SOLAR POWER SATELLITES
SPS-REPOSE

Review of mission cases
(WP 2000, 3000, 4000)

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Document type	Nb WBS	Keywords :

SUMMARY

This technical note summarises the work done in the WP 2000, 3000 and 4000. It has been prepared by the SPS-REPOSE team with EADS-Astrium and Université La Réunion.

It reviews a preliminary definition of the SPS system for several types of applications: to provide power to geostationary platforms, to interplanetary vehicles, to rover or human outpost on mars surface, to infrastructure on Moon, and as alternative to Darwin mission. An assessment of the interest of the SPS system is done for each application.

Document controlled by :

DOCUMENT CHANGE LOG

Issue/ Revision	Date	Modification Nb	Modified pages	Observations
Draft	10/05/04			Preliminary version
00/00	19/07/04			For comments
00/01	8/09/04			Final version

PAGE ISSUE RECORD
Issue of this document comprises the following pages at the issue shown

Page	Issue / Rev.	Page	Issue / Rev.	Page	Issue / Rev.	Page	Issue / Rev.	Page	Issue / Rev.	Page	Issue / Rev.
All	draft										
All	00/00										
All	00/01										

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1 INTRODUCTION

Power generation is one of the crucial elements of space vehicles as well as of elements implemented on other planets and Moons. The on-board available energy is limited by the mass and size of the vehicle or elements and by the launcher constraints, although the improvement of the technologies (solar cells, storage systems, etc) allows an increase of the power system performances (efficiency, energy density). Furthermore, the sizing of the power generation system is driven by the eclipse period, requiring storage system and solar arrays surface for batteries recharging, by the environment and by the solar irradiation intensity.

In the medium term, Earth orbiting platforms will require higher power levels. Interplanetary exploration vehicles face the problem of distance to the Sun, especially when large amount of power may be needed. Large infrastructures on Moon and planets, like Mars, are constrained by environment attenuation, long eclipse or distance to the Sun.

New systems and technologies have to be found, which go beyond simple improvements of the current technologies. Solar Power Satellite (SPS) systems, based on wireless power transmission, are attractive candidate solutions to provide power to space vehicles or to elements on planet surface.

Studies have been carried out for tens of years on providing renewable electrical energy on Earth from Space Power Satellites. Targeted range of electrical power is from hundred MW up to several GW.

This study aimed at assessing the utilisation of Space Power Satellite to provide power to satellites, platforms, interplanetary vehicles or elements /infrastructures on Mars or Moon surfaces.

In the first technical note, a review of the RF and laser transmission systems has been done. For RF transmission system a 35 GHz frequency has been retained for the space-to space and space-to-planets applications. For the laser transmission system, visible and near IR range, with photovoltaïcs cells as receiver, has been preferred, while keeping the CO2 laser as a potential alternative.

This technical note aims to assess several applications of the SPS system for delivering power to:

- Earth orbiting platforms, with the example of geostationary platforms
- Interplanetary vehicle, with example of a vehicle/satellite travelling from Earth to Mars
- Elements on Mars surface: either a small rover, or a large Mars base
- Elements on the Moon surface
- Alternative to the Darwin mission configuration

This assessment includes an assumption on the user needs, a n analysis of the SPS location, a preliminary sizing of the transmission system, a preliminary definition of the SPS system and an assessment of the interest of using SPS system.

This technical note has been written by EADS Astrium and Université La Réunion.

Université La Reunion was in charge of the RF power transmission system and has written the related sections 5.5.1 and 6.3.1

2 MISSION CASE 1: GEOSTATIONARY SATELLITE

2.1 MISSION NEEDS AND CONSTRAINTS

The main requirements and constraints for the GEO satellites are:

- Power: 20 kW are required for the platform and the payload, that means at the output of the solar arrays.
- The SPS has to provide the power at least during the eclipse phase
- The satellite solar arrays shall be used as receiver system. Thus, the solar arrays will be used to be illuminated either by the sun or by the SPS system
- Consequently, the SPS will be based on a laser power transmission system, in the sunlight wavelength range (0.5 to 1 μ).

2.2 SPS POSITIONING

Two candidates for SPS positioning could be considered

2.2.1 Equatorial orbit 45000 to 55000 Km

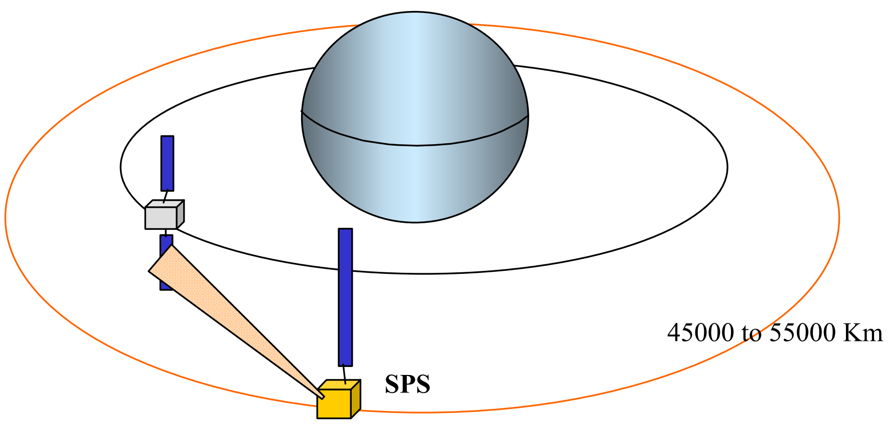


Figure 2.2/1: SPS position on equatorial orbit

The SPS is located in the equatorial plane, beyond the GEO. It can illuminate the communications satellite on a major part of its orbit, but a strategy of utilisation has to be discussed.

The SPS can be:

- Earth pointed; the power generation system (solar arrays, concentrators, other) has to be continuously pointed towards Sun (mechanisms), and the power transmission system (typically laser system) has also to follow permanently the target.

- Sun pointed; the power generation system is fixed, while the power transmission system has to follow the target with a higher range of pointing angle, according to the strategy of utilisation.

In both cases, the SPS will be subject to short eclipse around the equinox periods.

2.2.1.1 Distance and view angle between SPS and target satellite

The relative position of target satellite (TC) and SPS is shown on Figure 2.2/2

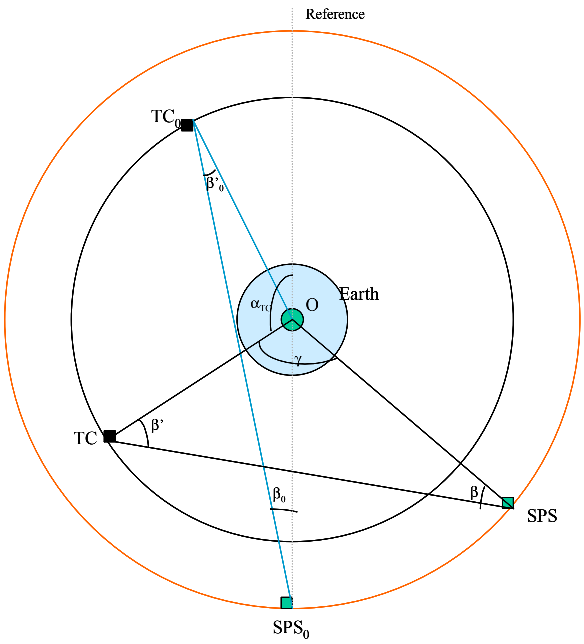


Figure 2.2/2: Relative position of SPS and coms satellite

- ***Initial conditions***

The line between target satellite and SPS is tangent to the Earth.

$\beta_0 = \text{Asin}(R_{earth}/R_{SPS})$

$\beta'_0 = \text{Asin}(R_{earth}/R_{TC})$

$D_0 = R_{SPS} * \cos\beta_0 + R_{TC} * \cos\beta'_0$

$\gamma = \pi - \beta - \beta'$

- ***Distance as function of time***

α_{TC} is the angle between the reference axis and the line 0-TC

α_{SPS} is the angle between the reference axis and the line 0-SPS. At t=0, $\alpha_{SPS0} = \pi$

$R_{SPS} = H_{SPS} + 6370$

$T_{SPS} = 2\pi * (R_{SPS})^{3/2} * \mu^{1/2}$

$\alpha_{TC} = \alpha_{TC0} + (d\alpha_{TC}/dt)*t = \alpha_{TC0} + 360*t/T_{TC}$ (T_{TC} is the period of the target satellite)

$\alpha_{SPS} = \alpha_{SPS0} + 2\pi*t/T_{SPS}$ (T_{SPS} is the period of the SPS)

$\gamma = \alpha_{SPS} - \alpha_{TC}$

$D = (R^2_{TC} + R^2_{SPS} - 2* R_{TC}* R_{SPS}*\cos\gamma)^{1/2}$

- ***Angle of view as function of time***

The evolution of angle between the direction of the target satellite and the direction of the Earth center, from the SPS, as function of time gives the evolution of the power transmission system pointing angle.

β = Asin (R_{TC}*(sinγ)/D)

The angle between the power transmission system direction (SPS-target line) and the direction to Sun (here the reference) is:

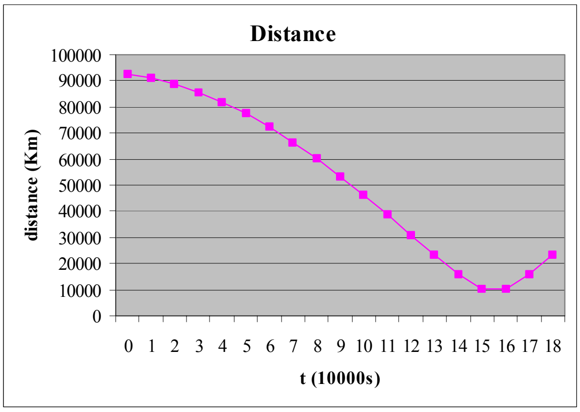
$\varphi = \beta + \alpha_{SPS} - \pi$

2.2.1.2 Case SPS at 45000 Km

The Figure 2.2/3 illustrates the evolution of the relative distance and of the true anomaly (γ angle) between the two vehicles as function of time. Initially, the SPS is located at SPS_0, with $\alpha_{SPS0} = \pi$. In these initial conditions (SPS-target sat line tangent to Earth), α_{TC0}=17,3°, the difference between true anomalies is γ_0= 162,7° and the distance between the vehicles is about 86300 km.

The period of the SPS orbit is 115910 s.

The SPS can illuminate the target satellite during several successive orbits of the satellite before being occulted by the Earth. Occultation occurs when γ is between 162,7° and 197,3°. The figure shows also that about 340000s will be necessary to find the two vehicles in the same position: that represents about 4 orbits for the telecom satellite, or 3 orbits for the SPS.



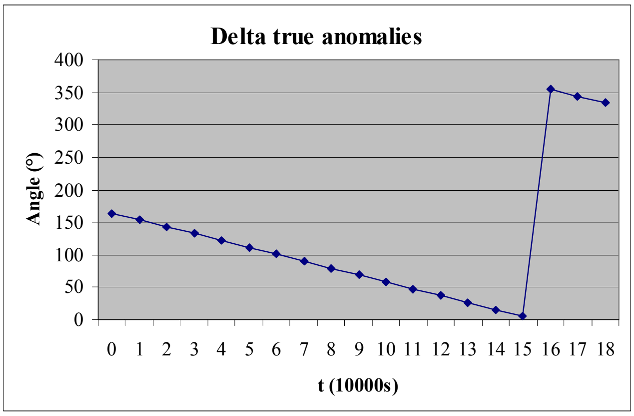


Figure 2.2/3: Distance and angle between the target satellite and the SPS as function of time (SPS at 45000 km)

In fact, the SPS will provide power to satellite only in a reduced range of relative position: it is proposed to provide power when the distance is lower than 25000 km, to limit the sizing of the SPS. With this assumption, the Figure 2.2/4 illustrates the relative distance and the angles (delta true anomalies, Earth line to target line, Sun line to target line).

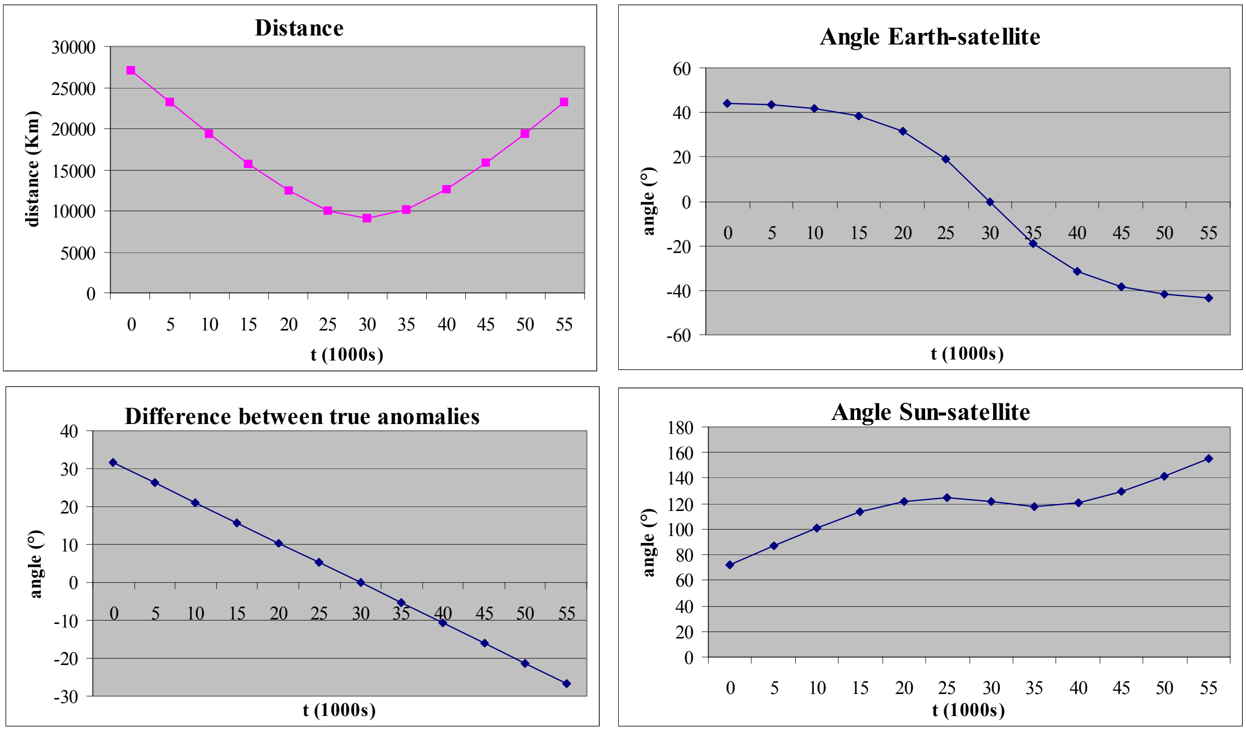


Figure 2.2/4: Evolution of relative position of both vehicles for a relative distance lower than 25000 Km

This period of utilisation of SPS could be limited as follows:

- Duration: about 50000s. It is more than half the orbit of the satellite. Such a period covers the eclipse period of the target satellite
- Delta true anomaly: the relative true anomaly varies between +30° and –30°.
- If the SPS is Earth pointed, the pointing angle of the power transmission system (angle between SPS-Earth direction and SPS-satellite direction, varies between –40° and + 40°, that means an evolution of 80° within 50000s, centered around 0. The required pointing velocity is 0,0016°/s.
- It the SPS is sun pointed, the pointing angle of the power transmission system (angle between SPS-Earth direction and SPS-satellite direction, varies from 70 to 160°, that means an amplitude of 90° and an angular velocity of 0.0018°/s. The power transmission system pointing angle is not limited in case of SPS sun pointed. However, if only eclipse periods of the target have to be considered, the evolution of the power transmission system pointing angle is limited.
- This period occurs every 4 orbits of the target satellite

2.2.1.3 Possible utilisations and impacts on target

Utilisation during eclipse and daylight

As quoted above, the SPS can illuminate the satellite during several orbits before being occulted by the Earth. The distance between the SPS and the satellite varies in a large range (9000 to 86000 km) and the power beam steering angle varies also in a wide range (-55 to +55°), even if the SPS is Earth pointed.

When the satellite is illuminated by both the Sun and the SPS, it receives a higher level of energy, with the laser energy focused on a restricted wavelength range.

However, the satellite solar arrays are facing sun, so that the SPS can provide additional energy to the satellite only if its relative location allows it to see the solar array side covered with photovoltaic cells.

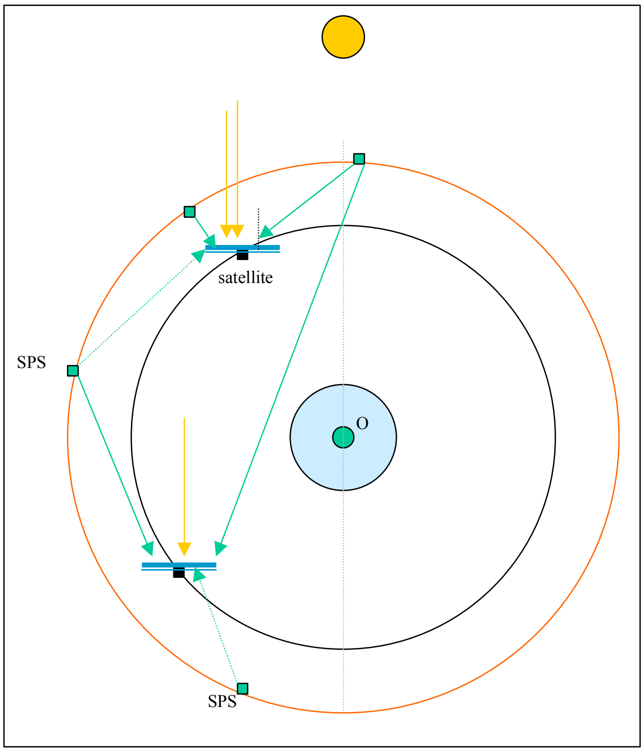


Figure 2.2/5: Energy from Sun and SPS

Indeed, as illustrated on Figure 2.2/5, the SPS power is not normal to the solar arrays surface, except in particular configurations, and the angle between the normal to the solar arrays surface and the power beam varies between 0 and 90° according to the respective positions of both vehicles. In addition, if the SPS is located in the anti-Sun side with respect to the solar arrays surface, the power beam will illuminate the backplane of the solar arrays (covered with OSR for heat dissipation) that will prevent from the nominal radiating function could increase the cells temperature with a risk of drastic degradation. As illustrated on the Figure, the probability of receiving energy from SPS is rather low when the satellite is close to the Earth-Sun line, while it increases when the satellite is on the anti-Sun side with respect to Earth.

When passing in eclipse phase, the satellite has to rotate its solar arrays towards the SPS to have them normal to the power beam. If the SPS power transmission system is designed to provide the same sizing energy density as the Sun, the solar arrays will provide to the user the same power as during daylight.

Therefore, the solar arrays surfaces have to be sized taking into account the lowest received sun energy density (summer case) and the highest one (winter). In addition, they would receive a variable energy from the SPS, varying with the configuration from zero up to twice the sun energy. The satellite power system will have to cope with such a large variation of received energy along an orbit in the design of distribution, heat dissipation, etc. Indeed, with the current configuration, to provide twice the sun energy on the cells would degrade or destroy these cells due to a too high resulting temperature. A modification of the solar array configuration would be necessary with more space between the cells and introduction of OSR for heat dissipation; as the solar arrays sizing case would remain the sun energy in summer (without additional SPS energy), that would lead to larger solar arrays. The use of a set of two or three SPS would lead to the same results.

If the SPS is used as primary source of energy, the satellite solar arrays will rotate to face the SPS. In that case, the Sun will provide an additional energy varying between the full sun energy (satellite SPS aligned with Sun direction with SPS on sun side), and no energy. In addition, the sun can illuminate the backplane, impacting the thermal control.

Therefore, the use of SPS during daylight and eclipse does not lead to reduce the size of the solar arrays surface; it would lead to modify the solar arrays configuration.

Utilisation during eclipse phase only

The SPS could cover only the eclipse phase of the target satellite. In that case, it would complement the sun to provide permanently the needed level of power to the target. The maximum transmission duration would be 4160 s, during which the SPS will perform 13° on its orbit.

As indicated above, the SPS can provide power to the target in eclipse area from any position on its orbit, except when it is located between –7.1° and +24.3° around the sun direction (red zone, target not visible from SPS due to Earth occultation) and between –7.1° and +7.1° around the anti Sun direction (SPS in eclipse).

If SPS provides power from anywhere on its orbit, the SPS-to-target distance will vary between 86000 and 9000 km. In case of SPS earth pointed, the beam angle will vary between –55° and +55°. During a transmission period, its variation will be up to 6°. Besides, as discussed hereunder, there will be impact on the satellite solar arrays. But this case has a high availability of the power supply to the satellite. A set of two SPS will ensure a permanent supply of power to satellite in the eclipse period.

There are two options (see **Figure 2.2/6**), with the objective to limit the requirements on the power transmission system beam steering angle:

- to have the SPS in the same side as the target satellite, in order to minimise the distance. It is in addition proposed to get the SPS within the range of 25000 km. This is illustrated on Figure 2.2/7. In that case, the SPS shall be located in the range of true anomaly of [147-200°], except the eclipse zone, that means –33° to +20° around the target satellite eclipse zone. The beam steering angle amplitude, if SPS is sun pointed, varies between 3 and 10°. In case of Earth pointed, the beam steering angle has a low variation during the period of transmission (typically 1°). The angle of view is about 53° on one side of the eclipse area, -53° on the other side. The satellite will have to turn towards SPS its solar arrays, which are basically facing Sun, even during eclipse. That impacts the logic of control of these solar arrays and assumes a transition period to be managed, although it is short.

- To have the SPS in the Sun side, such as to avoid the Earth occultation. Distances vary between 80000 and 92000 km. The SPS shall be located in the range of true anomaly of roughly [12-50°], or [-25 to –50°]. The beam steering angle amplitude, if SPS is sun pointed, could be up to 40°. In case of Earth pointed, the beam steering angle will vary during one period of transmission by 6°, with a rate of 0,0014°/s. However, according to the position of SPS on its orbit at the satellite eclipse time, the beam pointing angle may vary from –15 to + 15°. The satellite will have to turn its solar arrays towards the SPS, but the rotation angle will be much lower (typically less than 25°) than in the previous case.

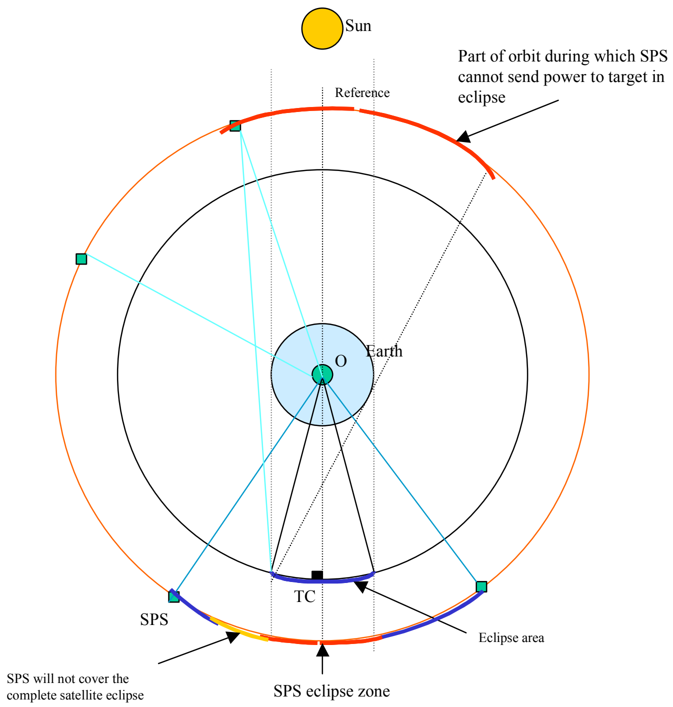
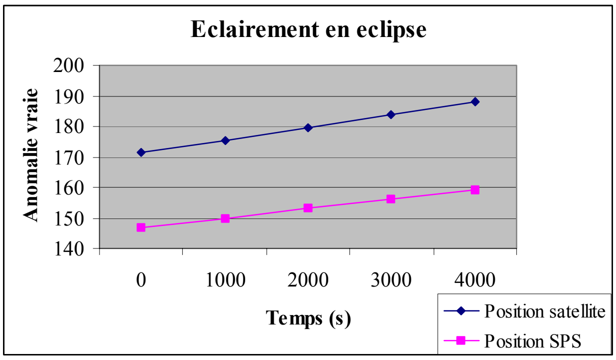


Figure 2.2/6: Illustration of the relative positions for power transmission in eclipse



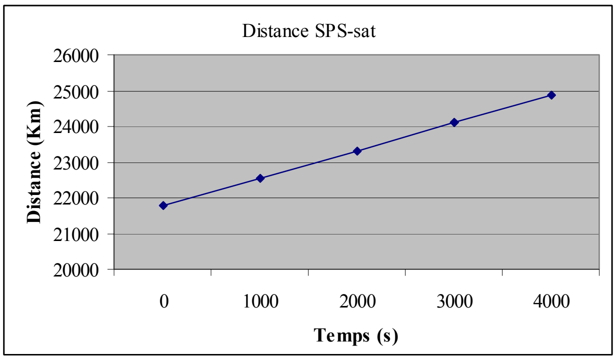
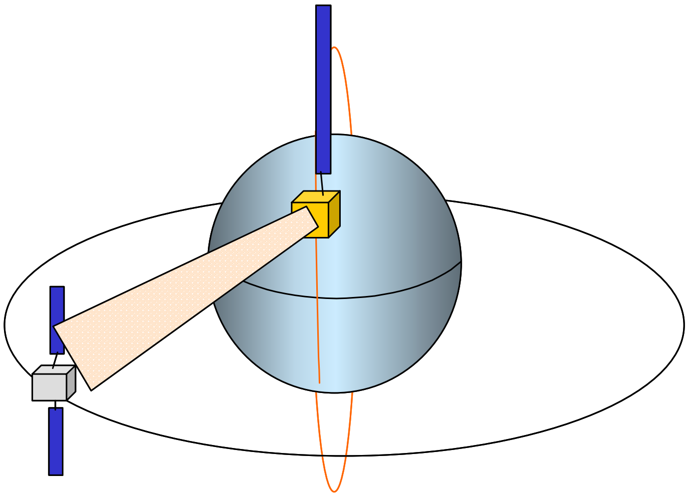


Figure 2.2/7: Illustration of relative position of SPS and satellite during the satellite eclipse. In that illustration the relative position of the SPS when the satellite enters the eclipse phase is such that the distance at end of eclipse is lower than 25000 km.

2.2.2 Sun synchronous orbit



The SPS is located on a sun synchronous 6-18h orbit. Altitude is assumed in order of 1000 km, so that it is permanently illuminated by the Sun.

Utilisation during daylight and eclipse

If the SPS has to transmit power to the target satellite at any time, the transmission distance will vary between 34 870 Km and 42750 km, as illustrated in the Figure 2.2/8

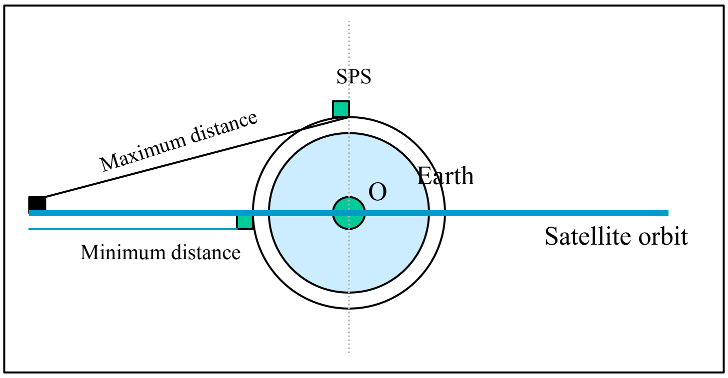


Figure 2.2/8: Maximum and minimum distances between the SPS and the target

Nevertheless, the permanent utilisation of the SPS raises two problems:

- During at least half of the satellite orbit, the SPS illuminates the backplane of the satellite solar arrays as being in the anti-Sun side with respect to the satellite. Thus, the SPS cannot provide power to the satellite at least during half the time. But it corresponds to the illumination of the satellite by the Sun. When the satellite is in the other half of its orbit (SPS in the sun side), the SPS beam power angle of view with respect to the solar arrays (facing Sun) varies between 0 and 90°, the most efficient configuration occurring during eclipse.
- The power beam angle, with an Earth pointed SPS, varies between 0 and 180°.

Utilisation during eclipse phase

The utilisation during the eclipse phase is illustrated on Figure 2.2/9. The SPS orbit plane is quasi perpendicular to the satellite-sun direction.

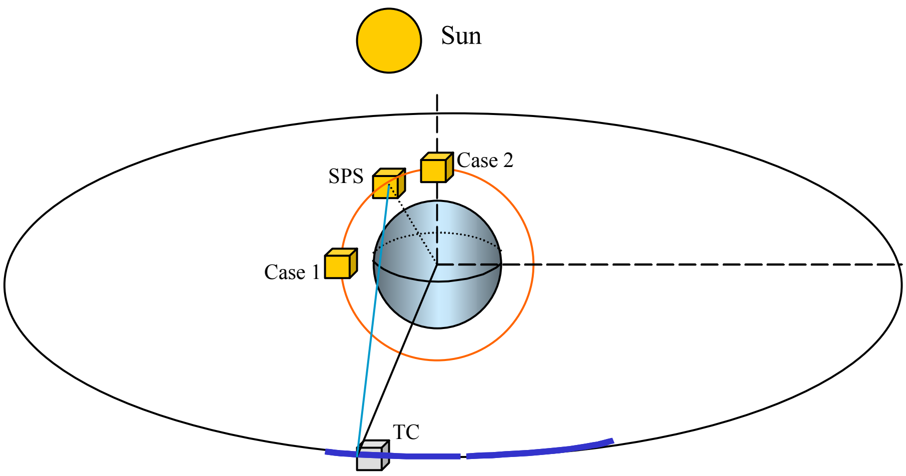


Figure 2.2/9: Power beam in eclipse phase

The evaluation of the distances between SPS and target and of the beam steering angle can be done in two envelope cases: the SPS in the equatorial plane and the SPS above the pole.

- SPS in the equatorial plane. The Figure 2.2/10 gives the evolution of the relative distance SPS to target and of the beam pointing angle when the target describes the entire trajectory during the maximum eclipse phase. It shows that the distance varies between 41700 and 44100 km, and the pointing angle between 1 and 19°.

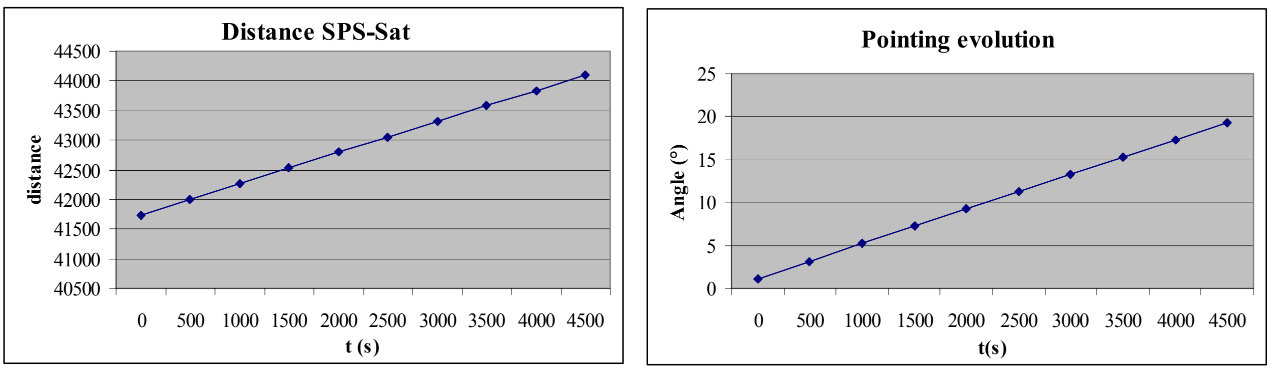


Figure 2.2/10: Distance and pointing angle evolution when the SPS is in equatorial plane

- SPS in polar plane. It is symmetrically located with respect to the eclipse phase trajectory. The Figure 2.2/11 shows that the distance SPS-to-target is quasi constant at 42880 km and the beam pointing angle varies between –9 to +9°.

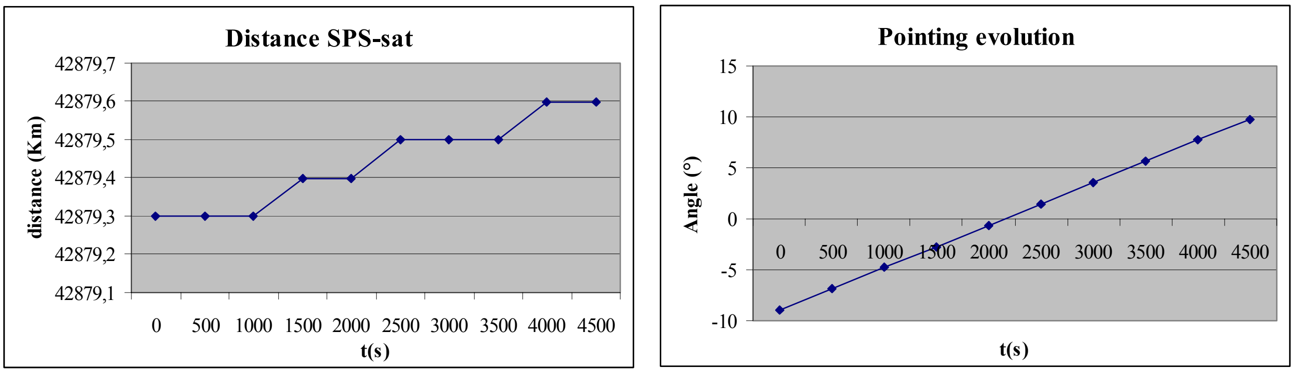


Figure 2.2/11: Distance and pointing angle evolution when the SPS is above the pole

Therefore, the sizing case for the SPS in sun synchronous orbit is:

Power to be provided: 20 kW

Power transmission during eclipse

Distance : 42000 to 44000 km

Transmission system pointing angle evolution (in horizontal plane): < 19°

Maximum azimuth: 80° (Direction of beam, normal to equatorial plane) (minimum azimuth is 90°). Beam pointing evolution is +/- 10° in vertical plane

Angular rate: < 0,0046°/s

Transmission duration: 4160 s

2.2.3 Synthesis

SPS utilisation

The utilisation of the SPS in daylight does not allow to reduce the size of the satellite solar arrays. Indeed, whatever the positioning of the SPS, and assuming the satellite solar arrays facing Sun, the additional energy brought by the SPS is variable between 0 and twice the solar energy density. Such an energy density (twice the sun energy) is not compatible with the current geostationary satellites solar arrays design, the risk being to degrade or destroy the cells. Adding both sun and SPS energies would require a modified design with insertion of OSR around and between the cells and finally larger solar arrays surfaces. Moreover, when the satellite is in a certain part of its orbit, the SPS illuminates the backplane of the solar arrays. This is also not possible as it prevents from heat dissipation via the backplane. This part of the orbit is variable with the SPS in equatorial orbit, and is the half orbit on the sun side when the SPS is in sun-synchronous orbit. The use of a solar arrays with cells on the two sides would require insertion of radiative material (typically OSR) between and around the cells) leading to larger surfaces and could be considered with only one side illuminated at a time. On thermal point of view the two sides cannot be illuminated simultaneously. The utilisation of SPS in daylight in addition to the Sun would result in high impacts on the satellite power generation system and on the design of solar arrays.

Therefore, the proposed utilisation of the SPS is only to cover the eclipse period for different telecommunications satellites. Consequently, it will be used only a part of the year, around the equinoxes.

Impacts on target satellite

The solar arrays have one face covered with cells, the other face being radiative. (OSR).

The target satellite is Earth pointed, the solar arrays rotating over 360° around an axis perpendicular to the orbit plane, as illustrated on Figure 2.2/12. The rotation of the solar arrays continues at the same rate during eclipse such as to retrieve the Sun with adequate pointing at the output of the eclipse period.

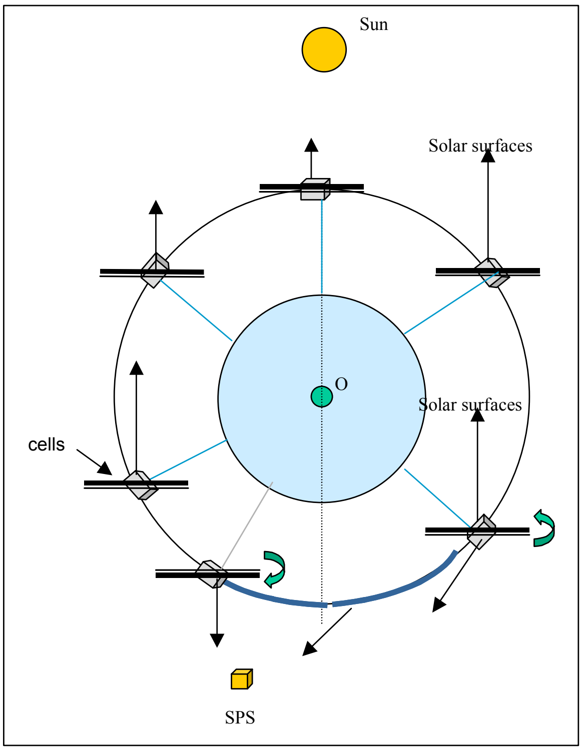


Figure 2.2/12: Satellite solar arrays configuration along its orbit

- ***SPS on equatorial orbit, on the eclipse side***

As illustrated on Figure 2.2/12, the satellite solar arrays have to be rotated by at least 150° to face the SPS power beam. It has to be done at a much higher rate than the nominal rotation rate of the mechanism (basically 12 h for 180°) in order to have a very short transient. Same rotation problem at the output of the eclipse period. That has a drastic impact on the solar arrays mechanism, on the solar arrays surface themselves (impact of a much higher rotation rate and resulting efforts on the large surfaces) and on the rotation control algorithm including SPS power beam acquisition, and Sun acquisition at the end of eclipse. Anyway, there will be a transient phase at beginning and at end of eclipse, and batteries will be necessary to cover this phase. The use of solar cells on two faces would avoid the transient, but would have impacts on the solar arrays definition and on the solar array thermal control. On availability point of view, due to evolution of relative true anomaly, the SPS will deliver power to the satellite during eclipse roughly every four orbits (25% coverage of the eclipse periods).

- ***SPS on equatorial orbit, on sun side***

The SPS uses the same side of the solar arrays as the Sun. The solar arrays will have to be reoriented to face SPS when the satellite enters the eclipse, but by 25° maximum. In that case, the satellite continues to receive power when entering in eclipse, with a small reduction of the power level (due to the angle). The power level will reach its nominal level at the end of the reorientation. The reverse rotation will be necessary at the output of the eclipse, with a worst case of 37°. To minimise, or avoid impacts on the solar arrays rotation mechanism and on the large surfaces, it would be possible to correct only a part of this angle or even to keep the nominal solar arrays rotation without correction. To that aim, the laser power transmission system should be sized so that the energy density on the satellite solar arrays is slightly higher than the sun energy density to compensate the angle of view. This configuration leads to a low availability, between 25 and 50%.

- ***SPS in sun synchronous orbit***

SPS is viewed as the Sun by the satellite solar arrays. Thus, when the satellite enters the eclipse period, it continues to receive power from the SPS. There is no power interruption. The satellite solar arrays receive the power beam under an angle being lower than 19° (angle between sun direction and SPS direction). Therefore, the solar arrays rotation mode remains nominal (no impact) and the laser power transmission system will be designed to take into account this maximum loss of energy due to the angle. This configuration allows a 100% availability of the power for the satellite.

Comparison of candidate positioning for SPS

For providing power to geostationary satellite during eclipse, the SPS will be **preferably located in a sun synchronous 6-18h orbit**. An altitude in order of 1000 km will allow the SPS to be permanently in view of the Sun. The advantages of this positioning with respect to the other candidate are:

- No impact on the satellite solar arrays (which keep their nominal rotation mode), as compared to equatorial orbit with eclipse side that leads to high impacts.
- Lower distances between SPS and target as compared to equatorial orbit with sun side (half the distance)
- Fixed SPS solar surfaces viewing permanently the sun

- Permanent view of the target satellite (no occultation), leading to 100% availability of the power for the target with one SPS (3 or 4 would be needed in equatorial orbit to ensure the same availability)
- Access (launcher performance) is easier in low sun synchronous orbit than in equatorial orbit beyond the geostationary orbit.
- The power beam is directed towards the space, while there is a risk of high power density on Earth with an SPS on equatorial orbit with a beam oriented towards Earth.
- Two degrees of freedom for the power beam steering: +/- 19° in horizontal plane, +/- 10° in vertical plane

2.3 POWER TRANSMISSION SYSTEM

2.3.1 Laser transmitting system driving parameters

The SPS will use a laser transmission system compatible with photovoltaic system of the satellite solar arrays. The laser will then be in the sunlight wavelength rage.

The optimum wavelength has to be in the visible range.

Other important and constraining parameter is the required optical power at communication satellite solar array (20 kW). The primary optical power has thus to be in the range of several tens of kW.

The receiving area is relatively large (several hundred of square meters), so the pointing accuracy and the power spot at receiver level can be relaxed accordingly.

We propose a SPS constellation composed of several satellites. Each satellite embarks a reasonably sized beamed laser system, which individually points in closed loop the communication satellite. A laser beacon will be implemented on the solar panel array of the communication satellite. The proposed system is considered as self redundant because the failure of one SPS will only reduce the SPS service accordingly but not the SPS mission.

2.3.2 Laser SPS system

2.3.2.1 Laser system

As explained in previous technical note (ref N° 04/SZ1/322NT/CC), laser diode pumped Nd laser seems to be the most suited for SPS applications. We will propose for most of the SPS applications a multi kW Nd laser.

The concept of the laser is presented in this chapter because this application is the most demanding mission. All the other Nd lasers proposed in the other SPS missions will be equivalent.

The key challenges for developing high-average power (HAP) solid-state lasers (SSL) are the thermo-mechanical problems, which occur in the gain medium as a result of pumping. Even with pumping by the efficient semiconductor laser diodes, as much as half of the pump energy is dissipated into heat, which is deposited into the SSL medium. The proposed concept is similar to the Active Mirror Amplifier concept, which is proposed for most of the development of Multi-kW laser that are presently studied for many applications (Scientific applications as laser fusion or military weapons). The present design is based on the Compact Active Mirror Laser (CAMIL) developed by BOEING. This concept shows a strong interest for average power laser scaling.

CAMIL uses a large-aperture laser gain medium mounted on a rigid, cooled substrate, as illustrated on Figure 2.3/1. The laser medium is a relatively thin disk, of about 2.5 mm thickness, and with a diameter typically between 5 and 15 cm. The substrate contains a built-in heat exchanger with microchannels on the front surface so that coolant can directly wet the back face of the laser medium disk. Except for the microchannel penetrations the front surface (facing the disk) of the substrate is ground to optical flatness. Diode pump arrays are arranged around the circumference of the gain medium disk. Pump radiation is first focused through a lensing element, concentrated in a conical hollow duct and then injected into the perimeter edge of the disk. Preferably, the disk edge receiving pump radiation has a tapered undoped section that channels the pump radiation into the gain medium. Such undoped extensions are analogous to those used in rod lasers and have been shown highly effective for reducing thermal stresses and Amplified Spontaneous Emission (ASE) losses. By appropriately positioning and focusing pump diode arrays, it is possible to uniformly pump the laser gain medium across the entire aperture. Principal advantages of side-pumped CAMIL are greater flexibility in choosing substrate material (does not have to be transparent) and capability to use the efficient Yb:YAG gain medium.

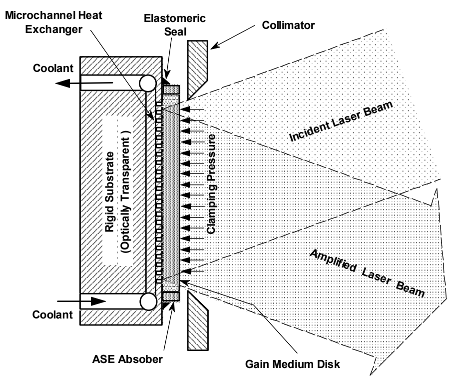


Figure 2.3/1: CAMIL concept (from ([11])

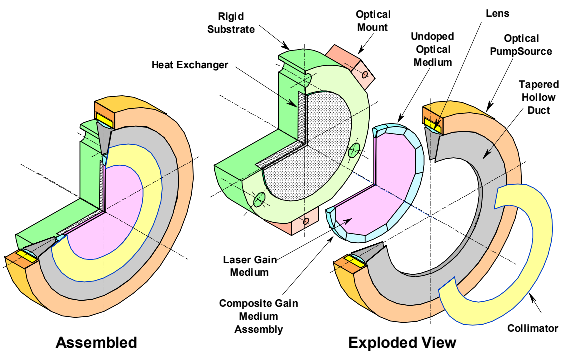


Figure 2.3/2: Side-pumped CAMIL module (from ([11])

CAMIL modules can be used to construct laser amplifiers and oscillators. A laser oscillator may typically use ten or more CAMIL modules to obtain a round-trip gain sufficient for high outcoupling from an unstable resonator as shown, for example, in FIGURE 2.3/3.

6 to 10 kW average output power laser can be obtained with 5 to 10 CAMIL modules of 4,5 cm diameter. The radiation is doubled to generate a 532 nm radiations optimised for solar arrays efficiency.

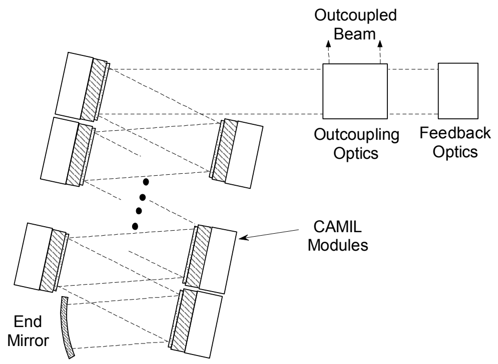


Figure 2.3/3: Laser oscillator with CAMIL modules (from ([11])

2.3.2.2 Telescope and Laser pointing system

Figure 2.3/4: SIC Telescope

A light weighted telescope will be implemented. This telescope has to be made with the ASTRIUM SiC technology, which has been space qualified up to 3.5 m diameter. It is illustrated on Figure 2.3/4.

A two mirrors pointing system has to be implemented for pointing the laser beam on to the communication satellite with sub-microradiant accuracy.

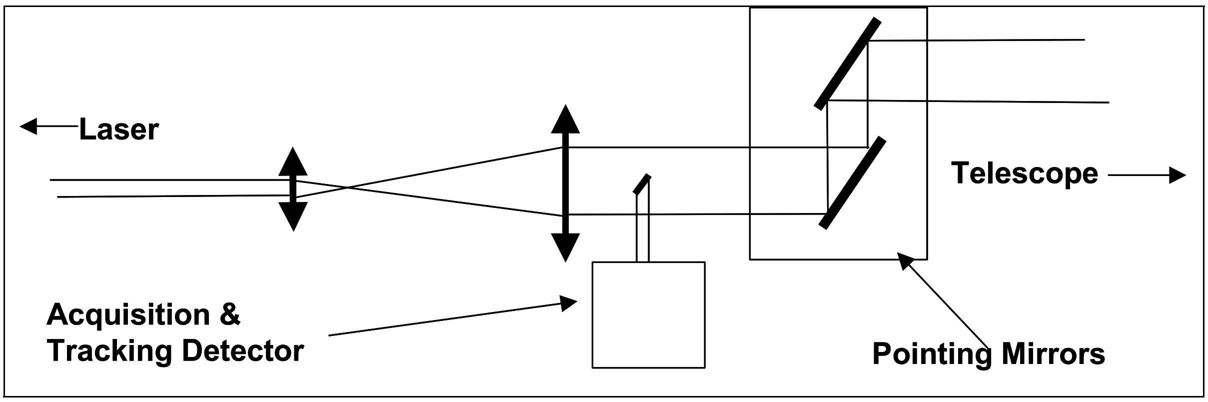


Figure 2.3/5: Pointing system

2.3.2.3 Laser system parameters

A set of 4 lasers, each emitting 10 kW, is proposed to provide the required 20 kW to the satellite. Each laser could be mounted on a different satellite.

The main parameters of the proposed laser system are given in the Table 2.3/1. Each telescope has a diameter of 2m, and the laser is operating in the 0,532µm wavelength. The mass and the power budget of each 10 kW laser are given in the Table 2.3/2 and Table **2.3/3**. The laser will require about 269 m² radiative panels to ensure a correct thermal control

Laser system parameters		
Laser power	10000	W
SPS satellite Number	4	
Operating Wavelength	532	nm
Telescope Diameter	2	m
operating distance	45000	km
Diameter of the Spot on target aera	14,6	m
50% Loss pointing Accuracy	192	nrad
Surface of the PV target area	300	m²
Efficiency of the PV Cell	50	%
Illumination at target level	238,82	W/m²
Generated electrical power	20000	W

Table 2.3/1: Laser system parameters

Sytem laser mass	Mass in Kg
Telescope	80
Pointing system	5
Opto-mechanical assembly of laser	200
Mechanical structure	200
Electronics	200
Total	685

Table 2.3/2: Laser mass budget

Power flow chart laser		
Emitted power	10000	W
SHG generation efficiency	70	%
Optical pump efficiency	50	%
Laser diode plug in efficiency	50	%
Power generation system		
DC/DC converter efficiency	85	%
Power required at SA output	70000	W
solar cell efficiency	30	%
solar power density	1300	W/m²
solar array size	176	m²

Table 2.3/3: Laser power budget

2.4 POWER GENERATION SYSTEM

Each SPS of the constellation brings a 10 kW laser. This laser, as shown on Figure 2.3/8, requires 57 kW at the diode input level. Assuming a DC/DC converter efficiency of 85%, this leads to a power requirement of 70 kW on the power bus.

Considering the power level, the proposed power architecture is illustrated on Figure 2.4/1. It is based on a 100V DC regulated power bus with Sequential Serial/Shunt Regulator. The switches can be sized for a voltage drop lower than 0.5 V by paralleling Mosfets, thus leading to a conversion efficiency of 99,5%. Solar arrays voltage matching is also very good because of the relatively constant solar flux and of the regulated bus voltage. In order to provide 70 kW at the DC/DC converter input, up to 176 m² of solar arrays are necessary, as illustrated on Table 2.3/3. These solar arrays are made of multiple-junction solar cells with small concentrators with an efficiency of 30%. The resulting solar arrays mass would be 264 kg, assuming a density of 1.5 kg/m².

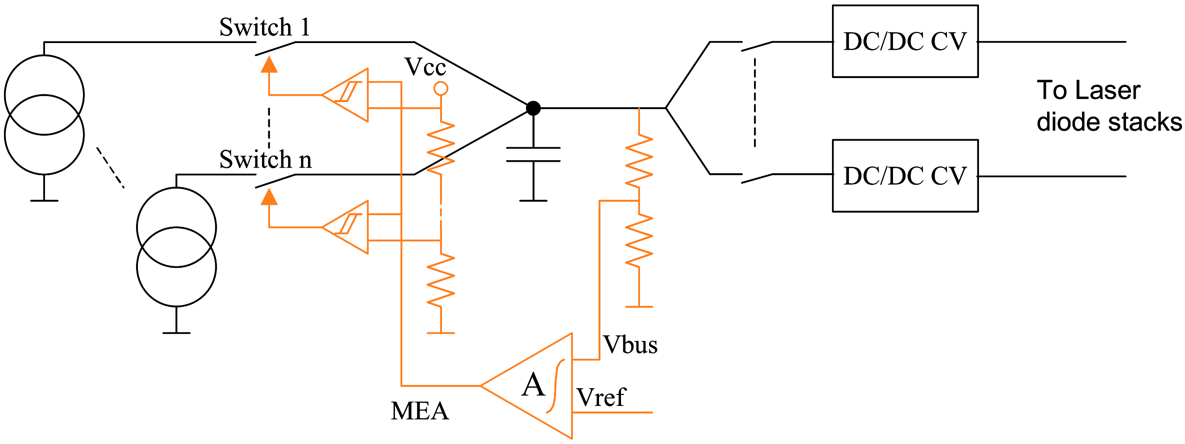


Figure 2.4/1: Electrical power system architecture

2.5 PRELIMINARY SYSTEM CONCEPT

The proposed SPS system is composed of 4 identical satellites, each having a 10 kW laser. These satellites are flying in formation, on the sun synchronous orbit; thus all the solar arrays are fully facing sun, without any occultation of one set of solar arrays by another satellite. The four beams are focusing on the geostationary satellite receiver arrays when in eclipse. An illustration of the SPS is shown in Figure 2.5/1.

Each SPS has a mass at launch around 14t.

The overall system efficiency is 6,25%. It represents the ratio between the power available at user interface (in that case at the output of the geostationary satellite solar arrays) and the power generated at the output of the SPS solar arrays. The Figure 2.5/2 summarizes the system performances. The on board system efficiency is 12,25% (17,5% for the laser system and 85% for the DC-DC converter), the receiver system has an efficiency of 50%.

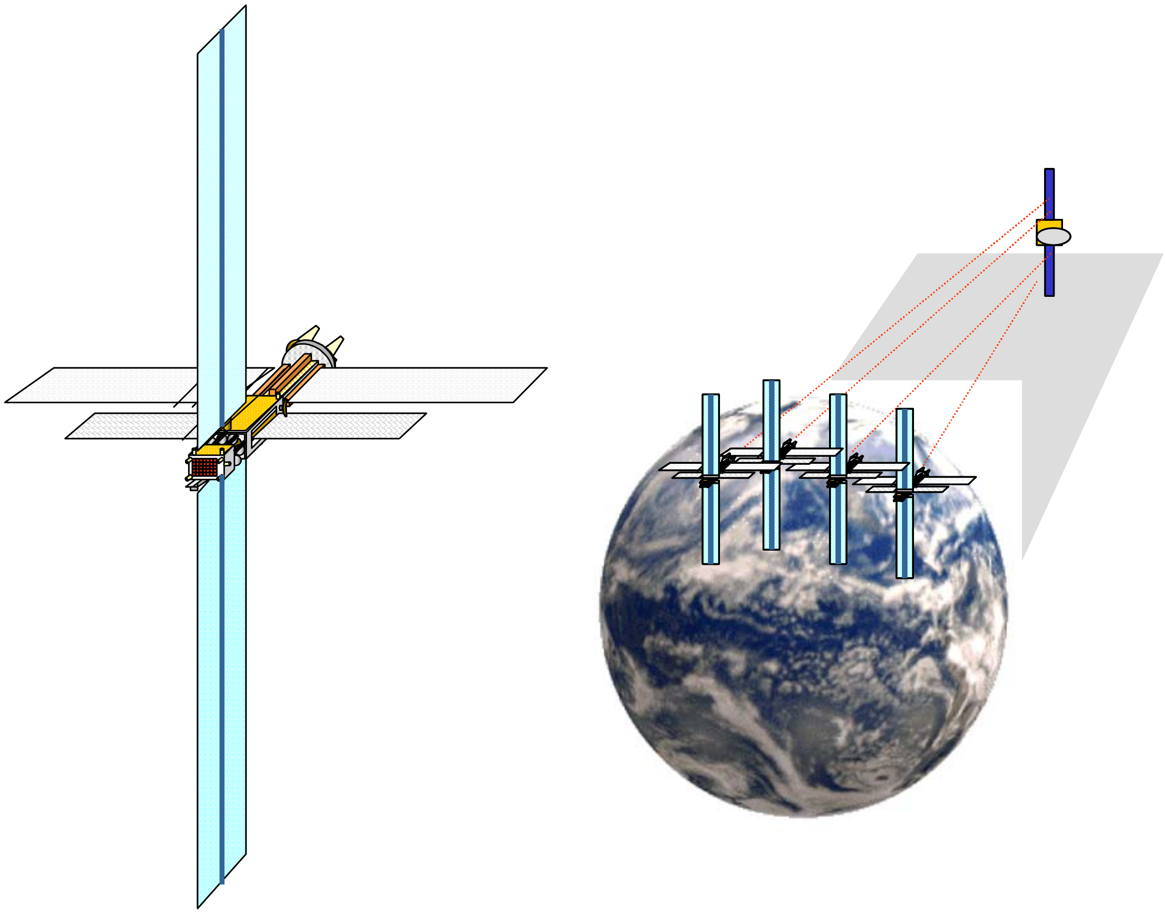


Figure 2.5/1: Illustration of SPS configuration and of SPS system

ON-BOARD SPS	SPS Solar Surfaces	Efficiency	Power distribution, DC-DC converter	Laser system	*Laser diode plug in* *Optical pump* *SHG generation*
	176 m²	**0,1487**	0,85	0,175	

(4 SPS) **Generated power: 70 kW** → Power @ laser system input 57 kW → **Emitted power 10 kW**

TRANSMISSION
4 telescopes, 2 m diameter typically
Distance: 45000 Km
Spot diameter on target: 14,6m
Power density @ target level: ~ 240 W/m²

RECEIVER SYSTEM
Received power: 40 kW
Surface: 300 m²
Efficiency: 0,5
Power available to user: 20 kW

Figure 2.5/2: Summary of SPS system performances

2.6 SPS INTEREST

The objective of the SPS is to provide power to the telecommunications satellites during their eclipse phase, so that they are permanently receiving the needed level of energy from external source.

The interest of the SPS is then to provide a service to all the geostationary satellites, for a long lifetime. Providing a service, it is assumed that there will be a redundant SPS to ensure the service continuity in case of failure.

In that case, the SPS would have to provide power to several satellites simultaneously in eclipse. The maximum size of the eclipse arc is about 17°. Currently, some parts of the geostationary arc are populated with satellite separated by 1 or 2°. Thus, in the worst case, there could be up to 10 satellites simultaneously in eclipse, but with lower power needs (5 to 7 kW) than the study assumption. It can be assumed that up to 5 satellites of 20 kW will be in eclipse at the same time. The SPS would have then to provide several power beams to feed these satellites simultaneously.

The potential gain for the satellite of using the SPS services would be to avoid the batteries and associated BCDU (battery charge/discharge control unit). The relevant mass would be roughly 500 Kg. However, the satellites will keep a power storage capability to cover the risk of SPS failure and service interruption. The availability of a redundant SPS ensures that the interruption will be short. In that case, the satellite batteries would be sized for ensuring the satellite communication service for a very short period (few minutes).

In addition, the minimisation of the size of the batteries could lead to reduce slightly the size of the solar arrays, as there would be no need to recharge the batteries from the solar arrays. However, that represents a small part of the solar arrays.

With these assumptions, the potential gain due to SPS would be roughly 400 Kg of battery and associated items. The gain of using SPS would be a few M€ for launch, plus the batteries. The alternative of using SPS is, for the geostationary satellites, to keep the power subsystem as it is with solar arrays and batteries. From the user point of view, this potential gain has to be compared with the cost of the service. The potential gain does not appear cost effective.

In this application, the SPS would be operational only during the eclipse phase that means a few weeks around the equinoxes. It could be interesting to look for other potential secondary or complementary applications, for a SPS on sun synchronous orbit.

3 MISSION CASE 2: DARWIN

3.1 DARWIN APPLICATION AND NEEDS

The objective is to transmit power from a laser power source to several small free flyers located at a short distance.

The ESA's Darwin will be an interferometric mission in the mid Infrared (6-18μ). The current instrument concept is based on six free flying 1.5m telescopes arranged in a planar, hexagonal configuration. It will be operating at Lagrangian point L2, where there are minimal gravitational gradients. The solar pressure induces torques which have to be continuously compensated. The impacts of thermal emission from the satellite on the observations have to be reduced.

A possible configuration considers the use of a large sun shield to prevent the free flyers from having slightly different solar pressure and solar thermal effects. It is illustrated on Figure 3.1/1. It includes:

- A large, multi layer solar sail of 550 m diameter, which shields the whole constellation from the sun at once. It has its own thrusters and attitude control system to prevent the solar pressure from pushing the sail to the telescopes
- Six free flyers telescopes, equipped with solar arrays and a central beam combiner
- A wireless power transmission based on laser system

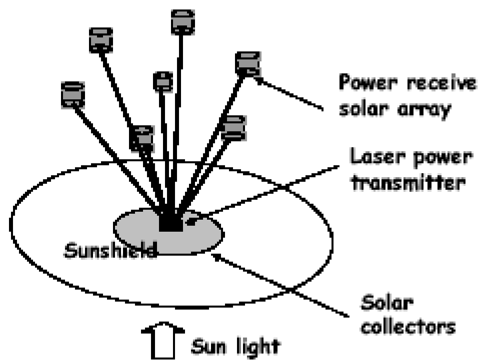


Figure 3.1/1: Illustration of Darwin configuration

The electrical energy is generated by a solar cell area located on the center of the solar shield. It is transmitted to the free flyers by a laser power transmission system, which feeds solar panels installed on each free-flyer. The laser pointing control is required as the attitude of the solar sail is by far less stable than the one of the interferometric array. The solar sail is pointed to Sun.

The main requirements are:

- Distance between the solar sail and the flyers: 200 m
- Distance between the flyers: 25 to 250 m

- Required electrical power (per flyer): 1,5 kW
- Maximum power density at the flyer solar panel level: 1 kW/m²
- The individual power flux has to be tuned to minimize the optical pressure differential torque

3.2 POWER TRANSMISSION SYSTEM

3.2.1 Laser system preliminary design

Darwin SPS requirements are different from the requirements of the other SPS. The distances between the target and the SPS are much shorter: a few hundreds of meters as compared to thousand of kilometres. The SPS laser solution will be then completely different

The proposed solution is based on non coherent laser sources. High power laser diode array coupled on multimode fiber laser is the ideal solution.

Such types of lasers are currently developed for welding application. The Figure 3.2/1 is depicting the Laser diode bar assembly

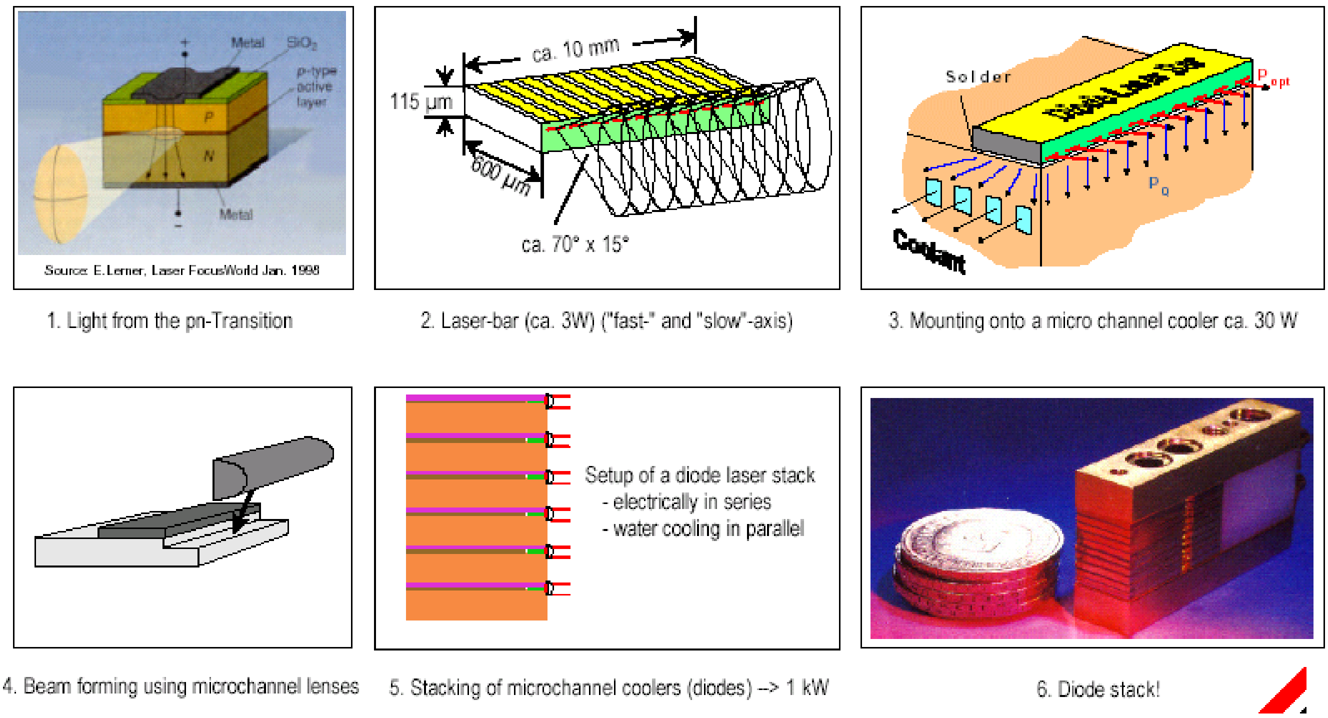


Figure 3.2/1: Laser diode Bar assembly (from [8])

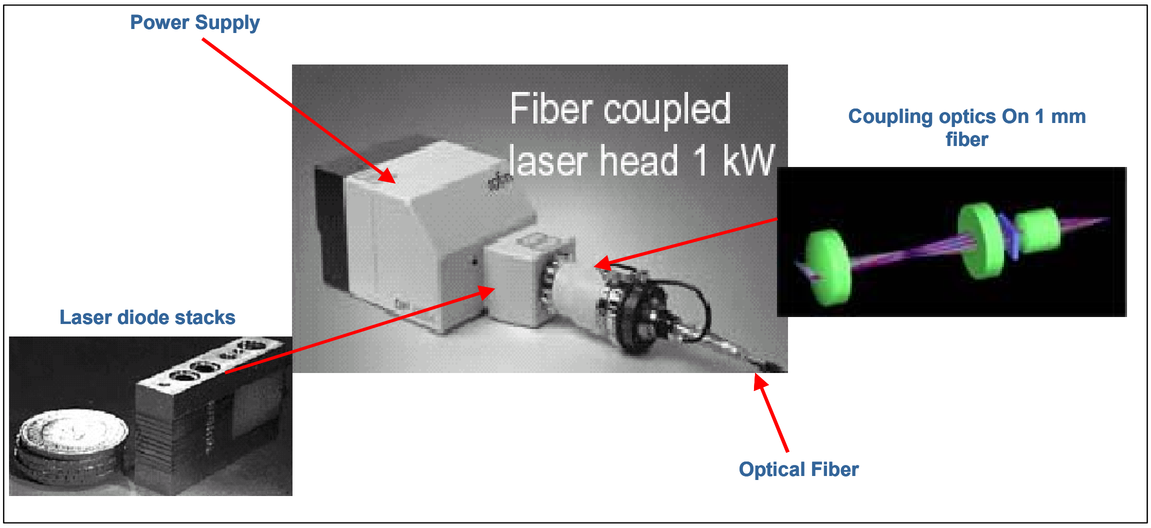


Figure 3.2/2: 1kW Fiber coupled laser (from [8])

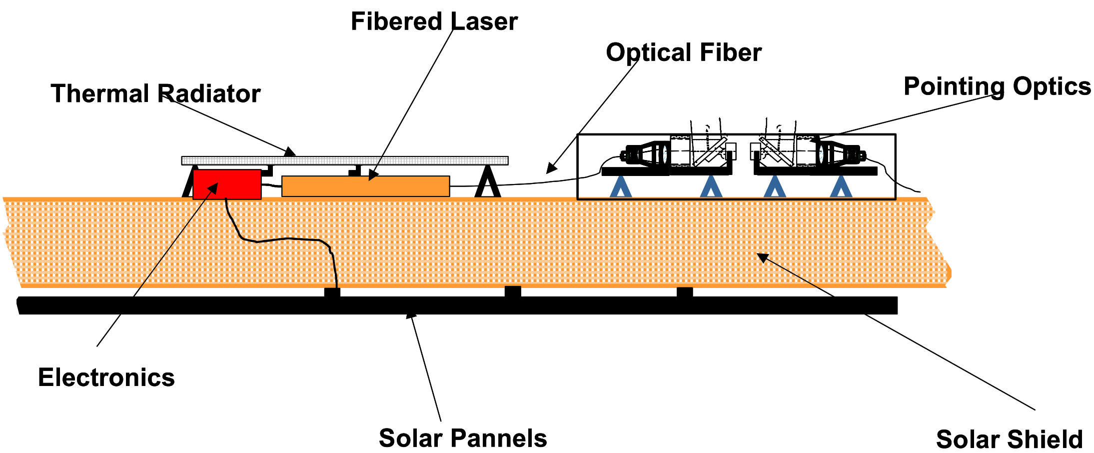


Figure 3.2/3: DARWIN SPS

The laser diodes are coupled on multimode fiber optics in fully integrated assembly. The core of multimode fiber is approximately 1mm diameter. 4kW CW output power were obtained with current silica core fiber. Higher power would be expected with new Hollow fiber. However such development is not fully completed.

The specifications of the laser for the Darwin application are:

– 4000 Watts CW

– λ = 808 nm

–Multimode Fiber coupled with 1mm fiber core

–Numerical aperture : 0.2

The fiber is coupled on a small telescope (0.2 m) and the beam is directed towards the free flyer thanks to a steering mechanism.

This proposed laser diode is an existing equipment which can be easily space qualified.

Seven laser assemblies are gathered on the bus supporting the sun shield. Each laser assembly is connected to its own solar panel on the sunny side and to radiative panel on dark side.

3.2.2 Mass and power budgets of the laser system.

The mass and the power budget of the 4 kW laser power system are given in the following tables.

Sytem laser mass	Mass in Kg
Telescope	3
Pointing system	2
Opto-mechanical assembly of laser	6,5
Mechanical structure	15
Electronics	45
Total	71,5

Table 3.2/1: Mass budget (for 1 laser)

Power flow chart laser		
Emitted power	4000	W
Fiber coupling efficiency	70	%
Laser diode plug in efficiency	50	%
Power generation system		
DC/DC converter efficiency	85	%
Power required at SA output	13445	W
solar cell efficiency	30	%
solar power density	1300	W/m^2
solar array size	34	m^2

Table 3.2/2: Power budget (for 1 laser)

The laser will require 70 m² radiative panels for one laser to ensure a correct thermal control.

3.3 POWER GENERATION SYSTEM

For one free flyer, the power assumptions on the SPS are:

- Emitted power: 4 kW
- Fibre coupling efficiency: 70%
- Laser diode plug efficiency: 50%
- DC/DC converter efficiency: 85%

This gives a power requirement of 13.4 kW (at the output of solar arrays) for one free flyer.

Assuming a solar cell efficiency of 30% and taking into account the Sun flux on the L2 Lagrangian point (1300 W/m²), the solar array, based on multiple-junction solar cells with small concentrators, has an area of 34 m² (see Table 3.2/2) and a mass of 52 kg.

For 7 free flyers, these figures shall be multiplied accordingly. That gives a power requirement of 94 kW for the laser supply. Assuming an additional 10 kW for all other systems of the SPS (data handling, AOCS, propulsion, thermal control ...), the solar array will have a total area of 267 m², and a total mass of 400 kg.

In term of configuration, a modular approach shall be preferred. The best way is to design independent systems per free flyer. This allows to adapt easily the system as more free flyers are ready for the mission.

In term of power supply, one PSS will supply one laser of one free flyer. The PSS architecture is illustrated on Figure 3.3/1.

Taking into account the environment (relatively constant solar flux) and the power consumption requirement, the best approach is a 100V regulated bus with Sequential Serial/Shunt Switching Regulator (S^3R).

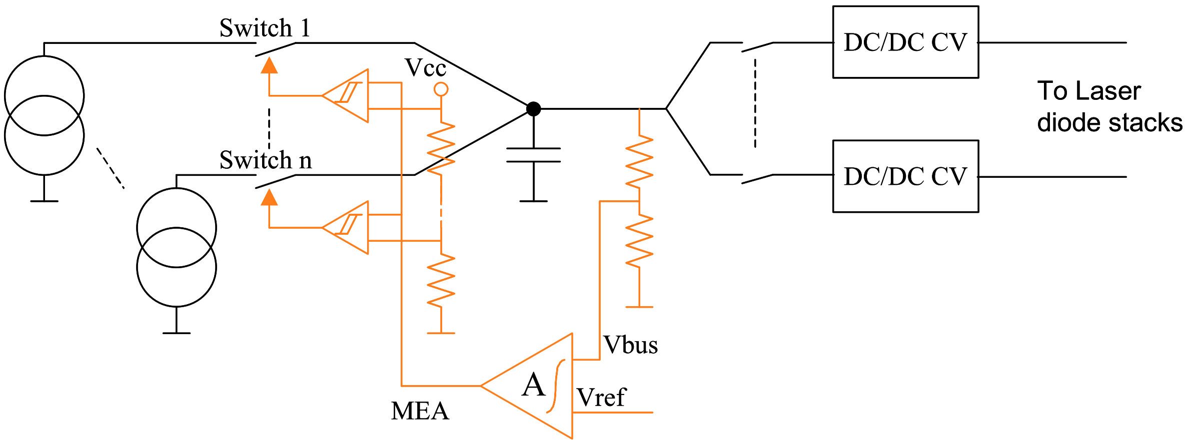


Figure 3.3/1: Architecture of power subsystem (for one laser)

SA switches can be sized for a voltage drop lower than 0.5 V by paralleling Mosfets, thus leading to a conversion efficiency of 99.5%.

SA voltage matching is also very good because of the relatively constant solar flux (±3.5%) and of the regulated bus voltage.

3.4 PERFORMANCES AND SPS PARAMETERS

The performances of the SPS system are depicted in the Table 3.4/1.

The solar panel on the free flyer that collects the laser transmitted energy has a size of 1m x 1m. The telescope has a diameter of 0,2m, and the beam spot at the target level has a maximum diameter of 0.5m. The resulting pointing accuracy is in order of 1.1 mrad, which is easily achievable. The power available at each user is 2240 W.

Parameters	Value	Unity
Input:		
Distance laser to constellation	200	m
Distance Minimum Hub to FF	25	m
Distance Maximum hub to FF	125	m
Size of the Square solar pannel of FF	1	m
Efficiency of the solar pannel	70	%
Useful part of the solar pannel	100	%
Emitter diameter	0,2	m
Couplig efficiency of optics	80	%
Fiber Core diameter	1	mm
Emmitted Wavelength	808	nm
Emitted Optical power	4000	W
Fiber NA	0,2	
Emitted angle at fiber level	0,40	radiant
Emitted angle at fiber level	23	Degres
Output:		
Emmitted Optical sytem		
Focal length	0,5	m
Angular Pointing	25	Degres
Pointing Accuracy	1,1	mrad
Pointing Accuracy	0,06	Degres
FF Array parameters		
Minimum distance Laser to FF	202	m
Maximum distance laser to FF	236	m
Received power at FF		
Max Dim. of The Focused Spot at FF solar pannel	0,50	m
Electrical Power at each FF	2240	W

Table 3.4/1: SPS system parameters

4 MISSION CASE 3: VEHICLE TO MARS

4.1 MISSION NEEDS AND CONSTRAINTS

This mission case is representative of interplanetary missions.

The objective is to provide power to a vehicle during its transfer from Earth to Mars.

This mission case is characterised by the very long and evolving distance between the SPS and the target. It is illustrated on Figure 4.1/1.

The interplanetary vehicle requirements and constraints are as follows:

- The needed power is 0,5 kW for small satellites, or 50kW for large vehicles for preparing manned infrastructure.
- The vehicle is on an Earth-to Mars transfer trajectory. It is launched when Mars and Earth are in adequate conjunction and the transfer duration is assumed 6 months. The vehicle can remain around Earth for assembly if necessary and preparing the boost for injection into Earth-to-Mars transfer orbit. This period can last several days. The transfer itself is assumed to last 6 months. At the arrival to Mars, the vehicle either performs a direct entry (hours), or is captured in orbit around Mars before performing entry. The SPS will provide power during the Earth-to-Mars transfer trajectory.

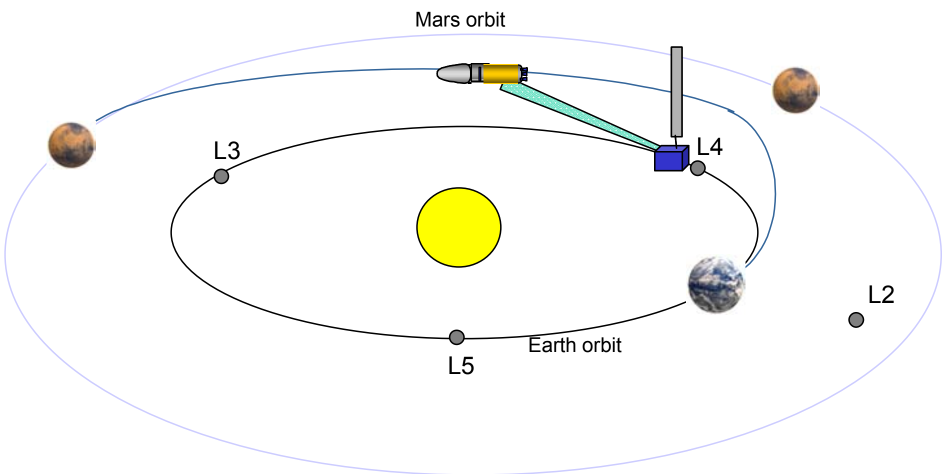


Figure 4.1/1: Transfer to Mars mission

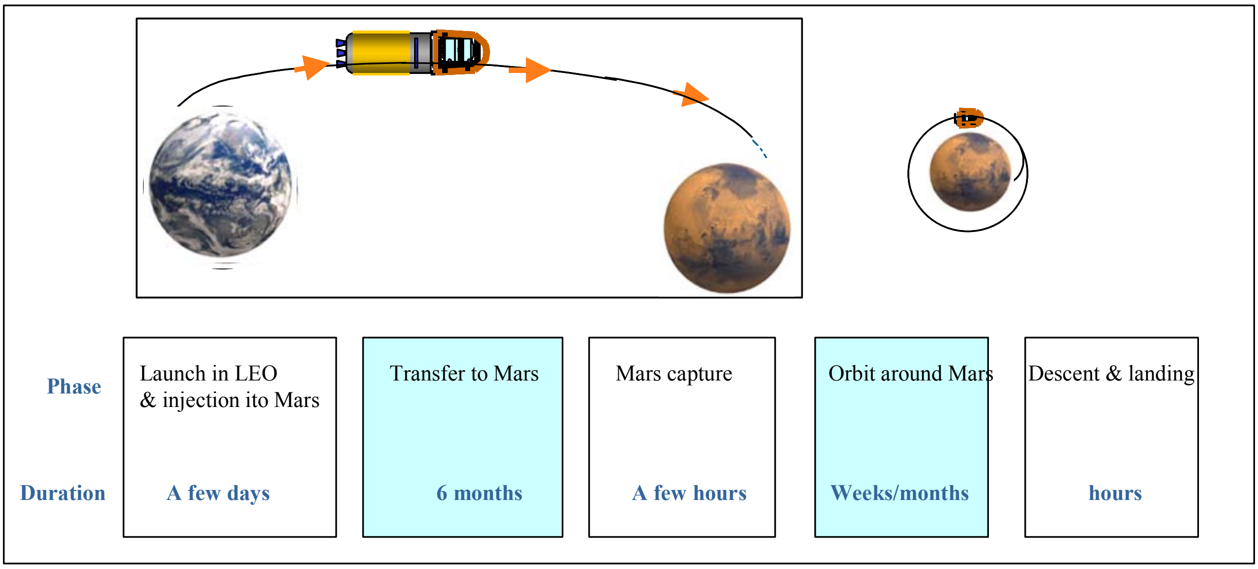


Figure 4.1/2: Illustration of main phases of the mission to Mars

4.2 SPS POSITIONING

The possible SPS locations (see Figure 4.1/1) are:

- At Lagrangian point L4 or L5. It can have its solar panels or collectors permanently facing sun.
- At Lagrangian point L2
- In orbit around Earth, typically on equatorial
- In orbit around Mars

A rough analysis of the evolution of the distance between the SPS located at any of these points and the vehicle has been done, with simplifying assumptions: circular orbits for Earth and Mars with respective radius of 1 AU (Astronomical Unit) and 1.52369 AU, and an Hohmann transfer orbit for the vehicle. The Figure 4.2/1 illustrates the evolution of the respective positions of both vehicle and candidate SPS locations along the transfer of the vehicle to Mars.

The evolution of the distance between the vehicle and the possible SPS locations is given in Figure 4.2/2 as function of the vehicle position (true anomaly), with an indication of the time at which the vehicle position is reached.

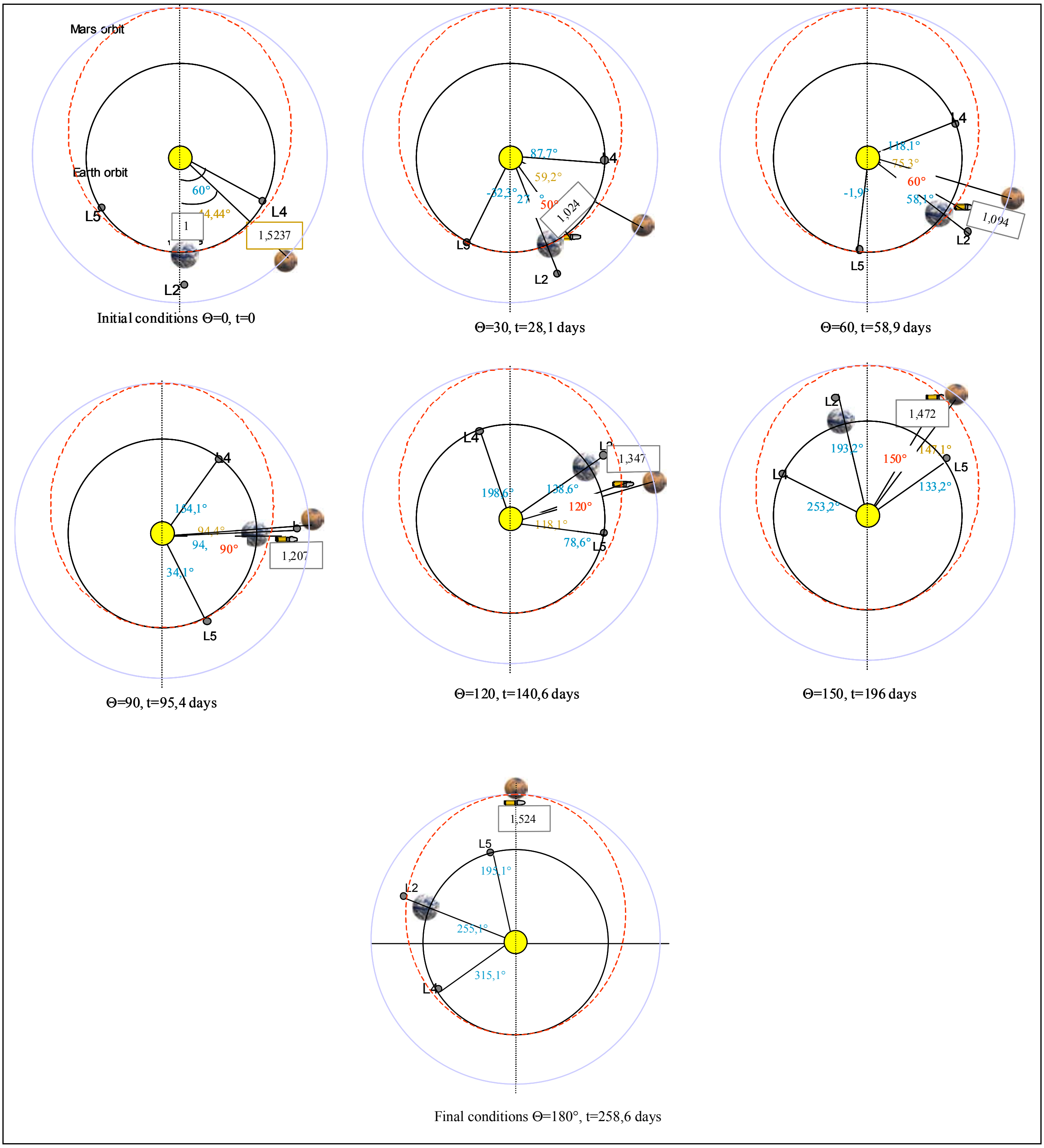
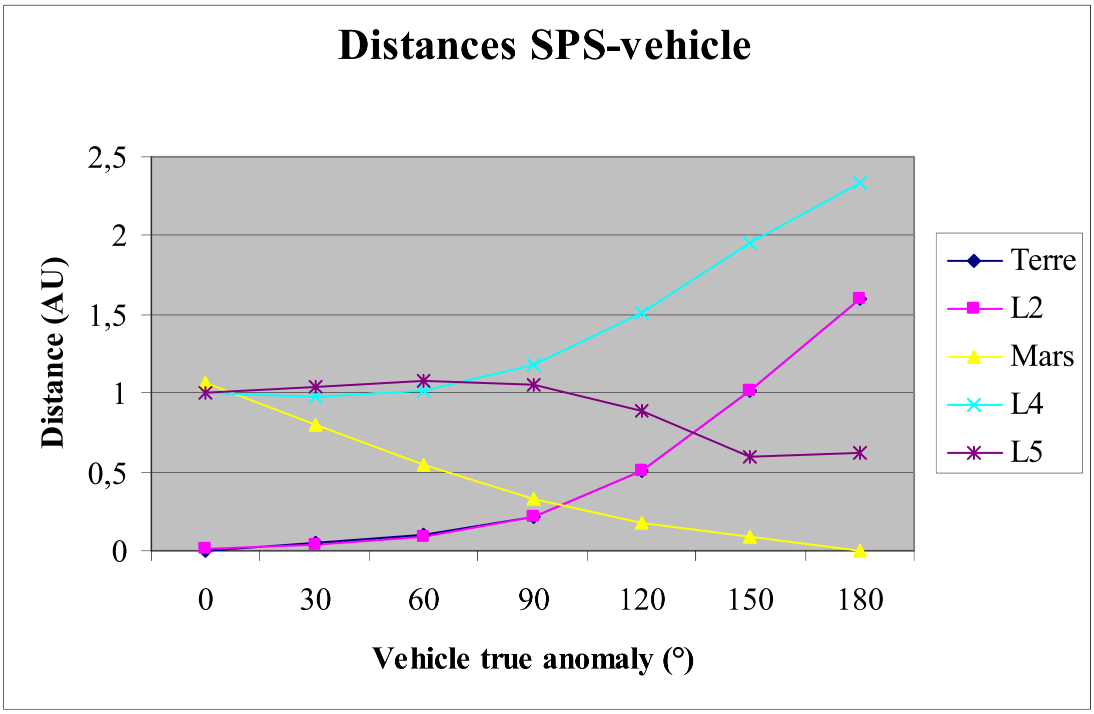


Figure 4.2/1: Evolution of the respective positions of vehicle and candidate SPS locations during the vehicle transfer to Mars



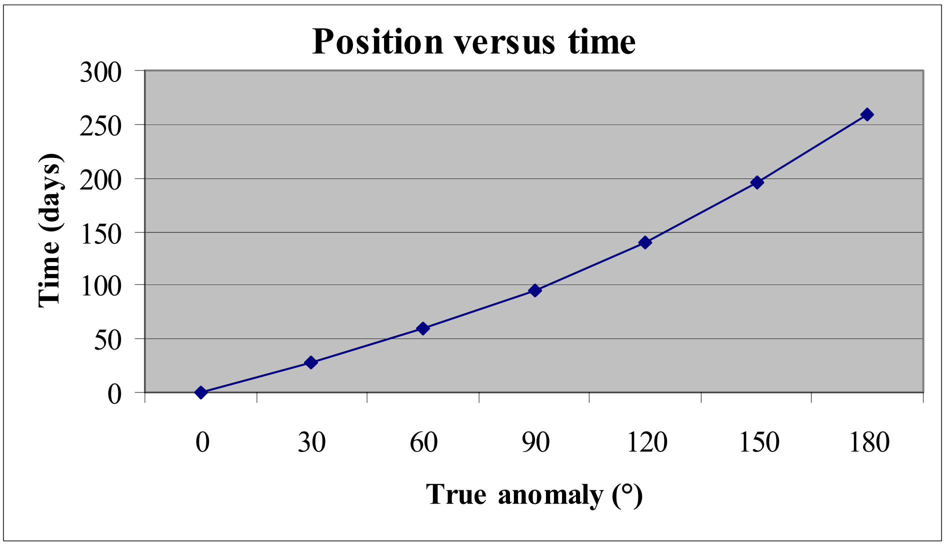


Figure 4.2/2: Evolution of the distance between the vehicle and the candidate SPS positions

4.2.1 Lagrangian point L4/L5

If located at L4, the distance between SPS and vehicle will increase from 1 AU up to about 2,34 AU when the vehicle will arrive at Mars. Indeed, L4 has initially a higher true anomaly than the vehicle, and is rotation faster. Only interest of L4 is the continuous view of the Sun by the solar surfaces.

L5 is located symmetrically to L4 with respect to the Earth. It starts behind the vehicle in terms of true anomaly (-30° against 0°). Therefore, the distance SPS-vehicle remains around 1 AU (maximum 1,08) during the first hundred days, and then decreases down to about 0,59 AU when L5 is at the same true anomaly as the vehicle, and starts to increase. Final distance is about 0,62 AU. This location is more interesting than L4 as it allows also a permanent view of solar surfaces on the Sun, and leads to lower distances than L4.

4.2.2 In orbit around Earth

If the SPS remains in orbit around Earth, the distance to the vehicle starts from a close distance typically a few 10000 Km if in geostationary orbit, up to 1,594 AU when the vehicle arrives at Mars. Besides, being in orbit around Earth, there will be periods of occultation of the SPS-vehicle visibility, and periods of eclipse for SPS. The maximum duration of each period is 1h20, so that power storage system could be sized on board the vehicle to cover these periods. The angle between the SPS-vehicle direction and the SPS-Sun direction varies between –70 and + 70°, and the beam steering angle will vary in the same range.

4.2.3 L2 Lagrangian point

The distance SPS-vehicle and relative positions are roughly the same as in the case SPS around Earth. The distance is slightly lower. Difference is that the SPS will permanently view the Sun and the vehicle. Neither occultation nor eclipse. The drawback with respect to Earth orbiting position is the launch and transport to L2 point.

4.2.4 In orbit around Mars

If the SPS remains in orbit around Mars, or instance in areosynchronous orbit, the distance to the vehicle decreases from 1, 07 AU down to 0. There could be some periods of eclipse for the SPS or occultation of the power beam, but for a short period, and the power storage system of the vehicle could be sized for these periods. The angle between the SPS-vehicle direction and the SPS-Sun direction varies between 40 and –60°, and the beam steering angle will vary in the same range. Nevertheless, the SPS located close to Mars will receive a lower energy density from the Sun than the SPS located close to Earth or in L2 or L5.

4.2.5 Synthesis

The key parameter for the sizing of the power transmission to a vehicle to Mars, or to an interplanetary vehicle, is the distance between the SPS and the vehicle.

When comparing the candidate SPS locations, the lowest maximum distances, which are the sizing parameter, are got with the SPS in orbit around Mars or at L5. The location in orbit around Earth or at L2 is interesting until the last part of the transfer (thus until 200 days).

However, in order to minimize the impacts on the SPS power generation system, the Sun energy density received on the solar surfaces has to be considered also. The size of the surfaces is roughly proportional to the energy density on one part, and to the square of the SPS-vehicle distance on another part. Taking into account an energy density of 500 W/m² close to Mars, and 1300 W/m² close to Earth and at L2 and L5, an equivalent distance (all other parameters being identical) could be computed: it gives for the Mars SPS positioning an equivalent sizing distance corresponding to an energy density equivalent to the Earth one. This is illustrated on Figure 4.2/3, which allows to compare the SPS candidate positioning with respect to the size of the SPS solar surfaces. The most interesting positioning becomes L5, with a maximum sizing distance of 1, 08 AU.

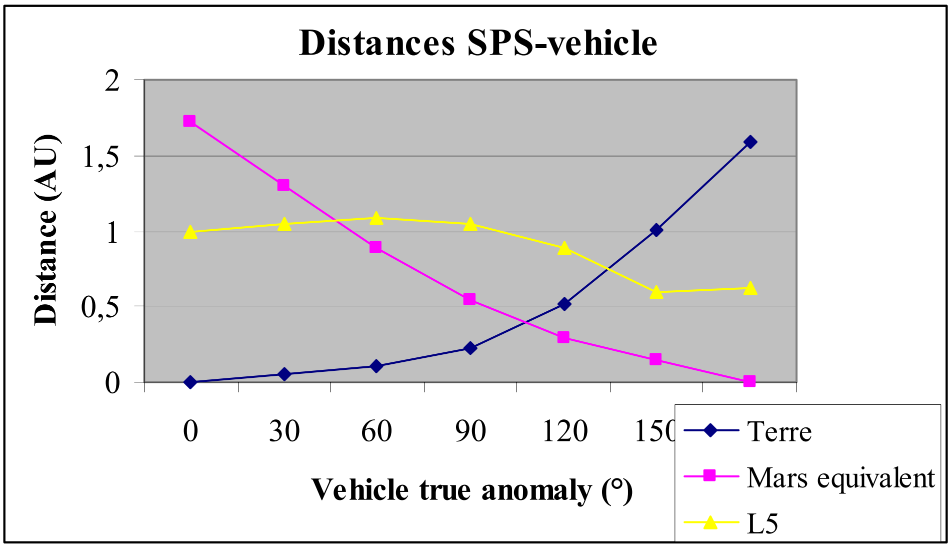


Figure 4.2/3: Equivalent distance between SPS and vehicle, taking into account sun energy density

Anyway, that remains a very important distance, about 162 Millions km, for the sizing of the power transmission system.

A solution to reduce this distance is to combine a SPS close to Earth (or at L2) and another one in orbit around Mars. In that case, the handover between the SPS around Earth and the SPS around Mars would occur after about 121 days of vehicle transfer, with a maximum distance between Earth SPS and vehicle of 0,37 AU, that means 55,5 Millions km. The distance between Mars SPS and vehicle would be lower, 0,233 AU or 35 Millions km, but this handover would be the optimum with respect to SPS solar surfaces.

Besides, the SPS around Earth (or at L2) and around Mars could be used for other applications.

Another solution to reduce the distance and have only one SPS is to have the vehicle powered by the Sun energy at the beginning of the transfer, and then by the SPS power when the sun energy density at the level of vehicle solar arrays becomes too low (for instance below 1000 W/m²). In that approach the vehicle solar arrays would be sized for a minimum solar energy density (thus a gain in surfaces with respect to sizing the solar arrays for the solar energy density in vicinity of Mars), and the SPS located in orbit around Mars would provide the same energy density on the solar arrays. If we consider for instance a value of 1000 W/m², the vehicle divides by almost a factor 2 its solar arrays surface, and the Mars located SPS will provide power from a distance of 0,435 AU, that means 65 Millions km, about 76 days after the vehicle has left the Earth.

SPS in L5	162 Millions km	- One SPS - High distance - No occultation, no eclipse - Solar energy density 1300 W/m²	High power density allowing reduced receiver surfaces Receiver surface movable for pointing
One SPS in Earth orbit or at L2 **One SPS in Mars orbit**	55 Millions km 35 Millions km	- Two SPS - Lower distances - Occultation or eclipse - Solar energy density 1300 W/m² for SPS close too Earth, 500 W/m² for SPS close to Mars - Possible applicability to other missions	High power density allowing reduced receiver surfaces Energy storage capability for occultation/eclipse periods Receiver surface movable for pointing
One SPS in Mars orbit	65 Millions km	- One SPS - Medium distance - Occultation or eclipse - Solar energy density 500 W/m² - Possible applicability to other missions	Limited power density due to utilisation of sun energy on a part of orbit Higher solar arrays surfaces Energy storage capability for occultation/eclipse periods Receiver surface movable for pointing

Table 4.2/1: Summary of comparison of candidate SPS positioning

4.3 POWER TRANSMISSION SYSTEM

The power transmission system will be based on laser system due to the high distances. Indeed, the utilisation of RF systems would lead to the use of relays to forward the power beam.

The use of a laser transmission system would be in line with the possibility for the vehicle to rely on the sun power on the first part of the travel, and on the SPS power beam on the second part.

4.3.1 Interplanetary Missions: Driving parameters

The main driving parameters is obviously the distance: 65 millions kms.

This system is very challenging and the only solution will be to use a large number of phase emitting systems in order to concentrate the laser power in a very narrow beam. The lasers are phased thanks to a common master laser (Master) which part of the output power is diverted towards the other lasers (Slaves). On each slave laser an optical phase control system (piston control) ensures the co-phasing of the laser and of the fine pointing of the array output beam. The coarse pointing is performed by the pointing system defined in chapter 2.3.2.2. The final pointing accuracy is in the range of nanometre (3 order of magnitude of the SILEX pointing accuracy).

The array is composed of 30 Slave laser satellites equally distributed in a 150 m array. Each individual laser system is identical to the laser system defined in chapter 2.3. The emitting wavelength is 532nm.

The control of the formation flying of such constellation arrays will be very complex.

We consider that 1 kW can be fed by the vehicle (small or medium satellite). If more power has to be fed the complexity of the array will increase accordingly (1500 Slave satellites operating in formation flying).

4.3.2 SPS parameters

The SPS parameters are given in the following table.

Laser system parameters		
Laser power	10000	watt
Number of slave laser	30	
Operating Wavelength	532	nm
Array dimension	150	m
operating distance	65000000	km
Diameter of the Spot on target aera	281,3	m
Efficacité de l'array	50,0	%
50% Loss pointing Accuracy	2	nrad
Surface of the PV target area	1000	m²
Efficiency of the PV Cell	50	%
Illumination at target level	2,41	W/m²
Generated electrical power	1207,2	W

Table 4.3/1: SPS parameters

4.4 SPS CONFIGURATION

The SPS system is composed of a constellation of satellites with one master laser satellite and TBD slave laser satellites as illustrated on Figure 4.4/1.

For example, a constellation of 30 slave satellites, each having a 10 kW laser operating at 0,532 µm with a 2m telescope, leads to the performances given in Figure 4.4/2.

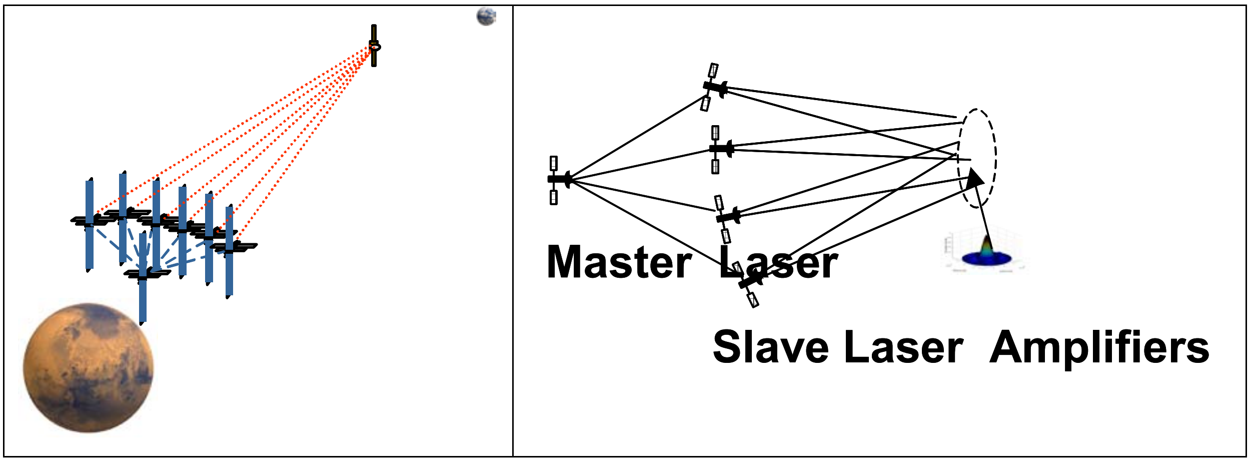


Figure 4.4/1: Illustration of SPS system for power delivery to interplanetary vehicle

ON-BOARD SLAVE SPS
Generated power: 70 kW
Wavelength: 532 nm
Efficiency: 0,1487
Emitted power: **10 kW**
Mass budget: ~ 21 t

TRANSMISSION
30 slave laser satellites
Equivalent array size: 150 m
Overall emitted power: 300 kW
Distance: 65 M km
Spot diameter on target: 281m
Power density @ target level: ~ 2,4 W/m²
Loss due to pointing accuracy: 50%

RECEIVER SYSTEM
Received power: 2,41 kW
Surface: 1000 m²
Efficiency: 0,5
Available power: 1,2 kW

Figure 4.4/2: Example of performance of SPS system (case of 30 satellites)

Each satellite has to generate 70 kW at the input of the DC-DC converter. That leads to a solar array surface of about 450 m². The required receiver surface (solar array on the vehicle) is 1000 m², and the power density at target level is only 2,4 W/m² that is too low with respect to the solar energy density. The power available at user interface is 1,2 kW, and the end-to-end efficiency is in order of 0,06%. An increase of the array diameter would be necessary in order to get a higher and more attractive power density.

4.5 SPS INTEREST

The key problem of the interplanetary vehicles is the drastic decrease of the solar energy density when removing from the Sun. Thus, in case of vehicle to Mars, the sun energy density is about 1300 W/m² (sizing case) in the vicinity of Earth, while it is only 500 W/m² in the vicinity of Mars. The vehicle solar arrays (or solar surfaces) have to be sized for the worst case, that means the mars vicinity case. At first approximation, the size of the solar surfaces is proportional to the solar energy density. For travel farther than Mars, other source of energy would be necessary due to the mow solar energy density. For instance RTG are already used. In case of mission to Mars, other source of energy could also be necessary for vehicle requiring a high level of power (tens of kW for instance).

The interest of the SPS is to provide an adequate and constant power energy density on the vehicle. It could be 1000 W/m², or more (at least significantly more than 500 W/m²). That would allow using photovoltaic cells while minimizing the size of the solar surfaces and associated mass.

Nevertheless, the large distances of transmission are penalizing for the SPS sizing.

For powering the vehicles travelling towards Mars, several candidate locations are possible for the SPS, but all leading to very high distances. Whatever the option (two SPS located in vicinity of Earth and Mars, or a single SPS in orbit around Mars), the distance is in order of 55 to 65 Mkm.

The result is a very low efficiency of the system (less than 1%) and also a low power density at the receiver level. In the example, only 2,4 W/m² is achieved at receiver level, which makes it not interesting as compared to using directly sun on the solar arrays. Large receiver surfaces are subsequently necessary. Besides, a high pointing accuracy is required. That could be improved by increasing the virtual diameter.

However, it appears that a very complex SPS system is made necessary to provide power to vehicle travelling to Mars, and that there will not be a high number of vehicles to be powered. Besides, the proposed SPS positioning is not necessarily applicable to vehicle travelling towards further planets (Saturne, Jupiter, etc). Positioned the SPS further than Mars would raise the problem of solar energy density at very long distance from the Sun, or the implementation of SPS relays.

5 PLANETARY MISSIONS- ELEMENT ON MARS SURFACE

5.1 MISSION NEEDS AND CONSTRAINTS

5.1.1 Elements on Mars surface

Two types of elements on Mars surface can be considered:

- ***Small automatic rovers***

(from NASA)

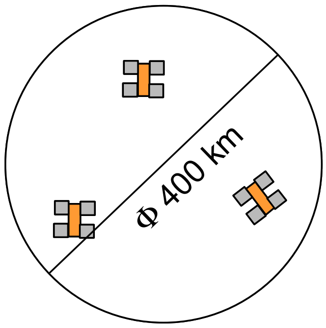


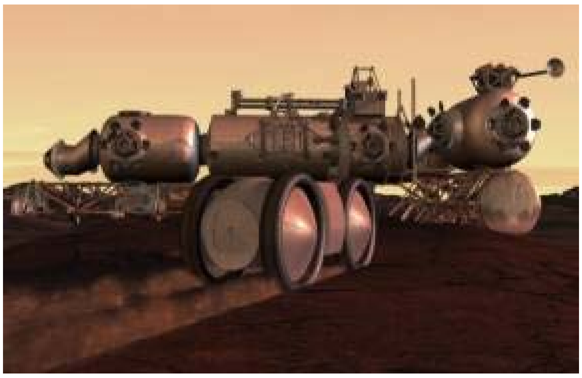
Figure 5.1/1: Small automatic rovers

A set of 2 or 3 rovers is assumed to be implemented in the same area. Each rover has an autonomy of 200 km and a velocity of about 10 cm/s. The SPS has then to provide power to rovers located in an area of 400 km diameter.

The rover has several modes of utilisation: operational modes (locomotion, analysis, etc) and waiting mode (minimum power consumption). Its power need is about 500 W in operational mode, and 50 W in dormant mode. Its receiver surface is limited in size to a maximum of 3 m. Its mass is up to 500 Kg.

- ***Large infrastructure (Mars base)***

The Mars outpost is implemented on the Mars surface and is composed of several elements. It is a manned base, so that a crew could be present. This base would be implemented on an area of about 1 km diameter. The SPS receiver antenna could be located at a safe distance from the base (typically 1 to a few km) to avoid the power beam on the base. There is no constraint on the size of the receiver antenna, except the transport, installation and deployment feasibility. The Mars base is fixed on the Mars surface. Its power need is 100 kW permanently.



(from S51 "European Mars Mission Architectures" study, 2002)

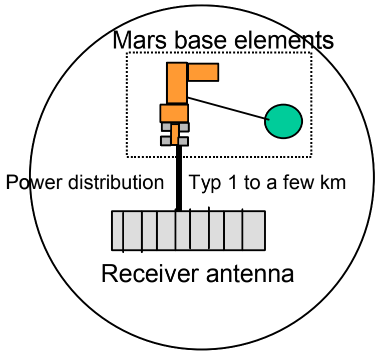


Figure 5.1/2: Mars base

5.1.2 Mars environment

The Mars environment, and especially the lower atmosphere and the winds, have impacts on the power transmission between the SPS and the Mars element.

5.1.2.1 Atmospheric composition of lower atmosphere

The elements included in the lower atmosphere are as follows:

Carbon dioxide (CO2):	95,32 %
Nitrogen (N2):	2 ,7 %
Argon (Ar):	1,6 %
Oxygen (O2):	0,13 %
Carbon monoxide (CO):	0,07 %
Water vapor (H2O):	10-1000 ppm (variable)
Neon (Ne):	2,5 ppm
Krypton (Kr):	0,3 ppm
Xenon (Xe) :	0,08 ppm
Ozone (O3) :	0,04-0,02 ppm (variable)

5.1.2.2 Surface pressure

The pressure at the surface of Mars is about 6 to 7 mbar

5.1.2.3 Winds

The wind velocities near the surface range typically from 2 to 10 m/s.

Velocities during a dust storm range from 17 to 30 m/s with a maximum that can exceed 40 m/s.

5.1.2.4 Atmospheric dust

The atmospheric dust is the major absorber of the solar irradiation. The mean size of the dust particles is about 1.5 μm, which is in the same order of magnitude than the sunlight wavelength range.

The dust particles could degrade the performance of the photovoltaic solar cells, as well as of optical sensor. The Material Adherence Experiment mounted on the Pathfinder Sojourner has measured the deposition of dust during 36 sols: a degradation of the solar array performance at a rate of 0,28% per day was recorded. This degradation could be reduced down to 5% of that rate by angling the array such that the wind or the gravity removes the dust, in combination with other countermeasures.

Nevertheless, wind velocities of about 20 to 30 m/s are necessary to raise typical dust particles from the surface.

5.1.2.5 Dust storm

There are three types of dust storms: local, regional and global.

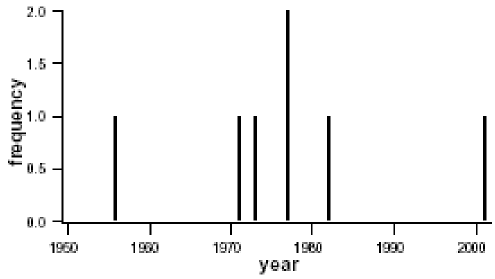
- **Local dust storm:** they concern an area of typically less than 2000 km. 771 local dust storms have been observed in 1999, 335 in the northern hemisphere, 436 in the southern one.
- **Regional dust storms:** they cover an area of more than 2000 km. 12 have been observed in 1999, 8 in the northern hemisphere, 4 in the southern one. They lasted more than 2 days.
- **Global dust storms:** they are planet encircling dust storms. They are rather sparse, as illustrated on Figure 5.1/3.

Most dust storms occur during perihelion period. Local dust storms occur near the polar cap edges and at mid latitudes in both hemispheres.

The occurrence of the dust storms is difficult to predict. According to a NASA model, illustrated on Figure 5.1/4, it could be assumed on the site of the target (Mars element):

- Two global dust storms every three years, lasting 100 days with a peak of optical depth of 6.
- Two regional or local dust storms, lasting TBD days (up to 100 days), with an optical depth increased from 0.5 to 3.

HST observation of the 2001 global dust storm



Number of observed global dust storms

Figure 5.1/3: Illustration of global dust storm (from S51 "European Mars Mission Architectures" study, DLR TN on Martian environment constraints, 2002)

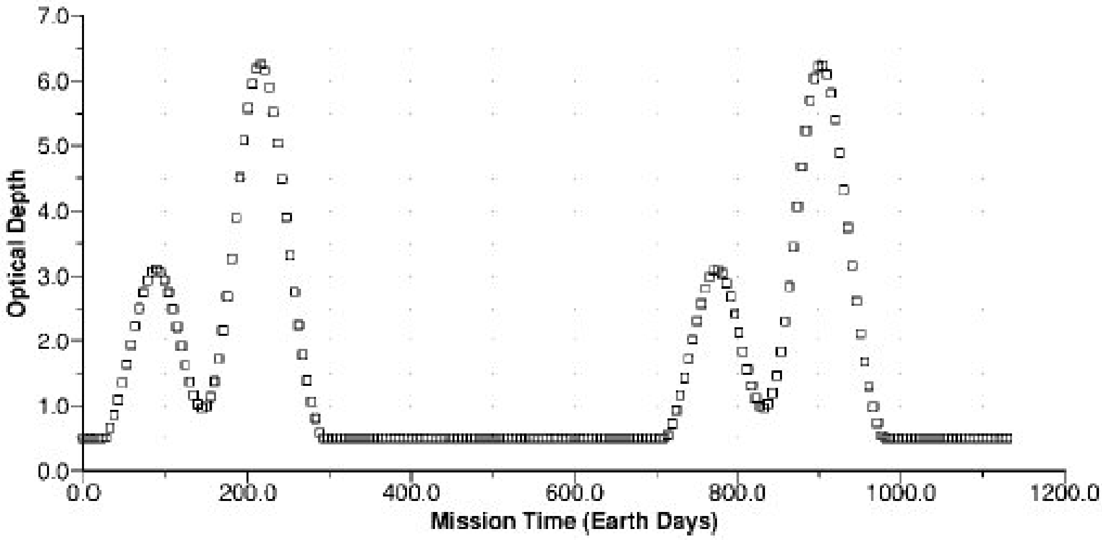
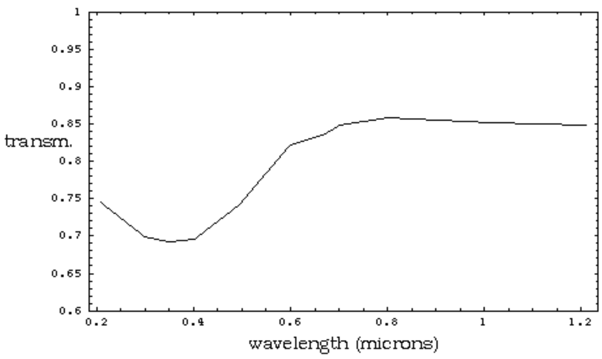


Figure 5.1/4: Dust storm occurrence assumption (from S54 study, Future Power Systems for Space Exploration, TN4, 2002)

5.1.2.6 Mars surface solar spectrum

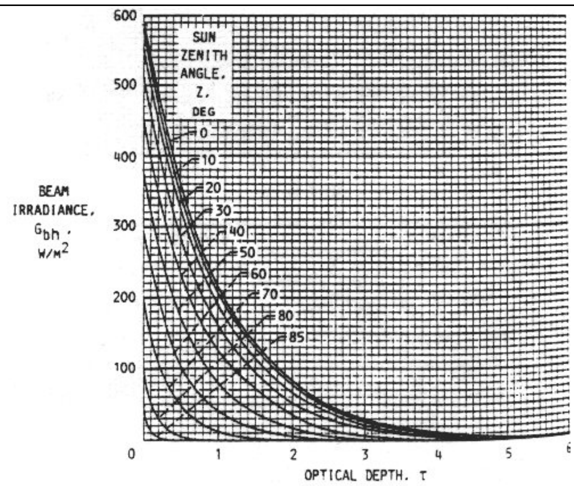
The solar spectrum on Mars, shown on the Figure 5.1/5, is different from that in Low Earth Orbit. Generally, the Mars spectrum has a lower blue content. Atmospheric solar energy transmission can be reduced by up to 30% at 0.35µm (in the blue wavelength) on a clear day. The spectrum contains a larger fraction of red light. The optical depth may vary from 0.5 (blue sky) to 3 (typical dust storm). It could reach peak of 6 in case of severe dust storm (the OD of 6 is the maximum recorded opacity).

The Figure 5.1/5 gives the variation of the direct solar energy as function of optical depth and sun zenith angle. It comes from data from Viking 1 and 2. It shows that the expected solar flux is typically below 350 W/m² in clear conditions (OD=0.5), and below 100 W/m² during a dust storm.



Mars spectrum

(from S54 study, Future Power Systems for Space Exploration, TN4, Qinetics, 2002)



Direct beam irradiance variation with optical depth and sun zenith angle, Viking lander 1 @ 22.3°N

(from S54 study, TN4 – Qinetics & [14])

Figure 5.1/5: Mars surface solar spectrum

5.1.2.7 Physical and orbital parameters

The main physical and orbital parameters are:

- Mass: 6,4 10^{23} kg
- Radius: 3397 km
- Gravity: 3,7 m/s²
- Mars orbit: perihelion: 206,6 10^6 km
  Aphelion : 249,2 10^6 km
  Period : 687 Earth days
- Mean solar day: 24h 39mn 6s
- Gravitational parameter: 42830 km³/s²

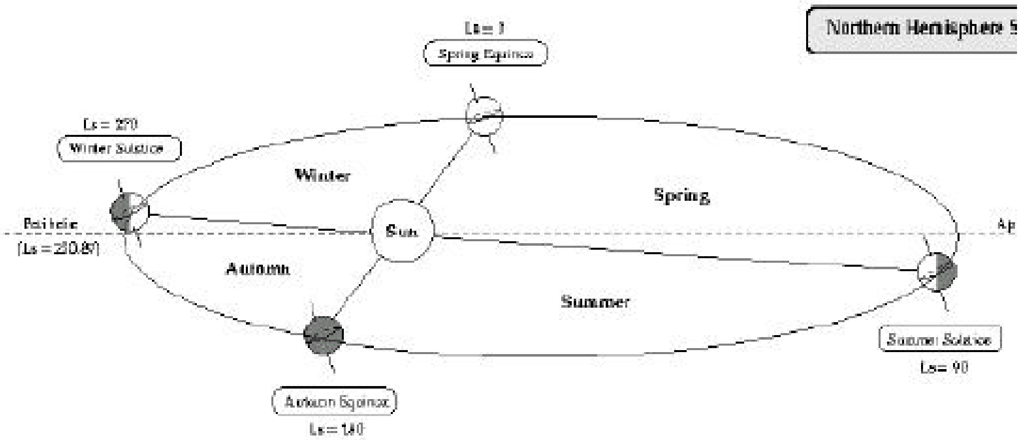


Figure 5.1/6: Illustration of Mars orbit

5.2 SPS POSITIONING.

The SPS could be located in:

- Areosynchronous orbit above the user area, at 17000 km altitude
- Low Mars orbit, equatorial or sun synchronous, at an altitude around 500 to 1000 km

5.2.1 Mars areosynchronous orbit

The SPS remains permanently above the area in which would be the rovers or the infrastructure, as illustrated on Figure 5.2/1.

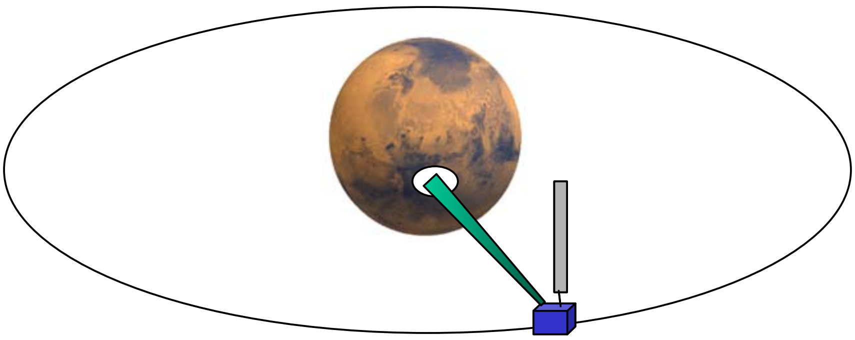


Figure 5.2/1: Illustration of Mars stationary orbit

5.2.1.1 Distance SPS-target

It varies between 17 000 km and 20600 km as function of the latitude of the element on Mars. Assuming the Mars rovers or infrastructure close to the equator, the distance is about 17000 km.

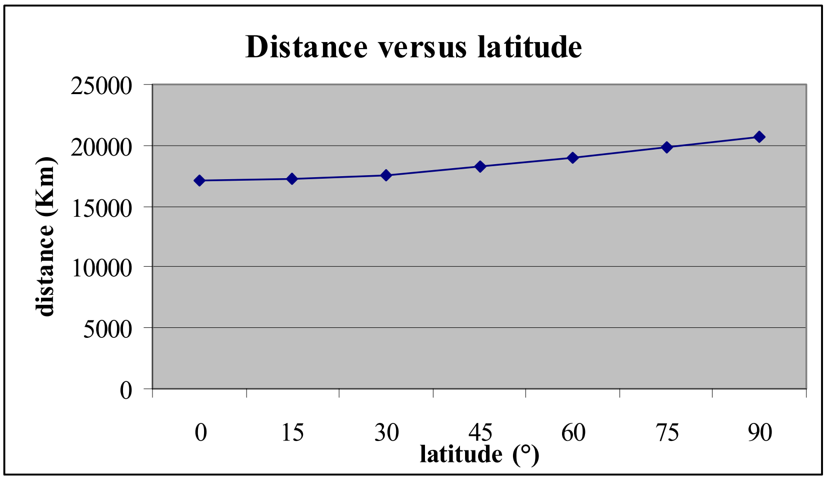


Figure 5.2/2: Distance SPS-target as function of the latitude of the target

5.2.1.2 Availability of the power beam

Eclipse

The SPS will be in eclipse for a short time around equinoxes, with a maximum duration (about 1h20) at equinoxes. During the eclipse phase, the SPS will no more transmit power to the Mars element. This is the driver for the sizing of the power storage system of the Mars elements. In case of rover, it can be switched in dormant mode. Alternatively, two SPS can be considered.

Geometrical visibility

Being on a stationary orbit, the SPS remains permanently in visibility of the target

Availability

The target could receive power quasi permanently. The only period of non-supply of power is the eclipse time, which is short and only at certain periods of the year. Only one operational SPS would be sufficient.

This partial conclusion covers only geometry and eclipse, thus is independent of the selected technology. Additional restriction will appear according to the technology (dust storm, etc).

5.2.1.3 Beam steering needs

Case of Mars infrastructure

The power beam remains quasi fixed for the Mars infrastructure. The receiver system, based on rectenna, concentrators or other, can be implemented close to the infrastructure (at hundred meters, depending on the beam diameter). The power can be distributed to the Mars infrastructure elements from this receiver system. That avoids to have a high energy density on the infrastructure elements, especially manned elements. The receiver can be fixed and oriented towards the SPS.

Case of rovers

Each rover shall have its antenna/concentrator. Due to their manoeuvering capability (autonomy and velocity), the power beam has to be moved to follow the target. The beam steering angle is less than 2°, corresponding to the evolution of the SPS antenna-to-target line. This angle would be lower depending on the area of the beam on Mars surface.

5.2.1.4 Impacts on SPS attitude and configuration

The SPS follows the daily rotation of Mars. Therefore either the solar surfaces or the power antenna has to rotate to compensate this daily motion:

- The SPS could be sun pointed to minimize the constraints on the solar surfaces, especially if a very accurate pointing is made necessary (concentrators, etc). The power transmitting antenna has to rotate (360° capability) to follow the target. Solution would be to de-couple the power transmitting system from the solar surfaces and power generation system.
- The SPS could be Mars pointed to keep the power beam and power transmitting antenna fixed towards the target. The solar surfaces need one degree of rotation to follow the sun.

5.2.1.5 Advantages and drawbacks

This SPS positioning has the capability to transmit power to the target quasi permanently (exception are eclipse periods). A de-coupling system would allow to have solar surfaces oriented towards sun while the power transmitter antenna can be kept pointed towards the target.

Installing the SPS on this orbit does not raise technical problems.

Nevertheless, such a distance between SPS and target is penalizing for the RF power transmission system, especially for feeding the rovers. Indeed, due to the very low size of the rover receiver antenna, the efficiency of the power transmission is very low (see hereunder chapters), in the order of 10^{-7}, leading to provide at the input of the RF system more than 1 GW to deliver 500 W to the target! For this application, a much lower distance is necessary. In the case of Mars base, there is more flexibility on the size of the receiver antenna and efficiency of 10^{-4} to 10^{-2} could be achieved depending on the respective size of antennas.

5.2.2 Mars low equatorial orbit

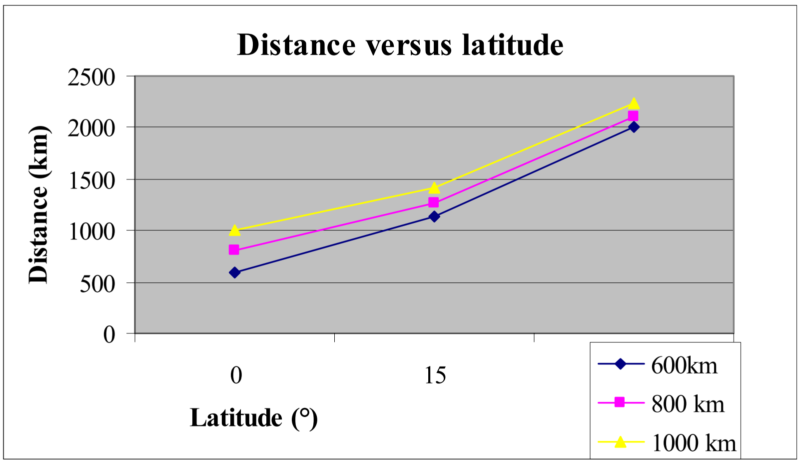
The interest of this orbit is to reduce the distance SPS-target in the case of RF transmission, and especially for the rovers. The SPS altitude is in the range 500 to 1000 km. A Mars equatorial orbit allows the SPS to fly over the target at every orbit, assuming that the target (rovers) remains at a reasonable latitude.

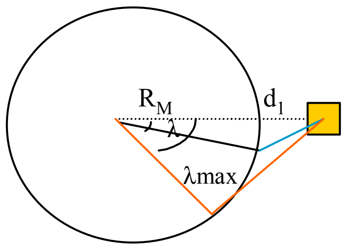
5.2.2.1 Distance SPS-target and periods

The Figure 5.2/3 illustrates the evolution of the SPS-to-target distance as function of the latitude of the target. This distance increases rapidly with the latitude. Besides, at such a short altitude, the SPS beam can only reach a limited range in latitude.

As shown on Figure 5.2/3, the maximum reachable latitude is between 32 and 39° according to the SPS altitude. In addition, to keep the distance within 1000 km, the rovers shall remain within 10° around the Mars equator. This is a strong constraint.

The period of the SPS as function of its altitude varies between 7380s (2h 3min) at 500 km to 8850 s (2h 28mn) at 1000 km.





Altitude (km)	600	800	1000
λ max (°)	31,8	36	39,4

Figure 5.2/3: Distance versus latitude for low equatorial Mars orbit

5.2.2.2 Availability of the power beam

Eclipse

The SPS will have an eclipse phase at each orbit. Nevertheless, it is not necessarily over the target at that time. When the target is in the eclipse zone, as illustrated on Figure 5.2/4, the duration of the period of non power delivery depends on the position, altitude and capabilities of the SPS. On the figure, the SPS is in eclipse between S1 and S2. Before arriving in S1, the SPS can provide power to the target, assuming a beam pointing capability of 2γ° total (it is the angle necessary to follow the target once acquired). In addition, a minimum time of power delivery is also assumed; thus, 3 minutes have been considered, leading to the point S0. The assumed worst case is then the acquisition of the target with a γ° beam pointing (M0) three minutes before entering eclipse. At the end of eclipse zone, the SPS can supply power to a target in M1 (γ° beam pointing). When the target is in M0-M1 zone, it cannot be powered from SPS even if SPS is above this zone due to eclipse.

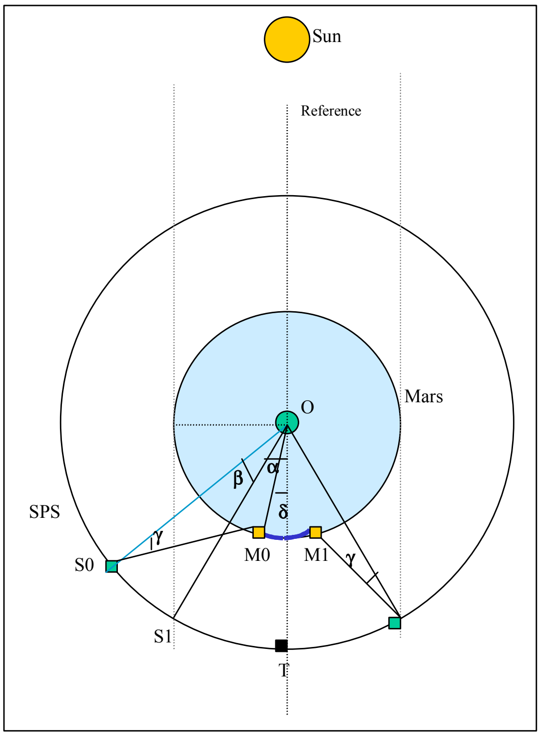


Figure 5.2/4: SPS eclipse illustration

$\beta = (180/T_{SPS})*360$

$\delta 2 =$ angle (S0-M0-O)$=$ Asin((R_{SPS}*sinγ)/R_M)

$\delta = \beta + \alpha - (180 - \delta 2 - \gamma)$

$\delta' = \alpha - (180 - \delta 2 - \gamma)$

Duration: $(\delta + \delta')*T_M/360$

The following table summarizes, for different SPS altitudes and two values of the maximum power beam pointing angle (15 and 30°), the duration of the period of non power transmission in seconds and in successive number of SPS orbits

SPS Altitude (km)	Duration (s)		Nb of successive orbits without power transmission	
	15° max	30° max	15° max	30° max
500	30910	29570	4,2	4
600	29400	27760	3,8	3,6
800	26740	24530	3,2	3
1000	24460	21620	2,8	2,4

Geometrical visibility

When the SPS is flying over the target, the duration of the power transmission depends mainly on the maximum beam angle and of the SPS altitude. The Figure 5.2/5 shows schematically the evolution of this duration for two values of the maximum beam angle (15 and 30°) and for different altitudes. This figure assumes that the target (mainly rovers) is fixed during the power transmission (more or less true with respect to the short duration) and is on the Mars equator (no effect of latitude taken into account).

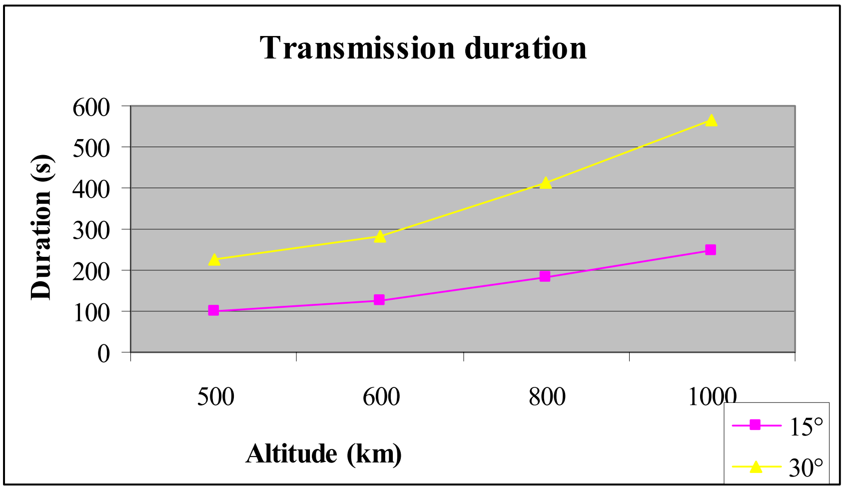


Figure 5.2/5: Duration of the power transmission to target as function of SPS altitude and maximum beam pointing angle

Availability

Taking into account the number of passes over the target per Martian day (number of SPS orbits per day), and the number of orbits during which both SPS and target are in eclipse, the availability of the SPS power to the target is shown in the Figure 5.2/6 in percentage of the Martian day.

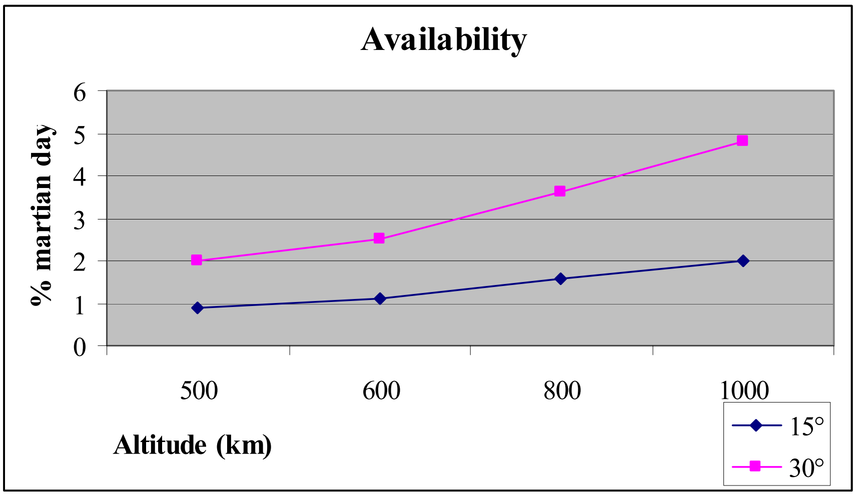


Figure 5.2/6: Availability of the power to the target as percentage of the Martian day

This figure, which results from a rough computation with simplified assumptions, shows that the power will be available only a few percent of the time with one SPS. Interest is to increase the altitude (distance SPS-target) and the maximum beam angle. Significant higher coverage would require a set of satellites.

5.2.2.3 Beam steering needs

The beam steering needs are driven by the relative motion of the SPS, and no more of the target (in case of rovers). During the fly over the target, the power beam angle will evolve between -γ° and +γ° (typically 15 and 30°). The beam angle steering rate is about:

Maximum pointing angle (°)	15	30
Altitude (km)		
500	0,3	0,27
600	0,24	0,21
800	0,16	0,145
1000	0,12	0,105

Beam angle steering rate (°/s)

5.2.2.4 Impacts on SPS attitude and configuration

The SPS has to point both the sun and the target. Therefore (see Figure 5.2/7):

- The SPS could be sun pointed to minimize the constraints on the solar surfaces, specially if a very accurate pointing is made necessary (concentrators, etc). The power transmitting antenna has to rotate (360° capability) to remain pointed towards the Mars surface. In addition, the power beam shall be steered between ±γ° (with respect to Mars direction) after target acquisition to follow the target. Solution would be to de-couple the power transmitting system from the solar surfaces and power generation system.
- The SPS could be Mars surface pointed to keep the power beam and power transmitting antenna fixed towards the target. The solar surfaces need one degree of rotation to follow the sun.

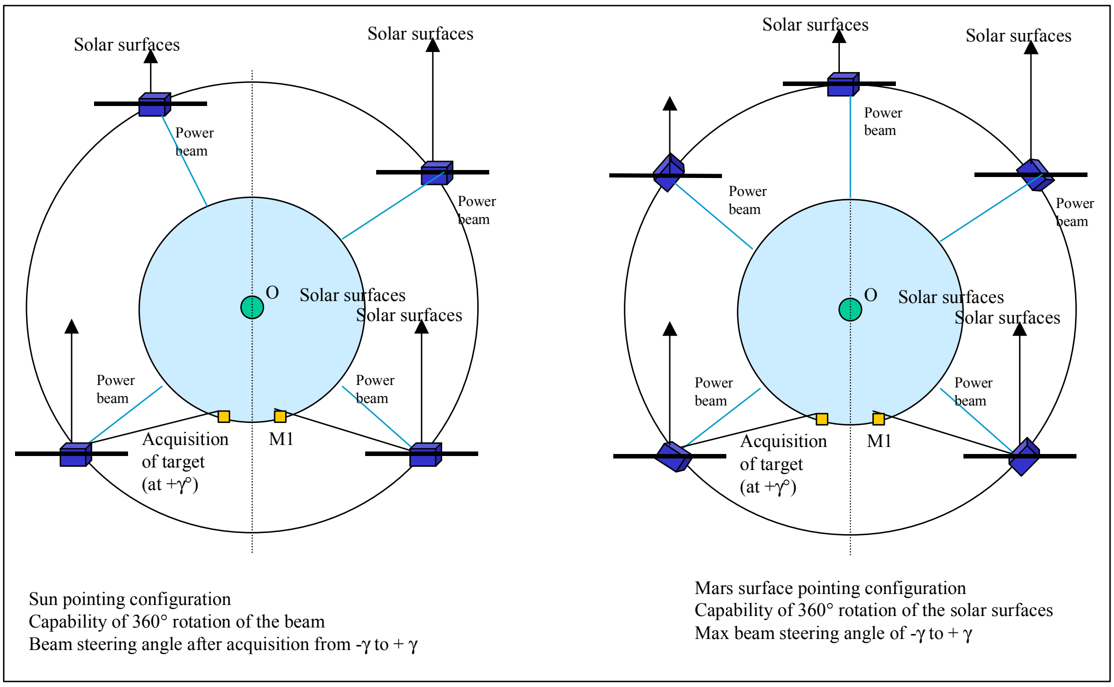


Figure 5.2/7: Illustration of possible SPS attitudes

5.2.2.5 Advantages and drawbacks

This SPS positioning allows to improve the efficiency of the power transmission system, in particular in case of RF technology. Overall efficiency of the RF system in the order of 10^{-4} could be expected for rovers powering with SPS altitudes lower than 1000 km and adequate projecting antenna size.

But this orbit does not allow the SPS to remain in visibility of the target, which will be powered only a short time per orbit, and thus a few percent of the Martian day. Several SPS would be needed to ensure a quasi permanent powering of the target.

In addition, the power beam shall be steered over an angle of ± 15 to 30° to ensure a few minutes of powering of the target.

5.2.3 Mars sun-synchronous orbit

The Figure 5.2/8 illustrates the sun synchronous orbit around Mars. The interest of this orbit is to reduce the distance SPS-target in the case of RF transmission, and especially for the rovers, and to fly over the target whatever the latitude. As for low equatorial orbit, the SPS altitude is in the range 500 to 1000 km.

5.2.3.1 Distance SPS-target and period

The period of the SPS as function of its altitude varies in first approximation between 7380s (2h 3min) at 500 km to 8850 s (2h 28mn) at 1000 km, as for the equatorial orbit.

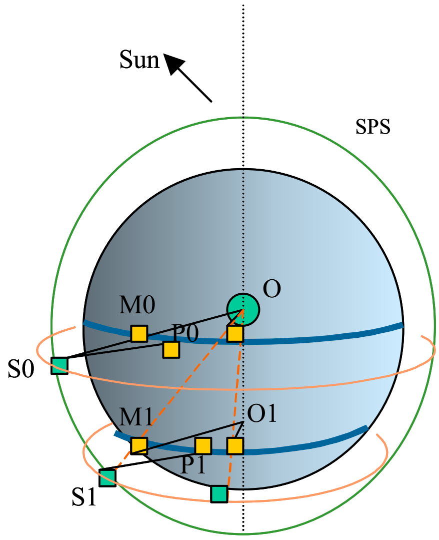


Figure 5.2/8: Low sun synchronous orbit for SPS

5.2.3.2 Availability of the power beam

Eclipse

This orbit allows the SPS to be quasi permanently in view of Sun. This requires a minimum altitude. There is no lack of powering due to eclipse.

Geometrical visibility

When the SPS is flying over the target, the duration of the power transmission depends on the maximum pointing capability of the projecting antenna system. The rough evaluation is the same as for the equatorial orbit (see Figure 5.2/5). It varies from 100 to 600s according to the altitude (500 to 1000 km) and the maximum pointing angle (15 or 30°).

5.2.3.3 Availability

There are very few orbits allowing the SPS to be in visibility of the target.

During one SPS orbit, the plane of the target rotates by 30 to 36° as function of the altitude (500 to 1000 km). When the SPS flies over the target, the analysis shows that at the next pass at the same latitude, the SPS is not necessarily in visibility of the target: that depends on the maximum pointing angle and of the latitude. Thus, as illustrated on Figure 5.2/8, with a pointing angle of 30°, the SPS can illuminate, at a latitude of 30°, an area located at about 5.8° at 500km, or 11.9° at 1000 km (angle counted from the Mars center). These values are lower than the rotation of the target and subsequently, the SPS cannot see the target. At higher latitude the angle increases.

Therefore, the SPS cannot power the target from two successive orbits, except for target at high latitudes (typically from 75° at 800 to 1000km, and from 80° at 500-600 km). As a consequence, the SPS altitude should be selected in order to have a repetitive orbit (fly over the same area every Martian day). Altitudes close to 500 km (12 orbits and 1000 km (10 orbits) are examples. However, it should be checked whether the orbit could be sun synchronous and repetitive within the altitude range.

The SPS will then fly over the target during two orbits per Martian day, and for short duration. The power will be available between 0,5 and 1,3% of the Martian time, depending on altitude, for a maximum power beam angle of 30°. This availability will be increased (at least doubled) if the target is located at high latitude (higher than 75°).

Significant higher coverage would require a set of satellites.

5.2.3.4 Beam steering needs

Identical to the case of low equatorial orbit (see 5.2.2.4)

5.2.3.5 Impacts on SPS configuration

This orbit allows the SPS to permanently see the sun. Therefore, the SPS can be Earth pointed, with the solar surfaces facing sun. No specific rotating mechanisms, except for the beam steering.

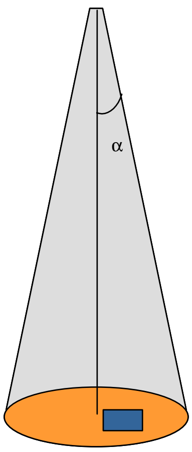
5.2.4 Trade-off and synthesis

Among the trade-off criteria, one is the achievable efficiency of the power transmission system in the different positioning, and the resulting impact on the SPS power generation surfaces.

5.2.4.1 Illuminated surfaces

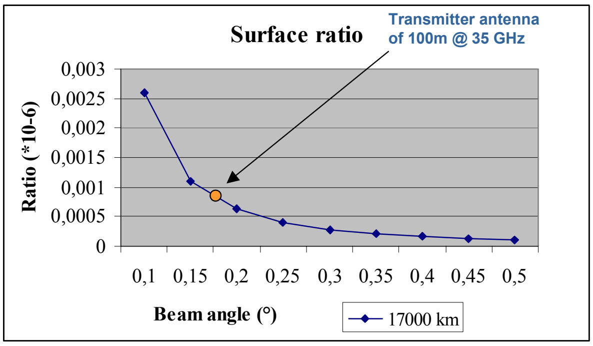
The Mars surface illuminated by the beam depends on the beam angle (function on the beam frequency and emitting surface diameter) and on the SPS-to-target distance

The ratio between the receiver surface and the beam illuminated surface drives the transmission system efficiency. Assuming a conical beam, the following table gives the illuminated surface and the ratio between the receiver surface and the illuminated surface for different beam angle and two receiver surface diameters (case of rover and case of Mars base).

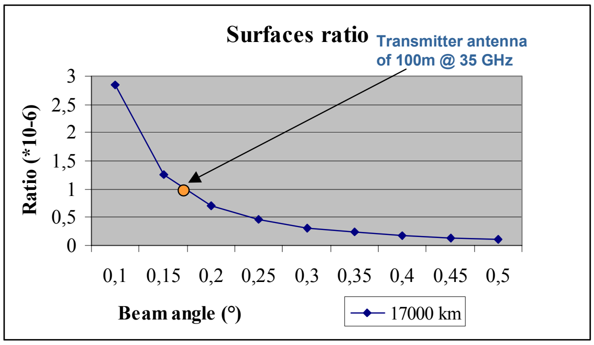


Distance (km)	Illuminated surface (km^2)			Ratio receiver surface/illuminated surface (10^{-6})					
				Receiver Φ3m			Receiver Φ100m		
	α= 0.5°	α= 0.25°	α= 0.1°	α= 0.5°	α= 0.25°	α= 0.1°	α= 0.5°	α= 0.25°	α= 0.1°
500	60	15	2,4	0,118	0,473	2,95	131	525	3280
1000	240	600	9,6	0,03	0,118	0,74	33	131	820
17000	69140	17280	2770	0,0001	0,0004	0,0026	0,114	0,454	2,84

The Figure 5.2/9 illustrates the ratio between the surfaces for a SPS-target distance of 17000 km and for the rover and the Mars base.



Rover case (3m receiver antenna)



Mars base case (100m receiver antenna)

Figure 5.2/9: Illustration of surface ratio for the rover and the Mars base

It can be seen that in case of RF transmission, the ratio between the surfaces is very low (les than 10^{-9}) in case of rover application when the SPS is on an areosynchronous orbit.

5.2.4.2 RF power transmission

Application to rovers

Due to its size, the rover cannot implement a large antenna. The antenna is deployed after the landing of the rover. The assumption is a typical diameter of 3 m (or an equivalent 7 to 8 m^2 of antenna surface) for this receiver antenna.

The efficiency (PR/PP) of the RF transmission system is given by:

$P_R/P_P = 0{,}556 * \eta_P * \eta_R * \eta_T * (D_P * D_R / \lambda_R)^2$

Where:

η_P is the efficiency of the transmitter system

η_R is the efficiency of the receiving system, including the antenna efficiency and the conversion

η_T is the attenuation along the transmission, due to the atmosphere

A rough evaluation of the overall efficiency of the system is illustrated in the Figure 5.2/10 to Figure 5.2/12. It assumes an efficiency value of 50% for the receiver, 30% for the transmitter and 100% for the transmission.

This evaluation has been done for an RF transmission frequency of 35 GHz and a receiver antenna diameter of 3m and two sizes for the projecting antenna (50 m and 100m).

If the SPS is at 17000 km, stationary positioned with respect to target, the efficiency of the transmission system will be in the order of 1 to 4 10^{-7}: the solar surfaces of the SPS have to provide 1 to 6 GW at the input of the RF transmission system to supply at the end 500 W to the rover!

Therefore, lower altitudes have to be considered.

One criteria to select the orbit altitude is the minimum overall efficiency acceptable.

This minimum is taken as 10^{-4}, which means to deliver 5 MW at the output of the solar surfaces in order to provide 500 W to the rover. That means an altitude of 500 km with a projecting antenna of 50 m, up to 1000 km with an antenna of 100 m (relevant efficiency of 1.02 10^{-4} with the above assumptions. Taking into account a possible improvement of other parameters, **the SPS shall be located at an altitude in the range 500 to 1000 km.**

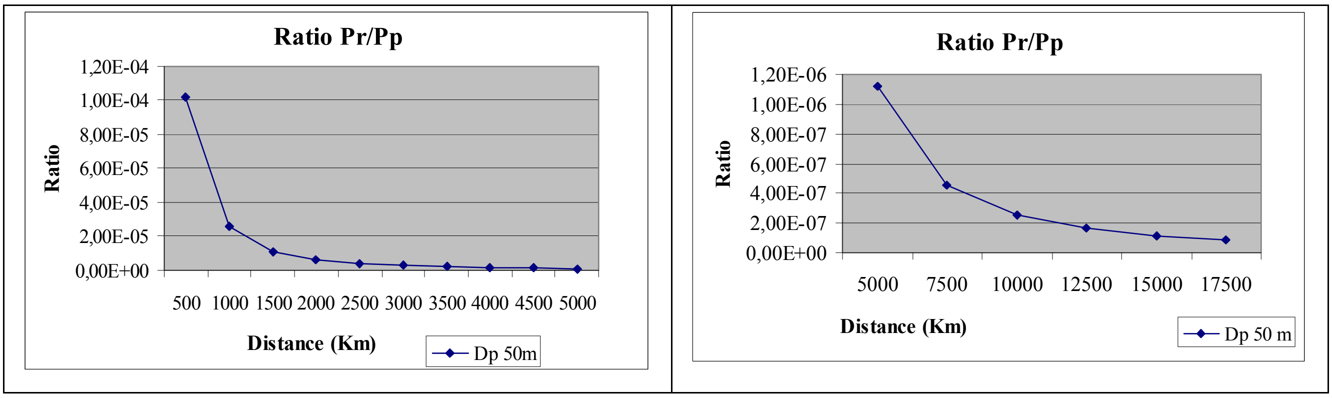


Figure 5.2/10: Evaluation of the transmission system efficiency for a projecting antenna diameter of 50m

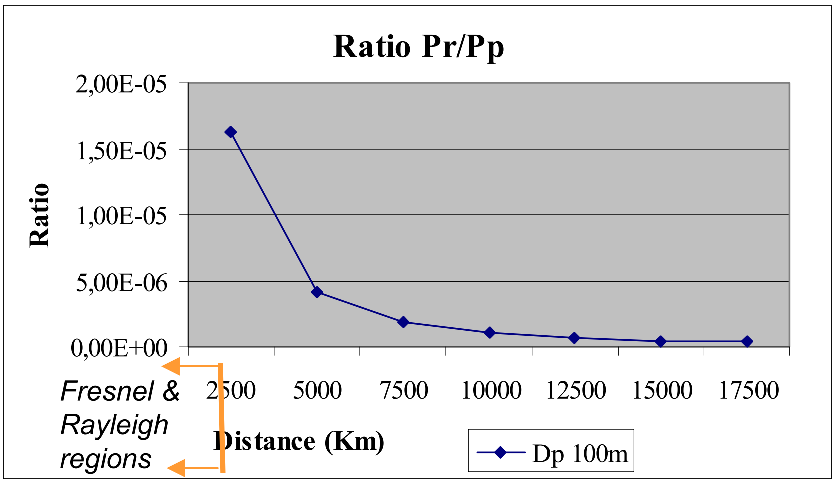
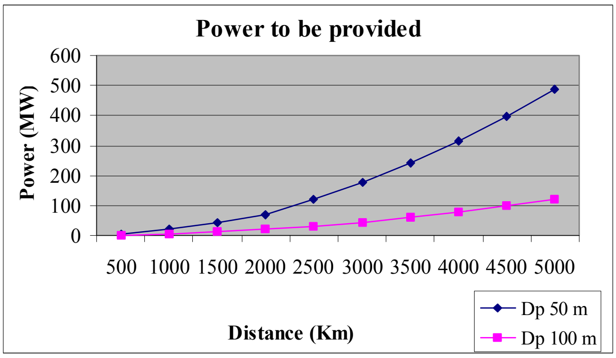


Figure 5.2/11: Evaluation of the transmission system efficiency for a projecting antenna diameter of 100m



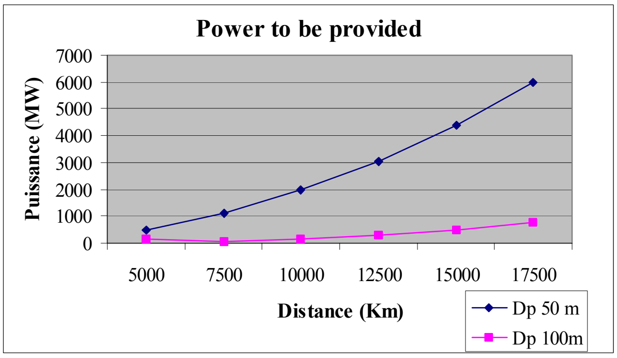


Figure 5.2/12: Evaluation of the power to be provided by the SPS power generation system for the rover case with projecting antenna diameters of 50 and 100m (SPS at 17000 km)

For a transmitter antenna diameter of 100 m, at 35 GHz, the Fresnel region is up to about 2500 km.

Application to Mars base

In the case of a Mars base, constraints on the receiver antenna size can be relaxed.

The Figure 5.2/13 illustrates the overall efficiency of the RF power transmission for an SPS at 17000 km, and projecting antenna sizes of 100m and 150 m. It gives the efficiency at 35 GHz for different sizes of the receiver antenna. This efficiency varies between 0,04% and 3%.

For powering the Mars base with RF transmission, the SPS could stay at the areosynchronous orbit (17000 km).

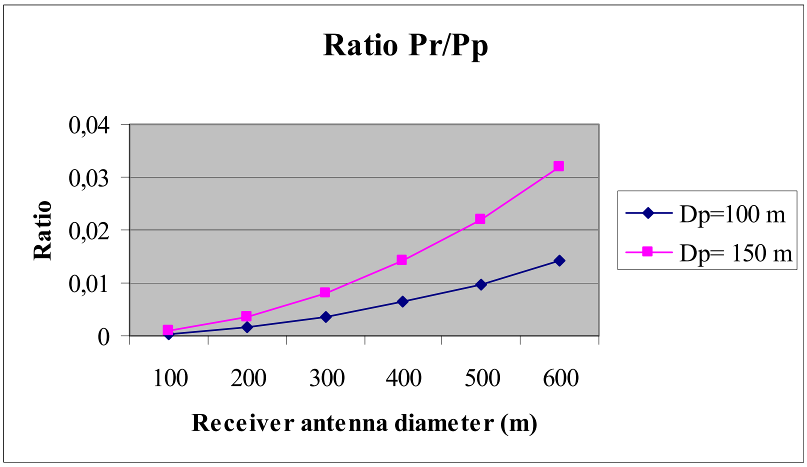


Figure 5.2/13: Evaluation of the transmission system efficiency for two projecting antenna diameters (100 m and 150 m) and different receiver antenna diameters (Mars base)

5.3 COMPARISON OF CANDIDATE SOLUTIONS

5.3.1 Case of rover

Two candidate solutions have been proposed for powering the rover:

- A RF system SPS located on low Mars orbit (equatorial, sun-synchronous or other)
- A laser system SPS located at 17 000 Km (areosynchronous orbit)

The RF system SPS located at the areosynchronous orbit was not retained due to the very low efficiency.

5.3.1.1 RF system SPS in low Mars orbit

An efficiency of at least 10^{-4} is expected, leading to a required power of less than 5 MW at the input of the RF system. To get this efficiency, a transmitter antenna size of at least 50m diameter is required. That requires the mastering of large antenna technology at 35 GHz; thus, to minimize the antenna losses, a phasing error lower than 5 mm is required over the antenna length in case of a parabolic antenna.

The maximum SPS-to-target distance is 1000 Km to cope with the minimum efficiency. A lower SPS altitude could be selected to take into account the necessary antenna beam steering angle. Indeed, the beam steering angle degrades the efficiency: it leads to an increase of the distance and the diminution of the equivalent size of the antenna proportionally to the cosines of the steering angle.

On such low orbit, the SPS will be in visibility of the rover only a few % of the Martian day, and power will be available to the rover accordingly.

The attitude and orbit control of the SPS will have to take into account the impacts of perturbations, large surfaces to be controlled. That will impact the propellant mass.

Different low orbits have been considered:

- Low equatorial orbit

  It is applicable if the rover is at a latitude lower than 10°. The SPS will fly over the rover at each orbit, but will have a short visibility duration; in addition, the SPS could be in eclipse during some passes. This type of orbit has impacts on the SPS configuration: either the rotation (360°) of the solar surfaces (one or two degrees) or of transmitter system. The beam steering rate would be up to 0.3°/s.

- Sun synchronous orbit

  It is applicable to all latitudes. There is no eclipse. There are two passes over the rover per day, with a very low period of power transmission. There is a possibility of having two successive orbits with target visibility for high latitudes, but still with a very few time of visibility. The SPS would be Earth pointed with solar surfaces fixed towards sun.

- Inclined orbit

  The interest is to have an orbit with inclination equal to the rover latitude. That would increase the visibility duration, which however would remain low. There are eclipse period, and a necessary rotation, with two degrees of freedom, of either the solar surfaces or of the transmitter system.

5.3.1.2 Laser system SPS in areosynchronous orbit

The key problem of the laser system is the impact of the dust storm. Assumptions have to be taken to assess this impact.

- **Assumptions on beam attenuation and laser transmission**

The Figure 5.1/5 (from the S54 study) gives the direct beam irradiance variation with the Optical Depth (OD). It corresponds to a particular region in north latitude. The received power is 325 W/m2 for OD=0.5, and 35 W/m2 for OD=3.

From this figure, it is proposed to take as hypothesis the following ratio of degradation of solar flux density:

100% for OD=0.5

10% for OD= 3

5% for OD=4

The degradation of the laser power density on the rover surface as function of OD is assumed in the same ratio as the direct solar irradiance. Thus, the power provided by the laser system on the rover could be:

650 W for OD=0.5 (as defined in the section 5.4 above)

65 W for OD= 3

32 W for OD = 4

These assumptions have to be confirmed by a detailed analysis of the transmission through the Mars atmosphere and dust winds. In particular in case of very accurate pointing.

It could be noted that the laser beam remains close to the normal of the rover surface while the direct solar irradiance angle varies with the daytime and latitude.

- **Dust storms models**

The dust storms models taken into account for this analysis are as follows, according to Figure 5.1/4:

- Local dust storms last only a few days with OD less than 1

- Regional dust storms: there is only one per year, with a duration of about 100 days with an OD peak of 3 lasting a few days

- Global dust storms: there is one global storm per year at the maximum, having a duration of about 100 days with an OD peak of 6 lasting a few days.

- **Impact of dust storm on laser power transmission**

The rover is assumed not operational during a dust storm. It shifts to dormant mode and remains fixed. Its power consumption is assumed 50W. Besides, it is assumed equipped with countermeasures to avoid opacification of solar cells by the dust.

Local dust storms:

During local dust storms, the OD remains lower than 1. The power received from SPS is higher than 50W.

Regional dust storms

The Figure 5.3/1 gives the evolution of the OD and of the level of power supplied to the rover. The laser transmission system would provide enough power to the rover, despite beam attenuation due to dust.

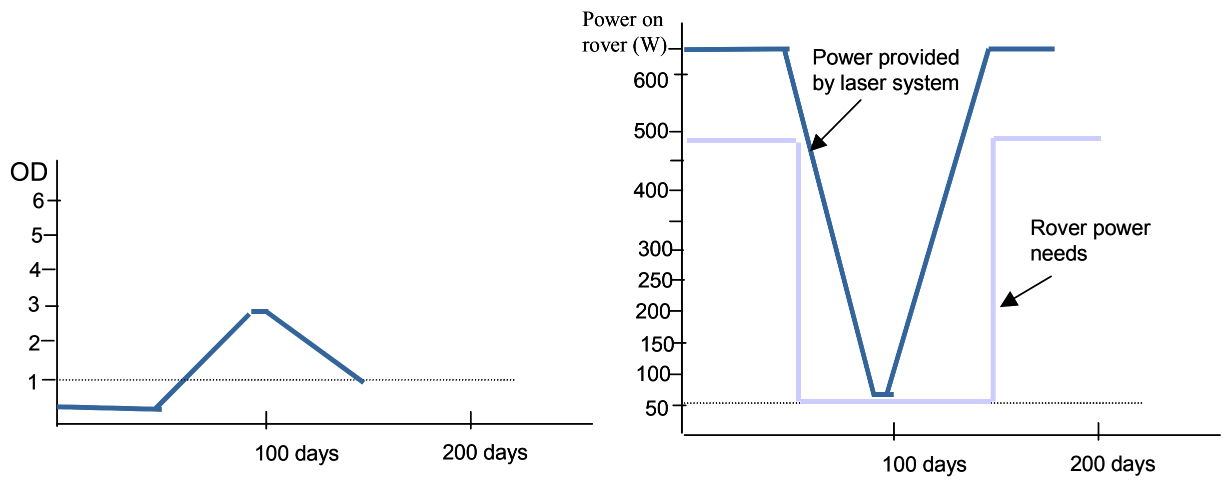


Figure 5.3/1: Illustration of assumed OD evolution and power supplied to the rover (regional dust storms)

Global dust storms

Evolution of OD and of power supplied to the rover is illustrated on Figure 5.3/2 for the assumed dust storm model

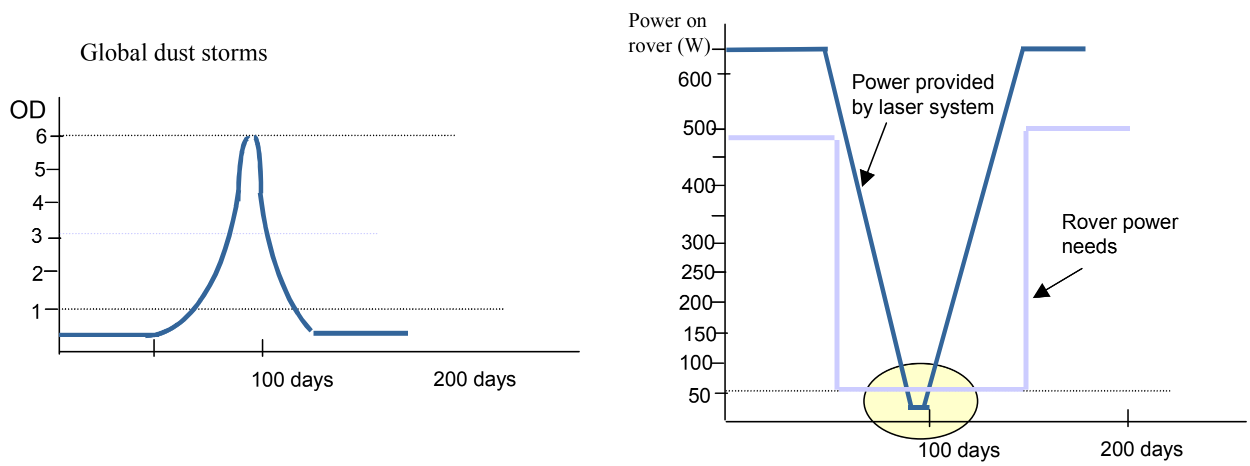


Figure 5.3/2: Illustration of assumed OD evolution and power supplied to the rover (global dust storms)

There will be few or no power supplied to the rover if the OD is above 3. The peak of storm is assumed to last a few days. Therefore, the rover battery has to be sized for the peak period.

For a peak of 5 days, the requested electrical energy is 6000 Wh. Assuming Li type batteries (for instance Li-ion) with a capability of 120 Wh/kg, the battery mass would be 60 Kg, including 20% margin, in line with a rover of 500 Kg. This mass could be reduced with improved or new technologies at that time.

These figures have to be considered carefully. The associated assumptions, such as the duration of the peak of the storm, and the diffusion of the laser beam through the dust have to be analysed in detail.

5.3.2 Case of Mars base

Two candidate solutions have been proposed for powering the Mars base:

- A RF system SPS located at 17 000 Km (areosynchronous orbit)
- A laser system SPS located at 17 000 Km (areosynchronous orbit)
-

5.3.2.1 RF system SPS in areosynchronous orbit

An efficiency of 1 to 5% could be expected, with adequate system (with respect to efficiency) and large antenna surface. That would lead to a requested power of 2 to 10 MW at the input of the RF system.

The transmitter antenna size would be in the range 100 to 200 m. Such size raises the problem of large antenna technology at 35 GHz, such as the manufacturing and the accuracy of the surface. The best compromise between the antenna size and the solar surface size (requested power at the RF system input) will have to be found.

Large receiver antenna size (rectenna) at the Mars base (100 to 400m) will be necessary. Here again, the best compromise between ground antenna, on-board antenna and on-board solar surfaces will have to be traded. The large antenna on Mars surface raises the problem of its implementation (transport, deployment, maintenance, etc), which will have to be analysed.

The beam steering angle will remain below 2°.

The SPS will provide a high availability of the power to the target, with a quasi permanent visibility. Only short eclipse periods around equinoxes will require power storage system on the Mars base.

The SPS configuration will have to cope with either rotating (360°) solar surfaces (one degree of freedom with a possible angle of the beam with respect to sun direction) or rotating transmitter system. In addition thermal control aspects will also impact the SPS configuration.

There will be lower impact on attitude and orbit control than in the case of low Mars orbit.

5.3.2.2 Laser system SPS in areosynchronous orbit

Efficiency in order of few % could be expected, leading to a requested power in order of 2MW or less at the input of the laser system in case of clear sky. The size of mirror is reduced (a few meters) as compared to the antenna.

The receiver system is composed of large surfaces of cells or of electrical-to-thermal conversion system. The photovoltaic cells will also receive the sun power flux, variable according to the daytime, so that the cells will have to be sized to withstand the peak of power density.

The beam steering angle is below 2°, as for the RF system. Likewise, the laser system has a high availability to provide power to the Mars base, as being quasi permanently in visibility, except for short eclipse periods around equinoxes.

But the main problem of the laser system is the impact of the dust storm. The Mars base shall be powered permanently, including during dust storm, with 100 kW (no dormant mode). There will be no enough power provided by the laser system in case of dust storm with Optical Depth of 3 or more, unless oversizing the SPS by an important factor (10 or more). Besides, sizing the Mars base battery system to compensate the loss of power during dust storm would appear too heavy.

The impact of the mission on the SPS configuration is identical to the RF case.

5.3.2.3 Synthesis

For rover application, the RF system has too low performances: low efficiency leading to a high sizing of the SPS solar surfaces, or too low availability in low orbit.

The laser system SPS in areosynchronous orbit is preferred: quasi permanent visibility to the rover, an efficiency of up to 1% could be expected (in case of blue sky), although it is penalized by its behaviour in case of dust storm. However, in the case of rover, some assumptions allow the utilisation of the laser system: at first, the rover remains in dormant mode (fixed with 50 W consumption), then the rover receives at least 10% of the power flux density at Optical Depth of 3, the dust storm peak (with OD higher than 3) lasts less than 5 days and finally the rover cell degradation or opacification is low. These assumptions have to be verified.

For Mars base application, the laser system appears too much penalized by the dust storms.

The RF system SPS in areosynchronous orbit is preferred: it offers a quasi permanent visibility of the target, an efficiency of 1 to 5% is expected, but it assumes large transmitter antenna on SPS and large receiver surfaces on Mars.

5.4 POWER DELIVERY TO ROVER

5.4.1 Laser power transmission system

5.4.1.1 On-board transmitting system description

The mission of the SPS system is to provide 500W to a Martian rover during solar Eclipse. Rover field coverage is approximately 400 km. We propose to reuse the same laser concept as described in Chapter 2.3. The main difference is related to the output power and the size of the emitting telescope.

The SPS system is composed by 4 independent laser systems that can be pointed independently. The 4 systems are embarked on the same satellite located in areosynchonous orbit. The pointing system is a loop controlled system. On each rover corner cubes are embarked. These corner cubes reflect the received beam. An acquisition process (with open loop scanning) is necessary to identify the Rover position and to point the SPS laser beam in the right direction. The line of sight is maintained in the rover direction thanks to the retro-reflected beam that acts as beacon.

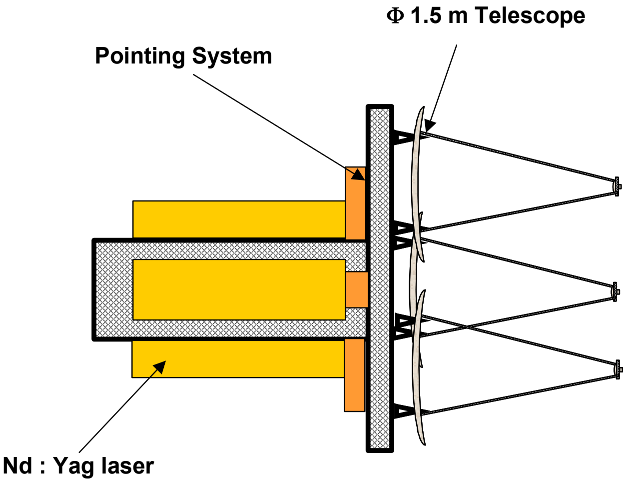


Figure 5.4/1: SPS Concept

5.4.1.2 Mass and power budgets of the laser system.

The mass and the power budget of the 6 kW laser power system are given in the following tables.

Sytem laser mass	Mass in Kg
Telescope	40
Pointing system	5
Opto-mechanical assembly of laser	100
Mechanical structure	150
Electronics	150
Total	445

Table 5.4/1: Mass budget (for 1 laser)

Power flow chart laser		
Emitted power	6000	W
Number of lasers	4	
Optical pump efficiency	50	%
Laser diode plug in efficiency	50	%
Power generation system		
DC/DC converter efficiency	85	%
Power for spacecraft	10000	W
Power required at SA output	122941	W
solar cell efficiency	30	%
solar power density	490	W/m^2
solar array size	836	m^2

Table 5.4/2: Power budget (for 4 lasers)

5.4.1.3 Performances and SPS parameters

The performances of the SPS system are depicted in the following table. The rover receiver surface is assumed a square panel with a 3 m side.

Parameters	Value	Unity
Input:		
Distance Satellite to Mars Surface	16700	km
Distance Minimum Rover vertical position	0	km
Distance Maximum Rover to vertical position	200	km
Size of the square solar pannel of Rover	3	m
Efficiency of the solar pannel	50	%
Useful part of the solar pannel	98	%
Nunber of associated lasers	4	
Emitter diameter	1,5	m
Emmitted Wavelength	1060	nm
Emitted Optical power	6000	W
Output:		
Emitted Optical sytem		
Angular Pointing	1	Degres
Pointing Accuracy	86,2	nrad
Rover parameters		
Minimum distance Laser to Rover	16700	km
Maximum distance laser to Rover	16701	km
Received power at FF		
Spot dimension at Rover level	14,40	m
Electrical Power at Rover	650	W

Table 5.4/3: SPS parameters

5.4.2 Power generation system definition

Power needs

The payload is made of 4 lasers. Each laser is composed of 12 laser diode stacks, as illustrated on Figure 5.4/2. Each stack requires an input power of 2 kW under 20 V.

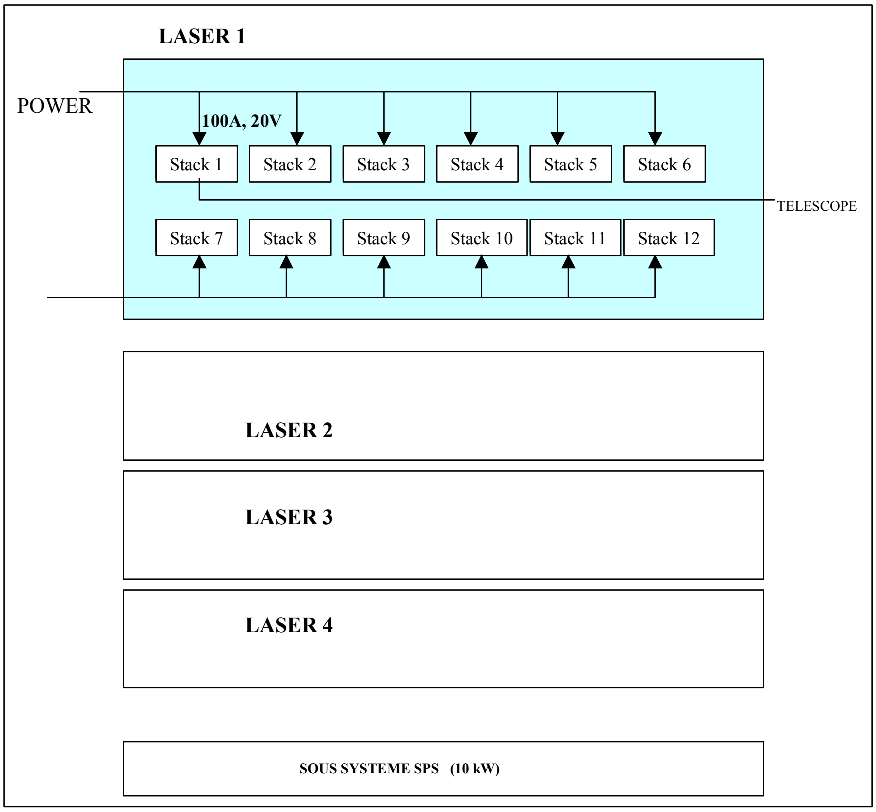


Figure 5.4/2: laser system requirement on power distribution (rover case)

Assuming a DC/DC converter efficiency of 85%, and a provision of 10 kW for the SPS own subsystems, a total power of 123 kW has to be generated, as shown on Table 5.4/2.

The same architecture as for the Darwin mission can be proposed. Each laser will be individually powered by an independent Power Sub System (PSS). A fifth PSS will provide the required 10 kW for the spacecraft. The solar arrays, based on multiple-junction solar cells with small concentrators, have an area of 837 m2 (192 m2 for one laser PSS and 69 m2 for the spacecraft) and a mass of 1255 kg (288 kg for one laser PSS solar array, 103 kg for the spacecraft).

This surface corresponds to slightly more than one ISS solar array (between two and three wings).

Illustration of existing system for 1000 m^2 surface

Such a surface of solar cells is already existing and flying. It concerns the ISS (International Space Station).

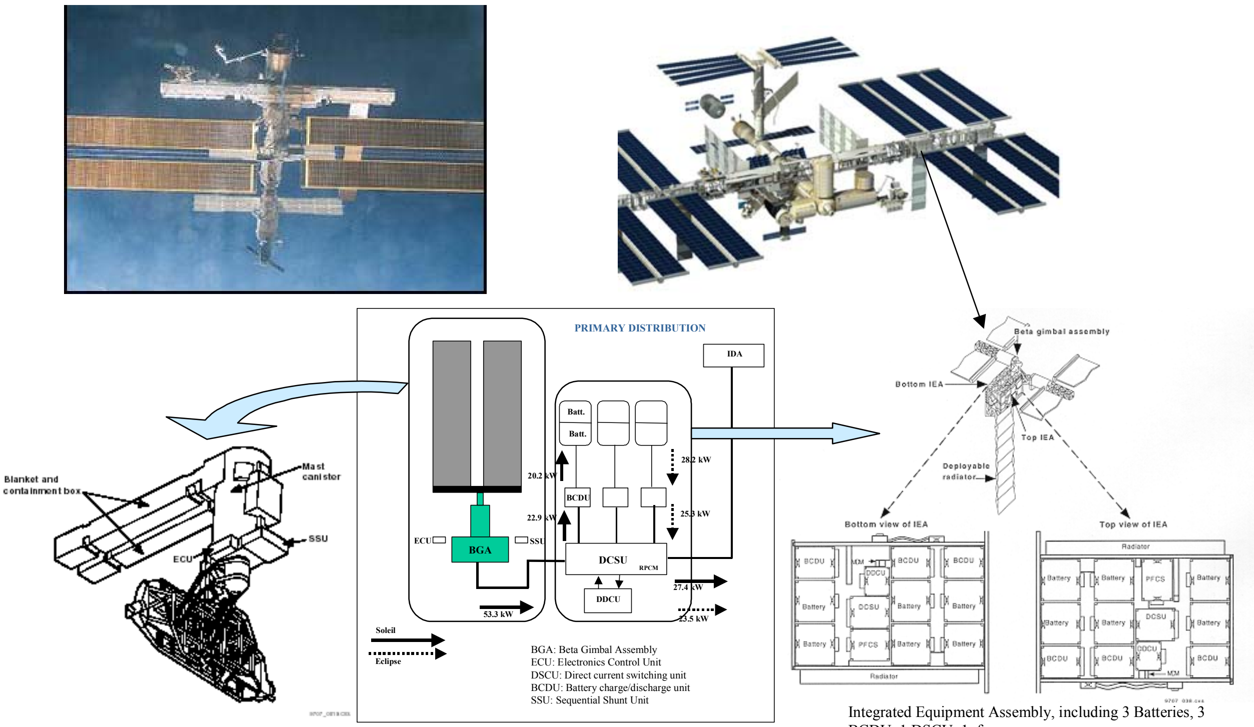
Indeed, at complete assembly, the ISS will have up to 3000 m2 of solar arrays (but not the same type of cells). The ISS comprises 4 large solar arrays units implemented on the truss, each consisting of 2 wings.

The Figure 5.4/3 illustrates the ISS solar arrays.

Each wing has a surface of about 360 m^2 (32.9 m x 11.3m).

Each solar array has two degree of freedom, made necessary by its orbital location (@ 51°6 inclination).

Each solar array assembly includes batteries, electronics, solar array rotation mechanism, and a deployable radiator.



Integrated Equipment Assembly, including 3 Batteries, 3 BCDU, 1 DSCU, 1pfcs

Figure 5.4/3: Illustration of ISS solar arrays (pictures from NASA)

Proposed power architecture

For delivering 123 kW, a S3R (Sequential Serial/Shunt Switching Regulator) power architecture is proposed. It is illustrated on Figure 5.4/4. Its main characteristics are:

- A regulated bus at 100 V
- A Solar Array voltage matching better than 97%, even when considering the Sun flux range in the vicinity of Mars, which varies between 490 and 720 W/m2.
- The dissipation at the level of switch can be reduced to less than 1% of the Solar Array power
- The division of the solar array into sections lead to easily manageable switches, which can be sized for a voltage drop lower than 0.5 V by paralleling Mosfets, thus leading to a conversion efficiency of 99,5%.
- Each of the four laser payloads can be individually controlled.

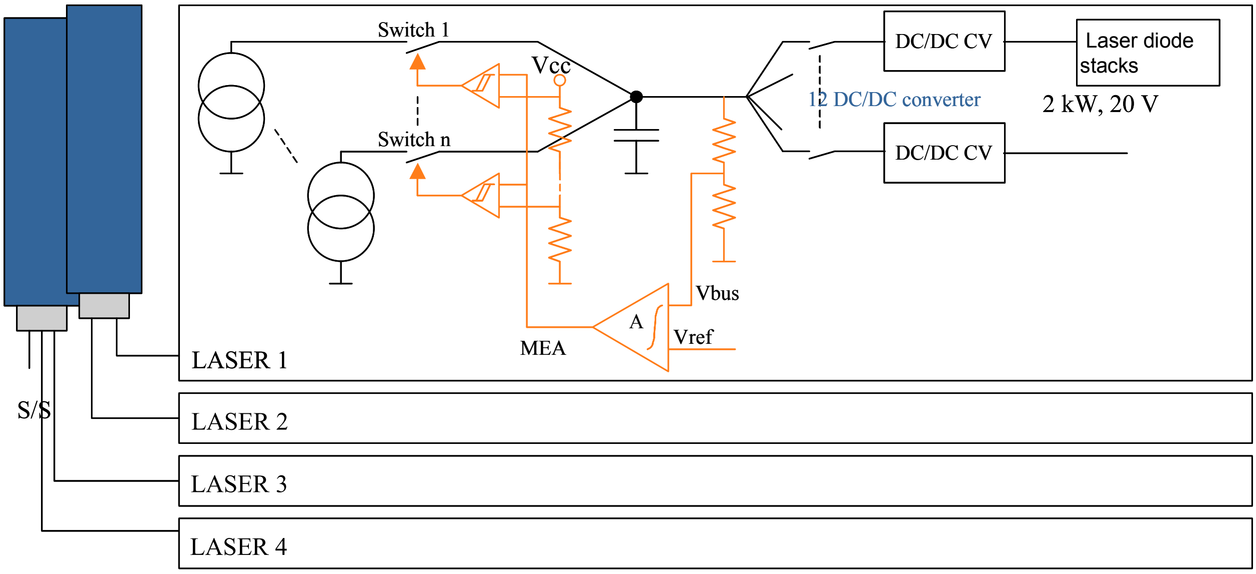


Figure 5.4/4: Proposed power architecture (S3R)

5.4.3 Preliminary definition of SPS

The SPS preliminary design includes the payload part, which is composed of the four lasers with the associated radiators, and the resource part with the solar arrays, power primary distribution, propulsion, communications and SPS management.

This preliminary SPS definition is illustrated on Figure 5.4/8. It relies on current or reasonably achievable technology, both for the laser system and for the SPS subsystems.

5.4.3.1 Main features

The SPS is located on an areosynchronous orbit, and is pointed towards the Mars surface (towards the rover area).

There are two solar arrays mounted on the North/South axis to avoid occultation. These solar arrays have two degrees of freedom, one for the orbit compensation (360°) around the North/South axis, and another one for the control of disturbances. Each solar array has a size of 10m x 42m. In launch configuration, as illustrated on Figure 5.4/5, each solar array is folded as in the case of the ISS solar arrays. That makes easier the implementation under fairing. 10 kW are foreseen for the power need of the SPS subsystems.

Batteries (typically Li-ions batteries) are mounted to provide power to the SPS subsystem during the short eclipse period. In these periods, the SPS does not provide power to the rover that relies on its own batteries. 40 kg of batteries could provide up t 2 kW during the 1h20 of eclipse. The SPS batteries are recharged from the solar arrays. Ancillary equipment (charge/discharge equipment, distribution unit, etc) are installed.

The thermal control system is based on active fluid loops and deployable radiators. A redundant pump assembly is foreseen on each loop. 2 deployable radiators (40 m^2 each, 3 x 13m) are installed close to the solar arrays level to dissipate the heat generated in the primary distribution. Two deployable radiators (120 m^2 each, 4x 30m) are mounted on the payload part, to dissipate the heat generated by the power losses. They are sized to dissipate about 100 kW. However, the definition of the laser system and power subsystem leads to a heat loss of about 72 kW (heat dissipation of the primary converter being taken into account by the other radiators), lower than the assumption, and thus would result in smaller radiators.

The communications with Earth are ensured with two systems. Two orientable X band antenna (typically 0.8m diameter) and associated transponders allow the transmission of a few kbps; the two antennae are made necessary partly for redundancy and partly to ensure the complete visibility in presence of large appendages. Two omni S band antennae are used in emergency to transmit a tens of bits. Recorders are foreseen to cover the Earth occultation. In addition, data formatting, multiplexing/demultiplexing equipment complete the communication system.

The navigation relies on various types of sensors to perform the transfer to Mars and the acquisition of the final orbit and of the target. The attitude control relies on sensors, inertial systems and reaction wheels. Attitude disturbances along the orbit could be partly compensated by the solar arrays control. Chemical thrusters (typically hydrazine) are foreseen in addition to electrical thrusters for attitude control.

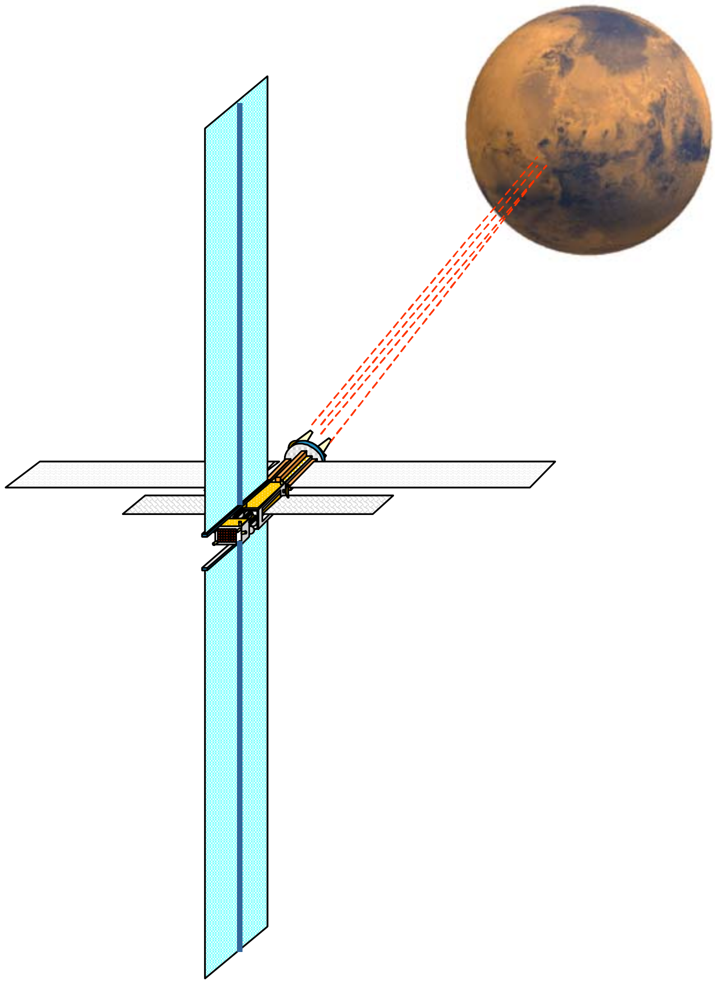
The transfer from the low Earth orbit (on which the SPS is injected by the launcher) to the final Mars orbit is performed with electrical propulsion. A thrust level of 100N is assumed, with a set of 1N thrusters. The solar arrays capability is sufficient to provide power (typically 100 kW) to the thrusters. The heat dissipation is ensured by the deployable radiators mounted around the solar arrays and if necessary one of the payload radiator. Necessary Xenon is stored in several (up to 7) small tanks under high pressure.

The SPS configuration in launch configuration (Figure 5.4/5) has a diameter of 5m and a length in order of 16 m, compatible with current long or extra long fairing.

5.4.3.2 System overview

The SPS system performances are summarized in the Figure 5.4/6. It has an end-to-end efficiency of 0,6%, mainly due to the required pointing accuracy and to the small size of the receiver as compared to the beam spot at target level.

The SPS mass is about 40t. In this mass budget, the size of some SPS subsystems relies on initial assumptions (radiators) and not on the definition of the laser system and other SPS subsystems. Taking into account these definitions would lead to a lower mass. The sharing of the mass between the different subsystems is given in Figure 5.4/6. The main contributor is the propulsion system that represents about 44% of the total mass.



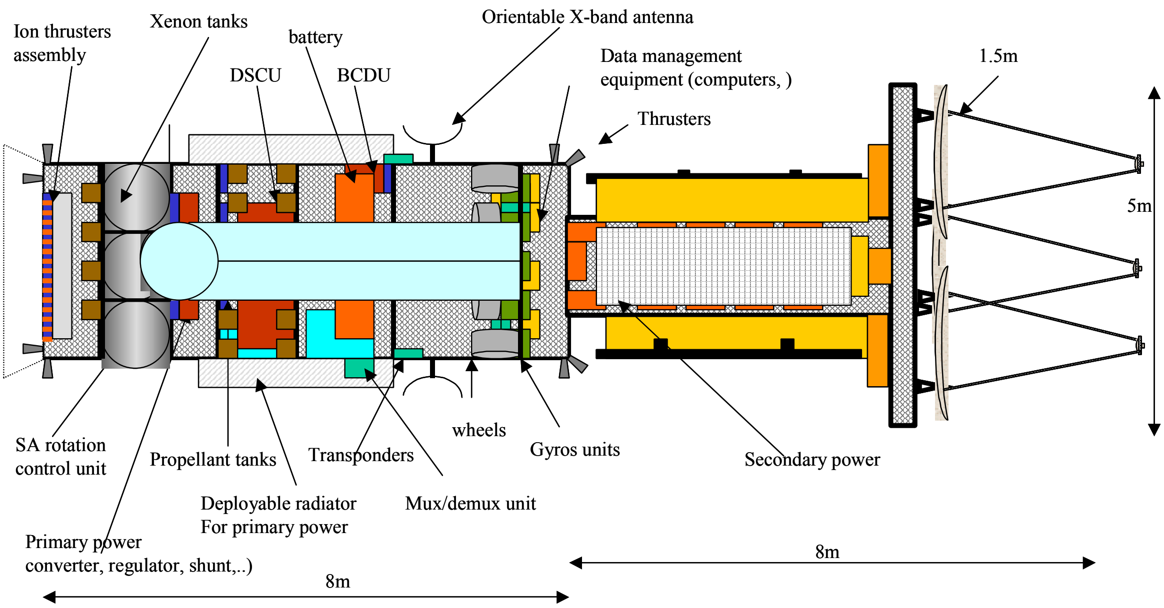


Figure 5.4/5: Illustration of SPS preliminary design for delivery power to rover on Mars surface

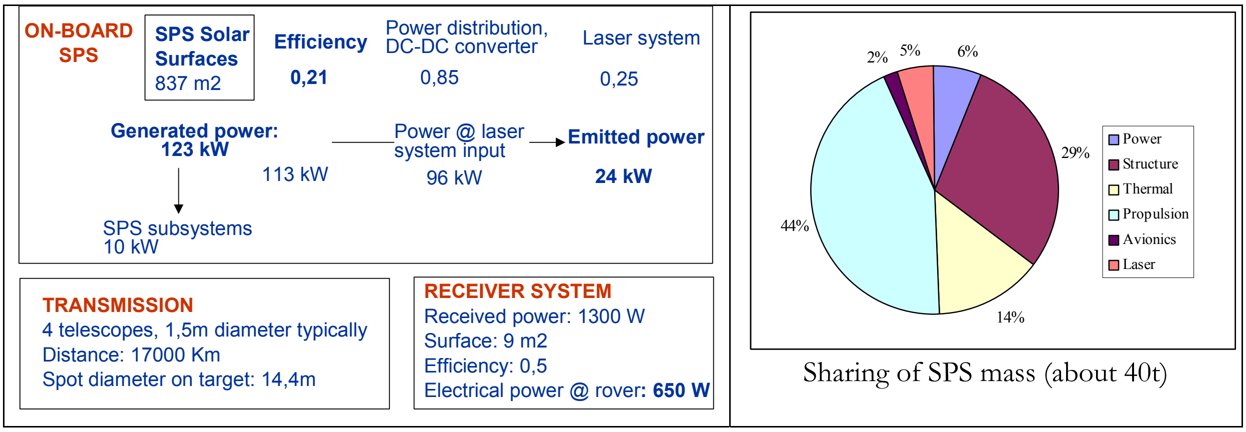


Figure 5.4/6: SPS system performances

5.4.4 Interest of SPS

The interest of using SPS is to power permanently small elements on Mars surface, like rovers or other, requiring low level of power. It would allow increasing significantly the lifetime and the operability of such elements that cannot have large batteries. However, these small elements have small receivers surface that is a strong constraint on the power transmission system. A laser transmission system is made necessary.

The critical point is the dust storm. It is necessary for the rover or other small element to have a dormant mode at low power consumption, and to be equipped with countermeasures to avoid or reduce the degradation of the receiver surface. The behaviour of the laser beam in the dust storm is an open point and analyses have to be done to evaluate the effects. Another point is the low end-to-end efficiency, which could be increased with advanced technologies.

Alternatives are either solar cells and batteries (current solution), or nuclear systems (RTGs).

5.5 POWER DELIVERY TO MARS BASE

5.5.1 RF power transmission system

5.5.1.1 Elements and methodology for the dimensioning of RF WPT systems

5.5.1.1.1 Methodology

The following method has been used for dimensioning the RF WPT system: first, to identify a possible rectenna technology for the frequency and to try to design the system so that the rectennas work at their best possible efficiency. This is done by fixing the power density at the collecting antenna to an optimum value, considering the antenna gain and efficiency. Then, to calculate what kind of projecting system could deliver such an optimum RF power density, that means what the size of the projecting antenna should be and what level of projected power is required. Final dimensioning takes into account the technology of projecting antenna and RF power sources to account for other losses. This allows to estimate the overall performance of the system.

5.5.1.1.2 Constraints

As already mentioned in the TN1 of this study, the frequency is chosen to be around 35GHz. This frequency will be used for all dimensioning purpose. However, other frequencies close to 35GHz, but which would be more convenient for regulatory issues could be used without affecting too much the overall design of the mission cases.

5.5.1.1.3 Rectenna technologies

Rectennas at 35GHz have been studied in only a few occasions. We will use results from previous published work so as to get validated and up to date data for efficiencies and rated power.

The reference work, which has been used, is the one from [4] (Yoo and Chang, 1992). This paper demonstrates that a rectenna can be designed at 35GHz, but it would be optimized to work at high power density. From this paper, we could estimate the level of incident power at the rectifying Schottky diode that should be used to get the highest efficiency. This power is approximately 150 mW.

Reaching such a power at 35GHz necessitates either to have a high power density at the collecting antenna aperture or to use a large gain collecting antenna to draw enough power from the beam to drive the rectifying circuit at its optimum power level. In any case, it would be difficult to achieve a high density of rectifying circuit. Using high gain antennas is also a drawback at this frequency because this is usually accompanied by a corresponding decrease in the antenna efficiency. The best solution would then be to use a lower gain antenna coupled to a rectifying circuit designed to work at lower optimum input power. We have not found evidence of such designs in the literature, but we believe this can be accomplished using adequate Schottky diodes and a correctly designed rectenna circuit. For this reason, we will arbitrarily use an optimum input power of 50mW for one of the alternative rectenna description that we will introduce in some versions of the dimensioning, and believe that it is within the reach of current technology or that it could be accomplished in a near future. A very low power density rectenna is also introduced, with a rectifying circuit optimized for an input power of 21mW but with this circuit, we only expect a maximum power conversion efficiency of 65%.

Below, we present the description of the four types of rectenna that we will use in the dimensioning of the mission cases, classed from the highest RF input power density to the lowest.

5.5.1.1.3.1 Rectenna type 1

This rectenna is conceived to work at higher incoming RF power density. It uses a lower gain antenna to provide the optimum 150mW at the input of a rectifying circuit. The antenna is a series patch antenna array. Example of such antenna is given on Figure 5.5/1.

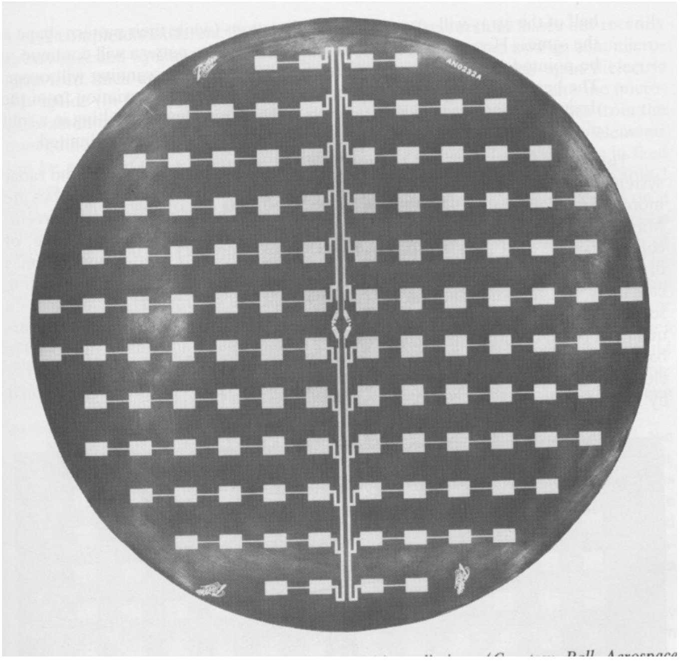


Figure 5.5/1: Example of series patch antenna technology that we could adapt to fabricate a rectenna element of type 1 (From "Antenna Handbook", Vol. III, pp17-115, Ed. Y.T. Lo & S.W. Lee, Van Nostrand Reinhold, N.Y., 1993).

The main characteristics of this rectenna element are listed below:

Property	Value	Comments
Antenna technology	Wave guide fed patch in series antenna	
Antenna gain	17dBi	
Antenna efficiency	70%	
Antenna dimensions	7cmx7cmx3cm	
Antenna effective area	2.96 cm^2	
Antenna mass	10g	Antenna effective mass 5kg/m2
Rectifying circuit	Schottky diode and filters	
Optimum input power	150mW	
RF to DC conversion efficiency	75%	
Optimized RF power density	**724 W/m2**	

Table 5.5/1: Main characteristics of rectenna element of type 1.

Another recent technique has been presented that could be suitable and more easy to integrate ([12] Van der Wilt and Strijbos). It is called a HPCA (Hybrid Coupled Planar Array). Figure 5.5/2 shows a realistic drawing of the antenna module. A waveguide runs through the middle of the antenna. Slots in the waveguide allow for the coupling of the incoming RF power to the lines of radiating patches in series (Figure 5.5/3). The connection flange and termination load are not represented.

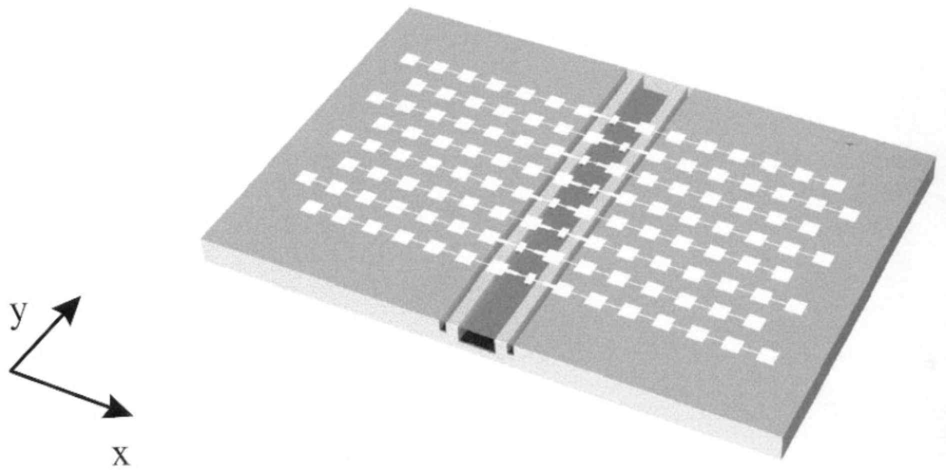


Figure 5.5/2: Realistic drawing of HCPA antenna ([12])

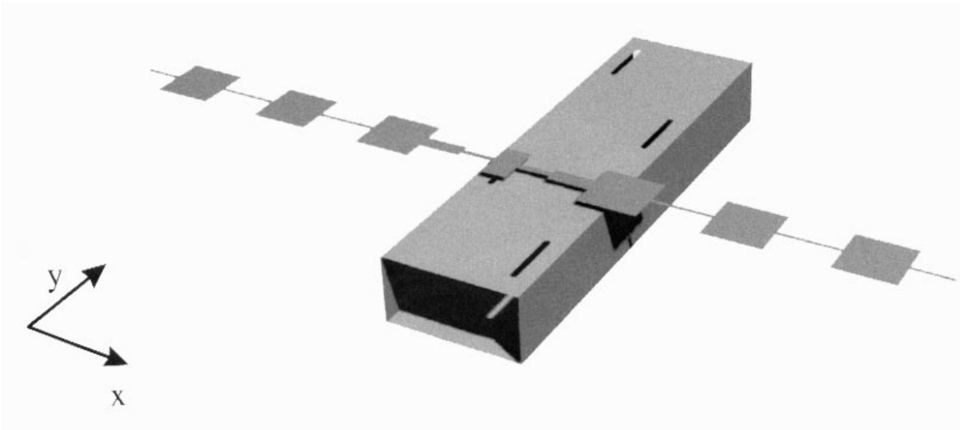


Figure 5.5/3: Detail of the HCPA antenna showing the geometry of the coupling of the waveguide slots to a line of radiating patches in series

5.5.1.1.3.2 Rectenna type 2

The rectenna consists of a planar assembly of 18cmx18cm square slotted waveguide array antennas. Each array delivers a power to the input port of a rectifying circuit whose role is to convert the incoming RF power into a DC electric power. It is assumed that the power density is almost constant on the overall rectenna aperture and consequently each elementary panel delivers the same voltage and power to a common load (or power management unit). Slotted array waveguide antennas have already been extensively studied. However, not many designs have been demonstrated at 35GHz. We use data from ([6] Sakakibara et al., 1996) to estimate the possible performance of such antenna at 35GHz.

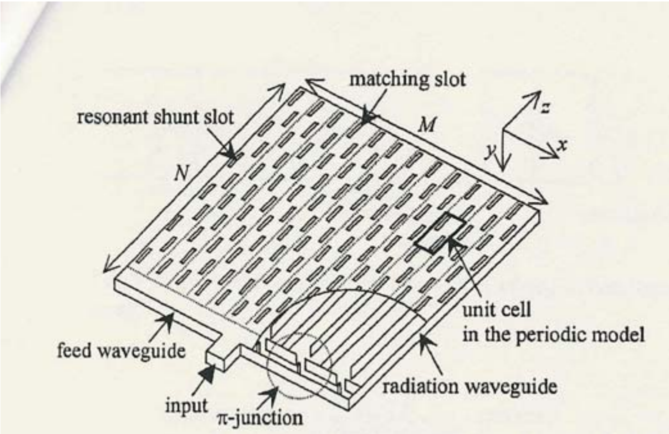


Figure 5.5/4: Schematic representation of a flat slotted waveguide array antenna (from [6]).

Figure 5.5/5: Example of a slotted waveguide antenna designed using a metallized foam technique. (from [5])

Fabrication technology and weight being a big concern, we would like to mention the work of (Himdi et al., [5]) at University of Rennes-IETR (France). They have designed and patented a technology called metallized foam to fabricate dielectric loaded waveguides and they recently demonstrated a 35GHz slotted waveguide using this technique, exhibiting efficiency in the order of 70%. This technology is now under production by the company "Antennes-Process" (35235 Thorigné-Fouillard, France). It is not proven, however, that it can be space qualified for Space. Perhaps a material choice should be conducted to find a light dielectric that could provide good mechanical and thermal properties and allow for its use in Space.

Using these techniques, it can be estimated that a 35dB gain antenna can be built with an efficiency of approximately 75%. This antenna would size approximately 18cmx18cmx2cm, and weight only 80g (without rigid backplane). Coupled to a rectifying circuit optimized to work at 150mW, a rectenna element with the following characteristics is obtained:

Property	Value	Comments
Antenna technology	Metallized foam flat slotted waveguide array antenna	
Antenna gain	35dBi	
Antenna efficiency	75%	
Antenna dimensions	18cmx18cmx2cm	Estimated from ().
Antenna mass (without rigid backplane)	80g	Antenna effective mass 3kg/m2
Rectifying circuit	Schottky diode and filters	
Optimum input power	150mW	
RF to DC conversion efficiency	75%	
Optimized RF power density	**10.75 W/m2**	

Table 5.5/2: Main characteristics of rectenna element of type 2.

5.5.1.1.3.3 Rectenna type 3

This rectenna is constructed using a Schottky diode rectifying circuit optimized to operate at 50mW input power. The antenna element used is a slotted waveguide antenna as already presented in the rectenna type 1 description. The main characteristics of this rectenna element are listed in the following table:

Property	Value	Comments
Antenna technology	Metallized foam flat slotted waveguide array antenna	
Antenna gain	35dBi	
Antenna efficiency	75%	
Antenna dimensions	18cmx18cmx3cm	
Antenna effective area	186 cm^2	
Antenna mass (without rigid backplane)	80g	Antenna effective mass 3kg/m2
Rectifying circuit	Schottky diode and filters	
Optimum input power	50mW	
RF to DC conversion efficiency	75%	
Optimized RF power density	**3.58 W/m2**	

Table 5.5/3: Main characteristics of rectenna element of type 3.

5.5.1.1.3.4 Rectenna type 4

It is identical to rectenna type 3 but constructed using a Schottky diode rectifying circuit optimized to operate at only 21mW input power. Expected RF to DC efficiency is only 65% because of the low input power. The antenna element used is a slotted waveguide antenna as already presented in the rectenna type 1 description. The main characteristics of this rectenna element are listed in the following table:

Property	Value	Comments
Antenna technology	Metallized foam flat slotted waveguide array antenna	
Antenna gain	35dBi	
Antenna efficiency	75%	
Antenna dimensions	18cmx18cmx3cm	
Antenna effective area	186 cm^2	
Antenna mass	80g	Antenna effective mass 3kg/m2
Rectifying circuit	Schottky diode and filters	
Optimum input power	21mW	
RF to DC conversion efficiency	65%	
Optimized RF power density	**1.51 W/m2**	

Table 5.5/4: Main characteristics of rectenna element of type 4.

5.5.1.2 Dimensioning cases

5.5.1.2.1 Main specifications

Property	Value	Comment
Average distance between SPS and Rectenna on ground	17000 km	
Mars base power need	100 kW DC power	
Beam steering	Fixed beam	

Table 5.5/5: Main specifications of mission case consisting of providing power to elements on Mars, SPS being in areosynchronous orbit.

5.5.1.2.2 Rectenna type 1

When rectenna type 1 is used, an incoming power density of 724W/m² should be present at the rectenna aperture.

Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 150mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 75%. Each element then delivers a DC output power of 110mW.

Each rectenna type 1 element occupies a surface of 49cm² and then, each square meter of rectenna delivers a DC output power of P_{units} = 22,45 W.

To provide the 100 kW needed it is then necessary to deploy a total surface of:

$$S_{total} = 100\ 000/22{,}45 = 4454 \text{ sq. m.}$$

This corresponds to a circular area of diameter: D_c = 75 m

Table 5.5/6 summarizes the main estimated characteristics of the rectenna area with a rectenna type 1.

Property	Value	Comment
Rectenna	Rectenna type 1	High power density rectenna
Power density at rectenna	724 W/m²	
Efficiency of antenna	70%	
RF power at rectifying circuit	150mW	
RF to DC conversion efficiency	75%	
DC output power	110mW	
DC output power density	22.45 W/m2	
Surface of rectenna needed for 100 kW total DC output power	4454 m²	
Diameter of corresponding circular area	75 m	

Table 5.5/6: Main estimated characteristics of the rectenna area with a rectenna type 1.

To produce an RF power density of 724 W/m² at a 17000 km distance, the projecting antenna aperture of diameter D_p and transmitting an RF power of P_{tr} must satisfy the following relation, if it is illuminated with a 10dB Gaussian taper:

$$dP = 0{,}709\ \frac{P_{tr} * D_p^2}{\lambda^2 * r^2}$$

In this relation dP is the produced RF power density, λ is the wavelength (8,6 mm at 35 GHz) which and r is the distance (17000km in this case).

Assuming a 1MW of projected power, an antenna diameter of 4.7km is needed. If we now assume a 4MW projected power, a 2.3km diameter antenna is needed.

Table 5.5/7 lists the estimated characteristics of the RF projecting unit with a rectenna type 1.

Property	Value	Comment
Option 1 (Low power large diameter antenna)		
Projected power	1 MW	
Antenna diameter	4.7 km	
Option 1 (High power lower diameter antenna)		
Projected power	4 MW	
Antenna diameter	2.3 km	

Table 5.5/7: Main characteristics of the RF projecting unit with a rectenna type 1.

5.5.1.2.3 Rectenna type 2

When rectenna type 2 is used, an incoming power density of 10.75 W/m2 should be present at the rectenna aperture.

Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 150 mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 75%. Each element then delivers a DC output power of 110mW.

Each rectenna type 1 element occupies a surface of 320cm² and then, each square meter of rectenna delivers a DC output power of P_{units} = 3.438 W.

To provide the 100 kW needed it is then necessary to deploy a total surface of:

$$S_{total} = 100\,000/3.438 = 29090 \text{ m}^2.$$

This corresponds to a circular area of diameter: D_c = 192.5 m

Table 5.5/8 below summarizes the main estimated characteristics of the rectenna area with a rectenna type 2.

Property	Value	Comment
Rectenna	Rectenna type 2	*Medium power density rectenna*
Power density at rectenna	10.75 W/m²	
Efficiency of antenna	75%	
RF power at rectifying circuit	150mW	
RF to DC conversion efficiency	75%	
DC output power	110mW	
DC output power density	3.438 W/m²	
Surface of rectenna needed for 100 kW total DC output power	29090 m²	
Diameter of corresponding circular area	192.5 m	

Table 5.5/8: Main estimated characteristics of the rectenna area with rectenna type 2.

To produce an RF power density of 10.75 W/m² at a 17000 km distance, the projecting antenna aperture of diameter D_p and transmitting an RF power of P_{tr} must satisfy the following relation, if it is illuminated with a 10dB Gaussian taper:

$$dP = 0{,}709 \frac{P_{tr} * D_p^2}{\lambda^2 * r^2}$$

In this relation *dP* is the produced RF power density, λ is the wavelength (8,6 mm at 35 GHz) which and *r* is the distance (17000km in this case).

Assuming a 2MW projected power, an antenna diameter of 400m is needed.

Property	Value	Comment
Projected power	2 MW	
Antenna diameter	400 m	

Table 5.5/9: Main characteristics of the RF projecting unit with rectenna type 2.

5.5.1.2.4 Rectenna type 3

When rectenna type 3 is used, an incoming power density of 3.58W/m² should be present at the rectenna aperture.

Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 50mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 75%. Each element then delivers a DC output power of 37.5mW.

Each rectenna type 3 element occupies a surface of 320cm² and then, each square meter of rectenna delivers a DC output power of P_{units} = 1.17 W.

To provide the 100 kW needed it is then necessary to deploy a total surface of:

$$S_{total} = 100\,000/1.17 = 85470 \text{ m}^2.$$

This corresponds to a circular area of diameter: D_c = 329 m

Table 5.5/10 summarizes the main estimated characteristics of the rectenna area with a rectenna type 3.

Property	Value	Comment
Rectenna	Rectenna type 3	*Low power density rectenna*
Power density at rectenna	3.58 W/m²	
Efficiency of antenna	75%	
RF power at rectifying circuit	50mW	
RF to DC conversion efficiency	75%	
DC output power	37.5mW	
DC output power density	1.17 W/m²	
Surface of rectenna needed for 100 kW total DC output power	85470 m²	
Diameter of corresponding circular area	329 m	

Table 5.5/10: Main estimated characteristics of the rectenna area with a rectenna type 3

To produce an RF power density of 3.58 W/m² at a 17000 km distance, a projecting antenna diameter of 232 m is needed assuming a 2MW projected power.

Property	Value	Comment
Projected power	2 MW	
Antenna diameter	232 m	

Table 5.5/11: Main characteristics of the RF projecting unit with a rectenna type 3.

5.5.1.2.5 Rectenna type 4

With rectenna type 4 is used, an incoming power density of 1.51W/m² should be present at the rectenna aperture. Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 21 mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 65%. Each element then delivers a DC output power of 13.65 mW.

Each rectenna type 4 element occupies a surface of 320cm² and then, each square meter of rectenna delivers a DC output power of P_{units} = 0.42 W.

To provide the 100 kW, it is necessary to deploy a total surface of: S_{total} = 100 000/0,42 = 234432 m².

This corresponds to a circular area of diameter: D_c = 546 m

Table 5.5/12 summarizes the main estimated characteristics of the rectenna area with a rectenna type 4.

Property	Value	Comment
Rectenna	Rectenna type 4	*Very Low power density rectenna*
Power density at rectenna	1.51 W/m²	
Efficiency of antenna	75%	
RF power at rectifying circuit	21mW	
RF to DC conversion efficiency	65%	
DC output power	13.65mW	
DC output power density	0.42 W/m2	
Surface of rectenna needed for 100 kW total DC output power	234432 m²	
Diameter of corresponding circular area	546 m	

Table 5.5/12: Main estimated characteristics of the rectenna area with a rectenna type 4.

To produce an RF power density of 1.51 W/m² at a 17000 km distance, a projecting antenna diameter of 151 m is needed assuming a 2MW projected power.

Property	Value	Comment
Projected power	2 MW	
Antenna diameter	151 m	

Table 5.5/13: Main characteristics of the RF projecting unit with rectenna type 4

5.5.1.3 On-board transmitting system description

In this section we give details on the technology that could be used in transmitting systems corresponding to two of the above discussed cases examined in section 5.5.1.2. These two dimensioning cases are:

- Dimensioning with rectenna type 2
- Dimensioning with rectenna type 4

5.5.1.3.1 Dimensioning with rectenna type 2

Table 5.5/9 lists the main characteristics that the projecting unit should exhibit when rectenna elements of type 2 are used to deliver 100kW of DC power to a large infrastructure on Mars. The average power density projected by the antenna is 16W/m², and the maximum power density at the antenna center is 40W/m². The antenna can be made of waveguide flat slot arrays (high gain and high efficiency) equivalent to the one used for the rectenna on ground. Using a metallized foam realization technique these antennas can be made light and compact.

The RF power can be delivered by a MPM (Microwave Power Module). One possible off the shelf reference for this application is the M1280 from L-3 Communications Electron Devices (Figure 5.5/6). The required voltage at the MPM input is 270 VDC and the power consumed by each MPM is 200 W (efficiency > 25%). Of course, this is current technology and off the shelf component. It may be possible to increase the efficiency to perhaps 50% on this kind of product for specific designs in the future.

MPM Type Number	Frequency Range (GHz)	RF Output Power (W)	Dimensions (inches)	Weight (lbs)	Prime Power (VDC)
M1200	2.0 to 8.0	50 to 100	14.0 x 4.0 x 1.2	5.0	270
M1220	6.0 to 18.0	60 to 100	7.5 x 6.25 x 1.0	3.75	270
M1221	6.0 to 18.0	60 to 100	7.85 x 6.5 x 1.0	4.0	28
M1280	18.0 to 40.0	20 to 60	14.0 x 3.0 x 1.25	4.0	270
M1340	40.0 to 46.0	25	14.0 x 3.0 x 1.25	4.0	270

Figure 5.5/6: Example of MPM references taken from L-3 Communication catalogs.

Taking into account the 65% efficiency of waveguide slot array antennas and the 25% efficiency of the MPM modules, the total power that must be supplied to the MPM modules is 2MW/(0.65×0.25)=12.3MW.

The number of modules needed is 12.3MW/200W = 61500. The antenna has a 400m diameter. It consists of 61500 1.4mx1.4m waveguide slot array elements, each fed by one MPM module.

The mass of each module can be estimated from the mass of the antenna and the mass of the associated MPM module. A supplementary 200g is added to account for retrodirective pilot sensor and phase conjugation circuitry (Sensor, PLL, phase conjugation, amplifier). This gives:

Mass of antenna element:	2 m2 × 3kg/m² = 6 kg
Mass of backing plate	2 m2× 3kg/m² = 6 kg
Mass of MPM module:	2kg
Mass of retrodirective circuitry:	0.2kg
Total mass/module:	14.2kg

This mass can be used to estimate the projecting equipment mass, not taking into account the supporting frame mass and any thermal management equipment. The total RF equipment mass is 61500×14.2kg = 873 tons.

Table 5.5/14 lists the main properties of the described projecting unit.

Property	Value	Comments
Dimensioning case using rectenna type 2		
RF power projected	2MW	
Antenna diameter	400	
Average projected power density	16W/m2	
Maximum projected power density	40W/m2	
Type of antenna modules	Waveguide flat slot array antenna	
Antenna efficiency	65%	
Size of modules	1.4mx1.4mx3cm	
Mass of antenna module	12kg	*Including backing plate*
Type RF generator	MPM	*M1280 from L-3 Communications or similar.*
RF generator efficiency (including EPC)	25%	
Mass of RF generator	2kg	
Phasing mechanism	Retrodirective phase conjugation	
Phasing circuit mass	0.2kg	
Number of modules	61500	
Total mass of projecting unit	873 tons	
Total power required at input	12.3 MW	

Table 5.5/14: List of main properties of a possible projecting unit in case of rectenna element of type 2.

5.5.1.3.2 Dimensioning with rectenna type 4

Table 5.5/13 lists the main characteristics that the RF projecting unit on board the SPS should exhibit to provide the necessary power density required on ground when rectenna elements of type 4 are used to provide the 100kW of DC power to the infrastructure on Mars.

Antenna elements may be made from waveguide slot array as in the previous case.

The average power density projected by the antenna is 146W/m², and the maximum power density at the antenna center is 374W/m².

Using antennas with 75% efficiency, this necessitates 194 W of RF power at the input of a 1 m² antenna. Such a power can be delivered by two high efficiency TWTs for each sq. meter of projecting antenna. One off the shelf reference from Thales Electron Devices TH4046 exhibits specifications very close to the one that should be used for this application. Such a device could deliver 100 W at 35GHz and weights only 0.8kg. Efficiency can be estimated to be around 50%. Power Conditioner with efficiencies of about 85% could be designed to supply power to this tube from a single DC voltage source, drawing approximately 235 W. We would need about 2 such devices for each sq. meter of projecting antenna. The number of devices needed can be estimated from the total projected power and the efficiency chain from the input power to the projected RF power. The input power should be 2MW/(0.75×0.5×0.85)=6.27MW.

With this in mind, each source drawing 235 W, we need 6.27MW/235W = 26681 RF projection modules.

Each module feeds an antenna of surface 0.5m².

The mass can be estimated from the mass of the antenna modules, the mass of the TWT, the mass of the electronic power conditioner and the mass of the retrodirective phase conjugation circuits:

Mass of antenna:	0.5×3kg/m² = 1.5kg
Mass of backing plate:	0.5×3kg/m² = 1.5kg
Mass of TWT:	0.8kg
Mass of EPC:	0.8kg
Mass of phasing circuitry:	0.2kg
Total mass/module:	**4.8kg**

The total mass of the projecting antenna is 26681×4.8kg=128 tons, without the supporting frame and necessary thermal management equipment.

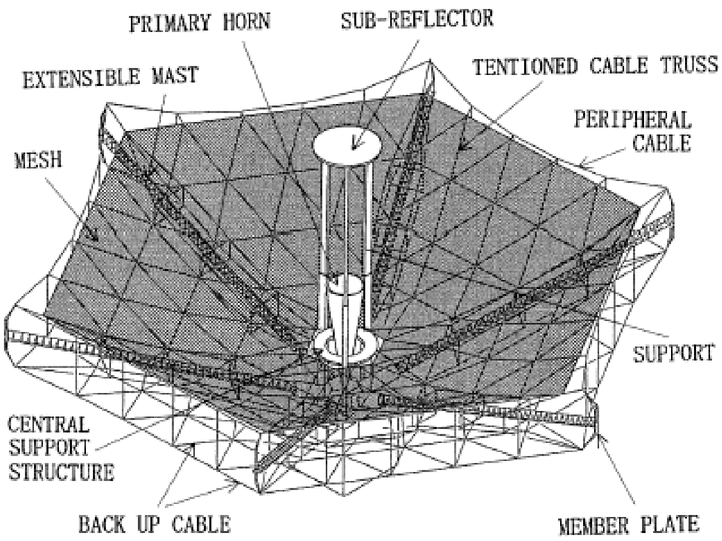
The Table 5.5/15 below lists the most important properties of the projecting unit described in this section.

Property	Value	Comments
Dimensioning case using rectenna type 2		
RF power projected	2MW	
Antenna diameter	132	
Average projected power density	146W/m2	
Maximum projected power density	374W/m2	
Type of antenna modules	Waveguide flat slot array antenna	
Antenna efficiency	75%	
Size of modules	0.7mx0.7mx3cm	
Mass of antenna module	3kg	*Including backing plate*
Type RF generator	TWT	*TH4046 from Thales EDD or similar*
RF generator efficiency (including EPC)	42.5%	
Mass of RF generator	1.6kg	
Phasing mechanism	Retrodirective phase conjugation	
Phasing circuit mass	0.2kg	
Number of modules	26681	
Total mass of projecting unit	128 tons	
Total power required at input	6.27 MW	

Table 5.5/15: List of main properties of a possible projecting unit for a dimensioning case using rectenna elements of type 4.

5.5.1.4 Other technologies for RF transmitter

5.5.1.4.1 Large deployable reflector antennas at millimeter wave frequencies



Large deployable antenna constitution ([13] Takano et al., 2002).

5.5.1.4.2 Retrodirective phase-conjugation at millimeter wave frequencies.

Some considerations on phase conjugation system performance at 35GHz.

5.5.1.5 Performances

In this part of the report, we provide a performance analysis of the proposed solutions for the two cases studied in the section 5.5.1.3 of this document. The objective of this study is to evaluate the sources of energy loss in the power transmission chain.

5.5.1.5.1 System dimensioned with rectenna type 2

The largest source of losses is during the propagation of the beam. This comes from the fact that we use a small rectenna array and only capture a small amount of the projected power. We could get much better results by using a larger rectenna and getting more power out of it, but for a given output power level, the only way to get a better collection efficiency would be to have a larger transmitting antenna, projecting a lower overall RF power. Because the projected power would be less, a lower area of PV cells would be necessary. However, an increase in the projecting antenna size requires a larger platform and deployment issues.

Chain element	Power in	Power out	Efficiency
Rectifying circuit	133.33kW	100 kW	75%
Collecting antenna	178kW	133.33kW	75%
Propagation	2MW	178kW	**8.88%**
Projecting antenna	2.67MW	2MW	75%
RF Generator	10.68MW	2.67MW	25%
Total WPT system	10.68MW	100kW	0.93%

Table 5.5/16: Listing of all sources of energy loss for a WPT system providing power to an infrastructure on Mars, using type 2 rectenna elements.

5.5.1.5.2 System dimensioned with rectenna type 4

Here again, the largest part of the losses comes from the RF beam capture by the collecting rectenna. The best solution would be to increase the size of the projecting antenna.

Chain element	Power in	Power out	Efficiency
Rectifying circuit	154kW	100 kW	65%
Collecting antenna	205.3kW	154kW	75%
Propagation (attenuation and collection efficiency)	2MW	205.3kW	**10.26%**
Projecting antenna	2.67MW	2MW	75%
RF Generator	5.34MW	2.67MW	50%
EPC	6.28MW	5.34MW	85%
Total WPT system	6.28MW	100kW	1.59%

Table 5.5/17: Listing of all sources of energy loss in the WPT system providing power to an infrastructure on Mars, using type 4 rectenna elements.

5.5.2 Power generation system definition

5.5.2.1 Architecture

The SPS payload is defined by a high number of TWT and its associated converter (EPC). The assumptions taken into account for the definition of the power architecture correspond to the initial RF power transmission system sizing: 27370 sets of TWT/EPC, with a power consumption of one converter equal to 190 W, leading to a total SPS consumption of 5.2 MW, and assuming some 10 kW for the other SPS subsystems functions.

Such a power requires a high voltage bus. As already proposed, the maximum manageable voltage is about 1 kV. Higher voltages lead to high problem of component availability and arcing in the cables, connectors, components and solar cells. 1 kV is already critical with regards to this problem.

As for the rover mission (5.4.2), a sequential Serial/shunt Switching Regulator architecture can be used.

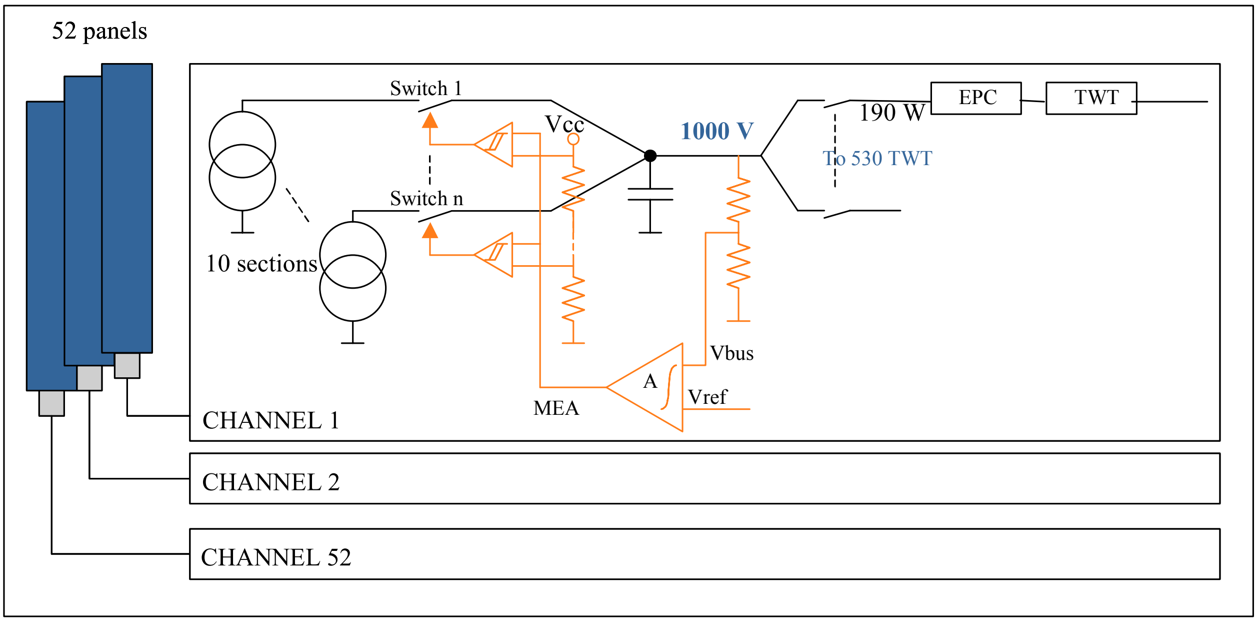
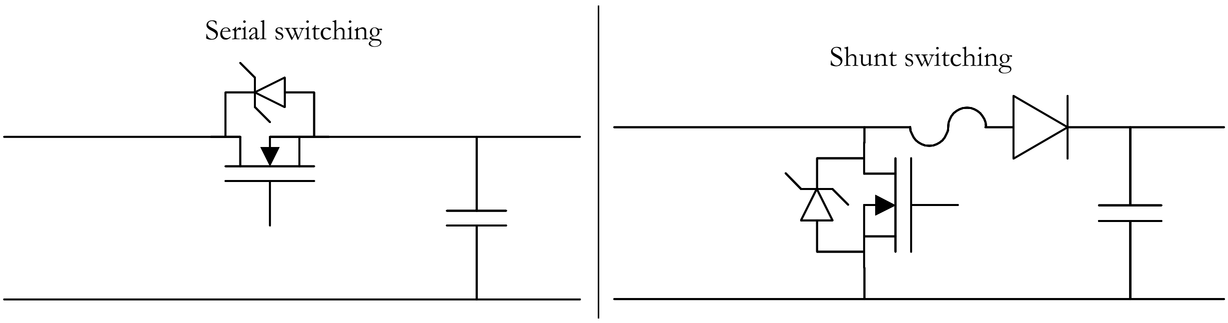


Figure 5.5/7: Power architecture for the Mars base application

The switch can be either a serial switch or a shunt. The choice between the two topologies is driven by the component derating.



In the case of the serial switching, the transistor shall be designed for a voltage equal to the maximum SA open circuit voltage minus the minimum guaranteed voltage of the bus. For a regulated 1 kV bus, the SA open circuit voltage will be about 1400 V. Assuming a minimum guaranteed bus of 900 V, this means a voltage across the transistor of 500V compatible with already available (but not space qualified) Mosfet and IGBT, assuming a derating factor of 50%.

For the shunt switching topology, the shunt transistor, the diode and the fuse (or redundant diode) shall be sized for a voltage equal to the maximum bus voltage, leading to 2 kV components in order to respect the derating requirements. No transistor (Mosfet and IGBT) has been found that meets this voltage requirement.

As a consequence, the serial switching topology is preferred. IGBT technology is also preferred because of their higher current capability than with Mosfet of the same packaging. As an example (TO-247 package),

- Mosfet IRFG50 from International Rectifier: 1000 V, 3.9 A @ 100°C
- IGBT IRGP20B120U-E from International Rectifier : 1200 V, 20 A @ 100°C
- IGBT SGW25N120 from Infineon : 1200 V, 25 A @ 100°C

Voltage drop will be about 3.5V leading to an efficiency of 99.6%. For a distributed power of 5.2 MW, the regulator loss is equal to 21 kW. This high dissipation is a major concern of very high power application.

5.5.2.2 Solar array

Trade-off between solar cells and Dish Stirling technologies is given here below:

	SA with small concentrators	Dish Stirling thermal machine
Collect area (m^2)	35120	30100
Mass (tons)	53	151

To close this trade-off, important elements should be added like:

- Impact on launch and SPS configuration
- In-flight constraints like pointing requirements and thermal,
- In-flight assembly complexity,
- Technology maturity and growth potential.

Only the SA with small concentrator solution is studied, main results being also applicable to the Dish Stirling solution.

The SA can be spit in several (52) panels of about 100 kW each. In order to avoid the management of 5.2 MW in only one point, it is proposed to split not only the SA but also the power management and distribution up to the TWTA.

This means that one panel, providing 100 kW will be managed by one dedicated electronic box, and distributed to about 530 TWTA. This makes easier the management of the solar array as each panel can be spit in about 10 sections (ten switches).

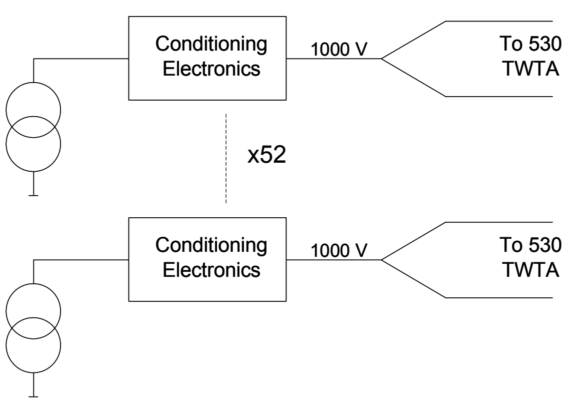


Figure 5.5/8: Solar array sharing

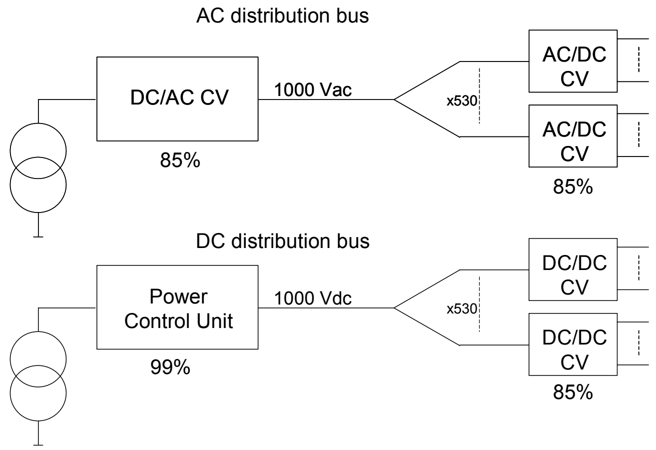


Figure 5.5/9: Power distribution architecture

5.5.2.3 Distribution

A preliminary trade-off between AC and DC distribution is presented on Figure 5.5/9.

The AC distribution requires an inverter that transforms the DC voltage of the solar array to AC. Then a converter (generally called EPC) allows to regulate the secondary voltages to the required ones (several high voltages typically from 500 to 4500 V). The DC distribution requires a power management unit as described in the section 5.5.2.1 and the EPC.

The requirements of power bus quality on the secondary voltages leads to the need of EMI filtering and voltage regulation. For this reason, the efficiency of the AC/DC converter cannot be significantly improved with regards to the DC/DC converter.

In term of overall efficiency, one can see that the DC bus solution is better. Other factors have to be considered like the skin effect on the cables and the EMC constraints. Assuming a frequency of 50 kHz for the AC bus, the thickness of the skin effect is about 300 µm. Cables shall therefore be chosen to maximize the perimeter and/or shall be made of multiple individual wires covered with silver. The power bus cleanliness with respect to EMC is also a problem in term of spurious emissions in the power bus frequency.

For these reasons, the DC bus technology is considered preferable for our application.

In term of harness, taking into account the considered lengths (around 250 m from the solar cells to the antenna's foot and a 232 m diameter antenna that supplies 27370 tubes), a budget of 20 tons has been estimated

5.5.2.4 Impact of last RF power transmission system sizing

The last sizing of the RF power transmission system leads to the utilisation of 26681 TWT/EPC, each TWT capable of 100 W RF. The power consumption of one converter (power available at the EPC input) is 235 W, leading to a total power at the output of the solar arrays of 6,27MW. The power system architecture would remain identical, with probably a sharing of the solar arrays in more channels.

5.5.3 Preliminary definition of SPS

The SPS definition is driven by the need for a 132 m diameter antenna emitting 2 MW at 35 GHz, and for solar arrays (or solar surfaces) generating 6,28 MW.

5.5.3.1 SPS with parabolic antennae

The utilisation of parabolic antenna (parabolic reflectors illuminated with sources) leads to consider a set of 8 deployable antennae of 50m diameter. Indeed, a 132 m diameter at this frequency is too large with respect to surface accuracy control. Maximum diameter at this frequency is around 50 m. The implementation of 8 antennae allows to have an equivalent to the 132 m antenna.

Therefore, 8 identical modules have been defined and are implemented around a central core. Each module, as illustrated on Figure 5.5/10, includes the antenna, one set of sources (typically 64) illuminating the opposite antenna, the TWT/EPC and associated electronics, a deployable radiator of about 1000 m², an active thermal loop.

The central core relies a long truss structure on which are mounted the 8 antenna modules. The antenna is deployed at about 50 m from the module. The central core includes a primary reflector, located in front of the modules that reflects the signal from the sources of a module to the main reflector of the opposite module. It includes also all the electronics for the command control of the SPS, part of the power subsystem, the communications equipment (X band and S band as in the case of the SPS for the rover), and the electrical propulsion with a capability of about 100 N. It also support the truss on which are mounted the solar surfaces.

The solar arrays are mounted in North/South axis along the truss (about 320 m). The truss allows to remove the solar arrays from the potential occultation due to the large appendages (radiators and antennae). They have two degrees of freedom: one for orbit compensation (360°), another one for control of disturbances along the orbit.

The SPS is illustrated on Figure 5.5/14

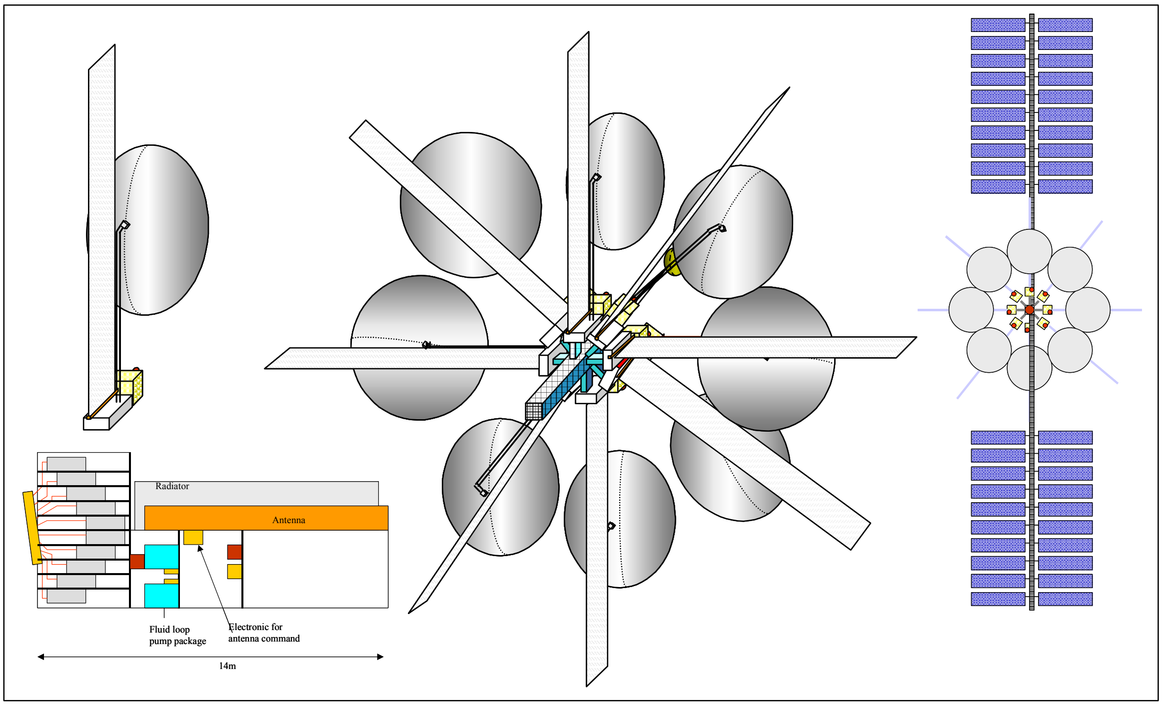


Figure 5.5/10: SPS definition with parabolic antennae

5.5.3.2 SPS with slot array antennae

A single waveguide flat slot array antenna of 132 m can be used. As illustrated on Figure 5.5/4 and Figure 5.5/11, the EPC/TWT is integrated to the waveguide element, with associated electronic and with its radiator. The backside of the antenna is sufficient to implement the necessary radiator surface. However, all the radiator has to be protected against Sun with baffles.

The antenna with its element is mounted on a central truss structure, as illustrated Figure 5.5/12. This structure houses the electronics for SPS command and control, part of the power subsystem, the communications system and the electrical propulsion with a 100N thrust capability. The solar arrays are mounted in North/South axis along a truss structure of 270m length. They are composed of a serie of panels. The truss has a first part (75m) free of solar panels in order to clear the sun visibility and avoid any potential occultation by the antenna. The solar arrays have two degrees of freedom: one for orbit compensation (360°), another one for control of disturbances along the orbit.

The SPS is illustrated on Figure 5.5/14.

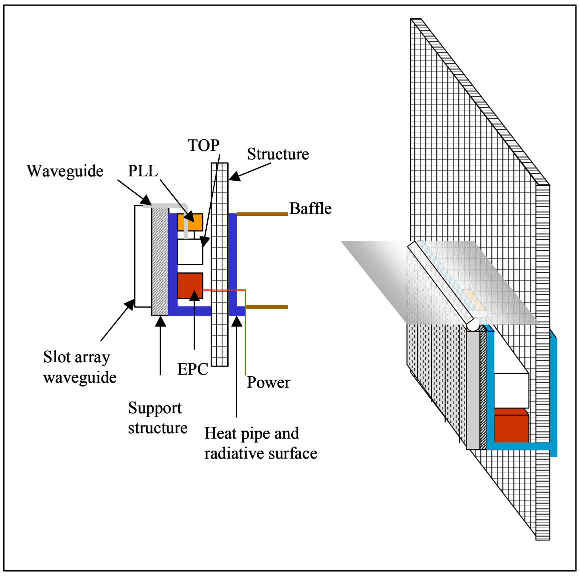


Figure 5.5/11: Illustration of an element of the slot array antenna

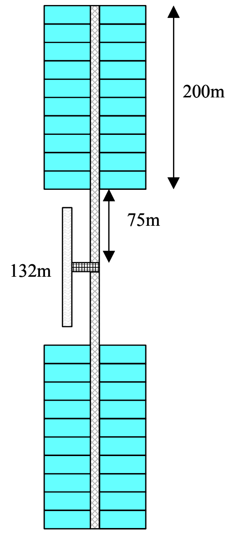


Figure 5.5/12: Schematic of the SPS with slot array antenna

5.5.3.3 SPS system overview and performances

The main performances of the SPS system are illustrated on Figure 5.5/13. To deliver 6,28 MW, the solar arrays have a size of about 42350 m2. The receiver area (rectenna) has a diameter of 546 m. The end-to-end efficiency is about 1,6%. The SPS mass is in order of 500t, with current or reasonably achievable technology.

The SPS electrical propulsion ensures its transfer from the LEO to the Mars orbit. However, The SPS has to be launched in several parts and assembled in LEO.

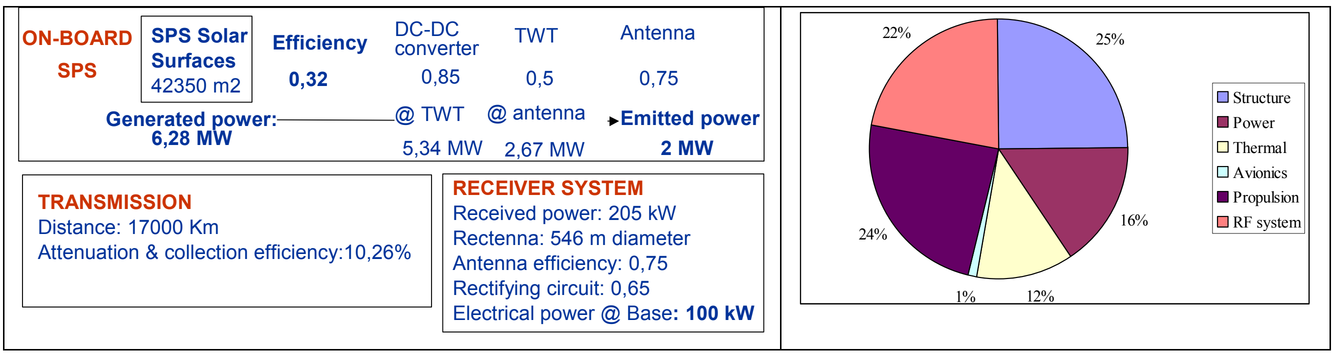


Figure 5.5/13: SPS system performances overview

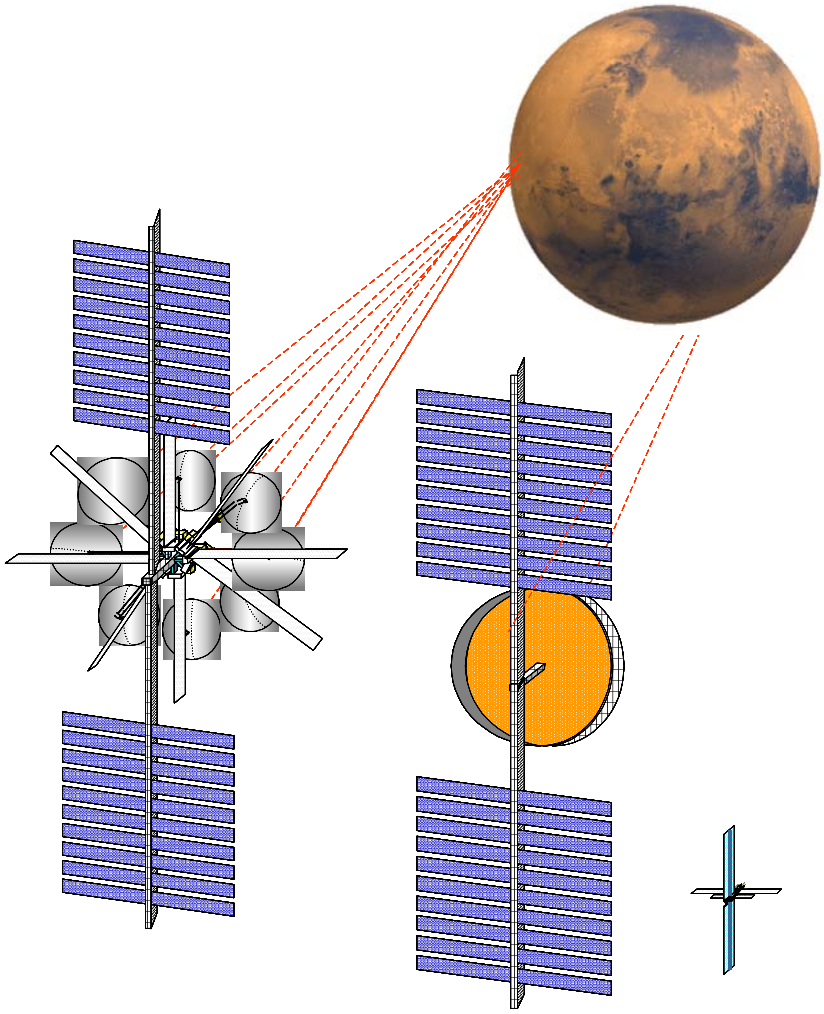


Figure 5.5/14: Illustration of the SPS configurations for power transmission to Mars base (small one is the SPS for power delivery to rover)

6 PLANETARY MISSIONS: ELEMENT ON MOON SURFACE

6.1 REQUIREMENTS

6.1.1 Target requirements

The element on the Moon would be a scientific infrastructure with observatory elements. This infrastructure would be implemented on an area of about 1 km diameter. The need in terms of power is about 20 kW. The element will be in eclipse during a maximum of roughly 14 days, depending on its latitude. The objective of the SPS is to provide power at least during this eclipse period. There is no constraint on the size of the receiver surface.

6.1.2 Moon environment

Main characteristics

Mass: 7.35×10^{22} kg (0.01230002 Earth mass)

Mass times gravitational constraints: 4909.793 km3/s2

Surface gravity: 1.622 m/s2

Mean equatorial radius: 1737.4 km

Temperature: + 130°C at equator with Sun at zenith

-200°C in permanently shadowed areas (near poles)

Moon orbit

Semi major axis: 384401 km

Apogee radius: 406700 km

Perigee radius: 356400 km

Inclination of orbit to ecliptic: 5.145°

Inclination of orbit to equator: 6.679°

Sidereal period (fixed star to fixed star): 27.32 days

Synodic period (new Moon to new Moon): 29.53 days

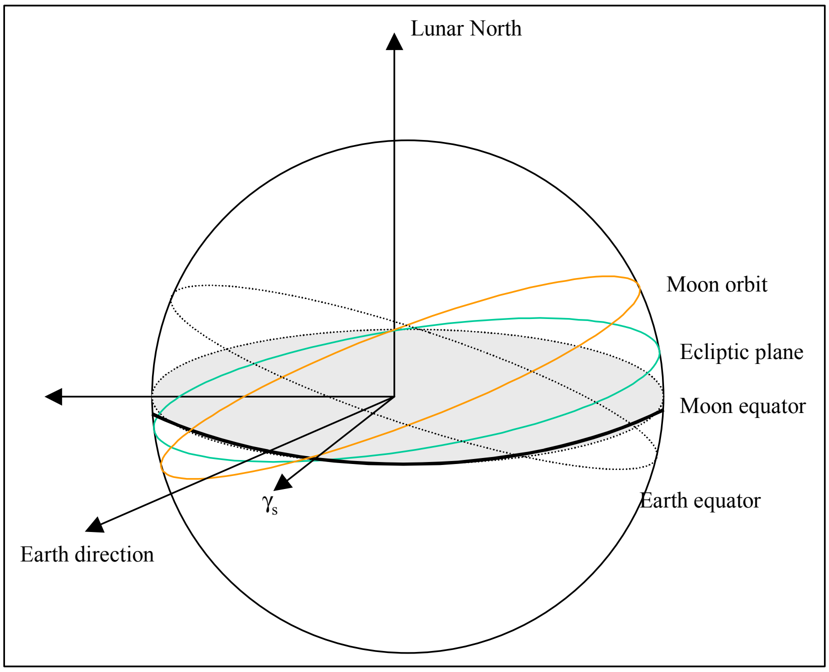


Figure 6.1/1: Moon characteristics

6.2 SPS POSITIONING

As illustrated on Figure 6.2/1, three possible locations can be considered for the SPS:

- On an orbit around Moon at a distance allowing a long visibility period (typically 5000 km) and minimising the size of the power transmission system
- At an Earth-Moon Lagrangian point L1 or L2
- At the Earth-Sun Lagrangian point L2

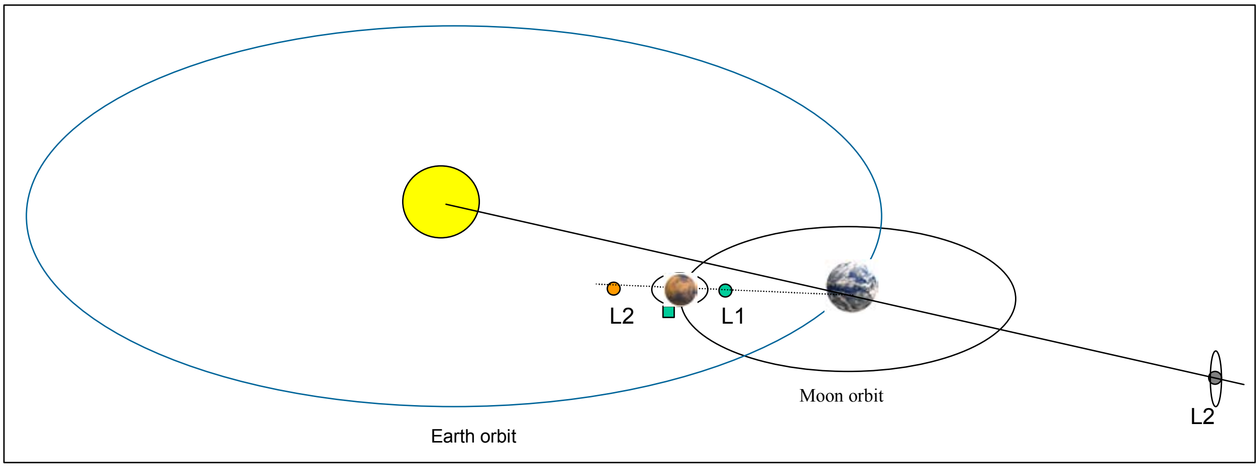


Figure 6.2/1 Possible locations of SPS

6.2.1 Equatorial orbit around Moon

A geostationary orbit would lead to a high distance between SPS and target. It is proposed to position the SPS on an orbit close to equatorial, at an altitude allowing a long duration of visibility of the ground element and not penalising the power transmission performance.

For instance, an altitude of 5000 km would lead to a SPS orbit period of about 14h, and a duration of power transmission of about 4,5h with a beam steering angle varying between –15 and +15°.

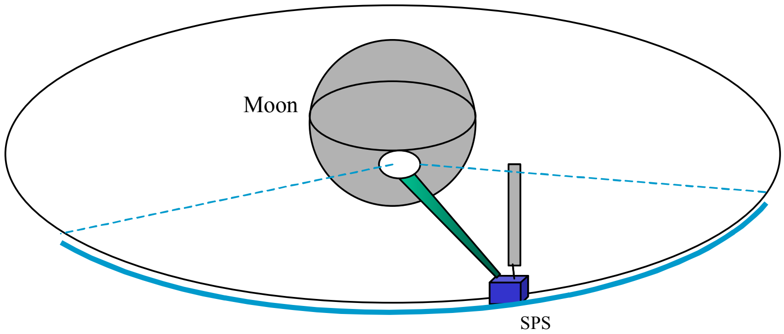


Figure 6.2/2: Case of equatorial orbit

However, to minimize the eclipse period (due to Moon occultation), the SPS orbit plane should have a low inclination, of at least 10°. Indeed, due to the low angle between the Moon equatorial plane and the ecliptic plane (about 5°), the SPS would have an eclipse period per orbit if in equatorial plane of with less than 10° inclination. Some eclipses could occur in particular configuration, but for a short time.

This positioning allows to illuminate all the elements located between –75 and +75° latitude. If the element is located at high latitudes, near the poles, a quasi polar orbit will be preferred.

This SPS positioning leads to an availability of the power to the Moon element of 33% of a day, so that a set of three satellites would be necessary to provide permanently power to the Moon element, except on high latitudes.

6.2.2 Earth-Moon Lagrangian points

Earth-Moon Lagrangian points which can be considered are L1, L2 on one part, L4/L5 on another part. Their respective positions and illumination are illustrated on Figure 6.2/3.

L1 and L2

L1 and L2 are located symmetrically with respect to Moon along the Moon-Earth axis at an average distance of about 58000 km. When positioned at L1, the SPS provides permanently power to an element located on the Moon side facing Earth. As shown on Figure 6.2/3, the SPS illuminates the element both during eclipse phase and during sunlight phase. From this point, the SPS can illuminate almost half the Moon: up to 88° latitude, and from –88° to +88° in longitude.

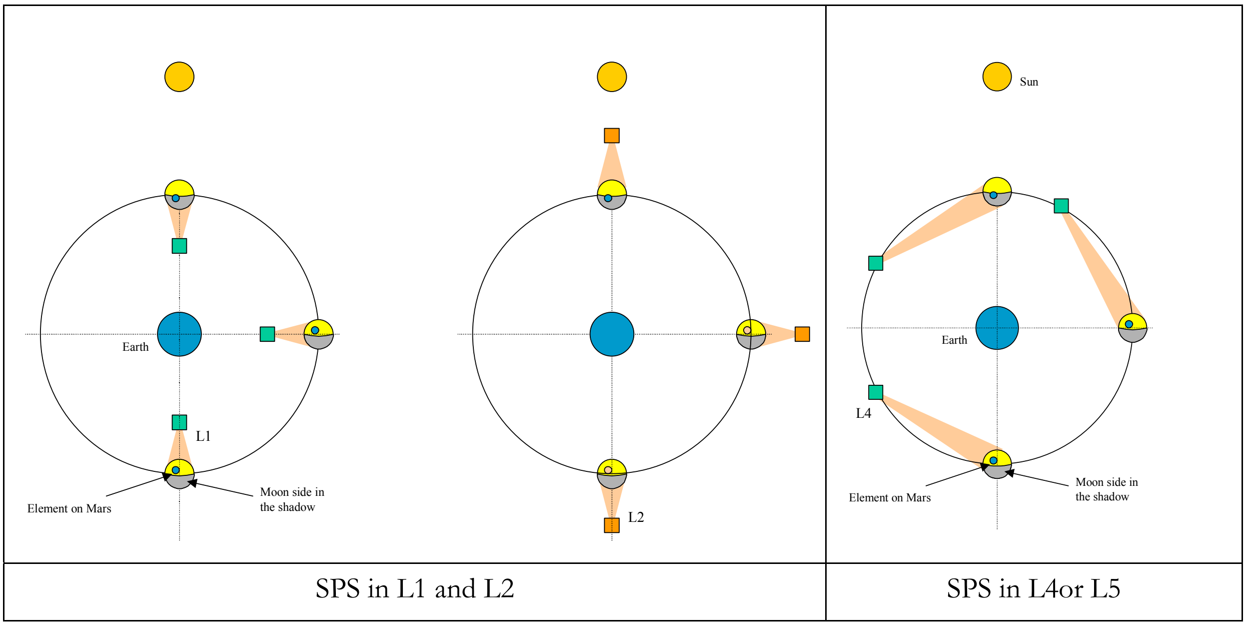


Figure 6.2/3: Possible SPS positioning in Earth-Moon Lagrangian points

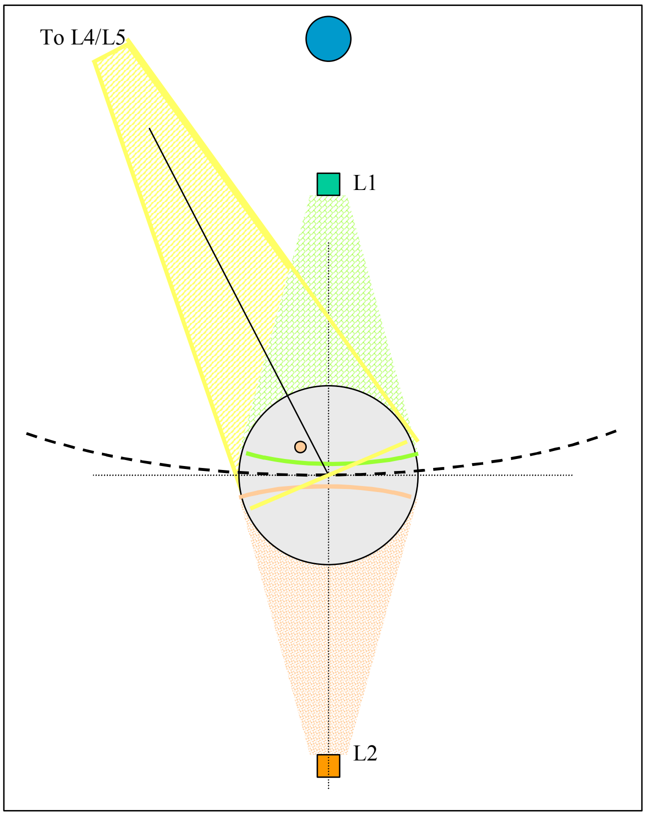


Figure 6.2/4: Illumination zones from the different SPS positioning at Lagrangian points

When positioned in L2, the SPS provides permanently power to an element located on the hidden Moon side. As for L1, the SPS illuminates almost half the Moon (up to 88° latitude, and from –88° to +88° in longitude) during eclipse phase and during sunlight phase.

SPS would be positioned in L1 or L2 depending on where is implemented the element on the Moon. To cover the complete Moon surface, two SPS would be necessary, one at each point, but a small zone including poles and perpendicular to the Earth-to Moon axis will not be covered. Due to the orbit inclination, the eclipse phase for SPS (due to occultation by Earth) will be scarce and short.

L4/L5 points

They are located on the Moon trajectory, at an average distance of about 384400 km. From those points, the SPS would illuminate roughly half the Moon surface permanently. The use of two SPS would cover almost all the Moon surface, but a zone located in the anti Earth side will remain not covered. These points are more stable than L1 or L2. However, they do not offer advantage with respect to L1 and L2 as the SPS will cover only half of the Moon. Besides, the distance is penalizing the sizing of the transmission system.

6.2.3 Earth-Sun Lagrangian points

The SPS could be positioned in L2, as illustrated on Figure 6.2/1. In that case, the SPS has permanent visibility on Sun and Moon. It will be in a Halo orbit around the L2 point to avoid any occultation of the Sun by the Earth. The distance between the SPS and the Earth is about 1.5 Million km, so that the distance between the SPS and the Moon surface will vary roughly between 1.1 and 1.9 Million km. As the point L2 is situated in the opposite side of the Sun with respect to Moon, the SPS illuminates permanently the Moon face in eclipse. Consequently, any element on Moon is illuminated either by the Sun, or by the SPS, which complements the Sun. Therefore, the utilisation of common receiver surface (typically photovoltaic cells) would be the best approach.

The drawback of this position is the large distance, which has a drastic impact on the sizing of the power transmission system. Only laser transmission system is applicable.

6.2.4 SPS positioning synthesis

Three candidate positioning can be considered:

SPS at low altitude (5000 km): *Short distances, but need for 3 SPS for a permanent illumination of the element*

- Low inclination orbit (about 10 to 15°)
- Low distance SPS-target
- Target illumination: 33% of a day. Need of a set of 3 satellites for a permanent illumination

- Coverage: all the Moon surface for latitudes between –75 and +75° (lower availability for surfaces located at higher latitudes)
- Maximum beam steering angle: 15°
- Applicable to RF and laser
- Positioning minimizing impacts on the size of the power transmission system
- Target power storage system to be sized for a few hours

SPS at Earth-Moon L1 or L2: *permanent illumination of a face of the Moon, with medium distances*

- Average distance SPS-target: about 56000 km
- Coverage: Almost half the Moon surface (either side facing Earth – L1, or opposite (hidden) side – L2). A slice of surface including poles and perpendicular to Earth-Moon line is not covered)
- Illumination: permanent illumination of the covered side
- SPS to be positioned in L1 or L2 according to the location of the element on Mars. 2 SPSs needed (one in L1, one in L2) for a quasi complete coverage
- Maximum beam steering angle: 2°
- Applicable to RF and laser
- Target power storage system to be sized for a few hours

SPS at Earth-Sun L2: *full complement of Sun illumination, but very high distances*

- SPS-target distance varying between 1.1 and 1.9 Million km
- SPS in Halo orbit to avoid occultation by Earth
- Coverage: the Moon side not illuminated by the Sun
- Illumination: permanent illumination of the anti-Sun side. Any surface of the Moon is illuminated either by Sun or by SPS.
- Same receiver surface for sun energy and SPS power.
- Applicable to laser
- High distance generates high impacts on power transmission sizing.

6.3 RF BASED SPS SYSTEM

The SPS is located at low altitude (typically 5000 km). It provides up to 50 kW to the infrastructure on the Moon. The other potential locations (L1 or L2) lead to too large distance and would yield to an excessively large SPS for the required power.

6.3.1 RF power transmission system

6.3.1.1 Elements and methodology for the dimensioning of RF WPT systems

We use the same methodology than in section 5.5.1 where we studied RF power transmission to elements on Mars. The frequency is also considered to be 35GHz. The rectenna element definitions of type 1 to 4 will also be used without modification in this section.

6.3.1.2 Dimensioning cases

Two cases will be considered. In the first case, the rectenna is built using type 2 elements (medium power density rectenna). In the second case, type 4 elements will be used (very low power density rectenna).

6.3.1.2.1 Rectenna type 1

When rectenna type 1 is used, an incoming power density of 724W/m2 should be present at the rectenna aperture.

Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 150mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 75%. Each element then delivers a DC output power of 110mW.

Each rectenna type 1 element occupies a surface of 20cm^2 and then, each square meter of rectenna delivers a DC output power of P_{units} = 55 W.

To provide the 50 kW needed it is then necessary to deploy a total surface of:

$$S_{total} = 50\,000/55 = 909 \text{ sq. m.}$$

This corresponds to a circular area of diameter: D_c = 34 m

Table 6.3/1 below summarizes the main estimated characteristics of the rectenna area when rectenna element of type 1 is used.

To produce an RF power density of 724 W/m^2 at a 5000 km distance, the projecting antenna aperture of diameter D_p and transmitting an RF power of P_{tr} must satisfy the following relation, if it is illuminated with a 10dB Gaussian taper:

$$dP = 0{,}709 \; \frac{P_{tr} * D_p^2}{\lambda^2 * r^2}$$

where *dP* is the produced RF power density, λ the wavelength (8.6mm), *r* the distance 5000km).

Assuming a 2MW of projected power, an antenna diameter of 969 m is needed. If we now assume a 4MW projected power, a 685 m diameter antenna is needed.

Table 6.3/2 lists the estimated characteristics of the RF projecting unit with rectenna type 1.

Property	Value	Comment
Rectenna	Rectenna type 1	High power density rectenna
Power density at rectenna	724 W/m²	
Efficiency of antenna	70%	
RF power at rectifying circuit	150mW	
RF to DC conversion efficiency	75%	
DC output power	110mW	
DC output power density	55 W/m2	
Surface of rectenna needed for 100 kW total DC output power	909 m²	
Diameter of corresponding circular area	34 m	

Table 6.3/1: Main estimated characteristics of the rectenna area when rectenna type 1 elements are used.

Property	Value	Comment
Option 1 (Low power large diameter antenna)		
Projected power	2 MW	
Antenna diameter	969 m	
Option 2 (High power lower diameter antenna)		
Projected power	4 MW	
Antenna diameter	685 km	

Table 6.3/2: Main characteristics of the RF projecting unit with rectenna type 1

6.3.1.2.2 Rectenna type 2

When rectenna type 2 is used, an incoming power density of 10.75W/m² should be present at the rectenna aperture.

Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 150mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 75%. Each element then delivers a DC output power of 110mW.

Each rectenna type 1 element occupies a surface of 320cm² and then, each square meter of rectenna delivers a DC output power of P_{units} = 3.438 W.

To provide the 50 kW needed it is then necessary to deploy a total surface of:

$$S_{total} = 50\,000/3.438 = 14545 \text{ sq. m.}$$

This corresponds to a circular area of diameter: D_c = 136 m

Table 6.3/3 below summarizes the main estimated characteristics of the rectenna area when rectenna element of type 2 is used.

Property	Value	Comment
Rectenna	Rectenna type 2	*Medium power density rectenna*
Power density at rectenna	10.75 W/m²	
Efficiency of antenna	75%	
RF power at rectifying circuit	150mW	
RF to DC conversion efficiency	75%	
DC output power	110mW	
DC output power density	3.438 W/m2	
Surface of rectenna needed for 100 kW total DC output power	14545 m²	
Diameter of corresponding circular area	136 m	

Table 6.3/3: Main estimated characteristics of the rectenna area when rectenna type 2 elements are used.

To produce an RF power density of 10.75 W/m² at a 5000 km distance, the projecting antenna aperture of diameter D_p and transmitting an RF power of P_{tr} must satisfy the following relation, if it is illuminated with a 10dB Gaussian taper:

$$dP = 0{,}709 \; \frac{P_{tr} * D_p^2}{\lambda^2 * r^2}$$

where *dP* is the produced RF power density, λ the wavelength (8.6 mm), *r* the distance (5000km).

Assuming a 2MW projected power, an antenna diameter of 118 m is needed.

Property	Value	Comment
Projected power	2 MW	
Antenna diameter	118 m	

Table 6.3/4: Main characteristics of the RF projecting unit when rectenna type 2 is used.

6.3.1.2.3 Rectenna type 3

When rectenna type 3 is used, an incoming power density of 3.58W/m2 should be present at the rectenna aperture.

Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 50mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 75%. Each element then delivers a DC output power of 37.5mW.

Each rectenna type 3 element occupies a surface of 320cm^2 and then, each square meter of rectenna delivers a DC output power of P_{units} = 1.17 W.

To provide the 50 kW needed it is then necessary to deploy a total surface of:

S_{total} = 50 000/1.17 = 42735 sq. m.

This corresponds to a circular area of diameter: D_c = 233 m

Table 6.3/5 below summarizes the main estimated characteristics of the rectenna area when rectenna element of type 3 is used.

Property	Value	Comment
Rectenna	Rectenna type 3	*Low power density rectenna*
Power density at rectenna	3.58 W/m²	
Efficiency of antenna	75%	
RF power at rectifying circuit	50mW	
RF to DC conversion efficiency	75%	
DC output power	37.5mW	
DC output power density	1.17 W/m2	
Surface of rectenna needed for 100 kW total DC output power	42735 m²	
Diameter of corresponding circular area	223 m	

Table 6.3/5: Main estimated characteristics of the rectenna area when rectenna type 3 elements are used.

To produce an RF power density of 3.58 W/m² at a 5000 km distance, following the previously quoted relation, an antenna diameter of 96.4 m is needed together with a 1 MW projected power

Property	Value	Comment
Projected power	1 MW	
Antenna diameter	96.4 m	

Table 6.3/6: Main characteristics of the RF projecting unit when rectenna type 3 is used.

6.3.1.2.4 Rectenna type 4

When rectenna type 4 is used, an incoming power density of 1.51W/m2 should be present at the rectenna aperture.

Considering the antenna effective area A_{eff} and its efficiency η_{ant}, this optimum power density allows the rectifying circuit to be driven by the optimum input power of 50mW and to convert this RF power into a DC electric power with the maximum power conversion efficiency of 65%. Each element then delivers a DC output power of 13.65 mW.

Each rectenna type 4 element occupies a surface of 320cm² and then, each square meter of rectenna delivers a DC output power of P_{units} = 0.42 W.

To provide the 50 kW needed it is then necessary to deploy a total surface of:

$$S_{total} = 50\,000/1.17 = 117216 \text{ m}^2$$

This corresponds to a circular area of diameter: $D_c = 386$ m

Table 6.3/7 below summarizes the main estimated characteristics of the rectenna area when rectenna element of type 4 is used.

Property	Value	Comment
Rectenna	Rectenna type 4	*Very Low power density rectenna*
Power density at rectenna	1.51 W/m²	
Efficiency of antenna	75%	
RF power at rectifying circuit	21mW	
RF to DC conversion efficiency	65%	
DC output power	13.65mW	
DC output power density	0.42 W/m2	
Surface of rectenna needed for 100 kW total DC output power	117216 m²	
Diameter of corresponding circular area	386 m	

Table 6.3/7: Main estimated characteristics of the rectenna area when rectenna type 4 elements are used.

To produce an RF power density of 1.51 W/m² at a 5000 km distance, following the previously quoted relation, an antenna diameter of 88.5 m is needed together with a 500 kW projected power. Projecting a higher power of 2MW would only necessitate an antenna radius of 44.25 m to get the same power density at the rectenna.

Property	Value	Comment
Option 1: Low Power Large Antenna		
Projected power	500 kW	
Antenna diameter	88.5 m	
Option 2: High Power Small Antenna		
Projected power	2MW	
Antenna diameter	44.25 m	

Table 6.3/8: Main characteristics of the RF projecting unit when rectenna type 4 is used.

6.3.1.3 On-board transmitting system description

This part describes the dimensioning of the projecting unit on board the SPS in low altitude orbit around the Moon. It also gives some details on the technology and components that could be used for producing the RF power beam and this subsequently allows for the estimation of the mass of the system.

Two dimensioning cases have been considered: using rectenna of type 2 on the Moon for the first case, of type 4 for the second case. For each case, we already evaluated the size and level of projected RF power; in this chapter, we complete this analysis by providing details on how to provide the required power and what kind of antenna element and technology could be used to achieve the desired specifications.

6.3.1.3.1 Dimensioning when rectenna type 2 is used

We have seen in section 6.3.1.2.2 that the required antenna diameter is 118 meters and the projected RF power level should be 2 MW. The maximum power density at the projecting aperture is 467 W/m2, with an average power density on the antenna of 183 W/m². Such a power level requires the use of the highest possible efficiency generators to avoid thermal management issues. This power will be provided by miniature TWTs, each delivering 100 W maximum output power, with an efficiency of 50%. They will be supplied by 85% efficiency electronic power conditioners capable of providing the necessary high voltages from a single DC voltage, thus drawing a maximum of 235W from the Power System. Because of the high power generators used, they must feed large antenna elements, with high gain. These high gain elements have estimated efficiencies of up to 60% at 35GHz. Then, an average power of 305W/m2 will be required at the input of each sq. meter antenna. 3 TWTs will be used for each sq. meter of projecting antenna leading to a total of 32807 TWTs with a total power of 7.71 MW. Each TWT feeds an antenna element of 57cm×57cm, which uses the HCPA technology proposed by Van der Wilt and Strijbos. It could be estimated from their work that an 80mm×43mm antenna element designed at 35GHz would provide approximately a 25dB gain with an estimated 75% efficiency. With these estimations, we can extrapolate the antenna gain of a 57cm×57cm antenna realized with this technology to be 44dB.

The projecting system would be consisting of 32807 antenna elements of 57cm×57cm×4cm, each fed by a 100W TWT and supplied with power by an 85% EPC.

The mass estimation is the following:

Mass of antenna:	32807×3kg	*3kg is the estimated mass of a 57cm×57cm antenna element*
Mass of TWTs:	32807×0.8kg	
Mass of EPCs:	32807×0.8kg	
Mass of phase control:	32807×0.2kg	
Total mass:	**32807×4.8kg = 157.5 tons**	

This estimation is made without calculation of the optimum size of the subarrays. Indeed, the subarray size is a crucial parameter in the case of electronic beam steering as it is the case here. It controls the position of the grating lobes and subsequently conditions the level of power lost in the grating lobes when the beam is steered from boresight to the maximum steering angle. However, many other aspects have been neglected at this point of the study and the above estimation provides a value that is meaningful, without having to run a detailed system analysis. This analysis would be necessary however, as the next step of in system design if this application appear potentially interesting.

Property	Value	Comments
Dimensioning case using rectenna type 2		
RF power projected	2MW	
Antenna diameter	118	
Average projected power density	183 W/m2	
Maximum projected power density	467 W/m2	
Type of antenna modules	HCPA antenna	
Antenna efficiency	60%	
Size of modules	57 cm×57 cm ×4cm	
Mass of antenna module	3kg	*Including backing plate*
Type RF generator	TWT	*M1280 from L-3 Communications or similar.*
RF generator efficiency (including EPC)	42.5%	
Mass of RF generator	1.6kg	
Phasing mechanism	Retrodirective phase conjugation	
Phasing circuit mass	0.2kg	
Number of modules	32807	
Total mass of projecting unit	157.5 tons	
Total power required at input	7.71 MW	

Table 6.3/9: Main features of on-board transmission system in case of rectenna type 2

6.3.1.3.2 Dimensioning when rectenna type 4 is used

We have seen in section 6.3.1.2.4 that the required projecting antenna diameter is 88.5 m, taking the low power option, with a projected RF power of 500 kW. The maximum power density at the projecting aperture is 207 W/m², with an average power density over the aperture of 81.28 W/m². This power could be provided by 50% efficiency TWTs delivering 80W, supplied with power by an 85% electronic power conditioner. The power drawn from the Power System is 188 W, for each device fed. High gain antenna elements will be used, with a maximum efficiency of 60%. A power of 136 W will be required at the input of each sq. meter of projecting antenna. A total RF power of 6152×136 = 837kW will be required at the input of the antenna terminals. Consequently, 10458 TWTs will be necessary. Each TWT feeds an antenna of dimension 75 cm×75 cm. The total power drawn from the Power System is 10458×188=1.97 MW.

The mass estimation is the following:

Mass of antenna elements:	10458×4.5kg
Mass of TWT:	10458×0.8kg
Mass of EPC:	10458×0.8kg
Mass of phase control circuit:	10458×0.2kg
Total mass of projecting equipment:	**10458×6.3kg=65.9 tons**

Property	**Value**	**Comments**
Dimensioning case using rectenna type 4		
RF power projected	500 kW	
Antenna diameter	88.5	
Average projected power density	81.3W/m2	
Maximum projected power density	207W/m2	
Type of antenna modules	HCPA antenna	
Antenna efficiency	60%	
Size of modules	75 cm×75 cm×4 cm	
Mass of antenna module	4.5kg	*Including backing plate*
Type RF generator	TWT	*TH4046 from Thales EDD*
RF generator efficiency (including EPC)	42.5%	
Mass of RF generator	1.6kg	
Phasing mechanism	Retrodirective phase conjugation	
Phasing circuit mass	0.2kg	
Number of modules	10458	
Total mass of projecting unit	65.9 tons	
Total power required at input	1.97 MW	

Table 6.3/10: Main features of on-board transmission system in case of rectenna type 4

6.3.2 Power generation system definition

This mission is similar to the Mars base in term of power requirements. 10458 tubes have to be supplied for a total power consumption of 2 MW.

The same architecture is therefore proposed

The solar array area is equal to 5000 m².

6.3.3 Preliminary definition of SPS

This preliminary definition of the SPS system concept has been based on current or reasonably achievable technology state-of-the-Art, and without any iteration for optimisation. The SPS definition is driven by its Moon pointed attitude, its large antenna and solar arrays.

The antenna, 88.5m diameter, is a flat one of HCPA type, with the TWT/EPC and ancillary electronics integrated on the backside, and the radiator also mounted on the back. Baffles are necessary to protect the radiators against Sun. Heat pipes may be used to transfer the dissipated heat from the antenna and TWT/EPC to the radiators. A degree of freedom (mechanical or electronic) is necessary to keep the beam pointed to the target during the relative motion of the SPS. A 30° beam steering capability (-15°, +15°) is needed.

The antenna is attached to a truss structure that houses control command and management equipment, part of power system such as batteries to provide power to SPS subsystems during eclipse, communications system equipment. Electrical propulsion is also implemented on the truss: thrusters on the backside, and fuel tanks in the truss.

The solar arrays are mounted on trusses along North/South axis, perpendicular to the central core to favour the sun visibility and avoid, in some configurations, the occultation of solar surfaces by other solar surfaces. Each side is composed of ten panels of 250 m² (5m x 50m). The panels are removed from the central core to avoid occultation by the antenna.

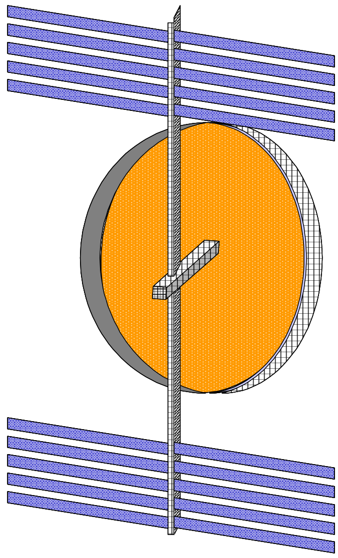


Figure 6.3/1: Illustration of SPS

The SPS mass at launch is in order of 180 t.

The main performances of the SPS system are summarized in Figure 6.3/2. The end-to-end efficiency of the SPS system is about 2.5%.

Being in low orbit around Moon, the SPS will stay only 33% of its time in visibility of the target. Therefore, a set of 3 SPS is necessary to ensure a permanent power delivery.

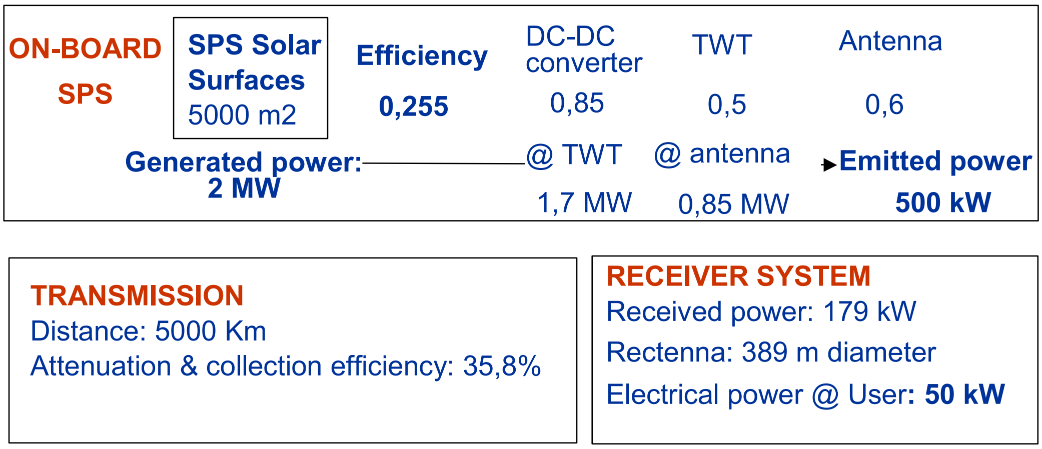


Figure 6.3/2: Main performances of the RF based SPS system for power delivery to element on Moon

6.4 LASER BASED SPS SYSTEM

The SPS is located at the Lagrangian point L1 or L2 and delivers up to 20 kW to the infrastructure

6.4.1 Laser power transmission system

6.4.1.1 On-board transmitting system description

The mission of the SPS system is to provide 20kW to a Moon station during solar eclipse.

The size of the solar panel can be relatively large (several tens of meters).

We propose to reuse the same laser concept as described in Chapter 2.3. The main difference is related to the output power and the size of the emitting telescope.

The SPS system is composed of 4 independent 10 kW laser systems that can be pointed independently. The 4 systems are embarked on the same satellite located in Earth-Moon Lagrangian point L1. The pointing system is a closed loop controlled system.

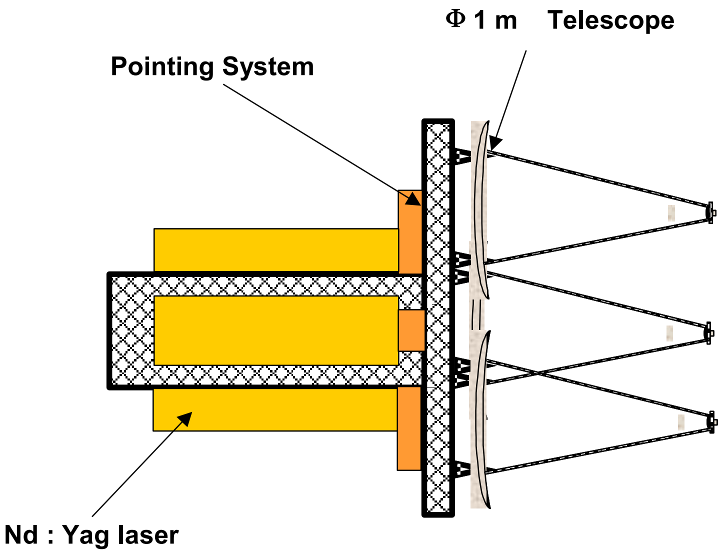


Figure 6.4/1 : SPS laser Concept

6.4.1.2 Mass and power budgets of the laser system.

The mass and the power budget of the 10 kW laser power system are given in the following tables.

Sytem laser mass	Mass in Kg
Telescope	20
Pointing system	5
Opto-mechanical assembly of laser	200
Mechanical structure	200
Electronics	200
Total	**625**

Table 6.4/1: Mass budget (for 1 laser)

Power flow chart laser		
Emitted power	10000	W
Number of lasers	4	
SHG generation efficiency	70	%
Optical pump efficiency	50	%
Laser diode plug in efficiency	50	%
Power generation system		
DC/DC converter efficiency	85	%
Power for spacecraft	5000	W
Power required at SA output	273908	W
solar cell efficiency	30	%
solar power density	1300	W/m^2
solar array size	702	m^2

Table 6.4/2: Power budget (for 4 lasers)

6.4.1.3 Performances and SPS parameters

The performances of the SPS system are depicted in the following table:

Laser system parameters		
Laser power	10000	watt
SPS laser system	4	
Operating Wavelength	532	nm
Telescope Diameter	1	m
operating distance	60000	km
Diameter of the Spot on target aera	38,9	m
50% Loss pointing Accuracy	833	nrad
Surface of the PV ground surface	10000	m²
Size of PV pannel (square)	100	m
Efficiency of the PV Cell	50	%
Illumination at target level	33,58	W/m²
Generated electrical power	20000,0	W

Table 6.4/3: **SPS laser system parameters**

6.4.2 Power generation system definition

The SPS is made of four lasers consuming 57 kW each at the diode input level (10 kW emitted power).

Assuming a DC/DC converter efficiency of 85%, this leads to a power requirement of 270 kW on the power bus.

Again, it is proposed to design one PSS dedicated to one laser. This leads to the same definition as for the Telecommunication mission.

6.4.3 Preliminary definition of SPS

This preliminary definition of the SPS system concept has been based on current or reasonably achievable technology state-of-the-Art, and without any iteration for optimisation

The laser based SPS is oriented towards the Moon surface and the target.

It is composed of a payload part, in front of the SPS with the telescope oriented towards the Moon, and of a resource part. The payload part includes the four lasers and deployable radiators to reject the dissipated heat. There is one radiator per laser module, with a surface of 120 m², rejecting up to 47 kW. Active fluid loop (or alternatively heat pipe) may be used to transfer the heat from the laser components to the radiators.

The resource part includes avionics, power subsystem, communications subsystem, electrical propulsion; solar arrays (two panels of 8 m x 45 m) are mounted along North/South axis. Two radiators are deployed perpendicular to the solar arrays, for the primary power distribution and other electronics, and electrical propulsion.

The main performances of the laser based SPS system are recalled in Figure 6.4/3. The end to end system efficiency is about 7,15%. The SPS mass is in order of 25t. The receiver system, on the Moon, made of photovoltaic cells, is typically 100m x 100m.

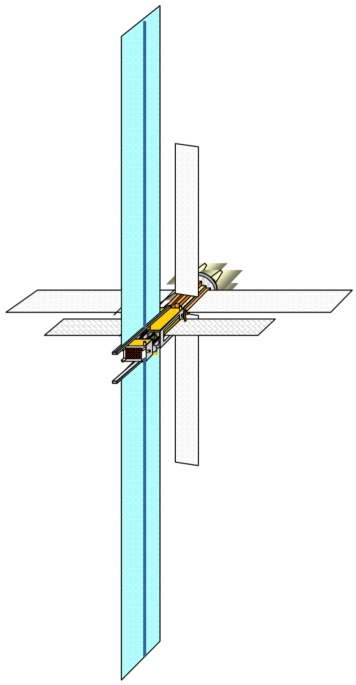


Figure 6.4/2: Illustration of laser based SPS

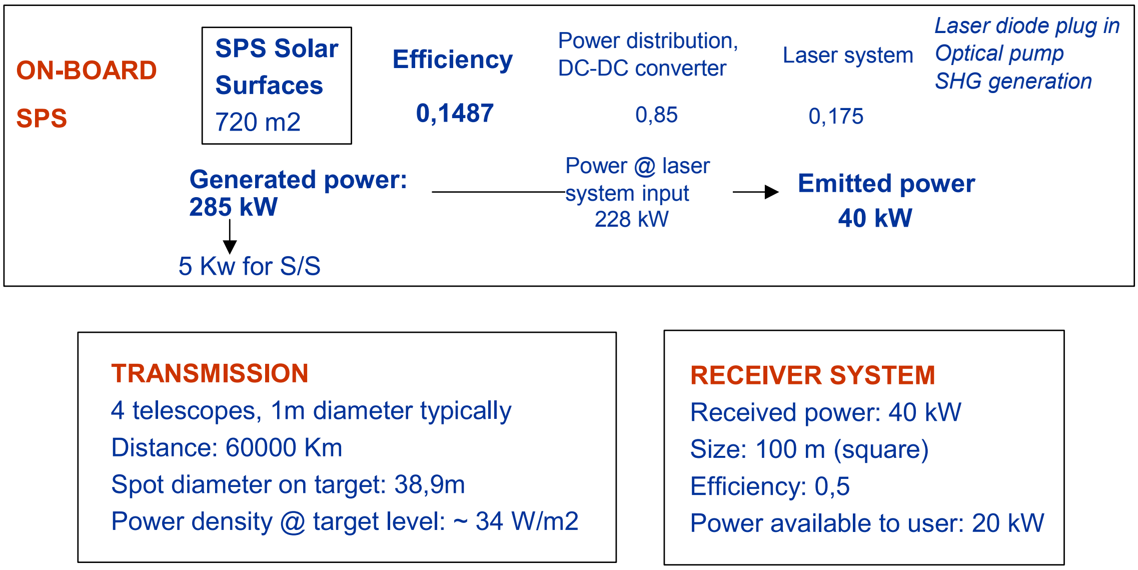


Figure 6.4/3: Main performances of the laser based SPS system for power delivery to element on Moon

6.5 INTEREST OF SPS

The utilisation of a SPS system to provide power to infrastructure on Moon surface could solve the energy problem raised by the long eclipse. Indeed, according to its location in latitude on the Moon, the infrastructure may have to cope with up to 14 days of eclipse (followed by 14 days of sunlight). Alternative solutions consist in using either energy storage systems (like fuel cells) leading to store at least 7 MWh for a 20 kW power need (if 100% efficiency), or nuclear power systems, which both are not renewable energy.

The proposed laser based SPS concept is located at L1 or L2 Lagrangian point (58000 Km). It allows a permanent illumination of half the Moon surface, and is subject to scarce and short eclipses. Preliminary evaluation leads to a SPS launch mass in order of 25 t, injected in LEO (single launch) and transferred with electrical propulsion, and a receiver surface of 100m x100m of photovoltaic cells. An increase of the telescope size could allow to reduce the receiver surface, but such an iteration has not been done.

7 SYNTHESIS

Applicability

A preliminary evaluation of the SPS system concept has been done. It has been based on current or reasonably achievable technology state-of-the-Art, and without any iteration for optimisation. This "conservative" approach has been followed in order to define a first basis for future work, as there was only few background on these missions.

The applicability of the SPS system to different types of missions has been assessed.

The SPS power delivery to geostationary platforms assumes that a concept of service for all satellites is considered. However, the potential gain for the platforms does not appear cost effective (unless the cost of the service be sufficiently low). The utilisation of SPS to deliver power to interplanetary vehicles has not been retained: the distance is too high, leading to complex system, and it is difficult to implement SPS beyond the Mars orbit.

On the opposite, the SPS system appears as an attractive solution for power delivery to elements on Mars or Moon surfaces. For small element on Mars, like rover, the laser based SPS allows to increase the autonomy; key issue is the dust storm, and laser performance through dust has to be analysed. For large Mars base, RF based SPS is preferred, due to dust. It is an alternative to the use of nuclear power system. For infrastructure on Moon, laser based SPS is preferred when the required power remains in order of tens of kW.

The SPS concept could also be an alternative solution to technological issues like in the case of the Darwin mission.

Technology and key issues

A preliminary evaluation of the SPS system concepts has been done, based on current or reasonably achievable technology. This leads to overall system efficiencies of a few percent, and to important SPS masses. This concept evaluation assumes an electrical propulsion system ensuring the SPS transfer from LEO to its final positioning. In the case of a laser transmission system, the SPS is compatible with a single launch (with a possibly heavy launcher) in LEO.

This preliminary evaluation is a basis to identify the key issues driving the performances.

The performance of the SPS system is driven mainly by:

- The RF system capability at 35 GHz, for the RF based SPS. In particular, the key features of this capability are:
  - o The optimisation of the on-board and ground systems: receiver surface, antenna diameter, emitting power
  - o The antenna size: for slot array type, key issues concern the large antenna to be assembled in orbit (with integrated signal generators), the low efficiency and the thermal stability of sub arrays; for parabolic type, key issues are related to maximum size, surface accuracy control, primary reflector implementation. Nevertheless, the low antenna efficiency taken into account reflects other types of losses (error on phase, amplitude, …)
  - o The DC to RF converter: there is very high number of equipment, and a low efficiency that impact mass, power to be provided by solar surfaces, and heat to be dissipated
  - o Rectenna on-ground: type of rectenna not optimised, leading to large surface and potential problem of deployment
- The laser capability. The key issues are the on-board efficiency, impacting SPS mass and needed power, and beam pointing accuracy.

- the laser technology, antenna and signal generators technology, the pointing accuracy and the receiver system efficiency.

The main drivers of the mass and size of the SPS are related to the SPS platform:

- large solar arrays surface, impacting deployment and pointing systems
- length of truss, to implement solar arrays and to prevent them from occultation
- deployable thermal radiators, requiring large surface, and system for transferring dissipated heat from equipment to radiators (active fluid loops, other)
- control of large and flexible structures (very low frequency)
- pointing stability
- control of disturbances along the orbit
- plasma propulsion

In addition, some aspects like beam steering have not been analysed, but might appear also a key issue in the SPS system sizing.

Axes of improvement

The interest of this preliminary evaluation is also to identify the areas of optimisation.

Several axes of improvement can be considered. They are:

- The **improvement of the overall system end-to-end efficiency**, which is rather low, especially in comparison to the overall efficiencies quoted in the Space-to-Earth studies. For instance, in the case of the RF based SPS for Mars base, an improvement of the efficiency from 1,6% to 5% would allow to reduce the SPS mass by at least 40%, all others identical.
- The utilisation of **new technologies for the power transmission systems**.
    - For RF system, the selection of 35 GHz raises currently technical limitations on the antenna and equipment as quoted here above. Possible axes of technology improvement are: to reduce the slot array antenna density, to improve the density and the deployment technology for classical parabolic antennae, to improve RF signal generator in terms of efficiency and mass by an adequate adaptation the 35 GHz, to optimise the rectenna (in terms of input power, efficiency) for the application
    - For laser system, current on-board efficiency is in the range of 12 to 21% and the overall efficiency in the range 0,6 to 7%. Axes to improve these performances are: to develop or improve new technologies like direct solar pumped laser, or new types of fiber laser, to improve SPS pointing and stability, to improve efficiency of receiver system.

- The utilisation of **advanced or new technologies in the mass driver area**. That is the case for instance for:
  - the power generation: it leads to large surfaces impacting deployment and pointing systems and leading to large structures. Possible axes consist in reducing the surface of solar surfaces (increase of cell performances, etc) and the density of the complete surface (film, structure, etc), or in analysing performances of possible alternatives, like Dish Stirling. For instance, the technology used in the preliminary definition (solar array made of multiple junctions cells mounted on a thin film substrate with small concentrators, with a target of 30% efficiency and 1.5 kg/m^2 density is not available to day in Europe. Objective at medium term is to have better performances (>30%, < 1.5 kg/m^2). The power system generates also high mass of cabling, in case of large platform and high power; to reduce this mass, possible approaches are to analyse new bus material (Aluminium bus bar for instance) or new technology (like supra conductor), and to use very high voltage from primary source to end user distribution (typically 1000V, with a feasibility of design compatible with such voltage to be demonstrated, and the space qualification of power regulation components for this voltage).
  - the heat transfer and radiator surfaces: axes of improvement are to reduce the surface of radiators (higher radiator performances) and its density, and to improve the heat transfer from the high consuming equipment to the radiators
  - structure and mechanism: technology improvement towards lighter system for large solar surface deployment and lighter deployable structure and antenna.
- The **optimisation of the complete system, SPS and receiver**. Thus, in the case of receiver surfaces installed on the planet surface, the drivers for the system optimisation have to be defined, in terms of mass in orbit, mass on the surface, mass on the surface including the descent and landing part, etc. This optimisation can be done through the selection of receiver element, the balance between SPS emitting parameters and receiver size, etc.
- The analysis of SPS concept alternative to the large platforms (case of RF based SPS for Mars base): **the utilisation of a network of satellites**, offering a larger virtual emitting diameter, could be interesting as solving a number of technical issues. It allows to have individual satellite compatible with a single launch and a flexibility in the implementation and maintenance scenario. For instance, they could be distributed following a known distribution function, thus forming what is known as an aperiodic array. But it presents other technical issues, which should be identified.

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