat consists of a planer solar array structure and partial bus built from graphite composite material. With the C4MJ Metamorphic Fourth Generation cell, the solar cell will generate power with a 30 - 40 % efficiency. This design will assume 30%

efficiency. A C4MJ Metamorphic Fourth Generation cell option has a 20 years configuration with a solar overall planform area of 54 m^2 and 112 m^2.

3.4.1. Satellite Design 1:

E-Sat 1 is a small-scale SSPS capable of generating a minimum of 21,870 W of power using a blanket area of 54 m^2 PV solar panels with 30 % efficiency. The number of solar panels is 112 with 0.707 × 0.707 m of one slack. The satellite's bus structure is a T-shape, where four cubes are integrated with each cube and are 1 m in height, width, and length. The transmitted microwave power is equivalent to 10 kW, generated using multiple Klystron or Magnetron, the baseline power amplifier with slotted waveguide as the radiating elements mounted on the satellite. The WPT and payload for E-Sat 1 are two opposite-sided MPT systems with the ends mounted with multiple (48) Conical Horn antennas, that configure the one-meter waveguide system. Model ATH2G8A is a wideband, high-gain, high-power microwave horn antenna that provides field intensities of up to 500 V/m. The phase control system utilizes an active reproductive array with a piloted beam with orbital elements for phase configuration. The opening mechanism of the solar PV cells array is a rolling mechanism, as Figure 3.4.

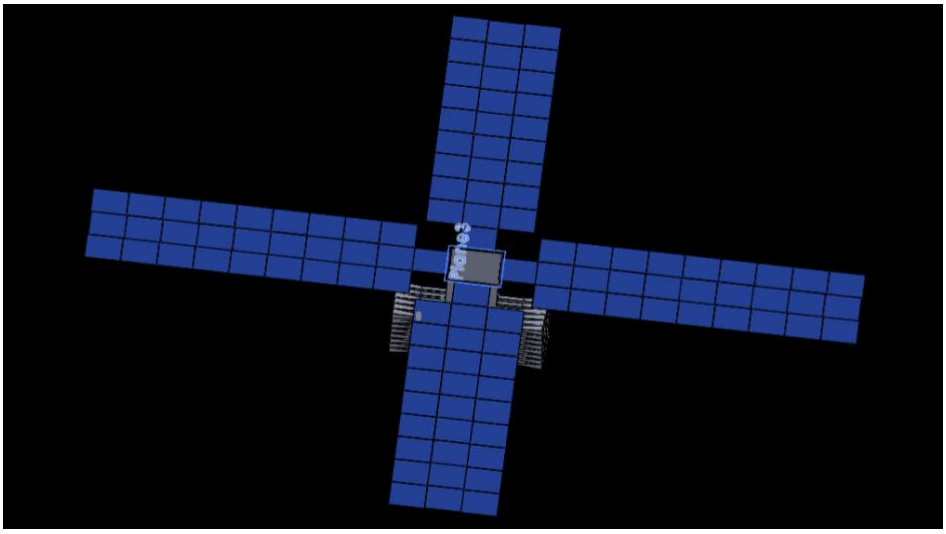


Figure 3.4 E-Sat 1 CAD model

The principal requirement is the transmission of energy of 10 kW in energy orbit/ from one satellite to another. The minimum converges area is 500 km, real-time response, 20 years lifetime. The payload is MPT based system with a frequency of 5.8 GHz. The inter-ground is transceiver by a 50 cm diameter antenna. Other specifications are as follows,

- high power microwave transmission and antenna design with high gain (> 20 dB).
- The highest Schottky diode rectenna efficiency, over 90 %, is achieved at 2.45 GHz
- The microwave power from the magnetron is extracted into a standard rectangular waveguide and their appropriate frequency bands.
    - Example: WR-340 waveguides, the frequency range of 2.2 to 3.3 GHz. WR-430 frequency range 1.7 to 2.6 GHz. Larger in size but more power propagated in the waveguide.
    - Size (W × H × D) =12.2 × 9.9 × 20.3 cm
    - Frequency range: 2.5 – 7.5 GHz
    - Power input (maximum): 12 kW CW
    - Weight: 1.18 kg
- PV cell:
    - Weight: 20.5 Kg
    - Structure: less weight using a graphite base structure. These structures must be extremely lightweight, which can be satisfied by a gossamer structure such as membranes, cables, and thin, flexible beams.

3.4.2. Satellite Deign 2:

The E-Sat 2 is a variant, and an upgraded version of E-Sat 1 with four reflectors attached between all four PV arrays, as shown in Figures 3.5 and 3.6. The main difference is the utilization of closed loop Brayton cycle system, it can produce more electricity using reflected light as heat. With 26 % efficiency, Brayton system will support and decrease the size of solar panels using the concentrator of heating on the working fluid inside the closed system. with 26 % efficiency, a concentrated heat of 500 °C will generate approximately 68 W/h with a minimum concentrator area of 21.486 m^2. Researcher achieved an efficiency of up to 54 % for hilum based closed loop Brayton cycle system. The transmission system is MPT-based with the same configuration of multiple horn antenna with wave guided system. However, the system is more applicable with the optical concentrator and for big size SSPS, with the limitation of E-Sat, it is difficult to incorporate within the mass ratio of system.

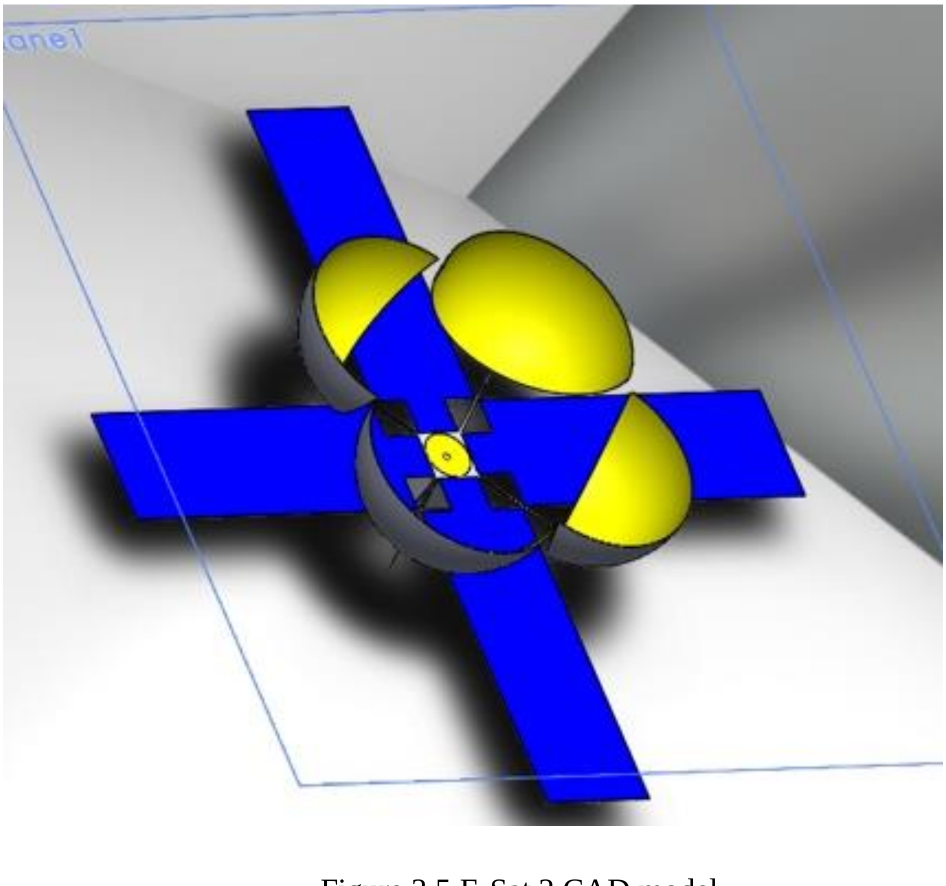


Figure 3.5 E-Sat 2 CAD model

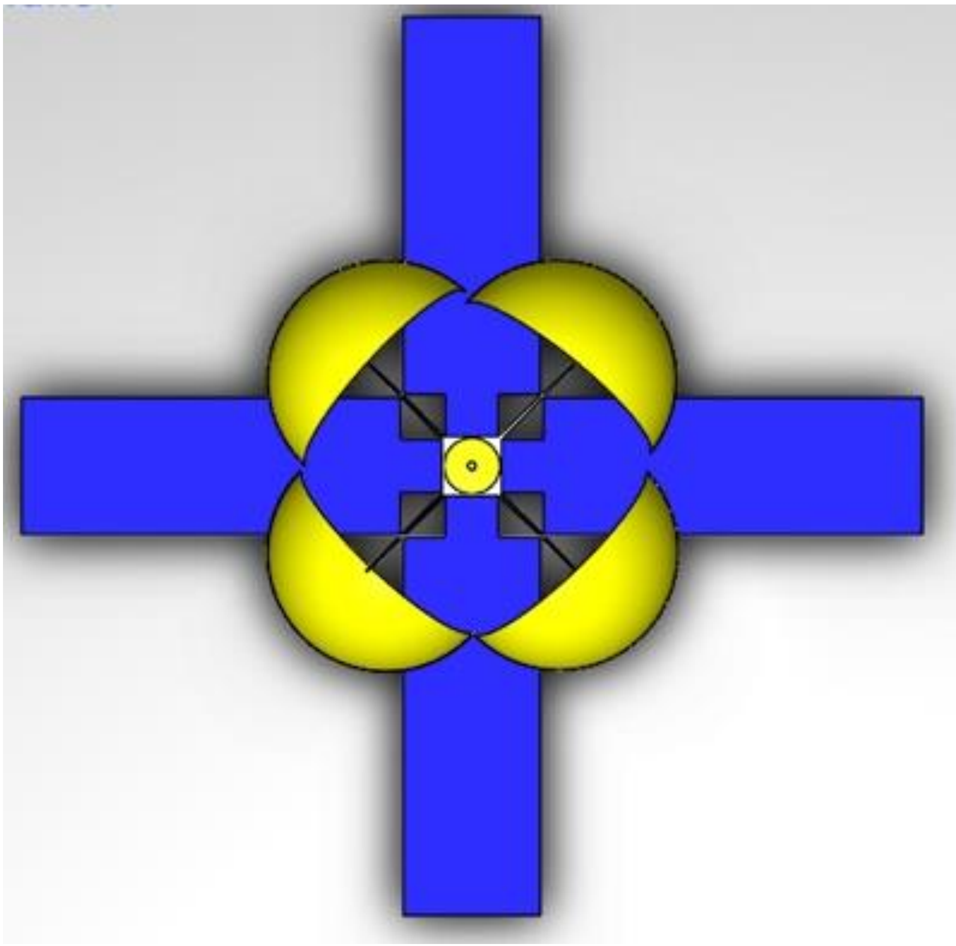


Figure 3.6 E-Sat 2 top view, CAD model

3.4.3. Satellite Design 3

The E-Sat 3 is an LPT-based payload system made of 3 units of 1 m^3 structure with a two-sided PV cells array and multiple reflectors. The reflector is used for reflecting on the rear side of PV cells array system. Using a reflector will increase the power generation system while reducing the weight of the satellite explained below in the concentrator section. The reflectors used are in two designed concave mirrors and rectangular plates for precisely reflecting the concentrated solar radiation on the solar panel in the clockwise direction. The design is simply similar to the previous E-Sat 1 design. The changes in E-Sat 3 is the WPT propagation method on LPT based with 50 Laser diode places with an optical stabilizer to focus on an object. The reflector comes in 4 groups with seven mirror elements to reflect respective PV solar panels, as shown in Figures 3.6 and 3.7. The structure is made of carbon fiber for lightweight efficiency.

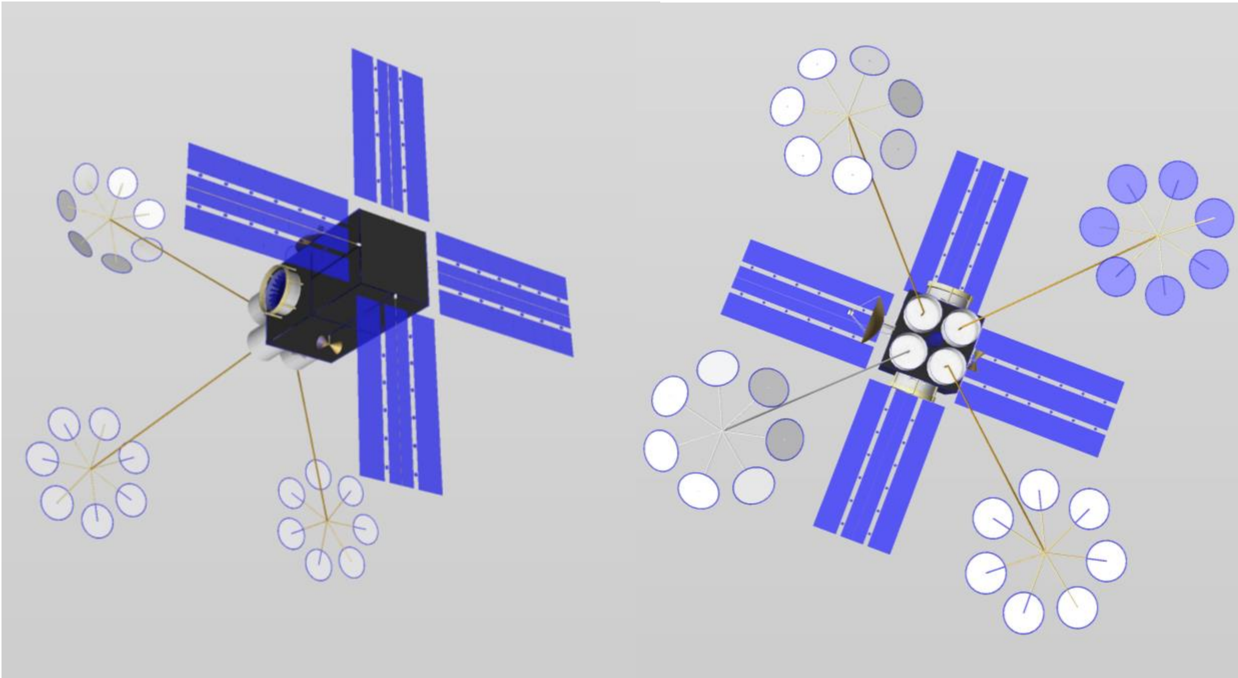


Figure 3.7 E-Sat 3, with reflector front and side view CAD model

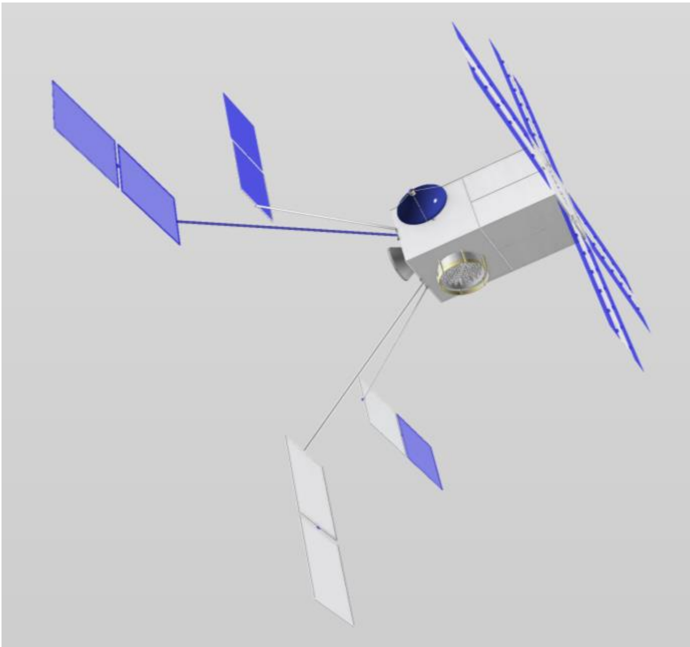


Figure 3.8 E-Sat 3, flat reflector CAD model

3.4.4. Satellite Design 4:

The design of E-Sat is based on 2020 technology that can be improvised with the idea of creating a system of compact SSPS to generate efficient energy using mirrors and a two-sided PV cell array: the PV cell array attached on the backside will absorb light reflected from the mirror to generate even more electricity. A thermoelectric generator will maintain the thermal temperature of both PV sides, which will produce more than 2 % of electricity on one side and cool down the other side of the PV cells array and vice versa using a sandwich structure. The small reflector is placed precisely, focused on each back PV cell array to produce twice the electricity in DC form. LPT generally uses power transmission with 1064 nm, [77] direct pinpoint transmission with narrow beam Figure 3.9, shows the CAD model of the final E-Sat. The E-Sat weighs approximately 250 kg, and a fully functional E-Sat can generate electricity using a PV cells array. The thermoelectric generator and reflector will be attached to advance the capabilities of the satellite. The primary payload of E-Sat is a Laser system. The single E-Sat will cost approximately 10 million USD.

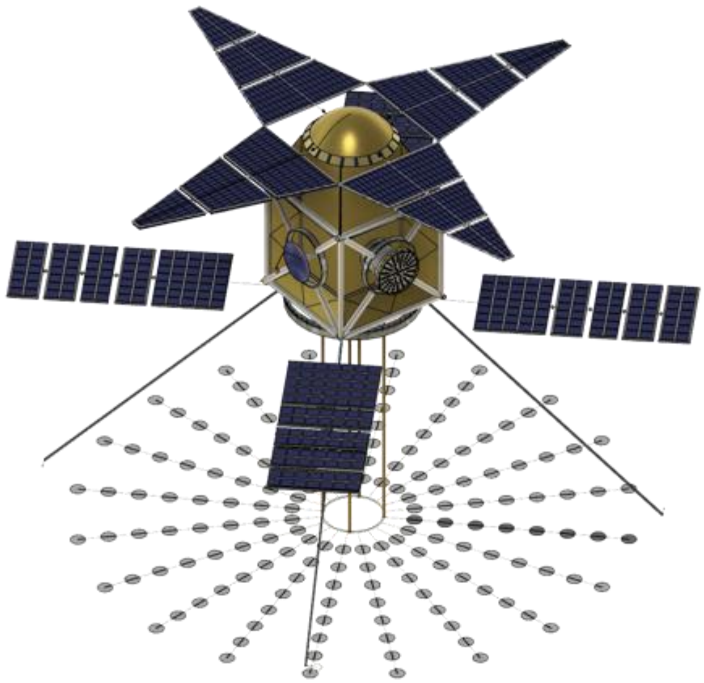


Figure 3.9 E-Sat 4 CAD model

3.5. Structure and Mechanisms Subsystem:

The structure of E-Sat has adopted the system aligned with the launch-vehicle adapter. The mission concept of the final E-Sat is based on potential reusability of satellites. The structure bus is intended to be 3×1×1 meter, resembling a small rocket nozzle structure. By the end or expiration of its operating life, the entire spacecraft will fold back to the original compact launching position; the reflector and PV array will cover the spacecraft and return to the Earth's atmosphere. Power transmission will be used by Laser-based wireless transmission. The laser component is difficult to burn out at reentry in the atmosphere, making it easier for capture using a sea-based fishing net at sea-based dropping location and utilize avionics for other satellites.

3.5.1. Concentrator

The PV cells work as heat engines, collecting energy from heat-radiated elements and cooling down with the PV effects. The sun is the heat source/element in space, and in satellites, the PV effect is taken by PV cells. The change in effective temperature of an orbiting element or a satellite is due to the direct line sight of solar irradiance and reflected energy from Earth and celestial body as Albedo effect. To gain more power, a multijunction bifacial PV solar cells array can be appropriately utilized, which generate electricity using direct solar radiation from the front side and absorb albedo effect with concentrator/reflector focused solar irradiance from the rear side of the solar panel (E-Sat 3 and 4) or concentrated directly to the heat-based engine (E-Sat 2). To properly design a concentrator for the SSPS system limits to the specific weight and considerable size of the reflector subsystem. A lightweight, thin, and highly reflective surface can be appropriately used for accurate reflection on satellites. In E-Sat designs, various innovative approaches were employed to sufficiently establish the concentrator/reflector to produce high reflection for increasing power generation efficiency efficiently. The concentrators are curved, convex, concave, and planer in a unique structure to properly focus on a perfect PV cell. For example, supporting an area of a reflective planer surface is 1 m^2 and, considering reflective efficiency of 90 %, can generate 368.28 W power on the 1 m^2 size solar panel using Equation 3.1 [88]

$$P_O = \frac{A \times \eta \times I_s}{100} \times \eta_{PV} \tag{3.1}$$

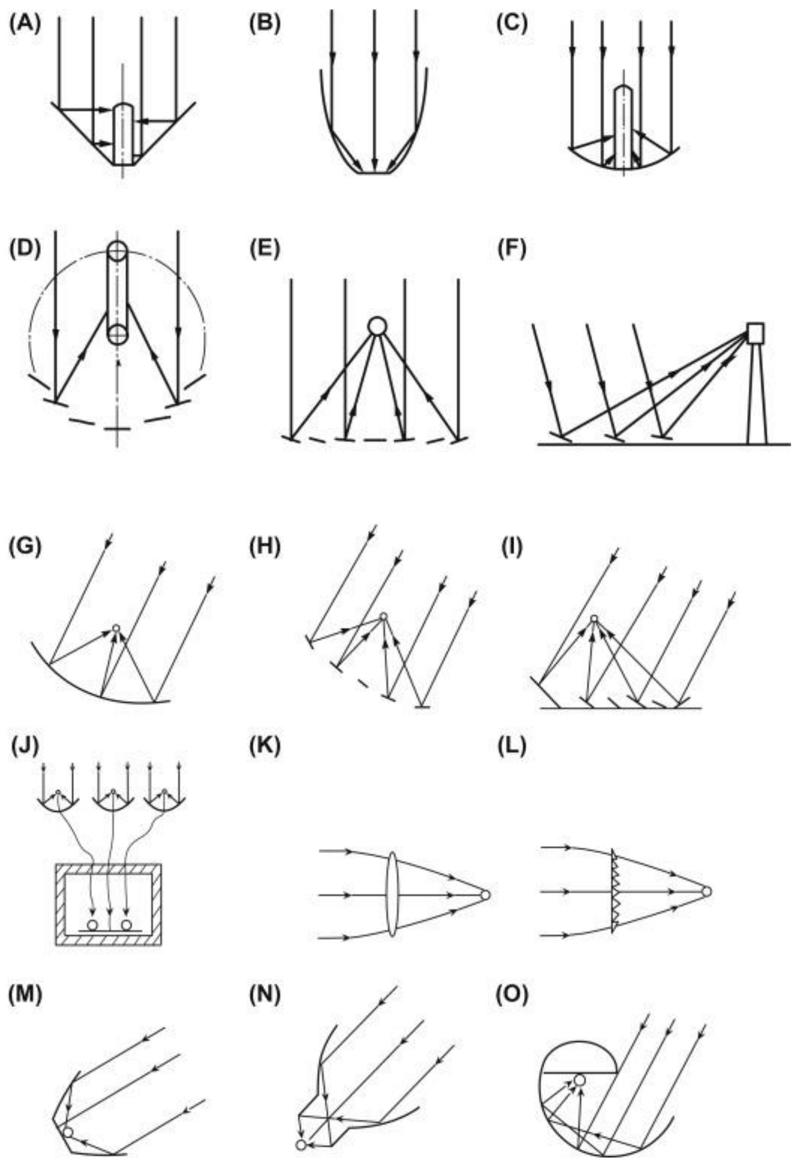


Figure 3.10 Various kinds of concentrating solar collector structures: (A) conical concentrator; (B) CPC concentrator; (C) sphere concentrator; (D) cylinder array concentrator; (E) array

Fresnel lens concentrator; (F) heliostat tower concentrator; (G) parabolic concentrator; (H) array parabolic concentrator; (I) reflective Fresnel lens concentrator; (J) small disk reflective concentrator; (K) convex lens concentrator; (L) transmittance Fresnel lens concentrator; (M) trough conical concentrator; (N) mirror dual-focus concentrator; (O) multicurved compound [89].

The Figure 3.10 shows the various type of structure design to incorporate high gain of concentrated solar light by the reflection. In many previously studied SSPS designs is incorporated parabolic and multireflection systems for example, SSPS OMEGA. Considering the solar concentrator mass between the range of 5 – 20 kg, each component is with diameter of 0.3 meters.

Circular: In E-Sat 3 and 4, the reflector used is a circular concave mirror. The utilization of a concave reflector benefits a higher concentration of solar irradiance on the solar panel shown in Figure 3.9.

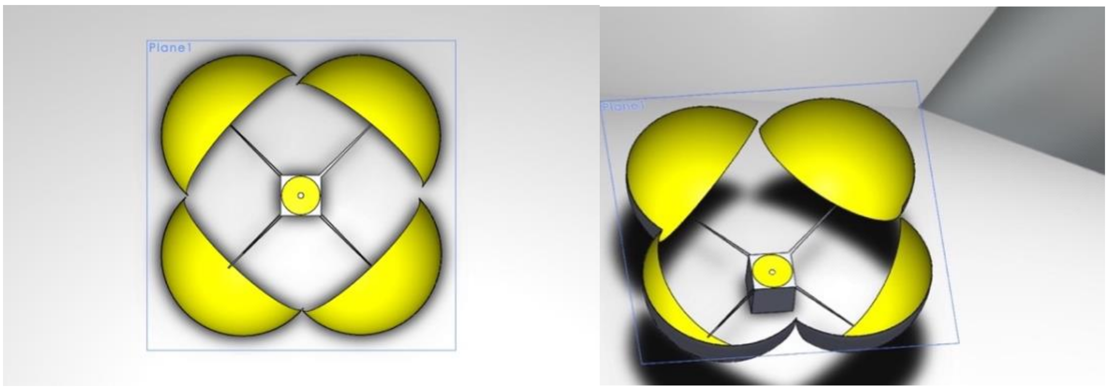


Figure 3.11 Front and side view of circular reflector CAD model

Curved: A curved and parabolic concentration system can obtain up to 3000 °C temperature. In E-Sat 2, the reflector is made from a single shape structure divided into four parts, the significance of this reflector, it is highly concentrated on a surface to generate heat which is used by the Brayton system as shown in Figure 3.11.

Planner: the planner reflector uses the same PV array size to reflect irradiance on PV.

3.5.2. PV array

In general, most space missions use PV cells for primary or secondary power generation systems. The battery combinations give continuous power supply to SC while orbiting around Earth or during interplanetary missions. The EPS of SC includes a power generation unit, storage, and distribution system. The PMAD system of SC makes power control easier. On the other hand, installing multiple solar panels on large structures such as SSPS will increase the weight, and thus the cost of satellites. Making a compact and in-pack design of solar cells requires mechanical moment, design complexity, reliability, and increased risk factor. The design of SC PV cells depends on the available light to SC, also called solar intensity, which varies with the inverse square to the distance of the Sun to the SC.

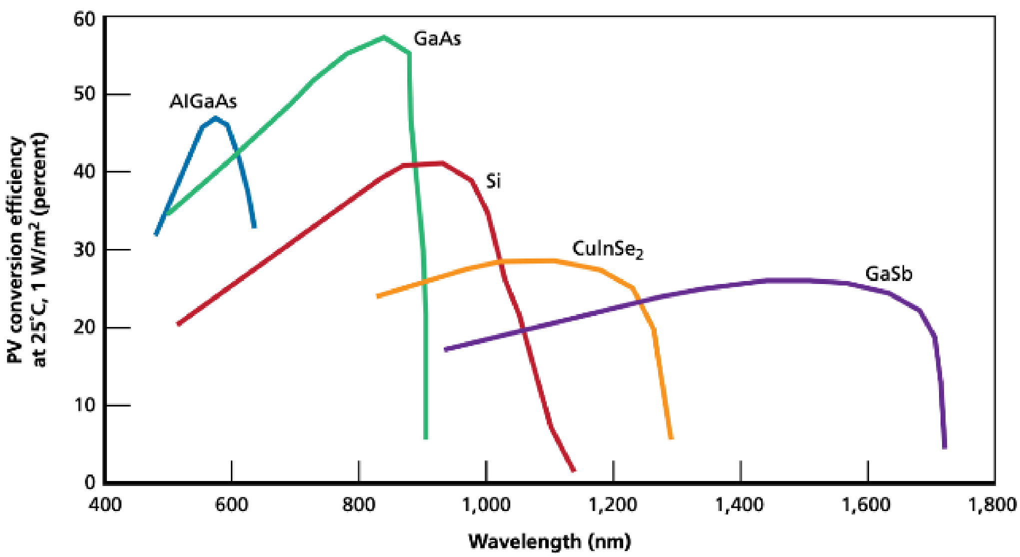


Figure 3.12. Spectral response of some PV materials. source [73]

A PV cell generates electricity by using photonic energy. The photon must have tremendous energy in the cells or be equal to the bandgap in the material used to create PV cells since the frequency is proportional to the photon's energy. The ideal light source should be monochromatic and with an ideal frequency consistent to that of sunlight. The most widely used PV cells are based on Si, GaAs, and InGaAs. Figure 3.12, the GaAs with 850 nm and Si with 900 nm gives the highest efficiency with the desirable range as discussed in the previous subsection [59]. As shown in Figure 3.12, GaAs and Si PV cells show a better response in desired frequency range than the LPT range described in the previous chapter. The other substrate also shows response but with poor efficiency. The aerospace industry utilizes a multijunction solar array for SC as it produces sufficient energy with average efficiency; triple-junction PV cells are primarily due to lower cost. The total solar energy output of the PV system can be calculated using Equation (3.2) and conversion efficiency is by Equation (3.3).

$$Solar\ Energy\ O/P\ of\ PV\ system = \ E\ = A \times r \times H \times PR \tag{3.2}$$

Conversion Efficiency

$$\eta\ (\%)\ = \frac{P_{mp}}{Incident\ solar\ energy} \tag{3.3}$$

Energy and power (kW), total solar panel area (m^2), efficiency (%), and considering coefficient for losses in range 0.5~0.9 (for calculation, default value 0.75). The PV array will be deployed using a rolling mechanism. Every PV cell is joined with three PV other cells, which can be opened one by one in all four directions.

3.5.2.1. E-Sat PV arrays system

E-Sat is primarily studied with an efficiency of 28 % to 32 %, as discussed previously. Several PV cells with different efficiencies are explained in Table 3.2. Higher efficiency of solar cells with appropriate costs will cut down the cost of satellite and structure mass, drastically decreasing the launching cost. E-Sats will generate energy with dual-sided solar panels, where PV cells mounted on the backside of solar panels will absorb energy from reflected rays from specially designed thermal white reflectors. All E-Sat is an origami-based structure adopted for reducing the opening of structure and mechanical elements. The thin ultralight PV cells can be adequately folded using one input mechanism, and it is easy to open without mechanical complexity. The solar panel can open and fold in E-Sat using the mechanical rolling system to deploy all solar panels in space.

Table 3.2 Space grade solar PV cells

PV cell name	**Efficiency**	**Maximum area per cell**	**Total minimum area of cells required to generate 50 kWh (m^2)**
Azurspace TJ 3J30C	30	30.18 cm^2	121.95
SolAero ZTJ Ω	30.2	80 cm^2	121.35
Spectrolab XTE prime	30.7	97.92 cm^2	119.33
Azurspace 4G30C	32	30.18 cm^2	114.41
SolAero IMMα	32	83 cm^2	114.41
4J minimodule (hybrid)FHI ISE	41.1	121 cm^2	89.12
For example,	36.5764		100
For example,	33.2513		110

3.5.2.2. PV mechanical Driver Circuit

A simplified analog circuit diagram for moving PV cells Array using the feedback from the light diode is shown in Figure, as the light source is moving the LD. Low luminosity will pass a signal to the transistor, and the motor will adjust it towards the appropriate direction. This is a simple diagram for solar panel management. However, the same instruction may utilize sun sensors and adjacent motor or thruster for the appropriate angle to get high power from the Sun. Below is a CAD figure designed for single-axis movements for a motor driver circuit. The simple circuit and PCB design are shown in Figures 3.13 and 3.14, respectively.

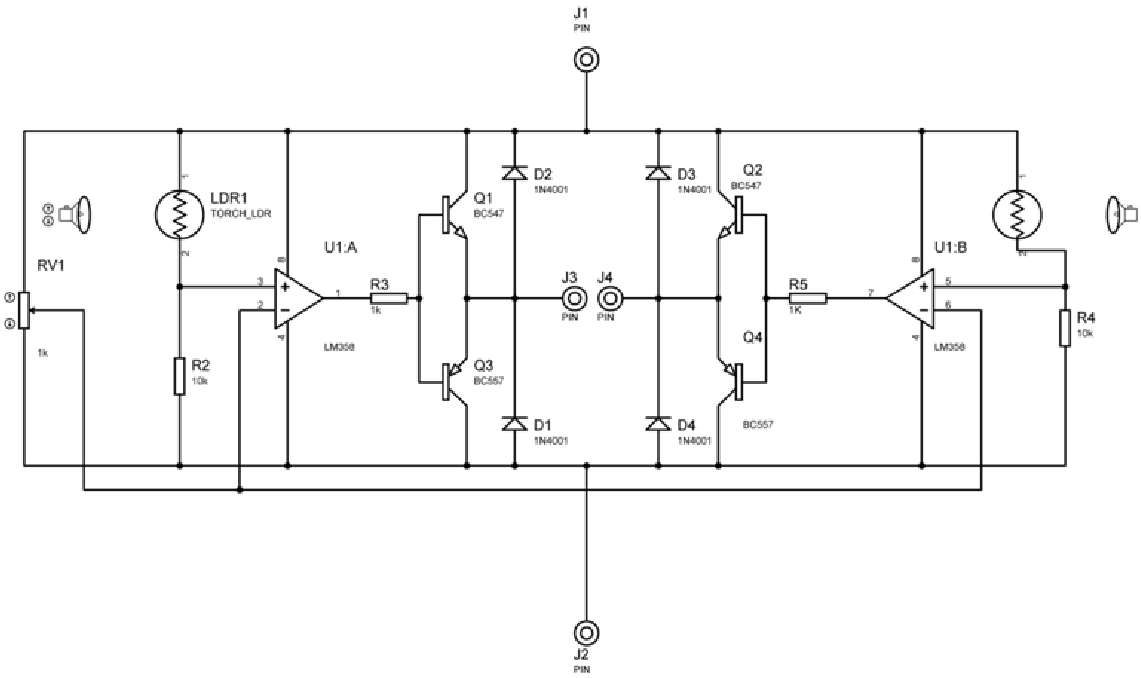


Figure 3.13 Schematic circuit diagram of light sensor diode for the feedback loop

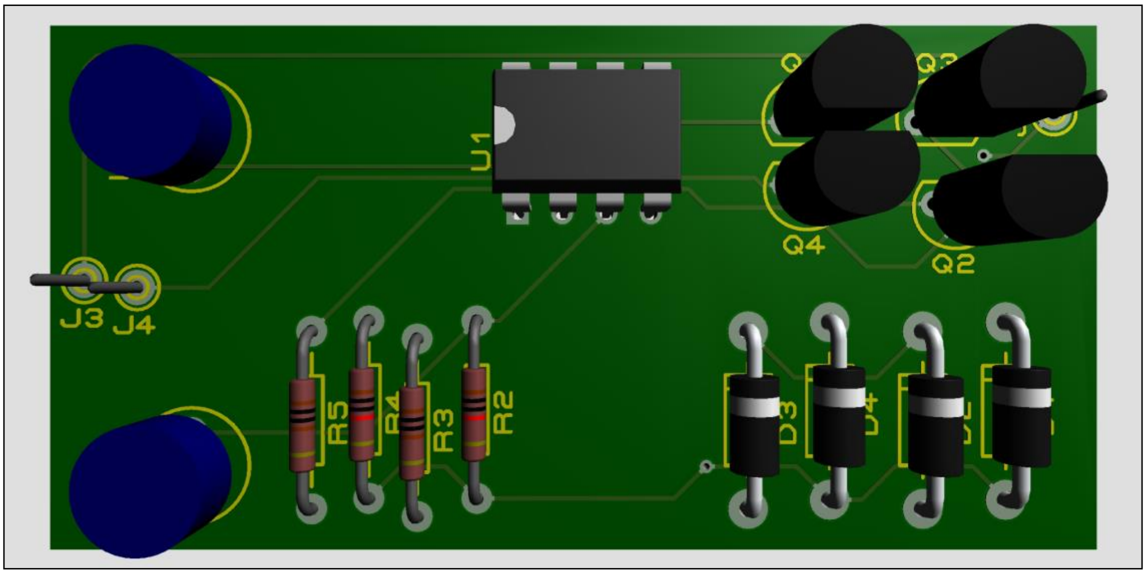


Figure 3.13 PCB design of light sensor diode for feedback loop

3.6. Payload

The payload carries the mission's central and essential purpose. It involves the architecture, receiving Payload of E-Sat, power generation and conversion unit, laser sub-system, optical calibration, and communication system. The Laser subsystem consists of 50 lasers with 200 watts each to transmit 10 kW of energy.

3.6.1. Microwave Power Transmission subsystem for E-Sat 1, 2

The detailed and historical microwave transmission experiments are described in chapter 2. In this section, the subsystem will emphasize the mechanism of the 10 kW MPT system. Before the transmission of power required a secure and ideally establish orientation of E-Sat for better transmission. The MPT subsystem will be utilized as payload with the configuration of multiple magnetrons and conical horn antenna incorporated with a one-meter waveguide system as shown in Figure 3.15. A total of 96 antennae is used for power transmission for propagating MPT of two-sided 10 kW power.

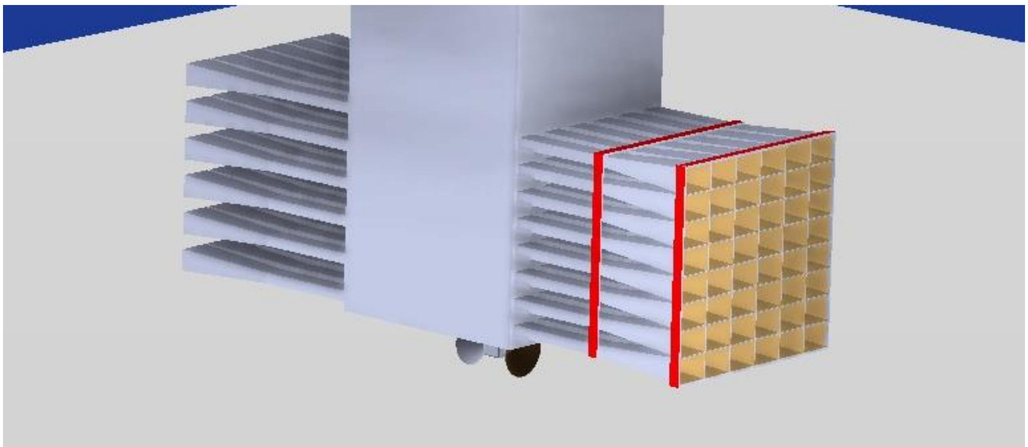


Figure 3.15 CAD model of multiple MPT antenna

The magnetron is used to generate microwave by motion of electron in a crossed electric and magnetic field. High power vacuum devices can be used for WPT systems from space to Earth, orbit and interplanetary missions. The magnetron can be in multi-cavity for high efficiency while utilizing duel port magnetron called Amplitron or Platinotron. For example, pulse magnetron is used in radar technology for identifying a moving object in the air. Magnetron frequency changes with change in voltage and current, using maximum voltage at 4.5 kV and cathode voltage at 3.7 V, and can be used for MW size WPT in SSPS system. For generating microwaves using magnetron, a reference frequency is locked, called an injection locking method. For this, the reference frequency is injected from the magnetron output antenna via a circulator. If there is difference in frequency between the magnetron and injected signal, it is symbolized as Δf, where f is frequency. It can be calculated using Adler's Equation (3.4).

$$\eta\ (\%)\ =\ \frac{\Delta f}{f} = \frac{1}{Q_E}\sqrt{\frac{P_i}{P_o}} \tag{3.4}$$

When the propagation distance is significant, the transmitting wave is considered a planer wave in an electromagnetic field. A key parameter in the WPT system is the efficiency between transmission system and rectenna; beam efficiency can be calculated using the Friis transmission Equation (3.5) and (3.6). Here d is distance between antenna, and A is the aperture area.

$$\eta = \frac{P_r}{P_t} = \frac{G_t A_r}{4\pi d^2} = \frac{A_t A_r}{(\lambda d)^2} = \frac{G_t G_r}{\left(\frac{4\pi d}{\lambda}\right)^2} \tag{3.5}$$

$$A = \frac{\lambda^2}{4\pi} G \tag{3.6}$$

In the Friis equation, the area of receiving antenna should be equal to or more than the area of transmitting antenna $A_r \geq A_T$ for better reception. For example, taking a transmission antenna of diameter 1 m and rectenna of 2 m separated with a distance of 1 m gives the 278.87

W for the power transmission of 10 kW. The beam of power density for the antenna of 1 m and 2 m diameter are 3183.1 W and 795.77 W/m^2, respectively.

Power at the receiving station the transmission system combines several antennae with a controlled phase to achieve a narrow beamwidth. Using Equation (3.7) and (3.8) is used for calculating beam efficiency η using τ of WPT between transmission and rectenna [30]. The system efficiency with the transmission antenna diameter of 1 m and rectenna diameter of 2 meters gives a significantly less efficiency concerning change in distance from 10 m to 500 km as shown in Table 3.3.

$$\tau^2 = \frac{A_r\, A_t}{(d\lambda)^2} \tag{3.7}$$

$$\eta = \frac{P_r}{P_t} = 1 - e^{-\tau^2} \tag{3.8}$$

Table 3.3 Efficiency of antenna with the change in distance

Distance	***Efficiency***
10	0.999
100	0.092
1,000	9×10^{-4}
10,000	9.28×10^{-6}
100,000	9×10^{-8}
500,000	3.71×10^{-9}

From a system standpoint, the critical derived parameter for the satellite is the specific power, S. S is the RF power radiated at the satellite per kg of mass in orbit. The larger S is, the less mass is required to obtain a fixed amount of radiated power. The goal is to maximize S while minimizing weight and managing risk and cost. Estimate, S, by Equation (3.9). The link budget for the SSPS system is modeled using a modification of the Friis formula [90], namely as in Equation (3.10),

$$S = \frac{\eta_{PV}\,\eta_{DCRF}\,\eta_{Tx}}{m_{SV}} A_{pV}(AM0) \tag{3.9}$$

$$P_r = [\eta_{PV}\ \eta_{DCRF}\ \eta_{Tx}\ \eta_{dif}\ A_{PV}\,(AM0)]\left(\frac{f}{c\,d}\right)^2 A_{PV}\,A_r \tag{3.10}$$

Effective antenna aperture can be calculated using Equations (10) and (11) as it can be seen the change of aperture with the change in gain from 1 - 100 dB, in Figure 3.16.

$$A_e = \frac{\lambda^2}{4\pi} = \frac{C^2}{f^2} \times \frac{G}{4\pi} \tag{3.11}$$

$$A_e = \frac{C^2}{f^2} \times \frac{10^{\frac{G(dB)}{10}}}{4\pi} \tag{3.12}$$

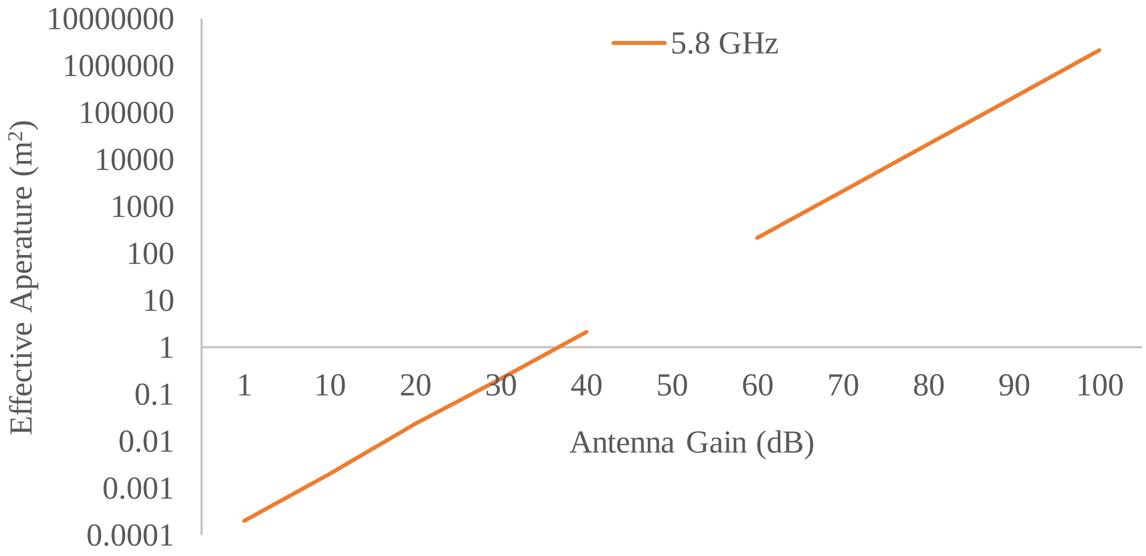


Figure 3.16 Effective Antenna Aperture with Gain

A parabolic reflector antenna gain can be calculated using the following Equation (3.13). Where k is the antenna's efficiency, a 5.8 GHz frequency antenna gain is shown in the following Table 3.4 with the different antenna diameters varies from 1 m to 1000 m.

$$G = 10 \times \log_{10} k \left(\frac{\pi D}{\lambda}\right)^2 \tag{3.13}$$

Table 3.4 Antenna Gain and Beamwidth with change in efficiency

Diameter (m)	***Efficiency (%)***	***Gain (dB)***	***Beamwidth (degree)***
1	100	35.66	3.62
1	90	35.21	3.62
1	80	34.70	3.62
1	70	34.11	3.62
1	60	33.45	3.62
1	50	32.65	3.62
1	40	31.69	3.62
1	30	30.44	3.62
1	20	28.68	3.62
1	10	25.66	3.62
1	1	15.66	3.62
2	100	41.69	1.81
2	90	41.23	1.81
2	80	40.72	1.81
2	70	40.14	1.81
2	60	39.47	1.81
2	50	38.68	1.81
2	40	37.71	1.81
2	30	36.46	1.81
2	20	34.70	1.81
2	10	31.69	1.81
2	1	21.69	1.81

The calculated value of effective antenna aperture for the antenna gain in the range of 1 – 100 dB for the 2.45 – 400,000 GHz frequency is shown in Appendix A, Table A.1. The parabolic reflector antenna gains with frequency range from 2.45 to 281,760 GHz with the beam width in Table A.2.

Table 3.5 Supposed Efficiency and Power for SSPS by Microwave

Elements	***Efficiency (%)***	***Power Level (kW)***
PV array	30	$P_{DC} = 21.4$
DC to RF power	80	$P_{RF} = 17.12$
Beam collection	90	$P_r = 15.4$
RF to DC Power	80	$P_{DC\ output} = 12.32$

Table 3.5 shows the general efficiency of the MPT [65, 67] system for the E-Sat and transmitted power to the daughter/ customer satellite. A transmission system diagram is shown in Figure 3.17, in which DC to microwave is generated and using microwave generator and using appropriately selected waveguide with multiple conical horn antenna for the propagation of MPT to receiving antenna. The receiving antenna has avionics to match the signal phase with a filtration system to convert the microwave into desirable DC power according to the payload selection for load or charging batteries. Furthermore, the total efficiency depends on the selection of frequency and PV cells. This systematics circuitry is basically used in all MPT based SSPS systems with desirable transmission frequency. The output DC power is approximately 10 kW with the $P_{DC\ output}$ and avionics efficiency. The Naval research laboratory demonstrated using Photovoltaic Radiofrequency Antenna Module Flight Experiment (PRAM-FX) in orbit as a sandwich model for space solar architecture. The experiment generated 8.4 W RF power with 37.1% DC to RF conversion efficiency, and the total module efficacy was 8% [54].

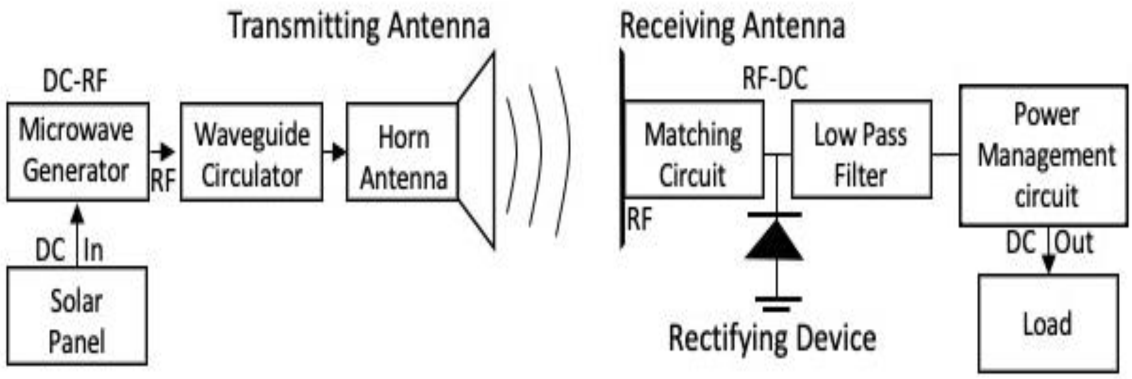


Figure 3.17 Schematic diagram of Microwave Power Transmission System

3.6.2. Laser Power Transmission Subsystem for E-Sat 3, 4

The Laser based payload system is used in E-Sat 3 and 4 with 50 LD of 200 W each. The detailed and historical laser transmission experiments are described in chapter 2. In this section, the subsystem is emphasizing the mechanism of the 10 kW LPT system. Transmission of power requires a secure and ideally establishment in the orientation of E-Sat concerning the customer satellite by the transceiving signal from the aiming and pilot signal subsystem.

The E-Sat sends a pilot and beacon signal to the customer satellites using a continuous laser pulse to verify a secure connection across the distance. The generated DC power is rectified and sent to the laser driver circuit to solid-state diode to generate a high-power laser

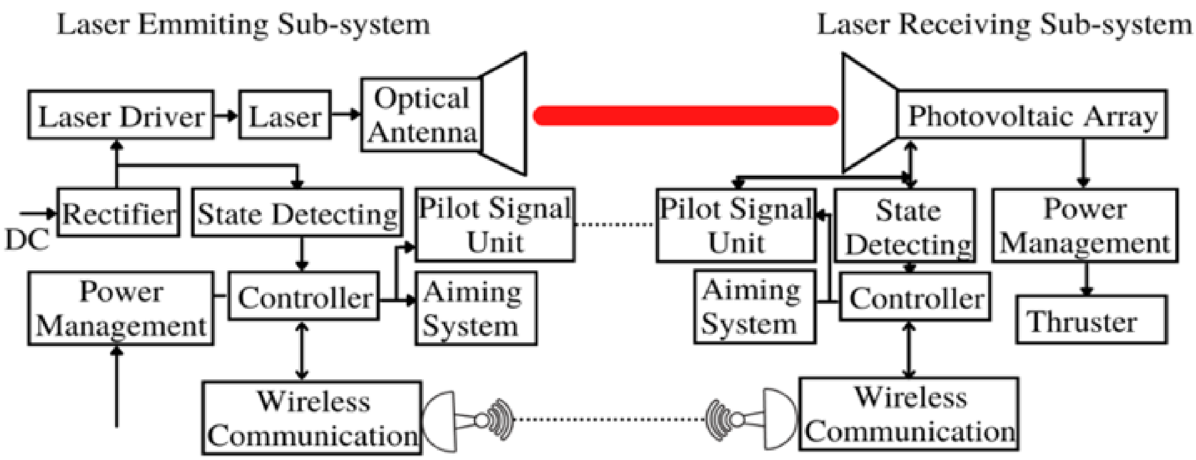


Figure 3.18 Laser Power Transmission subsystem

system and transmit through the optical antenna, as shown in Figure 3.18. The daughter satellite's PV cells receive the laser power and convert it to desired operational power, which will be utilized by the electro thruster, CPU, communication, and sub-mission payload. The designed E-Sat system output varies from 6.36 kW to 10 kW, significantly high for a typical operational satellite. However, recurring power can be adjusted according to the desired application by an inbuilt avionics system. Table 3.6 shows the supposed efficiency of the standard SSPS system using LPT. The beam collection and Laser to DC power efficiency depend on PV cells.

Table 3.6 Supposed Efficiency and Power for SSPS by Laser

Elements	***Efficiency (%)***	***Power Level (kW)***
PV array	30	P_{DC} = 21.4
DC to Laser power	56	P_L = 11.98
Beam collection	90	P_r = 10.78
Laser to DC Power	59	$P_{DC\ output}$ = 6.36

The lasers must be coherently arrayed at the source, or their beams may be incoherently combined at the receiver site. Generally, the diodes must be arrayed coherently and combined efficiently for transmission [82]. The applicable payload laser has three main categories, CW, pulsed and ultrafast, as CW lasers are used to produce undisruptive continuous waves of light for constant and very stable power. For example, the coherent beam from a 100 W laser diode of 808 nm wavelength gives the constant 58 % efficiency [91]. While pulse laser has several nanoseconds laser action time, the pulse is between 0.5 to 500 nanoseconds. Therefore, it stores the energy and releases energy very rapidly. The 975nm pulse diode delivers the repeatable rate of current pulses with 77 % output efficiency, while ultrafast are pulse diode with a pulse rate of 5 femtoseconds to 100 picoseconds [95, 96]. For example, the different types of LD with wavelength, operational current, operational voltage, and efficiency

Table 3.7 Different types of Laser with characteristics suitable for LPT

LD P_o (W)	λ (nm)	current (A)	voltage (V)	Efficiency (%)
100 [92]	975	10.5	24	77
200 [92]	915	12	35	48
250 [92]	915	18.5	27	50
300 [92]	940	70	1.7	58
400 [93]	808	60	20	45
400 [94]	915	60	20	45

are summarized in Table 3.7. The laser selection for effective transmission uses 0.8 um 1.64 um lasers. The 200 W CW laser with a diameter of 1 cm aperture, the max power density is 254.65 W/cm^2 using CW laser. Arranging 50 lasers coherently gives 127 W/cm^2. Using Chapter 2, Equations 2.7 and 2.8. The angle of divergence of the beam and final divergence is calculated in the Table 3.8. The angle of divergence is proportional to the wavelength. The final divergence of proposed laser is shown in the given table.

Table 3.8 The LD's angle of divergence of beam and final divergence

LD (W) P_O	λ (nm)	Angle of divergence of the beam (μrad)	Final Divergence (nm)
100	975	62.07	10.94
200	915	58.25	10.16
250	915	58.25	10.16
300	940	59.84	10.44
400	808	51.43	14.10
400	915	58.25	10.16
200	1050	66.85	18.32
200	1060	67.48	18.50

The Electricity to laser conversion Efficiency can be calculated using the following Equation (3.14), where the efficiency is calculated using Input laser power by the DC Power.

$$\eta_{el} = \frac{P_l}{P_{DC}} \tag{3.14}$$

The provided results and the associated efficiency for E-Sat 3 and E-Sat 4 are 40 ~ 62 %, an expected efficiency of 48%, which gives 10.27 and 18.832 kW, respectively for 200 W laser with a wavelength of 915 nm. A full E-Sat 3 and 4 efficiency is shown in Table 3.9.

Table 3.9 Supposed Efficiency and Power for E-Sat 3 and 4 by Laser

Elements	**Efficiency (%)**	**Power Level (kW)**	
		E-Sat 3	**E-Sat 4**
PV array	30	P_{DC} = 21.4	39.23
DC to Laser power	48	P_L = 10.27	18.832
Beam collection	90	P_r = 9.24	16.949
Laser to DC Power	59	$P_{DC\ output}$ = 5.45	10

Considering the transferring LPT from space to Earth, the losses are due to atmospheric and air losses, attenuation, and environmental effects. The Laser power attenuation depends on the distance and atmospheric quality. However, the E-Sat is specially designed for power transmission in a space-to-space system in LEO with a very small environment. So, the laser transmission efficiency can be calculated using the following Equation (3.15) and (3.16). Here $\propto$ is a laser attenuation constant; the size distribution of the scattering particle it is also depends on visibility.

$$\eta_{LT} = \frac{P_r}{P_l} = e^{-\alpha d} \tag{3.15}$$

$$\alpha = \frac{\sigma}{k} \frac{\lambda}{\chi}^{-\rho} \tag{3.16}$$

Laser to electricity conversation using PV cells array is shown in Figure 3.18, the PV cells voltage and current can be calculated using the load or payload work on customer/ daughter satellite. The load will satisfy the required power or 10 kW laser as payload for power-sharing between E-Sat and E-Sat. For example, if E-Sat o1s1 (o1: orbital plane number one, s1: satellite one) is generating and transmitting power using an SSO based to the a non- SSO based E-Sat o2s1 (o2: orbital plane number two, s1: satellite one) there will be a few minutes of eclipse time. Due to eclipse the power generation mechanism will not work because of non-linear power generation source at PV cells array as it is not in sunlight zone. In this situation, the pair of batteries will directly transmit the power to LD for the LPT system.

Furthermore, the required power for charging can be transmit using E-Sat o1s1. The power requirement for E-Sat o2s1 will require 1-10 kW to be transmitted from E-Sat o1s1. For this, the PV array will play an essential role in power receiving. This mechanism is also utilizing for old satellites with PV cells.

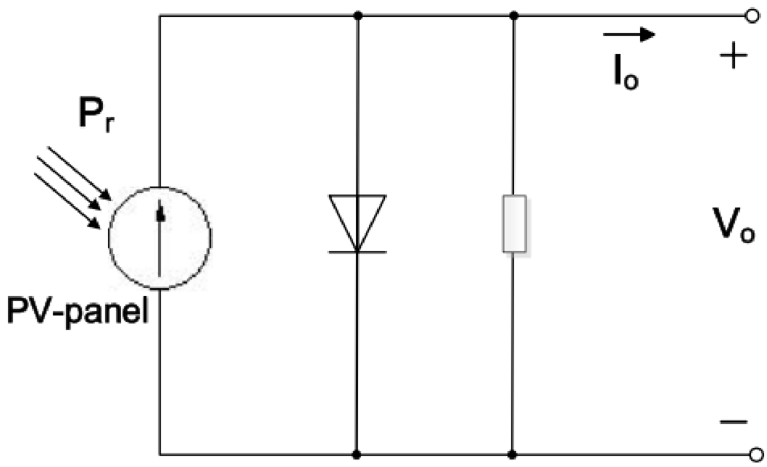


Figure 3.19 PV power conversion basic circuit

Figure 3.19 the PV voltage and current will define the receiving power system using Equation (3.17 – 3.20). In Equation (3.18), n is the PV ideality factor, h is planks constant 6.62607004 × 10-34 m^2 kg / s, and P_O is PV output power with I_O and V_O.

$$I_O = I_{SC} - I_S \left(e^{\frac{V_O}{V_m}} - 1 \right) \quad (3.17)$$

$$V_m = \frac{nkT}{q} \quad (3.18)$$

$$P_O = I_O \, V_O \quad (3.19)$$

$$\eta_{DC\ output} = \frac{P_O}{P_r} = \frac{I_O \, V_O}{P_r} \quad (3.20)$$

$\eta_{DC\ output}$ is the leser to electricity conversion efficiency, which gives the overall efficiency of the system approximately 6 % for the above configuration with 200 W laser LD. The laser beam is generated by individual laser with a 1 cm diameter, with a lens focal length of 100 m, beam quality M^2 is 1. By utilizing Equation (3.21 – 3.23) the LD's power conversion with collection of photonic energy is with beam size is calculated and represented signal gain and loss in Table 3.10.

Table 3.10 Signal gain and loss

LD P_o (W)	**λ (nm)**	**Current (A)**	**Voltage (V)**	**Efficiency (%)**	**E-Sat (DC-Laser)**	**Photon Energy (eV)**	**Beam spot size (mm)**	**Diffraction-limited angular resolution (μrad)**
100	975	10.5	24	77	26.28	1.27	12.4	119
200	915	12	35	48	18.83	1.36	11.7	112
250	915	18.5	27	50	19.62	1.36	11.7	112
300	940	70	1.7	58	22.75	1.32	12.0	115
400	808	60	20	45	17.65	1.53	10.3	98.6
400	915	60	20	45	17.65	1.36	11.7	112
200	1050					1.181	13.4	128
200	1060					1.17	13.5	129

$$2 \ \omega_0 = \frac{4M^2 \lambda f}{\pi D} \quad (3.21)$$

$$E_P = h \, v = \frac{h \, c}{\lambda} = h \, c \, k \quad (3.22)$$

Here D is the diameter of the lens, considered 1 cm. The photon energy calculation is calculated using Equation (3.22). Diffraction limited angular resolution of an optical laser system with the aperture diameter of 1 cm, and angle of divergence by Equation (3.23).

$$\alpha = 1.22\frac{\lambda}{D} \tag{3.23}$$

Table 3.11 Angle of divergence for 100 and 200 W LD with change in distance

LD (W) Po/p	λ (nm)	Angle of divergence of the beam (μrad)	Beam diameter After 1 m (mm)	Beam diameter After 10 m (mm)	Beam diameter After 100 m (mm)	Beam diameter After 1000 m (mm)	Beam diameter After 100 km (mm)	Beam diameter After 500 km (mm)
100	975	62.07	100	101 mm	106	162	6,307	31,135
200	915	58.25	100	101	106	158	5,925	29,225

Table 3.12 Area of Laser point for 100 and 200 W LD with change in distance

LD (W) Po/p	λ (nm)	Angle of divergence of the beam (μrad)	Area of laser point after 1 m (mm^2)	Area of laser point after 10 m (mm^2)	Area of laser point after 100 m (mm^2)	Area of laser point after 1000 m (mm^2)	Area of laser point after 100 km (mm^2)	Area of laser point after 500 km (mm^2)
100	975	62.07	7,863.73	7,951.7	8,859.2	20,629.8	31×10^6	761×10^6
200	915	58.25	7,863.13	7,945.75	8,795.62	19,668.78	27×10^6	670×10^6

Table 3.11 – 3.14 the angle of divergence, laser pointing, the intensity of laser and power density is showcase for the 100 W and 200 W laser with various frequency for the distance 1 meter to 500 km as the customer satellite change the relative position.

Table 3.13 Laser intensity for 100 and 200 W LD with change in distance

LD (W) Po/p	λ (nm)	Angle of divergence of the beam (μrad)	Laser intensity after 1 m (mW/mm^2)	Laser intensity after 10 m (mW/mm^2)	Laser intensity after 100 m (mW/mm^2)	Laser intensity after 1000 m	Laser intensity after 100 km (mW/mm^2)	Laser intensity after 500 km (mW/mm^2)
100	975	62.07	1,271	1,257	1,128	484	0.32	0.013
200	915	58.25	1,271	1,258	1,136	508	0.36	0.014

Table 3.14 Power density for 1050 and 1064 nm LD with change in distance

Distance (m)	**Power Density (W/ cm^2)**			
	100W (1050 nm)	**200W (1050 nm)**	**100W (1064 nm)**	**200W (1064 nm)**
1	7.12×10^9	3.56×10^9	6.9340×10^9	3.46×10^9
10	7.12×10^7	3.56×10^7	6.9340×10^7	3.46×10^7
100	7.12×10^5	3.56×10^5	6.9340×10^5	3.46×10^5
1,000	7.12×10^3	3.56×10^3	6.9340×10^3	3.46×10^3
5,000	284	142	277	138.68
50,000	2.84	1.42	2.77	1.38

Comparing the MPT and LPT systems with various frequencies, MPT is a better option in high-power transmission from space to Earth and is more mature than the LPT system. However, with compact design, where small power can be created within specific mass, the LPT can be utilize and provide significant power to small satellites using micro satellite SSPS structure as E-Sat. The rectenna size is bigger than the transmitting antenna for better propagation in the MPT system. At the same time, the LPT system receiving part is small and can be pinpoint for higher distances.

The most commonly used Laser is Nd: YAG laser, consisting of a diode with a plugin efficiency of up to 80 % at 795 - 850 nm. A combination of thousands of laser diodes allows more power transmission accuracy and efficiency in extensive power transmission.

The lasers must be coherently arrayed at the source, or their beams may be incoherently combined at the receiver site. Generally, the diodes must be arrayed coherently and combined efficiently for transmission [82]. As a result, the LPT system is not yet mature enough and has low transmission conversation efficiency than the MPT system for high power transmission, as shown in Figure 3.20. However, the LPT system can have long-distance transmission compared to MPT [83].

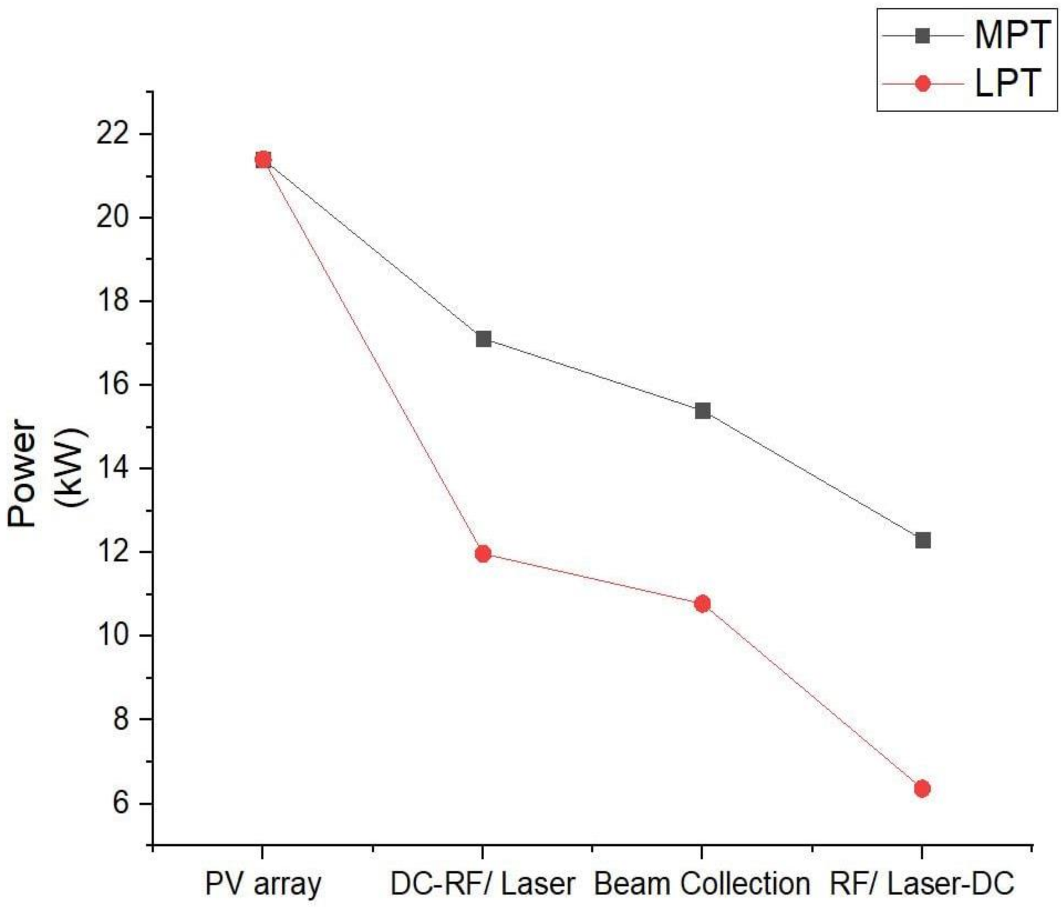


Figure 3.20 Generation and Transmission of Power between MPT and LPT

3.6.3. Laser Driver circuit

An essential system for transmission required a pulse triggering sequence and timing driver circuit. It will allow the exact timing of laser propagation to object satellite and with efficiently required thrust. A simplified diagram provides a means for understanding the operation of the Laser a Laser nomenclature of its input and output signals. A laser diode required a laser driver circuit vital to keep the Laser drive easy to control. Figure 3.21, the schematic circuit is for the laser diode driver is shown below, designed with a 75 W laser. Due to constraints with software, we used only a 75 W laser system. It is a simplified laser diode circuit for small watt laser and can be adjusted using variable LCR components according to designing requirements. The multiple transistors, diode and capacitor are used to enhance triggering pulses in its simple design in Figures 3.21.

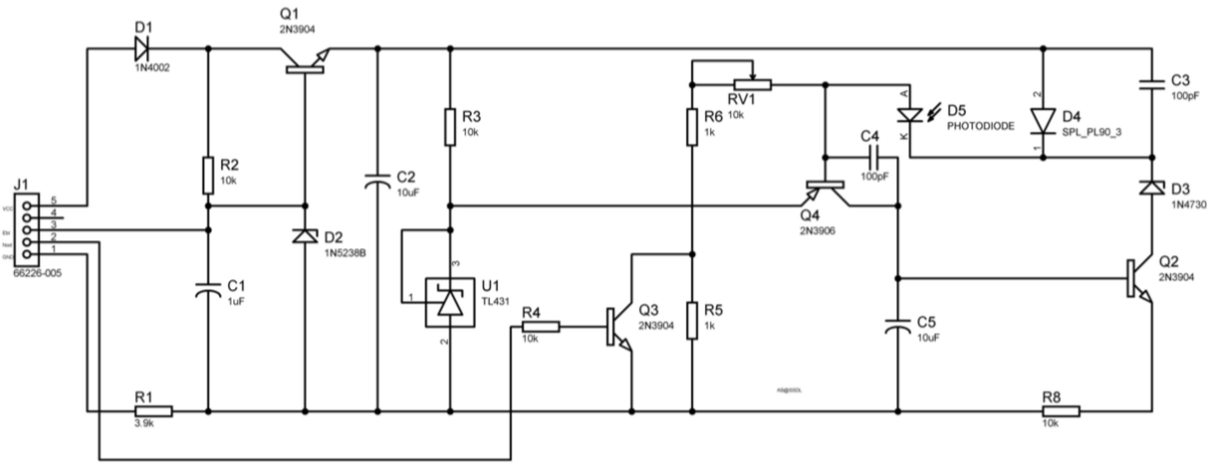


Figure 3.21 The schematic circuit is for the Laser Diode Driver

Typically, adjusting the power of laser transmission can be achieved by changing the value of electricity as the electricity is propositional to the radiated flux or power. Desirable output adjustment using circuit implementation can be achieved. As shown in Figure 3.22.

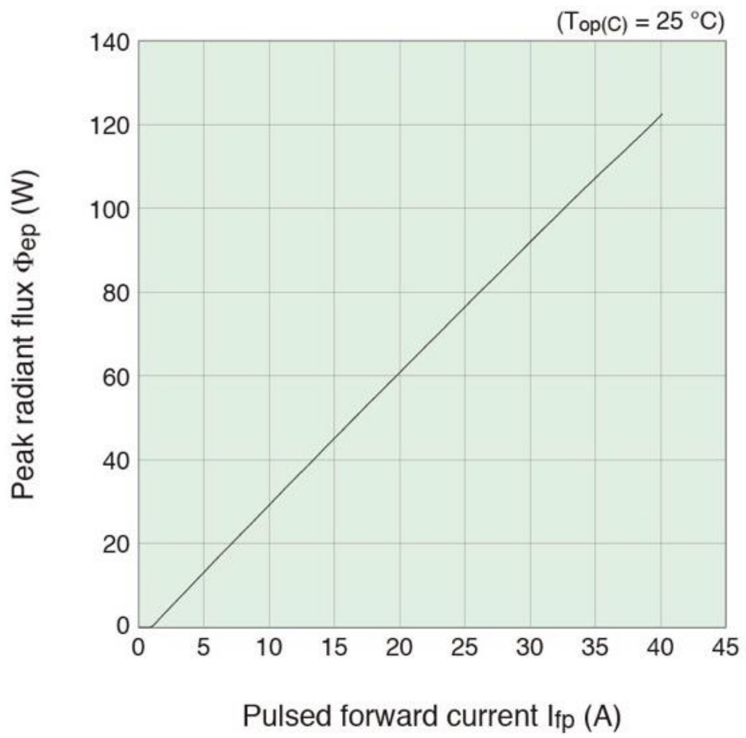


Figure 3.22. Radiant Output Power vs. Operating Current

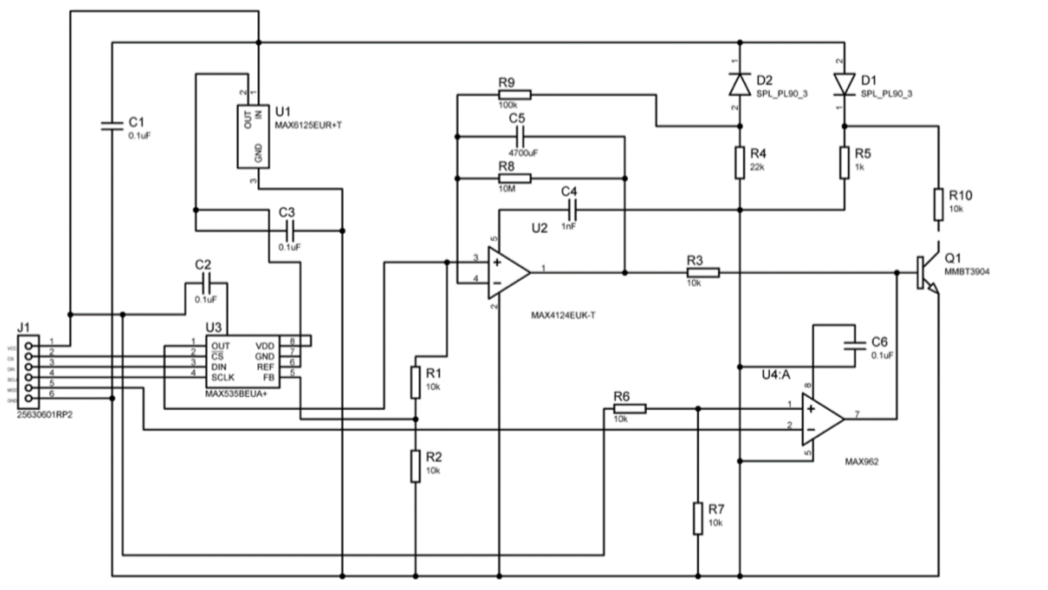


Figure 3.23. Laser Driver Diode Circuit

The following Figure 3.22 is the schematic circuit design of two LD with an amplifier, DAC IC, and pulsed trigger circuit as the driver circuit. The circuit containing a feedback loop for a voltage-driven system to fill the transmission requirement can be seen in Figure 3.23.

3.7. Power Management System in Spacecraft

The Designing of E-Sat with subcomponents power system distribution and management must maintain the payload in orbit. The PMS meets the restrictive voltage specifications of microprocessors, microcontrollers, PV, sensors, payload, and other harsh-environment space applications. Typically, PMS includes PV array, batteries, wire, heater, cooler, heating pipe, and subsystem for operational uses. The SSO-based E-Sat will generate electricity with solar irradiance of 1360 W/m^2. With an assumption of 30 % efficiency of PV array for power generation in E-Sat. Pair of batteries for backup is required for emergency and the E-Sat for storing energy for eclipse time to serve the required power at payload and onboard subsystem of satellites, NiCd, AgZn, NiH with a specific energy of 30, 60, 100 in W-hour per kg, respectively. Moreover, Lithium-ion batteries that are primarily in use can be optional for power storage. The connecting wires, transmission wires, sensor system, payload subsystem, motors, and onboard data handling system are also considered in PMS for regulating and controlling temperature and maintaining satellites properly in every condition.

3.8. Attitude and Orbital Control Subsystem

The AOCS performs and provides necessary information of the satellite’s orientation and orbital position using a sensor system to locate its position in the desired orbit using actuators accurately. It monitors and controls during the orbital maneuver and points the desired direction of sight for payload and solar PV cells array. An AOCS subsystem is a critical component of any SC. It contains the AOCS sensors, GNC, Filtration and processing unit, actuator management unit combination of AOCS actuators, and GNC actuators shown in Figure 3.24.

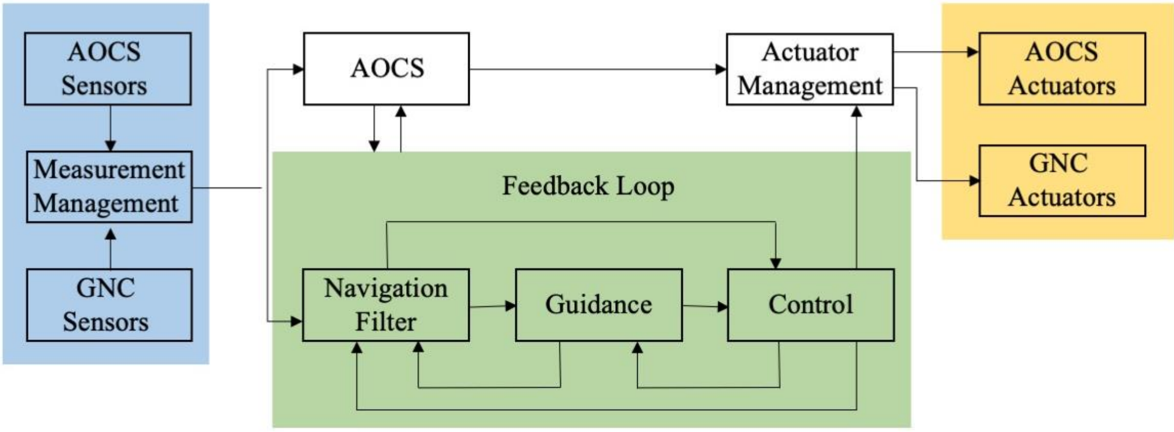


Figure 3.24 Block diagram of SC's AOCS

The programmed software is uploaded to the OBDH computer; it includes navigation filtration, guidance, and control algorithms. To understand the AOCS system, it is essential to know the attitude and orbital motion, parameters, and equation fully elaborated in appendix B.

Despite restrictions such as eclipses, a system of sensors is necessary to calculate a spacecraft's attitude, particularly rates and angular displacement. The attitude data must be updated regularly and with adequate precision. Sensors are divided into two categories:

(I) Reference sensors offer a reference or ‘datum' of an element's direction, including the Sun, a planet, or a star. However, this might be disrupted by an eclipse.

(ii) inertial sensors may give continuous attitude measurements but require attitude or calibrate correction for reference sensors due to inaccuracies, keeping the attitude error within highly narrow bands. A system of sensors, including reference and inertial sensors is utilized for a spacecraft adopting a balance of capability, mass/power utilization, and cost due to the numerous sensor ideas and restrictions [97].

3.8.1. Sun Sensor

Sun Sensor is used for detecting the orientation of the Sun to the SC. For SSPS, sun sensors play a significant role because of the continuous power generation system. The E-Sat is always turning its PV cells to 90 degrees to the Sun. For spacecraft in LEO to GEO orbits, the Sun is a bright body that may be represented as a point source in the sky (arc radius of 0.267°). As a result, distinguishing the Sun from other stars and planets is relatively straightforward. As a result, numerous Sun sensors have been developed over time, ranging from simple approaches that purely detect the incidence or deficiency of the Sun intensity to complex technologies that pinpoint the Sun's orientation to a few hundred of a degree of precision. These sensors can be classified into three basic categories: the Sun presence detector, analog sensor and digital sensor. The following sections briefly explain the operation of the sensors, including the hardware involved for better understanding. The sun sensor is divided into three classes;

1. Sun presence detector
2. Analog sensor
3. Digital sensor

3.8.1.1 Sun Presence Sensor

Just as its name suggests, the sun presence sensor detects when the sun vector enters its area of view. The input of the sun sensor is in light with a specific wavelength in the field of view which varies with the particular application for the SC. It is highly used for on/off mechanism for the telescope, star sensor and highly sensitive payload to sunlight can be seen Figure 3.25.

When the Sun is in its restricted field of vision zone, the shadow bar sensor recognizes it and sends a signal to the control units, covering the sensitive equipment. The sun presence sensor can be a slit and V-slit-based design. The slit sun presence sensor's photocells lies beneath slits. When the sun vector lies on the planes of slits, it generates the output signal in Figure 3.26. In a spinning satellite, two slit detectors are used to detect the spin axis of SC. When SC is spinning, the normal vector rotates, and Equation (3.24) give a cosine waveform, as the output can be found when n the normal vector and s as sun vector in the frame of the sensor box satisfy the following equation eight conditions

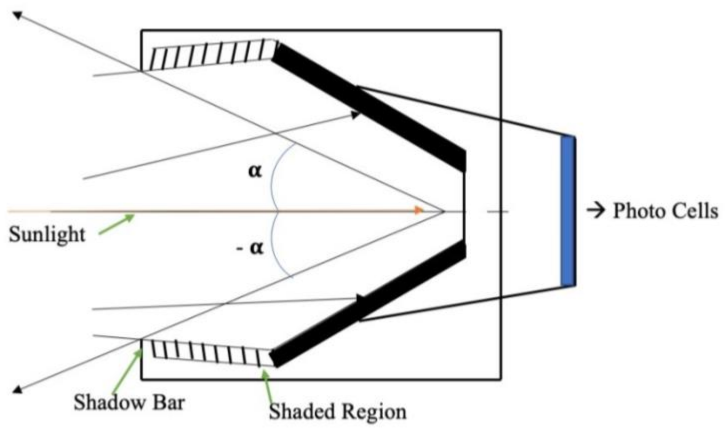


Figure 3.25 Sketch diagram of shadow bar sun sensor

$$n.s= 0 \tag{3.24}$$

V-slit sun sensors have two slits to find the spin axis and orientation to the Sun. The longitudinal slit leads straight to the spin axis, whereas the tilt slit follows an inclination of i to the spin axis. The Sun direction intersects the axis of the longitude slit and the tilt slit once at every rotation of the SC.

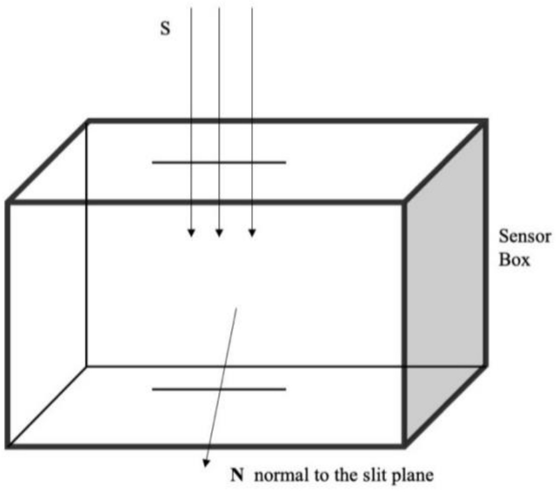


Figure 3.26 Two Slit sun sensor detector

Figure the sun sensor frame is X_{sb}, Y_{sb}, Z_{sb}, and SC body frame x_b, y_b, z_b. As their two slits in this, both normal and Sun vector should be zero, as shown in Equation (3.25). Figure 3.27 shows the sun angle with a spin angle of 45 degrees for two V-slits presence sun sensors.

$$n_1.s = 0 \qquad n_2.s = 0 \tag{3.25}$$

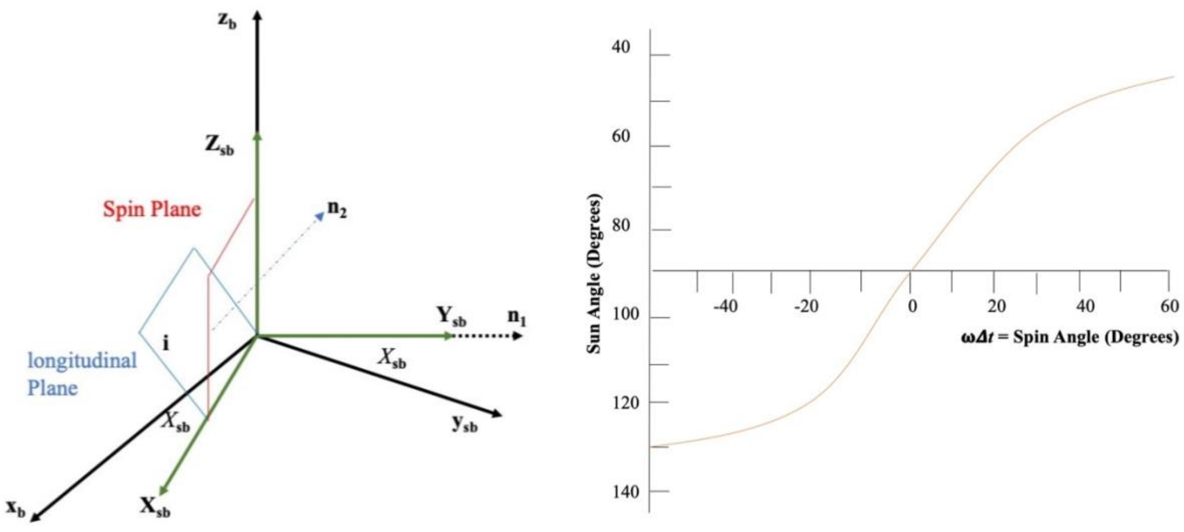


Figure 3.27 V slit sun presence detector

3.8.1.2 Analogue Sun Sensor

The primary principle for an analog sun sensor is to have an energy flux on the photo cell's surface is proportionate to the cosine of the Sun vector's incidence angle. It is also called the cosine law indicator. A single photodetector is used for single and two-dimensional sun vector representation by satisfying the computing Equation (3.26). The measurement of generated output current to the incident angle of sunlight lead towards the orientation of SC to the normal vector. A second solar cell is placed with its optical axis perpendicular to the first cell's axis for more accuracy, as shown in Equation (3.27)—two-photo detector sensor used for three-dimensional sun directional vector as shown in Figure 3.28 and 3.29. In a Single and two-axis analog sun sensor or cosine detector, A_c is a constant of the solar cell being used.

$$I(\alpha) = A_c \cos(\alpha) \tag{3.26}$$

$$I(\beta) = A_c \, cos(\beta) \tag{3.27}$$

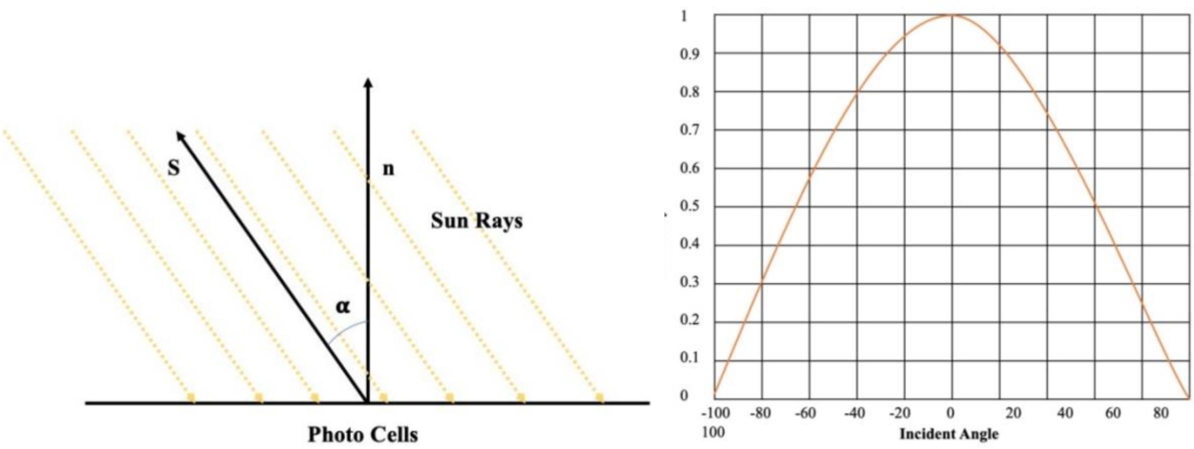


Figure 3.28 (a) Geometry (b) incident Sun angle as a function of current of single-axis analog sensor

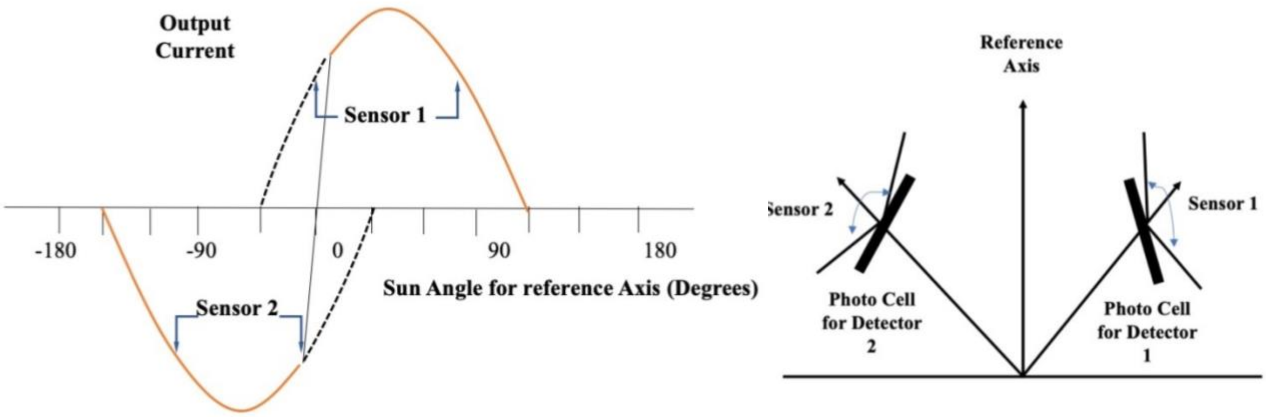


Figure 3.29 (a) Incident sun angle as function sensor (b) Geometry of single Axis analog sensor

For example, the Advanced Coarse Sun Sensor from Solar MEMS technology is used for LEO and GEO missions for sun tracking and attitude determination, as shown in Figure 3.30. It measures the sunlight in two orthogonal axes. It has ± 60 degrees of view angle with an accuracy of <1 degree for the mission of -40 to 85 °C.

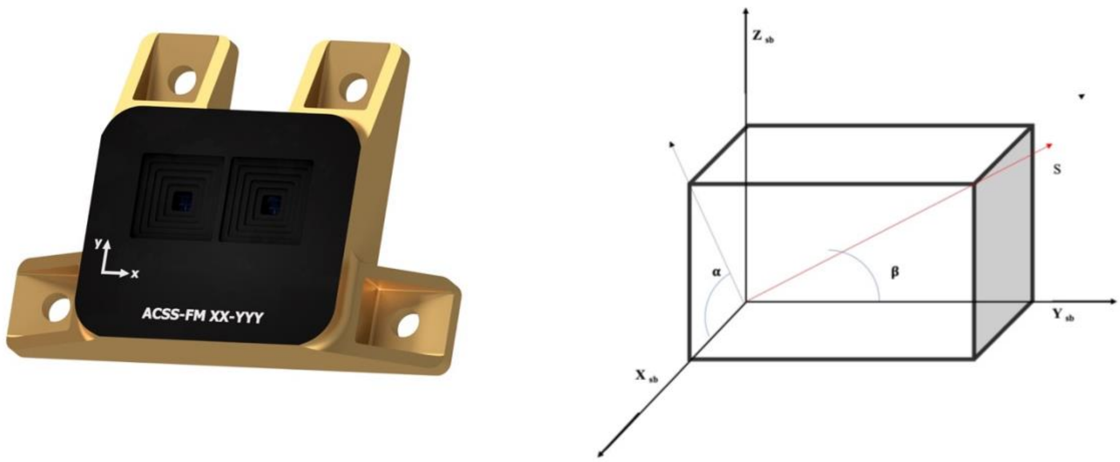


Figure 3.30 Two-axis Analog Cosine Sensor

3.8.1.3. Digital Sun Sensor

A digital sun sensor gives more precise and accurate satellite orientation to the Sun's ray than the analog sun sensor. To get accurate data, a two digital Sun sensor use at 90 degrees to the optical plane. It contains the optical head with a narrow slit as a sensor and a signal processing unit.

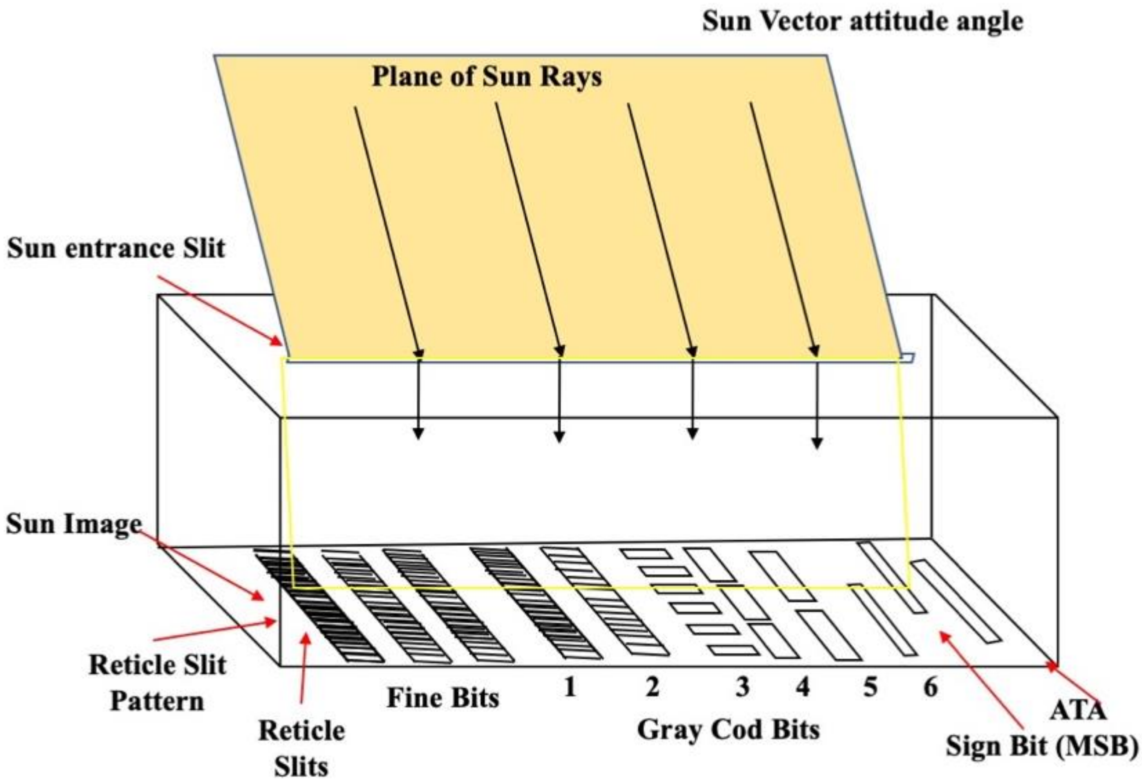


Figure 3.31 A fundamental principle of operation of Mask Sensor

As shown in Figure 3.31, the ATA is half the width of the photocells, which leads to the half current generated through the light passing through the slits. ATA's current is considered the reference point for calculating the position, and the other slit is illumination from the Sun. If the output voltage from any slit is greater than the two times of ATA, then the system passes and, on the signal, as ON bit else OFF.

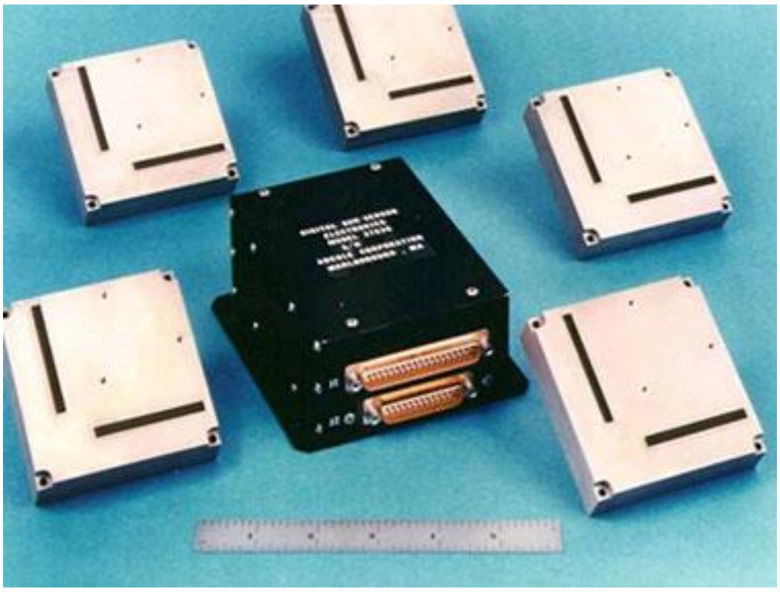


Figure 3.32 Digital Sun Sensor

With advancements in electronics and micro-electro-mechanical technology, small and compact size with low power budget, an active pixel sensor is being used with the CMOS and CCD technology. For example, as shown in Figure 3.32, Redwire Digital Sun sensor from Adcole Space Corp. Provide a 128 $\times$128 degree of field view with an accuracy of ± 0.25 degree [98].

3.8.2. Earth Sensor

Earth sensor provides the relative attitude of an SC to the Earth. As at nearby Earth orbit, it is challenging to make Earth as point source target compares to Sun. At LEO, the SC's 40% field of view is only occupied by Earth. In this scenario, the reflected light and albedo radiation can be utilized as the reference source for finding the attitude and orientation of spacecraft. From Earth, the reflected light is widely dependent on the surface as the reflection and absorption of wavelength are standard due to soil, water, forest, vegetation, and others and the time of the day. In this scenario, the IR sensor plays a significant role in keeping Earth as a reference point with the change in spectrum. The spectral range of 14 to 16 μm is highly used for horizon sensors. It has uniform distribution without any distraction due to the surface or day and night on Earth. The Earth sensor works on finding the horizon and keeping it as a reference using optical instruments with a processing unit. It is mainly placed at the spinning platform of SC, so when the SC spin and rotates, it detects the IR from Earth, or if it is on the steerable system, it will scan the sky as in Figure 3.33.

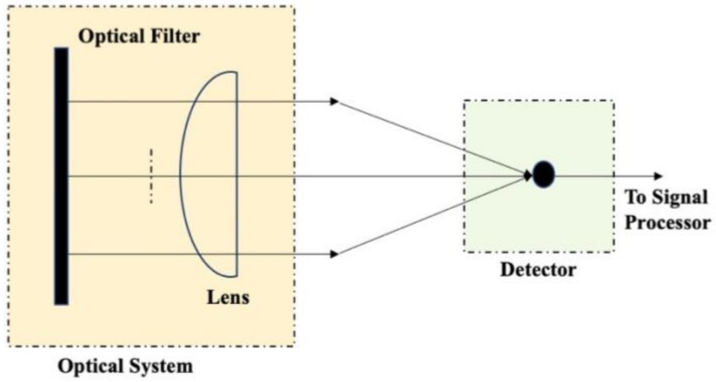


Figure 3.33 Sketch of operational Mask Sensor

The Earth sensor typically used a bolometer with a specific resistance or varies to incident radiation. Some of the detectors are near IR detectors, pyroelectric and thermopile devices. The sensitivity and accuracy of this detector are defined with the incident of radiation, as shown in Figure 3.34. The primary functionality for scanning Earth by making a cone by creating pulse generated per every rotation of the sensor scan. For example, as shown in Figure 3.35, IRES from Leonardo Airborne and Space System, two-axis IR earth horizon sensor for three-axis stabilization of spacecraft. It operates at a wavelength of 14 - 16.25 μm which works in temperatures of - 30 to + 60 °C. Operations include pointing, acquisition and chord mode in linear range as follow ±5.5° pitch; ±2.5° roll; ±11° pitch; ±2.5°; and ±23° pitch; ±14° roll respectively with an accuracy of < 0.05 ° random error and having digital and serial interface [99].

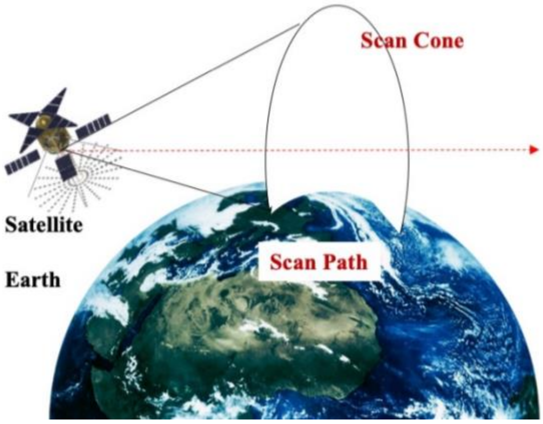


Figure 3.34 Horizon Crossing Indicator

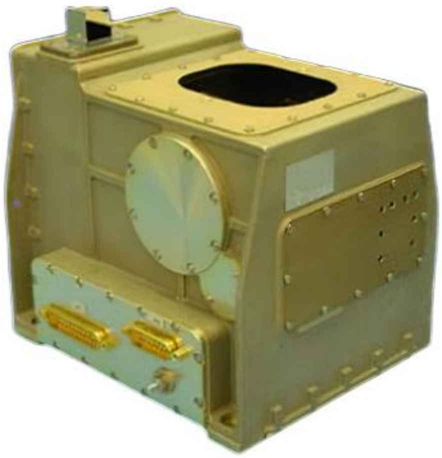


Figure 3.35 A two-axis IR Earth Horizon Sensor

3.8.3. Magnetometers

Magnetometers are also one of the foremost popular attitude determination sensors for satellites in LEO. This is due to their ease of use, durability, low cost, and negligible mass. They are used to measure the local magnetic field's strength and direction. The attitude of its satellite may be calculated using this information and a model of the Earth's magnetic field. Magnetometers can only offer coarse readings since the field is not well mapped and has numerous anomalies. Thus, they are generally coupled with additional sensors like star cameras and sun sensors. So, if magnetorquers are utilized for the stumbling period of the satellite, magnetometers are now used extensively. The firing of the magnetorquers must be timed to allow for the magnetic field breakdown before obtaining any data. The magnetometer's location must be carefully considered. It must be situated away from any potential sources of noise. As a result, they are frequently seen at the extremities of extendible booms.

3.8.4. Gyro

Gyros are used to calculate a vehicle's angular rates without the requirement for an external or objective reference. If a spacecraft's attitude is measured using an earth or sun sensor, the angular rates of the satellite's primary axes may be calculated by subtracting the sensor's angular position data. If the spacecraft encounters an eclipse, this will be an issue since it will no longer be feasible to determine the satellite's attitude and control it without using gyros-rate sensors. Another purpose for employing gyros-rate sensors is to regulate a spacecraft's angular rate and its angular location. While differentiating angular position with satellite other sensor creates noisy data, which further affects the AOCS feedback system. Gyros are mostly made of spinning wheel to create the rotating moments of inertia inside the SC. The other drawback of is gyro sensor is that its moving parts limit its operational lifetime. However, laser gyro, fiber optic gyro, and quartz rate sensors provide fewer noise data and a long lifetime. The noise is also dependent and inverse proportional to the range of measurement, less range higher accuracy, higher range lower accuracy. It is used in a cluster as one per axis, and the fourth one is for skewed configuration for redundancy. The cluster is called the inertial reference unit.

3.9. Attitude Control Actuators

The actuators are used for controlling the spacecraft in a feedback loop of AOCS sensors. This system controls the spacecraft and maintains a perfect orientation, attitude, and position inside the orbit. Furthermore, the actuators are divided into inertial and non-inertial actuators—the inertial actuators based on the generated torque to the angular moment of SC. Examples include, momentum wheels, reaction wheels, and control moment gyroscope. Non-inertial actuators are magnetic torques and thruster-based. As explained below;

- **Momentum Wheels**: provides a constant angular moment by stabilizing the gyroscopic moment in the ACS using orientation and controlling the spin axis moment to the regular rate.

- **Reaction Wheels:** It is used by increasing or decreasing the external torque generated by the wheel to stabilize the SC. The momentum and reaction wheels are based on the principle of conservation of the angular moment. In this way, they use a pair of wheels to produce torque and rotation movement in the angular momentum direction or the opposite direction inside the spacecraft, and the momentum is not changing or increasing/decreasing the system's total momentum. However, it is transferring the external momentum to the spacecraft. According to the system architecture, the inertia of the flywheel is selected to fulfill the requirement to the size, attitude, and mass of the SC. According to the spacecraft system analysis, it is well designed and well-engineered to get maximum wheel inertia for a given Mass.

A brushless DC motor is used to rotate side of the reaction and momentum wheels system. The complete assembly is integrated with the electronics subsystem with a pressurized cage, which helps to spin the wheel and acts as a pressure vessel to store the lubricant for easy movement. The lubricant also depends on the duty cycle of the wheel and the motor technology is incorporated to design the system. with the development of mechatronics, the design is now coming down to a low cost with the high life cycle. The momentum wheels are more significant than the reaction wheels, which help a smooth spin rate with a constant angular moment to maintain that attitude of an SC. A minimum of three non-coplanar wheels are used and are required to control the spacecraft. To avoid failure, a fourth wheel is incorporated to gain more accuracy.

- **Control Moment Gyroscope**: The momentum wheels use by the gimbal in one or two-axis provide a significant advantage of changing the direction of the momentum vector and a spinning wheel axis. It produces considerable angular torques to the system and is amplified by appropriately embedding the flywheel. For the gimbal, motors are used to produce angular momentum with the desired direction. The gimbal inertia with the system of analyzes spacecraft is according to mass, size, and attitude. The single gimbal momentum can be achieved using a single axis wheel with momentum in one axis direction. A double gimble system can work in an angular momentum vector with a degree of freedom with extra torque. However, it is costlier and more prominent with the complex design. The moment gyroscope is used inside a large spacecraft or space station to produce higher angular moment and higher stabilization under the external disturbance due to torques. Typically, 55 - 150 kg is used for 100- 1000 Nm of output torques.

- **Magnetic torquers**: Magnetic control is currently used in many space missions because of low-cost hardware and more reliable ACS, especially for a small satellite. The magnetic field of Earth with SC will generate a magnetic moment and control torque as shown in Equation (3.28). The direction of magnetic moment can be controlled with the proper sequence of magnetorquers firing. However, the Earth's magnetic field vector mainly relies on orbital position. Due to this, the control torque, which is orthogonal to the Earth’s magnetic field and magnetic moment of the spacecraft, is not controlled by the spacecraft axis in the specific orbital area. Another disadvantage of a magnetic control system is that desirable control might generate a disturbance in yaw, pitch, and roll for the other axis. To understand this, the Earth's magnetic field is assumed using the international geomagnetic reference field using a spherical harmonic model. The dipole vector can be expressed by Equation (3.28, 3.29) representing the geomagnetic field vector with the identic matrix.

$$N_M = M \times B \tag{3.28}$$

$$B = \nabla\left[\frac{R^T M_e}{R_s^3}\right] = \frac{[1-3RR^T]M_e}{R_s^3} \tag{3.29}$$

Magnetorquers can produce torque due to the number of coil winding s with the cross-section area of the coil while a certain number of current passes through the coil axis (u). Equation (3.30, 3.31). Usually only one magnetorquer is used per axis; however, with high altitude, the strength and accuracy are getting weaker to lower earth magnetic field. Magnetorquers do not need any fuel and external power system, and thus have an unlimited lifecycle.

$$M = n_c\, I\, A\, u \tag{3.30}$$

$$N_M = n_c\, I\, A\, (u \times B) \tag{3.31}$$

3.10. Thruster System

The external disturbance can be nullified using small thrusters on each axis. The thruster placement is also defined with the structure, mass, orbital configuration with the feedback thrust system for the AOCS control system, as shown in Figure 3.36. s compared to the Magnetic control system, the thrusters need an external consumable system. A consumable system can be a fuel to keep the perfect altitude and orbital parameters of spacecraft. The thrusters are expensive and heavy in size, and also complex to design. Examples include cold gas, electrohydrodynamic, electrostatic ion, electrospray, ion, and pulse plasma thruster. Many space agencies currently have highly researched and developed the electromagnetic propulsion system to provide high thrust efficiency to low mass ratio.

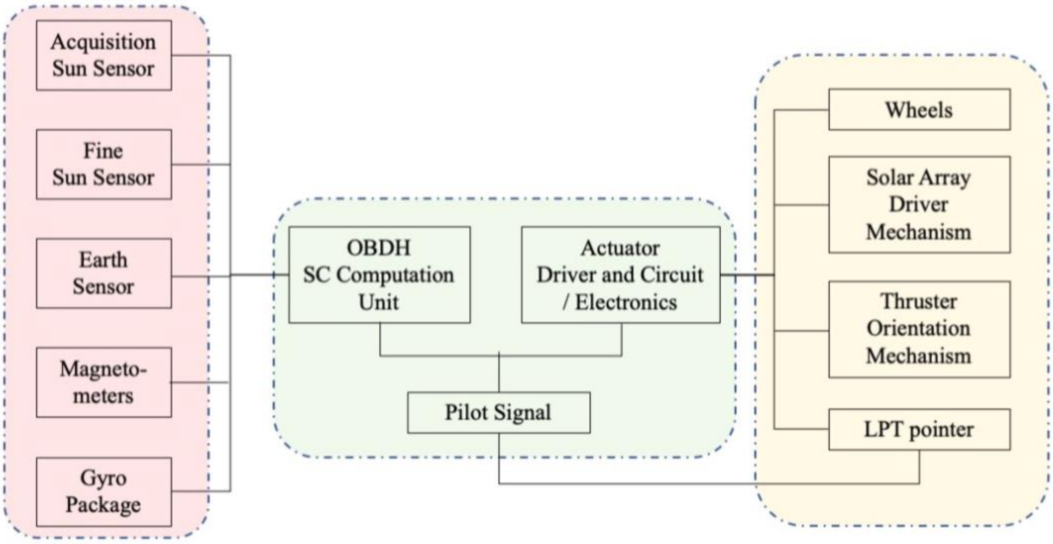


Figure 3.36 A feedback thruster Actuators system

3.11. Communication Subsystem

A communication system provides necessary instruction and transfer of required data between the satellite and ground stations. The communication subsystem consists of an antenna, RF, laser generator, encoder, decoder, transmitter, receiver, modulator, bandpass filters, and others to support the SC bus to provide constant support to the payload.

Antenna: It plays an essential role for SC trans reception of data and signals from the satellite or the ground. RF-based antennas are primarily used for front-end transceivers; nowadays, Laser communication-based optical antenna is also used. The RF antenna is an easy to transmit, less complex and highly adopted system in SC with the signal from long distances with big converge is on Earth can easily be transmitted without the worry of automation due to the Earth's atmosphere. The optical antenna is used for high-speed transmission as the light travels faster than the radio wave; it is chiefly used for inter-satellite linking, for example, Starlink. In the E-Sat, the RF signal is modulated with QPSK for downlink transmission and vice versa for the uplink signal in the satellite. This can be seen in Figure 3.37. The data is encrypted, modulated to the transmission system, and passed to the earth-based ground station through the microwave antenna. A minimum of two earth-based ground stations will be used for monitoring and controlling all E-Sat. The reception in the system is received by the receiving antenna and demodulated for operation and performs the command sent from the earth-based system. The modulation and demodulation of the system inside SC are called MODEM. The input of a signal to the MODEM required amplification and filtration of the signal for transceiving. TWTA/SSPA can achieve the amplification of the transmitted signal, and LNA can be used to remove noise for amplifying the receiving signal [100].

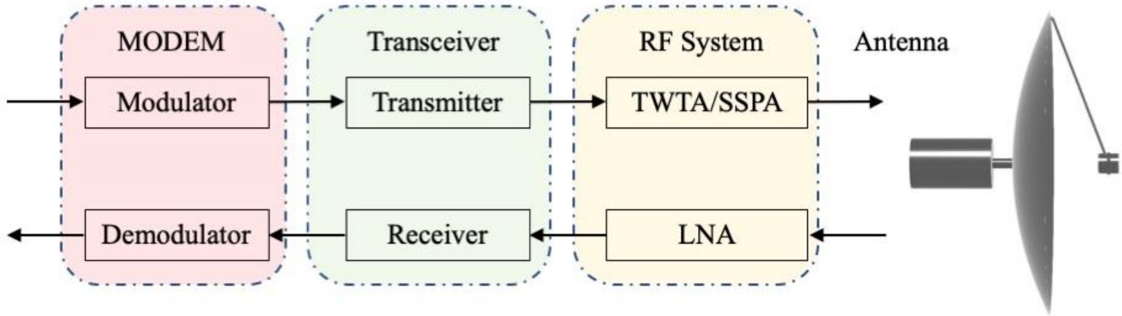


Figure 3.37 An RF front end transmission system

Interlinking of E-Sat to E-Sat and E-Sat to customer satellite will be achieved by sending an optical signal as pilot/ beacon beaming signal for confirmation of E-Orbit satellites using optical trans receiver system on each E-Sat and customer satellite. The satellite data will be transmitted and shared with each E-Sat for fast calibration and processing of power transmission configuration. A pilot signal will be delivered with the wavelength of 808 nm CW laser diode with the divergence of 9 mrad [101].

The following point should be considered for communication operation from the antenna.

- Extensive earth coverage with low-gain Omni & squinted-beam antennas.
- Satellite antennas with higher gain for medium earth range.
- Parabolic reflectors include multi-beam antennas with various feed systems to cover multiple users and limited areas.
- Deployable antennas, especially for achieving more finely focused beams and supporting considerably higher-gain multi-beam antennas.
- Scanning and hopping beams with phased array feed and phased array antennas.
- Laser beaming for pilot and interlink between satellite should be on a different wavelength of LPT wavelength.

3.12. Focusing of Payload Lens

Usually, for focusing of lens for optical resolution in-camera is to use with manual focus. Most mobile cameras use autofocus using CMOS chipset and digital zooming system with the current advancement in technology. The telescope uses the motor-based system in single-axis movement. A single axis-based movement lens for focusing on an object will be achieved by using LPT technology.

3.12.1. Using Stepper Motor

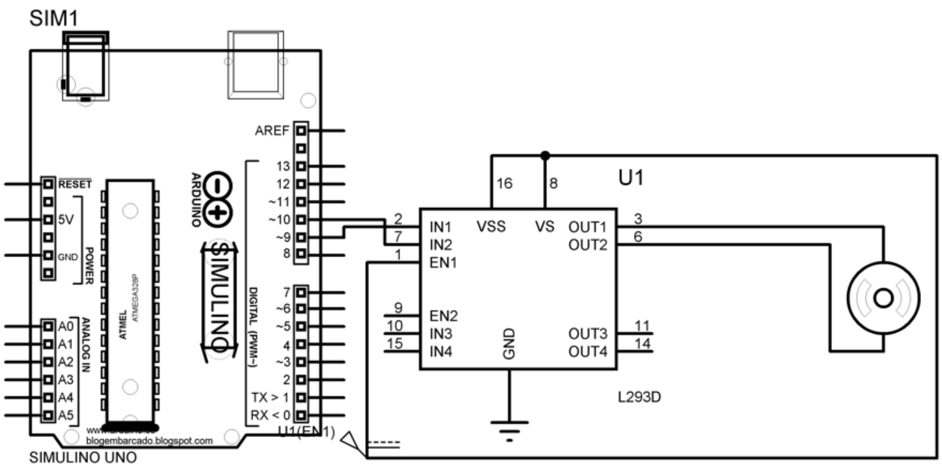


Figure 3.38 Arduino Uno based motor driver circuit

Stepper motor is widely used in robotics and IoT systems for single and multiple axis moments. A feedback loop of sensor and microprocessor system can be used for the stepper motor to move. A simplified example using analog components is shown in Figures 3.38 and

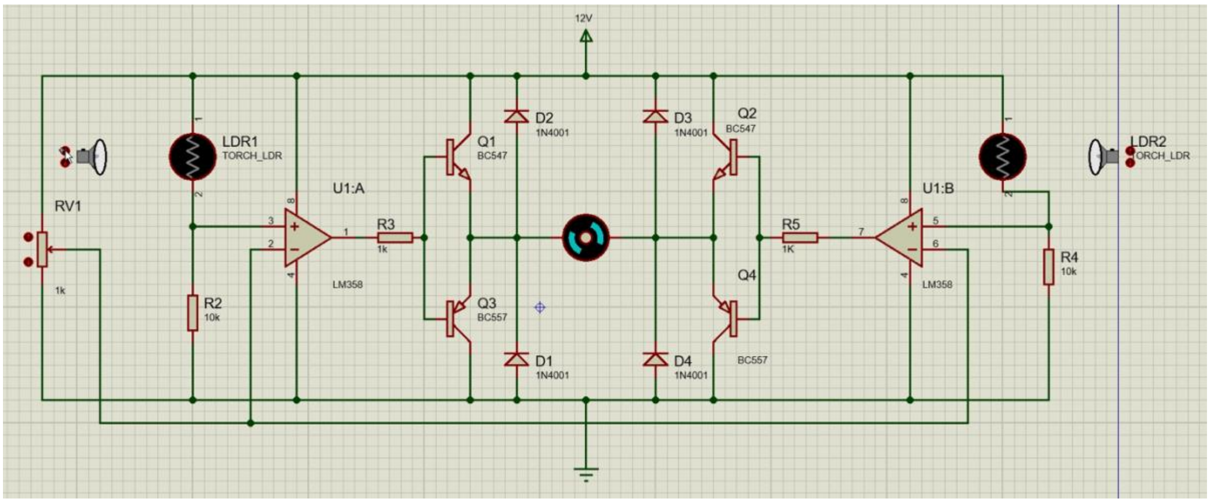


Figure 3.36 Feedback loop circuit for light diode for operation of motor

3.39. Moreover, for a basic experiment using an Arduino UNO board containing a microprocessor and small C programming design a simplified circuit for running a motor as shown in the Figure 3.36. Most actuators use DC and stepper motors for moving parts, such as antenna pointing mechanisms, linear actuators, rotary actuators, solar array drive assembly, optical filter wheel assembly, gambles, and many more.

3.12.2. Composite - Ceramic Inner Layer with Outer Piezoelectric Manipulator

Piezoelectric actuators form a new area between electronic and structural ceramics. An application like positioners, motors, and vibration suppressors. The precision manufacture of optical equipment, such as lasers and sensors, and the precision placement for manufacturing semiconductor chips, calibrated using solid-state actuators, are of the order of 0.1 μm. As far as traditional electromagnetic motors are concerned, small motors less than 1 cm are often needed in office or factory automation equipment and are difficult to manufacture with adequate energy efficiency. Ultrasonic motors with an insensitivity to scale are superior in the mini-motor region. Vibration using piezoelectric actuators, vibration in space systems and military vehicles using pie promising technique—for example, Multilayer Piezo – Electric Ceramic (Actuators) by Nikko.

3.11.3 Hydraulic Actuator/ Flexible Hydraulic Actuators

This actuator is primarily suitable for less mass lightweight and does not use motors due to the output torque inertia ratio limitation. The actuator is used with an oilproof rubber core tube with aramid fibers with sufficient internal pressure. It can generate up to 1500 N of output pressure. However, the focusing lens does not require that extension of pressure. The same concept of the oil actuator can be used for single axis moments like muscle movement.

3.11.4 Magnetic Field Manipulator

A magnetic field, which is transparent and relatively safe for small equipment and movement of the lens, is a powerful tool for remote actuation and wireless control of magnetic devices. It is highly used in robotics, especially small robots. Magnetic actuation systems offer a practical approach for remote equipment control via a dynamic magnetic field, as described in Figure 3.40.

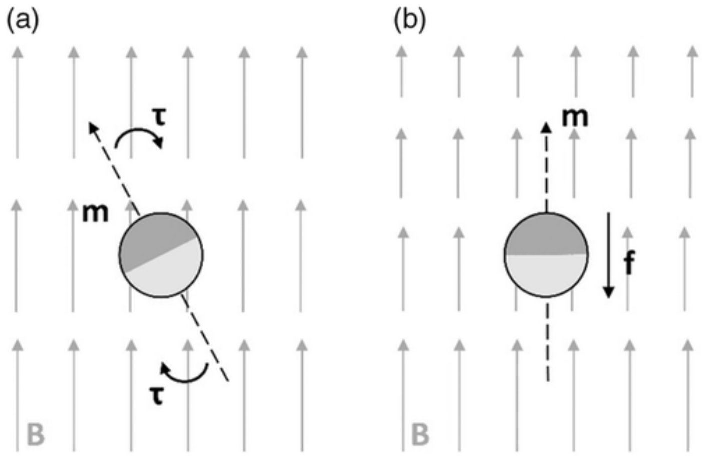


Figure 3.40 Diagram of magnetic interaction: a) pure torque under a uniform magnetic field; b) pure force under a nonuniform magnetic field.

3.13. Command and Data Handling Subsystem

C&DH subsystem is also referred to as OBDH subsystem, currently used by the digital processing unit to perform calculation, operational and internal communication as the brain of SC. The task and command are mainly performed on the OBDH system by the programmable software integrated and stored in the onboard storage system. It has the functionality to prepare the data synthesizer, encoding, encrypting, decrypting, transmission, and reception of a signal from the ground-based center or interlinking satellite data. It also processed with AOCS to find the appropriate solution for maintaining the spacecraft inside the perfect orbital system to perform the desired function and support the payload by giving continuous feedback to actuators and the AOCS sensor system. However, in the E-Sat, the C&DH main functionality, like any normal microsatellite, has to maintain the PV position to the normal to the sunlight vector, confirmation of other E-Sat interlinking and customer satellite within range by encoded and encrypted data sharing. Providing the information of satellite functionality like onboard battery status, the temperature of satellite, ACSS position, nearby satellite close approach and, fault management status to the ground-based segments. C&DH is also responsible for the cybersecurity, processing unit and SC subsystems health monitoring, up-downlinking of satellite [100]. All E-Sat work with the command passes through the Earth station transmitting and sending signals to E-Sat. A secondary station by another nearest satellite can be used to find the exact location of the new satellite using the pilot as a reference and handling communications traffic with QPSK modulation.

3.14. Handover and Interlinking of Satellite

Nowadays, interlinking and handover of LEO satellites are very much used for fast telecommunication between satellites and any point on earth station with the lowest latency. The handover can be achieved by Spotbeam, link layer, satellite, ISL, network layer, inter and intra. The hard and soft signal diversity, gateway and inter-system handover mobile station can reach a terrestrial network again which might be cheaper, has a lower latency.

3.15. Thermal Control Subsystem

Before launching any spacecraft in space, it is necessary to calculate and adequately analyze the thermal entity. The thermal control subsystem is an essential component of both spacecraft and instruments, keeping the temperature within the range needed to function correctly. The thermal control is achieved by balancing the energy emitted by the satellite as IR radiation against the energy dissipated by its internal electrical components plus the energy absorbed from the environment; atmospheric convection is absent in space. Establishing a thermal design for a satellite is usually a two-part process. The first step is to select a thermal design for the satellite's body, or primary enclosures, to serve as a thermal sink for all internal components. The second step is to select thermal designs for various components located within and outside the satellite body.

The essential heating elements in a spacecraft are instruments, heaters, or solar absorption when the only cause for heat loss is radiation. Thermal management uses heat pipes, thermal switches, heaters, and nozzles to transfer heat from the essential heat sources in the spacecraft to the radiators and away from the instruments. Traditionally, thermal management has become more passive; satellite is engineered so that the radiators are thermally attached to the heat source and retain a steady temperature. Heaters or louvers are also used to offer it some kind of control. For example, a standard solar panel in spacecraft operates from -100 to +150 °C. Due to a change in solar intensity and orbiting the Earth, the satellite comes and goes back to the sun side and returns to the eclipse side. As the E-Sat mission is LEO based at 900 km, it will complete a full rotation of the Earth in approximately 90-110 minutes. The halftime will be without Sun. This time the temperature suddenly drops down, and as soon as it comes in sun sight, the temperature will increase again. To avoid problems caused by a high glass coating and protection, a solar panel is required.

Moreover, the degradation of PV array is 3.75 % / year (Si) 2.75 % / year (Ge), due to change in radiation a 2.5 % year (Si), 2.75 % / year (Ge), the Aldo temperature effect from Earth to LEO spacecraft 18 °C, Average SSO orbit solar panel temperature gradient is above regular SC by 10 °C. However, the typical flat solar panel temperature at LEO is about 67 °, and the performance of conversion energy is getting low by 0.5 % per degree above 28° C. Meanwhile, if the satellite is Non-spinning, it will be 5° warmer a standard satellite. A working temperature describes the Table 3.15.

Table 3.15 Thermal performance of Satellite subsystem

Subsystem	***min (°C)***	***max (°C)***
Solar Panels	-100	150
VHF-UHF transceiver	-40	85
EPS	-40	85
OBC (CPU)	-40	85
ADAC	-20	85
Batteries	0	85
Laser	10	150
Microwave Transmitter	-40	85
Camera	0	60
Digital Electronics	-20	70
Analog Electronics	-20	70
IR detector	-269	35
Solid-state Particle detector	-35	35
Momentum wheel	-20	70

The only way to extract heat from a satellite is by removing heat from the radiator's surface to space. Traditionally, conventional louvers have been used to monitor the amount of cooling for a fixed-size radiator. The E-Sat system required a specific cooling and thermal dispersion system to continuously maintain temperature across the laser subsystem and electronics circuit as it is a 10 kW laser producing a pretty significant amount of heat. Maintaining the proper heat flow will require monitoring the cooling system across all subsystems, including Laser, OBDH, AOCS, and circuitry. It can be optimized and controlled by extricating heat using a heat sink, radiator, and other methods. There are two types of thermal control systems:

The ATCS uses a mechanically pumped fluid to perform heat transfer. Although this approach is more complex, the ATCS can handle much greater heat loads and provide a higher degree of control over how heat loads are managed. The PTCS consists of insulation, coatings, heaters, and heat-pipe radiators. Its components generally have few operational requirements and require low maintenance. In addition, PTCS components are also less complex and are easier to implement. There is different type of system is majorly used:

- Radiator
- Multilayer insulation
- Heat pipe
- Capillary pumped loop
- Thermal strap
- Heater
- Thermostats
- Thermoelectric cooling system
- Mechanical pumping
- Heat Sink:

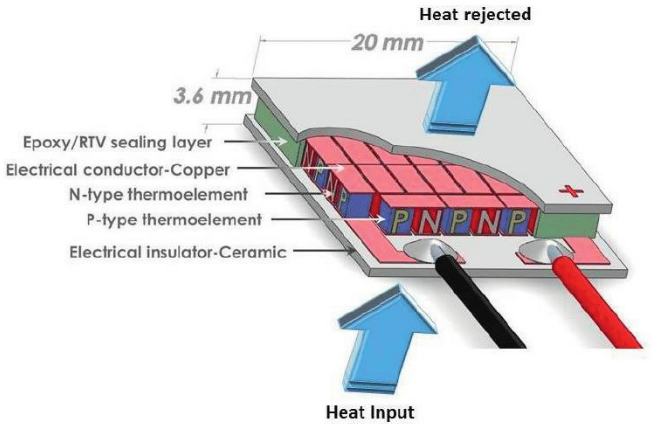


Figure 3.41 Thermoelectric Generator, Cooler

The temperature monitoring circuit can monitor temperature change by samples with eight thermistors, as in Figure 3.42. Six of these thermistors are located on the satellite's exterior to measure the temperature of the body-mounted solar panels. The other two thermistors will be located inside the satellite, one on the battery and the other on the transmitter. These circuits consist of the temperature monitoring circuit, the data/morse multiplexing circuit and transceiver, and the nichrome wire burn circuit for antenna deployment.

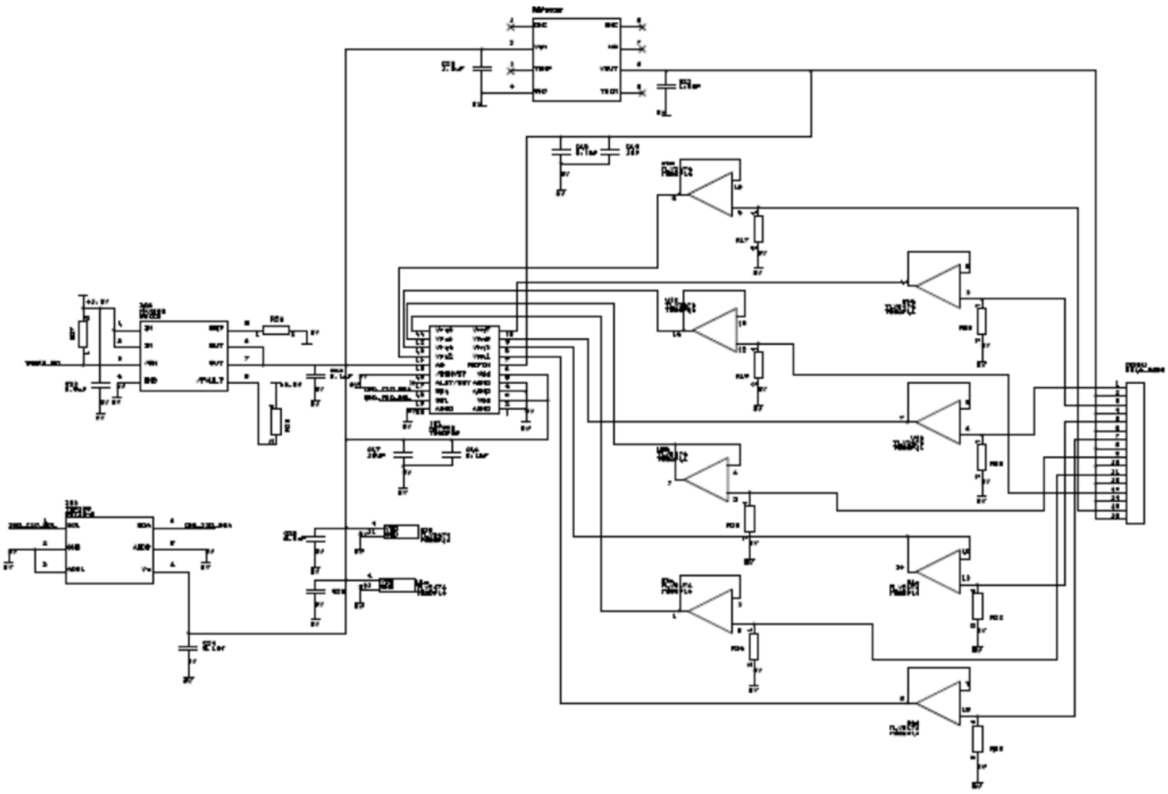


Figure 3.42 Temperature monitoring circuit

3.16. Orbital Perturbation Parameters

The E-Sat in orbit is experiencing the perturbation from different elements in orbit, likewise J_2, J_{22}, LGA, SGA, SRP, and drag due to atmosphere. The atmospheric drag is one of the most significant factors for deorbiting any spacecraft in LEO. While the SSPS system with mega construction in orbit placed in GEO experience massive perturbation due to the SRP and SGA. Using Figure 3.39, atmospheric drag highly works on the LEO satellite. As the altitude increases, the density of the atmosphere decreases. While SRP is constant, the SGA and LGA gave high perturbation near Earth with high altitudes. With the use of the Astrolib package designed by Professor Dr. T. Hanada, the perturbation is calculated for the 250 kg E-Sat at 900 km, and the test is run for the SSO orbit with an inclination of 98.6 ° using JAXA based model system for 2010 - 2011 condition as shown in Figure 3.41.

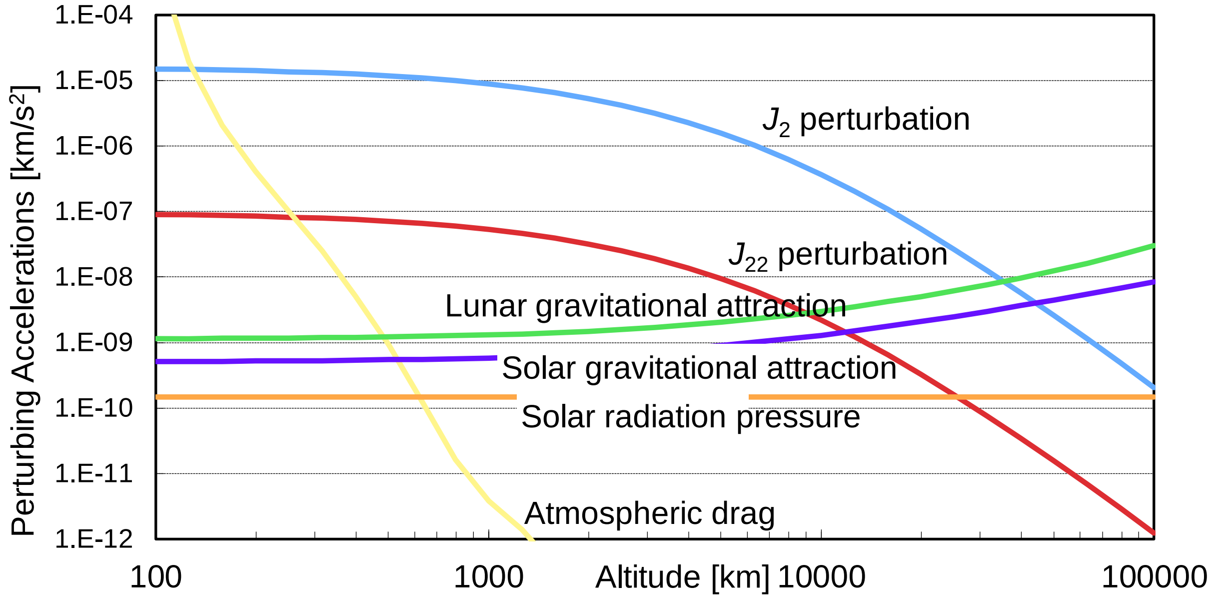


Figure 3.43 Perturbation near Earth

The E-Sat experience SGA and LGA as in Figure 3.42 and nominal drag due to atmosphere in one year of lifetime, as shown in Figure 3.43. The system is a small microsatellite that benefits traditional SSPS design, which is explained in the appendix. The satellite with current drag in the system can be overcome by a 0.1 N thrust system by a feedback loop with AOCS and GNC system.

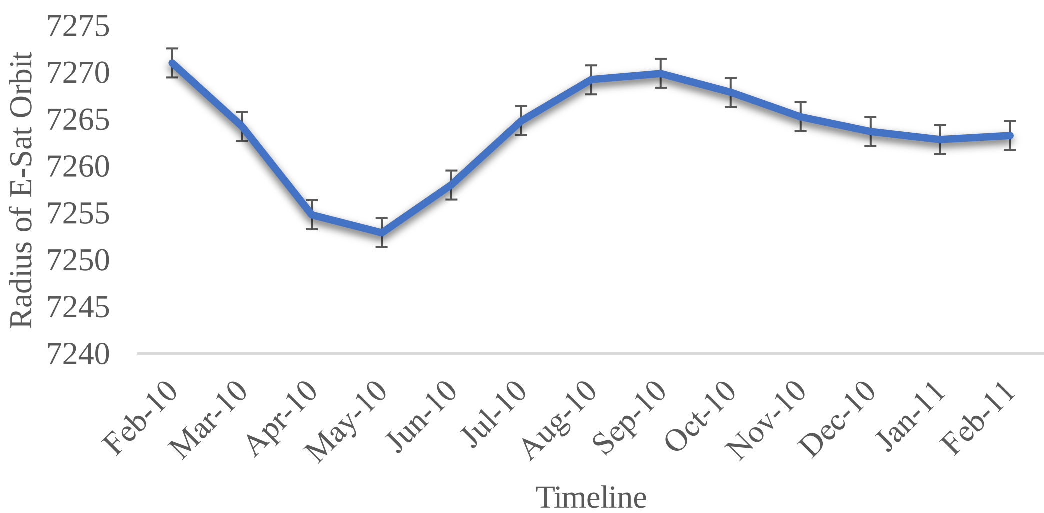


Figure 3.44 Overall Perturbation on E-Sat

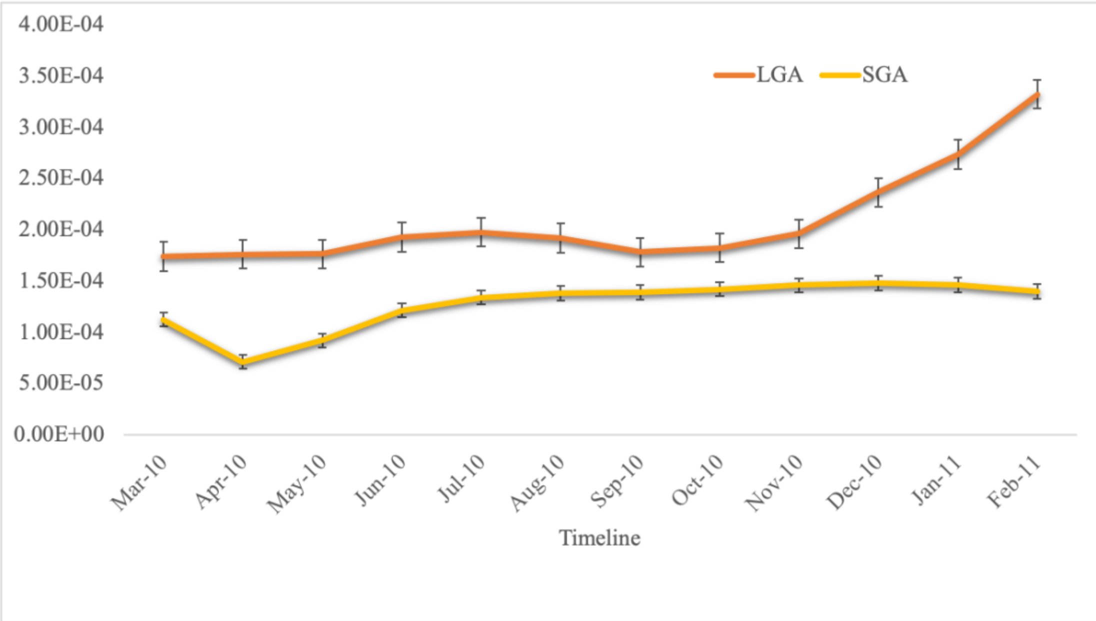


Figure 3.45 LGA and SGA effects on E-Sat for one year

Figure 3.46 Atmospheric Drag on E-Sat for one year

Chapter 4

Energy Orbit

Chapter 4

Energy Orbit

4.1. Introduction to Energy Orbit

E-Orbit is an LPT-based small SSPS constellation whose primary function is to generate, convert and transmit electricity to objective locations. E-Orbit functionality is based on the transmission of electricity in space to provide energy to the satellite industry. Currently, there are thousands of satellites planned by many space companies and agencies in LEO for better communication, GIS, space and Earth weather, and navigation system. It will increase up to 75,000 by the end of 2035, as in Figure 4.1. E-Orbit is capable of handling the PMS of these satellites. The constellation is divided into four phases for full E-Orbit demonstration.

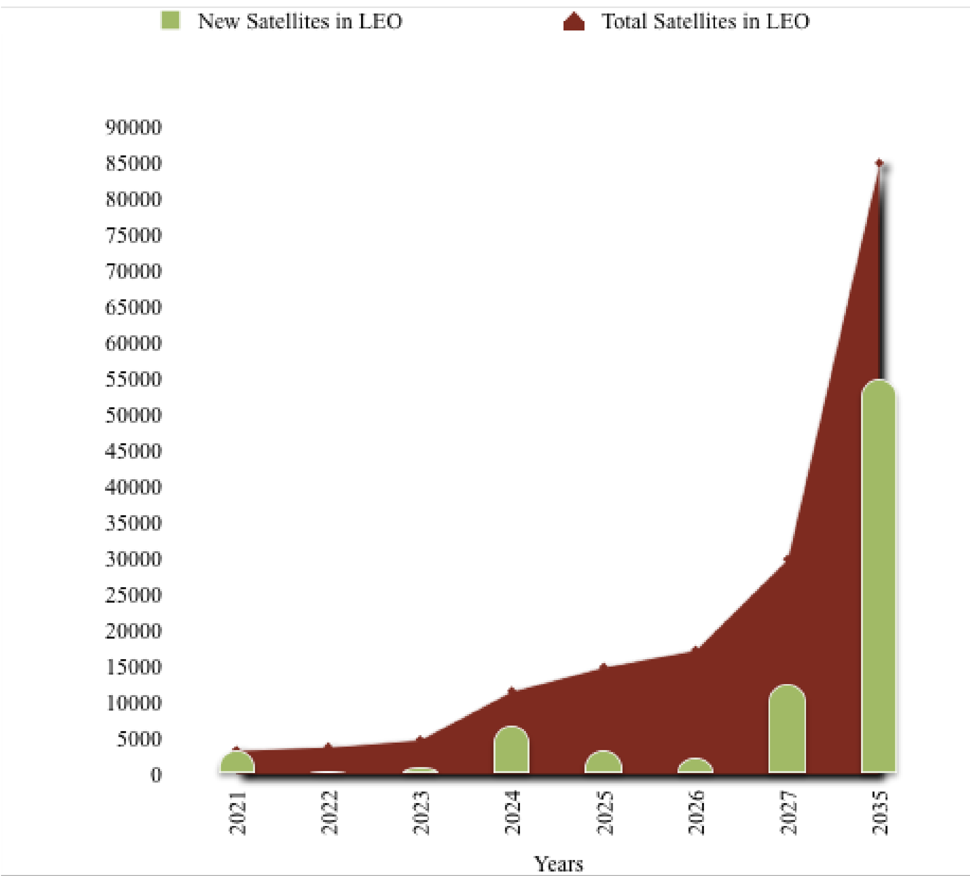


Figure 4.1 Projected future LEO satellite by 2035

4.2. Novelty of E-Orbit

The technological advancement with compact design opens many opportunities to space industries, as many GEO-based communication and navigation satellite constellations are in use. However, the GEO-based satellite only provides a high latency rate compared to low altitude satellite system. Many space agencies and organizations adopt the most straightforward solution of LEO-based mega constellation for communication, EO, and navigation system. Such systems include Starlink, OneWeb, Globalstar, and project Kuiper**.** Currently, 13 communication satellite constellations and six navigation satellites are orbiting around Earth. These essential functions of such constellations are to provide support and economical solution to Earth-based infrastructure. Until now, there is no constellation or satellite service system is design to support space-based infrastructure. The concept of E-Orbit is a solution for space sustainability and providing infrastructure around LEO for power transmission, debris removal, and laser propulsion system. The E-Orbit project proposes the following novelty,

- Provides continuous electricity in LEO around the 900 km altitude.
- The constellation of 1600 E-Sat to generate 16 MW of electricity and electrify the 500 km range from each satellite using the LPT system
- Every E-Sat is highly capable of transmitting 10 kW and receive the desired amount of electricity.
- E-Orbit will constantly track and provide three power transmission links at any time to customer satellites.
- A continues power in orbit even in the eclipse region.
- The production and cost of launching satellites will reduce by 10-15 %
- Increase the lifetime of the satellite.

- Provide power sustainability and wireless PMS system with small and compact design, and the satellite industries can utilize the old PMS area for high payload system without worrying about the power electronics system.
- Solar electric propulsion and thrust management system for orbital maneuver and deorbiting at EOL.
- Continuous support to the interplanetary missions, habitats, rovers, UAV and orbiters using laser emitter.
- The beacon signal in the interplanetary mission will continue to track and work as GPS.
- Using an LPT system, laser thruster can create thrust on a dead satellite for changing orbit.
- The 1600 E-Sat, remove a minimum of 1600 dead satellites from LEO at the EOL.

4.3. The E-Orbit Designing Phases

The E-orbit project consists of phases for developing a demonstration mission to support 3,200 satellites in LEO and optional support for up to 9,000 satellites. The phase is divided into four parts. The first phase is a demonstration mission as a mother-daughter satellite experiment by 2024. The mother satellite is E-Sat, and the daughter satellite is the customer satellite in orbit. The second phase is creating a five E-Sat constellation, E-Orbit in SSO, to handle ten active support and more than 15 optional support by 2027. The third phase is creating a mini E-Orbit with 35 E-Sat to handle 70 active and 100+ optional support by 2029. The fourth is handling LEO to support active 3,200 active support and more than 9,00 optional support by 1,600satellites by 2035. The next phase is creating Moon and Mars E-Orbit by 2045 and 2050 to support human explorations at interplanetary travel. The detailed mapped system timeline is explained in Figure 4.2.

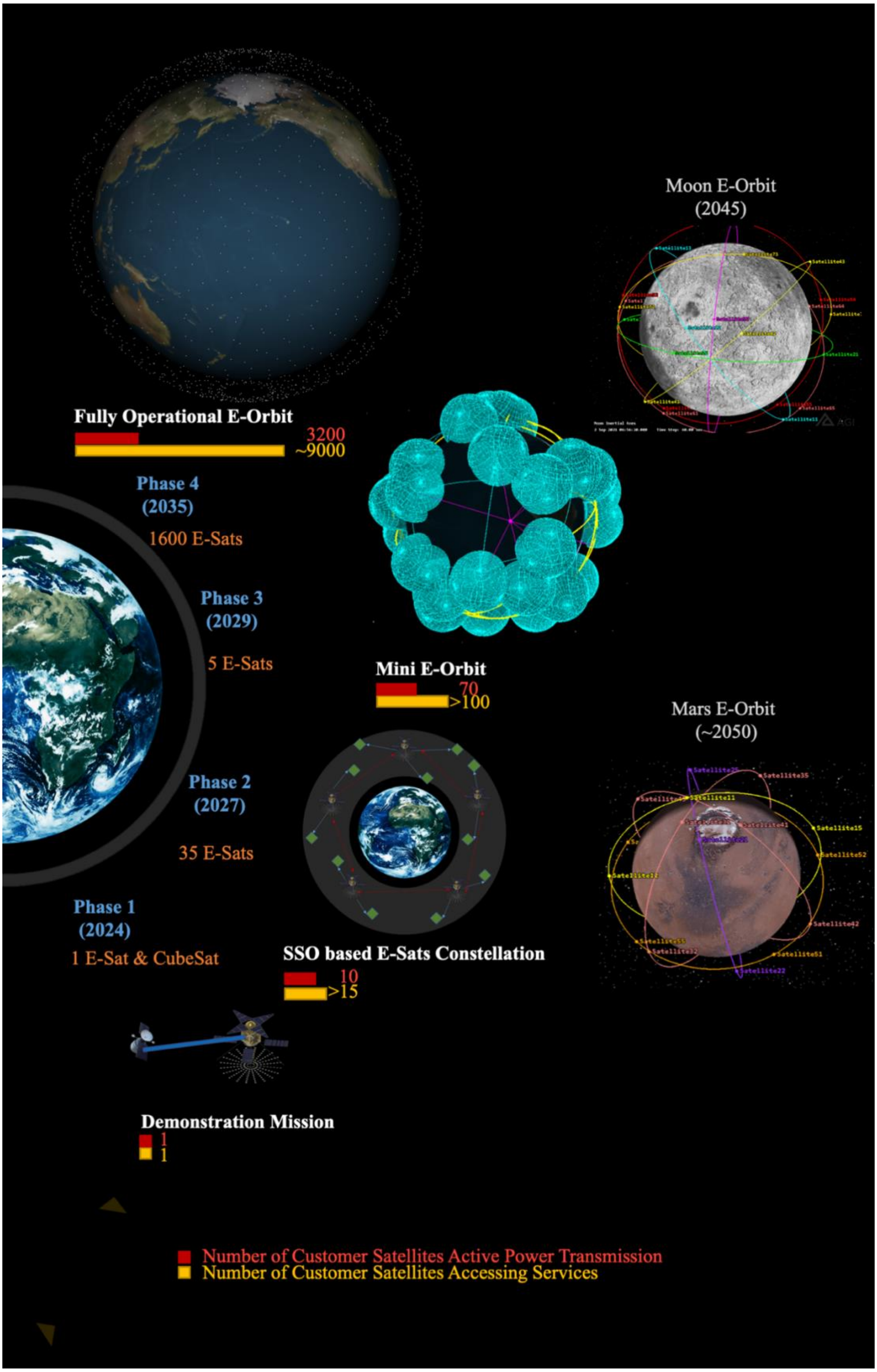


Figure 4.2 A timeline of E-Orbit Mission

4.3.1. Phase 1: Sending Experimental Mother and Daughter Satellites

Mother satellite is the first E-Sat capable of generating and transmitting 10kW of energy to daughter satellite. The payload follows the mother E-Sat that has LPT and PMS systems briefly described in the E-Sat section. The daughter satellite will carry an LPT receiver, electro thruster, PMS, and secondary payload of debris detection mission includes CMOS and LiDAR sensor. Mission will demonstrate and verify the WPT system between satellites and satellites to investigate effectiveness, efficiency, and capability for future E-Orbit. The daughter satellite will separate from the mother satellite and follow an HEO using the LPT system. The mission system working is explained in Figure 4.3. After securing the 900 km SSO location, the daughter satellite will start to distance from the mother E-Sat with a feedback loop by the onboard propulsion system. Then, it starts to send data across long distances via a communication channel located inside the satellite. The satellite will additionally carry a communication unit, attitude measurement sensor, orbital parameter detection sensor, and OBDH. In HEO, it will perform a second payload mission of debris detection using a sensor and return to near 900 km to recharge and generate thrust to follow back HEO. Using this experimental mission following experiments will be performed.

- Proof of concept of SSPS
- LPT for long-distance
- LPT for orbital maneuverings and thrust generation.
- Receiving and conversion of Laser to DC power (Proving WPT)
- Laser Propulsion Utilization
- Following HEO orbit
- LiDAR and CMOS based sensor for detecting HEO debris
- Recharge Using WPT

This experiment can use orbital maneuvering and thrust generation for active debris removal missions and orbital transfer. In the future, the satellite can be quickly transferred using laser thrust from LEO to MEO, GTO, and follow on as shown in Figure 4.4.

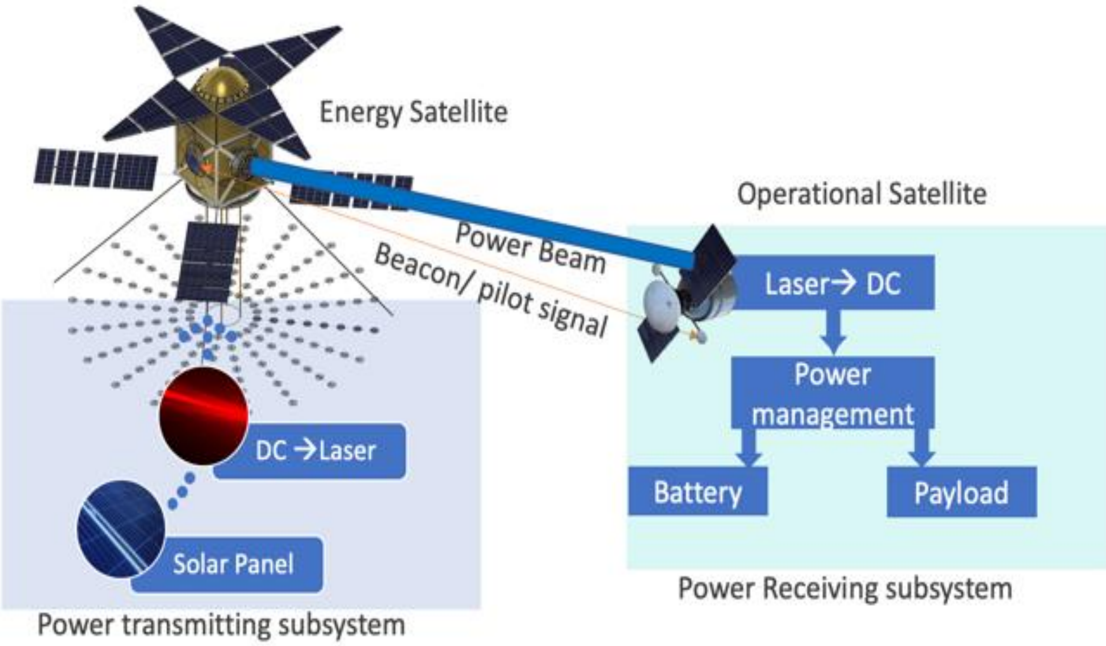


Fig. 4.3 Satellite to satellite laser power transmission mission architecture.

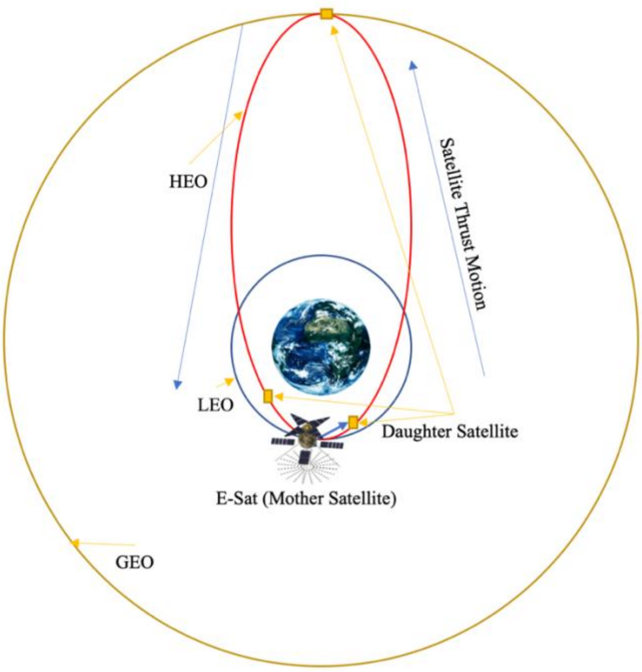


Figure 4.4 Orbital Transfer using phase 1 of E-Orbit

Table 4.1 Satellite-to-Satellite mission Architecture details

Elements	Description	Mother Satellite (SSPS)	Daughter Satellite (CubeSat)
Subject	WPT and onboard experiment	Scientifically investigate and demonstrate WPT from SSPS to achieve a significant level of efficiency of the transmission system and the entire spacecraft. - Pinpoint LPT for long-distance.	-Receive WPT by rectenna. -feedback to the mother sat. - Convert MPT & LPT into electricity to maintain orbit and maneuver in space using the electric thruster. - Extra experiments like debris detection
Payload	Hardware system	PV Cells, Microwave generator, laser setup, avionics, Central processing Unit (CPU), orbit maintenance instrument sensor, and tracer. Communication unit.	Reception and conversion unit, CPU, communication unit, electric thruster, data processing unit. Extra experimental sensor and detection unit
Frequency	Selection of frequency	MPT (5.8 GHz) and LPT (282.823 THz or 1060nm)	
Spacecraft Bus	Customized	Crated to efficient power collection and transfer	
Orbit	Best operational orbit	SSO orbit 900km with an inclination of 98.6 degree	
Data handling	Communication between satellite and ground segment	Direct communication between SSDL and both satellites.	
Responsiveness	Time delay and task management	Real-time data trans-receiving system	
Availability	Level of redundancy	0, available for 24x7	
Ground segment	Mission control center and satellite communication system	SSDL satellite communication system at KU.	
Launcher	Cost-effective to put in the desired orbit	Long March 3B, Polar satellite launch vehicle (PSLV), space X falcon 9, HII	

In the mother-daughter satellite mission, an ideal launch vehicle will be low-cost and capable of placing in SSO. The PSLV, Long March 3B, Falcon 9, or H-II are appropriate and affordable launch vehicle systems. Due to the small size of the demonstrative Mother-Daughter satellite, share riding in this launch vehicle firing system will be used. The aiming and pilot RF signal play an essential role in aligning satellites. The pilot signal follows QPSK modulation to determine the daughter satellite's position from the mother satellite. The attitude, orbital orientation, and orbital data will be sent back to the mother satellite. Then, with the help of a feedback loop, the daughter satellite will adjust the orientation and align with the mother satellite by performing onboard thruster to receive power, as described in Fig. 4.3. Table 4.1 shows the mission architecture requirement and operational details to fulfill the SS power transmission system. The target of SS WPTS to achieve real-time transmission with zero redundancy. The spacecraft bus is customized and can be launch from the above-provided launcher services. Kyushu University operates ground communication and mission operational systems.

4.3.2. Phase 2: Creating a Constellation of Five E-Sat at SSO.

The first E-Orbit will be created in SSO with an altitude of 900 km, at an inclination of 98.6°, and all five satellites are separated 72 ° apart from each other. All E-Sat will generate power 24×7 for a year with a mission life of a minimum of 5 years to support ten satellites within a 500 km radius at any time. These conditions are ideal for satisfying the LEO and SSO based satellites, as the Figure 4.5, 4.6, and 4.7 show the SSO based are satellite are more inclined near 95-100 degree and most of this satellite are generating power for operation of the satellite are bellow 500 W and this are microsatellites.

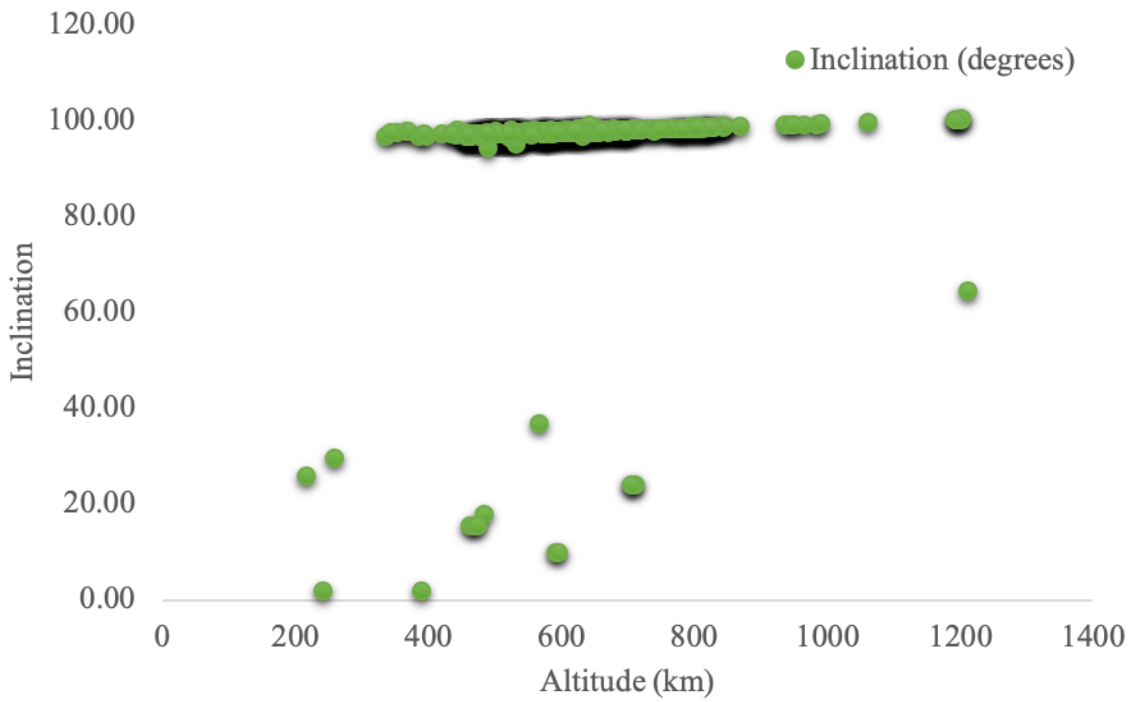


Figure 4.5 Satellite in SSO with the inclination

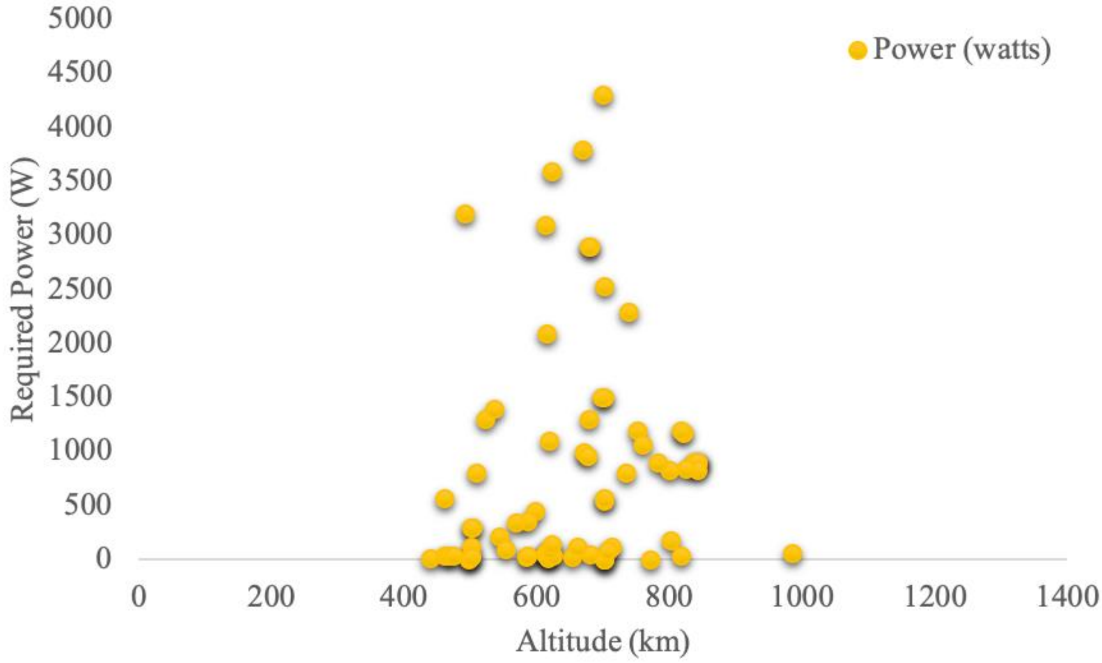


Figure 4.6 The power generated by Each SSO based Satellite

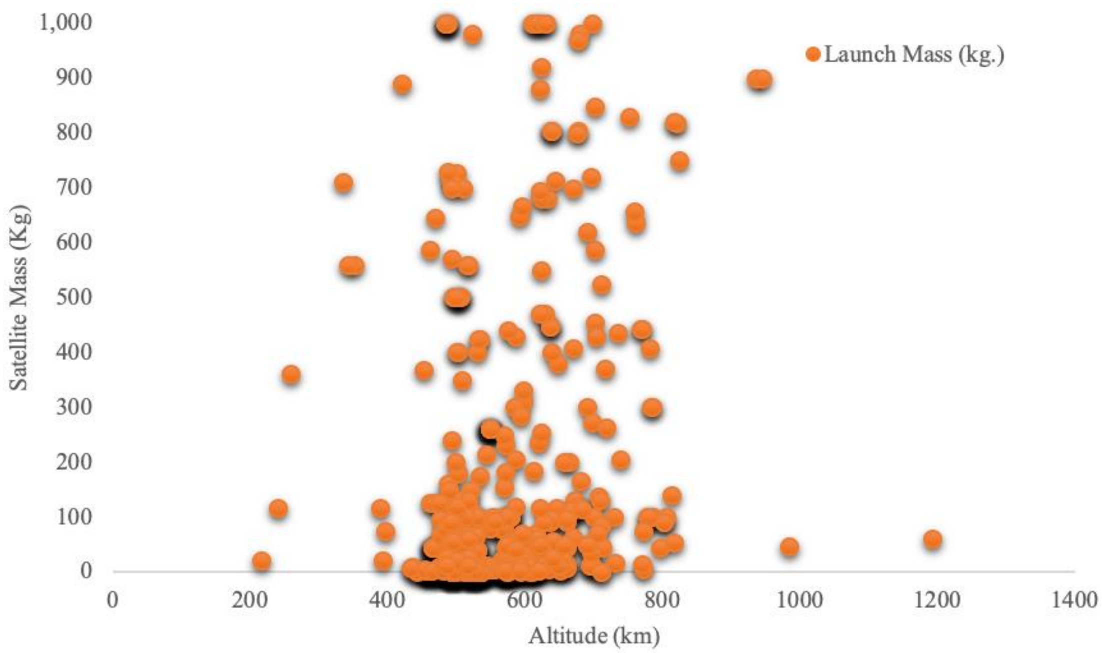


Figure 4.7 The mass of SSO satellite at launching

Sun Synchronous Orbit designing: The SSO-based E-Orbit design can be seen in Figures 4.8 and 4.9. The system is developed with SSO-based orbital parameters and Equations (4.1 – 4.7) to design this. For this, the Z-axis constant towards the x-axis (Earth side) is so ($Z^T X_E = constant$). The sun-Earth direction rotates about Z_E with an angular velocity equal to the Earth's orbit mean motion

$$T_E = 365.25\ days \tag{4.1}$$

$$n_E = \frac{2\pi}{T_E} \approx 1.9991\ \times 10^{-7}\ rad\ s^{-1} \tag{4.2}$$

The nodal effect is due to the Earth's gravitational coefficient J_2 is

$$\Omega = -\frac{3}{2}\frac{R_E^2}{a^2(1-e^2)^2}\ nJ_2 cosi \tag{4.3}$$

$$a = \sqrt[3]{\frac{P^2 GM}{4\pi^2}} \tag{4.4}$$

$$P = \frac{2\pi\sqrt{(R+h)^3}}{\sqrt{GM}} = \ 1\ h\ 43\ min \tag{4.5}$$

$$n = \sqrt{\frac{\mu}{a^3}} \tag{4.6}$$

$$\dot{\Omega} = n_E \tag{4.7}$$

Where,

- The altitude of satellites (h = 900 Km) is 7278.1363 km.
- a is the orbit semi-major axis 7278.1363 km.
- G is the gravitational constant = 9.80665 m/s^2.
- M is the combined mass of primary and secondary bodies.

The mass of the secondary body may be ignored when it is insignificant compared to the mass of the primary, as is the case for a spacecraft orbiting a planet.

- J_2 = 1.0826269×10−3 is a coefficient of an expansion representing the Earth's oblateness effect,
- The orbit means motion is n.
- μ = 398,600.4415 km^3 /s^2 is the Earth gravitational parameter.

Therefore, the Sun-synchronicity approximated condition is achieved by Equation (4.8)

So

$$cosi = -4.774.10^{-15} a^{\frac{7}{2}} (1-e^2)^2 \tag{4.8}$$

Alternatively, as if we are taking as circular orbit, the Equation (4.9) is when an orbit will be Sun-synchronous when pression rate ρ equal the mean motion of Earth about the Sun. which is approximately 360 degrees. Or $1.99096871 \times 10^{-7} rad\ s^{-1}$

$$c\frac{\Omega}{P} = \rho \tag{4.9}$$

$$cosi \approx -\frac{2\rho}{3J_2 R_E^2 \sqrt{\mu}} a^{\frac{7}{2}} \approx -\left(\frac{\mathrm{a}}{7278.1363\mathrm{km}}\right)^{\frac{7}{2}} \approx \left(\frac{\mathrm{P}}{1.43\mathrm{h}}\right)^{\frac{7}{2}} \tag{4.10}$$

Another critical parameter for SSO is the angle Δλ between two successive ascending nodes. This angle, after each orbit nodal period, is represented by Δλ = (Ω˙ − α˙E) T_Ω, where α˙ E = 7.2921 · 10−5 rad/s is the Earth's spin rate and the rates of change for the initial mean anomaly and the perigee argument due to J2 perturbation can be calculated using Equation (4.11 – 4.13)

$$T_\Omega = \frac{2\pi}{n + M_0 + \omega} \tag{4.11}$$

$$M_0 = \frac{3R_E^2 J_2}{4p^2} n\sqrt{1-e^2}\,(3\cos^2 i - 1) \tag{4.12}$$

$$\omega = \frac{3R_E^2 J_2}{4p^2} n\,(5\cos^2 i - 1) \tag{4.13}$$

Designing for five satellites in SSO orbit ($N_s = 5$), and all satellites are in the same orbit having same inclination angle, the argument of perigee, eccentricity, and semi-major axis. Moreover, all five satellites are separate by evenly equally angle can be calculated using Equation (4.14).

$$\boldsymbol{\Delta\Omega = \frac{2\pi}{N_s} = \frac{2\pi}{5} = 72°} \tag{4.14}$$

The five satellites separate by 72-degree angle from each other and multi sun-synchronous constellation is define by that this angle is precisely m times the angle $\Delta\Omega$ that is, is ($J_n\ \Delta\Omega$) and since the relative velocity between the node and Sun directions is ($n_E - \Omega$) Equation (4.15) and MSSC rational parameter by Equation (4.16)

$$J_d T_\Omega (n_E - \Omega) = J_n\,\Delta\Omega = \frac{J_n 2\pi}{N_s} \tag{4.15}$$

$$\eta = \frac{J_d}{J_n} \tag{4.16}$$

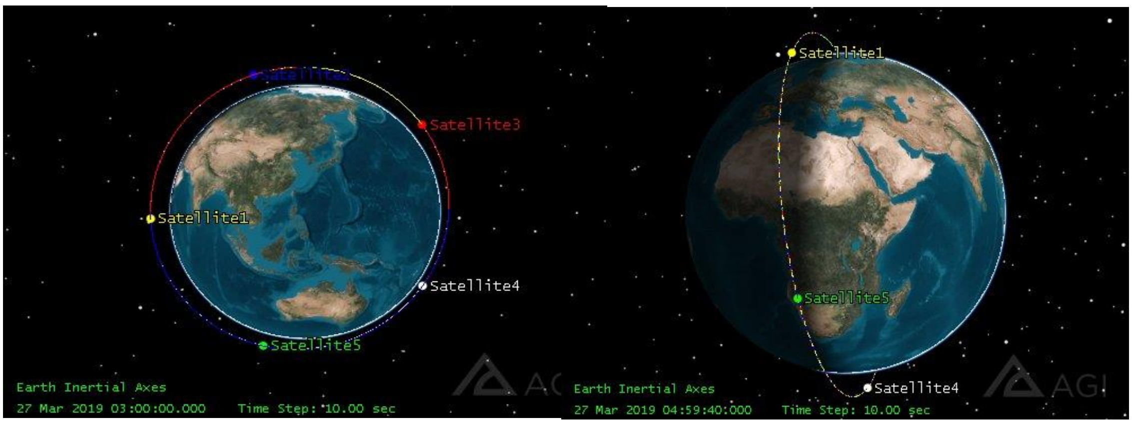


Figure 4.8 Animated design of SSO based E-Orbit (a) view from sun sight (b) side view

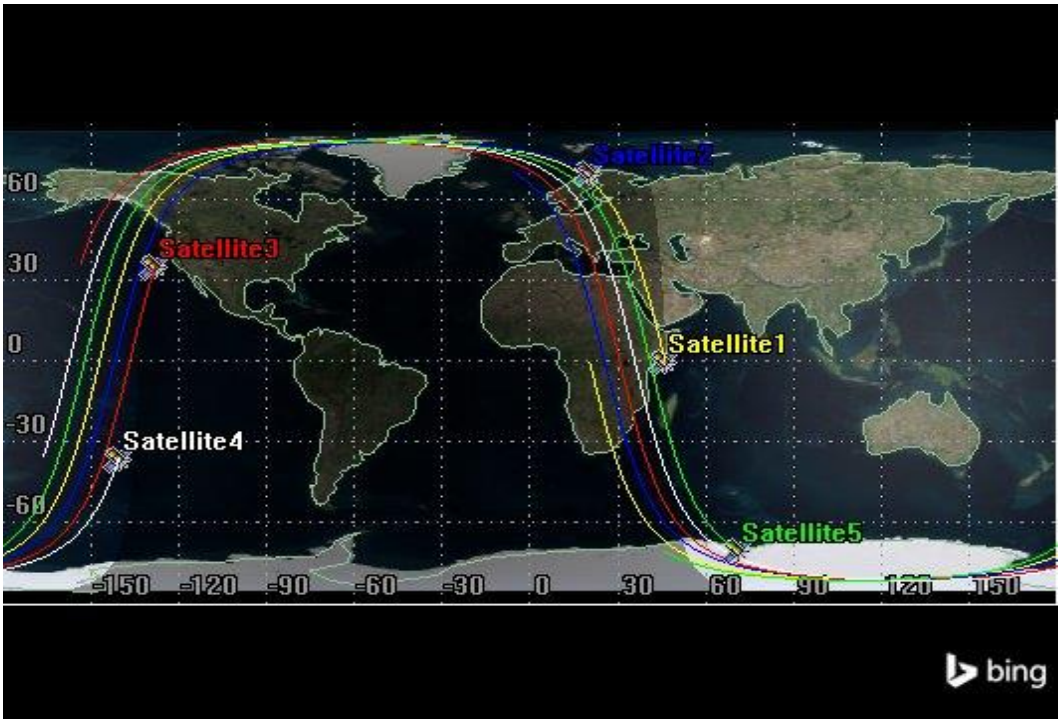


Fig. 4.9 2D Earth tracking of the satellite

4.3.3. Phase 3: Creating a Semi-E-Orbit Constellation

Currently, there are more than 3,328 satellites are in LEO. The majority of these satellites are around 400- 900 km altitude. As shown in Figure 4.10, most of this satellite is inclined with nearby 50 degrees and 95-100 degrees to follow the SSO. A majority of the future constellation

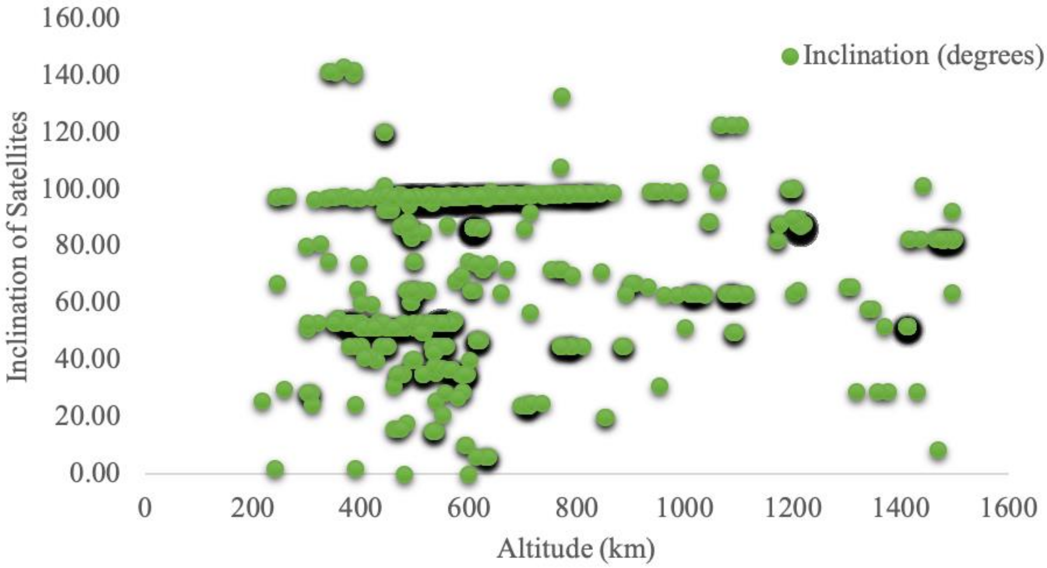


Figure 4.10 Satellite in LEO with an inclination

will be around this section, the power generated from this satellite can be seen in Figure 4.11. Most satellites in LEO utilizes a low power payload system due to the altitude and period of rotation around Earth and the eclipse region. Most of this satellite is a microsatellite, as shown in Figure 4.12.

After establishing SSO E-Orbit, a mini constellation of E-orbit will be created to satisfy the demand of power around this orbit with another six orbital planes created with 35 E-Sat in all seven different planes around 900 km altitude with an inclination of 98.6, 0, 45, 135, and eccentricity of 0, the circular orbit. The RAAN as 300, 0, 25, 45, 156, 260, and E-Sat has separated by 72 degrees apart from each other, as shown in Figure 4.13. Figure 4.13 (a) shows that the 500 km radius using cyan color and pink line links with customer satellite while orbiting in 900 km E-Orbit.

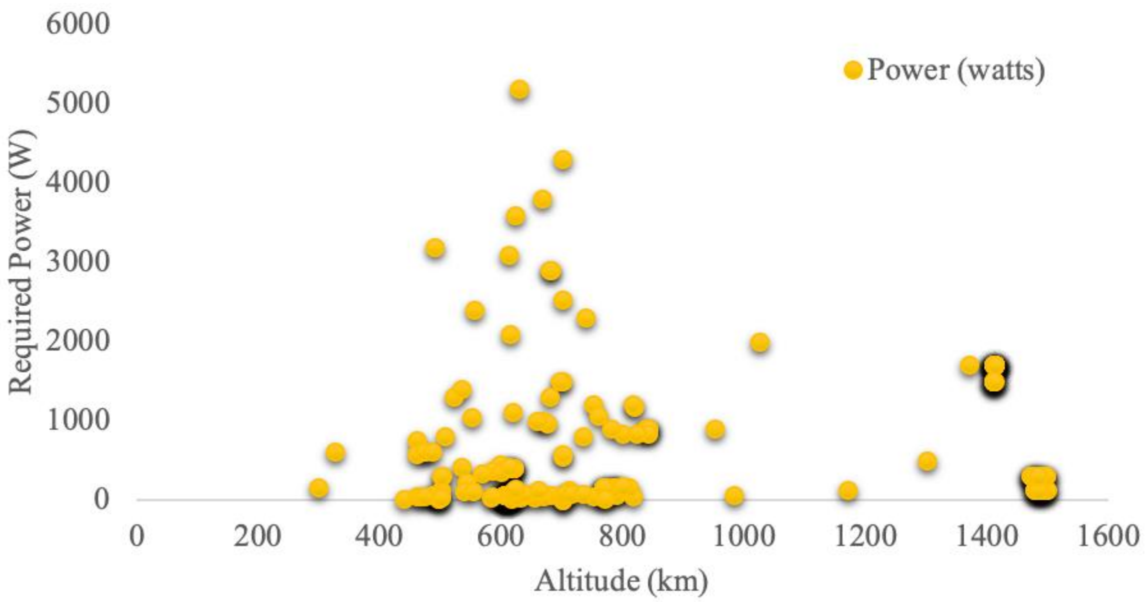


Figure 4.11 The power generated by Each LEO based Satellite

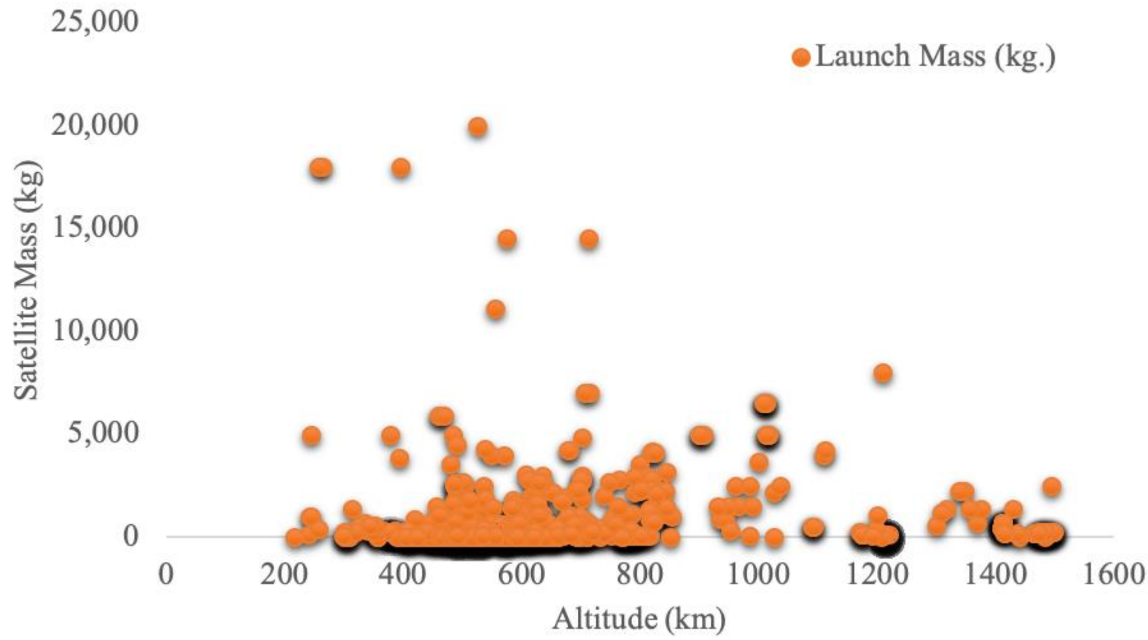


Figure 4.12 The mass of LEO satellite at launching

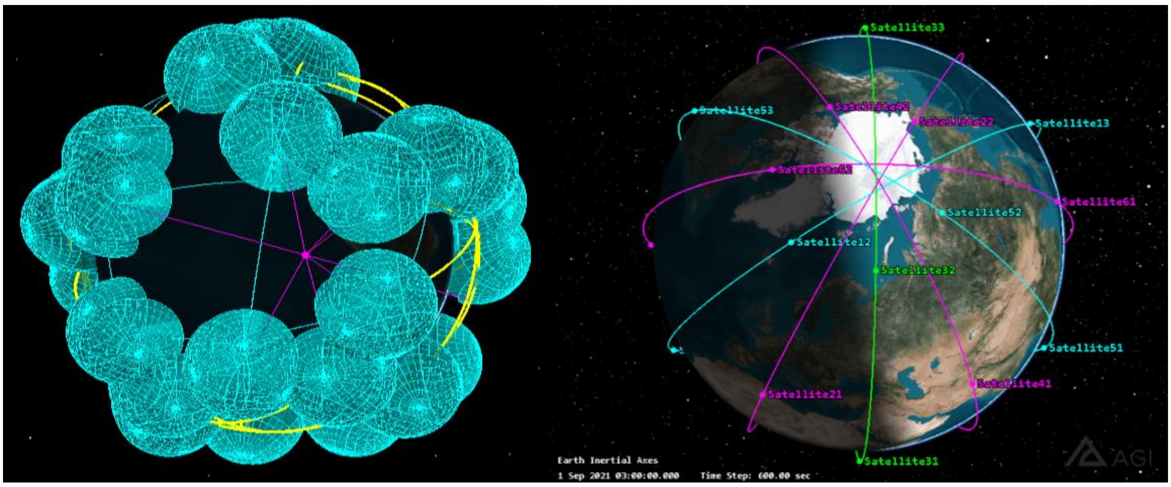


Figure 4.13 Animated semi E-Orbit constellation (a) E-Sat with tracking customer satellite (b) polar view of Earth with E-Orbit

The AGI STK-based software is used for orbital designing, parameter calculation, and system analysis. In contrast, the system is defined as an Astrolib based code designed to check the strength of formation flying in the 35 E-Sats, the mini-E-Orbit constellation for one year as shown in Figure 4.14 dummy satellite to interact and most considerable distance to achieve power transmission. With the 35 E-Sats, the satellite has to powerfully transmit the power for up to 9,000 km to satisfy all the LEO active Power transmission.

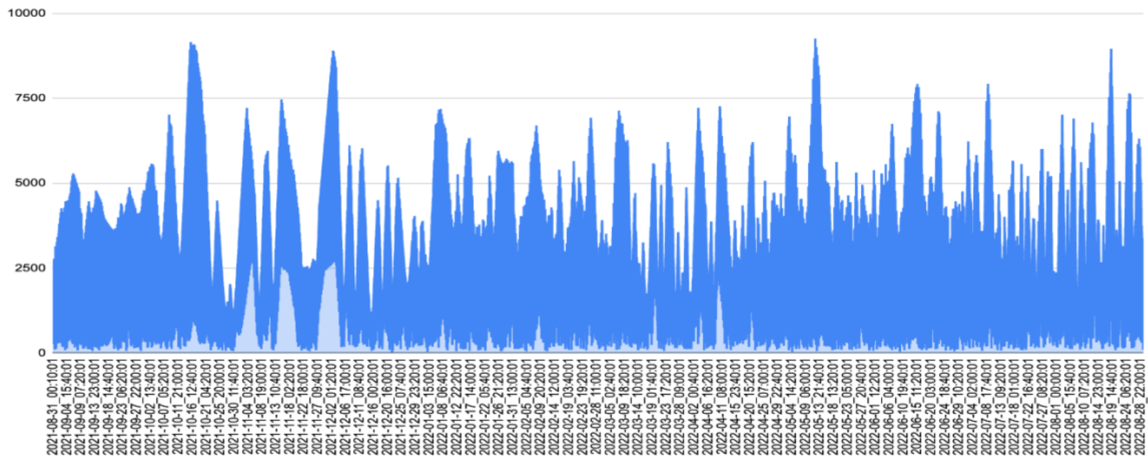


Figure 4.14 Distance of Dummy Satellite During 1 Year Time to E-orbit (35 Satellite)

The formation flying of mini E-Orbit shows that the continued interaction with a Dummy Satellite during the 1-year time to E-Orbit is minimal while in LEO, as shown in Figure 4.15. On the other hand, the availability of E-Sats is more likely available during the one-year simulation, as shown in Figure 4.16.

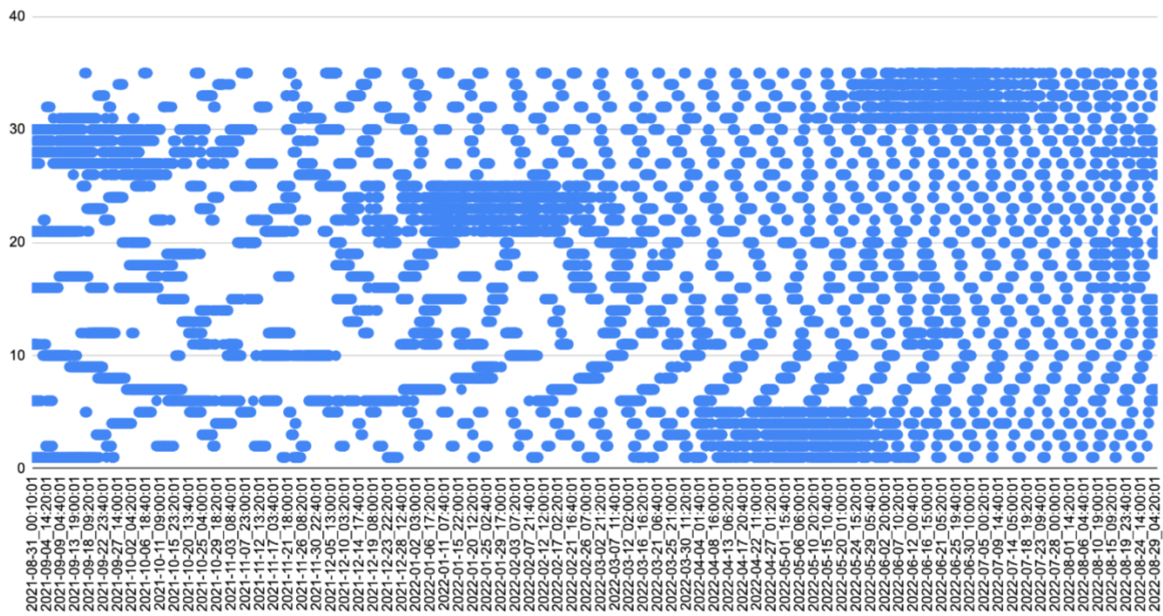


Figure 4.15 Interaction to a Dummy Satellite during the 1-year time to Mini E-Orbit

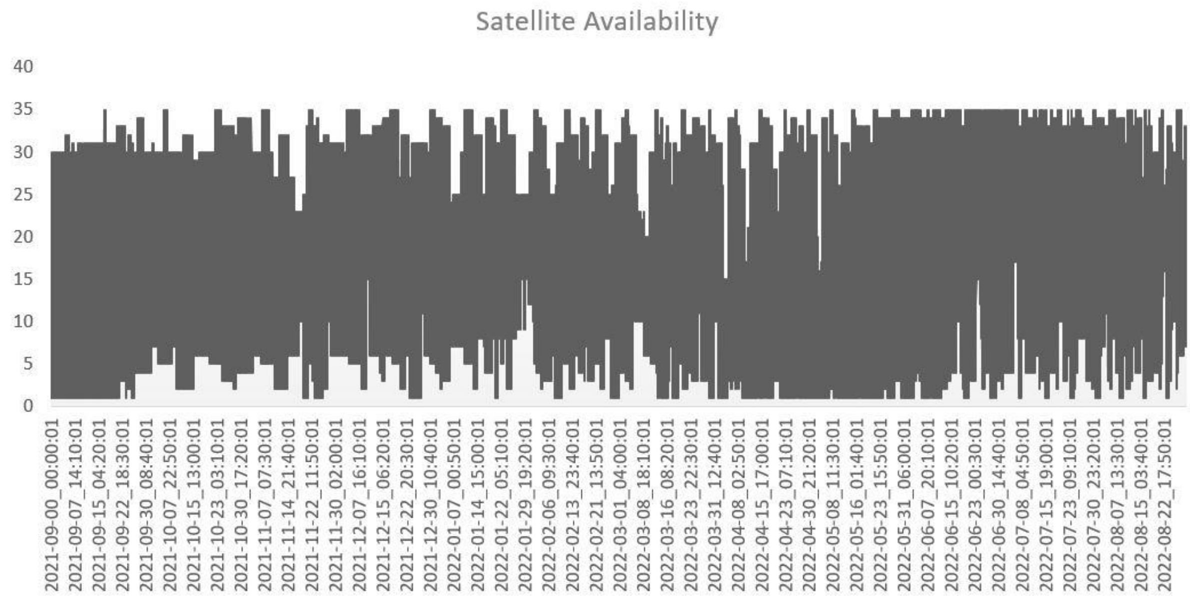


Figure 4.16 Availability of E-Orbit to A dummy Satellite during the 1-year time

4.3.4. Phase 4: Creating an Entire E-Orbit Constellation

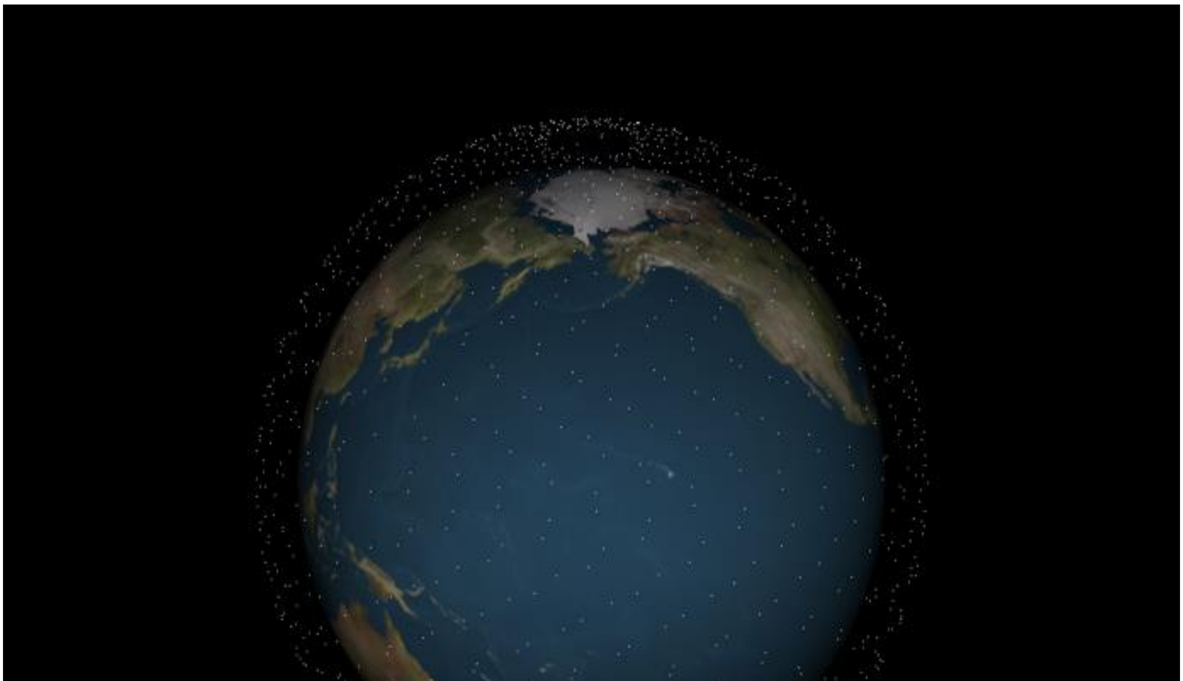


Figure 4.17 Animated entire constellation of E-Orbit with 1,600 E-Sat.

Phase 3 can only satisfy 70 active connections and 100 support missions. However, with an increasing number of constellations, the new format of the entire orbital system is required with a minimum of 1,238 E-Sat to orbit. Nevertheless, developing a constellation of 1,600 E-Sat across the Earth will provide a constant and continuous power supply of 10 kW in the range of 500 km using LPT-based E-Sat, as shown in Figure 4.17. The customer satellite will just have to carry a rectenna without worrying about limitations in power supplies. A single customer satellite will be attached constantly with three conjugative satellites for better transmission. If necessary, satellite connections can be increased for larger power supply. Figure 4.18 shows the utilization of E-Orbit in Orbit for one day, and its self-interaction with a dummy customer satellite rotating nearby E-Orbit provide the data interaction with the E-Orbit. The distance covered to allow the satellite to interact from the 900 km altitude to the customer satellite can be seen in Figure 4.19.

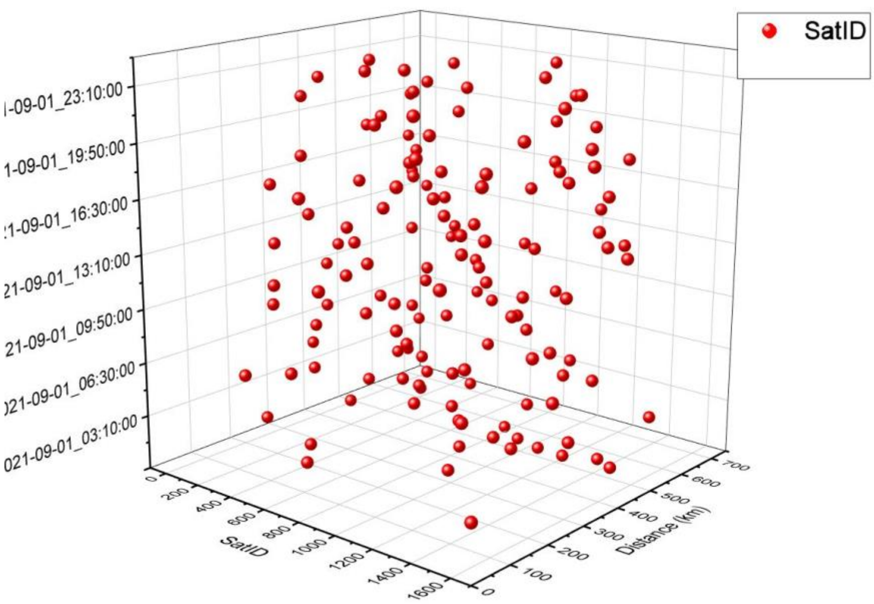


Figure 4.18 E-Orbit interaction to dummy customer satellite for a one-day

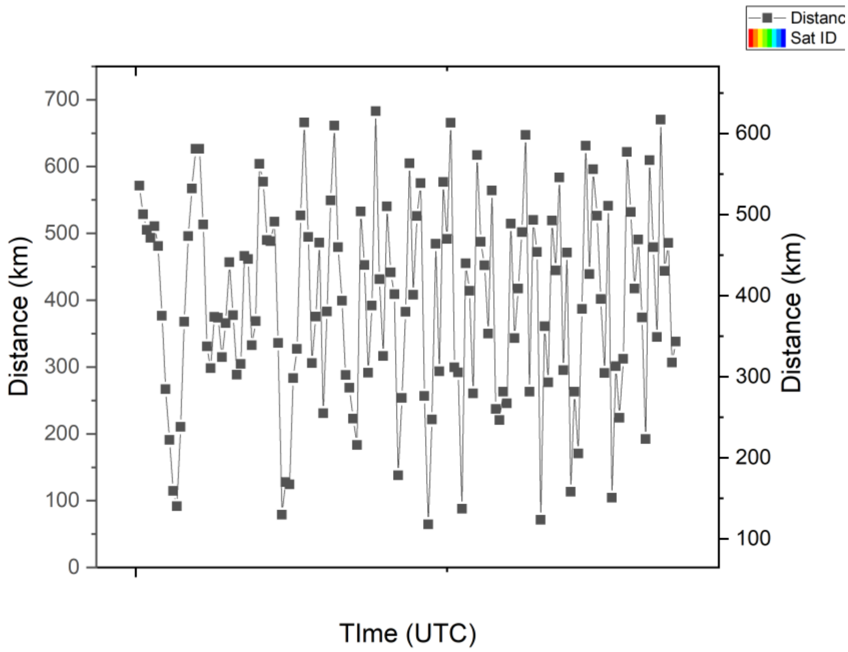


Figure 4.19 The distance between customer and E-Orbit Satellite range for one day

While all E-Orbit satellites orbit for one month will cover and interact with the satellite for power transmission and interlinking for equalizing power, as shown in Figures 4.20 and 4.21.

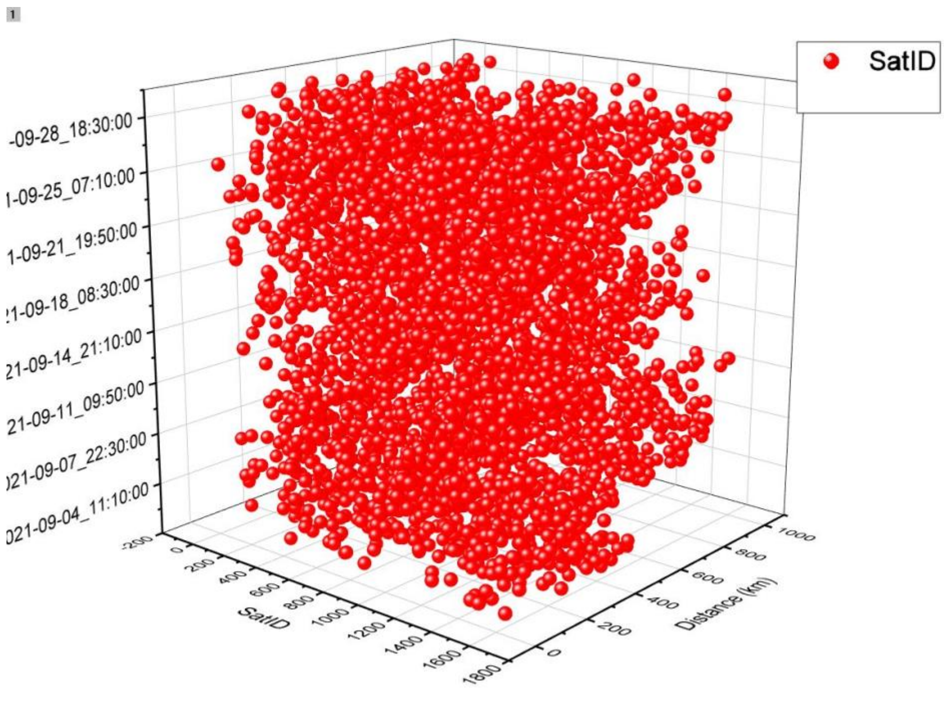


Figure 4.20 E-Orbit interaction to dummy customer satellite for a one-month

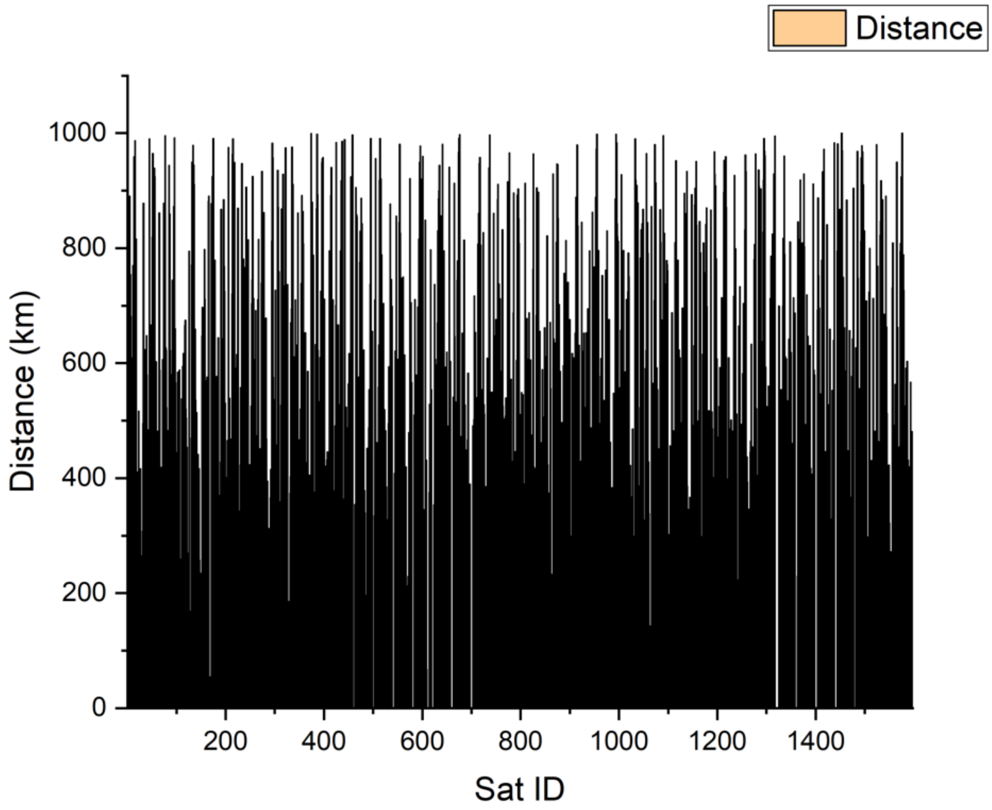


Figure 4.21 The distance between customer and E-Orbit Satellite range for one month

The new startups and researchers will not have to worry and trade-off the technology and payload for the power generating unit. They can rent electricity from E-Orbit. It will also lower the project's cost. Furthermore, renting electricity will minimize losses occurred from power management systems even if the mission fails. Typically, a cube satellite 1U power system will cost from 3,000 USD to 8,000 USD. Using E-orbit will also provide more space opportunities for research and space sustainability. The distance of the entire operation satellite is less than the 700 km range to cover for LPT can be seen in Figure 4.22. Moreover, the satellite's view angle when interacting with any customer satellite can be seen in Figure 4.23.

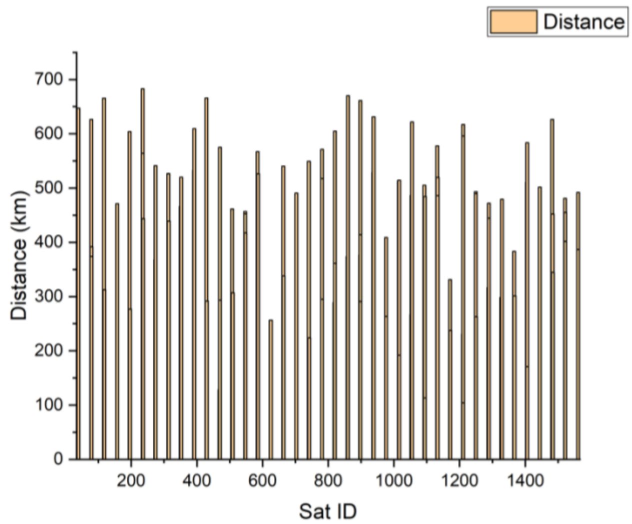


Figure 4.22 The distance between customer and E-Orbit Satellite range for one year

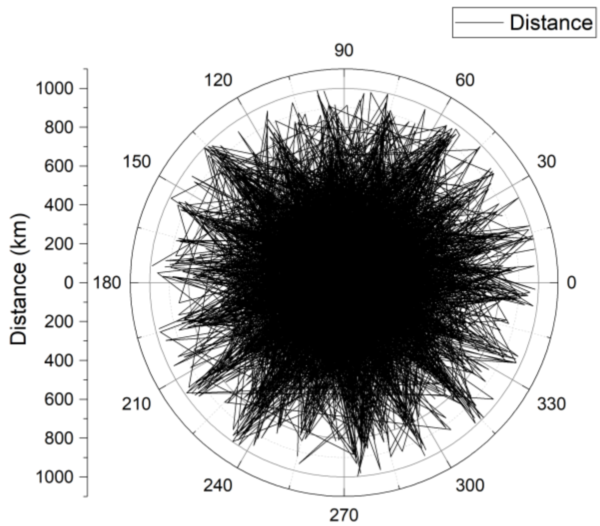


Figure 4.23 The satellite interaction with customer satellite for a year by E-Sat

For sustainable and better space resource utilization, the E-Orbit follows the SDGs as E-orbit offers affordable and clean energy transfer, manufacturing traditional fuel propulsion systems that impact the environment will be stopped (SDG 7). The new market of reflector manufacturing will be helpful for economic growth without affecting the PV-cell manufacturing industry (SDG 8). The innovation of reflector, dual-sided solar panel, Laser, and rectenna will be helpful to build new infrastructure in the space industry (SDG9). This will bring sustainable consumption and production pattern (SDG 12). The global partnership will be increased due to technology to fight against cost factors (SDG 17).

Table 4.2 E-Orbit basic functionality

Sr. No.	Requirement	The factor which Typically Impacts the Requirement	S2S SSPS System
		FUNCTIONAL	
1.	**Performance**	The primary objective, payload size, orbit, pointing	Transmission of power (min 10 kW) The solar array of 54 m^2 and 112 m^2, SSO, multi- 900 km
2.	**Coverage**	Orbit, number of satellites	SSO 900km Min 5 E-Sat, Max 1,600 E-Sat
3.	**Responsiveness**	Communication architecture, processing delays, operation	Real-time data transceiving system
4.	**Secondary Mission**	For mother-daughter experimental satellite	Debris
		OPERATIONAL	
1.	**Duration**	Experiment or operations, level of redundancy	20 years
2.	**Availability**	Level of redundancy	0, available for 24×7
3.	**Survivability**	Orbit, hardening, electronics	Natural environment only
4.	**Data Distribution**	Communication architecture	Real-time communication between satellite to satellite and ground stations
5.	**Data Content, Form, and Format**	User needs, level, and place of processing, payload	Across energy orbit up to 10 kW of power
		CONSTRAINTS	
1	**Cost**	Crewed flight, the number of SC, size, and complexity of orbit	14 billion USD
2.	**Schedule**	Technical readiness, program size	Three years of final operating capability in 4 years, 2024
3.	**Regulation**	Law and policy	LPT regulation
4.	**Political**	Sponsor, government	On public demand
5.	**Environmental**	Orbit, Lifetime	Natural
6.	**Interfaces**	Level of the user and Operator infrastructures	To the sponsor and distributed between satellite users who will use E-Orbit
7.	**Development Constraints**	Sponsoring organization	They are launching by the international collaboration of space agencies to creating a unified E-Orbit.

The E-Orbit will also be built for Moon and Mars missions with 25 and 30 satellites for the continuous orbiter and surface support. The WPT for interplanetary missions provides mobility of habitat and constant power use for multifunctionality regardless of time and availability of sunlight. Three conjugative satellites can provide continuous power and, at the same time, work as a GPS for tracking and movement of the system on the Moon and Mars.

4.4. Orbital Maneuvering

The transportation technology is developing rapidly, currently with RLV technology the price of launching is dropped up to 5,000 USD for LEO and 30,000 USD for GEO. Space researchers and engineers are looking to drop the price even more for easy launching down to approximately 100-500 USD which will be an appropriate price for launching any prominent structure SSPS in GEO. In comparison, the E-Orbit can solve most problems by continuing Laser arbitration thrust to any given object within the range of orbital parameters. It has different application as follow:

- Orbit Transfer
- Orbit Keeping
- De-Orbiting at EOL

Orbit Transfer*:* As the above explains, launching any satellite depends on launching vehicle, place, and size of the structure. Current technology of the launching system can place any satellite in LEO, MEO, GEO, GTO, SSO, HEO and others. However, most launching vehicles are designed for the LEO-based satellite delivery system. While most of the rocket-based start-ups can transfer the spacecraft at nearby 600 km, only a few launching services are capable of specific inclination and higher orbit. The E-Orbit can use its Laser propulsion system by creating thrusts on the spacecraft from a different angle to achieve an elliptical orbit to transfer the satellite in HEO and send it directly to a nearby GEO place. This method will cut down most tasks for transferring and utilizing launching vehicles for the MEO and GEO. A single object or customer satellite will get reception from all 1600 E-Sat to maintain the HEO path. The exact mechanism can be used for lowering the customer satellite which is discussed below.

Orbit Keeping: On most occasions, a satellite orbiting in a particular orbit gets perturbation due to different orbital parameters resulting in the satellite losing its orbital parameters. To maintains the exact orbital parameters, an AOCS, and propulsion system are required. The propulsion system is expensive and increases the size of any satellite. That is why a small Nanosatellite (CubeSat) is more likely to fly without the propulsion system. Using the E-Orbit system, the satellite in orbit can utilize the thrust generated by Laser propulsion from E-Sat for orbiting within the range and performing the payload application freely. Any customer satellite can utilize the Laser propulsion from E-Orbit at any single time. At the close approach and close conjunction probability, the E-Orbit can use a thrust mechanism to change the orbit of the customer satellite on demand. It will help to serve as a tool for sustainable space and preventing future debris creation, which is explained in the debris removal section. Using this, the lifetime of any satellite and the operating system will increase the economic value of customer satellites.

De-Orbit at EOL: Every satellite has a particular life cycle of operation in orbit. After that, it will stay in orbit and become space debris or de-orbit by itself. Currently, all nations agree to de-orbit the LEO-based satellite by its parent/operation organization before dying by utilizing the onboard fuel system. E-orbit provides a thrust system at EOL of satellite to de-orbit from the particular location and provides significantly economical value to the company by utilizing the payload for years and generating millions of USD. A continuous thrust for orbital maneuvering will be used for de-orbiting to lower the satellites' orbit from a specific location within the E-Orbit range. At the EOL, the satellite will utilize the debris removal procedure from E-Orbit without worrying about the propulsion system. Details of the debris removal section will be briefly explained in the next section.

4.5. Debris Removal

The launch of the first spacecraft Sputnik 1 (Спутник-1) in 1957 signals the beginning of the space age marked by tonnes of rockets, spacecraft, and instruments that have been launched into space since. Initially, there was no plan for the end matter of such devices and instruments. Numbers continue to increase with explosions and collisions in space, creating hundreds of thousands of dangerous space debris shards. By now, more than 250 known explosions have already created thousands of space debris which is estimated to create more than 500 break-ups, explosions, collisions, or anomalous events resulting in fragmentation of more than 128 million space debris. Most of the debris is smaller than 1 cm, and 900,000 pieces are the size of 1-10 cm, and 34,000 are in the size of more than 10 cm and these numbers are increasing day by day. Twenty-three thousand artificial objects are revolving nearby Earth including 4,084 satellites currently operating in space and LEO 3,328 satellites as depicted in Figure 4.24 [84]. Most of the debris is in LEO but will eventually recede to the Earth's atmosphere due to drag and different perturbation types. Meanwhile, there is also debris in GEO, which is drastically unsafe for new satellites in orbit [102].

The most prominent case of space satellite collision accident occurred in 2009. Two communication satellites, Iridium 33 and Kosmos 2251, weighing 560 kg and 950 k,g respectively, accidentally collided at the speed of 11.7 km/s at an altitude of 789 km, as in Figure 4.25. This collision produced thousands of big, small, and sub-millimeter-sized debris, still orbiting Earth [103]. In 2015 a US military weather satellite DMSP-F13, the weight of 750 kg at SSO 850 km attitude, completed its 20 years of service life and accidentally exploded, creating a cluster of space debris due to the energy storage unit explosion. Approximately 147 debris of sizes approximately equal to a baseball orbit the. Approximately ten explosions are due to batteries or an energy management system. Prof Toshiya Hanada investigates the study of debris created by energy storage. There are very old satellites in GEO that become debris or inactive within a few years and become dangerous for the space environment.

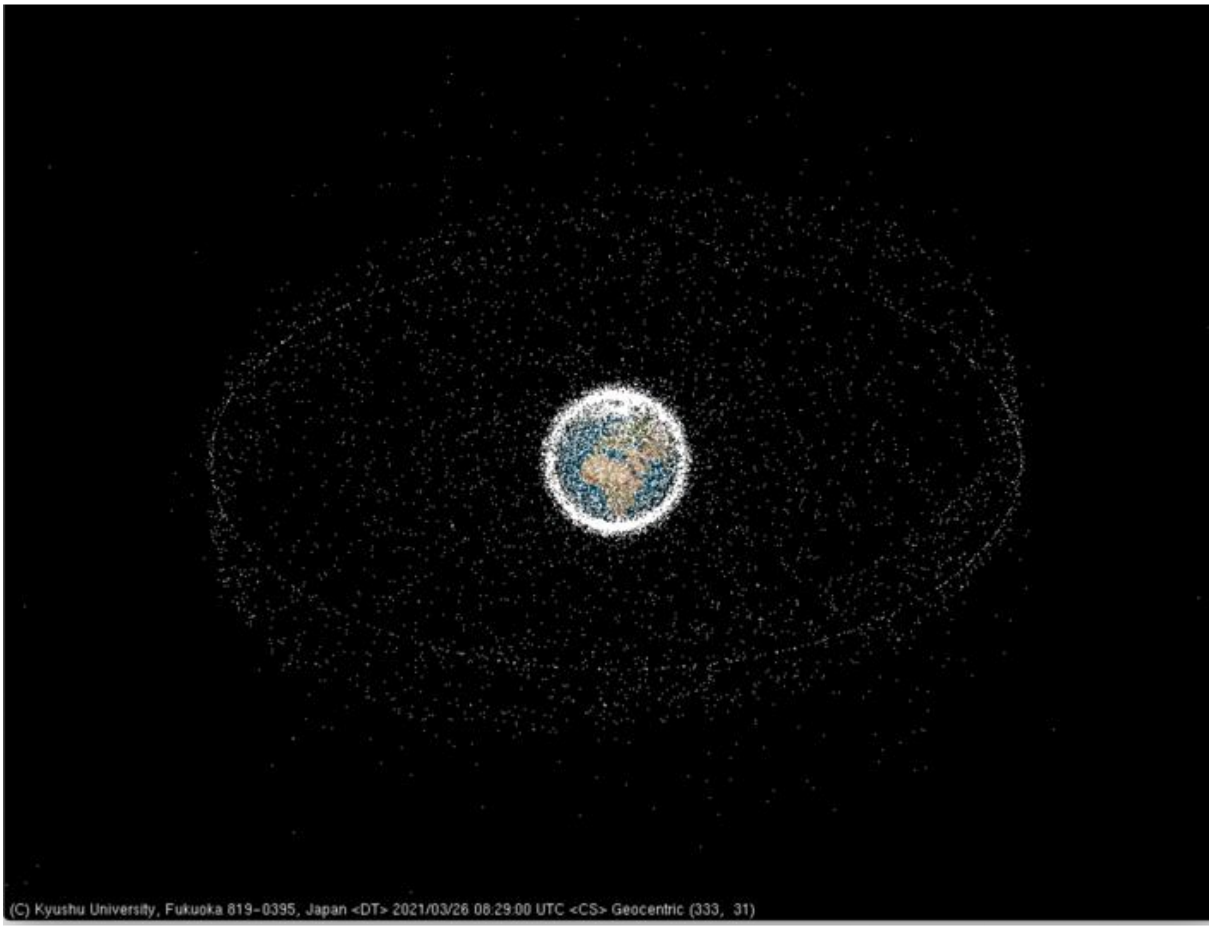


Figure 4.24 Space debris across near-Earth environment

Currently, many government and private companies are coming up with ideas for debris removal systems. For example, Japan-based company Astroscale uses a chaser in LEO-based satellite and docking using a magnetic capturing method by End-of-Life Service by Astroscale demonstration or ELSA-d project launched in 2020. The ClearSpace is used for dead satellites to capture and remove the debris from outer space. European Space Agency uses the net system to capture satellites and remove the big dead satellite from space. However, similar to the E-

Orbit thesis, the J-Sat and SSDL lab is working on a 100 W laser-based debris removal project collaborating with RIKEN, Nagoya University, and J-SPARC to develop a non-contacting method to remove debris from a distance of 100 m. The same idea is can be adopted used by E-Sats and E-Orbit. A 10-kW laser can be used for power transmission with a propulsion system, using two methods:

- LPT for propulsion payload system
- Laser-based thrust system

LPT for propulsion payload system: the transmitted power can be stored in the PMS system and utilized for payload and orbital maneuver using an electro thruster system. There are many electro thrusters, for example, Ion, plasma, non-ion, and electrospray thrusters. This thruster has a compact size with a high advantage over small spacecraft for orbit keeping. The received power can be utilized for the operation of this thruster.

Laser-based thrust system: a 10-kW laser is pointing to a specific location on an objective satellite can be utilized for orbital maneuvering. A customer satellite within the range of E-Orbit can use power and maneuver using the installed thruster or laser arbitrary thrust generated from E-Sat. In the event of a close approach or probability of conjunction, the E-Orbit, on-demand, will thrust multiple 10 kW, one with a particular thrust that focuses on lanes on the object satellite to avoid a collision. At the end of E-Sat's EOL, it will deorbit itself while utilizing its laser thrust mechanism on a dead satellite for share riding towards the Earth's atmosphere. When E-Sat is deorbiting, it will follow the targeted satellite in its orbit and closely follow within 100 m distance. With a non-contact laser thrust mechanism, the 10-kW Laser can generate 0.1N thrust within the 100-meter range and deorbit any satellite with a mass of 150 kg from 1,200 km to the Earth's atmosphere from 8-14 days, as seen in Figure 4.26. Circulating the target satellite continuously and providing a continued propulsion system will degrade any satellite within two weeks, with high safety and avoiding any collision, crating any fragments and close approach.

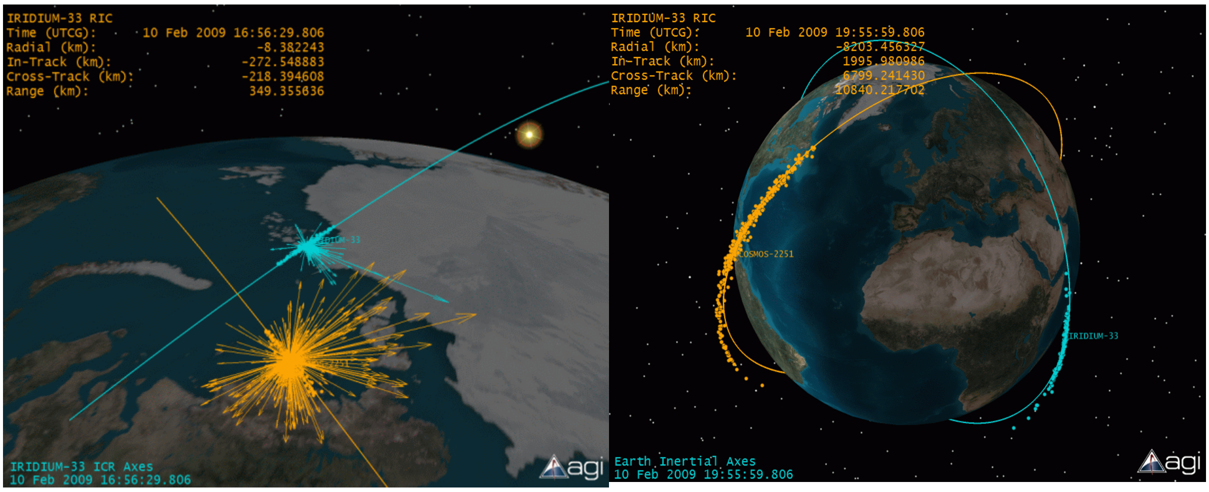


Figure 4.25 (a) View of Iridium 33 and Cosmos 2251 orbit and debris 10 min post collision, and (b) 180 min post collision

Two components are needed for such an ideal debris removal system. First is the target debris/ object and the propagator model. Currently, at SSDL, the LEO, GEO, and NEO are already investigated and crated. However, an HEO-based debris model is also required. The above section of orbital maneuvering explains how to use an LEO-based launching system to transfer a satellite in GTO/ GEO via the HEO propagation method that required an HEO propagator from E-Orbit. As the example of the HEO system, the probability of debris entering HEO is relatively high. If a collision occurs in GEO and LEO, this debris with different perturbation and velocity can enter elliptical order and become stay as debris in HEO. If a breakup occurs in an HEO, the error significantly affects the fragment's semi-major axis and eccentricity; therefore, improvement of close approach analysis or development of a new estimation method is an urgent task [104]. The investigation of HEO is significant as the HEO is also used for LEO, geostationary transfer orbit, and moon transfer orbit. Research on the HEO-based model is essential for safe, secure, and sustainable space research and investigation. Using experimental-based simulation can be created to scale down the fragments of artificial satellite components and to analyze the physical characteristics of general debris, including the six-axis moment, motion and velocity, and fragmented object orientation.

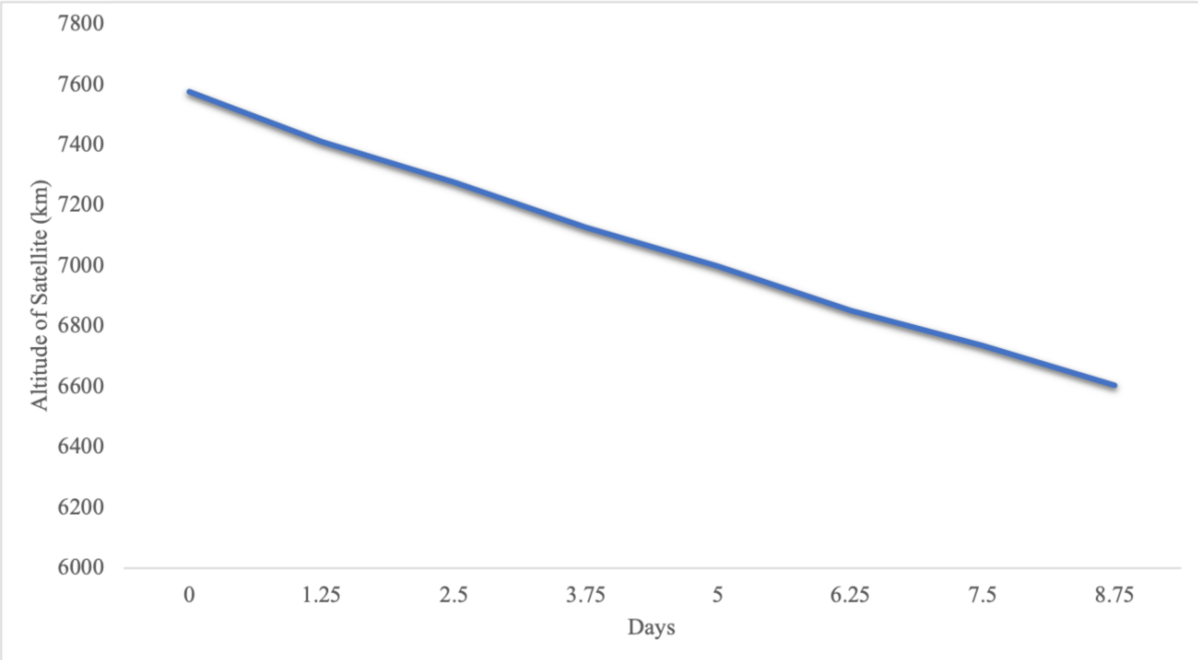


Figure 4.26 The thrust generated by E-Orbit for Debris Removal from 1200 km to reentry to Earth

If a breakup model occurs in HEO, the fragment velocity will change, and movement will be unpredictable; for example, COSMOS 917, a Soviet satellite launched into a Molniya orbit that broke up in 1977. Therefore, the high probability of propagation accuracy can be obtained even in long-term orbit propagation and needed. Currently, the perigee is 800 km, apogee is 39534 km, few fragments of space debris are still unknown, moving in a highly elliptical orbit.

To precisely calculate space debris, researchers use several perturbations in the force model created by Kyushu University under the guidance of Professor Toshiya Hanada. His work includes creating LEODEEM, GEODEEM, and NEODEEM [105-107]. However, when calculating the satellite force at a perigee above 2500 km, the atmospheric drag is negligible, while solar radiation pressure, solar gravitation attraction, lunar gravitational attraction J_2, J_{22}

perturbation are applied space objects as explained in Chapter 3 concerning the E-Sat model. If an object is in HEO, it is difficult to measure using the above force model. The objects/ debris moving fast at low altitude but slow at high altitude will get different perturbation forces with time. The study of debris in HEO is not carried out precisely. The study and creating the first HEO orbital model will benefit all force models for safe, secure and sustainable space launch. To carry out a mathematical simulation to create a perfect model for the HEO-Propagator: HEO Debris calculation model for accurate propagation, conjunction analysis, close approach, and collision probability. The current numerical approach-based model takes a long time with the help of an evolution integrated HEO-Propagator. It will give fast and accurate results to create better NEODEEM. HEO-based future projection is critical and helpful to track lost fragments that get high inclination or enter an elliptical orbit. As phase 1 of E-Orbit is S2S LPT demonstration mission will be used to analyze power transmission from E-Sat to daughter satellite and LPT efficiency concerning distance in HEO. At the same time, a CMOS and LiDAR-based debris detection payload will be added to create the perfect HEO-propagator model. In this, the sensor will continuously follow HEO and recharge at the LEO-based E-Sat. It will give excellent results for future LPT, laser thrust, and orbital transfer using HEO. A secure and reliable place for sustainable space application.

GEO satellites have many advantages and functionality for emerging nations. GEO satellite uses a stationary positioning and is essential for continuous service over a specific region, while HEO satellite provides more information during the approach to, and descent from, apogee. Bodies moving through the long apogee appear to move slowly and remain in high altitude over high latitude ground sites for more extended periods. This solves the GEO latitude problem. HEO is an elliptical orbit with high eccentricity used by telecommunications, navigation, and environmental study satellites. For example, the Molniya and Tundra orbit (used mainly by Russian and European space agencies) is used in the arctic region for hydro metrological observation chosen for its high advantages of long dwell time at a specific location or point. The normal perigee of HEO is 1000 km and apogee 35756 km.

The HEO-Proagotor will estimate collision probability in HEO. However, the chances of impact are much lower than LEO, but the same for semi-synchronous orbit, GTO, and GEO. In the proposed research plan, the LEO-based data collection and measurement of debris is more accessible than GEO-based debris with orientation and rotation of six-axis moments. Cataloguing and creating an HEO debris system's directory as LEO magnitude is high, GEO and medium earth orbit. As the speed is slower at high altitudes, it is easy to track and predict using mathematically for the next 100 years model. At the same time, the object's velocity at LEO will be higher than the ordinary object in LEO, which will help catalog the object orientation and inclination concerning HEO motion. It will help extend NEODEEM and GEODEEM models as GEODEEM has lower accuracy than LEODEEM. The complete system will be helpful for every system tool kit used by every space agency to send and create a new space mission. The Quasi-Zenith Satellite System (QZSS), also known as Michibiki Japanese and BeiDou satellite from China, which uses the same HEO system, will also help mitigate and prevent the point of an incident due to impact through HEO based debris system.

4.6. Market Analysis

Overall, The E-Orbit benefits over traditional power management and generation units like PV cells array, batteries, and MMRTG. The current total cost of MMRTG manufacturing and research is about 192 Mn USD for approximately 110 W, and it is used in curiosity Mars rover. The nominal cost of creating one satellite will require approximately 6-8 Mn USD. With PSLV launching, it will cost 18 - 28 Mn USD. At any single launch, a minimum of 4 – 8 E-Sat will be launched. As the launching in LEO is 5,000 USD and GEO is 30,000 USD, while the ride-sharing using SpaceX Falcon 9 costs 1 million USD for 200 kg, and Rocket lab charges at

20,000 USD per kg with 6 million USD for 300 kg, the new Virgin Orbit LauncherOne cost 24,000 USD per kg with 12 million USD for 500 kg ridesharing with the entire structure economic system details in the table.

Table 4.3 Projected Cost of E-Sat and E-Orbit

Parameters	E-Sat	E-Orbit
Number of satellites	1	1600
Cost of manufacturing	6-8 Mn USD	8-9.6 Bn USD
Cost of launching	25 Mn USD	5-5.5 Bn USD

The primary target market of E-Orbit is governmental, nongovernmental, private institutes. The space technology developer with the mega constellation capability to secure space with low cost of manufacturing and launching any satellites in space increases the payload capacity with EOL. The E-Orbit sale

- Baseload power to any SC in the prior range or within the E-Orbit system with minimum cost.
- Providing maneuvering of system, debris removal, and trust system on demand.

With the utilization of the E-Orbit system, the market price for space systems will drop by 15-20 % in the coming of the year with or without the RLV transportation system. The secondary market will provide infrastructure to the SSPS society to create technology for future clean, continuous, and renewable energy sources from space to Earth at any location on Earth. With the capability of every E-Sat is 10 kW, the system can govern and create market awareness for the SSPS project for future creation of power supply, space transportation and cleaning space. It includes

- Advanced Concepts of SSPS and Technology Research imitative;

- WPT Maturation and Demonstrations;
- GW System Demonstrations & SSPS Prototypes

Table 4.4 System analysis of SSPS

Parameter	SSPS ALPHA [38]	E-Sat Moon	E-Sat Mars
Total Power transmission	2000 MW	>10 kW	>10 kW
WPT	2.45 GHz (MPT)	1070 nm (LPT)	1070 nm (LPT)/5.8 GHz (MPT)
Solar Power Generation Efficiency	60%	30%	30%
WPT Efficiency	80%	60%	60% (LPT) / 70% (MPT)
Launching Cost (USD)	500/kg	165/kg * [108]	5358/kg * [109]
Cost to First Power (USD)	~31 B	~350 M	~500 M
Lifetime	>>30 Years	>>5	>>5
Levelized Cost of Electricity (USD)	0.09 per kW-hr	2.5 per kW-hr	4 per kW-hr

* using reusable launch vehicle

[118], while the cost of 10 kW E-Sat SSPS cost approximately 500 million USD. The total cost will be reduced with reusability for launching and placing an SSPS in Earth, the Moon and Mars orbit will be possible as it will deliver the power from 0.09 USD to 4 USD per kW as shown in Table 2.

4.7. Deep Space Mission

For outer planet operations, the solar intensity is too faint to conveniently allow solar energy to be used for spacecraft beyond the orbit of Jupiter. However, at Earth orbit and throughout the inner Solar System, SSSP technologies will provide a solution for high power for the electric propulsion system for deep space missions [38]. A continuous satellite array of SSPS will provide power transmission across the solar system. However, it will cost too much and will have a limitation of WPT depending on the distance. The deep-space mission will use RTG and a small nuclear reactor for further investigation and running the payload.

4.8. Comparison of E-Orbit and Solar Power Beaming Concept (2009)

Currently, there are specific SSPS constellations are made for in space power transmission systems. the closes comparison is based on LPT and low mass for power transmission can be done with the concept of Solar Power Beaming (2009) -SPB is made for

Table 4.5 Comparison of E-Orbit with Solar Power Beaming project

Elements	E-Orbit	SPB
PV	30 %	
Concentrator	Multiple	Multiple
Weight (kg)	250×1600	9125
Dimension (meters)	$3 \times 1 \times 1$	
Location	• LEO • Moon Orbit • Mars Orbit	• LEO
WPT	LPT	LPT
WPT Efficiency	48 %	50 %
Total Power (MW)	16	1
Power generation Method	• Bifacial PV • Multiple concentrators • Thermoelectric Generator	• PV • Hydrogen generator
Functionality	• WPT • Debris Removal technology • Orbital maneuvering • Laser thrust and propulsion system	• WPT

LEO-based 1 MW power transmission system, using LPT. The E-Orbit is unique design with multitasking approach.

4.9. Advantages and Disadvantages of Energy Orbit

All recent conceptual designs are based on modern technology; the total mass in orbit, cost, and complexity of the entire system decreased substantially, indicating massive progress towards a more efficient power system, as shown in Table 2.1. The primary SSPS design showed the orbit, shape, dimension, and maximum power transmission capability of conceptual design from 1973 to 2020. This design is capable of transmitting power to any remote location on Earth or constantly to one point. The majority of the project is based on MPT based system. However, the advancement of the high-power Laser will significantly reduce the size of the rectenna and can be used directly to pinpoint any desired location with the use of PV cells.

One of the main advantages of E-Orbit is to provide a continuous power supply to LEO-based space infrastructure. Elevate the power generation unit from SC and provide constant support in day and eclipse region in orbit. Equally power generated in orbit and distributed across the E-Orbit for maintaining power in all E-Sats. The E-Sats are designed based on 2020's technology. The E-Sat can be manufactured and assembled on Earth. A minimum of 6 E-Sats can be launched in orbit using Falcon 9 rocket. Cost-effective and create an economical solution from the next day in orbit. The disadvantages include its small size, the limited transmission of power over long distances, the lower accuracy of continuous tracking and power distribution with relative velocity without damaging the satellite.

The Laser based SSPS has disadvantages with the Earth's atmosphere. However, LPT is more beneficial for the Moon missions due to the moon's negligible atmosphere. A laser will transmit beams directly to a pinpoint location, allowing mobility and less complexity to the base station. The rover can work continuously in the crater and on the far side of the Moon without replenishing power. The Mars base system can control the MPT and LPT systems. It will allow stabilizing an entire city without relying on a nuclear generator or solar panel will allow mobility and continuous power transmission even during sandstorms.

4.10. Recommended Space Law and Policies for E-Orbit

The E-Orbit will elevate the dependency of PMS and resolve the space debris problem from space. The system provides potential benefits over the space traffic management to space utilization resources, interstellar travel, planetary defense system. The utilization of the Laser system for space application comes with the responsibility to share the space environment with other parties safely. This can include the intentional and non-intention misuse for the war creation scenario in space using a Laser system. the awareness of planetary protection and withstand cooperation within the member parties, national, international government to understand and awareness for the usefulness of the LPT system for humankind. Within the next decade, the number of satellites around Earth will increase by more than 80,000 satellites. The satellite conjunction and close approach between satellites will be a big task to resolve, with the stand to traffic management provide the new technology to demonstrate the space resources. The satellite-based ranging system will provide benefits over the space sustainability over the mega constellation designing. In LEO, drag plays an essential role in pulling the satellite towards the Earth, using a Laser-based propulsion system that can provide significant clean propulsion energy to any requested satellite. The projected modeling by 2029, there will show 2.5 million collision warnings, and to solve this, the E-Orbit can play a substantial role. Another important finding is that approximately 23,00 near-Earth asteroids and approximately 100 near-Earth comets are recorded by 2020. The planet protection agency is looking for a hazardous system that can create a catastrophic event on Earth. Using Laser technology can be used to protect the unidentified foreign object or registered foreign object to stop or change the impact direction using the Laser propulsion system. The E-Orbit Moon and Mars will provide an essential role for habitats and interplanetary missions in the upcoming future. With this, the MW and GW power generation system for Earth and Interplanetary will solve most energy problems from Earth and protect space from catastrophic events. The following recommendation should be following for better utilization of LPT and SSPS systems for sustainability.

- The peaceful utilization with the international parties’ agreements
- The acknowledgment and parties’ agreements should be held on the international level.
- The project should sponsor intergovernmental, international bodies to avoid inter conflict and space war scenarios.
- Avoiding the intercepting of any other satellite without any priors consent of manufacturer, operator, or governmental bodies.
- The Power transmission and beaming on satellite should be on perfect objective location, avoiding the blinding any optical sensor on the satellites.
- They were avoiding dazzling, damaging satellites bus or their subsystems.
- It should be not be treated as an antisatellite laser system, and it should only provide societal and economic benefits to the parties.
- The maneuvering of dead space debris to pursue a kinematic motion to target another dead debris or satellite should be avoided in any case.

Chapter 5

Interplanetary Energy Orbit

Chapter 5

Interplanetary Energy Orbit

5.1. Interplanetary Mission

It is the nature of humankind to build beyond what is perceived and pursue the unimaginable; ancestors migrated to all corners of the Earth, created settlements, constructed vast cities, started industrial revolutions, and visited the gray satellite of Earth, all from one basic sentiment: curiosity. It has led us to surpass the limits of humankind, one hallmark being the launching of the first artificial satellite Sputnik by the USSR in LEO on October 4, 1957, which spurred the launching of thousands of satellites into space and across the solar system for a variety of purposes. The search for a new habitat prompted the mission to the Moon Luna 2 in 1959, and currently, the Mars 2020 Mission, Perseverance rover, and the Ingenuity helicopter drone, 2020. Such missions to the Moon and Mars require continuous electric power, especially with discovering water on the Moon through the Chandrayaan 1 mission using the Moon Mineralogy Mapper M^3 [110]. The success of the ISRO would be the turning point for future Moon outposts on the south and north pole.

The Moon has 14 days, nights, and 14 days of sunlight, making it difficult to establish a permanent base without a nuclear reactor or large battery storage capacity. Moreover, the two polar regions of the Moon have permanent shadows for several months. With the current battery technology, only nuclear energy or a stored battery pack will be able to fully fuel research across the far Dark Side of the Moon. It is appalling that power storage technology has not improved since the first satellite to date. Thereby, we propose that the WPT will play an essential and significant role in exploring lunar night and the far side of the Moon.

On the other hand, the solar irradiation on Mars is significantly reduced than on Earth due to the extended distance from the Sun. Thereby, most Mars missions that depend on nuclear reactors or photovoltaic cells would generate significantly lower power at the surface due to sandstorms and reduced sunlight. Current power generation and management technology, consisting of nuclear reactors and photovoltaic battery cells, are insufficient to power a habitat. Thereby, this paper discusses using a constellation of small SSPS to provide a constant power source via WPT to the Moon/Mars orbiter, habitat stations, rovers, and UAV, regardless of location or time, or the presence of light [111]. This expands into alternative possibilities of mobile habitat systems in the South Pole {Aitken Basin at the Moon [112], or even on Mars. The SSPS constellation will be referred to as E-Orbit, Moon, and Mars, E-Orbit Moon and E-Orbit Mars, respectively.

The leading technologies used for delivering wireless electricity involve microwaves and lasers. WPT microwave technology has had a significant amount of scientific research, development, and testing conducted to advance frequency range use with high efficiency. Moreover, far-distance transmission using such technology is highly discussed around the SSPS community [113]. Recently, Japan and the USA have already performed several experiments using lasers for far-field transmission. Meanwhile, the laser WPT system as an energy source on Earth has drawbacks such as scattering and divergence due to its high-density atmosphere. However, it would play a significant role in Moon missions, as the Moon contains an exceptionally low atmospheric density at 1,000,000 molecules per centimeter cube, considered a vacuum on Earth. The density of the atmosphere at the Moon's surface is comparable to the density of the outermost fringes of the atmosphere of Earth, where the International Space Station orbit lies [114]. The laser WPT can pass directly without divergence or scattering caused by clouds or any atmosphere. The E-Orbit satellite will take advantage of the Moon's low atmospheric density for directing a pinpoint laser beam for continuous power supply missions on the Moon and the surface of Mars. The Moon and Mars E-Orbit will be discussed in this paper.

To date, humankind has sent numerous research-oriented orbiters, landers, and rovers to the Moon, Mars, and on asteroids. All missions consist of general tasks, including various operations, analysis, and transmission of research data back to Earth. Simultaneously the compression of traditional power generation methods for the orbiter, lander, and habitat on Mars and the Moon. The interplanetary mission where the space engineers work on the satellite, conceptual design of space habitat, and exploration system. State of the art in those missions relies on the radioisotope thermoelectric generator or solar panel attached with batteries to store power, plagued by certain limitations. Such tasks require power generation and management source units. The systems are bulky and have limited functional capacity and lifetime, further affected by unique space environments. For instance, when a spacecraft moves away from the Sun, energy collection efficiency on the solar panel reduces. If the temperature drops below -100 Celsius, the attached solar array shows deteriorated performance than expected due to the unpredictable degradation of individual solar cells. The far side of the Moon has 14 days of continuous darkness. On Mars, the incessant sandstorms and further distance to the Sun decrease the efficiency of concurrent power generation technologies.

Besides, these power generation unit satellites carry a pack of batteries to store energy, and the total system makes more than 10 – 25 % of the mass of the satellites. In hindsight, space solar power satellite serves a potential for a better energy transmission source than the traditional method. Space agencies have already studied space solar power station design concepts for gigawatts wireless power transmission systems from space to Earth to fulfill global electricity demand. Nevertheless, another field where space solar power satellites can be applied is for rovers and habitats, as shown in Figure 4.23. On Mars, rovers experience difficulty collecting sunlight due to sandstorms affecting energy collection at the attached solar panel.

Moreover, the rovers must investigate at the far side of the Moon, where sunlight is unavailable for a few days. This challenge can be suitably overcome by employing space solar power satellites, which can be used for wireless power transmission, independent of its location. Such techniques have possible applications towards power transmission for unmanned aerial vehicles for faster mapping purposes. As such, the dependence of those aerial vehicles towards fixed energy storage becomes alleviated. This chapter advocates the design and usage of a constellation of small Space Solar Power Satellites as Energy Orbit for future habitats, orbiters, rovers, and unmanned aerial vehicles for fast mapping, research, and investigation. The idea of Energy Orbit and how it will increase the lifetime, payload capacity, and reduce the mission's cost.

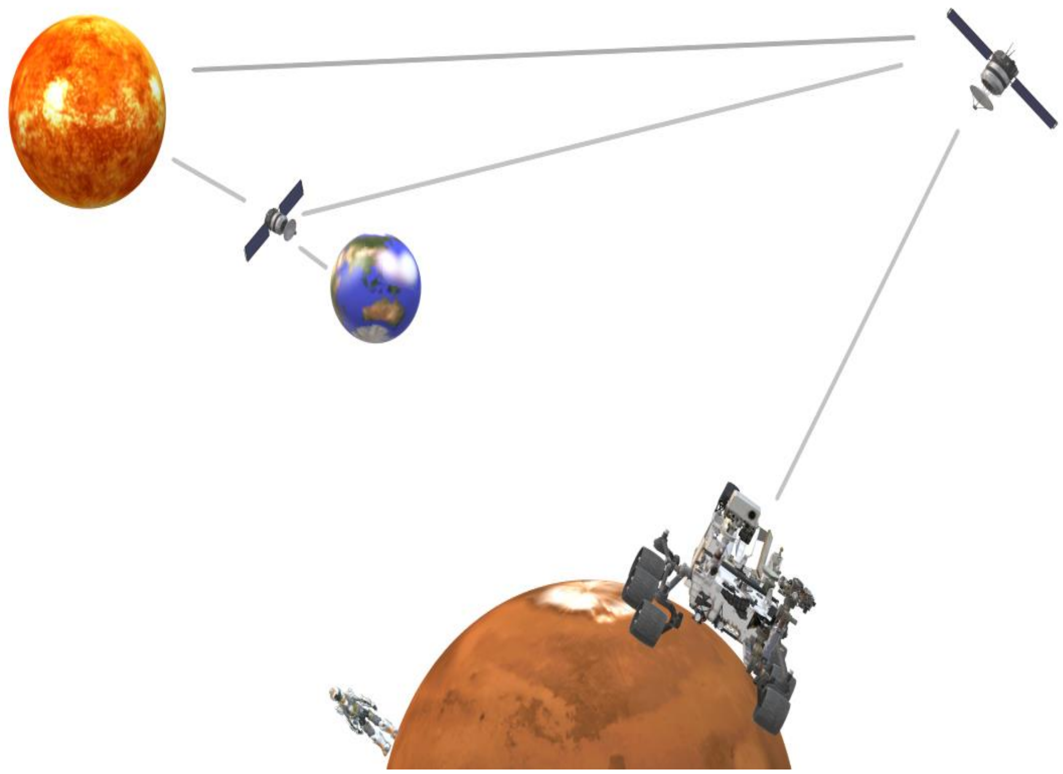


Figure 5.1 Conceptual design of E-Sat for powering Interplanetary Mission

5.2. Wireless Power Transmission for Interplanetary Mission

In general, an effective WPT must firstly enable transmission through the atmosphere, secondly, be a simple design with a directional emission, and thirdly, be easy to convert from an energy source (e.g.}., solar, electricity, heat) to its transmittable form ({e.g.}., Laser, microwave) and be able to convert back into an electrical energy form. By these standards, MPT is superior to laser power as the former has been highly researched incredibly for long-distance transmission [18, 19, 54]. MPT can be transmitted in more dense atmospheric conditions compared to LPT. However, the Moon has a low-density environment, and eventually, missions for colonizing Mars would need a continuous power beam where LPT will perform a significant role for interplanetary WPT [74].

5.3. Power Transmission from Satellite to Receiving Site

The previous section chapter the SSPS concept for Earth. The space utilization of WPT is still in development, and the conceptual design needs verification and demonstration. Therefore, a small SSPS will fill the gaps in the demonstration of technology [113]. The interplanetary mission can utilize the primary SSPS concept to demonstrate the feasibility of design and efficiency by using a small SSPS design to use WPT to connect with rover and orbiter on the Moon via an LPT-based system in Figure 4.24.

This system is designed in Space System Dynamic Laboratory, and the small SSPS are called E-Sat. The system will be generated and transmit approximately 10 kW of energy to any receiving site. The primary rectenna system can be used as PV cells of a rover, lander, and orbiter. It will provide continuous power to the remote location and inside the crater. The satellite is a 54m^2 PV solar panel with an efficiency of 30% to collect solar irradiance on the PV cells. The satellite will demonstrate the uses of power capability with performance to demonstrate the next generation of SSPS for the Earth.

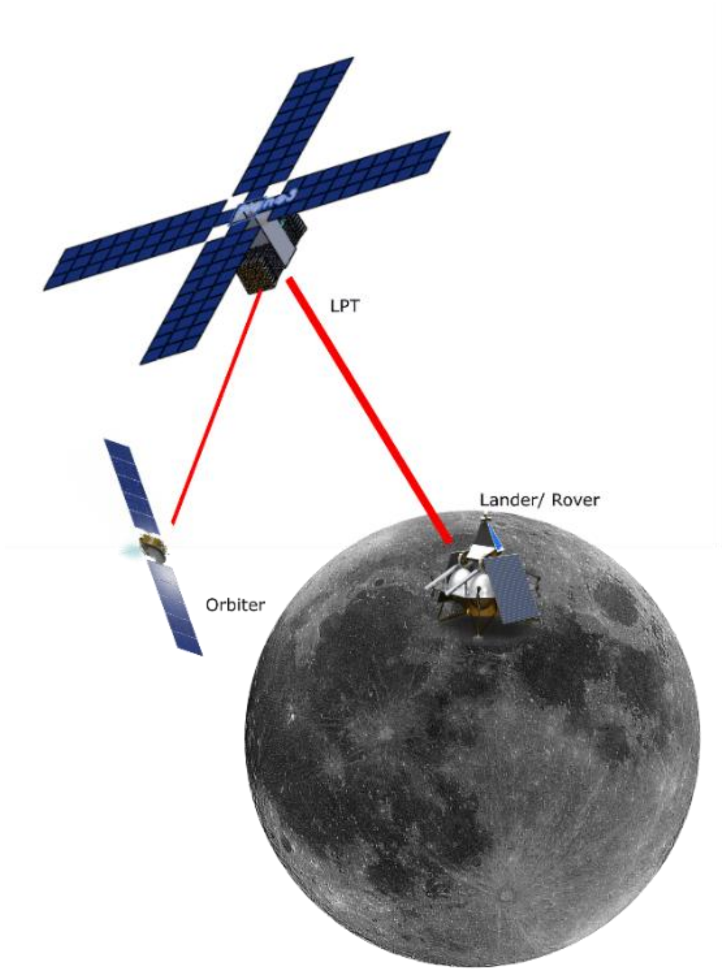


Figure 5.2 SSPS conceptual design for Moon Mission

5.4. Problem on Interplanetary Mission

The SSPS conceptual mission plans and different types of design that will solve significant earth problems are discussed and highly researched in the scientific community and space agency. JAXA, CNSA are trying to design and place it by the 2030s. However, in the present scenario, going back to the Moon and Mars will need continuous non-disruptive power sources with mobility capability. The SSPS design for an interplanetary mission will help investigate the Moon, Mars, and asteroid. It will play an essential role in exploration and for the deep space mission. This section describes why SSPS is important for Moon, Mars, and Orbiter around Earth.

5.4.1. Moon

The Moon plays an essential role in investigating and creating a permanent habitat station for further interplanetary traveling. USA, Russia, India, and China sent their rover to the Moon’s surface to search for the water, and the Chandrayaan-1 component M^3 has detected OH/H_2O on the uppermost surface of the Moon [115]. The most habitable place on the Moon is the polar region and crater over there. However, most of the craters are in the permanent or few months of the shaded region. The temperature over in the crater is lower than compared to other parts of the surface. The LCROSS mission was to detect hydrogen concentration using LEND on the LRO spacecraft collected data and send back to Earth. The data collected at the Cabeus crater, using an impactor, showed the highest number of hydrogen concentrations in the South Pole of the Moon. Data suggests that 0.5 to 4 % of water exists in the form of ice [116]. Finding water molecules on the Moon opens a new way to sustain and develop an outpost for research and create a permanent station for an interplanetary mission. However, the temperature in the craters is cold, and due to the lack of sunlight for a few months, we will need a continuous power supply of power to heat the habitat module. Figure 5.3 shows the temperature in a South Pole crater at daytime, nighttime, and an annual average temperature by LCROSS; the white dote in the figure in D section is the captured ice module inside the crater.

Moreover, data suggests that the solar wind from the Sun implanted helium3 on the moon's surface. As the atmosphere on the Moon is negligible, the surface has a layer of the precious elements at the far side Maria region and includes TiO_2 near the surface side mare region on the Moon. The helium three on Earth is rare, and it is approximately 20 tonnes. It is a radioactive element used as a fusion reactant, with a high potential to fuel the rocket for a deep space mission [117]. A few companies started the space-mining project. Mining in space will require a continuous power supply to support prominent mining instruments. Besides, 14 days of sunlight and 14 days of darkness limit the capability of landers and rovers for exploring on the surface. However, space agencies are using modern RTG for providing sufficient electricity to run equipment on the surface. The RTG has limited functionality and increases the payload of the rover. The rover uses photovoltaic cells that only generate electricity during the peak time. The solar irradiance at the Moon is similar to near-earth orbit 1365W/m^2; however, the generation time is only limited to 14 days with peak time. To stabilize a habitat and research in the crater will need continuous power generation and management units.

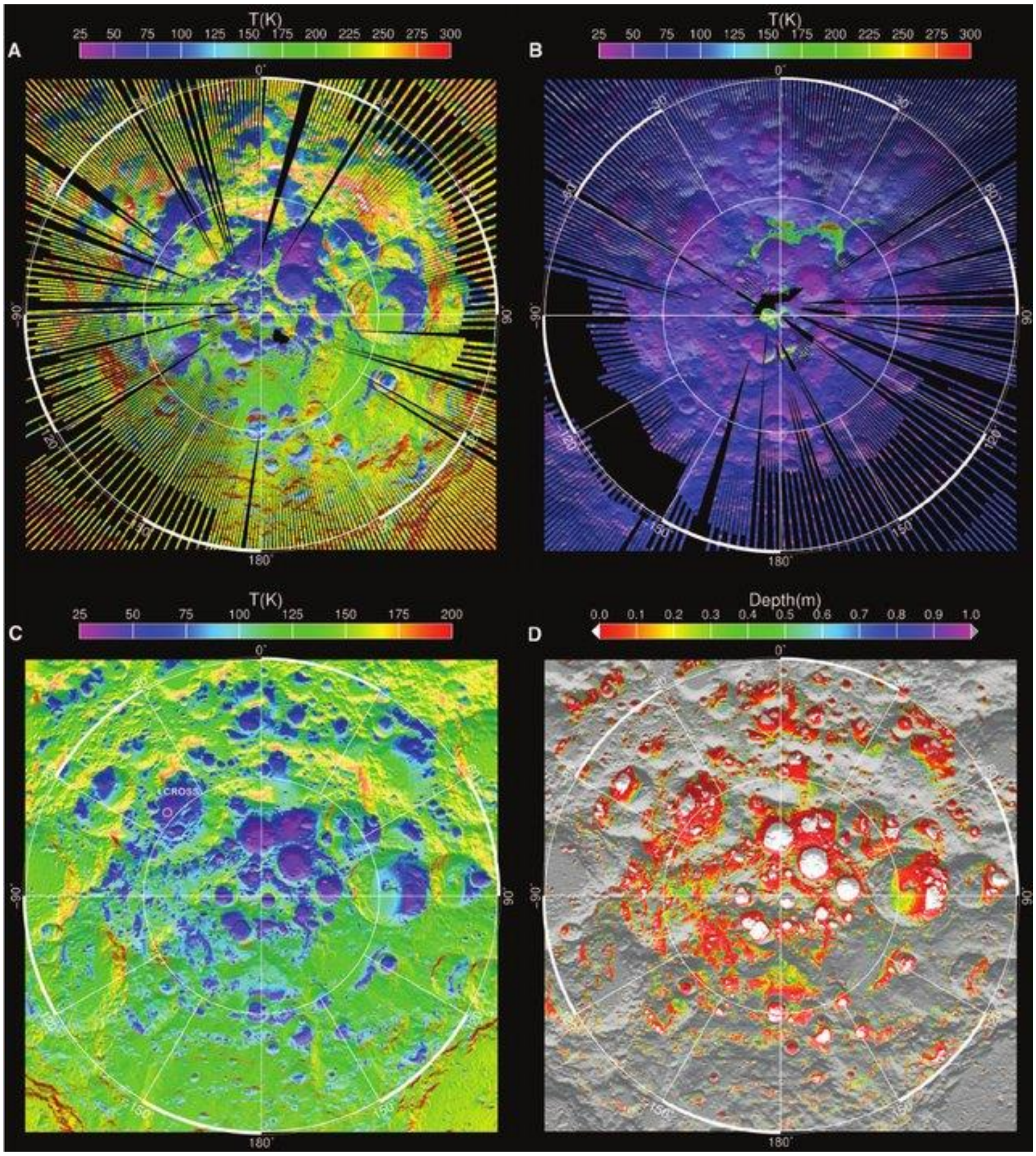


Figure 5.3 Maps of measured and model-calculated surface and subsurface temperatures in the lunar South Polar Region. The outer circle on all maps is 80° south latitude. Observations were acquired between 6 September and 3 October 2009 as the Moon approached the southern summer solstice. (A)Diviner-measured daytime bolometric brightness temperatures were acquired between 11.4- and 13.6-hours local time (5). (B) Diviner-measured nighttime bolometric brightness temperatures acquired between 21.41- and 1.66-hours local time (5). (C) Model-calculated annual average near-surface temperatures and the location of the LCROSS impact in Cabeus Crater. (D) Model-calculated depths at which water ice would be lost to sublimation at a rate of less than 1 kg m^{-2} per billion years. The white regions define the locations where water or ice can currently be cold trapped on the surface, the colored regions define the upper surface of the lunar ice permafrost boundary, and the grey regions define locations where subsurface temperatures are too warm to permit the cold trapping of water ice within one of the surfaces. Source [118]

5.4.2. Mars

Problems on Mars are significantly higher in comparison to the Moon. Mars is the fourth planet in the solar system. The distance from Sun to Mars is approximately 210.96 million km. Due to this, the photonic energy on the surface and in orbit is significantly lowered; the solar irradiance on Mars is around 590 W/m^2. Moreover, the peak time for power generation using PV cells is only two hours. Due to this, most Mars missions rely on RTG. The RTGs provide significant power to run avionics for small research-based rovers or landers. For example, the RTGs on Mars2020's Perseverance rover use MMRTG to deliver 110 W for functionality. Curiosity is using 100 W while a lifetime of 10-15 years, and due to the use of radioisotope as fuel, it degrades by a few percent per year. Mars sandstorm would most likely cover the solar panel, which significantly stops the power generation unit on the rover.

The colonization of Mars is becoming more popular. Space agencies proposed different structures and mission planes to becoming an interplanetary species. The first step is to create an RLV. SpaceX already demonstrates and started a transportation system between Earth and ISS using the Falcon series. They are in the progress of making a considerable rocket call Starship. SpaceX plans the Starship will carry astronauts and civilians to mars to establish a fully functional city. The United Arab Emirates space agency already started to research building an entire city on Mars by 2117. The Mars500, Sirius, and Mars Desert Laboratory are already investigating the possibility and challenges for living on Mars by long-term analog astronaut mission.

However, the designing of habitat and an entire city will need continuous power of energy. The small RTGs will not full fill the demand of the habitat system. The GW modern nuclear power plant will be the option for Mars city. However, it is dangerous to carry a significant amount of radioisotope using a rocket. A sufficient electricity system will require a renewable, safe and continuous power supply that is also mobile. To make Mars a habitable place for thousands of people.

5.4.3. Orbiter

ISS, satellites, and orbiters require a continuous power supply for on-running avionics and payload. The ISS alone is a size of a football ground. However, the solar array takes more than half of its size to produce 120 kW to maintain functionality. The agency plans to make lunar Gateways and Artemis project for a fast way to occupying Mars. The Artemis will require approximately 100kW to sustain and maintain the mission outside the Earth orbit.

The orbiter and satellite in the near orbit of the Moon and Mars use a solar panel to produce electricity to sustain the avionics and payload for scientific data collection and communication between the lander. However, in a shadow region or when sunlight is unavailable, RTG must be used to go back to the dark side or during the eclipse. The total power generation and management unit take up approximately 10~25% mass of any satellite, including batteries. Using WPT and SSPS, all the above problems can be solved. It will provide continuous, undisruptive power with more mobility to use the high payload capacity. For this, the concept of SSPS can be applied not only on Earth but for the interplanetary mission, as discussed in this paper.

5.5. Power Management for Interplanetary Mission

The Sun is a big power hub, luminating a tremendous amount of energy with the luminosity being approximately 3.827 x 1026 W/second. Nearby the Earth's orbit, the solar irradiance is approximately 1360 W/m^2 with slight changes according to the distance. The Earth rotates around the Sun following an elliptical path with distances between 147 million km to 152 million km to the Sun. Meanwhile, the Moon also follows an elliptical orbit around the Earth, sometimes getting as close as 363,000 km, and other times as far as 406,000 km.

The closest point that the Moon can get to the Sun is when the Earth is also at its closest point in orbit to the Sun, and the Moon is the most distant from the Earth. The nearest point that the Moon can get to the Sun is 146,692,378 km, while the furthest distance of the Moon would be 152,503,397 km. This indicates the importance of power generation using SSPS in the Moon orbit and at the Lagrange point. However, the power generation can occur only in daylight time when it phases towards the Sun. On the other hand, on Mars, the solar irradiance on the surface of Mars in the afternoon with maximum sunlight is 590 W/ m^2, equating to the Sun's intensity at Earth at 36 degrees. The distance is too large compared to Earth and Moon with 208.23 million km from the Sun. Besides, Mars has continuous sandstorms at its surface, which would affect the current power generation systems. Similarly, the average temperature on Mars and the Moon is -63 °C and ranges from 107 °C to -153 °C, respectively, making it a unique challenge to mandating a thermal unit and power generation system that can avoid heating effects. At the same time, the habitat would need to maintain standard temperature to stay alive. Thereby establishing habitats away from Earth would heavily rely on a continuous power source.

The small SSPS constellation- E-Orbit will generate continuous power 24 x 7 on the Moon even when it is on the dark side. The constellation is designed to produce power and transmit energy from one satellite [15] to the other even when they are not in direct sunlight.

5.6. Energy Orbit for Interplanetary Mission

E-Orbit is an LPT-based small SSPS constellation. It consists of 6-7 planes depending on the planet or celestial body, with five satellites in each orbit, 72 degrees apart from one other. The power is generated and transmitted to all energy satellites in orbit using PV cells. The E-Orbit for Moon and Maras can be seen in Figures 4.26 and 4.27.

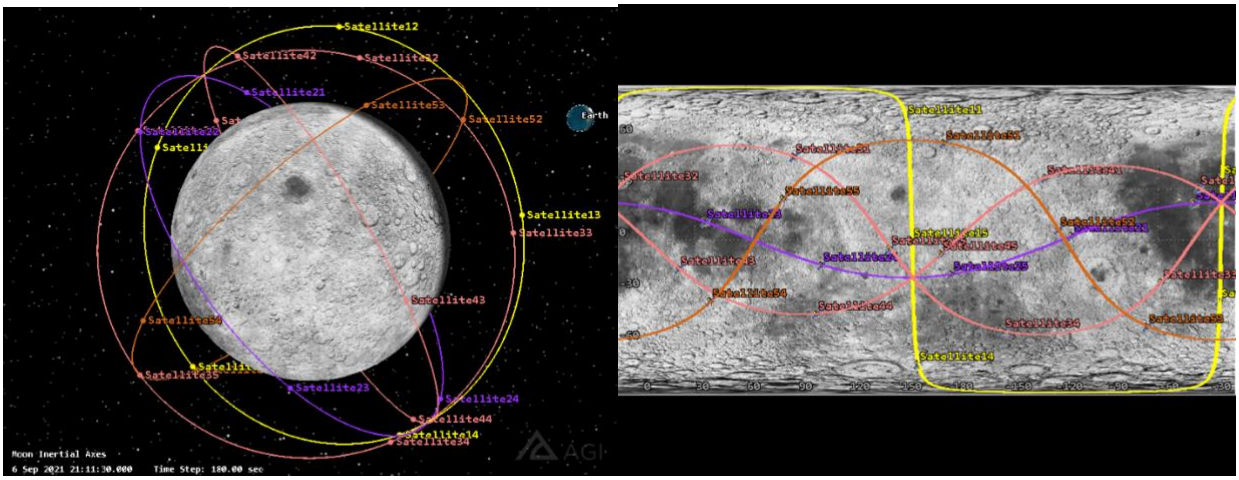


Figure 5.4 E-Orbit satellite constellation for Moon simulated in Systems Tool Kit

(a) E-Sat constellation in orbit (b) Satellite tracking on the surface

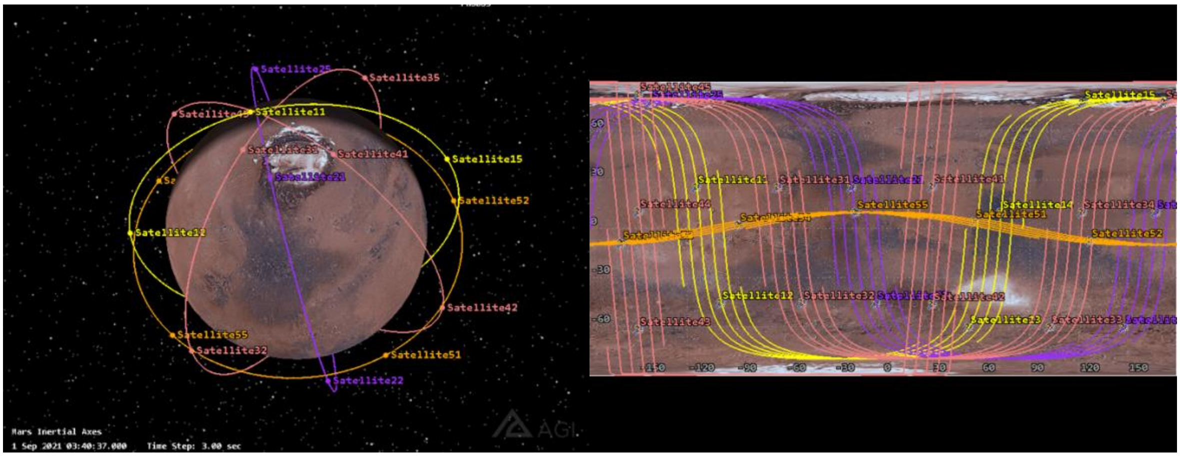


Figure 5.5 E-Orbit satellite constellation for Mars simulated in Systems Tool Kit

(a)E-Sat constellation in orbit (b) Satellite tracking on the surface

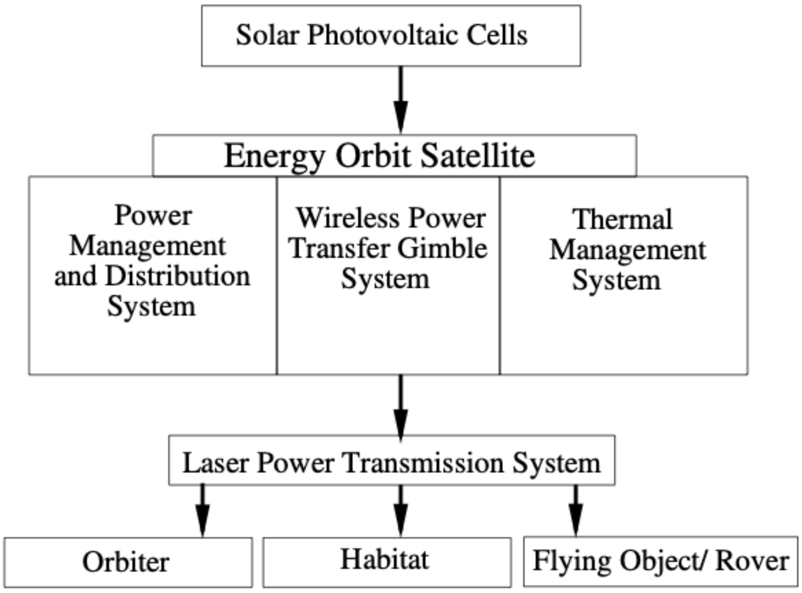


Figure 5.6 E-Orbit functional architecture

The transmission method would depend on the planet, according to the distance from the Sun, and other different specifications. The Moon Energy Orbit consists of five planes with 25 satellites, as shown in Figures 4.26 a and b. The satellite is at 900 km in the Moon E-Orbit with an inclination of 98, 08, 135, 45, 80 degrees and RAAN of 70, 70, 70, 70, 340, degrees respectively. E-Orbit is in 5 planes with 25 satellites at 900km for the Mars orbit with an inclination of 80, 80, 80, 45, 10 degrees, and RAAN of 50, 145, 100, 70, 240 degrees Figure 4.27 a and b. The Mars E-Orbit satellite represents the design for mission in polar region support. Orbiter, space application, and ground support include missions using rover, habitat, and UAV, as shown in Figure 4.28. Every surface-based objective power receiving device will be interlinked with three satellites for continued undisturbed signals to receive its required power transmission.

The pilot signal for secure connection also functions like the GPS for the object, located on its surface for assistance. The entire constellation of SSPS will provide an additional benefit for establishing a continuous GPS designed for intended missions. The orbiter and satellite are interlinked with a minimum of six SSPS satellites for improved connectivity using the LPT system. It will also track the satellite and help reduce the production of space debris in the future.

The interlink connection is established by confirming the pilot and aiming subsystem signal from an E-Orbit Satellite to other objective devices like orbiters, satellites, or any habitat modules, including mobile modules or permanent fixed modules, rovers, flying objects. For 10kW Laser, multiple 200W laser systems are used to pinpoint a single object. If a mobile object gets concentrated power, the system can use a high-power electronics system on the surface and orbiter for mapping and research.

5.7. Comparison: Traditional Method vs. E-Orbit Wireless Power Transmission

Typically, > 99 % launched mission uses PV array for power generation unit including Earth orbiter in LEO, MEO, GEO, HEO, planetary fly byes, orbiter, Ion propulsion mission: DS-1, Mars, Jupiter, Venus, mercury, surface mission on Mars and Moon and ISS. the power system is about 20-30 % of spacecraft mass and cost 20% of the budget which majorly used as Power management distribution, power generation, and energy storage [119] ISS arrays are about 2,500 m^2 to collect energy into 27 W/Kg and cost around 3,500 USD/W, and 12.9km long wired line for transmission across the station. In 2017 ISS upgraded with 24 new lightweight lithium-ion battery cells installed. The cost of upgrading, launching, and maintaining the space energy system is high, and it can be lover down with this technology. A standard communication satellite in GEO with a total eclipse load of 5-10 kW, the electric power system is about 30 % of a satellite's total dry mass, 40 % is structural and other is, and 30 % is the payload for all spacecraft [120]. Sometimes RTG and Solar panel GaAs solar cells for power and batteries are used for storage. E.g., Mars 2020 Perseverance rover would use the Multi-Mission RTG system for 14 years of the operational lifetime, 45 kg in weight to generate 110 W (assuming battery life deteriorates a few percent per year) with lithium-ion batteries. Chandrayaan-1 panels generate an average of 750 W supported by Li-ion batteries. Figure 5.7 shows the electric power necessity by the missions. Usually, an electrical power and solar paddle subsystem account for 15-20 % of a satellite's total weight [121].

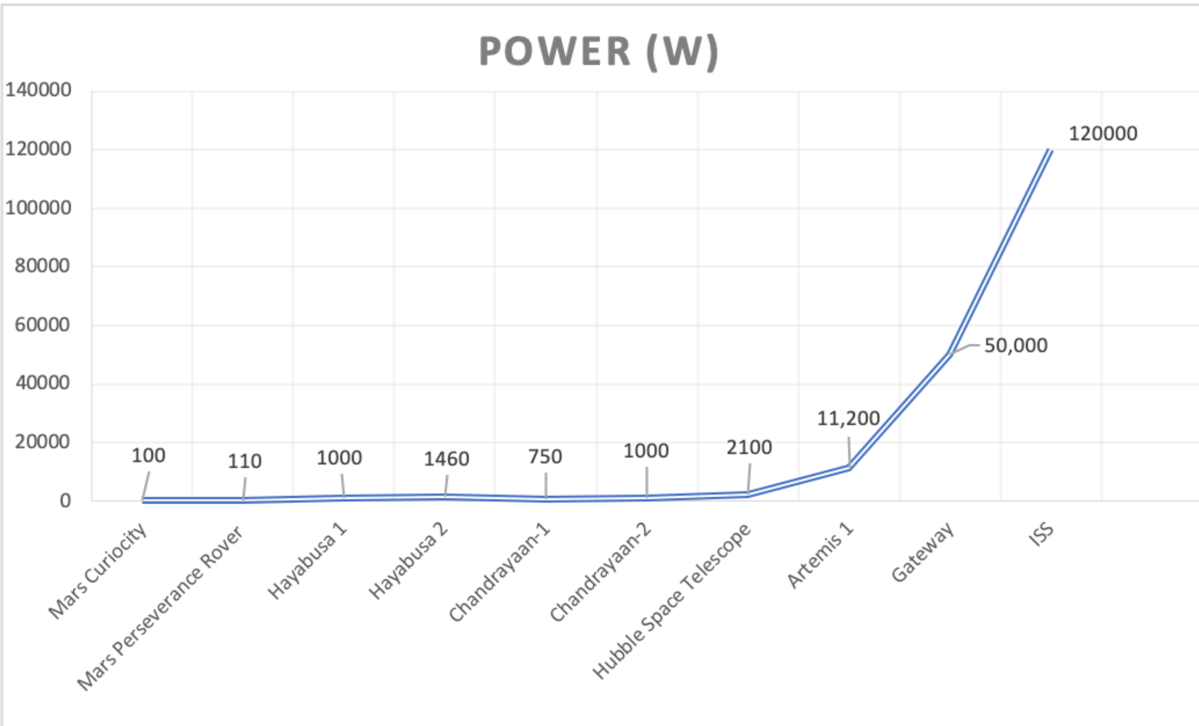


Figure 5.7 Electric power usage in different space mission

Previously, when the space shuttle was in operation, it cost 54,500 USD per kilogram; now, due to competition and the majority of space startups as SpaceX Falcon9, ISS is accessible with just 5,000 USD per kilogram [122]. In the future, it will be lower and more affordable. As mentioned above, 30 % of dry mass will be directly cut down if the WPT technology will be used, and just approximately 81,000 USD can be saved for LEO with standard launch vehicles like Soyuz PSLV or reusable launch vehicles.

The other proposed SSPS design focuses on transmitting a GW of power to the Earth-based ground station. The proposed space design cable to transmit up to 230W m^2 to earth-based rectenna. The LPT-based system significantly improves solar PV cells' efficiency by twice, allowing E-Sat meaningful LPT use. The technology demonstrated by SPS will require billions of US dollars and more to develop reusable transportation vehicles to place. Moreover, designing a small SSPS capable of 10 kW can show capabilities and reduce the cost and utilization. The reduced mass of satellites can be predicted and used for other payload systems. The 10kW power system will provide an interplanetary mission opportunity to utilize high-power electronics devices for research. Therefore, it will cut down to 30 % of space energy management and utilization system by replacing batteries, PV array, and reducing the launching weight will lower the space research mission's price. However, small equipment will be used for the conversion and transmission of the LPT system on objects. In account, it will be less than 5 % of the total current space power management system.

Transmission is more straightforward in comparison. Every E-Orbit Satellite capable of 10 kW transmission will be easier to launch using current transportation vehicles to LEO, Moon, and Mars orbit. E-Sat's weight will be minimal, and eight E-Sat can be transported using Long March 3D space launching vehicle. The demonstration and capability of SSPS will change the way of power use and transmission. WPT demonstration using E-Orbit will allow operation in remote locations on Earth, Moon, and Mars. The constellation is designed for the continuous WPT to any receiving site. In E-Orbit, the receiving site can connect with a minimum of three other E-Sat for a non-disruptive signal. This feature is distinctive from other proposed SSPS Concepts. If the receiving site requires more than 10 kW of power, other nearby flying E-Sat can transmit the additional required power.

Chapter 6

Conclusion

Chapter 6

Conclusion

5. Conclusions

The SSPS represents an enormous project, the recently proposed conceptual design requiring cooperation from all space agencies and researchers to launch hundreds of rockets. The considerable cost of launching and stabilizing SSPS in orbit presents a significant challenge for every researcher. However, the demonstrative prototype project can be introduced by employing a Moon or Mars-based small SSPS system that supports a rover or orbiter. The current challenge to an SSPS design is developing an efficient WPT system. The type and place of application will define the proper selection of WPT. For a typical example, an MPT-based SSPS is more appropriate for the Earth-based system for GWs system and will have higher application than the LPT system. The MPT is already thoroughly developed and investigated. It possesses the capability of penetration from the Earth environment with minor distortion. On the other hand, the LPT application is wholly appropriate for low atmosphere applications, such as space to space power transmission, on the Moon and Mars. A small SSPS can support rovers, landers, and continuous flying unmanned aerial vehicles.

Specialty design SSPS modules can be advantageous to suit the needs for various sectors, including military, national defense, commercial and industrial plants which all require power on specific locations. Many leading researchers proposed the prominent constellation of SSPS, which has significant worldwide use for providing carbon-free energy to any convenient location on Earth. In the near future, a Constellation of small SSPS or E-Orbit can potentially provide continuous power everywhere.

The novel E-Orbit concept properly represents an application-oriented design for creating an economic SSPS system for space application that is significant to the current technology. The previously proposed design is futuristic; however, the current abstract of E-Sat and the prominent constellation of E-Sats, E-Orbit, will subtly change the social perspective of energy generation and effective utilization in space. The E-Orbit will directly cut down the satellite's weight by changing PMS to a wireless PMS system. It will generate an economical solution with greater efficiency than any other current method. Just by changing the PMS system, costs for manufacturing and launching satellites can be directly cut down. At the same time, a changed PMS system can provide a highly efficient, high-quality, power electronics-based payload system for space application with its compact design. As PMS is a critical factor for defining EOL for any specific space mission, the E-Orbit will provide a significant advantage for extending satellite life.

In addition to contributing to energy efficiency, SSPS can also be used to sustain the special environment. Debris is one of the significant hurdles to particular expansion due to the increasing number of mega-constellations in space. Using thrust maneuvering, the Debris can be removed, and additionally, at the EOL of E-Sat, it can rideshare a dead satellite to the lower orbit. Space agencies are looking forward to substantial wealth across the space sector by extending satellite life will create wealth, and at the same time, removing the debris by orbital maneuver and de-orbiting will enhance a significantly sustainable space environment. The changing orbit will drastically cut down the considerable cost of satellite industries for higher altitudes by orbital laser propulsion.

E-Orbit can support the colonization of Mars and properly provide mobility to habitats. It can provide effective communication and positioning systems to accurately track and efficiently transfer WPT to the desired location. The WPT system will define the future technology of Earth and Space Exploration. Interplanetary research and space missions will require significant development in developed technology and avionics in which WPT will execute vital operations that will enable missions to remote locations anywhere on the Moon/Mars surface and their orbiters. The location constraints will no longer hinder current developments and will also open doors to scientific investigations on the far side of the Moon, not to mention easy access to polar regions for settlement.

The prominent constellation of SSPS will create a perfect E-Orbit, and with its continuous backup supply of power to the surface and space, it can deliver tremendous power to all technology. The constellation will also help develop the GPS for Mars and the Moon, evaluating and positively confirming its connected devices' precise location while efficiently delivering power. Simultaneously, it will allow the receiving satellites to perform with higher payload capacity. As the E-Orbit will provide direct power, all mission vehicles will have reduced weight and size that would have been used for carrying power generation and management units for the orbiter. E-Orbit will directly cut down the manufacturing and launching costs by 20-25%. Also, such mission vehicles will not be relying on their power generation unit. Using the E-Orbit would benefit mission vehicles by extending their mission lifetime and increasing their mobility.

E-orbit can be applied towards the sustainable space environment and utilize the non - SSPS potential application such as solar electric power and propulsion for exploration of orbital transfer vehicles for Earth orbit transportation, interplanetary, robotic science, and human explorations precursor mission. The future applications of SSPS include its role as a point of energy source from Lagrange point and asteroid belt for deep space missions using solar sail technology. The energy generation system can be planted near Earth, at the Lagrange point, and in the interplanetary orbit. Use the Sun energy from the solar storm and utilize it for high power application in the solar system.

Reference

[1] World Population, Worldometer, 2021 https://www.worldometers.info/world-population/ (accessed 25 July, 2021)

[2] Electricity Production, Global Energy Statistical Yearbook 2021, Enerdata https://yearbook.enerdata.net/electricity/world-electricity-production-statistics.html (accessed 25 July, 2021)

[3] Electricity Domestic Consumption, Global Energy Statistical Yearbook 2021, Enerdata, https://yearbook.enerdata.net/electricity/electricity-domestic-consumption-data.html (accessed 25 July, 2021)

[4] IRENA (2018), Global Energy Transformation: A roadmap to 2050, International Renewable Energy Agency, Abu Dhabi. (accessed 25 July, 2021)

[5] Energy consumption in Japan, WorldData, https://www.worlddata.info/asia/japan/energy-consumption.php (accessed 25 July, 2021)

[6] H. Grant, P. Heptonstall, R. Gross, and D. Handley. "Cost estimates for nuclear power in the UK. Energy Policy 62 (2013) 431-442.

[7] P. E. Glaser, Method and Apparatus for Converting Solar Radiation to Electrical Power, 3,781,647. (1973).

[8] S. T. Goh and S. A. R. Zekavat, Space solar power orbit design and cost analysis, 7th International Conference on Recent Advances in Space Technologies (RAST), Istanbul, (2015) pp. 753-758.

[9] J. Huenteler, Tobias S. Schmidt, and K. Norichika, Japan's post-Fukushima challenge–implications from the German experience on renewable energy policy. Energy Policy 45 (2012) 6-11.

[10] T. H. Bakken, A. G. Aase, D. Hagen, H. Sundt, D. N. Barton, and P. Lujala, Demonstrating a new framework for the comparison of environmental impacts from small-and large-scale hydropower and wind power projects. Journal of Environmental Management, 140, (2014) pp.93-101.

[11] P. E. Glaser, Satellite solar power station. Solar Energy 12, no. 3 (1969) 353-361.

[12] S. Abdelhady, An Entropy Approach to Tesla's Discovery of Wireless Power Transmission, J. Electromagn. Anal. Appl., Vol. 05, No. 04, (2013) pp. 157–161.

[13] A. Kalyan, S. S. Thanapal, and D. Ranjan, Ranking renewable and fossil fuels on global warming potential using respiratory quotient concept. Journal of Combustion (2018).

[14] V. Knap, L. K. Vestergaard, D. I. Stroe. A review of battery technology in cubesats and small satellite solutions. Energies. (2020 Jan);13(16):4097.

[15] A. Ball, J. Garry, R. Lorenz, & V. Kerzhanovich, Planetary landers and entry probes. Cambridge University Press, (2007).

[16] A. Baraskar, Small Satellite to Satellite Space Solar Power Using Wireless Power Transmission System. In Гагаринские чтения-2019 (2019), pp. 1306-1306.

[17] P. E. Glaser, Power from the Sun: Its Future, Science 162,3856,57–861. (1968).

[18] P. Jaffe, B. Kaylin, B. Colin, C. DePuma, L. Longbottom, H. Nisar, and V. Simlot. Opportunities and Challenges for Space Solar for Remote Installation. Naval Research Lab Washington DC United States, (2019).

[19] P. E. Glaser, E. Owen, J. J. R. Mackovciak, and E. I. Ralph. Feasibility study of a satellite solar power station. (1974).

[20] M. A. Minovitch, Orbiting Solar Power Station Inventor, United States, (1978).

[21] M. John and C. Mankins, SPS-ALPHA: The First Practical Solar Power Satellite via Arbitrarily Large Phased Array (A 2011-2012 NASA NIAC Phase 1 Project) FINAL REPORT to, (2012).

[22] NASA Technical Memorandum 58236 Satellite Power System Concept Development and Evaluation Program Volume I1 System Definition July 1981, National Aeronautics and Space Administration, (1981)

[23] Z. A. Cheng, X. Hou, X. Zhang, L. Zhou, J. Guo, and C. Song, In-orbit assembly mission for the Space Solar Power Station, Acta Astronaut., vol. 129, (Dec. 2016) pp. 299–308, doi: 10.1016/j.actaastro.2016.08.019.

[24] D. Goto, H. Yoshida, H. Suzuki, K. Kisara, K. Ohashi, & Y. Arimoto, The overview of JAXA laser energy transmission R&D activities and the orbital experiments concept on ISS-JEM. In International Conference on Space Optical Systems and Applications (ICSOS), (2014, May), pp. S5-2.

[25] A. Raj, K. Kalapala, K. B. Sivaraju, and M. Director, International First Place in International SunSat Design Competition-2014 at ISDC-2014.

[26] X. Meng, K. Du, X. Bai, J. C. Mankins, and C. Liu, Numerical investigation on improvement of energy transfer in solar power satellite, Renew. Energy, vol. 148, pp. 103–112, (Apr. 2020), doi: 10.1016/j.renene.2019.11.120.

[27] DOE US, NASA Reference System Report, Satellite Power System: Concept Development and Evaluation Program. (1978) DOE/ER-0023.

[28] M. Nagatomo, Development of the SPS 2000 System Concept and its Future. Institute of Space and Astronautical Science, (2001) p.1.x

[29] N. Kaya, Space Solar Power Satellite II. Japanese Space Solar Power Satellite NEDO Research and Research. Journal of the Institute of Electrical Engineers of Japan. D, Journal of Industrial Applications, 115 (8), (1995) pp.958-968.

[30] J. C. Mankins, A fresh look at space solar power: New architectures, concepts and technologies. Acta Astronautica, 41(4-10), (1997) pp.347-359.

[31] C.K. Carrington, H. Feingold, and J.T. Howell, Space solar power concepts: demonstrations to pilot plants (2002).

[32] W. Seboldt, M. Klimke, M. Leipold, and N. Hanowski, European sail tower SPS concept. Acta Astronautica, 48(5-12), (2001) pp.785-792.

[33] T. Fujita, H. Suzuki, H. Yoshida, T. Sezaki, K. Kisara, Y. Hisada, Y. Fukumuro, and M. Mori, Status of studies on space solar power systems (SSPS) in JAXA. Technical report of IEICE, (2008) pp.2008-04.

[34] S. Hahn, H. Ishii, T. Ozaki,K. Kiritani, T. Funyu, T. Okamoto, T., & I. Mikami, Power Beaming Technology Demonstration Satellite for Solarbird (R) Space Solar Power System. In 21st International Communications Satellite Systems Conference and Exhibit, (2003), pp. 2359.

[35] Y. Kobayashi, S. Mihara, T. Saito, H. Kanai, Recent activity of SSPS at USEF(FY2007), Technical report of the Institute of Electronics, Information and Communication Engineering (2008) pp.5-10.

[36] M. Mori, H. Kagawa, and Y. Saito, Summary of studies on space solar power systems of Japan Aerospace Exploration Agency (JAXA). Acta Astronautica, 59(1-5), (2006) pp.132-138.

[37] A. M. Rubenchik, J. M. Parker, R. J. Beach, & R. M. Yamamoto, Solar Power Beaming: From Space to Earth (No. LLNL-TR-412782). Lawrence Livermore National Lab. (LLNL), Livermore, CA, United States, (2009).

[38] J. C. Mankins, N. Kaya, and M. Vasile, SPS-ALPHA: the first practical solar power satellite via arbitrarily large phased array (a 2011-2012 NIAC project). In 10th International Energy Conversion Engineering Conference (2012) p. 3978.

[39] H. Xinbin, L. Meng, N. Lili, Z. Lu, C. Ying, C. Zhengai, & J. Haipeng, Multi-Rotary Joints SPS.

[40] K. Henson, Thermal power satellite design and mass calculation. In 2015 IEEE Conference on Technologies for Sustainability (SusTech) (2015, July) pp. 15-19.

[41] Y. Yang, Y. Zhang, B. Duan, D. Wang, and X. Li, A novel design project for space solar power station (SSPS-OMEGA). Acta Astronautica , 121, (2016) pp.51-58.

[42] I. Cash, CASSIOPeiA–A new paradigm for space solar power. Acta Astronautica, 159, (2019) pp.170-178.

[43] A. Baraskar, Y. Yoshimura, S. Nagasaki, and T. Hanada, Wirelessly Powering Satellites Using Laser Emitter, Proceedings of the Global Space Exploration Conference (GLEX 2021), St. Petersburg, Russia, June 14-18, (2021)

[44] T. Lind, M. Pellegrini, L. E. Herranz, M. Sonnenkalb, Y. Nishi, H. Tamaki, & T. Sevon, Overview and outcomes of the OECD/NEA benchmark study of the accident at the Fukushima Daiichi NPS (BSAF), Phase 2–Results of severe accident analyses for unit 3. Nuclear Engineering and Design, 376, 111138, (2021).

[45] H. Matsuoka, Global environmental issues and space solar power generation: promoting the SPS 2000 project in Japan. Technology in society, 21(1), (1999). pp 1-17.

[46] M. Suga, CO2 emission of SPS (4) total effect. Proceedings of Sixteenth ISAS Space Energy Symposium, (1997):126–30.

[47] M. Nagatomo, An approach to develop space solar power as a new energy system for developing countries. Solar Energy, 56(1), (1996). Pp. 111-118.

[48] N. Kaya, Solar Power Satellite in Japan -Study by NEDO Reported, Kobe University,

https://www.jstage.jst.go.jp/article/ieejias1987/115/8/115_8_958/_pdf (accessed 25 July, 2021)

[49] T. Nakamura, N. Sekiya, T. Yamagami, H. Mizoguchi, & K. Tanaka, Thermal Analysis of Hybrid Structure of Power Generation and Transmission Panel for SPS. Space Solar Power Systems, 6, (2021). Pp. 9-14.

[50] N. Shinohara, History and Innovation of Wireless Power Transfer via Microwaves. IEEE Journal of Microwaves, 1(1), 218-228, (2021) https://doi.org/10.1109/JMW.2020.3030896

[51] N. Lior, Power from space. Energy Conversion and Management, 42(15-17), (2001) pp. 1769-1805.

[52] Li, X., Duan, B., Song, L., Yang, Y., Zhang, Y., & Wang, D. A new concept of space solar power satellite. Acta Astronautica, 136, (2017) pp. 182-189.

[53] S. S. Gosavi, H. G. Mane, A. S. Pendhari, A. P. Magdum, S. Deshpande, A. Baraskar, M. Jadhav, and A. Husainy, A review on space based solar power," Journal of Thermal Energy Systems, vol. 6, no. 1, (2021). pp. 16-24, doi:10.46610/jotes.2021.v06i01.003. [Online]. Available: https://doi.org/10.46610%2Fjotes.2021. v06i01.003.

[54] C. T. Rodenbeck, T. Christopher, P. I. Jaffe, B. H. Strassner II, P. E. Hausgen, J. O. McSpadden, H. Kazemi, N. Shinohara, B. B. Tierney, C. B. DePuma, and A. P. Self. Microwave and Millimeter Wave Power Beaming. IEEE Journal of Microwaves 1, no. 1 (2021).

[55] K. Kumar, & M. Bhattacharya, Modelling and Efficiency Analysis of Microwave Wireless Power Transfer System. In 2019 Innovations in Power and Advanced Computing Technologies (i-PACT) (2019, March), Vol. 1, pp. 1-7.

[56] W. Zhou, and K. Jin. Power control method for improving efficiency of laser-based wireless power transmission system. IET Power Electronics, 13(10), 2096-2105, (2020) https://doi.org/10.1049/ietpel.2019.1372

[57] T. N. Le, (2009). Conceptual design of a solar power beaming space system. (accessed 25 July, 2021)

[58] D. Goto, H. Yoshida, H. Suzuki, K. Kisara, K. Ohashi, & Y. Arimoto, The overview of JAXA laser energy transmission R&D activities and the orbital experiments concept on ISS-JEM. In International Conference on Space Optical Systems and Applications (ICSOS), (2014, May), pp. S5-2.

[59] K. Jin, and W. Zhou., Wireless Laser Power Transmission: A Review of Recent Progress, IEEE Trans. Power Electron., Vol. 34, No. 4, (2019), pp. 3842–3859, Doi: 10.1109/TPEL.2018.2853156.

[60] R. G. Madonna, Use of an Iterative Research and Development – System Engineering Approach for the Caltech Space Solar Power Project, 6th IEEE International Conference on Wireless for Space and Extreme Environments (WiSEE), Huntsville, AL, USA, (2018) pp. 200-205.

[61] M. Arya, N. Lee, and S. P;eegrino, Ultralight Structures for Space Solar Power Satellites, in AIAA Science and Technology Conference, San Diego, CA, (2016).

[62] J. Umali, L. Wilson, and S. Pellegrino, Vibration response of ultralight coilable spacecraft structures, in AIAA Science and Technology Conference, Grapevine, Texas USA, (2017).

[63] H. T. Friis, A note on a simple transmission formula, Proceedings of the IRE, vol. 34, no. 5, (1946) pp. 254–256.

[64] C. Balanis, Antenna Theory - Analysis and Design, 3rd Edition, Hoboken, NJ, Wiley-Interscience, (2005).

[65] X. Li, J. Zhou, B. Duan, Y. Yang, Y. Zhang, & J. Fan, Performance of planar arrays for microwave power transmission with position errors. IEEE Antennas and Wireless Propagation Letters, 14, (2015), pp. 1794-1797.

[66] E. Gdoutos, C. Leclerc, F. Royer, M. D. Kelzenberg, E. C. Warmann, P. Espinet-Gonzalez, & S. Pellegrino, A lightweight tile structure integrating photovoltaic conversion and RF power transfer for space solar power applications. In 2018 AIAA Spacecraft Structures Conference (2018). pp. 2202.

[67] S. Mizojiri, & K. Shimamura, Wireless power transfer via Subterahertz-wave. Applied Sciences, 8(12), (2018), pp. 2653.

[68] D. E. Raible. High intensity laser power beaming for wireless power transmission, Master’s Thesis, Department of Electrical and Computer Engineering, Cleveland State University, Cleveland, OH USA, (May, 2008).

[69] http://www.al.t.u-tokyo.ac.jp/lecture/Chap4(LaserTransmission).pdf (accessed 28. 09. 2020)

[70] F. Steinsiek et al., Wireless Power Transmission Experiment as an Early Contribution to Planetary Exploration Missions, in Proc. International Astronautical Congress, (2003) pp. 169-176.

[71] K. J. Duncan, Laser based power transmission: component selection and laser hazard analysis, in Proc. PELS Workshop on Emerging Technologies: Wireless Power transfer, (2016) pp. 100-103.

[72] R. Mason, Feasibility of Laser power transmission to a high-altitude unmanned aerial vehicle, Rand Corporation, (2011)

[73] W.F. Krupke, R. J. Beach, S. A. Payne, V. K. Kanz, and J. T. Early, DPAL: A New Class of Lasers for CW Power Beaming at Ideal Photovoltaic Cell Wavelengths, in 2nd International Symposium on Beamed Energy Propulsion, Melville, N.Y., UCRL-CONF-155610, (2003).

[74] A. Kobayashi and Y. M. Day, Consider the Situation and Future Prospects in The ITU Related to Wireless Power Transmission Frequency Management and Work Planning Commission Chief the Times Space Solar Power, (2016).

[75] R. R. King Et Al., Solar Cell Generations Over 40% Efficiency, In Progress In Photovoltaics: Research And Applications, Vol. 20, No. 6, (2012), pp. 801–815, Doi: 10.1002/Pip.1255.

[76] C. Bergsrud, and S. Jeremy. A Space-To-Space Microwave Wireless Power Transmission Experiential Mission Using Small Satellites, Acta Astronaut., Vol. 103, (2012), pp. 193–203, Doi: 10.1016/J.Actaastro.2014.06.033.

[77] C. Emanuel, U.S. Naval Research Laboratory, (accessed 20 Aug 2020), https://www.nrl. navy.mil /news /releases /researchers- transmit- energy- laser-power -beaming-demonstration (2019).

[78] K. Walter. Solid-State Laser Engineering, 3rd ed., Springer-Verlag. ISBN 0-387-53756-2 (1992).

[79] G. Christian. Analysis of Wireless Transmission of Power for Physical Security Use. No. SAND2019-12233. Sandia National Lab.(SNL-NM), Albuquerque, NM (United States),,doi:10.2172/1570551.(2019).

[80] Scientific Library, Laser, http://www. scientificlib. com/ en/Physics /Laser /Laser.html (accessed 25 July, 2021)

[81] J. Kochiyama, N. Kaya, T. Fujiwara, H. Yasui, & H. Yashiro, (1997). U.S. Patent No. 5,666,127. Washington, DC: U.S. Patent and Trademark Office.

[82] R. M. T. Dickinson, (2002). Wireless Power Transmission Technology State-Of-The-Art. IAF abstracts, 34th COSPAR Scientific Assembly, 741.

[83] L. Summerer, & O. Purcell, (2009). Concepts for wireless energy transmission via laser. Europeans Space Agency (ESA)-Advanced Concepts Team.

[84] Union of Concerned Scientists, UCS Satellite Database, https:// www. ucsusa. org/ resources/satellite-database, 2005 [accessed 14 February 2021].

[85] Satellite Industry Association, State of the Satellite Industry Report, 2019 Top-Level Global Satellite Industry Findings, (accessed 25 July, 2021)

[86] M. Taherbaneh, H. Ghafooifard, A. H. Rezaie, & K. Rahimi, (2011). Evaluation end-of-life power generation of a satellite solar array. Energy conversion and management, 52(7), 2518-2525.

[87] A. Baraskar, H. Chen, Y. Yoshimura, S. Nagasaki, and T. Hanada, Verify the Wireless Power Transmission in Space using Satellite to Satellite System. International Journal of Emerging Technologies, 12(2): (2021), pp. 110–118.

[88] T. Green, I. M. Opio, C. Rolf, and R. Andrew. Energy Calculator for Solar Processing of Biomass with Application to Uganda. Energies 13, no. 6 (2020), pp 1485.

[89] H. Zheng, Chapter 2—Solar Energy Utilization and Its Collection Devices. Solar Energy Desalination Technology. (2017 Jan) 1: pp 47-171.

[90] N. Shinohara, ed. Recent wireless power transfer technologies via radio waves. River Publishers, (2018).

[91] Northrop Grumman, Diode Bars – 100W CW https://www.photonicsolutions. co.uk /upfiles /LaserDiodeBar100WCWDatasheetLG29Jan18.pdf (accessed 25 July, 2021)

[92] S. Library. Laser, [Online]. Available: http://www.scientificlib.com/en/Physics/ Laser / Laser.html (accessed 25 July, 2021)

[93] P. Crump, J. Wang, S. Patterson, D. Wise, A. Basauri, M. DeFranza, S. Elim, W. Dong, S. Zhang, M. Bougher, J. Patterson, S. Das, M. Grimshaw, J. Farmer, M. DeVito, and R. Martinsen, Diode laser efficiency increases enable >400-W peak power from 1-cm bars and show a clear path to peak powers in excess of 1-kW, Proc. SPIE 6104, High-Power Diode Laser Technology and Applications IV, 610409 (accessed 25 July, 2021), https://doi.org/10.1117/12.649552

[94] F. Dorsch, F. X Daiminger. 2kW cwfiber coupled diode laser system. SPIE. Vol. 3889. 2000.

[95] K. K. Chen, S. Alam, J. R. Hayes, D. Lin, A. Malinowski and D. J. Richardson, 100W, single mode, single polarization, picosecond, ytterbium doped fibre MOPA frequency doubled to 530 nm, 2009 High Power Diode Lasers and Systems Conference, (2009), pp. 1-2, doi: 10.1109/HPD.2009.5366532

[96] Coherent Inc., Lasers: Understanding the Basics, https://www.photonics.com/ Articles/Lasers_Understanding_the_Basics/a25161 (accessed 25 July, 2021)

[97] B. Wie, V. Lappas, & J. Gil-Fernández, (2014). Attitude and orbit control systems. In the International Handbook of Space Technology (pp. 323-369). Springer, Berlin, Heidelberg.

[98] Digital Sun Sensor, Adcole Space, https://adcolespace.com/product/digital-sun-sensor-%C2%B164/ (accessed 25 July, 2021)

[99] INFRARED EARTH SENSOR Leonardo S.p.a., https://www. leonardocompany. com/documents/201423150107/IRES_NE_Attitude_Control_Sensors_LQ_mm077 87 _.pdf?t=1538987566236, (accessed 25 July, 2021)

[100] H. H. Nguyen, & P. S. Nguyen, (2020). Communication Subsystems for Satellite Design. In Satellite Systems-Design, Modeling, Simulation and Analysis. IntechOpen.

[101] M. Toyoshima, K. Toshiaki, W. Klaus, T. Masahiro, T. Hideki, S. Yozo, T. Yoshihisa et al. "Overview of the laser communication system for the NICT optical ground station and laser communication experiments on ground-to-satellite links." Journal of the National Institute of Information and Communications Technology 59, no. 1.2 (2012): pp. 053-075

[102] H. Klinkrad, (2010). Space debris. Encyclopedia of Aerospace Engineering

[103] CelesTrak, Satellite Orbital Conjunction Reports Assessing Threatening Encounters in Space (SOCRATES), http://celestrak. com/ SOCRATES. (accessed 25 July, 2021)

[104] Y. Itaya, K. Fujita, and T. Hanada, Precise time estimation of on-orbit satellite fragmentations. Acta Astronautica, 152, (2018), pp.415-425.

[105] K. Uramoto, & T. Hanada, Development of orbital debris environment model in the Low Earth Orbit. Advances in the Astronautical Sciences, 117, (2004), pp. 927-938.

[106] S. Furuta, T. Hanada, K. Fujita, & K. Takezono, Discussion on the necessity of orbital debris removal in the geostationary region. In 65th International Astronautical Congress 2014, IAC 2014: Our World Needs Space, Vol. 2, (2014). pp. 1239-1246. International Astronautical Federation, IAF

[107] S. Kawamoto, N. Nagaoka, T. Hanada, & S. Abe, Evaluation of active debris removal strategy using a debris evolutionary model. Proceedings of the International Astronautical Congress, IAC, 2019-October, (2019) [IAC-19_A6_2_10_x53577]

[108] S. Dinkin, A lunar vision at $2,000/kg, December 6, 2004, https://thespacereview.com/article/284/1 (accessed 25 July, 2021)

[109] J. E. Werner, S. G. Johnson, C. C. Dwight, and K. L. Lively, Cost Comparison in 2015 Dollars for Radioisotope Power Systems--Cassini and Mars Science Laboratory (No. INL/EXT-16-40218). Idaho National Lab. (INL), Idaho Falls, ID USA (2016)

[110] C. M. Pieters, J. Boardman, B. Buratti, A. Chatterjee, R. Clark, T. Glavich, R. Green et al. The Moon Mineralogy Mapper (M^3) on Chandrayaan-1. Current Science : (2009), pp 500-505

[111] A. Baraskar, Moon Habitat Power station and Communication equipment. In Гагаринские чтения-2018, (2018), pp. 341-343.

[112] D. A. Kring, D. Daniel, A. Durda. Global Lunar Landing Site Study to Provide the Scientific Context for Exploration of the Moon, LPI-JSC Center for Lunar Science and Exploration. (2012)

[113] H. Hiroaki, and T. Hanada. Using a Small Satellite to Verify the Wireless Power Transmission in Space. In The Twenty-ninth International Symposium on Space Technology and Science. (2013).

[114] NASA, Is There an Atmosphere on the Moon?, accessed 22 Aug 2020 https: // www.nasa.gov /mission_pages /LADEE /news /lunar-atmosphere.html (2013).

[115] C. Pieters, J. N. Goswami, R. N. Clark, M. Annadurai, J. Boardman, B. Buratti, J-P. Combe et al. Character and spatial distribution of OH/H2O on the surface of the Moon seen by M3 on Chandrayaan-1. science 326, no. 5952 (2009): pp. 568-572.

[116] I. Mitrofanov, V. Boynton, G. Chin, J. B. Garvin, D. Golovin, L. G. Evans et al. Hydrogen mapping of the lunar south pole using the LRO neutron detector experiment LEND. science 330, no. 6003 (2010) pp. 483-486.

[117] J. R. Johnson, D. S. Timothy, and P. G. Lucey. Estimated solar wind-implanted helium-3 distribution on the Moon, Geophysical Research Letters 26, no. 3 (1999) pp 385-388.

[118] D.A. Paige, M.A. Siegler, J.A. Zhang, P.O. Hayne, E.J. Foote, K.A. Bennett, A.R Vasavada, B.T. Greenhagen, J.T. Schofield, D.J. McCleese, and M.C. Foote, Diviner lunar radiometer observations of cold traps in the Moon's south polar region. Science, 330(6003), (2010) pp.479-482.

[119] P. Beauchamp, R. Ewell, E. Brandon, and R. Surampudi. Solar power and energy storage for planetary missions. Jet Propulsion Laboratory and California Institute of Technology (2015).

[120] M. R. Patel, Spacecraft power systems. CRC press, p24 (2004).

[121] I. Minoru, H. Kusawake, S. Shimada, A. Ishii, Y. Kikuchi, K. Aoki, J. Shimizu, and T. Ito. Lightweight Power Control Units and Power Distribution Control Unit for Satellites. The Journal of Space Technology and Science 28, no. 1 (2013).

[122] W. W. Cobb, How SpaceX lowered costs and reduced barriers to space. The Conversation (2019). (accessed 25 July, 2021)

Appendix A

A.1. For Parabolic Reflector Antenna Gain

The given formula is for calculating the gain of a parabolic dish antenna:

$$G = 10 \times \log_{10} \eta \left(\frac{\pi D}{\lambda}\right)^2$$

For frequency 2.45 GHz

Diameter (m)	Efficiency (%)	Frequency (GHz)	Gain (dB)	Beam width (degree)
1	100	2.45	28.18	8.57142857
1	90	2.45	27.72	8.57142857
1	80	2.45	27.21	8.57142857
1	70	2.45	26.64	8.57142857
1	60	2.45	25.96	8.57142857
1	50	2.45	25.17	8.57142857
1	40	2.45	24.20	8.57142857
1	30	2.45	22.95	8.57142857
1	20	2.45	21.19	8.57142857
1	10	2.45	18.18	8.57142857
1	1	2.45	8.18	8.57142857
2	100	2.45	34.20	4.28571429
2	90	2.45	33.74	4.28571429
2	80	2.45	33.23	4.28571429
2	70	2.45	32.65	4.28571429
2	60	2.45	31.98	4.28571429
2	50	2.45	31.19	4.28571429
2	40	2.45	30.22	4.28571429
2	30	2.45	28.97	4.28571429
2	20	2.45	27.21	4.28571429
2	10	2.45	24.20	4.28571429
2	1	2.45	14.20	4.28571429
10	100	2.45	48.18	0.85714286
10	90	2.45	47.72	0.85714286
10	80	2.45	47.21	0.85714286
10	70	2.45	46.63	0.85714286
10	60	2.45	45.96	0.85714286
10	50	2.45	45.17	0.85714286
10	40	2.45	44.20	0.85714286
10	30	2.45	42.95	0.85714286
10	20	2.45	41.19	0.85714286
10	10	2.45	38.18	0.85714286
10	1	2.45	28.18	0.85714286
100	100	2.45	68.18	0.09

100	90	2.45	67.72	0.09
100	80	2.45	67.21	0.09
100	70	2.45	66.63	0.09
100	60	2.45	65.96	0.09
100	50	2.45	65.17	0.09
100	40	2.45	64.20	0.09
100	30	2.45	62.95	0.09
100	20	2.45	61.19	0.09
100	10	2.45	58.18	0.09
100	1	2.45	48.18	0.09
1000	100	2.45	88.18	0.009
1000	90	2.45	87.72	0.009
1000	80	2.45	87.21	0.009
1000	70	2.45	86.63	0.009
1000	60	2.45	85.96	0.009
1000	50	2.45	85.17	0.009
1000	40	2.45	84.20	0.009
1000	30	2.45	82.95	0.009
1000	20	2.45	81.19	0.009
1000	10	2.45	78.18	0.009
1000	1	2.45	68.18	0.009

For frequency 5.8 GHz:

Diameter (m)	Efficiency (%)	Frequency (GHz)	Gain (dB)	Beam width (degree)
1	100	5.8	35.66	3.62
1	90	5.8	35.21	3.62
1	80	5.8	34.70	3.62
1	70	5.8	34.11	3.62
1	60	5.8	33.45	3.62
1	50	5.8	32.65	3.62
1	40	5.8	31.69	3.62
1	30	5.8	30.44	3.62
1	20	5.8	28.68	3.62
1	10	5.8	25.66	3.62
1	1	5.8	15.66	3.62
2	100	5.8	41.69	1.81
2	90	5.8	41.23	1.81
2	80	5.8	40.72	1.81
2	70	5.8	40.14	1.81
2	60	5.8	39.47	1.81
2	50	5.8	38.68	1.81
2	40	5.8	37.71	1.81
2	30	5.8	36.46	1.81
2	20	5.8	34.70	1.81
2	10	5.8	31.69	1.81
2	1	5.8	21.69	1.81
10	100	5.8	55.66	0.36

10	90	5.8	55.21	0.36
10	80	5.8	54.70	0.36
10	70	5.8	54.12	0.36
10	60	5.8	53.45	0.36
10	50	5.8	52.65	0.36
10	40	5.8	51.69	0.36
10	30	5.8	50.44	0.36
10	20	5.8	48.68	0.36
10	10	5.8	45.66	0.36
10	1	5.8	35.66	0.36
100	100	5.8	75.66	0.04
100	90	5.8	75.21	0.04
100	80	5.8	74.70	0.04
100	70	5.8	74.12	0.04
100	60	5.8	73.45	0.04
100	50	5.8	72.65	0.04
100	40	5.8	71.69	0.04
100	30	5.8	70.44	0.04
100	20	5.8	68.68	0.04
100	10	5.8	65.66	0.04
100	1	5.8	55.66	0.04
1000	100	5.8	95.66	0.004
1000	90	5.8	95.21	0.004
1000	80	5.8	94.70	0.004
1000	70	5.8	94.12	0.004
1000	60	5.8	93.45	0.004
1000	50	5.8	92.65	0.004
1000	40	5.8	91.69	0.004
1000	30	5.8	90.44	0.004
1000	20	5.8	88.68	0.004
1000	10	5.8	85.66	0.004
1000	1	5.8	75.66	0.004

For frequency 100 GHz

Diameter (m)	Efficiency (%)	Frequency (GHz)	Gain (dB)	Beam width (degree)
1	100	100	60.40	0.21
1	90	100	59.94	0.21
1	80	100	59.43	0.21
1	70	100	58.85	0.21
1	60	100	58.18	0.21
1	50	100	57.39	0.21
1	40	100	56.42	0.21
1	30	100	55.17	0.21
1	20	100	53.41	0.21
1	10	100	50.40	0.21
1	1	100	40.40	0.21
2	100	100	66.42	0.11

2	90	100	65.96	0.11
2	80	100	65.45	0.11
2	70	100	64.87	0.11
2	60	100	64.20	0.11
2	50	100	63.41	0.11
2	40	100	62.44	0.11
2	30	100	61.19	0.11
2	20	100	59.43	0.11
2	10	100	56.42	0.11
2	1	100	46.41	0.11
10	100	100	80.40	0.02
10	90	100	79.94	0.02
10	80	100	79.43	0.02
10	70	100	78.85	0.02
10	60	100	78.18	0.02
10	50	100	77.39	0.02
10	40	100	76.42	0.02
10	30	100	75.17	0.02
10	20	100	73.41	0.02
10	10	100	70.40	0.02
10	1	100	60.40	0.02
100	100	100	100.40	0.002
100	90	100	99.94	0.002
100	80	100	99.43	0.002
100	70	100	98.85	0.002
100	60	100	98.18	0.002
100	50	100	97.39	0.002
100	40	100	96.42	0.002
100	30	100	95.17	0.002
100	20	100	93.41	0.002
100	10	100	90.40	0.002
100	1	100	80.40	0.002
1000	100	100	120.40	0.0002
1000	90	100	119.94	0.0002
1000	80	100	119.43	0.0002
1000	70	100	118.85	0.0002
1000	60	100	118.18	0.0002
1000	50	100	117.39	0.0002
1000	40	100	116.42	0.0002
1000	30	100	115.17	0.0002
1000	20	100	113.41	0.0002
1000	10	100	110.40	0.0002
1000	1	100	100.40	0.0002

For frequency 281,760 GHz

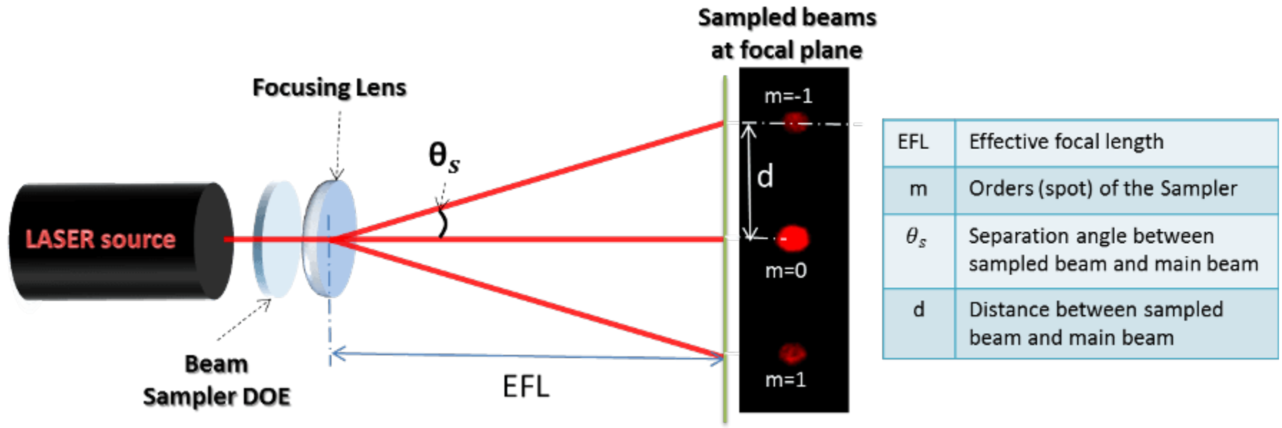
Diameter (m)	Efficiency (%)	Frequency (GHz)	Gain (dB)	Beam width (degree)
1	100	281760	129.40	0.000075
1	90	281760	128.94	0.000075
1	80	281760	128.42	0.000075
1	70	281760	127.84	0.000075
1	60	281760	127.18	0.000075
1	50	281760	126.38	0.000075
1	40	281760	125.41	0.000075
1	30	281760	124.16	0.000075
1	20	281760	122.40	0.000075
1	10	281760	119.39	0.000075
1	1	281760	109.39	0.000075
2	100	281760	135.41	0.000037
2	90	281760	134.96	0.000037
2	80	281760	134.45	0.000037
2	70	281760	133.87	0.000037
2	60	281760	133.20	0.000037
2	50	281760	132.40	0.000037
2	40	281760	131.43	0.000037
2	30	281760	130.19	0.000037
2	20	281760	128.42	0.000037
2	10	281760	125.41	0.000037
2	1	281760	115.41	0.000037
10	100	281760	149.39	0.0000075
10	90	281760	148.94	0.0000075
10	80	281760	148.42	0.0000075
10	70	281760	147.84	0.0000075
10	60	281760	147.18	0.0000075
10	50	281760	146.38	0.0000075
10	40	281760	145.41	0.0000075
10	30	281760	144.16	0.0000075
10	20	281760	142.40	0.0000075
10	10	281760	139.39	0.0000075
10	1	281760	129.39	0.0000075
100	100	281760	169.39	0.00000075
100	90	281760	168.94	0.00000075
100	80	281760	168.42	0.00000075
100	70	281760	167.84	0.00000075
100	60	281760	167.18	0.00000075
100	50	281760	166.38	0.00000075
100	40	281760	165.41	0.00000075
100	30	281760	164.16	0.00000075
100	20	281760	162.40	0.00000075
100	10	281760	159.39	0.00000075
100	1	281760	149.39	0.00000075
1000	100	281760	189.39	0.00000007
1000	90	281760	188.94	0.00000007

1000	80	281760	188.42	0.00000007
1000	70	281760	187.84	0.00000007
1000	60	281760	187.18	0.00000007
1000	50	281760	186.38	0.00000007
1000	40	281760	185.41	0.00000007
1000	30	281760	184.16	0.00000007
1000	20	281760	182.40	0.00000007
1000	10	281760	179.39	0.00000007
1000	1	281760	169.39	0.00000007

A. 2. Antenna Gain with different frequency

Antenna gain (dBi)	*Frequency (GHz)*									
	2.45	5.8	10	25	50	100	1,000	100,000	281,760 (1060 nm)	400,000
1	0.002	0.0002	0.00009	0.00001	0.000004	0.0000001	0	0	0	0
10	0.012	0.002	0.0007	0.000114	0.00003	0.000007	0	0	0	0
20	0.119	0.023	0.0071	0.001144	0.000286	0.000072	0.000001	0	0	0
30	1.192	0.213	0.0715	0.011443	0.002861	0.000715	0.000007	0	0	0
40	11.915	2.126	0.715	0.114433	0.028608	0.007152	0.000072	0	0	0
50	119.151	21.261	7.152	1.144331	0.286083	0.071521	0.000715	0	0	0
60	1191.515	212.606	71.521	11.44330	2.860827	0.715207	0.007152	0.000001	0	0
70	11915	2126.06	715.206	114.4330	28.60826	7.152066	0.071521	0.000007	0.000001	0
80	119151	21260.60	7152.066	1144.330	286.0826	71.52066	0.715207	0.000072	0.000009	0.000004
90	1191514	212606.0	71520.66	11443.30	2860.826	715.2066	7.152066	0.000715	0.00009	0.000045
100	11915146	2126060	715206.6	114433.0	28608.26	7152.066	71.520665	0.007152	0.000901	0.000447

A.3. Beam Sampler Calculation and Gaussian beam calculation using LPT aperture for 10 kW with aperture diameter of 10 mm.



LD (W) Po/p	λ (nm)	Efficiency (%)	Effective Focal Length (m)	Distance (m)	Minimum input beam diameter [um]	Separation angle θs between sampled beam and zero order (deg)	Separation distance (d) between sampled beam and zero order (m)	Full distance between the two sampled beams: (m)	Power through Aperture (W)	% Power through Aperture (%)
100	975	77	1	1	4	45	1000	2000	10000	100
			100		293	0.5729	1000	2000	10000	100
			10000		29250	0.0057	1000	2000	2084.5	20.8
			500000		1462500	0.0001	1000	2000	0.9350	0
			1	10	3	84.2894	10000	20000	10000	100
			100		29	5.7106	10000	20000	10000	100
			10000		2925	0.0573	10000	20000	10000	100
			500000		146250	0.0011	10000	20000	93.0699	0.9
			1	100	3	89.4271	100000	200000	10000	100
			100		4	45	100000	200000	10000	100
			10000		293	0.5729	100000	200000	10000	100

			500000		14625	0.0115	100000	200000	6074.3665	60.7
			1	10000	3	89.9943	9999999	19999999	10000	100
			100		3	89.4271	10000000	20000000	10000	100
			10000		4	45	10000000	20000000	10000	100
			500000		146	1.1458	10000000	20000000	10000	100
			1	500000	3	89.9999	500005530	1000011060	10000	100
			100		3	89.9	499999999	999999999	10000	100
			10000		3	88.85	500000000	1000000000	10000	100
			500000		4 `	45	500000000	1000000000	10000	100
200 [90]	915	48	1	1	4	45	1000	2000	10000	100
			100		275	0.5729	1000	2000	10000	100
			10000		27450	0.0057	1000	2000	2331.2167	23.3
			500000		1372500	0.0001	1000	2000	1.0617	0
			1	10	3	84.2894	10000	20000	10000	100
			100		28	5.7106	10000	20000	10000	100
			10000		2745	0.0573	10000	20000	10000	100
			500000		137250	0.0011	10000	20000	105.6092	1.1
			1	100	3	89.4271	100000	200000	10000	100
			100		4	45	100000	200000	10000	100
			10000		275	0.5729	100000	200000	10000	100

			500000		13725	0.0115	100000	200000	6541.3560	65.4
			1	10000	3	89.9943	9999999.9	19999999.9	10000	100
			100		3	89.4271	10000000	20000000	10000	100
			10000		4	45	10000000	20000000	10000	100
			500000		137	1.1458	10000000	20000000	10000	100
			1	500000	3	89.99	500005530	100011060	10000	100
			100		3	89.9885	500000000	1000000000	10000	100
			10000		3	88.8542	500000000	1000000000	10000	100
			500000		4	45	500000000	1000000000	10000	100

Appendix B

B.1. The Cost details of Space launching system, debris cleaning charges by various company and cost of E-Orbit with the working on debris removal, power transportation and orbital maneuver. The Table B.1 is the economical revenue generate from the E-orbit system for power transmission and other for one year and 20 years.

Table B.1 is the economical revenue generate from the E-orbit system

Services	*Per unit*	*Per year (Mn USD)*	*Up to EOL (Bn USD/ 20 years)*
Energy Transfer (for transferring 1 kWh)	18 USD/ kWh	504	10.09
Orbit transfer (200 Satellite a year)	1 Mn USD	200	4
Orbit Keeping (200 Satellite a year)	1 Mn USD	200	4
De-Orbiting at EOL (200 Satellite a year)	1 Mn USD	200	4
Debris removal using ridesharing	1 Mn USD		1.6

The Figure B.1 is the compering debris removal technology to the E-Orbit cost, as the most of companies charge from 6 Mn USD to 117 Mn USD while E-Orbit Sat charges 1 Mn USD.

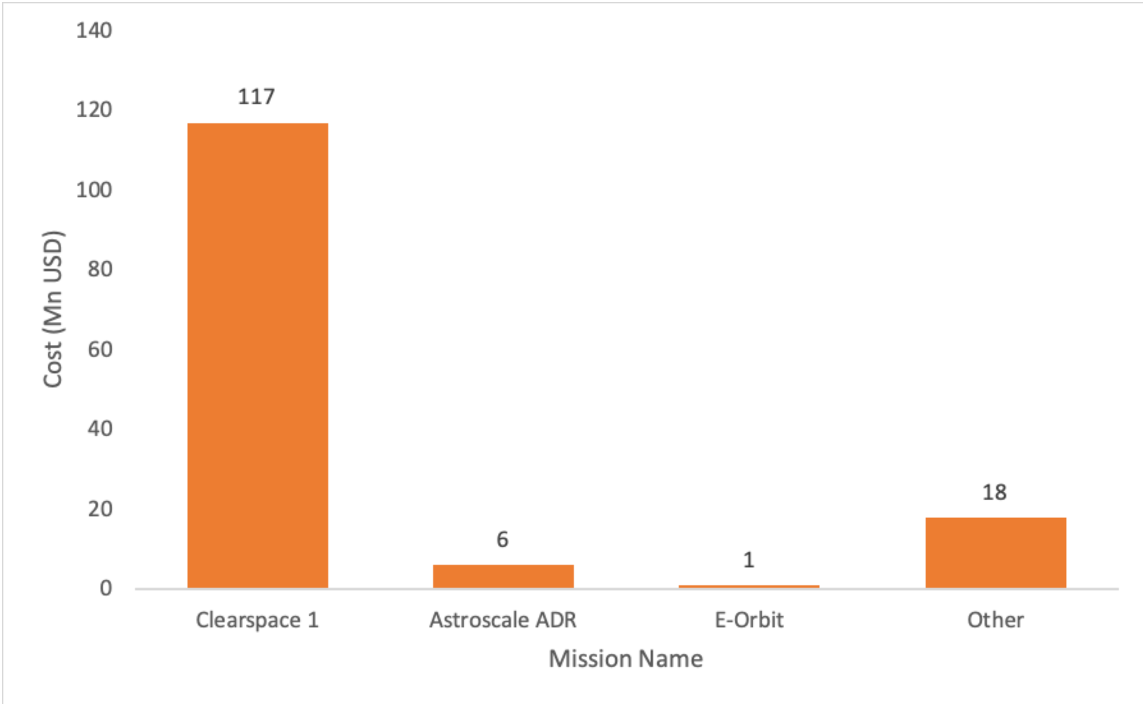


Figure B.1 the cost of debris removal technology in Mn USD

Table A.2. Launching of one kg in LEO

Launch Vehicle	*Payload cost per kg (USD)*
Vanguard	1,000,000 [1]
Space Shuttle	54,500 [1]
Electron	19,039 [2, 3]
Terran 1	9,600
Ariane 5G	9,167 [1]
Long March 3B	4,412 [1]
Proton	4,320 [1]
Falcon 9	2,720 [5]
Falcon Heavy	1,400 [5]

The Table A.2 is about the cost of launching one kg in orbit with the capable launch vehicle around the worldwide. And Table A.3 is launch vehicle details for the appropriate launching vehicle for E-Orbit with payload capacity with cost of launching.

Table B.3. Launch vehicle details, payload and cost of launching

Launcher [6]	*Payload (kg)*		*Cost (Mn USD)*
	LEO	GTO	
Starship	150,000	40,000	2 expected
Falcon Heavy	63,800	26,700	90–150
Long March 5	25,000	14,000	150
Delta IV	23,040	13,130	215
New Glenn	45,000	13,000	
Vulcan	17,800–34,900	7,400–16,300	99
Ariane 6	21,650	11,500	115
Ariane 5	21,000	10,735	165-220
Atlas V	18,850	8,900	109-153
Falcon 9	22,800	8,300	61.2
H-II, IIA & IIB	19,000	8,000	(190), 90, 112
H3	4,000-28,300 (base-heavy)	7,900-14,800	
Proton (UR-500)	23,000	6,920	65
Zenit	13,740	6,160	45-90
Long March 7	20,000	5,500-7,000	87.45
Long March 2-3-4	12,000	5,500	20-102
GSLV Mk.III (LVM3)	10,000	4,000	51
Angara A5	14,600–35,000	3,600–12,500	90-105
GSLV Mark II	5,000	2,700	47

Soyuz	8,200	2,400	35-48.5
ULV	4,500–41,300	1,500–16,300	
PSLV	3,800	1,200	18-28
Cyclone-4M	5,000	1,000	45
Minotaur IV & V	1,735	640	50

Reference:

[1] Space Transportation Costs: Trends in Price Per Pound to Orbit. yumpu.com. Futron Corporation. (Accessed on July 30, 2021).

[2] Rocket Lab points out that not all rideshare rocket launches are created equal. TechCrunch. (Accessed on July 30, 2021)

[3] rocklabusa.com (PDF). Rocket Lab https://www.rocketlabusa.com/ assets/Uploads/ Payload-User- Guide-LAUNCH-V6.6.pdf (Accessed on July 30, 2021)

[4] Terran 1. relativityspace.com. Relativity Space. (Accessed on July 30, 2021)

[5] NASA Technical Reports Server (NTRS). ntrs.nasa.gov. NASA. (Accessed on July 30, 2021).

[6] Comparison of Orbital launcher family https://en.wikipedia. org/wiki/ Comparison _of_orbital_launcher_families (Accessed on July 30, 2021)

Appendix C

C.1. The Drag profile on different SSPS design.

1. **CaSSIOPeiA Geostationary 4-sun SSSC**
   a. Radius (Hight from center of earth) = 42157.0 km
   b. MASS = 2045000.0 kg
   c. SOLAR_RAD_AREA = 2941225.0 m^2

The CaSSIOPeiA is well designed with knowledge of cosine system to maintain in GEO. The 4 Sun SSSC will deliver 1.99 GW of energy from space to earth with mass of 2045 tons. Via 2.45 GHz based MPT system. the rectenna is 68 km^2. The total cost to design and place will be approximately 7,754 Mn USD.

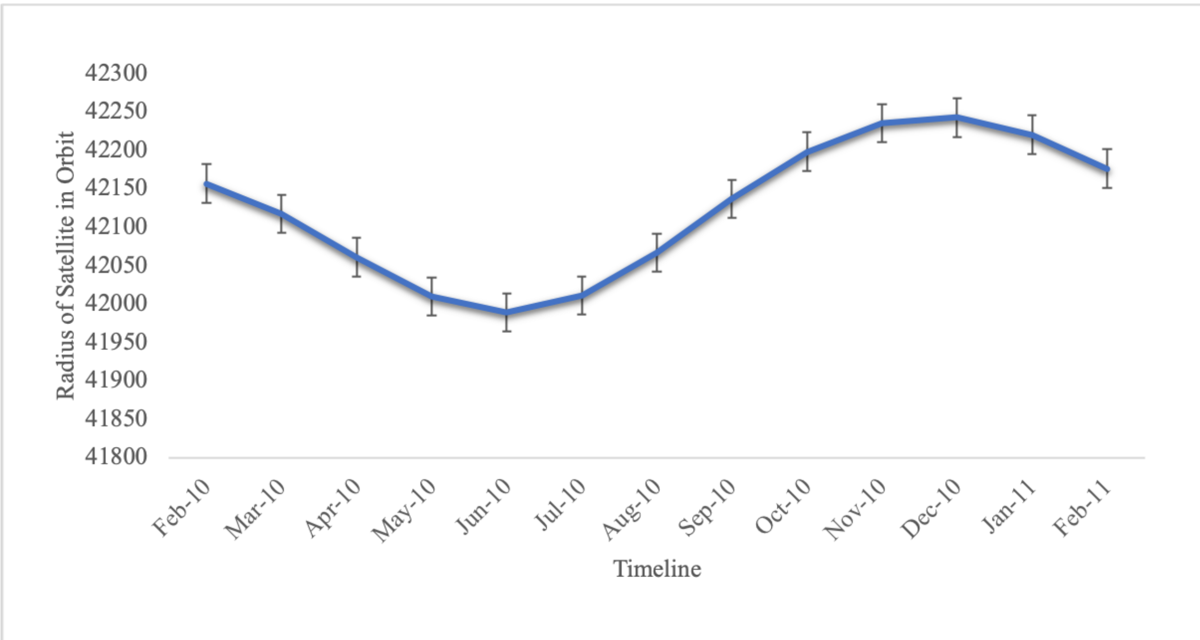


Figure B.2. the overall perturbation on the CaSSIOPeiA

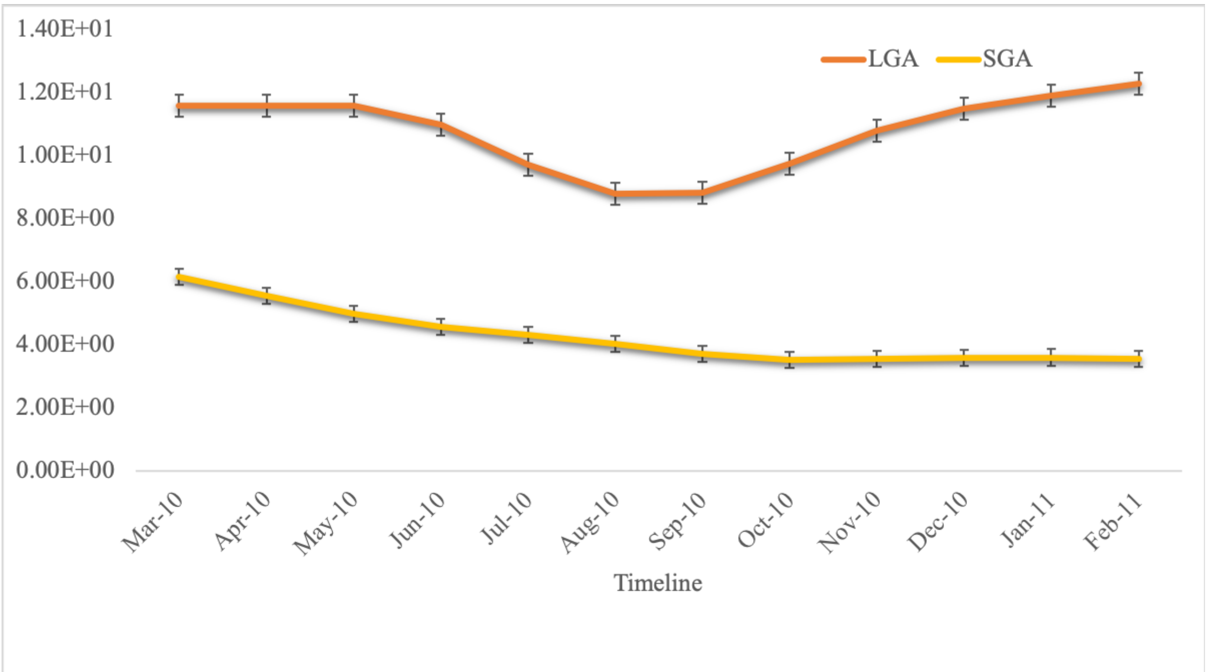


Figure B.2. the overall LGA and SGA on the CaSSIOPeiA

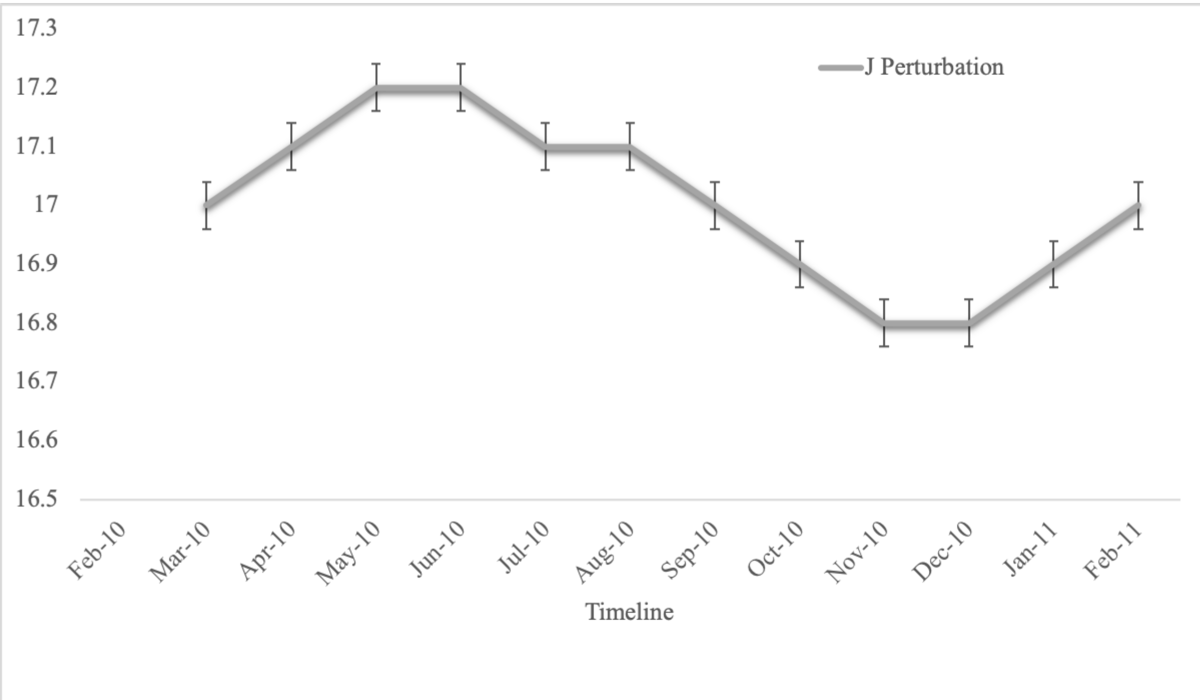


Figure B.3. the J perturbation on the CaSSIOPeiA

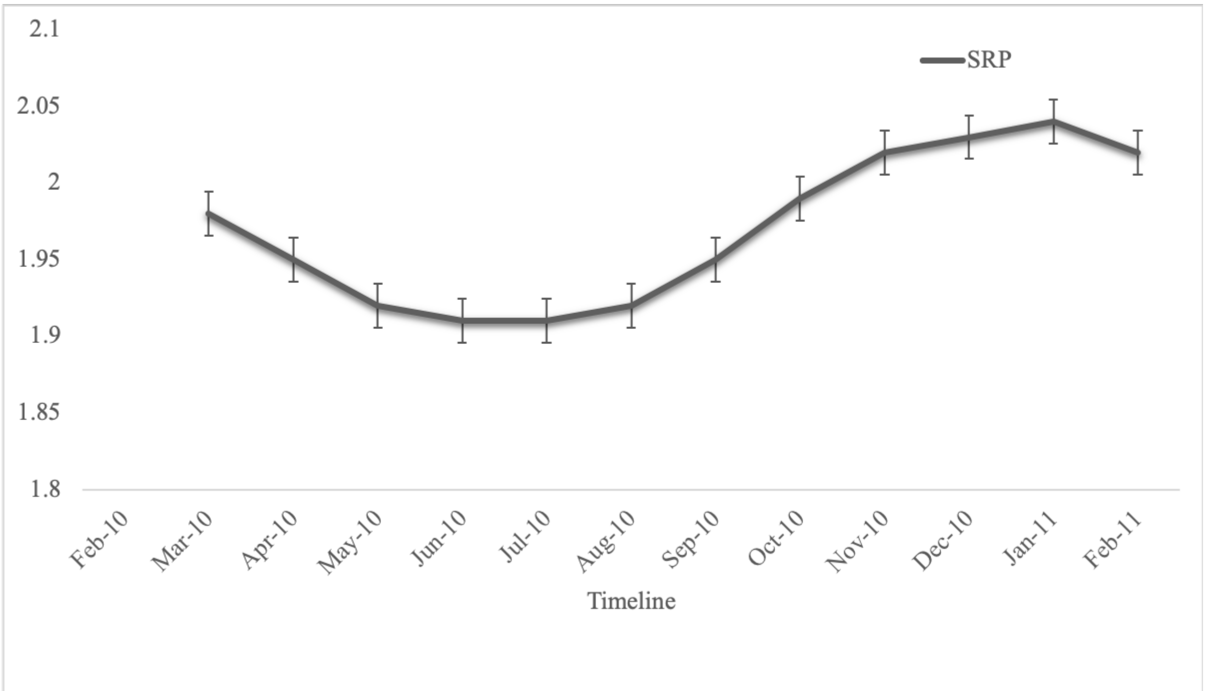


Figure B.4. the overall SRP on the CaSSIOPeiA

2. SUN tower SSPS

Hight = 7871.0 MASS = 20000.0

Solar Radiation Area= 500000.0 Drag Area = 125000.0

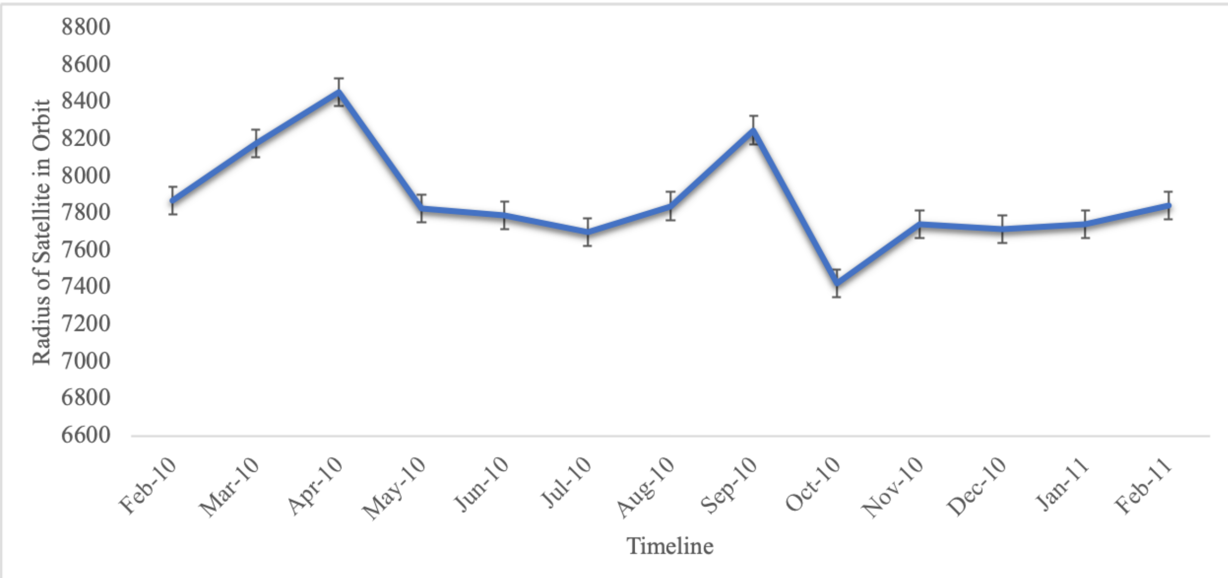


Figure B.5. the overall perturbation on the Sun Tower

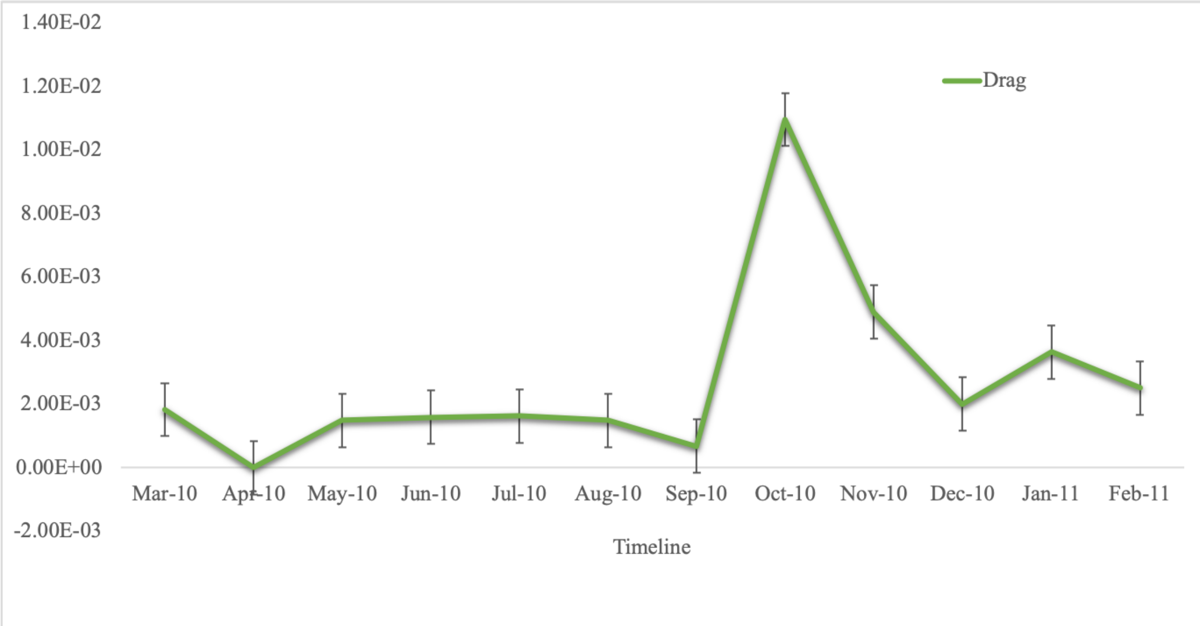


Figure B.6. the overall Drag on the Sun Tower

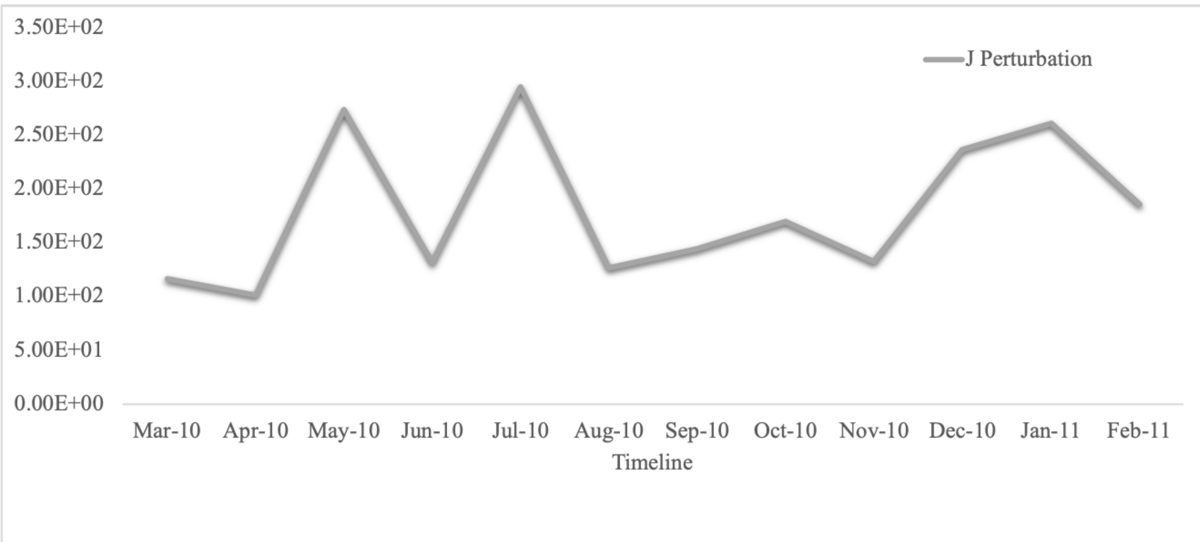


Figure B.7. the overall J perturbation on the Sun Tower

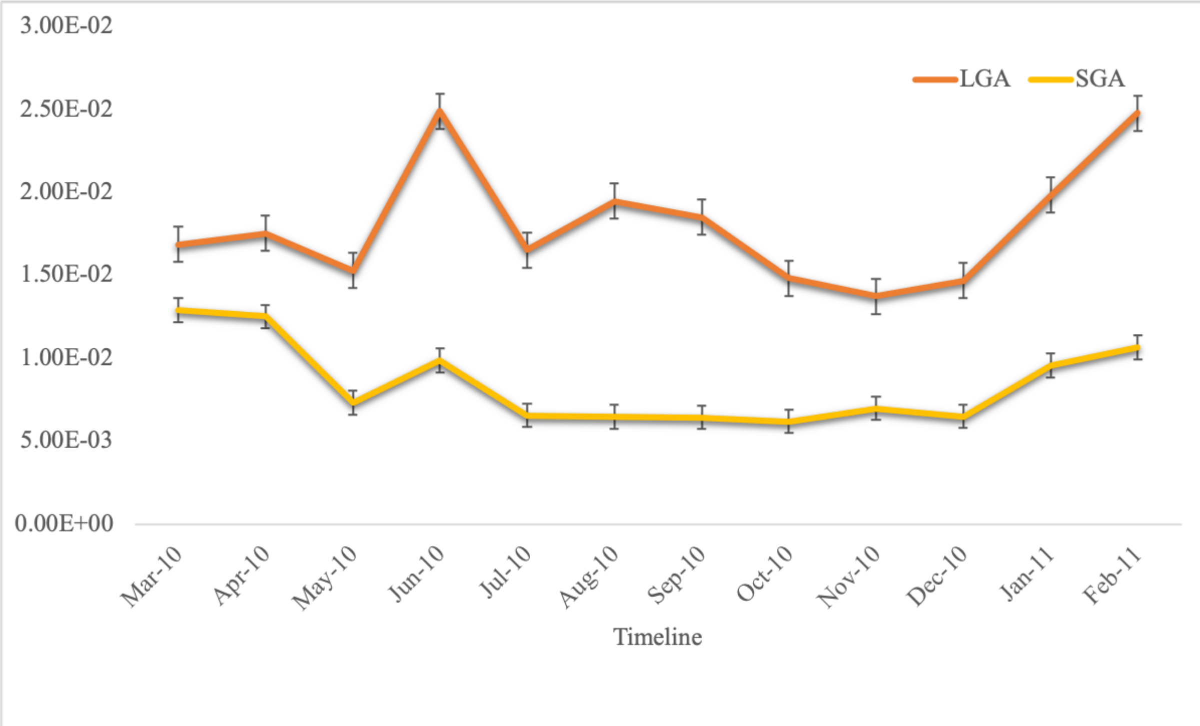


Figure B.8. the overall LGA and SGA on the Sun Tower

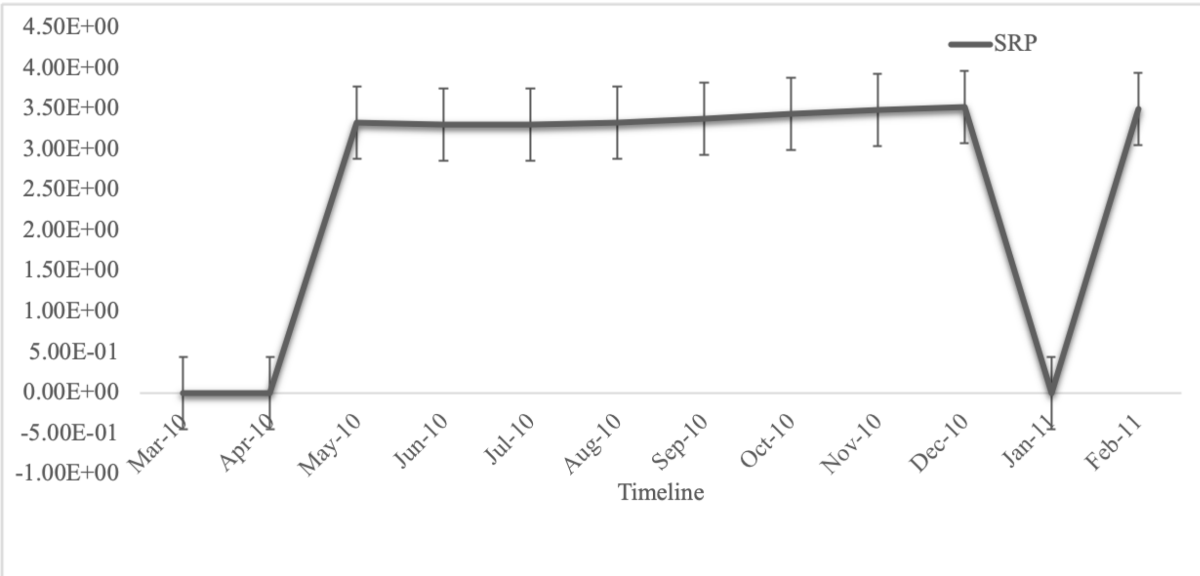


Figure B.9. the overall SRP on the Sun Tower

3. DoE NASA 1979_GEO

Hight = 42157.0 MASS = 250000.0

Solar radiation Area = 50785400.0 Drag Area = 14571350.0

During the program it found that the system of DoE NASA SSPS 1979 Is unstable to keep in space and it unbalance and fall out the structure within few days.

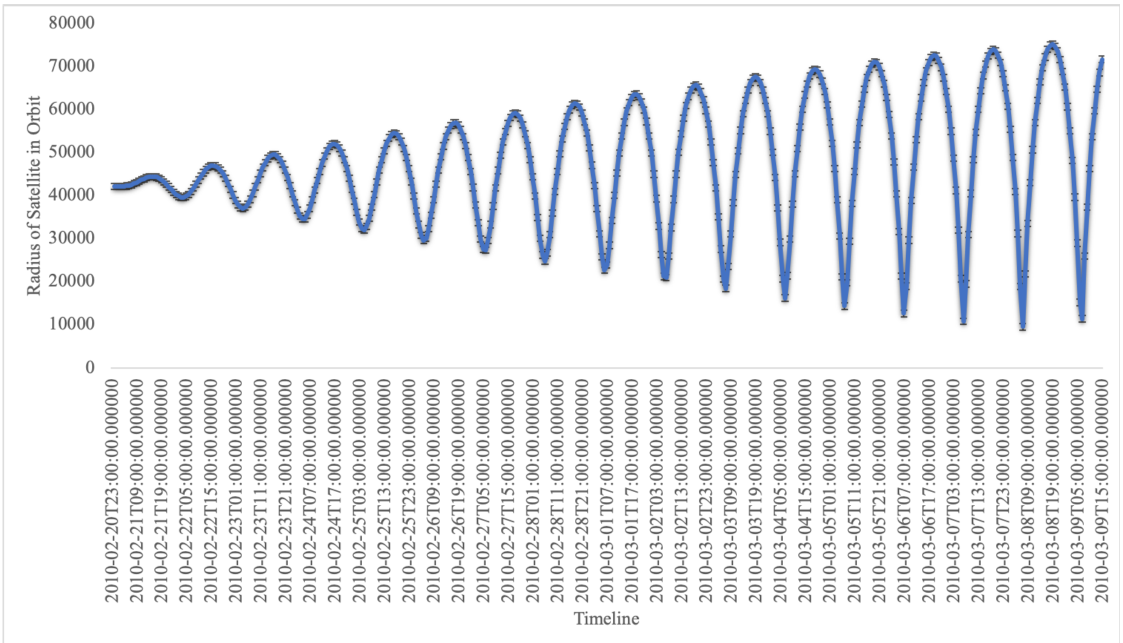


Figure B.10. the overall perturbation on the DoE NASA SSPS 1979

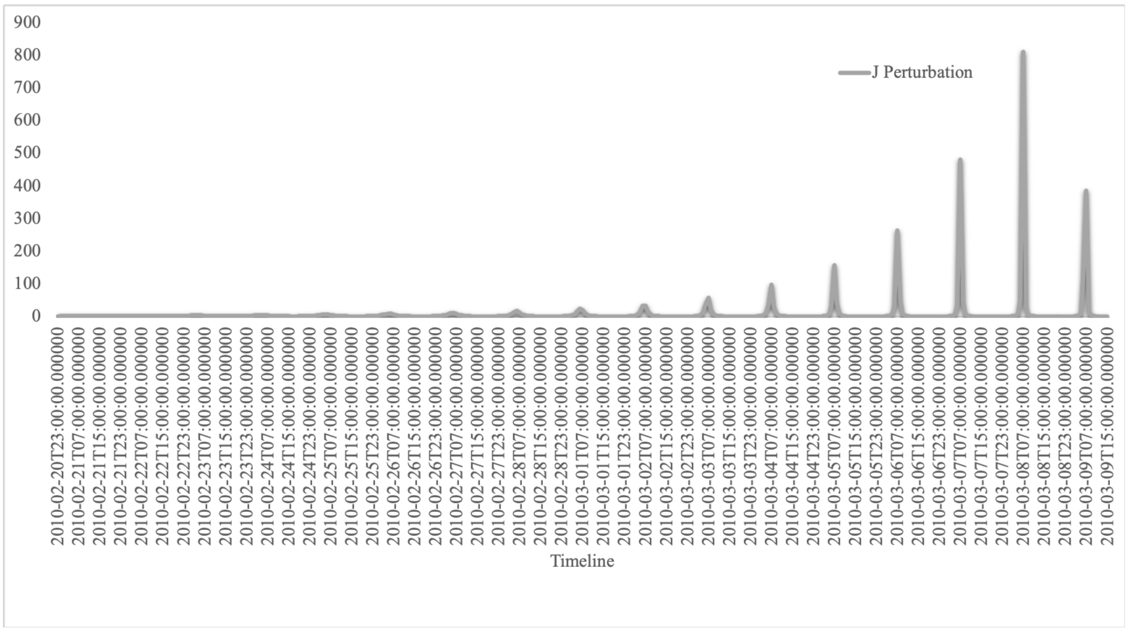


Figure B.11. the overall J perturbation on the DoE NASA SSPS 1979

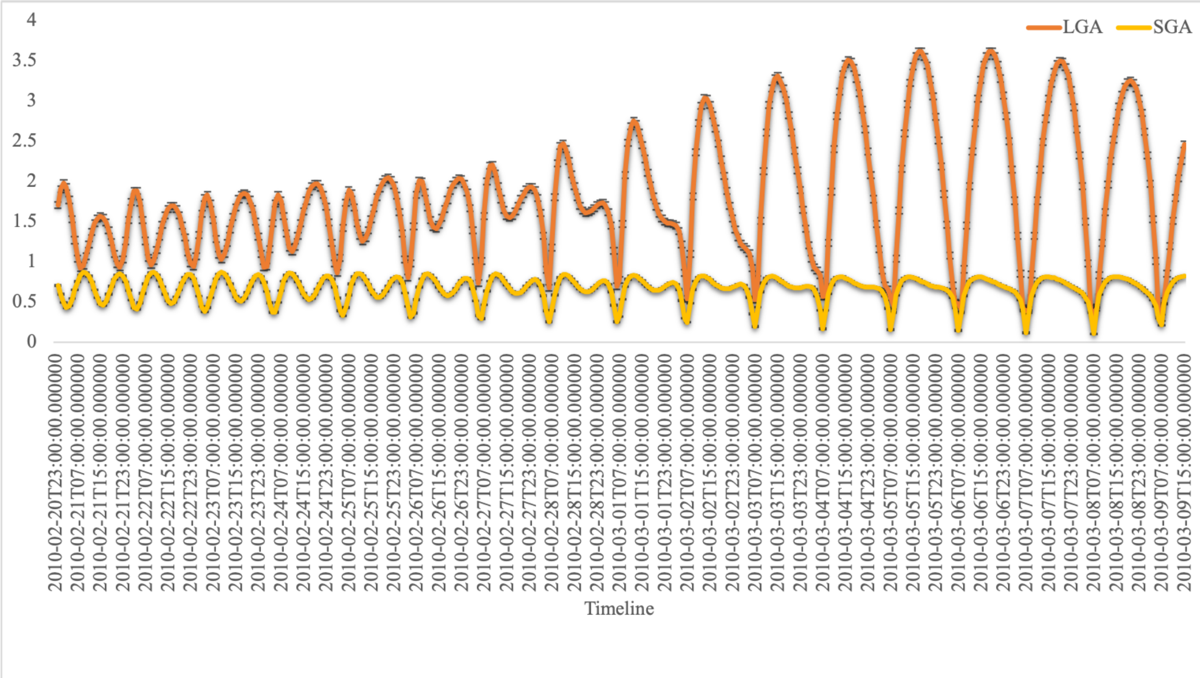


Figure B.12. the overall LGA and SGA on the DoE NASA SSPS 1979

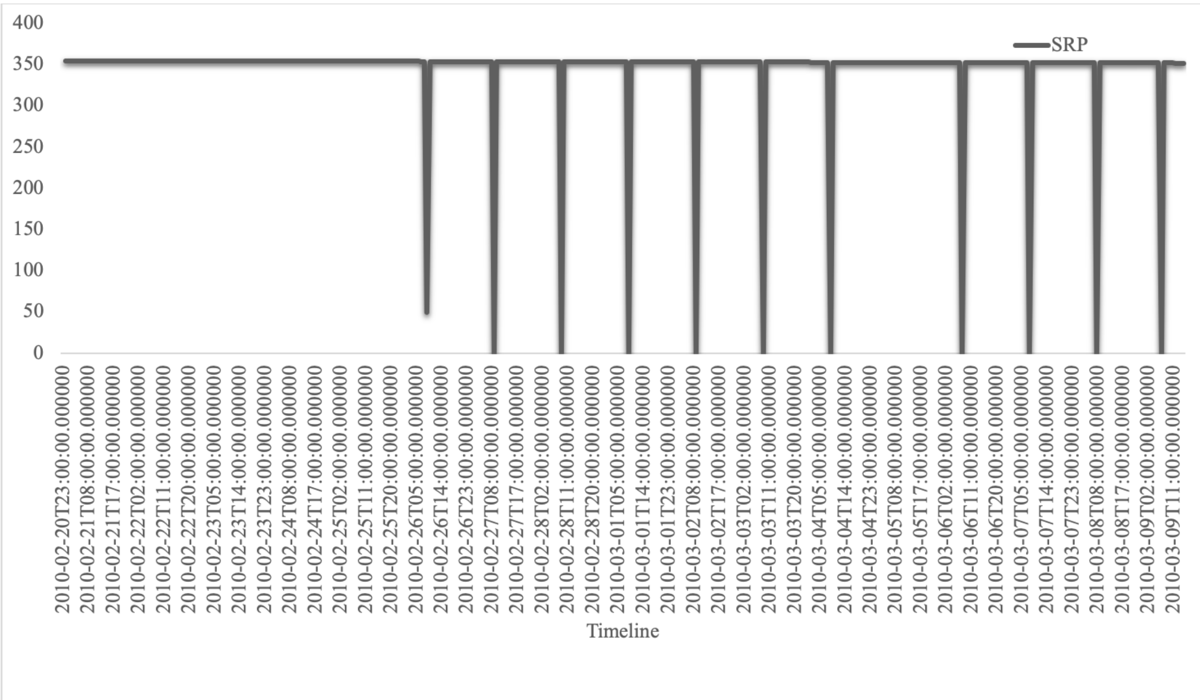


Figure B.10. the overall SRP on the DoE NASA SSPS 1979

ACKNOWLEDGEMENT

Those who walk the difficult path to success never rest at this destiny; they walk ahead towards greater success. We consider ourselves privileged to work under the guidance of such talented and experienced people who guided me all through the preparation of my thesis.

I am indebted to my supervisor, Professor **Dr. Toshiya Hanada**, for his support during the formation of this work. Who supported me with his vast knowledge, experience, and suggestion. Only their inspiration has made this doctoral work easy and exciting. I am also grateful to having supervision from Assistant Professor Dr. Yasuhiro Yoshimura, Mr. Shuji Nagasaki, and Dr. Hongru Chen. Their unwavering guidance, genuine friendship, and immutable enthusiasm for the topic were evident throughout its duration and acted as a catalyst for me to complete the D. Eng. successfully.

I appreciate the feedback from examiners and reviewers; Professor Dr. Naoki Shinohara and Associate Professor Dr. Koji Tanaka. Their comments improved the quality of this thesis.

Additionally, I would like to recognize Dr. Paul Jaffe (U.S. Naval Research Laboratory U.S.A) and Ian Cash (International Electric Company Limited, United Kingdom) for their constructive thoughts, opinions, insights, and advice. I feel deeply beholden to them since this truly helped shape the direction and positive outcome of my research.

Furthermore, I like to thank Miss. Hee Yung Woo (Georgia State University U.S.A.) for helping me learn Japanese, entrepreneurship and help me for constructive writing during writing my thesis and being a good friend.

I would like to express my hearty thanks to the students I met at Space System Dynamic Laboratory student, a researcher for their great support during my time at Kyushu University. My cordial appreciation to:

Kazunobu Takahashi	Shun Matsumoto
Daisuke Sakai	Zhang Yuang
Takahiro Maruyama	Luo Li
Kokichi Kagatsume	Shun Isobe
Taiki Urashi	Mahiro Tanahashi
Takahiro Shimizu	Kenta Nakajima
Syuta Fukii	Wu Long Hoang
Kakeru Masaki	Shingo Motodani
Kenji Imamura	Habibiroudkenar Pejiman

I also like to thank Peeyush Barskar, Mukul Bishwas, Vivek Baraskar, Shubham Gosavi, and all researchers from Entropy Research and Development Pvt. Ltd. India for enthusiastically supporting with active research. And the enormous depth of appreciation goes to the countless individuals who were gracious enough to assist me over the duration of my D. Eng. Without their collective contributions, the quality of this project would have been greatly diminished. Definitively, I heartily wish to express a huge thanks to my parents Chandrarekha and Chandrabhan Baraskar and my brother Peeyush Barskar for their love, support, and encouragement throughout this tenure project.

ADITYA BARASKAR