8 Space Transportation

Space transportation is usually the single most expensive service used by a space program. In view that about 25 to 40% of the cost is spent for as little as a 10 minute service, for a project that lasts many years, it is clear that special attention has to be given to this topic.

In each step of a Space Solar Power Program development plan, different types of space transportation are needed. The first demonstration missions are envisaged as being near term, small in size —a few kW—and can utilize current launch systems. The first operational missions are larger in size —100 kW—and may be 10-15 years away. The missions would profit by using systems under development or updated versions of today's launchers. The future commercial missions are yet larger, 1-100 MW, and 1-10 GW. These missions are dependent on more future, reliable, inexpensive and dedicated transportation.

In the first and second space transportation section (8.1, 8.2) the basic data is given for the launchers and upper stages (OTV) that are operational or under development. This is primarily used as basic data upon choosing a transport for the different first demonstration missions of the stepped evolution of the Space Solar Power Program.

The third section (8.3) covers the launch systems under development today and informs the reader of what is available for the first operational space based solar power satellites in the first years of the 21 th century.

The fourth section (8.4) summarizes the space transportation envisaged by previous "SPS" studies. Especially all the future transportation concept of the NASA/DOE studies in the late 1970's are reviewed.

The four following chapters look into the next generation of available space transportation systems. 8.5 discusses perceived SPS customer requirements. Specific suggestions are made for preferred new transportation systems and their possible changed characteristics. Not only cost but also reliability, availability and resiliency are addressed. Section 8.6 describes different technologies that could improve all types of space transportation systems. In specific improved propellants, new materials and nuclear and laser power is discussed. section 8.7 goes into more detail about the technical options and possible systems used for future lunar transportation.

The final section 8.8 discusses the development schedule of the space transportation for the stepped Space Solar Power Program development program.

8.1 Operational Space Transportation Systems

This section will discuss the world's current orbital launch and piggy-back capability.

8.1.1 Review and Analysis of Earth To Orbit Launchers

This section will examine the current 1992 launcher market. Any space power demonstrations will be launched on one of the vehicles discussed below. The discussion starts with a review of operational launchers, This is followed by a market analysis. The section finishes with a discussion of costs.

Review of Operational Launchers

Modern operational launchers can be organized into categories according to payload lift capabilities and orbit that payload is placed into. For the purposes of this review, we will use the following classification system:

System Classification	Payload to LEO	Payload to GTO
Small	Below 1000 kg	N/A
Medium	1001 kg to 10,000 kg	500 kg to 5000 kg
Large	10 001 kg to 20,000 kg	5001 kg to 8000 kg
Heavy Lift	Above 20,000 kg	Above 8000 kg

A detail discussion of small launchers is presented in the next section.

Medium launchers are the workhorses in the world's launch stable. Medium launchers could carry a modest Space Solar Power Program demonstration payload in the near term at a predictable price. Many nations or multi-national organizations have medium launchers. The Europeans have the Ariane 4, the US has the Atlas, Delta II, and Titan II, and the Japanese have the H-1. The H-1 has just been retired from the Japanese stable. The Martin Marietta Titan II is headed for use as a low-cost military launcher and should have little or no effect on the commercial launch industry. The Chinese have the Long March CZ-4 and CZ-3A and the Russians have the Zenit 2/SL-16, Tsyklon/SL-14. Arianespace, launching the Ariane 4, provides transport to orbit for over half of the "open market payloads" Open market payloads are available for competition of launch providers. Another market type is a closed market; an example is the USA military payloads. Only USA launch providers may launch these payloads. This policy is under question and review however, a change allowing non US launch providers to launch US military payloads is highly unlikely given the current commercial position of major US launch providers. The recent Russian and Chinese entry into the commercial launch markets introduce some market uncertainty. The Chinese's Long March series provides commercial launches but only hold a small market share. Their stable of modern medium launchers can provide customers with a large range of payload capabilities and orbits; also, the Chinese offer attractive commercial arrangements with extremely competitive pricing. The Russians are in a similar position to the Chinese. Currently, their vehicles are available at extremely attractive prices. However in both cases, allowing open market payloads to be launched on either Russian or Chinese vehicles would financially affect US and European launch providers. Recently, the US gave approval for an INMARSAT that contains US technology to be launched on a Russian vehicle. A Long March 2 CZ-2C and Long March 3 CZ-3 have launched open market payloads and agreements exist for future launches. One analysis for this situation is the number of launches allowed (approximately nine) will not have a great financial impact on Arianespace, General Dynamics, McDonnell Douglas or Martin Marietta.

The evolution of the medium sized commercial launch vehicle market should be interesting for Space Solar Power Program to follow. An internationally sponsored Space Solar Power Program demonstration payload could make a good argument for use of a lower cost launcher. If the country providing the launch were also collaborating in the demonstration, then governments might allow the launch of their technology on these vehicles. The Chinese and Russians currently offer launchers with excellent performance at a below current market launch price. The main question is what will the ultimate World response be to the Chinese and Russian ability to undercut current market prices. Many launch providers are somewhat secure in their closed markets and these closed market launches help keep the production lines open. Russian and Chinese launch prices cannot be expected to remain low indefinitely. Their costs most likely will increase for a variety of reasons: 1)increased worker or material expense, 2)political changes affecting the commercial environment, or 3)the tried and true method of charging what the market will bear. Politics have a large effect in the market place right now; but, the future has a need for medium launchers: what the price will be and who will supply the service remains an open question.

Few large launchers are available in the world today. Although there are many vehicles in development, the need for this class of vehicles has not yet returned to the levels of the late 1960s and early 1970s. The only the US launcher in this class is Titan III. The only other operational large launch vehicles are in the Russian stable: the Proton and the Zenit 3. The Titan III has few orders. Martian Marietta is not aggressively marketing the Titan and is concentrating on the military launch vehicle Titan IV. The Russian vehicles are available but have the same problems as outlined in the above paragraph. The Russians are aggressively marketing the Proton and have signed an exclusive contract with US marketing agents. Glavkosmos and a group of Australians have proposed launching the Zenit 3 from Cape York in Australia. A Proton or Zenit launched from Cape York would offer excellent performance at a very competitive price.

Only two heavy lift vehicles exist today: Shuttle and Energia. The US Space Transportation System is currently making six to eight flights per year. The time to be manifested on a shuttle flight can be at a minimum three years to a maximum that has yet to be seen. Given current US economic conditions, a fifth shuttle, requested by NASA, is not likely. The shuttle rarely carries its full payload capacity. The Russian Energia also has limitations on its operational use. Only three Energia are known to exist; production rates are only one to one and a half vehicles per year. The Buran and the Energia programs have been placed on hold indefinitely with no planned flights. In general, the capability exists to launch payloads over 20,000 kg but payloads that have this much mass are rare. Typical commercial payloads are in the 4000 kg to 5000 kg range to GTO. Large mass payloads are not yet flying again in great numbers.

Analysis of Operational Launchers

The first satellite was launched 35 years ago. Then only two national launch capabilities existed. Today, many countries and companies have orbit launch capability. These organizations range from relatively small commercial corporations, as in the case of Orbital Sciences Corp, to the multinational European Space Agency. The world's launcher systems come in a variety of lift capabilities, prices, and launch terms. Getting into orbit is not a question of technology; rather, it is usually a question of cost, launch availability, and lastly, but many times the critical factor, political permission. The world's primary space-faring nations are: Russia and to a limited extent a few other former Soviet republics, the European Community in the form of the European Space Agency, the United States, Japan, and finally China. Other countries that have small space programs are Brazil, Canada, Israel, and India. In the past, national governments have been the primary providers and users of launch services. Today, providers may be a purely commercial company, a quasi-commercial company owned either entirely or partially by a single government or a group of governments, or a completely governmental organization. Because of the evolution of launch vehicles, a purely commercial environment with no governmental involvement (except for launch licenses) is highly unlikely in the foreseeable future.

The traditional launch providers are concerned with losing market share with the entry of the Chinese and Russia into the commercial market. They comment that the Chinese and Russian vehicles are heavily subsided by their governments. These companies most likely will use all their influence to 1) keep these new providers out of the open payload markets, 2) delay their entry into the market place as long as possible, or 3) limit the damage of their entry by allowing them as few launches as possible. They have taken the last option for dealing with the new competition. The Chinese and Russians have gained entry into the market, however small. The question for the future is will the newcomers market share grow or will the established launch providers be able to hold them off with arguments of unfair government subsidies and adverse technology transfer. If the western launch providers lose their influence or the political winds change, then these new providers could gain more market share in the open launch market.

Procuring a launch takes quite a bit of time and energy. There are numerous issues to be dealt with. The technical issues are fairly straight forward: find a vehicle with adequate payload capability that meets the orbital requirements, plan the integration of the payload with the rocket, etc. But, the obstacles have just begun. If you are launching on the Ariane, the back log is over 30 satellites. The shuttle takes over three years, at a minimum, to fly most payloads. Atlas and Delta launchers aren't quicker. The Chinese, with their robust stable of launchers, appear to be able to provide launches quicker but have a problem with launching US technology. The United States has allowed the Chinese to launch satellites using US technology. It is unknown whether this trend will continue. Shopping for a smaller payload launcher might offer a slight advantage in speed to launching. The small payload launchers are dominated by smaller companies. These smaller companies are eager to meet the customers' needs. Orbital Sciences' Pegasus and Taurus and International Microspace's Orbital Express are examples of vehicles that are operational or in development. Many small entrepreneurial companies are trying to enter this market. Their success at finding paying customers has been marginal; however there are some bright spots. In August 1992, International Microspace won a contract for one launch and options for nine more from the US Department of Defense

Current launchers have a variety of problems that would prevent them from being used for any solar power mission other than a small scale demonstration system. These characteristics, as outlined by R.J. Hannigan in his paper "SPS Transportation Requirements: Which System? (Space Power, Volume 10), are:

- high yearly operating costs from $1 - 5 Billion, (fixed and recurring)
- high operating costs from $60 - 100 million per flight
- relatively low reliability from around 90 - 98% (Aircraft system reliability is at least 99.9999%)
- no abort capability, except shuttle in some cases
- low flight rates and launch opportunities of 5 to 10 per year
- long lead times from purchase to launch of 2 to 3 years
- frequent delays measured in weeks, months or even years
- no system resiliency, long standdown times after failures (months to years)
- few servicing or payload recovery opportunities (STS only, 2-3 years wait)

- limited launch sites

The requirements of launch systems needed to open space for commercial development and the characteristics of current launch capabilities are almost in opposition to each other.

Cost Analysis and Pricing of Operational Launchers

The cost of launching payloads is high. To date, no launcher has ever repaid it own development costs. With the exception of a few small privately financed vehicles, all the development costs of most world's launchers have been paid for by a government. This observation is quite in line with the notion that space launchers are still, in large part, the domain of governments rather than corporations and markets. The final destination for revenue derived from launching payloads or the payer for the launch service is an indication of the environment of the world's launch industry. The world has not yet seen a privately developed launch vehicle carrying for profit payloads. The character of this government dominated launch industry is high cost, low rate of innovation, and lengthy, complicated launching processes.

Table 8.1 on the following page is a summary of launch costs and performance.

Table 8.1 Operational Launchers

Launcher Name	LEO Payload (kg)	GTO Payload (kg)	Launch mass (kg)	Launch Thrust (kN)	Launcher costs Unit (M$)	Cost/kg (k$)
Ariane 40	4900	2000	240000	2710	60.0	12.24
Ariane 42P	6100	2700	339000	4150	62.0	10.16
Ariane 42L	7400	3200	400000	4040	85.0	11.49
Ariane 44P	6900	3000	358000	5590	65.0	9.42
Ariane 44LP	8300	3900	420000	5480	90.0	10.84
Ariane 44L	9600	4400	470000	5380	110.0	11.46
Ariane 5	18000	6900	725000	15492	100.0	5.56
Atlas I	5580	2250	164300	3630	65.0	10.95
Atlas II	6395	2680	187600	3950	70.0	10.95
Atlas II A	6760	2810	187700	3950	70.0	10.36
Atlas II AS	8390	3490	234000	3950	110.0	13.11
Long March 1 CZ-1D	750	200	80000	1120	10.0	13.33
Long March 2 CZ-2C	3200	1000	191000	2840	16.0	5.00
Long March 2 CZ-2E	9200	3370	464000	6000	35.0	3.80
Long March 2 CZ-2E/HO	13600	4500	460000	6000	50.0	3.68
Long March 3 CZ-3	5000	1500	202000	2840	20.0	4.00
Long March 3 CZ-3A	7200	2500	240000	3000	30.0	4.17
Long March 4 CZ-4	...	1100	249000	3000	18.0	4.50
Delta 6920 (Delta II)	3990	1450 (Δ6925)	218000	3580	40.0	10.03
Delta 7920 (Delta II)	5045	1820 (Δ7925)	230000	3650	45.0	8.92
H-2	10500	4000	266000	4050	90.0	8.57
M-5	1950	1215	128000	4220	40.0	20.51
STS (Shuttle)	23500		2040000	28590	130.0	5.53
Titan 3	14515	5000	680000	17220	130.0	8.96
Titan 4	17700	8620	860000	19820	150.0	8.47
Titan 2	...	...	155000	4180	43.0	
Pegasus	455	125	19000	490	10.0	21.98
Energia SL-17	88000	...	2400000	34880	80.0	0.91
Proton SL-13	20000	5500	705000	8840	30.0	1.50
Zenit SL-16	13740	4300	466000	7260	25.0	1.82
Soyuz SL-4	7000	...	290000	4029	15.0	2.14
Vostok SL-3	4730	...	290000	4029	14.0	2.96
SS-18 (ICBM)	4400	...	...	...	13.0	2.95
Tsyklon SL-14	4000	...	190000	2970	12.0	3.00
Kosmos SL-8	1350	...	120000	1486	6.0	4.44
Vysota SSN-8	150	...	...	...	5.0	33.33

8.1.2 Piggy-back Options & Small Launch Vehicles

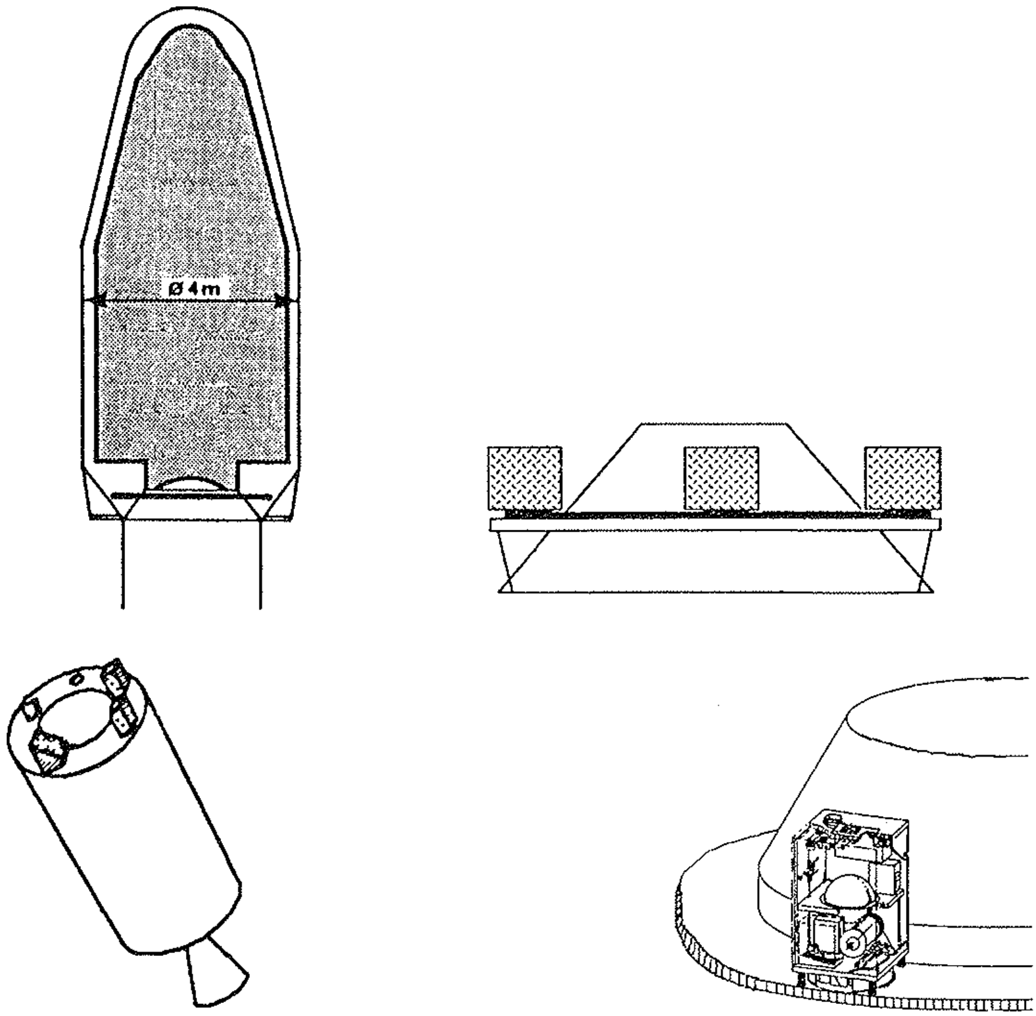
For the first space solar power demonstration mission in the 1990s, the mass of the satellite could be below 1000 kg, so there are piggy-back and small rocket options to launch it. The principle of piggy-back launch system is based on the fact that most launch vehicles do not charge precisely on the basis of the satellite mass. Excess capability can be used to launch secondary small satellites without modification of the launcher. The available mass depends on the mass of the main passenger and the performance of the launcher. The secondary payloads are placed into the same orbit as the primary payload.

The number of launch opportunities is limited. To date there are 15 to 25 commercial launches per year with expendable launchers (US and Europe) and about 6 to 8 flights for the US Shuttle. For an inexpensive and near term space solar power demonstrator we need the combination of minimum launch costs and maximum launch opportunities so piggy-back options appear to be very attractive. In this case the mass of the satellite is between 50 and 500 kg. If the mass is higher, small launch vehicles can be used. They are capable of lifting payloads ranging from 100 to 1000 kg into low Earth orbits.

Review and Analysis of Piggy-back Options

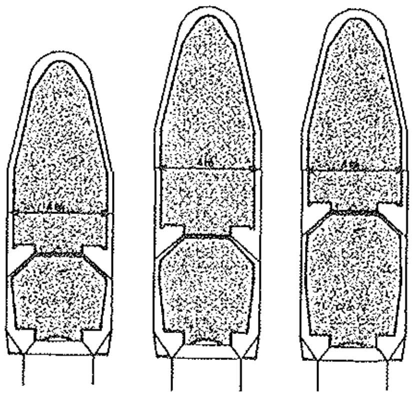
Here is a list of piggy-back options:

Ariane 4 - ASAP (Ariane Structure for Auxiliary Payloads)



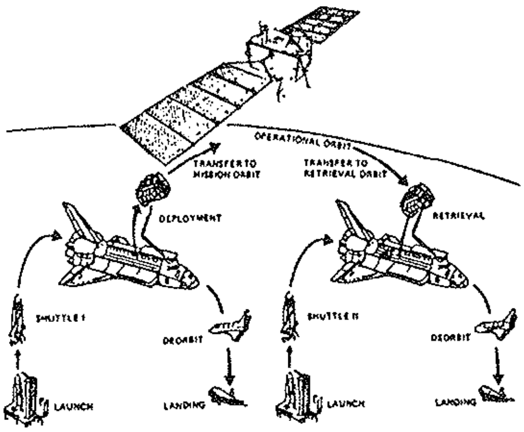
ASAP (Ariane Structure for Auxiliary Payloads) is a small payloads carrier below the prime passenger of the European Ariane 4 launch vehicle. It can carry up to six small payloads up to a total of 200 kg, with a maximum of 50 kg for any one particular satellite. This circular plate extends into the base of the fairing volume. ASAPs are used for LEO polar missions but it is also possible to use it for launches into GTO missions.

Ariane 4 - Ultra Short Spelda (USS)



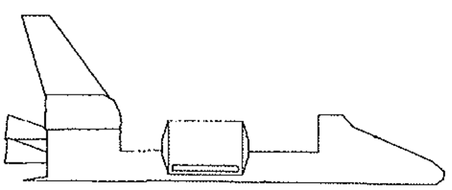
SPELDA (*Structure Porteuse Externe Lancement Double Ariane* / Ariane dual launch external bearing structure) has been developed to have a dual launch system on Ariane 4. This carbon fiber structure surrounds the inner spacecraft and supports the upper spacecraft and the fairing. There are three sizes: long, short and now ultra short. The USS can be used to launch one small secondary payload into GTO or polar orbit with a 400 to 800 kg mass.

Eureca Platform



Eureca (EUropean REtrievable CArrier) is a multi-disciplinary platform that is launched on US Shuttle mission (first one in 1992), left to function autonomously for six to nine months, and then is recovered by a later Shuttle. The mass available to payload is 1000 kg with a volume of 8.5 m^3.

SpaceLab



SpaceLab was built as the European contribution to the US Space Shuttle program and consists of a pressurized module and an unpressurized pallet structure. This structure is fixed in the US Shuttle cargo bay and can be used to expose payloads (about 100 kg) to the space environment.

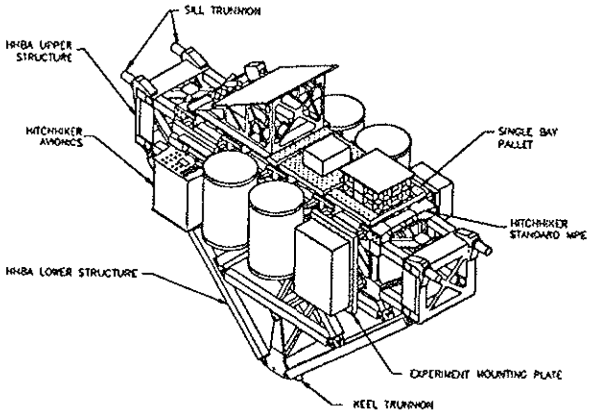
SpaceHab

SpaceHab is a pressurized module integrated into the Shuttle bay and can be configured to hold up 72 mid-deck lockers (capacity of 27 kg).

GAS Can & GAS CAP

The Get Away Special Canister and GAS Complex Autonomous Payload are systems attached to either site of the US Shuttle cargo bay. For the GAS CAP the maximum ejection mass is about 90 kg. Payloads can either be returned with the Shuttle or ejected into LEO.

Hitchhiker System



This system is a combination of the GAS systems (various configurations of GAS Can and GAS CAPs). The maximum mass for the payloads is about 900 kg.

SPAS

SPAS (Shuttle PAllet Satellite) system is made up of the same basic structural elements as Eureca and is capable of carrying up to 900 kg of payload.

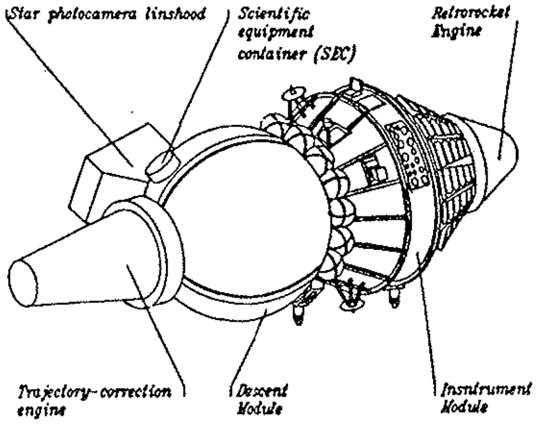
Atlas 2 & Delta 2

With US Delta and Atlas launchers, the piggy-back system can launch 200 to 500 kg.

RUSSIA (CIS)

With Russian launchers, piggy-back options are:

- Resurs system (variant of the Vostok manned capsule)



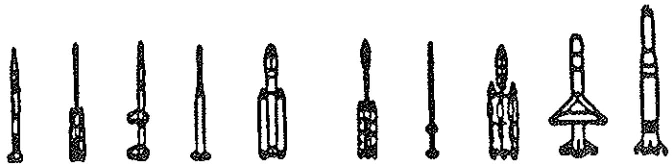
- Foton Capsules
- RAPUNZEL piggy-back system for deployment of tethered satellite or re-entry capsule (Rope attached piggy-back unit zooming on environmental data at low cost).

These capsules are able to support a range of piggy-back launch opportunities in a Low Earth Orbit.

CHINA (FSW)

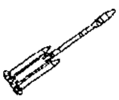
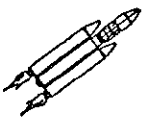
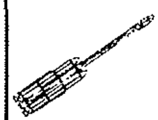
The Piggy-back service called FSW platform is provided by CAST (Chinese Academy of Space Technology) using Long March 2 launch vehicle (CZ-2C).This platform is capable of carrying various kinds of equipment with recoverable and non-recoverable payload from 50 to 300 kg.

Review and Analysis of Small Launch Vehicles



All countries started their space programs by building small launch vehicles and there is a growing range of them available today. These launchers could launch payloads below 1000 kg (LEO) and can be used for launching a small space solar power demonstration payload. Table 8.2 shows the list of small launchers available today and currently at the end of the development phase.
Even though these vehicles are more expensive than other piggy-back launch opportunities, they offer the lowest dedicated launch costs (see Table 8.2).

Table 8.2: Small Launch Vehicles

Name	Country	Company	Payload LEO (kg)	Remarks
Pegasus 	US	Orbital Sciences Corporation	430	
Scout 1	US	LTV	200/260	
Scout 2 	US / Italy	LTV / SNIA	450	Under development
Conestoga 	US	Comet SS inc.	450	Under development
Cosmos L-V	CIS		1250	
Shavit	Israel		145	
ALV	Australia		910	
VLS	Brazil		135	Under development
ASLV 	India	ISRO	150	
CZ-1D (Long March)	China	CGWIC	750	
Orbital express	US	Microspace	180	Under development
M-3SII	Japan	ISAS	460	
J-1	Japan	NASDA / ISAS	1000	Under development

8.2 Review and Analysis of Upper Stages/Orbital Transfer Vehicles

8.2.1 Definitions

Orbital Transfer Vehicles (OTV's) are important parts of the Space Transportation System. Their missions start from a stable Earth orbit—typically a Low Earth Orbit (LEO)—established by a launch vehicle, and their task is the transportation of payload(s) into other orbits and/or back -mainly higher energy orbits as :

- Sun Synchronous Orbit (SSO)
- Geostationary Transfer Orbit (GTO)
- Geostationary Earth Orbit (GEO)
- Interplanetary orbit

Due to their different task and the resulting type of vehicle it is useful to distinguish OTV's and OMV's (Orbital Maneuvering Vehicles). These OMV's are also dedicated to payload transfer but only in a zone near an orbiting system, i.e. Space Station Freedom or MIR. Only small orbit changes are part of their task and generally a ΔV of 250 m/s is considered as the upper limit for this kind of vehicles. Figure 8.1 shows typical mission profiles for each vehicle type.

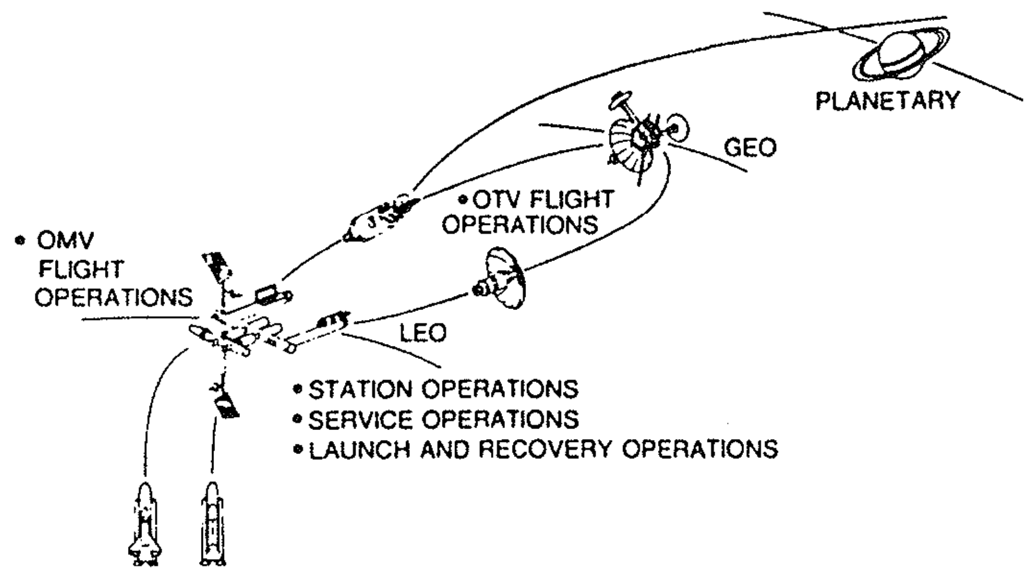


Figure 8.1 OMV/OTV Missions

Now the discussion will focus on OTV. We are going to classify the missions it could achieve. In fact the most important parameters to design those vehicles and to select the most efficient concept are the ΔV (Velocity Increment) they have to provide to the payload and the orbit they have to reach. Thus the classification is generally realized in term of ΔV and mission profile.

Type 1. To deliver payload from LEO or Space Station to GTO
$\Delta V \approx 2500$ m/s

Type 2. To deliver several payloads (in one mission) from LEO to higher orbits (GTO, SSO, GEO) -Bus stop mission-
2500 m/s $\leq \Delta V \leq$ 5000 m/s

Type 3. To perform servicing missions in GEO with return to LEO
for example : LEO-GTO-GEO-GTO-Aerobraking-LEO
$\Delta V \geq$ 6500 m/s

Type 4. To deliver payload(s) from LEO into lunar or interplanetary trajectories
$\Delta V \geq 4000$ m/s (which is the average value to escape the Earth)

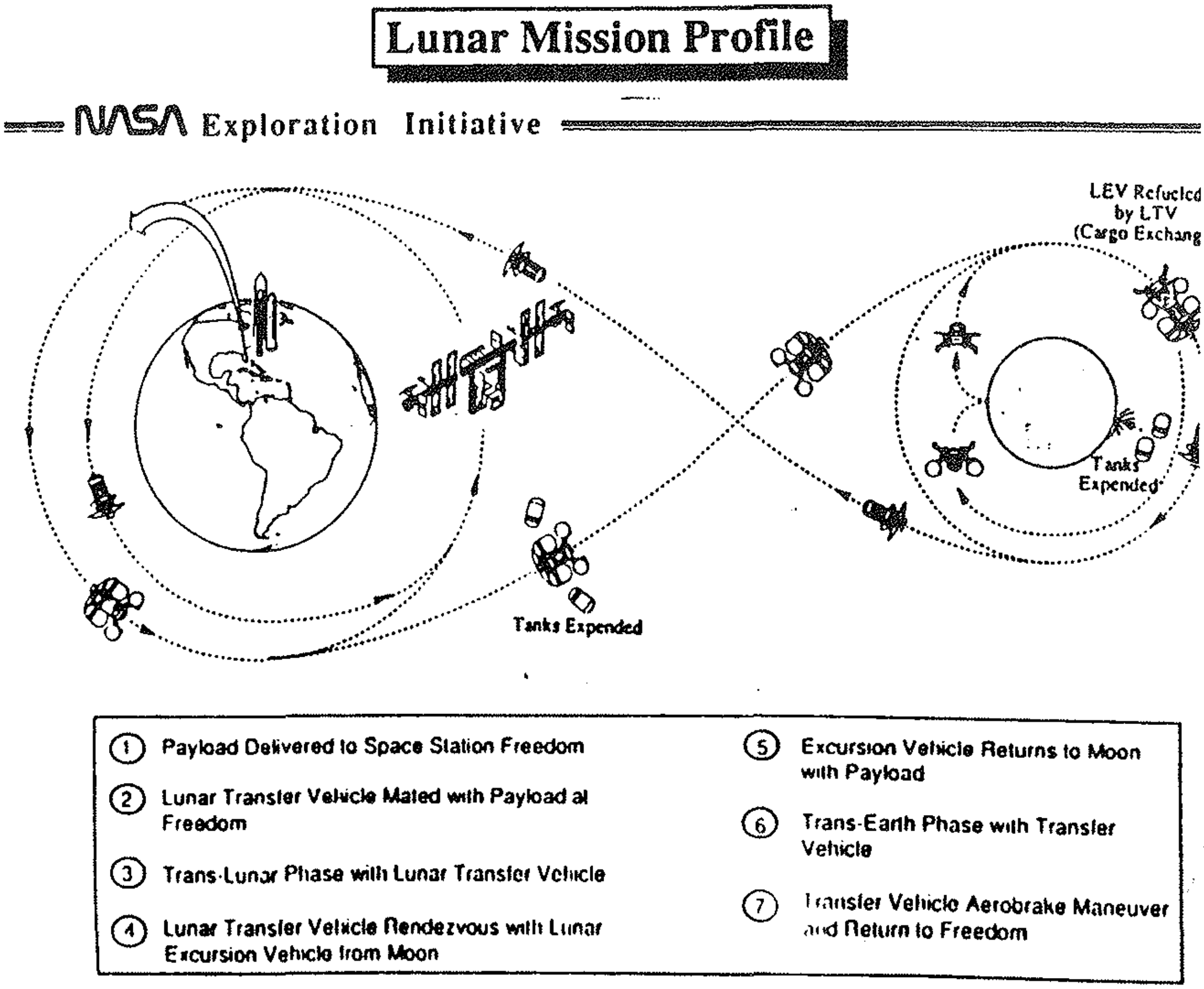


Figure 8.2 Example of Type 4.

As future launchers (reusable) shall be optimized for LEO capabilities, OTV's will be an important part of Space Transportation System fleet and will require particular vehicles designed for the mission type described above. And each of them will use an optimized propulsion system which can be :

- Storable propulsion
- Cryogenic propulsion
- Nuclear propulsion
- Electric propulsion
- New kind of propulsion

This subject will be widely treated in the later sections of this chapter.

8.2.2 Present Status of Upper Stages/OTV's

Current conventional launch systems (Ariane 4 for example) directly inject payloads into the envisaged target orbits of OTV's and don't need staging in Low Earth Orbit. This means that the upper stages work as OTV. But changes of strategy (as American NSTS or Ariane 5) with reusable orbiters requires a LEO staging.

Consequently new upper stages (i.e. EPS-L9 for Ariane 5) are the first step in the OTV field. The Table 8.3 summarizes the main characteristics of OTV like vehicles currently in operation or in development (and available before 1997).

Remark :The functions and characteristics of all these systems have three common features which are really typical of the current Space Transportation Systems i.e.

1. Dedicated to a particular payload transfer mission (Mass + ΔV)
2. Expendable
3. Using a chemical propulsion system (solid or liquid rocket engine).

Table 8.3: Main Characteristics of OTV-like vehicles

NAME (COUNTRY) / CHARACTERISTICS	PAM-D (US)	PAM-DII (US)	TOS (US)	IUS (US)	CENTAUR G (US)	TRAN STAGE (US)	EPS (EUROPE)	MODULE D (Former USSR)
STAGE : manufacturer	MDAC	MDAC	MMC	BAC	GD		MBB-ERNO	Proton 4th stage
length (m)	2.0	2.0	3.05	4.7	9.0		3.35	5.5
diameter (m)	1.2	1.6	3.4	2.9	4.6		3.65	4.
ENGINE : manufacturer	THIOKOL	THIOKOL	CSD	CSD	P & W	AEROJET	MBB	
type	STAR 48	ISTP	SRM-1	SRM-1 & -2	RL10A	AJ10-138	Pres. fed	Pump fed
number	1	1	1	2 stages	2	2	1	1
propellant (s)	solid	solid	solid	solid	LH2+LO2	Az50+MON	Mmh+MON	Ker.+LOX
TOTAL THRUST (kN)	64.5	78	200	200 81	17	71	27.5	85
Specific Impulse (sec)	205.6	-	294	293 301	328	302.2	318	362
Burn time (sec)	85	121	150	153 105	-	-	810	600
STAGE : Total weight (Tons)	2.2	3.5	10.75	14.8	27.8	-	8.12	17.3
Propellant weight (T)	2.0	3.25	9.7	12.6	20	-	7.0	14.6
Dry mass (T)	0.2	2.5	1.05	2.25	7.8	-	1.12	2.7
Airborne sup. equip. wt	1.15	1.6	1.45	3.35	N/A	N/A	N/A	N/A
PAYLOAD : to GTO (Tons)	1.25	1.9	5.9	7.71	N/A	8.3	N/A	N/A
(one way) to GEO (Tons)	0.64	0.95	3.0	2.3	5.9	4.7	6.8	2.6
SCHEDULE : Status	Operational	Operational	Operational	Operational	Operational	In develop.	In develop.	Operational
(Operational date)	(1982)	(1985)	(1986)	(1982)			(1995)	(1992)
Type of development	Commercial	Commercial	Commercial	US GOV'T	US GOV'T		EUR. GOV'T	USSR GOV'T
(sponsor)	MDAC	MDAC	OSC	USAF	NASA		(ESA)	

8.2.3 OTV Analysis

Market Survey

First of all let's present a rough near term market analysis to show that such vehicles have a real potential market outside the Space Solar Power Program needs.

A dedicated study was performed in Europe (based on previous studies of Arianespace and satellite operating organizations) in order to identify the coming market. Eastern world needs (mainly former USSR and China) were not taken into account and it was assumed that total number of commercial payloads can be extrapolated from the previous years. Table 8.4 presents the average results in term of commercial payloads to be launched per year.

Table 8.4 Average Commercial Payloads Per Year

MISSION	Payload Mass (Kg)	1989-1991	1992-1996	1997-2001	2001-2015
GTO - GEO Telecommunication	GEO, BOM 1900-2500 kg	15 - 16	13 - 19	10 - 13	15 - 20
Meteorological	600 - 1000 kg	1.6	2	1 - 2	2 - 4
SSO (polar) Earth observation	SSO, BOM 2000 - 5000	1 - 1.5	1 - 2	1 - 2	2 - 4
LEO Navigation, Scientific,	1200 - 2000	3	2.5 - 4	3	3
Astronomy, Micro-g., etc	1000 - 10000	1	1	1.5	1.5 - 4.5
International Space Station	13900 - 17900	-	-	12 - 15	8 - 11
European Space Station :	17200	-	-	-	3 - 6
Space Tourism	?	-	-	-	?
Space Power Station	?	-	-	-	?

Looking at those figures, with an available market for OTV's between 15 to 30 units per year after 1997, it becomes clear that there is a commercial slot for those vehicles even without Space Solar Power Program needs.

Performance Requirements

The required performance is first characterized by the velocity requirement (ΔV) and the payload mass. But considering the high level of transport cost in Earth orbit i.e. 20 k$/kg in GTO for Ariane 5 (ESA/CNES cost target), for each mission type a complete mass optimization has to be performed in order to save every kg of payload which has to be delivered in orbit.

Another input of such optimization is the possible reusability of OTV (or parts of them). But due to the cost of carrying kg of propellants in high orbit just to return the vehicle, it can be demonstrated that a reusable OTV is cost effective even with the use of braking devices (i.e. aerobraking) only if the propulsion system can provide a specific impulse of more than 800 s, which widely exceeds the current (or near future) engine capacities. Thus all *Space Power System projects have to be aware that economically speaking for the near future only an expendable vehicle has to be considered* except if for technical purposes it is necessary to bring back payload in Earth orbit from higher orbits or from the Moon.

Also, important means for OTV propulsion system selection/optimization is staging. For given design and performance parameter (Isp) the number of stages can be derived by theory. Obviously multi-stage OTV's is much more complicated than single-stage ones. Thus,

- solid rocket systems are limited to $\Delta V < 3000$ - 3500m/s
- storable liquid systems are limited to $\Delta V < 4000$ - 4500m/s
- cryogenic liquid systems are limited to $\Delta V < 5500$ - 6000m/s
- nuclear or electric propulsion systems have higher limitation (respectively in the range of 15,000 & 30,000 m/s)

Moreover, when for a certain mission a propulsion system is selected it is very important to enhance as much as possible the specific impulse of its engine because of mass savings. For example for an Ariane 5 LEO-GEO transfer (18000 kg in LEO) one more second of Isp allows to gain from 15 to 35 kg on payload (i.e. from 0.3 to 0.7% of the GEO mass capacity)

At least several factors apply to the thrust level selection of OTV main engine,

- gravity losses make the use of small thrust engine (< few kN) out of interest,
- high thrust level is limited because of payload constraints (acceleration),

so the best level of thrust is in the range of several tens of kN.

Beside all those performance limitations there are also boundary conditions which influence or will influence the real worth of OTV : Handling and ***Environment limitations***. This last aspect would certainly limit the use of a promising system in term of performance - nuclear propulsion system - to non Earth orbit applications at least.

Here above we set up one of the most important requirements that must be deal with for an OTV selection/design and roughly it leads to the following classification ;

type 1. missions are dedicated to liquid storable propulsion system

type 2. missions are dedicated to liquid cryogenic propulsion system (or very high performance storable system)

type 3. mission is out of interest with the current technologies

type 4. missions are dedicated to nuclear (if allowed) or electric propulsion system.

OTV Cost

Estimating the price of Orbital Transfer Vehicles is a very tough job, mainly because it is difficult to fix assumptions and the level of technology used Therefore figures hereafter are only order of magnitude estimates.

The average cost per OTV unit (using chemical propulsion) is currently between 1 and 20 M$ (US$).

Conclusion - OTV in the Near Future (> 1997)

We would like to put emphasis on the fact that the market analysis result is confirmed by the numerous and various studies on OTV and their propulsion which have been conducted up to now (mainly in the U.S.A. but also in Europe). For example ESA is currently performing a study on a high performance OTV-engine using space storable propellant under the name *Advanced Technology Engine* (which could be used on Ariane 5 OTV concept -cf. fig. 8.3) and some developments for the propulsion systems are underway in the US (USAF XLR 132 engine). Consequently, coming with the next generation of Space Transportation System (after 1997) there will a commercially available OTV and *Space Solar Power Program should take into account the use of such vehicles* if needed

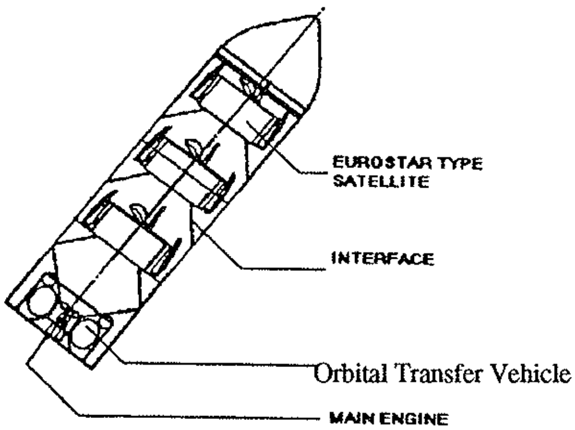


Figure 8.3 OTV Concept for a Three Satellite LEO to GEO Bus Stop Mission (from Ariane 5)

8.2.4 Future In-Orbit Vehicles

Electric Propulsion Orbital Transfer Vehicles

The technology of electric propulsion will allow reductions in the cost of space transportation. It enables this through:

- The reduction of the mass launched into orbit
- The reduction of the size of the propulsion system
- The reduction of the size of the launch vehicle required to place a payload into orbit.

Electric propulsion differs fundamentally from chemical propulsion in several ways. First, a large electrical power system is carried on-board the vehicle to provide energy to electric thrusters. This electrical energy is provided to thrusters which accelerate a propellant at very high speeds. This acceleration produces a very high I_{sp}. Because of the high I_{sp}, the total mass of the vehicle can be significantly reduced over chemical propulsion. This is especially true for the very high energy missions. The performance of an electric OTV is strongly dependent upon the power technology, power level and the I_{sp} of the thrusters. For each mission type, a series of trade studies and an optimization of power level and thruster performance is therefore recommended.

Secondly, the thrust levels with electric propulsion produce a low thrust-to-weight ratio: less than 10^{-4} and typically 10^{-6}. This requires the vehicle to thrust for a long period of time to accomplish its mission. Also because of the large power systems used, the vehicle is often very large. Because of the low thrust levels produced, the structure can be lightweight and flexible.

There are several thruster technologies that are appropriate for orbital transfer vehicles. They are ion, arc jet and Magneto-Plasma-Dynamic (MPD) thrusters. Each system can use various propellants and the performance is dependent upon the propellant selection. Ion propulsion will typically use inert gas propellants, such as xenon, krypton or argon. Arc jet thruster may use hydrazine, ammonia or hydrogen. For MPD thrusters the propellants may be argon, hydrogen or even lithium for very high efficiency engines.

The other important parts of an electric OTV are the structure, guidance system and other subsystems typical of chemical OTVs. Tables 8.4 & 8.5 provide the list of subsystems and the masses of a chemical and electric OTV for a lunar mission. The payload delivered to lunar orbit is 35,000 kg. The most massive part of the electric OTV is the power system [Palaszewski, 1988].

Both solar arrays and nuclear reactors may be used for powering these transfer vehicles. With a space solar power system, however, it may be advantageous to use the same power technology for the transfer vehicle as with the satellite. This would reduce the overall development cost because only one power system would need to be developed. Therefore, nuclear reactors may not likely be developed for Space Solar Power Program if only solar power were part of the system design. On the other hand, a nuclear-electric OTV is very efficient in terms of mass and is less sensitive to the radiation degradation during transfers through the Van Allen Radiation Belts. Solar arrays can be very sensitive to this radiation. Nuclear electric OTVs may also be developed for lunar transportation, Mars missions or other interplanetary missions. Thus, the selection of the power technology may be driven by many factors other than the Space Solar Power Program.

Conclusions

Very powerful benefits of reduced LEO mass and transportation cost reduction are possible with electric propulsion. Though the cost of development of the power system will be significant and perhaps the most expensive part of electric propulsion, the benefits in terms of long term cost reduction are undeniable. The performance of electric propulsion and its ability to reduce the mass of propulsion systems are discussed in the sections on advanced technologies for cost reduction and lunar transportation.

Table 8.5 Chemical Propulsion OTV Mass Summary*

Oxygen/Hydrogen Propulsion: Mp = 40,000 kg

Subsystem	Mass (kg)
Aerobrake	2973.21
Propulsion Main Engines	167.83
Propellant Storage and Feed	1039.24
RCS	1137.65
Power	291.66
Structure	2000.00
Thermal Control	172.72
ACS, Telecom, CDS	251.00
Residuals	609.14
Contingency	864.25
Total	9506.70

For the lunar transportation system, the chemical OTV uses two stages of the mass detailed in Table 8.6. The payload delivered to Low Lunar Orbit (LLO) is 35,000 kg.

Table 8.6 Electric Propulsion OTV Mass Summary*
Xenon Ion (1-MW) - Nuclear
Mp = 13,291.54 and Power System Mass = 10 kg/kW

Subsystem	Mass (kg)
Propulsion Main Engines	128.10
Propellant Storage and Feed	3098.89
RCS	1137.65
Power System and PPU	10164.40
Structure	930.42
Thermal Control	18.70
ACS, Telecom, CDS	251.00
Residuals	216.42
Contingency	1594.56
Total	17540.14

The payload delivered to Low Lunar Orbit (LLO) by the Nuclear Electric OTV is 35,000 kg.

Total	17540.14

The payload delivered to Low Lunar Orbit (LLO) by the Nuclear Electric OTV is 35,000 kg.

8.3 Space Transportation Systems Under Development

In the next ten years, few new launcher developments are to be awaited. In fact, it is most probable that all the new launchers that will appear before 2002 are already under development. Here is a list of the main ones:

Ariane 5 (Europe)

Ariane 5 is the successor of Ariane 4. It is developed to lower the launch costs, to increase the European payload capability in GEO, to have a large payload (20 tons class) in LEO and to provide Europe with a manned launcher (therefore its reliability objectives are higher). In addition, Ariane 5 offers a larger diameter shroud (the maximum payload diameter is 4570 mm).

This launcher is a completely new design. It includes the following main elements:

- a central cryogenic core, the *Etage Principal Cryotechnique* (EPC) with one O2/H2 engine, the Vulcain;
- two solid rocket boosters strapped to the side of the EPC, the *Etage d'Accélération à Poudre* (EAP);
- a second stage, the *Etage à Propergols Stockables* (EPS);
- a Vehicle Equipment Bay (VEB);
- a 5400 mm diameter fairing.

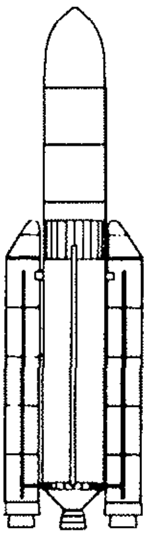


Figure 8.4 The Ariane 5 launcher

The launcher is presented in its unmanned version.

The first flight of Ariane 5 is at the present time scheduled in October 1995, with the Cluster science mission. The development phase is running smoothly and the new ELA-3 launch installations are now being completed in French Guiana.

Like the preceding Ariane rockets, Arianespace will market Ariane 5 and the launcher will be available as soon as it is declared operational (in 1996). So far, no payload has been manifested on commercial Ariane 5 flights. The launch price is 100-110 M$ (EC 1990) [Isakowitz, 1991].

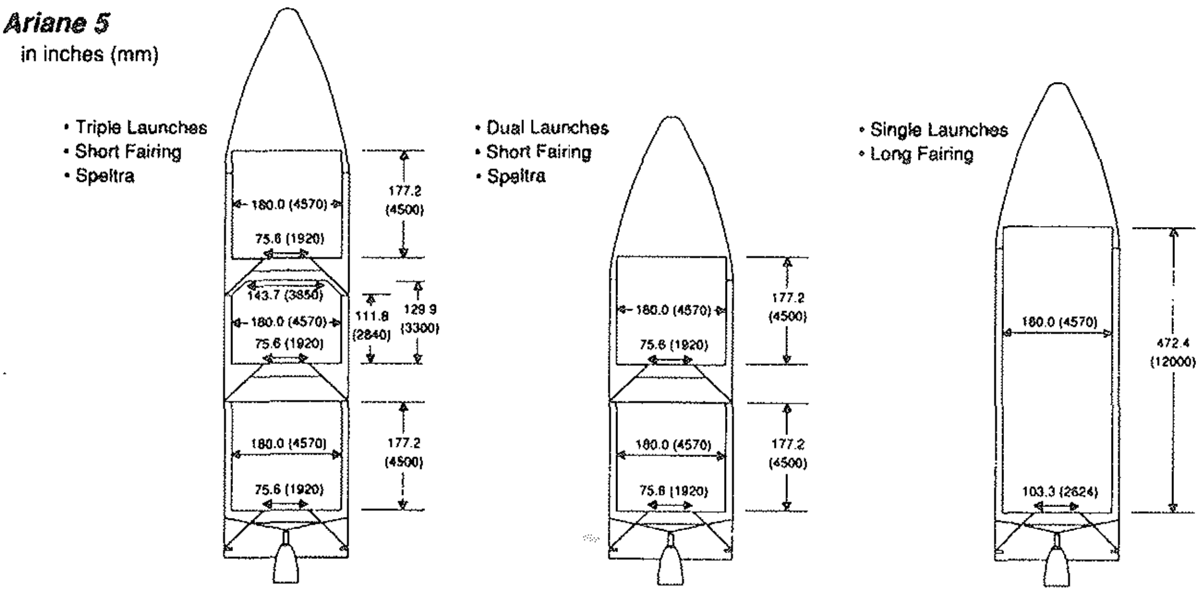


Figure 8.5 Payload accommodation on Ariane 5

[Isakowitz, 1991]

After the end of the development of the Ariane 5 launcher, it is at the present time envisioned to look for derivatives in two directions: an increased launch capability with an upgraded Vulcain (Ariane 5 Mark II) or two Vulcain's instead of one, and small launchers to replace Ariane 40, based on the Ariane 5 solid rocket booster, with Ariane 4 or to-be-developed upper stages (projects "DLA-V" or "DLA-S"). None of these projects is likely to give birth to an operational launcher before 2002, with the exception of Ariane 5 Mark II, which could offer a payload increase of two tons in LEO, which corresponds to two Atlas IIAS-class satellites in GTO.

H-2 (Japan)

The H-2 replaces the H-1 which performed its last flight at the beginning of 1992. This new vehicle is being developed fully by the domestic technology of Japan. It consists of cryogenic first and second stages and a pair of solid strap-ons. It will be capable of 4 times more payload to GEO than H-1. First flight is scheduled for 1994 with the OREX payload, but since it is fully booked until 1997, it can only be considered for the second Space Solar Power Program demonstration.

Japan has expressed its intention to eventually market this launcher on a commercial basis, but there are constraints on the launch windows due to the fishermen around Tanegashima Space Center. It is so far limited to two launch windows per year, which would authorize a maximum of four launches a year. The launch price is 100-120 M$ (EC 1990) [Isakowitz, 1991].

A configuration, called H-2D, with six Solid Rocket Boosters instead of two on the basic H-2 is being studied as a growth version. This launcher is necessary to launch the large 20 ton class HOPE, an unmanned winged vehicle accommodated on top of the H-2.

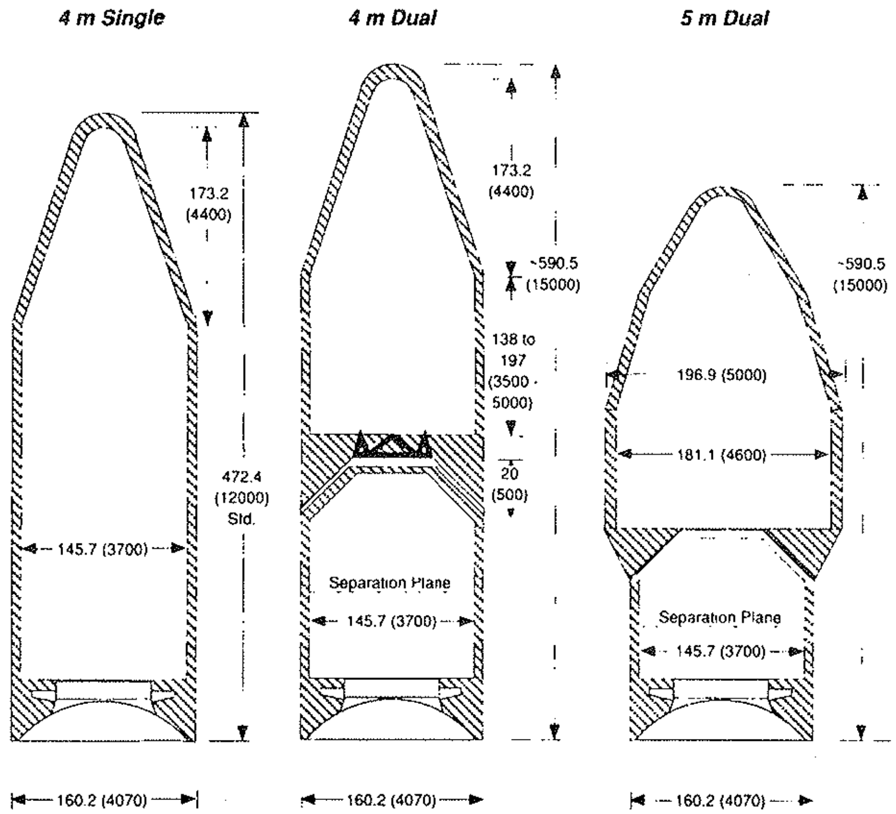


Figure 8.6 Payload accommodation on H-2

[Isakowitz, 1991]

Another planned derivative is the J-1, which is a small launcher derived from the SRB of the H-2. The first flight is scheduled for 1995, from the former H-1 launch pad.

Taurus (United States)

Orbital Sciences Corporation was awarded a contract in July 1989 by DARPA for a demonstration launch of a four-stage, inertially guided 3-axis stabilized solid propellant standard small launch vehicle (SSLV) called Taurus. The Taurus vehicle configuration is derived from Pegasus and Peacekeeper stages. The first demonstration launch is about 1992.

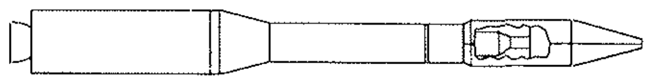


Figure 8.7 The Taurus Launcher [Isakowitz, 1991]

8.4 Previous Studies

This section presents a literature review of the work compiled in the NASA/DOE report for the Space Transportation Systems proposed to launch and put into orbit large power satellites. These studies were conducted in the late 1970's and early 1980's. Some criticisms are proposed, considering the experience gained during the past decade and today's state of the art in Space Transportation.

8.4.1 Satellite Power System (SPS) Reference Concept Description

The system configuration considered in the NASA/DOE report is summarized in Figure 8.8. It consists of 60 satellites of 50,000 tons each and an output power of 5 GW, to be posted in geosynchronous orbit at a rate of 2 per year, with a pre-assembly phase in LEO (480 km) and a final one in GEO (35,800 km). A permanent construction crew of 600 was estimated.

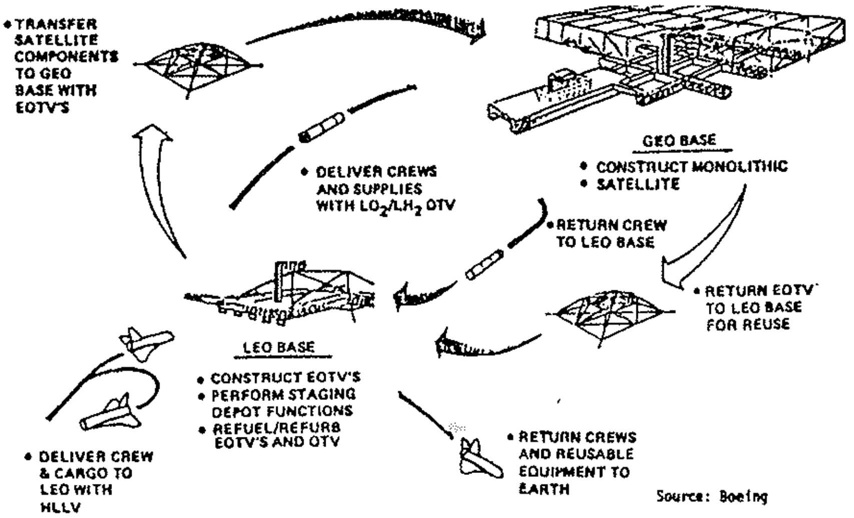


Figure 8.8 SPS GEO Construction Concept.

8.4.2 Space Transportation Systems (STS) Studied.

Earth-to-Orbit vehicles.

The main configurations of Heavy Lift Launch Vehicle (HLLV) were studied for the cargo payload (Figure s 8.9 & 8.10). They resulted from low cost configurations (large payloads, reusability and frequent operations) and state of the art technologies (developed for the Space Shuttle for example).

The *BOEING HLLV Reference concept* has a payload capability of 420 metric tons to LEO and is a winged two-stage, series burn configuration designed for vertical take-off and horizontal landing. Both stages are fully reusable. The booster (first stage) uses 16 LOX/LCH4 engines with high Isp (352 sec) and a vacuum thrust of 9.8 MN each (4.7 times the SSME), with an air breather propulsion system for flyback to the launch site. The orbiter has 14 LOX/LH2 SSMEs and 4 LOX/LH2 Orbital Maneuvering System (OMS) engines.

The main technology requirements for this configuration is the TPS. The conservative ideas of this system were the key factor to its selection as "HLLV Reference Concept" for the SPS project in 1980.

An *Alternate BOEING HLLV concept* with a payload capability reduced to 120 metric tons was also proposed. Its configuration is directly derived from the previous one. The booster uses 4 LOx/LCH4 engines and 4 high thrust air breather engines for flyback. The orbiter has 6 SSMEs. Despite slightly higher recurring costs (greater number of construction crew, more propellant consumed) and more frequent flights, this configuration was recommended by Boeing because of lower non-recurring costs (more commonalty with Space Shuttle), lower facilities costs and a size more appropriate for alternative missions.

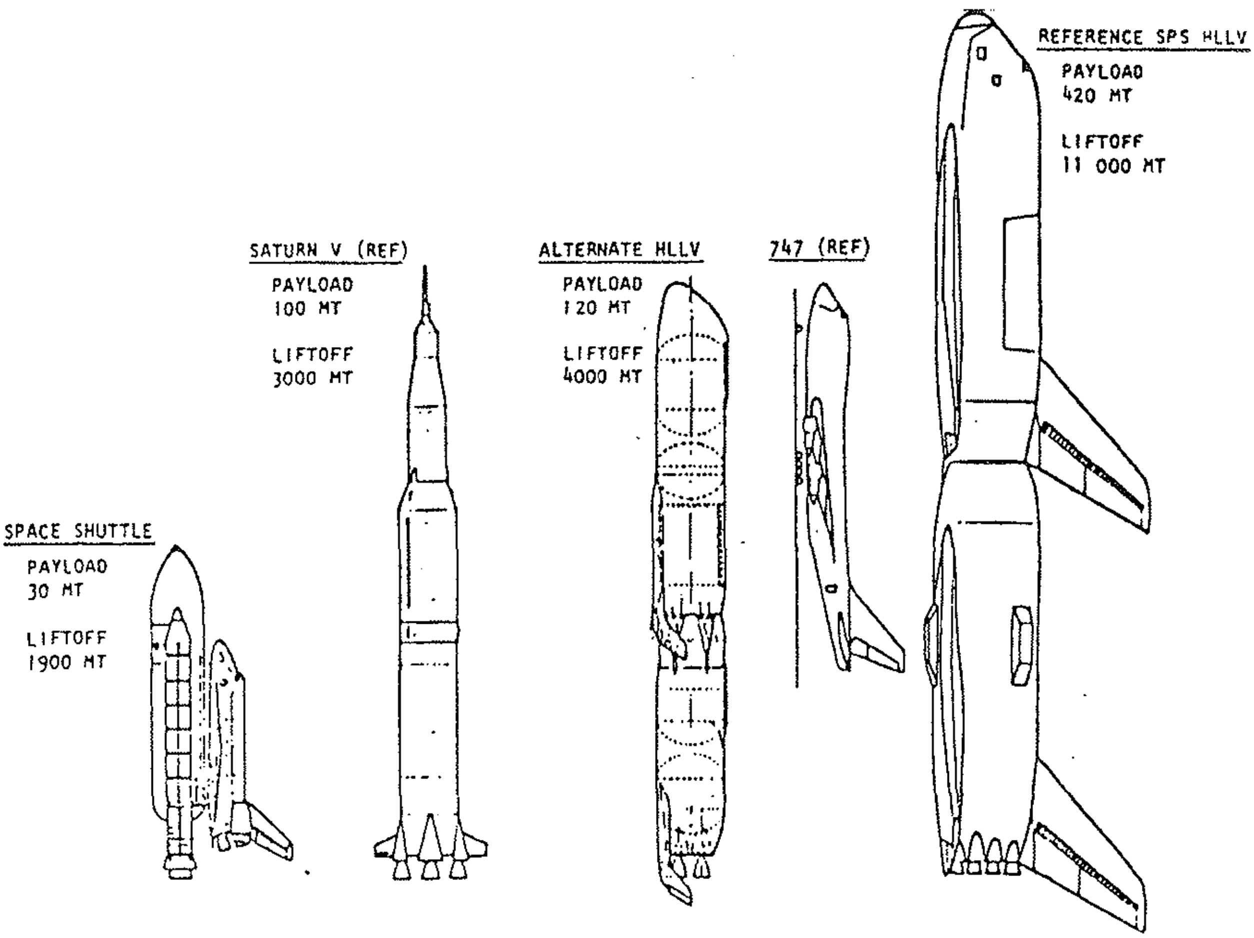


Figure 8.9 Boeing Studies and Comparison with Existing Launchers.

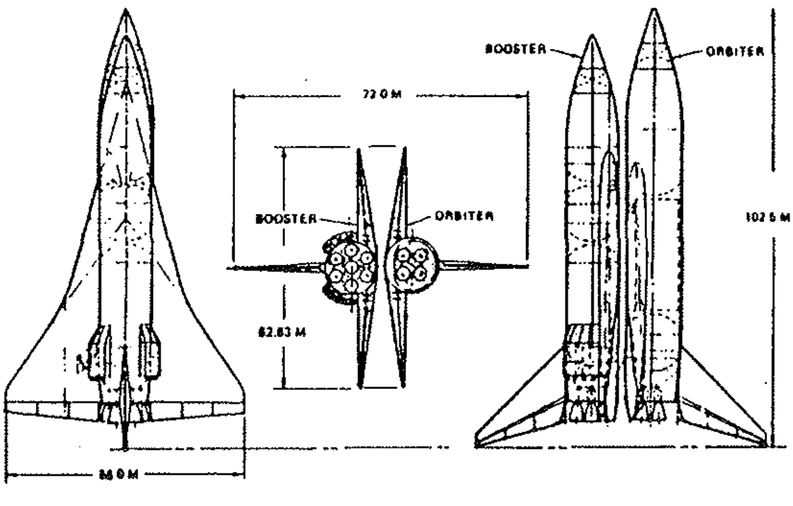


Figure 8.10 Rockwell Configuration.

The *ROCKWELL HLLV concept* has a payload capability of 230 metric tons to LEO and is a winged, two-stage, parallel burn configuration designed for vertical take-off and horizontal landing. Both stages are fully reusable. The booster has 7 LOX/RP-1 (kerosene) engines that operate on a gas generator cycle (with high chamber pressure and relatively high Isp (352 sec)), and an air breather propulsion system for flyback to the launch site. The orbiter has 4 LOX/LH2 engines derived from the Space Shuttle Main Engine (SSME) technology but much larger and with a higher Isp (467 s versus 455 s).

The most critical problems associated with this concept are the new material technology requirements (Thermal Protection System (TPS) and Propulsion System) and very high reliability of the few engines needed.

The requirements for *Personnel Launch Vehicle* (PLV) is to transport SPS construction (600) and maintenance (30) personnel. The Shuttle derivative approach provides a required capability at low investment cost and risk, but advanced PLV concepts were also studied. Boeing proposed a "modified Shuttle SPS Transportation System" (Figure 8.11), using a winged liquid propellant flyback booster powered by 4 LOX/LCH4 engines similar to the Boeing HLLV booster engines, a smaller version of the Shuttle External Tank and the Space Shuttle Orbiter (up to 60 passengers).

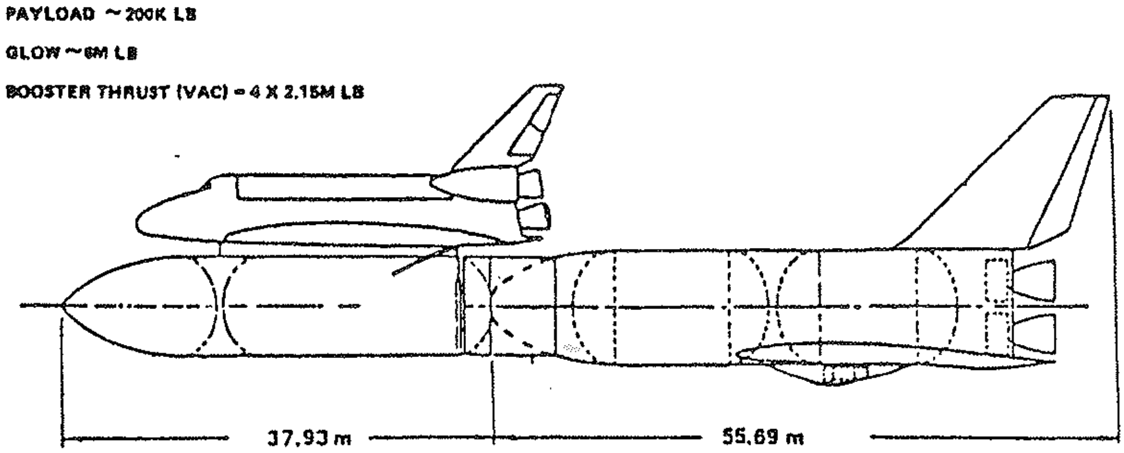


Figure 8.11 Boeing Configuration.

Rockwell proposed a very challenging (for mass and propulsion system performance) SSTO concept (Figure 8.12) using a multi-cycle air breather propulsion system. This vehicle would require major technology developments, in many areas.

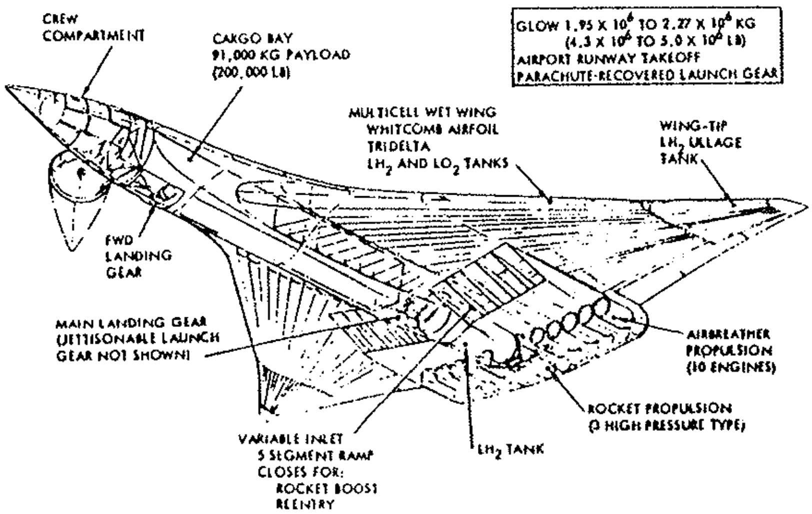


Figure 8.12 Rockwell Configuration.

Orbit Transfer Vehicles.

The cargo *Electric Orbit Transfer Vehicle* (EOTV) concept is shown in Figure 8.13. This electric propulsion vehicle is used to transport satellite components from the LEO staging depot to the GEO construction base. It has a payload capability of 4000 metric tons and is fully reusable. It consists of a solar array (1 km by 1.5 km, portion of a future SPS platform), a payload mounting and docking platform, and ion thruster modules at the four corners of the vehicle (Isp = 8000 s 200 MW power or 220 W/kg for specific mass).

Major (costly) technology advances are required to develop the EOTV, including: fabrication and assembly in LEO, solar cells technology, large structure management. These issues are very similar to those associated to the solar power satellite itself.

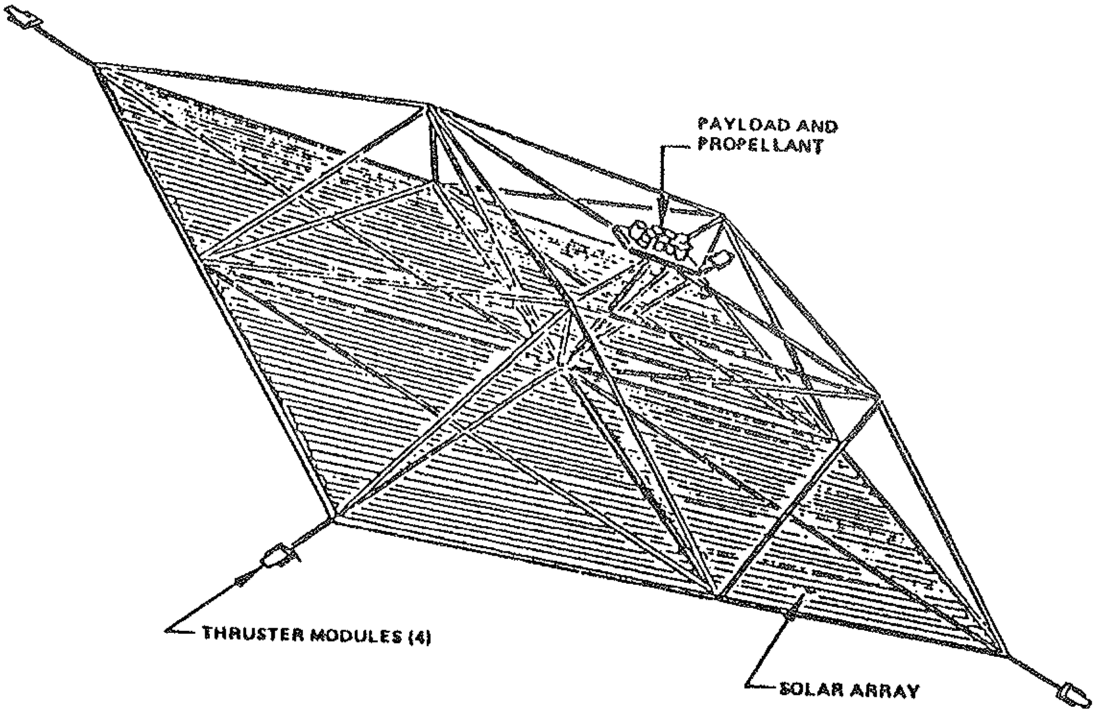


Figure 8.13 Cargo Electric OTV.

The *Personnel Orbit Transfer Vehicle* (POTV) is shown in Figure 8.14. It is a single stage vehicle designed to transport personnel and priority cargo (up to 90 metric tons) from LEO to GEO. Propulsion is provided by five LOX/LH2 90,000 Newton thrust staged-combustion engines (SSME technology) that would operate at an Isp of 470 seconds. Refueling propellant in GEO would be provided by the EOTV.

Storage of cryogenic propellants during the EOTV mission to GEO (180 days) and relatively high propellant mass fraction of the POTV (0.94) are critical technical issues for this concept.

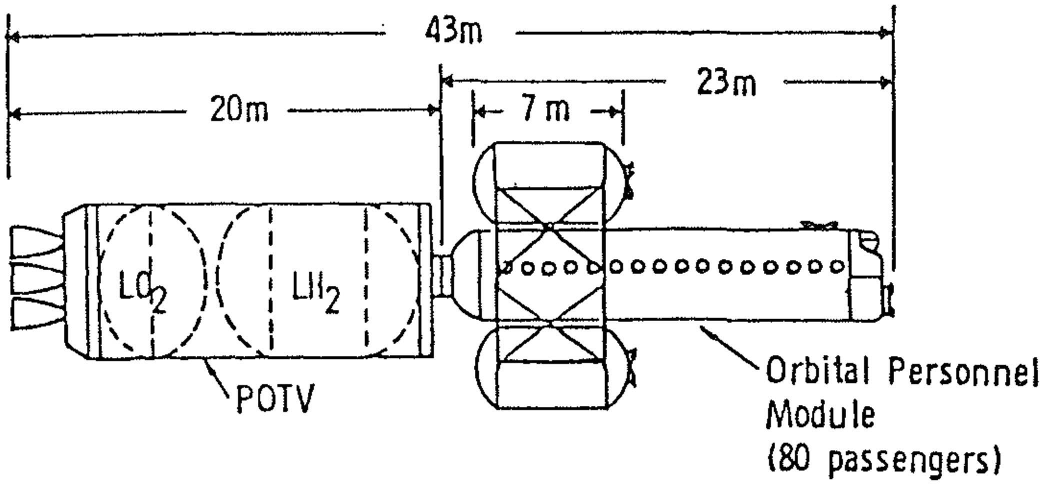


Figure 8.14 Personnel OTV.

An *Intra Orbital Transfer Vehicle* (IOTV) with a 1 or 2 man crew for inspection and service was also evoked in the STS, but no specific study of this concept has been undertaken.

Transportation cost analysis.

Detailed cost studies were carried out for the Boeing Reference Transportation System. They include the ground operations and system maintenance (Kennedy Space Center as reference), and a fleet of: 6 HLLVs, 2 PLVs, 23 EOTVs and 2 POTVs. For each vehicle, the number of flights, the cost per flight, the cost per SPS, and an estimated total transportation cost for DDT&E are summarized in Figure 8.15 (all costs are expressed in 1980 US Dollars). Figure 8.16 shows the relative costs of the transportation system in the total SPS analysis.

In the development phase, the STS represents 45% of total expenditures, and it becomes 25% during the operational phase (with HLLVs accounting for 70% of the STS costs).

DDT&E

o Heavy Life Launch Vehicle	$11,202
o Cargo Orbit Transfer Vehicle	2,247
o Personnel Launch Vehicle	2,616
o Personnel Orbital Transfer Vehicle	1,012
Total Transportation DDT&E	$17,077

Average Transportation Cost per SPS

o HLLV	$\frac{\$10.1m/FLT \times 11,606\ FLTS}{60\ SPS}$	=	$1,954m/SPS
o EOTV	$\frac{\$40.7m/FLT \times 847\ FLTS}{60}$	=	575m/SPS
o PLV	$\frac{\$10.7m/FLT \times 1458\ FLTS}{60\ SPS}$	=	260m/SPS
o POTV	$\frac{\$1.3m/FLT \times 587\ FLTS}{60\ SPS}$	=	12.7m/SPS

Total transportation cost per SPS $2,802m/SPS

Figure 8.15 Total Transportation Cost Summary (in "1980s" millions).

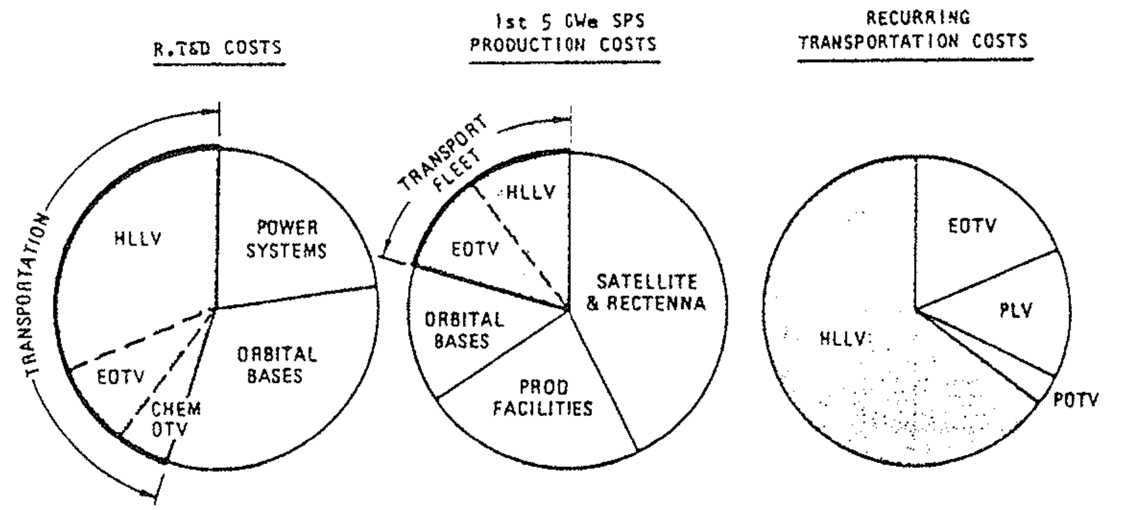


Figure 8.16 SPS Space Transportation Costs.

Some 1992's Comments.

The main technical developments required to make this STS feasible, with the performance assumptions presented above, have been outlined in the previous sections. Although some progress has been made since 1980, there has been no major breakthrough in propulsion systems or material sciences, and the development planning proposed in Figure 8.17 appears today very optimistic without significant efforts (financial) in R&D.

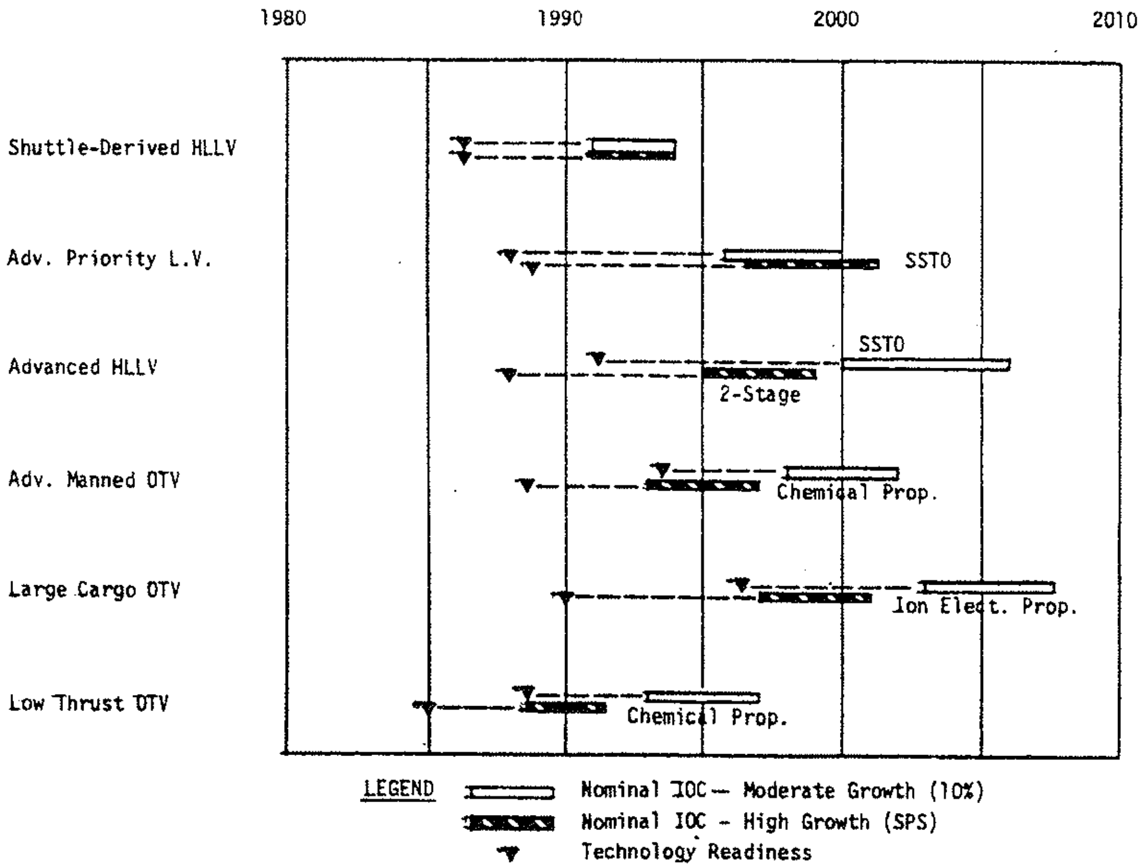


Figure 8.17 Development Planning (in 1980)

Moreover, all technical and transportation cost extrapolations are based on the Shuttle development experience (in 1980) and assume 100% reliability. Space Shuttle operations during the past decade have shown dramatic and unexpected technical limitations (97% reliability, TPS damages, relatively low reusability and high maintenance costs) and significant cost overruns : 5330 (1980) US Dollars/kg of payload for 100 predicted in 1980. Today's current prices for large launchers range from 1000 (with Energia) to 9000 (with Titan 4) (1990) US Dollars/kg of payload. These figures must be compared to the Boeing estimates of 25 (1980) US Dollars/kg of payload with the Reference HLLV!

8.4.3 Previous Heavy Launchers

Introduction

Why must we review previously heavy launchers? A couple of reasons account for this necessity. First of all, we should reaffirm numerous problems that previous launchers had suffered from and its solutions not to come across the same problems. Even if we come across a new issue, reviews of previous heavy launchers may give us the first step to solve it. Particularly in developing a new transportation system, reviews may show us any facts to be considered. Secondly, we can not throw away the previous technologies only because it is old fashioned. Table 8.7 shows payloads of various lineage, launch masses and various historical launchers.

In the following part the Saturn family and the G-1-e launcher are described in more detail.

The Saturn Family (United States)

The Saturn program came from Dr. Werner von Braun's proposal in April 1957 developing a launcher capable of placing payloads between 9,100 and 18,200 kg into a low Earth orbit. The Saturn I consisted of the S-IB first stage, S-IV second stage, and Instrument Unit (IU) and it carried payload up to 10200 kg into east LEO. The S-IB first basic design concept incorporated Jupiter and Redstone components because of their reliability and qualification status. The enlargement and refining of the S-IV stage design and the replacement of six Pratt & Whitney RL10 engines with a single Rocketdyne J-2 engine, produced the S-IVB stage used on the Saturn IB vehicle. This improvement within two years, made it to increase the payload delivered into a LEO up to 16600 kg. On top of this ability Saturn IB can escape payloads of 4,300 kg from the Earth Orbit. The Saturn IB was used to develop and test Apollo hardware and software in rehearsal for lunar mission with the Saturn V. Only 21 months after the first flight of Saturn IB, February 26 1968, Saturn V was launched. The payload capacity placing into LEO was increased orderly up to 119000 kg as compared to 16600 kg for the previous Saturn IB. Also payload escaping from the Earth Orbit was increased up to 50,000 kg. To our surprise, conservative Saturn design margins enabled the launch of the last three Apollo missions although 13% more performance was needed than ordinary required. This was a remarkable progress. In the present situation a similar progress looking at the space transportation in the United States cannot be recognized. It is due to some disadvantages of the US. space shuttle concerning its unreliability and unavailability. Even after 10 years of operation, the space shuttle still suffers from chronic technical difficulties and record of repeated flight delays.

G-1-e Launcher (Soviet Union)

The G-1-e (113m length, 17m in diameter at the base, and 30 engines for the first stage creating a total thrust of 45 MN) was designed to support the Soviet lunar program. The vehicle was designed by Sergei Korolev. With three stages the rocket reaches Earth orbit. The two top stages were used for trans-orbit injection. A factor which may have been more than a little responsible for significant development delays and four launch failures could have been the choice of inexperienced rocket engine manufacturers.

In fact, the G-1-e had been test flown only once, on 21 February 1969 when it suffered an explosive failure at an altitude of 12200 m. The second flight was a rehearsal mission whereby a 3-man Soyuz would dock with the G-1-e in orbit. However, on 3 July 1969, before the Soyuz launch, the G-1-e failed again. This accident sounded the death knell for the Soviet lunar program, even though the Soviet tried twice more to prove their G-1-e in 1971 and 1972. Both flights ended in failure. In the 1971 flight, the vehicle developed an uncontrolled roll seconds after liftoff. The on-board computer shutdown the engines and the G-1-e stack again collapsed back, completely destroying the second pad and gantry, which had been badly damaged in 1969. The last 1972 flight almost made into first stage cutoff. Staging had begun when the first engines were shutdown. Suddenly, a longitudinal pogo oscillations caused a rupture in a propellant line. Fire was followed by an explosion and, at 107 sec in to the flight, the computer shutdown the 24 outer engines still burning. The G-1-e was climbing to a 40 km altitude when the range safety officer destroyed the vehicle. After this successive failures the program was canceled.

Table 8.7 Historical Launchers

Launcher	Country	Launch Site	Latitude	Payload(Kg)	First	Total Num.	Success	Launch		Payload
Name				LEO	Launch	Launched	Rate	Mass(Kg)	Thrust(kN)	Ratio
Europa I	Europe	Woomera	31.1°S	1,350(east)	1968	3	0%	112,000	1332	1.21%
Europa II	Europe	Kourou	5.2°N	1,440(east)	1971	1	0%	112,000	1332	1.29%
Europa III	Europe			5,500(east)	-				2160	?
Diamant A	France	Hammaguir	31.0°N	85(east)	1965	4	100%	18,400	294	0.46%
Diamant B	France	Kourou	5.2°N	160(east)	1970	6	67%	25,000	343	0.64%
Diamant BP4	France	Kourou	5.2°N	200(east)	1975	2	100%	?	392	?
Black Arrow	Great Britain	Woomera	31.1°S	110(east)	1970	2	50%	?	?	?
N-1	Japan	Tanegashima	30.4°N	1,200(east)	1975	7	100%	90,400		1.33%
N-2	Japan	Tanegashima	30.4°N	2,000(east)	1981	8	100%	135,000		1.48%
B-1	Soviet Union	Plesetsk	62.8°N	600(east)	1962	?	[144]	?	?	?
		Kapustin Yar	48.4°N							
G-1-e	Soviet Union	Tyuratam	45.6°N	100,000(east)	1969	4	0%	?	45,000	?
Jupiter C	United States	CCAFS	28.5°N	9(polar)	1958	6	50%	?	334	?
Juno II	United States	CCAFS	28.5°N	41(polar)	1958	10	50%	?	667	?
Mercury Reds	United States	CCAFS	28.5°N	suborbital	1960	5	100%	?	347	?
Saturn I	United States	CCAFS	28.5°N	10,200(east)	1964	6	100%	506,000	7,295	2.02%
Saturn IB	United States	CCAFS	28.5°N	16,600(east)	1966	9	100%	588,000	7,295	2.82%
Saturn V	United States	CCAFS	28.5°N	119,000(east)	1967	13	92%	2,910,000	34,500	4.01%
Thor	United States	CCAFS	28.5°N	770(east)	1958	179	94%	51,500	?	1.50%
		VAFS	34.7°N							
TAT	United States			1,000(east)	1963	67	95%	?	?	?
LTTAL	United States			1,360(east)	1967	17	100%	?	?	?
Thorad-Agen	United States			1,360(east)	1966	22	91%	?	?	?
Vangurd	United States	CCAFS	28.5°N	9(east)	1957	11	73%	10,250	120	0.09%
CCAFS; Cape Canaveral Air Force Station										
VAFB; Vandenberg Air Force Base										

8.5 Future Space Transportation Systems

One of the major show stoppers of the Space Solar Power Program project has been the high transportation costs. This was evident especially from the NASA/DOE studies in the late 1970's. Hence many other studies have pointed out that in the future inexpensive transportation to be essential. This is true not only for Space Solar Power Program but all major space projects.

8.5.1 What is Insufficient with Today's Space Transportation Infrastructure ?

The space transportation infrastructure provided today is a rocky path rather than a highway. So bad is the infrastructure that no industry can be established in space and prosper. Before only governments could afford to walk the rocky path. Today even some governments are finding it to expensive.

In the case of the telecommunication the satellite has to be built to match the transport. Hence a satellites is more like a handmade off-road " Ferrari" instead of a "Volkswagen".

In a space project today the transportation cost is often conceived to make up 25% of the cost. Second order effect due to the bad reliability and availability of the transport increases the true cost to more than 75% of the overall project cost.

To give an example a telecommunication satellite pays 25 to 40% in launch cost. Due to the launcher reliability and an other 20% is paid for insurance. Due to the availability, often a one year delay is customary which would give 10% interest costs. But the big dark area is how much of the extra redundancy in the design is driven by the fact of bad launch availability and hence the long unserviced lifetime.. This factor could be as high as 25%. Hence 75 to 90% of the project cost are transportation related.

To continue the telecom example; imagine instead if a DHL/Federal Express was available for overnight transports to space. How would a satellite be designed

- would it be built for 10 year unserviced life with a lot of redundancy

- would it take 7-10 years in design and manufacturing before it was earning revenue.

With a good transport it would probably be simple, with a backup on ground /in space. The telecom satellites could be manufactured "en masse" and customized late in the design.

In conclusion if transportation's costs are reduced by at least a factor of ten and the reliability and availability is increased then the satellite/space segment cost will also be greatly reduced.

Why do We Have the Current "Rocky Path' Space Transportation Infrastructure ?

Well maybe the customer could afford a gold plated off-road Ferrari. Maybe the industry was pleased to supply what the governments wanted

Today's space transportation rockets have been developed by government to maximize performance and not to minimized cost. One reason for this is due to that the mass of individual payloads have steadily increased. However the most important factor is probably that space activity is conceived as a high tech area. Performance is a good driver for technology and quality. To develop a new rocket from existing technology to minimize cost is not interesting for a government who wants a high tech space industry.

Why then has there been no private initiatives to develop such a vehicle ? Well partly because the industries who have the knowledge have done pretty well making government rockets. Furthermore, the way current aerospace companies operate is not always compatible with the making of a cheap commercial rocket. The few companies who have tried to make cheap launchers have had too little experience or too little funding and a first limited markets

Can and will this change ? Yes for two possible reasons

- Yes if government recognize that low cost is possible and a must for the commercial development of space.

- Yes if for policy reasons the western governments will allow non market economies launch payloads at a substantially lower cost forcing the west to develop new cheaper rockets

What are the Current Thoughts for Near Term Decrease the Cost ?

Within NASA and ESA and in industry the current thought are probably best summarized by the US plans for the NLS. The NLS concept is to reduce cost a factor of ten by cutting a third of today's cost could in design, production and operations,

1/ a factor of three reduction in cost through design by.

- decrease in performance
- increase in design robustness (engine out)
- concurrent engineering to better satisfy manufacturing and operation aspects.

2/ a factor of three reduction in cost through manufacturing by.

- longer series.
- allowing for more structural mass
- decreasing operating temperature and pressure
- using known material
- simplified subsystems i.e. fewer parts, less welding and more casting

3/ a factor of three reduction in cost through operations by

- increased launch rates enables more efficient use of staff and facilities.
- inherently robust systems
- factory checkout simplifies operation
- fewer or more reliable electronic parts

All in all this should enable the launch cost to reduce by at least a factor of 10

What are the Thoughts for Future Improved Space Transportation Systems? What are the Future Customers Requirements ?

It is important to bear in mind that not only the cost but also the availability, reliability, resiliency and launch environment are important factors for a future launch system. A future customer putting up 1 GW of power stations per year would typically need transportation 10 000 000 kg /year. To serve this big mass to LEO a diversified fleet should be used. The payload can divided into Personnel Transport, Priority Cargo and Bulk. Each having different demands on 1/ cost, 2/ reliability, 3/ accessibility, 4/ launch environment and 5/ resiliency. A human cargo would focus on reliability and accessibility. The priority cargo would focus on cost and launch environment. The bulk would focus on cost. In such an scenario the fleet of possible vehicles could be as below

Personnel transport	SSTO: NASP or Delta clipper
Priority Cargo	HLLV: big dumb boosters, SATURN 5 derivatives
Bulk	HLLV or mass driver, RAM accelerator, laser propulsion

8.5.2 Personnel Transport

To enhance reliability a SSTO would enable the use of the fewest possible parts. If it is winged then more failure modes are recoverable. If it is ballistic the flying time is shorter. By making it reusable with a smooth land recovery the operability and hence cost would be enhanced over expendable. Two projects of this type are described. Ballistic and winged SSTO.

Ballistic SSTO Space Transportation systems

The trend in the word is definitely towards fewer stages. The four stage vehicles of the fifties have been reduced to two stage today. By reducing the stages the development cost can be reduced, the cost per launch decreases, the reliability increases and operations are simplified. All of this due to that fewer hardware parts and hence less parts need to be developed, manufactured, assembled and operated.

What are then the drawbacks. The suggested SSTO concepts such as Delta clipper and Beta 2 are designs that presses the SoA in materials, propulsion and operations.

This can be seen by using the most basic of rocket formula for calculating the added velocity given by a stage. This is the "Tsiolkovsky " equation. In this case it is used to calculate the total velocity during the ascent to orbit.

$\Delta V = g*Isp*\ln(Mig / Mbo) = g*Isp*\ln(1+\Pi / \Pi)$

The ΔV is normally for Ariane about 9350 m/s for a Earth to LEO launch. However for a SSTO a "minimum" total ΔV of 8250m/s [Koelle, 1992] has been suggested. The lower Δv suggested is possible due to the use of an extreme low orbit 70 x 200 km.

g is the gravity constant 9.81

Isp is the average specific impulse during the entire launch. Current design on high thrust and high performing cryogenic engines for Earth to orbit (SSME, Vulcain, LE-7) use fixed area ratio nozzles. The current European Vulcain engine has an area ratio of 45 and has a ground level Isp of 335 s and a vacuum Isp 430 s. SSME has 365s and 450s respectively [Heald, 1991]. The average of these engines are some 410-420s. If instead a nozzle design had a variable area ratio or was otherwise adaptable, then this mean Isp could be increased to 425s to 435 s

ln (Mig / Mbo) is the natural logarithm of the ratio between the rockets mass at ignition versus the mass at burnout. This can be restated using the structural factor Π :Mstructure / Mpropellant or the propellant fraction 1 / Π. Today's large cryogenic rocket stages typically have a structural factor of 0.1 i.e. 10% of the stage mass is the structure and the engines.. Numbers for Ariane 5, H-II first stages are 0.097 and 0.13. The STS external tank (which is only the tank) but 5-10 more propellants has a Π of 0.04. New material could replace today's materials then the Π would change from 0.1 to 0.09, 0.07 or even 0.05

Results

Assuming a vehicle with about 400 ton total mass and running the formula with today's values of ΔV 9350 m/s an Ispav= 410 s and a Π=0.1, would result in negative payload (14 tons) if allowing for additional masses in the order of 18 tons

If that same vehicles would be build in near term with the use of deployable nozzles having - Ispav 425 - and state of the art commercial materials -Π=0.09- and by using the lower ΔV of 8250 this would result in a payload 8 tons, a 2% payload fraction

However it is most probable that the SSTO system would be reusable. Hence the recovery system and increased life of components engine would lead to a heavier system. By assuming that a 50% higher structural factor would account for this the resulting vehicle would again produce a negative payload.(-9 ton)

However by using near term technology for adaptable nozzles which would result in an increased Ispav of 435 and a structural factor of 0.08 and assuming that the 18 ton factor can be decreased to 10 tons, then the delivered payload is again 7-8 tons.

The conclusion is that a successful SSTO vehicle seems feasible, if done by taking a stepped approach starting by demonstrating the available technology with an expendable vehicle. The vehicle can then later be updated which new technologies to achieve reusability

Space Transportation Systems for the 21st century—Spaceplanes

Trends Towards Next Generation Space Transportation Systems

Constructions of usable and low-cost space transportation systems are under development, for use as Earth to Orbit vehicle. During the initial stages of spaceflight there is a significant amount of air present. If new engines using air-breathing technology can accelerate these systems up to more than Mach 18 using atmospheric oxygen, these can achieve orbit using a single stage, like an airplane. The United States, former U.S.S.R, China, Japan and France, the Space developed countries, are all researching and developing original launch vehicles with this goal. These developments are summarized in Table 8.8. Of the concepts in this table both MOLNIYA and HOTOL plan to use existing transportation systems to gain sufficient altitude and velocity for air-breathing to function. SANGER is a new hypersonic vehicle under development which will jettison its ramjet when the atmosphere becomes too sparse for effective operation and revert to conventional rocket engines. X-30 and SSTO/NAL are conceived as single stage Spaceplanes.

As MOLNIYA and HOTOL are launched in the air, they have an advantage from the view point of risk analysis during development.

A two stage Spaceplane like SANGER is at first sight a feasible concept. With increases in fuel efficiency due to the air breathing engine and reductions in system weight due to use of advanced materials and structural design, it will realize more in the long term. But there are a lot of problems with it as the next generation transportation system. These are

1. The need to develop two types of transportation (Booster, Orbiter)
2. The problem of vehicle separation at supersonic velocity
3. Due to the utilization of horizontal take-off, it is difficult to plan for emergencies, both in terms of emergency infrastructure and possible crash sites. This is also a serious problem for X-30 and SSTO.

system studies for Spaceplanes are being developed based on the use of air-breathing engines both at The best solution appears to be reusable, horizontal take-off and landing Spaceplanes. All aspects of research, from studies of aerodynamics, structure, materials, flight control, numerical simulation to NASP office in the United States and NAL in Japan.

Spaceplane - A Reusable Winged Single Stage to Orbit

The Spaceplane concept is designed to support a new generation of manned spaceflight. Spaceplanes must emphasize safety, reliability and crew comfort and have unrestricted access for international use. One of the main purposes of Spaceplane development is to reduce launch costs due to reusability of the launcher.

The characteristics of Spaceplanes - reusability, horizontal take-off and landing and air-breathing engine technology - have not yet been tested in space transportation systems. Research in this area, however, suggests that air-breathing technology could be critical in making spaceflight earlier in the 21st Century.

Table 8.8 Space Planes

	MOLNIYA	HOTOL	SANGER	X-30	SSTO/NAL
Main projector / planner	MOLNIYA (Russia)	BAe (UK)	MBB (Germany)	NASP (the United States)	National Aerospace Laboratory (Japan)
Purpose of development	Reduce cost of launch and use spacecraft freely	Reduce cost of launch by 1/5 and use spacecraft freely	To transport materials and persons safely from Europe and reduce cost of launch by 1/10~1/5 and establish the technology of supersonic flight	Establish SCRAMjet engine in order to realize the system of reusable single stage Spaceplane	Establish safe and comfortable manned transportation system
Total length (m)	18.2 (Assumption)	about 52	83 / 28 (lower stage / upper stage)	46 ~ 61	94

Total width (m)	13.0 (Assumption)	-	41 / 16	21 ~ 28	29
Total height (m)	7.4 (Assumption)	about 10.5	-	9 ~ 12	16
Total weight (at launch)	24 t (without external tank)	250 t	254 t / 112 t	113 t ~ 136 t	350 t
Persons and payload weight	2 persons + 7 t (8 t at unmanned)	7 t (no person)	4 persons + 3 t	2 persons	5 persons
Form of engines	2 rocket engines (RD-170) (Assumption)	4 rocket engines (RD-170) (Assumption)	Turbo Ramjet / Rocket engine	SCRAMjet + rocket engine	LACE + SCRAMjet + rocket engine
Form of launch	Launch in the air from vehicle of An-225	Launch in the air from vehicle of An-225	Horizontal take-off	Horizontal take-off	Horizontal take-off

Two Stage to Orbit

This launch vehicle uses an air breathing first stage and accelerates the second stage to a speed of approximately Mach 6-10. The first stage is a winged airplane. A rocket powered stage then proceeds to orbit after staging. The air breathing stage flys back to an airport for refurbishment. The second stage may be either a payload canister or a reusable flying vehicle. The operations of this vehicle can potentially be very simple compared to traditional rockets. It is also a potential interim step prior to developing the NASP. Turbojets and scramjets on the TSTO will produce much lower velocities than that for the NASP. Therefore the air breathing technology is much more near term than the Mach 25 scramjets needed for NASP.

8.5.3 Priority Cargo

To allow cost reduction through increase in size, to keep a reasonable launch environment and to allow for large pre integrated structures; a Heavy Lift Launch Vehicle (HLLV) could be used. Three concepts are looked at. The historic SATURN 5 and a to be developed "Big Dumb Booster" and a single stage to orbit concept.

Saturn 5 - Feasibility of Improvement

Only the historic Saturn 5 and the present Energia have a payload capability of over 100 tons into LEO. In the following part an update of the Saturn 5 system technology is discussed. The Saturn 5 vehicle is the most powerful rocket ever developed and flown in the United States. Nowadays, the Saturn 5 is not in use because it was only developed for the moon race against the Soviet Union. It carries a payload of 119 tons into LEO. An improved Saturn 5 could be a possible future launch transportation system for the Space Solar Power Program. Two solutions to improve Saturn 5 are namely, (1) improvement of the first stage engine and (2) using additional rocket booster for the first stage. To minimize the cost of this improvements risky new technologies should be avoided.

Improvement of First Stage Engine

Saturn 5's first stage engine (F-1 engine) is a simple gas generator cycle engine, and it produces huge thrust under very low chamber pressure, as shown in Table 8.9 below.

The specific impulse of the F-1 engine is lower than the RD-170 engine, mainly because of the much lower chamber pressure. It is possible to increase the chamber pressure by using the modern technology of a high pressure fuel feed system. This would increase the performance of the main engines.

Another solution of improvement is using the Energia main engine. In this case, the Saturn 5's payload will roughly increase from 119 tons to 145 tons. In terms of cost this solution seems to be better.

Table 8.9 Engine performance - comparison between F-1 and RD-170

	F-1 (Saturn 5)	RD-170 (Energia)
Oxidizer/fuel	Oxygen-Kerosene	Oxygen-Kerosene
Mixture ratio	2.27-1	2.6-1
Initial thrust	6770 kN	7257 kN
Chamber pressure	66.8 bar	257 bar
Specific impulse	265 sec	309 sec
Area ratio	16-1	22-1
Mass	8444 kg	9833 kg
Height	5.59 m	3.66 m
Width	3.66 m	3.99 m

Additional Boosters for the First Stage

Another method of improvement is adding solid/liquid boosters for the first stage. Candidates are the Solid Rocket Boosters (SRB) of the US Shuttle and the liquid booster of the Energia. It is more efficient to use the Energia boosters, because of the higher performance of this boosters. They use liquid propellant and therefore they have a higher specific impulse. In this case structure changes would be required to attach the boosters on the first stage.

In our point of view the future use of the updated Saturn 5 seems to be an interesting possibility. Analyzing the first ideas given above it looks feasible, but needs some more detailed studies.

Big Dumb Booster - Pressure Fed with Large High Thrust Engines.

This system has been considered in the early 1960's as a method of placing large payloads into orbit. The vehicle uses tankage made in shipyards rather than in the very precise environment of a typical aerospace clean room. The reason shipyards are used is because the size of the tankage is extremely large and very thick. This is in contrast to the thin-walled tankage designs that are typical of flight systems like the Space Shuttle or Ariane. Thick tank walls are required because the tank pressure is high: (50 to 100 Bar).

8.5.4 Bulk

The bulk cargo would only be concerned with cost and resiliency. Depending on strategy a HLLV, mass driver / RAM accelerator or laser propulsion could be used. Having a transport thats launches only bulk i.e. metal, water, glass would mean on orbit production of metal structures, LH2 / LOX and solar panel covers. This would be economic if launch cost was low enough. Also it would prepare for materials delivered from a future lunar base.

RAM Accelerator

The ram accelerator is a chemically driven mass driver. Tests of speed up to 3 km/s and masses up to 20 grams have been demonstrated. The basic idea is to fill tube with premixed gases of hydrocarbons/hydrogen and oxygen. By having separators within the tube, difference in mixtures and hence also the Mach number can be obtained. By firing a free flying ram like projectile into the tube a combustion is obtained by the shock wave behind the projectile. This combustion can presently accelerates the projectile with over 30.000g. The whole system is robust and inherently scalable. It is today envisaged that a 4 km tube on the side of a mountain could accelerate 2 ton projectiles with 500 gs to 10 km/s. In the projectile a rocket engine for orbit injection would be included. The resulting payload fraction to LEO could be as high as 40%. The orbital cost for this bulk transporter is significantly -at least two orders of magnitude- lower then current launch costs.

8.6 Technology Assumptions

The characteristics of a space transportation system is directly related to the state of the available technologies. The purpose of this section is to define, for each of the basic technologies, the forecast level of performance from years 2002 to 2042. The following sections will mainly consist in an extrapolation of the currently existing technologies and studies. In fact, one should remember that 50 years ago, in 1942, chemical propulsion development was only beginning. So, some new technologies may be discovered in the far future and modify the results.

Lowering the Cost of Space Transportation

There are many alternatives to reduce the cost of space transportation. Combining some of the methods of improved propulsion and different structural concepts may make the idea of cheap space transportation a reality.

8.6.1 Metallized Propellants

Metallized propellants are gelled liquid propellants with metal particles suspended in them. By using these propellants, we can increase either the specific impulse or the propellant density, or both. Also the safety of the propulsion system is increased because the propellants are gelled liquids. Gelling the fuel reduces size of any accidental spillage and also reduces the explosion hazard if a high-velocity particle (micro meteoroid, accidentally dropped hammer, etc.) were to hit the vehicle's tankage.Past studies have shown that the density and/or the specific impulse increases of metallized propellants can increase the payload of the Space Shuttle (Palaszewski, 1991) by 14 to 35 percent. Other applications of metallized propellants can significantly increase the payload of upper stages: up to 100 percent increases are possible for planetary missions and 19 percent for LEO-GEO missions. The major technical challenges that occur with metallized propellants are the assurance of high engine efficiency with metal combustion. Particle combustion and the erosion of engine nozzles are very critical factors in using metallized fuels. Work is continuing to improve the performance of these engine with sub scale testing in the USA and Japan. The redesign of the propellant feed system to use non-Newtonian gelled fuels is also required.

8.6.2 Lightweight Upper Stages

One powerful way to reduce the cost of current space transportation is to replace existing upper stages with lighter weight systems. By reducing the mass of the upper stage, the launch vehicle does not have to lift so much into orbit. This can reduce the class of the launch vehicle one uses for a flight. Alternatively, the same launch vehicle can take more payload into orbit. This approach can dramatically reduce the size and cost of the required launch vehicle or improve the launch efficiency. The reductions in stage weight are possible with either electric propulsion, high energy chemical propellants or reduced structural masses. Each of these areas will be discussed below.

Electric Propulsion

A general discussion of electric propulsion is provided in the section 8.2.4. By using electric propulsion, the Isp of the upper stage propulsion system is increased very significantly: up to 5000 s versus the typical values of 300 s for storable chemical propulsion. An example of reducing the launch vehicle size is the use of solar electric ion propulsion for the deployment of Global Positioning Satellites (GPS). By using ion propulsion rather than the current Inertial Upper Stage (IUS), a smaller class of launch vehicle can be used. To launch the GPS/IUS combination, a Titan 4 is needed; only a Delta launcher is needed for the GPS launch with the ion propulsion upper stage. The cost difference between these two options is $250M for the Titan versus $80M for the Delta. Over the life of the GPS system, the total cost savings would be many billions of dollars.

Chemical Propulsion

High energy chemical propulsion systems can also provide high leverage for reducing costs. Examples of this are the use of high temperature materials to increase the performance of storable propellants. Because the engine does not require film cooling, the propellant is used more efficiently and will deliver higher Isp.

High performance cryogenic O_2/H_2 engines are also planned for advanced missions. These engines can deliver up to 485 s (4757.9 N-s/kg) using high chamber pressures (1000 to 1500 psi) and high expansion ratios (up to 1000).

Light weight Structures

Filament-wound tanks are used to store high pressure liquids or gases. This type of tank can reduce the mass of propellant and pressurant tankage by 10 to 50 percent. The greatest savings are only enabled at very high pressures of several thousand psi. For simpler pressure-fed propulsion systems, filament wound tankage may provide a lower weight alternative to all-metal tankage made of aluminum or titanium alloys.

8.6.3 High Energy Density Propellants

These propellants include metastable molecules and free radical atoms. A very high energy can be released from these materials. This high energy can be translated into a very high specific impulse: up to three times higher than current O2/H2 propulsion. For example, the Space Shuttle Main Engine delivers approximately 455 s (4463.6 N-s/kg). Atomic hydrogen may ultimately deliver 1500 s (14,715 N-s/kg). Studies of HEDP have shown that the mass of launch vehicles can be reduced by 50-80 percent [Palaszewski, 1990].

One such free radical propellant is atomic hydrogen [Palaszewski, 1990]. Whereas molecular hydrogen (H_2) is used in current space propulsion systems, atomic hydrogen (H) is a single atom. It can be stored in a matrix with solid molecular hydrogen. Atomic hydrogen would be used as a mono propellant. Because only one propellant is used, it may simplify the design of future launchers. Additional detailed understanding of the physics of high energy density propellants is needed. The main challenge for the vehicle designer will be to operate the propulsion system at 2 to 4 K temperatures. Also, atomic hydrogen releases its energy through recombination of the atoms. This recombination occurs very rapidly if not checked and controlled. Thermal control of the propellant is therefore extremely important. Designs for cryogenic solid particle (or two-phase flow) feed systems to take the solid matrix from the storage tank to the rocket engine will be required.

The major challenges that must be overcome are the production, storage, lifetime and utilization of these fuels. A matrix with up to 2 percent atomic hydrogen in H2 has been made. The total mass of atomic hydrogen stored is several nanograms. Very large amounts that would be needed for launch vehicles (100's of tons) have not yet been created.

8.6.4 Aerobrake/Aerocapture

Aerobraking uses the atmosphere of a planet to slow down and go into a low altitude orbit after returning from higher altitudes. This is in contrast to aerocapture which is used to brake into orbit around a planet after traveling on an interplanetary trajectory. Aerobraking can reduce the mass of chemical propellant transfer vehicles by 50 percent. This technology can reduce the transportation cost and mass for lunar and GEO payload delivery missions.

The aerobrake is a large aerodynamic structure that provides a protective thermal barrier against the heat of atmospheric entry. It can be very large in diameter: 20 to 30 m for a GEO or lunar transfer vehicle. In the selection process for the aerobrake, the total vehicle mass with and without the aerobrake must be determined. There are applications where the mass of the aerobrake may exceed the propellant required for the return from high orbit. Careful selection of the type and configuration of the brake is also required. The thermal heating during atmospheric entry may require a design that completely surrounds the vehicle.

8.6.5 Air Breathing Propulsion

An alternative to pure rocket propulsion is air breathing propulsion. With this technology, the air from the atmosphere can be used as the oxidizer in the vehicle engine. For Earth to Orbit vehicles, the total mass of propellant can be reduced very significantly if air is used in lieu of liquid oxygen. With a vehicle using oxygen and hydrocarbon fuel, the typical mixture ratio is 2.6. A mixture ratio is the ratio of the oxidizer mass to the fuel mass. This means that (2.6/3.6) or 72.2 percent of the propellant mass would no longer be carried on board the rocket. For a rocket using oxygen and hydrogen, the mixture ratio is 6.0 and total savings would be (6/7) or 85.7 percent of the propellant mass. This is a critical part of the development of the National Aerospace Plane (NASP) that is planned in the United States. Another potential application is the Two Stage to Orbit (TSTO) vehicle.

8.6.6 Slush Hydrogen

Another method of storing hydrogen fuel is called slush hydrogen. This technique allows the density of hydrogen to be marginally increased. The thermal heat capacity of the slush is also significantly higher than liquid hydrogen. This improves the ability of the fuel to cool the external surfaces of the aircraft during ascent and reentry. This is critically important for applications such as the NASP, which has a very large hydrogen tank. By using a higher density fuel, the propellant tank is considerably reduced in size. The drag reduction and the thermal advantages of slush hydrogen make it an enabling technology for the NASP. Extensive testing of slush hydrogen is underway as part of the NASP program in the USA.

8.6.7 In-Situ Propellants

In-situ propellants are fuels and oxidizers produced on other planets, moons or other bodies in the solar system. By producing propellants on extraterrestrial bodies, the transportation around the solar systems can be greatly improved and more cost effective. This is because all of the mass for propulsion does not have to be lifted from the Earth. Taking less mass along with you on your vehicle can allow a faster mission, and improve the payload delivered to the final destination.
Using in-situ propellants, the mass launched for a Mars Sample Return Mission can be reduced by more than 50 percent. For lunar transportation, the total mass lifted from the Earth can be reduced by 33 to 50 percent.

Current research is underway at a modest level in the rocket engine technology for using both O2/Al for the moon and O2/CO propellant for Mars. Additional technology is being developed at a low level to address the production of propellants on both the Moon and Mars.

8.6.8 Mass Drivers

A mass driver can electrically accelerate a payload to high speeds. The payload will experience a very high acceleration: from 100's to 1000's of gravity. It is therefore not recommended for human flight. Mass drivers are particularly applicable to launching payloads from the Moon toward a liberation point or perhaps into lunar orbit. Past studies of these devices have shown that they may reduce the cost of launching objects from the Moon for space construction projects. There has been consideration of these devices for Earth launch as well. The size of sub scale experiments with mass drivers has been relatively small: several grams to one kilogram. Very extensive scale increases would be needed to launch the payloads for Space Solar Power Program development.

8.6.9 Gun Propulsion

An alternative to rockets for Earth launch is gun propulsion. This type of system literally uses the same principle as a gun: expanding gases accelerate a projectile or payload to high speeds. As with the mass driver, high velocities and very high accelerations are produced. As with the mass driver, scale experiments have been conducted.

8.6.10 Laser Propulsion

This technology uses a laser to heat a propellant to high temperatures. The resulting gas is then expanded through a rocket nozzle to produce thrust. The performance of such an engine can be 1000 s (9810 N-s/kg) using hydrogen as a fuel. Laser propulsion can be used for Earth to orbit transportation. The payload for an Earth to Orbit (ETO) laser system will have to be fairly small: 1000 kg. This is because the laser on the ground will be limited in size: under 100 MW. Sending large payloads (> 10,000 kg) into orbit would require many 1000's of MW in laser power. Such giant lasers would be potentially be impractical in size. Small manned laser propelled vehicles as well as cargo vehicles have been studied. Very small scale experiments have been conducted.

8.6.11 Nuclear Thermal Propulsion

Nuclear thermal rockets use a nuclear reactor to heat a working fluid to very high temperatures. The fluid is expanded through a DeLaval nozzle to produce thrust. These engines have the potential to produce up to 1000s of Isp. The reactor technology for this system is under development in the United States by a joint program between the Air Force, the Department of Energy and NASA. A demonstration of the engine is possible in the next 15 years.

A primary consideration with nuclear thermal propulsion is the radiation environment surrounding the engine during and after operation. Also, after placing the reactor in space, the orbit is planned to be high enough to prevent an accidental reentry. These safety restrictions would also be very important for any type of space reactor, whether it were for space power, nuclear electric propulsion, or nuclear thermal rockets.

Several engines have during the 1960s been designed, built and tested. Nevertheless, none of those engines ever flew. For instance, the characteristics of the alpha 2 engine developed in the USA for the NERVA Program are presented below. This engine was designed on the basis of a so called "solid core engines" using high temperature carbide components such as UC and HfC. The characteristics of the ALPHA2 Engine are a thrust of 71.7 kN and a specific impulse of 860 s.

It is possible to improve the efficiency of these engines by increasing the temperature of the core. The evolution of the specific impulse of thermo-nuclear engines related to the core temperature of nuclear core technology is presented thereafter. For instance, a specific impulse of about 3000s could be reached with self contained gas core engines. The main challenge related to the development of these engines is the development of light weight materials which should be both heat resistant and radiation resistant. The current technology development performed in the field of fusion reactors for power production have shown important progresses in this domain. In consequence, this technology can be considered to be available in 50 years

Nevertheless, the main topic related to thermo-nuclear engines is the safety aspect du to the use of highly contaminant power sources. As a consequence, the use of this technology for space transportation systems operating from the Earth, or in low Earth orbit such as OTV should be considered with a lot of care. This technologys' benefits are discussed in the lunar transportation section.

8.6.12 Materials

The performance of engines or transportation system are directly related to the materials used for their manufacturing. For instance, the ΔV achieved with a given engine and mass of propellants depends on the mass of the launcher and therefore of the mechanical resistance of the materials. On the other hand, light weight heat resistant materials allows to increase the temperature of the combustion chamber or to avoid costly cooling systems and so to increase the efficiency of the engine.

Structural Materials

Fiber reinforced composite materials demonstrate higher specific mechanical resistance and stiffness than conventional metallic alloys. Some improvement of their characteristics and knowledge will be achieved in the following use:

- Organic matrix composites: These materials have been developed for many years and are already use for manufacturing of space transportation systems. Their performances could be increase in the following ways:
- increasing of mechanical resistance by improvement of the fiber characteristics as, for instance, very high resistance carbon or aramid fibers,
- increasing their mechanical behavior at high temperature by using polyamide organic matrix,

For instance, the use of filament wounding Kevlar reinforced organic matrix composites for the manufacturing of case for solid propellant rocket engines for military missiles have allow a important reduction of the mass of the engine. The adaptation of this technology for tanks of liquid rocket engines (i.e. Shuttle one) or to the case of solid rocket boosters (i.e. Shuttle, Ariane V) would increase the mass of payload for those transportation systems. This would required the development of large capacity manufacturing and non destructive testing facilities
- Metal matrix composites: those materials have been developed for many years. But at the opposite of organic matrix composites, almost no application of those composites has been developed for space transportation systems. Compared to monolithic metallic alloys MMC present, depending on the matrix and reinforcement present:

- increased mechanical resistance at room temperature,
- higher stiffness,
- improved mechanical behavior at high temperature,

- low thermal expansion

Nevertheless, the maximum temperature of use of those materials is only increased of 100 or 200 degrees centigrade compared to the monolithic alloys. Otherwise, Intermetallic Compounds (i.e. Titanium aluminide) are currently being developed. MMC made with these alloys present, compared to titanium alloys matrix composites:

- improve mechanical behavior at high temperature,
- increased oxidation resistance
- lower density
- Glass matrix composites: Those materials are made with glass matrix (amourpheous) or glass ceramic (partially crystallized). Those materials present low density and good mechanical behavior at high temperature (till 870 K for glass matrix and 1270 for glass ceramic matrix)

Heat Resistant Composite Materials

Ceramic Matrix and carbon carbon composites have been developed during the last ten years. Those materials present a very good behavior at high temperature and a non-brittle mechanical behavior, compared to monolithic ceramic or polycrystalline graphite. Moreover, Ceramic Matrix composites such as C/SiC and SiC/SiC present, due to the characteristics of the matrix an improved oxidation resistance. Those materials present a low density (2 to 2.5) when that of super alloys (Nickel based) is about 8.

Those materials offer new solution for the design of rocket engines to increase their efficiency. For instance, it is possible:

- to increase the temperature of the combustion chamber, for example for bi-propellant liquid rocket engines,
- to reduce or suppress the cooling system for combustion chambers of nozzle extension.

For example, uncooled nozzle extension made of C/SiC as been tested on an HM7 cryogenic engine (3rd stage of Ariane 4 launcher)). Compared to the dump-cooled metallic one, this nozzle extension lead to an increase of about 60 kg for the GEO payload of this launcher.

8.6.13 Mission Applications

The areas where these technologies can be applied are listed as a function of the mission application. Because the Space Solar Power Program is a very long term program, the precise technologies that would be used are not clear. Within the next 10 years, current launch systems and upper stages will be used for any demonstrations. However, technology development is difficult to project beyond the 10 year time frame. A list of potential technologies is therefore provided to summarize the options that are available and currently under consideration. As the Space Solar Power Program becomes more defined and the funding level is determined, a more focused vision of the transportation technologies can be developed.

Earth to Orbit:

Metallized Propellants
High Energy Density Propellants
Slush Hydrogen
Gun Propulsion
Mass Driver
Laser Propulsion

Orbital Transfer

Lightweight Upper Stage
Solar and Nuclear Electric Propulsion
Nuclear Thermal Propulsion

Lunar

In Situ Propellants

Mass Drivers

8.7 Lunar Transportation

The utilization of lunar raw materials to build up a large scale solar power satellite seems to be a promising way to lower the initial LEO mass. Therefore this section gives a brief discussion of lunar transportation systems that would support this activity. Reducing the initial LEO mass that has to be transported from Earth into LEO offers the opportunity to lower the overall costs of building up a Space Solar Power Program. According to the lower gravity of the Moon compared to Earth, only about $^1/_{20}$ of the energy is required to reach space from Moon than from Earth. In this case we assume a large lunar industrial infrastructure which is capable of producing Space Solar Power Program construction parts economically. According to different studies, the range of lunar materials of a GEO Space Solar Power Program is between 90% and 98%.

This section presents a comparison of cislunar transportation systems with different propulsion systems in order to find out the most promising strategy for transporting construction materials from the lunar surface to the Space Solar Power Program construction site. The following types of propulsion are addressed:

1. Conventional chemical LO_2/LH_2-propulsion
2. Electric propulsion
3. Nuclear propulsion
4. Mass driver

A laser propulsion system is not considered because operation requires a high power source in the order of at least 100 MW transmitted by a laser beam. Because large power sources may not be available during the Space Solar Power System construction phase we did not consider laser propulsion.

8.7.1 Conventional Chemical LO_2/LH_2 Propulsion

The first LOX/H2 propulsion system was used in the Centaur upper stage whose development started in 1958 with the first R&D flight in 1962. Of the chemical propulsion systems, LOX/H2 -propulsion represents the propulsion system with the highest achievable specific impulse (Isp). At present an Isp of about 455 seconds is achieved by the SSME (Space Shuttle Main Engine). Currently the Advanced Space Engine (ASE) using an expander cycle is under development in the USA with the aim of increasing the Isp to about 480 seconds. A higher Isp on the order of 485 seconds is expected for the future due to higher combustion chamber pressure (more than 200 bar) and higher expansion ratios on the order of 1000 especially in vacuum operation. At present chemical propulsion provides the highest ratio of thrust to weight with a ratio up to 100 Figure 8.18. This high ratio enables very short inter orbital transfer times between LEO and LLO on the order of 3 days.

With regard to the high thrust and throttling demand of lunar descent/ascent, LOX/H2-engines provide a high performance to fulfill the main mission requirements. Furthermore due to long-standing experience, LOX/H2-engines offer a relatively high reliability which is very important especially for manned missions. However, a disadvantage is the handling and storage of the cryogenic propellant for several days - especially of hydrogen - which results in high boil-off rates, compared to storable propellant such as hydrazine. Additional insulation material should solve this problem and keep the boil-off rate below 0.1% per day.

To obtain maximum performance the LO_2/LH_2-engine is operated at a weight mixture ratio (LO_2/LH_2) of about six which means that about 85% of the entire propellant is oxygen and about 15% is hydrogen. Due to the fact that oxygen is richly abundant on the Moon (the lunar regolith consist of about 45% of oxygen by weight) a lunar oxygen production facility could substitute the Earth-derived oxygen and provide lunar oxygen for the space vehicles. The initial LEO mass can be decreased in the order of up to 50% so that a cost benefit is expected due to a LEO mass saving.[Michael REICHERT, 1992],

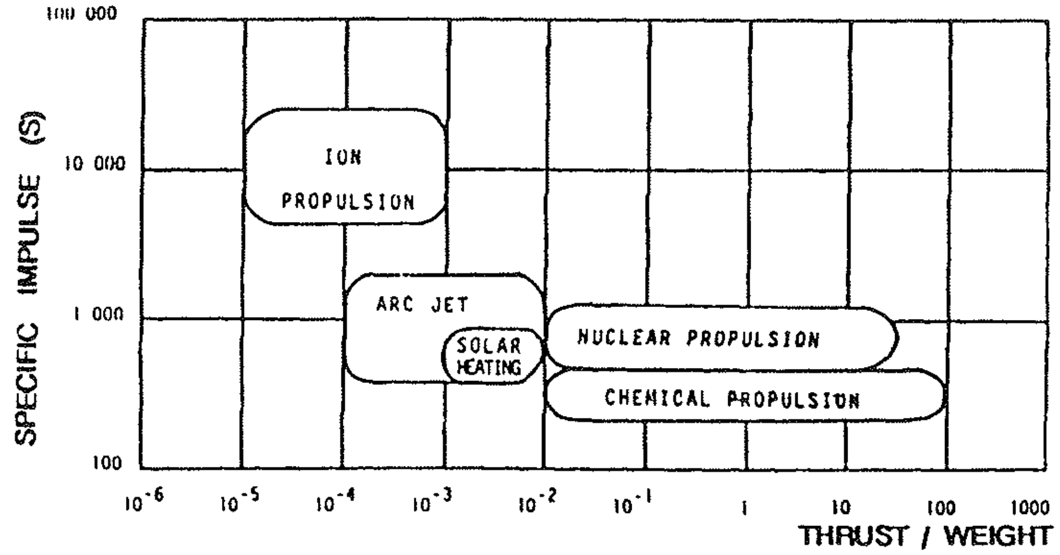


Figure 8.18 Ratio of Thrust to Weight for Different Propulsion Systems

Due to the fact that almost all technology is available a cislunar transportation system could be realized within less than a decade.

Lunar Bus (LB)

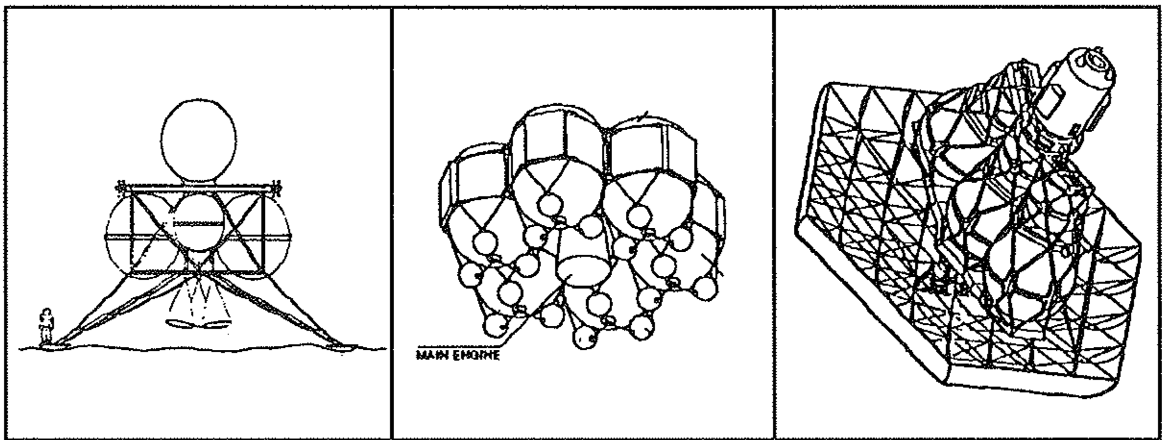
Payload transport between the lunar surface and LLO is carried out by a Lunar Bus Figure 8.19 which is designed to accommodate the special requirements of lunar ascent/descent. The ΔV-requirement for lunar ascent/descent is assumed to be 1900 m/s respectively 2000 m/s (higher reserves for landing maneuver) These Δv-requirements include losses due to gravity and thrust vector control and reserve propellant. The flight time for ascent or descent is less than one hour.

Orbital Transfer Vehicle (OTV)

Payload transfer between the LLO and GEO is carried out by an Orbital Transport Vehicle. The Δv-requirement between LLO and GEO for one interorbital flight amounts to 2000 m/s and the transfer time is about 3-4 days.

Aeroassisted Orbital Transfer Vehicle (AOTV)

Payload transfer between LLO and LEO is carried out by an AOTV. After leaving LLO the AOTV reaches Earth where it dives into the upper layer of the Earth's atmosphere to reduce its velocity using an aerobrake maneuver from about 11 km/s to the velocity of LEO to about 8 km/s and then insert into the space station orbit. The Δv-requirement from LLO to LEO is about 1000 m/s (800 m/s are required for lunar escape acceleration and midcourse maneuver and 200 m/s are required for Earth's atmosphere entry and leaving to reach the 400 km space station orbit in LEO). From LEO to LLO the AOTV accelerates to Earth's escape velocity ($\Delta V \approx 3200$ m/s) and when reaching the moon the AOTV decelerates in order to go into LLO ($\Delta V \approx 800$ m/s including midcourse correction). The flight time for each interorbital transfer is about 3 days. The technology for an aerobrake maneuver of an AOTV in the Earth's atmosphere will need to be developed.



LB (Lunar Bus) for lunar descent/ascent	OTV for interorbital transfer between LLO and GEO	AOTV for interorbital transfer between LLO and LEO

Figure 8.19 Space Vehicles of a Lunar Transportation System

8.7.2 Electric Propulsion

NASA has studied the manned exploration of the lunar surface and its use for a large scale exploration [Bryan PALASZEWSKI, 1988]. We attempt to use these studies as a basis for describing the efficiency of electric propulsion for the round trip transportation to the Moon.

Electric propulsion is characterized by its high Isp (1000 s to 10000 s) and low thrust. The ratio of thrust to weight of this type of propulsion is less than 0.01 see Figure 8.20. Of course, the performance strongly depends on the technology and also the power of the generator. The low thrusts do not allow liftoff capability from the Earth or from the Moon. Thus this type of propulsion will be used for an OTV from LEO to LLO. In the transfer, the OTV departs from LEO at an 500 km altitude orbit; the LLO is at an 100 km altitude 0.0 degree inclination orbit. Transfer to the Moon will require a combination of chemical propulsion and electric propulsion. In part, the step from Earth to LEO and from LLO to the Moon surface require chemical propulsion.

A major difference between chemical and electric propulsion concerns the ΔV required for orbit transfers. High thrust orbit transfers from LEO to LLO require ΔV of 4000 m/s, for example. This includes LEO departure, trajectory correction and LLO insertion. In the low thrust case, the ΔV is 8000 m/s. Thus, electric propulsion systems need twice as much ΔV than chemical propulsion for a lunar mission. This is mainly due to the gravity losses and the attitude control losses.

Among electric propulsion two main categories exist: the solar electric propulsion and the nuclear electric propulsion. In order to illustrate both of them we address the following examples:

-nuclear electric OTV:	Xe Ion OTV(1MW)
	H2 Arcjet OTV (1MW)
	Magneto Plasma Dynamic OTV (1MW)
-solar electric OTV:	Xe Ion OTV (300 kW)

Figure 8.20 shows a nuclear electric lunar transfer vehicle design. The vehicle uses a dynamic conversion nuclear reactor to power the electric propulsion thrusters. The payload and the reactor are separated by a boom. This boom minimizes the radiation exposure of the payload to the reactor. The main propulsion system is near the center of mass of the transfer vehicle. This design is a side thrust configuration.

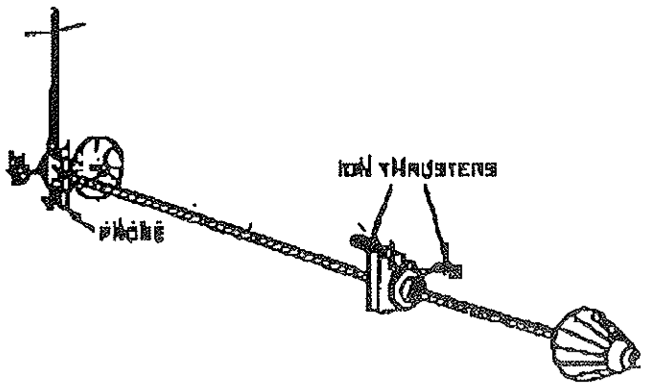


Figure 8.20 Nuclear Electric LTV

The specific impulse (Isp) of electric propulsion systems are given in Table 8.10.

Table 8.10 Propulsion System Performance for Future Applications

Propulsion Technology	Isp (s)	Total System Efficiency (%)
O2/H2	480	-
Arcjet	1500	49
Ion	2000 -10000	60-85
MPD	2000 -10000	50

Figure 8.21 shows the initial LEO mass for different propulsion technologies. The mission profile used for this study assumes that a payload of 35000 kg is delivered from LEO to LLO and no payload is returned. This Figure also gives the trip time for that mission.

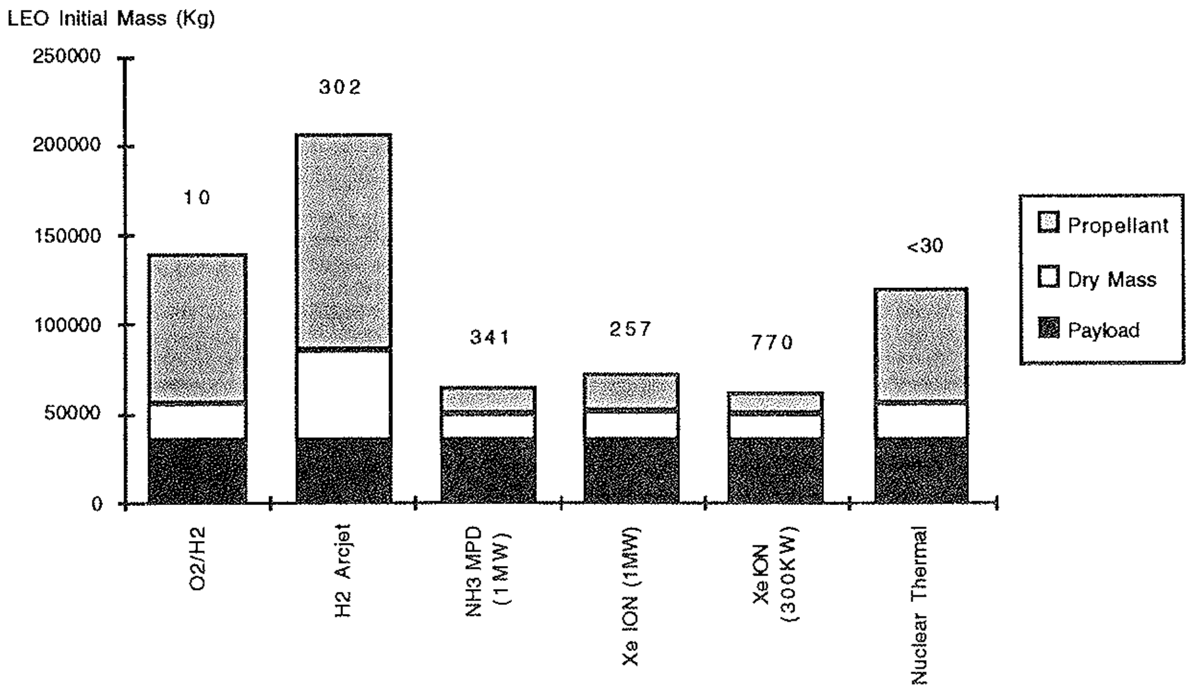


Figure 8.21 Propulsion Technology Comparison and Trip Time Round Trip (LEO-LLO) - Payload Mass : 35 Tones

The H2 arcjet propulsion system is the least efficient among nuclear electric systems. Due to the level of Isp that it delivers this system is also more massive than a chemical one. Because the ΔV a required for lunar round trip mission is so high, the propellant mass for the arcjet system is very large. A propulsion system with an Isp of 1500 s would not reduce the propellant mass sufficiently over the H_2/O_2 system.

The most efficient systems with respect to the propellant and dry mass are the Ion and MPD systems. They can save up to 50% of the total mass over the chemical propulsion system. Moreover it is important to stress that the ratio [payload / dry mass and propellant] is above 1. These masses strongly depend on the values of Isp and the specific mass of the reactor. The specific mass (kg/kW) reflects the performance of the reactor. The lower the specific mass, the higher the performance. The assumptions made in Figure 8.21 for the arcjet, the MPD and Ion system (1 MW) are:

P= 1MW Isp= 5000 s Specific Mass = 10 kg/kW

The duration of the transfer will be relatively long (200 to 400 days) because the level of thrust capability with MPD and Ion propulsion systems is low (10 to 100 N). Thus, these missions will be unmanned. The conventional chemical propulsion will be used for the transportation of the crews. These missions could be possible around the years 2020 and beyond.

Solar electric OTV requires big solar arrays. Generated electricity can be used to ionize Xe propellant. A 300 KW solar electric power system is considered. Here Figure 8.22 shows the decreased mass resulting from this system over the chemical one. This overall gain is around 50%. The Isp is of the same order of magnitude than for nuclear electric propulsion. Meanwhile the low thrust in comparison with a 1 MW power system will raise the travel duration dramatically (770 days). Figure 8.22 gives an idea of what could be achieved within the next ten to fifteen years.

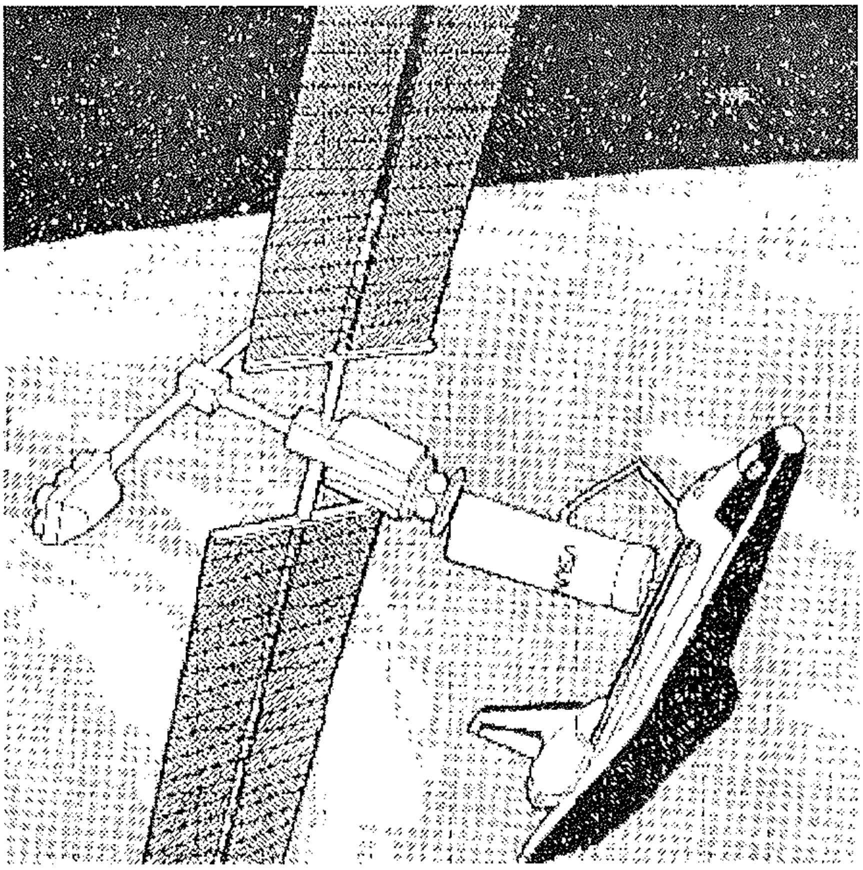


Figure 8.22 An Example of 300 KW Solar Electric OTV

8.7.3 Nuclear Thermal Propulsion

Nuclear thermal propulsion is quite similar to chemical propulsion and it will be used for mission profiles from LEO to LLO and back. A nuclear reactor heats an H2 propellant that provides the thrust through the nozzle. Achievable Isp is about 900 s and the reduction in LEO mass is about 20%

Figure 8.21. At present the ratio of thrust to weight can reach up to 30, thus the time for inter-orbital transfers can be in the order of less than 30 days depending on payload size and thrust. The first investigations concerning thermal nuclear propulsion which were carried out in the USA was the NERVA program which was canceled in 1973. This kind of propulsion has stimulated the interest from many Nations in the last twenty years. It might be the next non chemical propulsion system to be developed and to be used for large scale applications within the next thirty years.

8.7.4 Mass Driver

The purpose of the mass drivers is to accelerate payloads of materials to a high velocity by the transformation of electrical energy (using an electromagnetic field) to mechanical energy of motion. In our application the mass driver would be placed on the lunar surface, and it would accelerate the payloads to the escape velocity of the Moon (2340 m/s). These payloads would be collected at a point in space (Lagrangian point of the system Earth-Moon) to serve as the material depot for space manufacturing or to be sent to the Earth. Several Mass Drivers have been built until today. Mass Driver III has achieved over 1800 gravity acceleration. The length of the lunar machine to obtain the Moon escape velocity requires 160 meters. The mass that had been carried was about 500 grams. Many studies are conducted today on advanced Mass Drivers.

Summary

As shown in Figure 8.21, electric propulsion can reduce the initial LEO mass which is required to transport materials for building up Space Solar Power Program. This would have enhanced benefit to Space Solar Power Program (with the exception of H_2-arc jet propulsion) as compared to conventional chemical or nuclear thermal propulsion. The reason for this is the higher specific impulse which can be achieved by electrical propulsion. However electric propulsion provide a low thrust which will cause relatively long inter-orbital transfer times. So electric propulsion seems only appropriate for cargo missions. For manned missions high thrust chemical propulsions seems to be the most promising solution because of short inter-orbital trip times of about 3 days. The use of lunar oxygen for chemical space vehicles can reduce their initial LEO mass requirement on the order of up to 50 percent. If the safety and reliability of thermal nuclear propulsion can be enhanced, medium thrust nuclear propulsion could compete with chemical propulsion.

8.8 Scheduling

The scheduling of space transportation activities is based on the assumption that the first three demonstrations (Demo 0, 1 and 2) will only require existing or presently under development launchers. The only milestone for these three demonstrations will therefore be the choice of a specific vehicle.

Demo 3 and future demos, on the contrary, will need tailored developments in the field of Space Transportation (personnel, priority cargo, bulk, etc.). Therefore, a set of technology studies and the design of launch vehicles will have to take place before the launch and assembly of the demonstration spacecrafts.

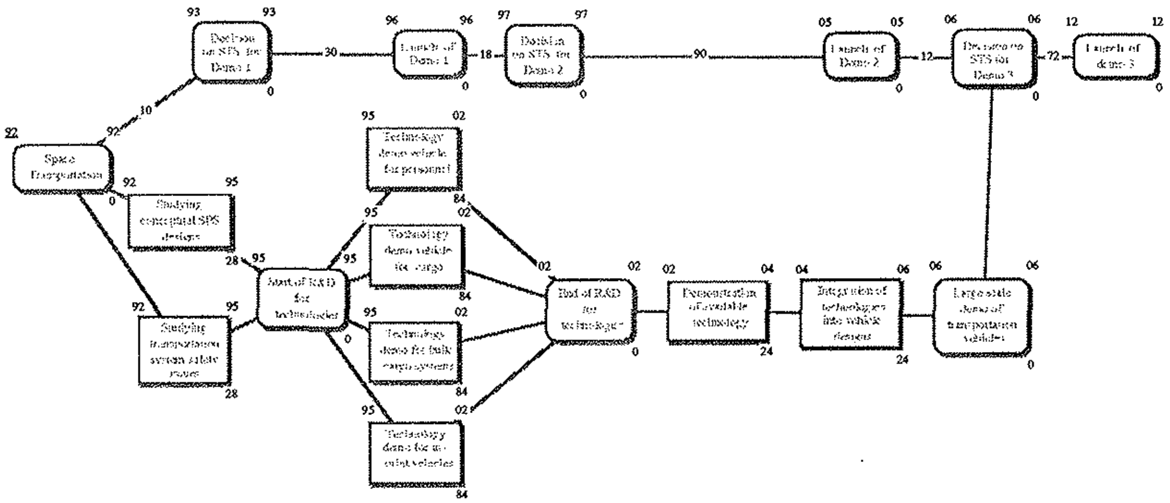


Figure 8.23 Research and Development Tasks for Space Transportation Technologies

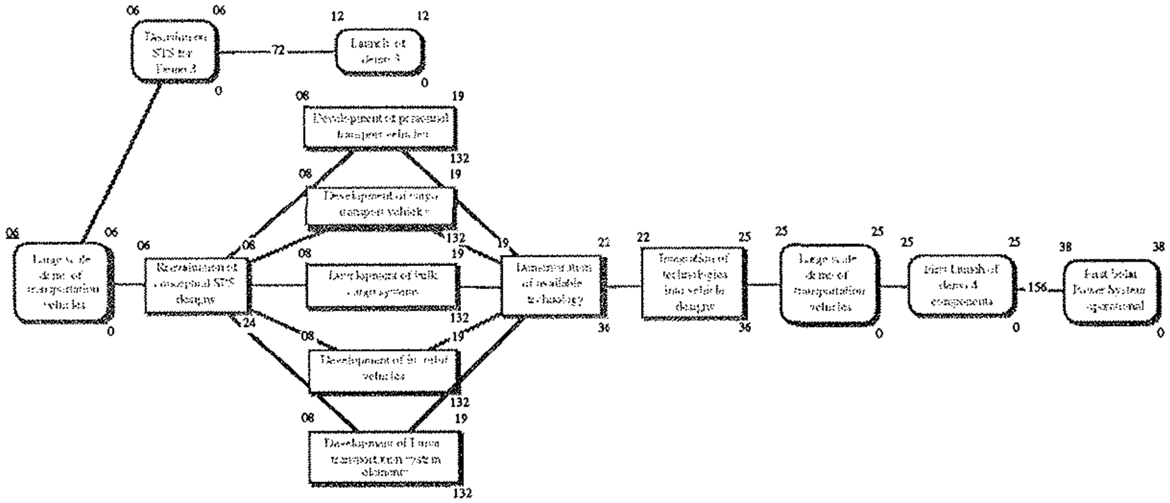


Figure 8.24 Advanced Space Transportation Development Tasks

8.9 Conclusions

Only by reducing the costs of space transportation can solar power from space become feasible. With many past studies of solar power satellites, the transportation system cost has been 25 to 40 percent of the total program cost. Even with current space projects, the cost of space launch services is terribly high. Without active measures to bring down the costs of space access, the viability of any large space program is questionable. It should also be clear that these "costs" include not only dollar value of the booster, but also the transportation system reliability, accessibility, launch environment and the vehicle resiliency. All of these factors can increase cost and defeat our purposes in space. Only through the application of innovative technologies and streamlined space launch operations will humankind attain the height of perfection and low "cost" in space flight.

There are many options for launching payload for a space power system. In the near term, there are numerous capabilities to deliver large and small payloads to LEO and beyond. Over the next ten years, there will be little change in the capacity to move satellites there are few developments in the planning stages other than incremental vehicle payload improvements. Beyond the ten year horizon, new launch vehicle designs, propulsion and materials technologies have the potential to make exciting leaps in payload delivery efficiency. Vehicles using Two Stage to Orbit and Single Stage to Orbit have the ability to reduce operational costs of payload launches. Simplifying these operations is a major stumbling block to making our access to orbit affordable.

Many technologies are available for space transportation systems of the future. The final selection of which technologies are used is very dependent on the time frame of the solar power system development. Based upon this report's development plan, the first launch vehicle developments for any large scale power satellites would be in the 2005-2010. The first satellite would be launched in 2035-2040. Because of long time until the first vehicle flight, it would be unwise to select a specific technology or set of technologies for the transportation system. Also, the specific architecture of the space solar power system will determine the relative importance of the transportation technologies. If a large scale power system is required, the need for lunar resources may become crucial. On the other hand, a smaller satellite constell ation would most likely not use extraterrestrial-based resources. The propulsion technologies that would be used would be advances reflecting the potential of Single Stage to Orbit and other improvements in propulsion technology to increase the energy density of propellants (such as metallized propellants and high energy density propellants). Light weight or high temperature materials will also play a vital role in reducing the cost of space operations and space access. Only time will tell how ambitious and exciting our global technological future will be in space transportation.

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9 Space Manufacturing, Construction, & Operations

9.1 A Matter of Scale

The task of this chapter is to examine how to better build a solar power satellite. This is by no means a trivial undertaking and probably represents the most daunting technical obstacle to the large-scale implementation of space solar power. According to the NASA/DOE reference system, a 5 GW SPS will weigh about 100,000 tonnes and will be equipped with a 10.7 km by 5.3 km array and a 1 km diameter transmitting antenna. To put the immensity of this structure in perspective, Space Station Freedom, which is currently planned to be the largest object put into orbit in the next ten years, will weigh just 305 tonnes, and its largest dimension is approximately 90 m. Thus the massive SPS contemplated by NASA/DOE will be at least a factor of 5000 times Freedom's volume and 300 times its mass.

One might argue that it will not be desirable to build solar power satellites in geostationary orbit (GEO) on the tremendous scale of gigawatts—a multitude of smaller satellites in low Earth orbit (LEO) may be more suitable. However, there is a limit to how small one can productively make these satellites: they need to be at least on the order of several hundred megawatts if they are to produce significant amounts of energy on Earth. Therefore, only a factor of 10 or so can be saved in terms of size and mass. This savings still yields a satellite that is 500 times the volume of Freedom and 30 times its mass..

One might also argue that advances in collection efficiency, either directly through solar cell improvements and solar dynamic generators, or indirectly through solar concentrators and reflectors, could reduce the collecting area required. However, the efficiency is already assumed to be almost 20%, and since virtually no solar conversion systems claim to have theoretical (let alone practical) efficiencies greater than 60%, this reduction can only realistically save a size and mass factor of about three.

Consequently, a best case scenario results in a satellite producing hundreds of megawatts with 60% solar conversion efficiency that is still well over 100 times the size and 10 times the mass of Space Station Freedom, which, incidentally, will take NASA several years to assemble. Therefore what one has, inevitably, is a very large space structure.

The theoretical engineering considerations of large-scale structures are detailed in Section 9.2, which deals with the modelling and control theory of large space structures. Specifically, the topics of multibody dynamics, modal representation, linear/nonlinear control, and robust control are discussed. The actual construction of a large space structure is the topic of Section 9.3. The advantages and disadvantages of both erectable and deployable structures are outlined. Also addressed in detail is the assembly of Space Station Freedom, extravehicular activity (EVA) experiments involving erectable structures, packing efficiency, trusses, inflatables, and adaptive structures.

The general conclusion of these sections is that the assembly and maintenance of solar power satellites, whatever their exact final size, will be of sufficient complexity to absolutely require, at least in our view, an assembly and maintenance oriented demonstration before any such system of satellites can be installed. The primary purpose of this last prototype would be to show not only the ability to build the solar power satellite, but also the related but yet quite different ability to maintain it, for a period of about 5 years (based upon a desired 30-year satellite lifetime).

In the near term, the most important demonstrations would consist of small-scale assembly experiments (for example NASA's shuttle bay EASE and ACCESS assembly experiments). Also, small-scale demonstrations of inflatables, such as the inflatable rectenna proposed for the Arecibo 10 Million US$ design example (see Sec. 10.1) could be of great importance. Then comes a possible watershed event for space solar power: the construction of Space Station Freedom. The successful manned assembly and manned and telerobotic maintenance of Freedom would represent a major confidence-inspiring milestone for large space structures in general and space solar power in particular. Failure in either of these two critical areas for Freedom could set back development of space solar power for over a decade. Once the first two space stations of the 21st

Century—Freedom and Mir 2—are operational, more detailed and complex manned vs. robotic assembly productivity tests, of the sort that a solar power satellite might entail, need to be performed on these stations in order to determine which of the two construction options is best. From this standpoint, the 10 MW space to Earth SPS 2000 demonstration, proposed by ISAS and potentially operational in ten to twenty years, is also extremely interesting as a demonstration of robotic construction, since it relies upon the teleoperated assembly of 10 separate Ariane 5-launched components. Such a fully automated assembly would be unprecedented (even Space Station Freedom is not planning to use robots for assembly). Within the context of a program for space solar power, the automated assembly in and of itself would probably justify much of the cost of the demonstration, regardless of the amount of power actually delivered to the ground.

Problems on Earth

Returning to the figure of 100,000 tonnes for each 5 GW SPS, note that even if it were decided that smaller solar power satellites should be built, the overall tonnage/GW would remain approximately constant, since more satellites would be needed to produce the same amount of energy. To put this truly tremendous amount of mass into perhaps even better perspective, since the launch of Sputnik 1 in 1957, only 30,000 tonnes of payload have been placed in orbit. This means that the emplacement into orbit of one SPS will require more than a tenfold increase of the entire world's launch capabilities, from 1000 tonnes to at least 20,000 tonnes per year. [Hannigan, 1991] Furthermore, if several SPS are to be constructed simultaneously, then the global launch rate will have to increase by another order of magnitude.

There are two problems with this increased terrestrial launch rate scenario. First of all, at current space transportation cost levels of 10,000 US$ per kg or 10 Million US$ per tonne, a 100,000 tonne SPS would cost 1 Trillion US$ to launch into orbit, a figure almost equal to the annual US Gross Domestic Product. So not only does the actual launch rate have to increase by two orders of magnitude, but the effective launch costs have to decrease by at least two orders of magnitude in order for any space solar power system like the SPS reference system to be economically feasible. As mentioned in the previous chapter on Space Transportation, only a factor of ten decrease in launch costs can be realistically expected in the foreseeable future (see Sec. 8.5.1)—and even that much may be too optimistic. Secondly, there is the matter of the possible deleterious environmental effects of a hundredfold increase in the terrestrial launch rate on the upper atmosphere. This problem is discussed in more detail in Sec. 6.2.2. Because of these assorted technical, financial, and environmental potential "showstoppers" to the building of large SPS with terrestrial materials, the use of nonterrestrial resources has been seriously considered.

The Lunar Solution

Contemporaneous to the NASA/DOE studies, Gerard K. O'Neill, to whom this report is dedicated, first proposed the use of lunar materials for the construction of Solar Power Satellites [O'Neill, 1978]. He argued that the raw materials needed for construction of these satellites could be delivered to GEO from the lunar surface at one-twentieth the transport cost of their delivery from the Earth. This argument is based on both the Moon's substantially lower gravity well and its lack of atmosphere, which allows the use of electromagnetic launchers called mass drivers, whose viability O'Neill also helped demonstrate. The general idea is that these mass drivers, which would only have to be approximately 160 m long in order to impart lunar escape velocity, would propel these raw materials to a mass catcher located in a halo orbit at the Lagrangian point L_2 [Farquhar, 1971], some 60,000 km behind the Moon, from whence the raw materials could be cheaply delivered to GEO.

The NASA/DOE study itself was constrained to consider only the terrestrial resource option. However, NASA did commission two studies on lunar resource utilization for Solar Power Satellites from General Dynamics [Bock et al., 1979] and MIT [Miller and Smith, 1979]. More recently, the Space Studies Institute funded two studies by Space Research Associates [Kelso et al., 1985 and Tillotson,1989] to determine what mass fraction of solar power satellites could be built from nonterrestrial materials. All four studies determined that at least 90% of solar power satellites could be built from nonterrestrial materials at great reduction to overall system cost. The 1985 SRA study even concluded that as much as 99% of solar power satellites could be built from lunar resources. Resources found on the Moon that are of potential use for solar power satellite fabrication include silicon, aluminum, glass, and iron.. Also, oxygen can be used as a propellant throughout the space solar power cislunar infrastructure. However, some of these resources are more difficult to extract and utilize than others, and this and other nonterrestrial resource issues are the subject of section 9.4.

In general, terrestrial knowledge about nonterrestrial resources is quite poor. For space solar power, we need to know what materials are potentially available in order to optimize the final solar power satellite system design, and the earlier we have this information the better. For this reason it is vital to the success of space solar power that the entire Moon be spectroscopically mapped in detail from lunar orbit. Additionally, near Earth asteroid missions such as the US Department of Defense's Clementine, scheduled for launch in 1994, and NASA's proposed NEAR, could also be of great import to space solar power. Lastly, Shuttle External Tanks or Energia Cores represent additional "nonterrestrial" resources that may be utilized by space solar power. It is important to establish orbital control of such structures in order to ascertain the viability of using these tanks for manufacture of solar power satellites.

The various nonterrestrial resource studies referred to above tend to equate reduced terrestrial mass content of solar power satellites with economic desirability. However, Woodcock notes that these studies overlooked the problem of manufacturing complicated space hardware from raw nonterrestrial materials [Woodcock, 1989]. Lunar resource manufacturing, as well as space manufacturing in general, is the subject of section 9.5. Before lunar resources can be incorporated into solar power satellite designs, the ability to handle and process these materials in a microgravity environment should be demonstrated. Therefore, we recommend the testing of physical processing methods of lunar simulants on either Space Station Freedom or some other microgravity laboratory. Also, since chemical processing of lunar raw materials might be more efficiently done on the lunar surface than in orbit, it is suggested that in situ chemical processing tests on the lunar surface be performed by penetrators, rovers, and perhaps even automated bases.

Base Power

The presence of a fully operational lunar base capable of delivering raw materials to Earth orbit could greatly reduce the cost of building solar power satellites. In his 1989 paper, Woodcock attempts to answer what is essentially the fundamental question with regard to the use of lunar resources for solar power satellites: at what level of operation will the energy/cost savings provided by the use of lunar resources outweigh the initial capital investment needed to establish both the cislunar infrastructure and the manufacturing capabilities? Because this manufacturing capability does not yet exist, the latter half of this question is very difficult to quantify. However, by employing a detailed parametric analysis that takes into account the cost of money, Woodcock is able to conclude the following [Woodcock, 1989]:

- Generally, the analysis indicates that lunar resources are economically beneficial
- Early attempts at "bootstrapping"—another idea popularized by O'Neill—by using lunar oxygen to increase the base's self-sufficiency are likely to pay off
- The analysis indicates that lunar resources to be used in the assembly of solar power satellites have a "strong positive economic return," but this conclusion is uncertain due to the lack of reliable orbital manufacturing costs
- The "most critical technology affecting positive economic return" is "lunar surface electrical power supply." Photovoltaic systems are likely to suffice for early lunar oxygen production, but "support of Solar Power Satellite materials production will require nuclear power or a power beaming system."

Woodcock's analysis provides the justification for the synthesis of two of O'Neill's ideas: "bootstrapping" and lunar-derived solar power satellites. Space solar power systems can benefit from "bootstrapping" by initially providing beamed power to a lunar base, which in turn could provide the system with the materials needed to build more solar power satellites. This power could be implemented in a staged way, as energy might first be provided on the order of kilowatts for a lunar rover (see Appendix B). Then, for an automated base, power can be beamed on the scale of hundreds of kilowatts, and ultimately megawatts can be provided for the manned base. Considering the dearth of near-term and possibly even mid-term Earth-based markets, and the terrestrial problem of atmospheric attenuation which would not be a problem on the atmosphere-free Moon, beaming power to the lunar surface may be the best way to convincingly demonstrate space solar power while simultaneously enhancing its long-term economic viability.

9.2 Structures

9.2.1 Modeling

In the early days of space exploration, Spacecraft were relatively small, compact, and mechanically simple. They were modeled as rigid bodies for purposes of motion simulation, stability determination, and active control design. Even then this approximation, introduced by neglecting flexibility, was sometimes unwarranted as demonstrated by the instability of the Explorer I Spacecraft in 1958. The abnormal behavior of this satellite was attributed to energy dissipation induced by vibration of the long wire turnstile antennas, which protruded from the cylindrical housing of the satellite.

These and subsequent experiences led to a vigorous program of research in multibody systems with flexible components. The approximate analytic and numerical techniques developed in the course of this research proved to be quite successful in designing Spacecraft with modest size and flexibility. However, large flexible structures required for solar power generation in space present new challenges to accurately model their dynamics and develop control procedures. In general, these structures are characterized by interconnected flexible bodies having small structural damping and low, closely spaced frequency spectra. The tasks of controlling the rigid body rotations (librations) for pointing accuracy and stabilizing the flexible structure vibrations pose dynamical and control design problems, never encountered before. This is even more the case for the unprecedented size of space solar power program structures.

A question arises: why not conduct ground based experiments before deploying a structure or its subassemblies in space? Unfortunately, ground-based experiments have their limitations as accurate representation of the gravitational, magnetic, solar radiation, free molecular and other fields has proven to be elusive. Thus, refined mathematical models and comprehensive control simulation techniques will be necessary to accurately and reliably predict complex dynamical interactions in large space structures.

Moreover, as Spacecraft become more complex and architecturally metamorphical, development of precise dynamic models and derivation of the corresponding equations of motion for transient and evolutionary stages become overwhelming. Hence, considerable attention has been directed towards development of computer algorithms to automate the dynamic simulation process for complex systems. In an effort to make these programs applicable to a large class of systems, the number of structural members constituting a system is considered a variable, i.e. left arbitrary. The phrase "multi-body computer program" has been coined to denote applicability of the code to a system with an arbitrary topology.

Multibody Dynamics

In the field of Spacecraft dynamics, the first paper describing a general multibody dynamics formulation was published by Hooker and Margulies in 1965. This work was based on Newton-Euler equations and is applicable to a point connected set of rigid bodies in a topological tree, where the constraint torques are obtained via Lagrange's multipliers. At about the same time, Roberson and Wittenburg treated a system with n-rigid bodies independently and derived the dynamic equations in the matrix form.

Ever since, a number of multibody formalisms have been reported in the literature. Ness and Farrenkopf, and Ho and Gluck extended the above models to the flexible n-body system. Ness and Farrenkopf chose the unified approach to deal with the nonlinear equations for the total motion of the system, while Ho and Gluck opted for the perturbation approach to deal with the flexibility dynamics. Kane and Levinson employed D'Alembert's principle and the concept of angular momentum for derivation of the equations of motion for a flexible tree type topological system. Vu-Quoc and Simo have proposed multibody formulations for both open chain and closed-loop structures. Modi and Lips have presented a general Lagrangian formulation for the librational dynamics of cluster type Spacecraft with an arbitrary number of deploying flexible appendages. Modi and Ibrahim extended the above model to include shift in the center of mass, changing central rigid body inertia and offset of the appendage attachment point. Modi and Ng further extended the multibody formulation to include an all flexible two-tier tree type configuration, incorporating thermal deformations and appendage deployment maneuvers.

Other multibody formalisms documented in the literature include the studies by Kurdila (Maggie's approach), Keat (velocity transform method), Ho (direct path technique) and Meirovitch (perturbation approach). Jerkovsky presents an excellent overview of the relative merits and disadvantages of selected momentum (Newton-Euler) and velocity (Lagrange) formulations mentioned above.

The earlier multibody derivations were based on Newton-Euler approaches. The methods of Newton and Euler are generally recognized as useful in understanding behavior of relatively simple systems, such as particles in space or gyroscopes. However, it is believed that Lagrange gave us superior procedures for deriving equations of motion for complex mechanical systems: it yields the governing equations of motion whose structure is independent of the system geometry. Also, the equations are readily amenable to stability study and well suited for control design. Finally, if equations are to be derived by symbolic processing, the primary criterion for selecting a derivation procedure would be amenability to automation, which encourages reduced dependence on engineering judgment.

The pioneering research in multibody dynamics was driven largely by the allure of the equations themselves, and not by the need of computer programs to simulate Spacecraft dynamics being designed at the time. Nowadays, the situation is quite different; the research efforts are governed by the need to develop tools for design and testing of Spacecraft and other systems now committed for development. Several general purpose computer codes aimed at studying dynamics of multibody systems have been commercially available for sometime. They include DISCOS, ALLFLEX, TREETOPS and SD/FAST. These are primarily suitable for systems with large rigid body motion with flexible members undergoing small deformations.

Modal Representation

Flexible multibody simulation algorithms employ discretization of the continuum based on the classical assumed modes method. This method proposes that the deformation field for each flexible component in the multibody chain can be expressed as a series of spatial and temporal functions. In general, the spatial functions can be any admissible function satisfying geometric boundary conditions and they are often referred to as mode shapes while the corresponding temporal functions are termed generalized coordinates. A daunting task facing dynamicists and control engineers is the choice of modes in discretizing structural deformation. In particular, the focus is on selecting the modes which adequately capture the interaction dynamics involving system parameters, control characteristics and intial disturbances.

To establish a framework for selection of modal functions, consider a typical Spacecraft with a rigid hub and elastic appendages. A hierarchy of modes would need to be selected in order to faithfully represent deformation history of the appendages. Either component modes may be used in which the appendages vibrate with respect to the central body but independently of each other; or system modal representation may be performed where the entire structure vibrates accounting for dynamical interactions throughout the Spacecraft. The `component modes' method was pioneered by Hurty in the early 1960's. It involves determination of the appendage admissible functions with enforced geometric compatibility between the adjacent elements of the system. Since temporal generalized coordinates are associated with each mode for a given component, the size of the problem is directly dependent on the number of appendages and modes. Another drawback of this method is in the development of a dynamically faithful set of admissible functions for geometrically complex structures with interconnected flexible components. Further investigations on modal selection by Craig and Bampton, Benfield and Hruda and Hughes were aimed at improving modal convergence by more precise specification of geometric conditions at internal boundaries between the substructures. On the other hand, with 'system modes', frequently obtained by the finite element method, the size of the problem is independent of the geometric complexity of the structure. The study by Hablani suggests that, for a given order of discretization, prediction of the Spacecraft's dynamics improves as one migrates from the component to the system modes. Furthermore, system modes are physically more meaningful, since a modal frequency represents resonance of the the entire structure.

Another important issue is to model accurately a flexible lattice structure. For example, in the case of the Space Station, should the truss structure be considered homogeneous, and if so, how? Design of the lattice structure, which constitutes the main truss, must be highly reliable since it cannot be tested full scale in its operational environment prior to the flight. On the other hand, a detailed finite element analysis of the truss structure using truss bar elements would involve a large number of elements and nodes thus becoming uneconomical, especially at this initial design phase when the structure and its associated systems are subject to modifications.

Techniques presently used to study space lattice structures fall into three categories: (i) matrix methods, (ii) field techniques, and (iii) continuum modelling. The first consists of numerically solving a set of algebraic equations in a direct manner. The other two are analytical approaches. Numerical methods such as the finite element and finite difference procedures, use the matrix approach based on discrete element idealization. Field techniques attempt to describe lattice structures or a pattern of elements analytically. The popularity of this approach is due to the fact that the elemental nature of the lattice bay is preserved in the governing equilibrium equations. In comparison with the numerical methods, a field analysis does not increase the problem size as the number of bays in the truss structure is increased.

The last method consists of approximating a repetitive lattice by an equivalent continuum. This ensures that the continuum model exhibits equivalent energy levels as the actual discrete lattice. Here qualitative decisions reduce the dimensionality of the mathematical model and physically identify the nature of the deformation (e.g. warping, bending, shear). Furthermore, as the number of repeating modules is increased, the accuracy of the response improves although the model size does not increase.

This energy equivalence concept has been demostrated in a variety of investigations. For instance, continuous systems involving particular types of beam and plate type lattice structures have been developed by Berry et al., and Juang and Sun. Their studies suggest the necessity to model large truss structures by the geometrically nonlinear Timoshenko beams. While shear and rotary inertia lead to small corrections to the Bernoulli-Euler theory for the lower modes of long and thin beams, significant errors may be introduced if they are neglected when dealing with thicker beams, or for the higher modes of any beam.

9.2.2 Control

The subject of attitude and vibration control in Large Space Structures (LSS) has received considerable attention and has evolved quite rapidly in the last thirty years. Balas and Meirovitch have presented an excellent overview of approaches to the control of LSS. Unlike the rigid Spacecraft design, LSS control is an interdisciplinary subject drawing on structural mechanics, continuum representation, optimization and identification. Issues in modelling and control design include control/structure interactions, actuator and sensor selection and placement, controllability and observability, control and observation spillover, sensitivity and robustness, modelling uncertainties and errors, to name a few.

The primary requirement of the flight control system is to maintain the LSS attitude within 5 degrees with reference to the orbital frame. Control Momentum Gyros (CMGs) will be utilized as the primary actuating devices for most of the assembly sequence due to their greater torque capability for given weight and power consumption, as opposed to momentum wheels. However, they have limited momentum storage capabilities before reaching saturation. Therefore, a scheme for desaturation of the CMGs will have to be developed to remove secular momentum build-up. Desaturation methods include the use of magnetic torquer bars, aerodynamic torques, fluid desaturation, reaction control systems, and gravity gradient torques.

When suppressing structural vibration, the closely spaced modal frequencies, coupled with the uncertainties in structural modelling, place stringent robustness requirements on the control system. A frequently used approach to ensuring robustness is to colocate the sensors and actuators, resulting in an energy dissipating configuration. However, because of physical limitations on hardware placement, this approach is often not feasible, resulting in unstable control/structure interactions.

We describe now several design techniques, from the most common (Classical and Optimal-Quadratic) to others that either improve the description of the system (Nonlinear Control) or emphasize uncertainty of the system (Robust Control) at the expense of more complicated formulation and more computational effort.

Classical, Optimal-Quadratic and Nonlinear Control Design

In application of linear control theory to flexible orbiting systems, three procedures have been commonly used to develop control laws for large flexible structures: (a) decoupling techniques; (b) pole placement; and (c) optimal linear regulator theory.

The decoupling technique can be applied in two situations where: (i) the linear state equations are decoupled using state variable feedback; and (ii) the open loop linear equations are first transformed into a decoupled set in the modal coordinates and then control laws are developed independently for each mode. Thus it becomes necessary to transform the control laws as expressed in modal coordinates to the actual control in the original coordinates.

In the pole clustering method the overall transient requirements of the system are considered instead of concentrating on the behaviour of the individual coordinates. The linearized equations of motion are recast in the state space form and the feedback control law selected.

The linear regulator theory allows one to set, a priori, distinct penalty weighting functions on the control effort as well as the state variables. The feedback control law is selected such that a quadratic performance criterion is minimized.

Both the linear regulator problem and the pole clustering method can result in some of the closed loop frequencies being orders of magnitude greater than those of the uncontrolled system. These higher frequencies may also correspond to the frequencies of higher modes not included in the previously truncated model. To account for such effects the order of the original system model will have to be increased in order to avoid the effects of spillover. On the other hand, these methods have the advantage of being applicable even in situations where the number of actuators is less than the number of modes in the mathematical model, in contrast to the the decoupling methods.

Bainum et al. have provided considerable insight into the behaviour of complex large space systems by modelling simple systems such as flexible beams and plates in orbit. Many more studies towards control of flexible Spacecraft have been documented in the literature such as the works by Denman and Jeon (eigenvalue relocation), Ih et al. (adaptive control), Meirovitch et al. (perturbation approach), Young (decentralized control) and Williams and Juang (pole/zero cancellation), to name a few.

The linear control optimization procedures, based either on the Bellman principle of optimality or the Pontryagin maximum principle, have served as efficient algorithms to develop control tools and strategies in the design of a large class of dynamical systems. However, in real situations, the nonlinear character of the system may be important so as to warrant linearization unadequate. The mathematical theory of bilinear systems and the general nonlinear controllability and observability theory have been investigated via certain aspects of the differential geometric theory such as Lie algebras to yield some understanding of the behaviour of these nonlinear control systems. Dwyer and Batten have proposed an attitude motion controller for a rigid Spacecraft based on the inversion of a nonlinear input-output map. Singh and Bossart[attempted a linear representation of the nonlinear dynamics of the rigid Space Station using feedback linearization theory. More recently, Karray and Modi extended the application of the feedback linearization technique to flexible systems. The nonlinear control strategy based on this technique has the advantage over controller designs based on linearized dynamics that a linearized model is only approximate, and valid only near operating points.

Robust Control Design

A recently developed (during the last decade) control-theoretic approach that holds promise for addressing control problems for Large Space Structures (LSS) and others similarly challenging, is the so-called "Robust Control" approach. This methodology produces control designs which are less sensitive than conventional ones to inaccuracies in modeling and varying and/or unknown parameters, at the expense of being more computationally intensive. Application papers and experiments in aerospace and other areas have been extensively produced. It must be mentioned that the LSS control problems have helped stimulate these theoretical developments, and happen to be very much suited for them.We can mention, among others, the following robust control techniques:

1) The H-infinity approach. This is basically an optimal control theory which uses a different measure for the "size" of the sub-systems involved ("infinity" norm instead of quadratic norm),

which helps achieve better robustness properties. A recent advance, representing a breakthrough in the computational aspects of this mathematical technique, is the paper by Doyle *et al*, 1989.

2) The Structured Singular Value (also called μ or mu) analysis. This theory combined with H-infinity theory results in the so-called mu-synthesis approach. Here, explicit modeling of the dynamic and parametric uncertainty of the system is provided, so that the control design is satisfactory for a set of models in the neighborhood of a nominal model. Only physically meaningful uncertainty is considered (structured uncertainty description) to avoid conservativeness in the design. For the fundamental theoretical basis of this technique and its application to the control of LSS[GJ Balas, 1990, GJ Balas et al, 1989]..

Algorithms for these methods have already been developed and implemented in various control-aided-design software packages (MATLAB™ among others). Further research in these areas that will greatly enhance our capability of predicting and controlling behavior of the large space solar power program structures, will involve research in both improved system identification (modeling) methods for uncertain systems and large-scale computational schemes for all these "robust control" techniques.

Control of Large Space Structures: A Reduced Order Model (ROM)/Residual Mode Filter (RMF) Design Concept

During the design of any control system for a Large Space Structure (LSS) possibly containing an infinite number of modes, there is usually a constraint on the number of frequencies a certain design can accommodate. This is a phenomenon usually caused by the lack of computer speed to compute the necessary gains for the large number of modes associated with Distributed Parameter Systems (DPS). To get around this limitation, Reduced Order Models (ROM), which contain only some of the modes of vibration of the actual system (n modes) are employed during the control law and observer design of the control system. Figure 9.1 shows the closed loop with matrix A being the model of the structure, matrix B being the input matrix, matrix C being the output and matricies L_n, K_n and G_n being gain matricies within the controller.

Unfortunately, in the closed-loop with the controller, some of the modes not directly included in the ROM design become unstable (q modes), although some of them do not (r modes). This problem of Controller Structure Interaction (CSI) is easily solved with the introduction of a parallel set of frequency-locking filters or Residual Mode Filters (RMF), equal in number to the number of q modes [Balas, 1988, Davidson 1990, Ouyang, 1987]. These filters can be added after the original ROM based controller has been designed, and produce very little degradation of designed performance while yielding acceptable stability margin for the closed-loop operation. In general, this ROM/RMF design allows for low order control of very large systems (the limit has not yet been found, i.e. distributed parameter systems are in infinite dimension space) and has been successfully employed experimentally and computationally [Reisenauer, 1990, Balas and Quan, 1989].

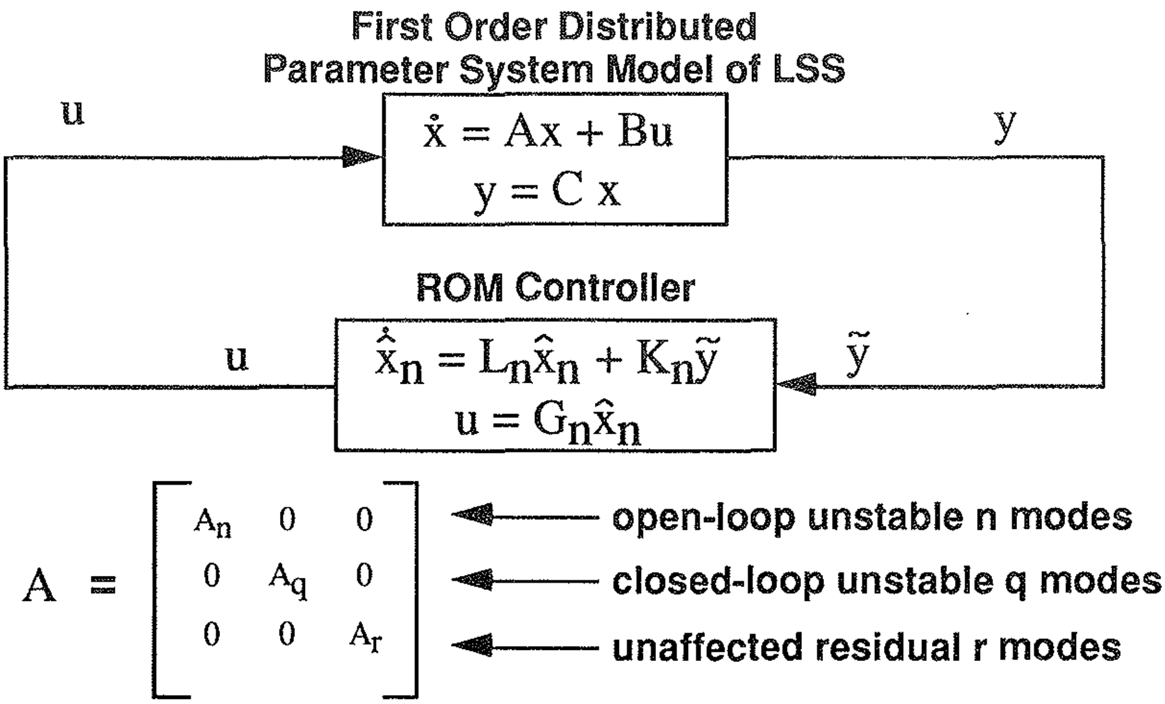

Figure 9.1 Block diagram of Reduced Order Model Design. This design sometimes creates closed-loop unstable q modes.

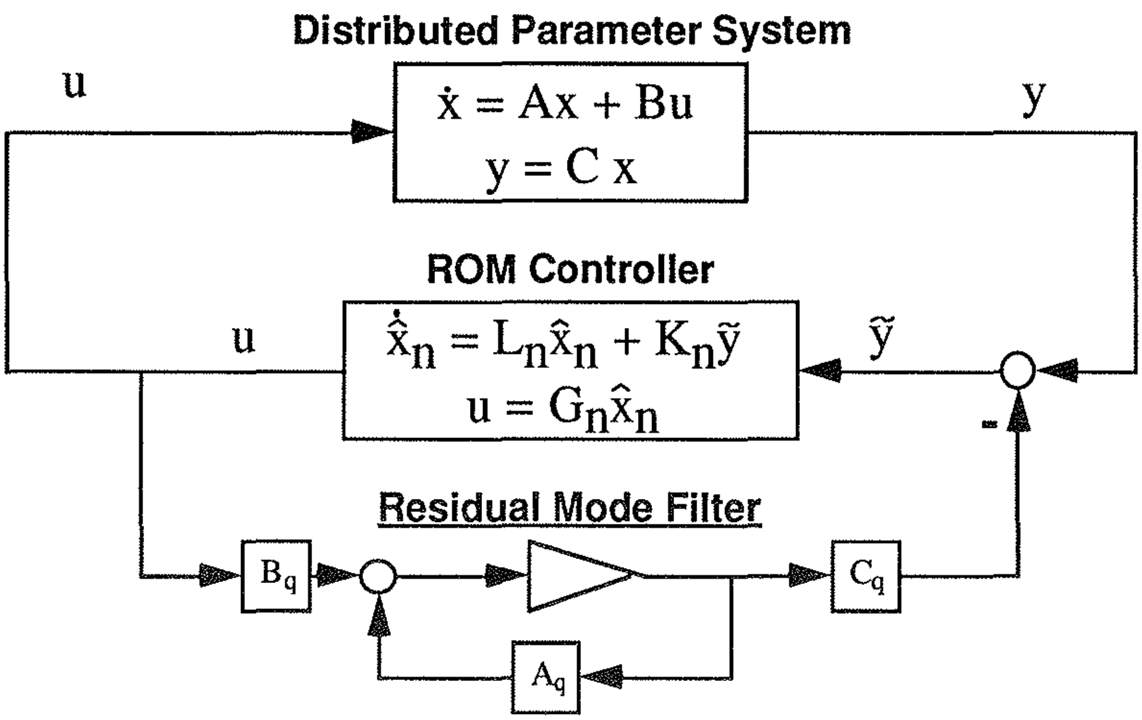

Figure 9.2 Reduced Order Model design with improved Residual Mode Filter that ensures exponential stability of the DPS.

9.3 Construction/Assembly Operations

9.3.1 Construction of Erectable Structures

As previously mentioned, the near-term space solar power program experiments and demonstrations are likely to be performed using small, inexpensive, deployable satellites. The larger systems envisioned in long term, commercial satellites will, however, almost certainly require construction and assembly capabilities in space. Many of these capabilities such as EVA construction and optimal construction configuration selection are currently being developed while others such as joining systems and engineering overlap issues are better understood. To better evaluate construction of erectable structures, a basic knowledge base is needed in each of these areas.

One of the first issues to be addressed in the erectability of space systems is joining systems. These can be separate into quick disconnect systems, basically systems that take a minimal amount of time and require very few tools, and permanent systems. Quick disconnect systems include Space Station Freedom(SSF) Quick Attachment Joints (QAJ), quick disconnect fluid lines and attach-disconnect electrical harnesses. These have reached a point where even such constraints as the use of bulky astronaut gloves have been overcome. The example of the QAJ demonstrates this in that an out-of-axis, easy to use, assembly joint was developed. Permanent systems, shown in Table 9.1 on the other hand, have been developed quite differently. [Nii, 1990] The use of welding for on-orbit repair of Soyuz-12 and the successful demonstration of welding and brazing on Skylab have been the primary milestones. [Anderson, 1988, Stuhlinger, 1975] It is through these successful demonstration that operational studies of erectable systems has been allowed.

Table 9.1 Classes of Permanent Joining Systems.

1. WELDING & BRAZING	2. MECHANICAL FASTENERS	4. ADHESION
Electron Beam	Bolting	
Exothermic	Riveting	5. BAYONET
Explosive	Snaps	
Laser	Threaded	6. SWAGING
Resistance	Wire Ties	Mechanical
Tungsten Inert Gas		Explosive
Ultrasonic	3. BONDING	
Brazing	Weldbond	
	Rollbond	
	Explosively Bonded	

The second major issue for quantifying the effectiveness of space construction is Extra Vehicular Activity (EVA). Currently, large scale EVA operations have been in maintenance tasks by the former Soviet Space Program and have been very limited in the area of construction EVA operations. The primary reason for this is because of the high cost of shuttle flights and because of the limited simulation capability of the neutral buoyancy facilities listed in Table 9.2.

Table 9.2 US. Neutral Buoyancy Facilities

NAME	LOCATION
NASA MSFC Neutral Buoyancy Simulator (NBS)	Huntsville, AL
NASA JSC Weightless Environment Training Facility (WETF)	Houston, TX
NASA ARC Neutral Buoyancy Facility (NBTF)	Mountain View, CA
University of Maryland Neutral Buoyancy Facility	College Park, Maryland
McDonnell Douglas Underwater Test Facility (UWTF)	Huntington Beach, CA

To date, within the US space program, there have been only three construction EVA experiments - Experimental Assembly of Structures in EVA (EASE), Assembly Concept for Construction of Erectable Space Structures (ACCESS), both on STS-61B, and the ASM space station truss assembly test on STS-49. The purpose of all of these experiments was to demonstrate the construction of large truss structures. These experiments demonstrated the assembly of trusses both large and small, from foot restraints within the Shuttle and in free flight along side the truss, and in many other conditions. A test of the maneuverability of a truss by an astronaut was also demonstrated. In general, these experiments together demonstrated a variety of construction abilities as well as confirmed assembly times taken in the neutral buoyancy facilities (Table 9.3).

Table 9.3 ACCESSAssembly Times vs. NBS Times.

TASK		TIMES (min:sec)	
	NBS Avg all tests	NBS Trained	Flight Trained
Setup	4:00	3:04	3:31
Assemble 10 bays	30:13	21:44	25:27
Disassemble 10 bays	18:45	15:00	18:52
Stow and Close up	5:23	4:30	4:41
TOTAL	58:21	44:18	52:31

Unfortunately, simulated zero-g facilities only give a static understanding of the tasks and are incapable of simulating the dynamics of the apparatus' with which the astronaut must work. This problem was very noticeable on STS-61B when capture of an INTELSAT satellite was attempted. In attempts to attach the capture bar to the satellite, the primary tool used in this task, astronauts found the dynamics of the Spacecraft to be incompatible with the tools they had. The process of trying to use the capture bar to dock the astronaut to the satellite would induce motions by the satellite that the crew had not anticipated or trained for. Finally, after three attempts with the capture bar, the mission was accomplished by using astronauts to control the dynamics of the satellite by holding the satellite in place while the capture bar was attached. This lack of dynamics and control simulation in the neutral buoyancy facilities will continue to be a problem for simulating EVA unless a computer controlled simulation can be developed.

In a space construction environment, the assembly of the structure part of the Spacecraft, which has been main area of emphasis of experiments in space construction, is only one part of the total assembly. Usually, members have fluid lines, electrical wire harnesses, and thermal control devices attached to them. This additional hardware complicates erectable construction because each non-structural connection requires verification and testing. Sometimes this requirement makes it necessary to assemble parts of the structure on the ground due to large concentrations of these connections, their difficulty in verification, or the crucialness of the connection.

An example of a design where this necessary was in the Space Station Freedom. Here large elements were chosen to be preassembled terrestrially instead of in space constructed because of the high EVA times associated with assembly, although the original goal was full erectability of the station. Unfortunately, this process of using large pieces decreases the efficiency with which a station can be packed into a launch vehicle. A more preassembled Spacecraft leads to the volume limit of the launch vehicle being reached before the mass limit, and thus, forces the need for more launch vehicles to put the station in space. In most cases, there is a trade off between fully erectable structures and structures delivered to orbit in large piece, with launch and EVA costs as the main parameters. This trade off is usually unique for each Spacecraft and is not done in a systematic Design I way, although research is being done to automate this decision [Jolly, 1992].

Engineering Overlap Issues

In addition to the choice of the size of pieces, many other engineering areas have issues that overlap with erectability of space structures. In general, the configuration of the design as well as the sizing issues with the overall Spacecraft and its individual sub-assembly pieces are concerns that are dual to engineering and construction and must have inputs from both areas before a final design is chosen.

Configuration issues usually deal with what kind of designs will be chosen and how certain elements will be connected together. Questions of whether to use a box truss or a tetrahedral truss

(Figure 9.3) or whether to go with something totally different often arise in support truss designs because of the lack of one totally superior design and the large number of design options [Fuller, 1975] Even with these restraints, methods to quantify and trade different types of truss structures have been attempted with some success.

For example, in 1988 Lichwala quantified several box trusses types and attempted to optimize the design (Figure 9.4). This was one of the first attempts to trade-off different types of box trusses, although it was incomplete in examining all the possible box truss types shown in Figure 9.5. Trade-off parameters such as fail-safe characteristics, operational access, and pallet mounting were included as well as the usual engineering parameters such as stiffness and torsional characteristics. The final conclusion reached was that the lattice truss which has all the diagonals crossed when looked at from any of the six box faces (RI. truss) had the best properties overall, although other truss types did have some better characteristics in some areas (Table 9.4).

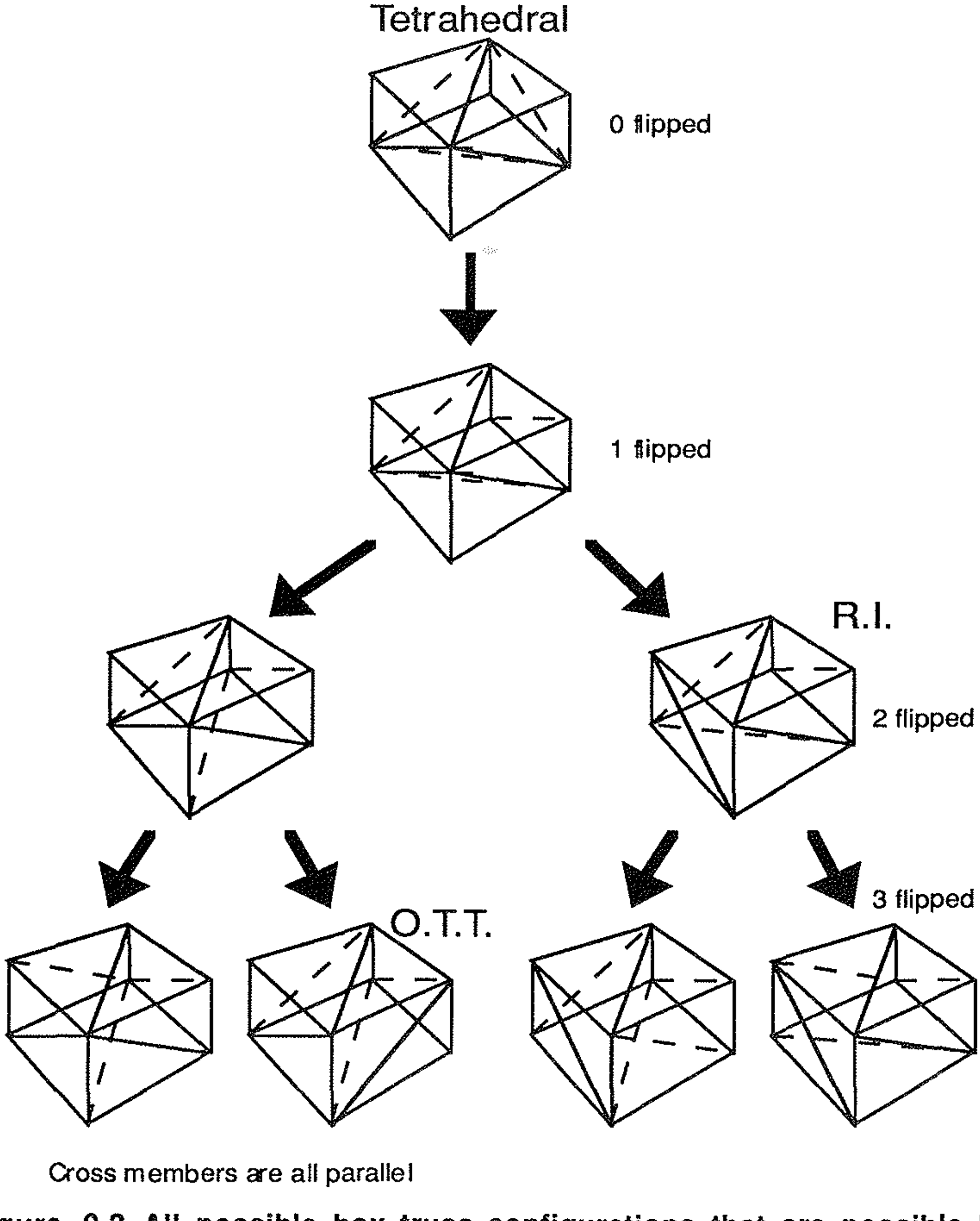

Figure 9.3 All possible box truss configurations that are possible by flipping the diagonal members on each of the six faces.

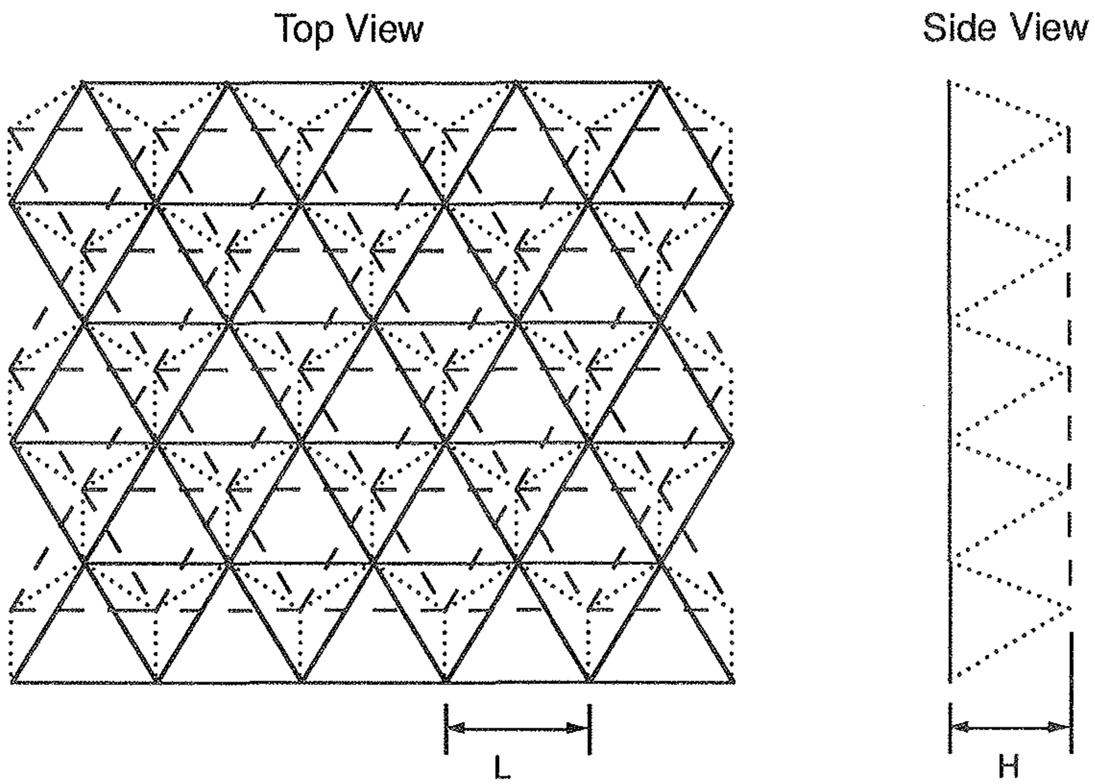

Figure 9.4 Top and side view of a planer tetrahedral truss.

Other attempts to study truss types was also done by Mikulas, et. al. in 1977 in their engineering characterization of a tetrahedral truss (See Figure 9.3). There analysis included not only the structural stiffness, strength and dynamics characteristics, but included sizing analysis' such as number of columns per square kilometer versus column length (Figure 9.6), column load due to orbital transfer and many others. Quantification of the structure also was done for several sizes and column lengths. Issues of this kind are paramount for considering the cost and feasibility of large scale projects such as an SPS project.

Table 9.4 Selection Criteria and Concept Evaluation (with bold dilating best choice)

CRITERION	CONCEPT A	CONCEPT B	CONCEPT E
Weight/Strength Stiffness		**Inherent Torsional Stiffness 50% > Concept E** **Uncoupled Behavior Simplifies Stress/Dynamic Analysis**	Coupled Behavior Reduced Stiffness Reduced Strength Margins
Fail-Safe Characteristics	Loss of Batten Disadvantage	**Stability Maintained With Loss of Select Joints**	
Operational Access Pallet Mounting and EVA Travel	9 Member Clusters at Joints are a Disadvantage	8 Member Clusters at Joints Not Preferred	**7 Member Clusters at Joints are Preferred**
Assembly Time	Not a Discriminator	Not a Discriminator	Not a Discriminator
Cost	Slightly Increased Packaging Cost	**Uncoupled Behavior Slightly Favors B**	

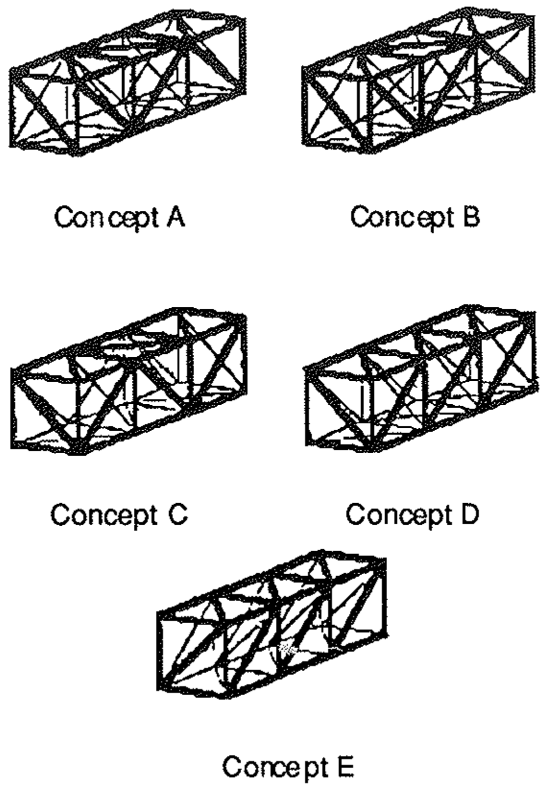

Figure 9.5 Five candidate trusses. Notice concept A, C, D and E (internal diagonals are aligned) and concept B (internal diagonals alternate)

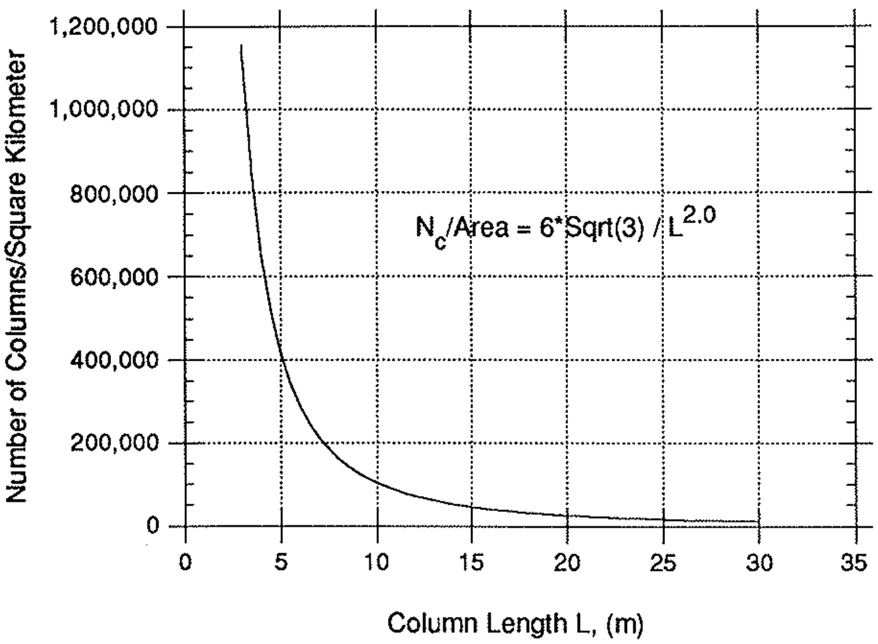

Figure 9.6 Number of columns per square kilometer in a tetrahedral truss as a function of column length.

With respect to lunar projects, many studies have been done as to sizes of lunar materials to be used as the building blocks for a lunar base. Ideas such as domes, both geodesic and single piece structures, bricks made of lunar concrete and basalt and many others have been proposed. All these varying scenarios all require different types and sizes of construction equipment. Construction issues such as crane sizes, furnace size, and number of people must all be considered in any structural design.

Advantages and Disadvantages

With the construction of erectable structures comes many advantages and disadvantages. These are outlined in Table 9.5.Many of the advantages associated with erectable construction come from the versatility and experience already in place, while many of the disadvantages stem from the harsh and expensive space environment. Trade offs between erectable structures, both EVA and EVR assembled versus deployable systems can be made if each of these advantages and disadvantages are well understood. Characterization of the trade offs and a set of rules has been developed [Smith, 1992], but is still yet to be used because of a lack of information about the system.

Table 9.5 Advantages and Disadvantages of Erectable Systems

Advantages
Versatile in Construction Operations
Low Sensitivity to Changes in Operations
Highly Adaptable to Operation Perturbation
High Amount of Experience
Current System Availability
High Volumetric Packing Efficiency
Short Assembly Time
Disadvantages
High Costs
Repetitive Tasks
Low Time Available Work Time (Pre-breath/Post-breathTimes)
High Training Time
Many Safety Issues

As SPS moves into its commercial phase, space construction of erectable systems will have to move from an experimental phase to more of an operational phase. EVA is just beginning to be understood but more work on simulation and dynamics modeling is needed before it becomes a common task. Optimal decision making of sizing basic assembly elements and how much should be terrestrially built versus on orbit assembled also is just beginning to be researched. In contrast, many of the engineering overlap areas are well developed and are ready to be implemented, but in general many of the necessary technologies needed for the erection of structures are currently being developed to a degree such be used in an SPS by the time large scale commercial development takes place.

9.3.2 Deployable Structures

Due to the limiting capability of current space transportation systems, and the necessity for simple construction operations, deployable structures are an option for manned and unmanned missions in the near and distant future [Natori, 1985]. Currently, deployable structures are used in solar arrays and reflector structures. They are classified as, the type of structural materials needed to carry out a mission profile, and whether the system requires a back-up structure or not. Various concepts can use combinations of both.

Design Considerations

The rationale for the design of all structures must start with a requirements definition and the environment of the mission. The design of deployable structures involves consideration of three different environments; manufacturing, transportation, & operations. Each of these impact the structure in varying degrees. The largest forces to be considered are the vibration and acoustic noise incurred at launch. Next, would be the gravitational and handling force the structure is subjected to on Earth. The least would be the operational forces on the structure from the space environment. However, if the designed structure were to respond exactly to the loads incurred at launch, the structure would be over designed. This is due to the fact that deployables are a packaged payload on the Spacecraft, therefore the loads incurred by each individual member during launch is not as great because of the distribution of the forces over the entire package and not to the individual members.

Deployable structures are also subjected to external space forces. The effects of gravitational forces as well as atmospheric drag have large elastic deflection effects and cause instability of the structure [Natori, 1987]. Thermal effects of solar radiation, cause thermoelastic deformation. In addition to thermal solar radiation effects, the structure is subjected to three types of pressure, absorbed radiation, diffusely reflected, and most importantly specularly reflected. These forces are important in the consideration and the choice of the technology as well as the design configuration of the structure. Although, these forces are negligible compared to the terrestrial manufacturing and transportation loads due to the 1G force on the structure as well as the stresses incurred by handling and testing. Therefore, the primary environment that a deployable must be designed for is the terrestrial manufacturing environment.

Deployable Structures

Deployable structures use many different technologies and employ different structural concepts to achieve their mission objective. The following is a brief description of the different technologies and their appropriate applications.

The most near term referenced structural shape is linear (the boom). Deployable truss structures have played an important role in satellite development for antennas and booms in the last 35 years. High stiffness properties have triggered much interest and research. Truss structures fall under two types of headings, one dimensional beam structures and two dimensional planar structures. Extendible beams have been used as supporting members for substructures and instruments as well as main structures. Some deployment concepts of beams include telescopic masts, coilable "astromast", collapsible mast, and variable geometry trusses (fig. 9.7a). These trusses utilize geometric forms of polyhedra in various combinations. The basic building block for a typical geodesic beam is the octahedron. Geometrical transformations of polyhedra to collapsed configurations by folding and/or changing the length of the members allows for efficient packaging (Figure 9.7b). Applications of these structures would be as a substructure to a solar array, or rectenna

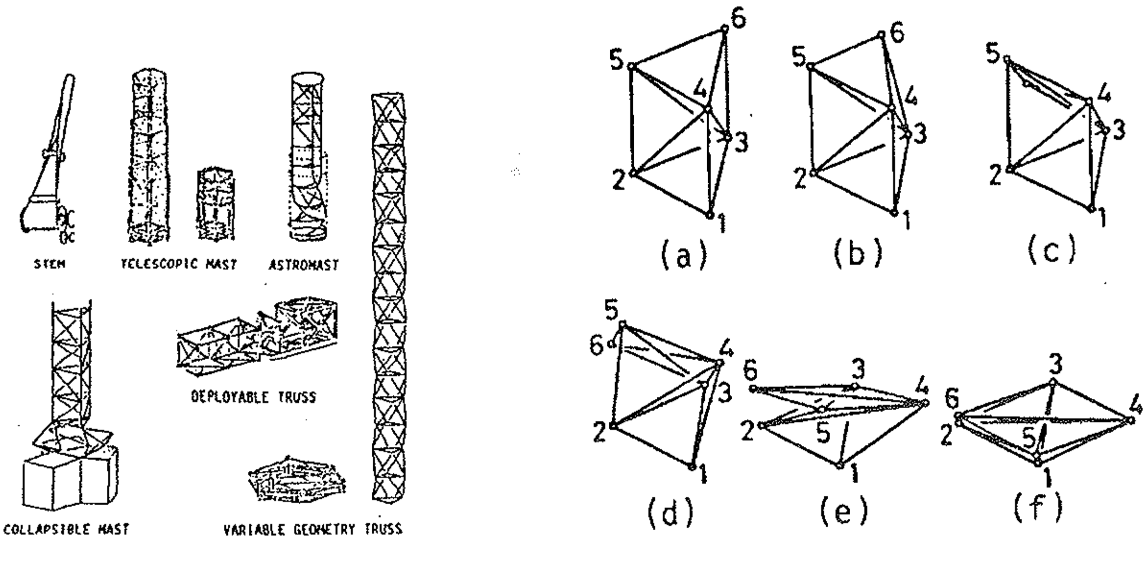

Figure 9.7 (a) Beam Deployment Concepts (b) Tetrahedral Fold
[Natori and Miura, 1985] [Natori et al, 1986]

The translation of a one dimension russ to a two dimensional truss could be executed by adding two tetrahedrons to the octahedron. Combinations of these elements form an octet truss (Figure 9.8a). Polyhedra in various combinations can also form different planar truss structures. Generating a curved truss from a flat planar truss can be accomplished by changing the length of the surface members or changing that of the diagonal members (Figure 9.8b). These planar structures allow for more control and stiffness over large spans as well as ease of deployment and efficiency packaging. Applications to space power could be used in combinations with modular inflatable structures for deployment of large antenna.

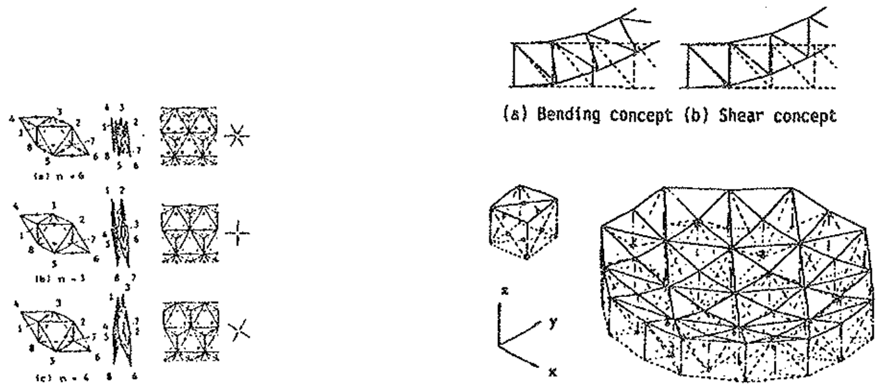

Figure 9.8 (a) Octet Fold [Natori et al, 1986] **(b) Parabolic Surface Truss** [Natori et al, 1986]

The difficulties with large scale space structures occur because of their size. It is extremely difficult to fabricate, handle, and test these structures on the Earth. Current fabrication concepts are not feasible and thus the realization of many projects are greatly influenced [Miura et al, 1986]. Breakthroughs in membrane structures offer solutions to this problem. A combination manufacturing and deployment folding system has been developed by Miura, Natori, and Sakamaki for large planar membranes (Figure 9.9 a,b). Large membranes coupled with deployable supports could be applicable to Space Solar Power Program solar array deployment as well as transmitting and receiving antenna for future space power satellites. Applications of these translate into fewer launches and possible less EVA. This suggest the possibility of a viable Space Solar Power Program in the near future.

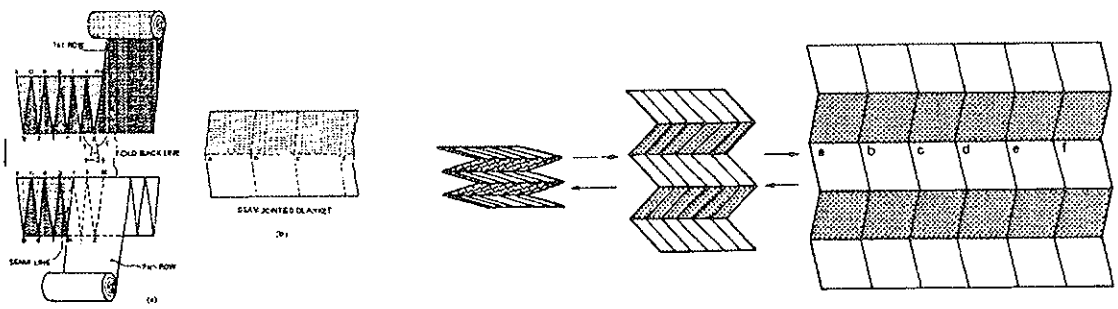

Figure 9.9 (a) Fabrication Method [Miura et al, 1986] **(b) 2-Dimensional Deployment** [Miura et al, 1986]

Membranes are also applied to inflatable structures. Inflatable surfaces show high accuracy when used as an parabolic shape for antenna and high efficiency in packaging. However their are difficulties with inflatable surfaces. For example reflectors must be manufactured to be one entire structure, and for such a large membrane element, precise accurate manufacturing process and overall handling treatments on the ground are required. Also, due to the lack of internal hard points shape control of the surface is difficult. The larger the surface becomes, the surface root mean square (rms) error increases [Kato et al, 1989]. Modularized inflatable structures (fig. 9.10a) with truss structure supports are projected to be a way of combating these difficulties. Large rigidized inflatables offer several advantages for space applications by good stiffness and thermal stability. Ongoing work is centered on the realization of scaled 3 m. reflectors to be subjected to mechanical and electrical tests in the immediate future (Figure 9.10b) [Bernasconi, M. C. 1984].

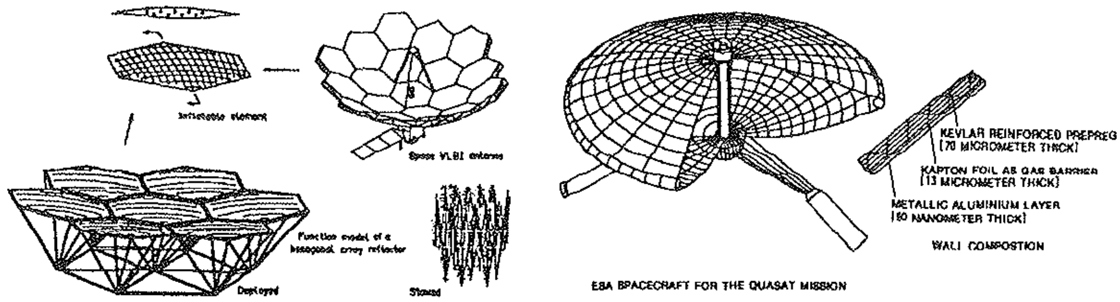

Figure 9.10 (a) Modularized Inflatable (b) Rigidized Inflatable
[Kato et. al. 1989] [Miura et. al. 1986]

Relative to other structural concepts, adaptive trusses are new to space structures (Figure 9.11). The basic premise is to vary the geometrical configuration of the truss by automatic extension and contraction of specific members for deployment and arbitrary change. These structures are studied for their application to docking structures, space cranes and the control of the shape of antenna structures.

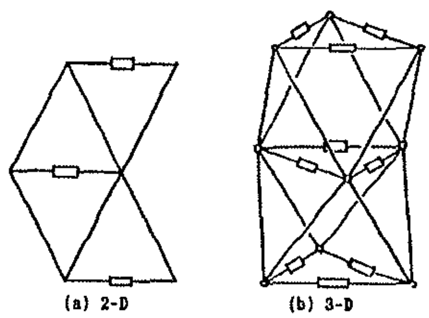

Figure 9.11 Adaptive structure
[Matunaga et. al. 1990]

Conclusions

As the need of large scale structures becomes a reality in space solar power, there will be the need of further development of deployable high-precision elements. Considerations of packaging efficiency, manner of deployment, and the number of joints in the system are equally important. Efficient packaging will minimize the number of space flights, while reducing the number of hinges and nodes is highly desirable for reliability and accuracy. Many times these systems will conflict with each other, therefore, it is important to have many deployable concepts available for space applications [Natori, Miura, 1985]. Our ability to develop these technologies should not limit the experiments that can be done quicker and cheaper to explore fundamental scientific questions. Nevertheless, construction and manufacturing technologies must be ready when the larger systems call upon them.

Table 9.6 Advantages and Disadvantages

Advantages of Deployable Structures	Disadvantages of Deployable Structures
Minimum human interfacing required	Decrease in structural stiffness
Structures require min. payload area	Difficulties in terrestrial testing
Easy transport	Lack of structural accuracy
Membranes require no mechanisms	Require high-precision elements

9.3.3 Schedule Issues for Deployable and Assembled Structures

Certainly, our ability to assemble and deploy large structures will be a driver in the very large satellites envisioned for a space solar power program. This program is not the only one in which construction methods will be required, though. The assembly of Spacecraft to be used in Mars missions and the installation of a lunar outpost are two mega-projects in their own right which will be enabled by the use of these technologies. Deployable structures (to date usually antennae, solar panels, and booms) have gotten larger as space missions have gotten more ambitious. One can see, then, that the development of these technologies has a schedule of its own which interacts with many programs. Milestones in any one of these programs can be considered advances for all of them. Figure 9.12 diagrams these milestones in a logical possible order.

Programs and milestones which are important for assembly techniques of large structures include:

- Reliable techniques for GN&C of expanding structures during the construction phase. This is especially important during the very dynamic periods when vibrations are occuring and mass properties change continuously or in large steps. This will be demonstrated during the assembly of Space Station Freedom.
- Assembly and construction of a Spacecraft with high pointing accuracy (as might be required for beamed energy transfer.) A test article in ~2005 is suggested.
- Operation and control of large, flexible Spacecraft with vibration modes which could interfere with its mission. Operation of Space Station Freedom will be a major milestone in this regard. This might be especially important for those Spacecraft which have a high requirement for pointing accuracy.
- Trades should be performed for each project where the optimum point of EVA vs robotic assembly lies. This will change based on the structures themselves as well as the capability of space robotics at that time.
- Trades must also be performed on the assembly of many small parts, which can be densely packaged in a launch vehicle, vs the launch of volume inefficient, preassembled structures.

Programs and milestones which might be important for deployable structures and methods include:

- Increased reliability of advanced deployable structures.
- Development of deployable high-precision elements. This might include the demonstration of a deployable solar dynamic generator with both high pointing accuracy and high precision reflector requirements. This could, perhaps, be the same test as one suggested for in-space manufacturing (9.5.3, below) wherein chemical vapor deposition to produce a reflector is suggested.
- The assembly and deployment of the many structures at a Lunar Outpost will provide skill in assembly in general as well as the infrastructure on the Moon for the use of non-terrestrial resources.

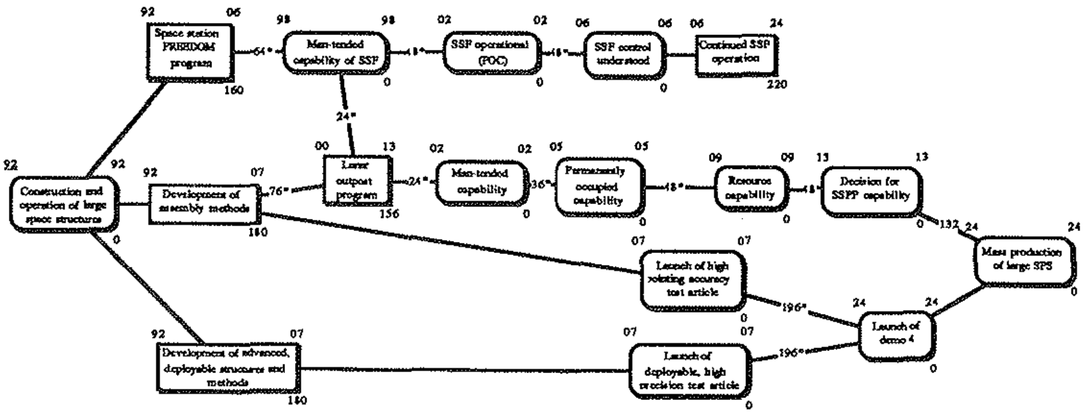

Figure 9.12 Construction, Assembly, and Deployment Task Schedule

9.4 Non-Terrestrial Resource Utilization

Within the near-term time constraints of the design examples prepared for this study, it is difficult to propose a viable role for on-orbit construction, in-space manufacturing, or non-terrestrial materials utilization. There have been several studies which have concluded that large-scale SPS's are not feasible with only Earth-launched material, and have looked at the possibility of using lunar material to provide valuable resources for the construction of Solar Power Satellites [Maryniak, 1991], [Leonard, 1991], [Lewis, 1991]. Analysis of the support required for such a scheme [Woodcock, in Glaser, in press] emphasizes the technology development yet to be done and the infrastructure necessary on the Moon as a source of uncertainty in such plans. A program for developing these technologies is necessary so that we have the capability to realize the larger systems which may be proposed for the long term, if they are justified by the prior experiments. The near term goals and products of these programs should make sense in and of themselves, i.e., they should be justified by their benefits vs. the alternatives, regardless of a solar power program. These trades should include not only the economics of mass pay-back, but also ease of operation, mission accomplishments, ease of program evolution, and similar intangibles which are difficult to quantify.

In their critique of the NASA/DOE study, the National Research Council cautions that

> "A decision to proceed with an SPS should not invoke a concurrent decision to develop the capability to use lunar of other non-terrestrial resources. For the next several decades, it would be more practical to use materials from Earth, thus minimizing the new technologies that would have to be developed to construct an SPS" [National Research Council, 1981.]

Nevertheless, the mass which must be placed into orbit for any of the commercial, base-load SPS systems proposed to date is just very large. Unless breakthroughs in several technology areas occur, these systems are likely to remain large. Therefore, one can propose that, if commercial SPS's are to be developed in the future, we must consider developing the ability to use resources which already exist in space. These resources could include materials indigenous to the Moon, asteroids, or even refined material such as empty external tanks brought to orbit by the U.S. Space Shuttle.

9.4.1 Lunar Resources

The concept of lunar resource utilization gives a new meaning to the phrase "living off the land." Lunar soil may be used to produce the oxygen, water, and radiation shielding astronauts will need to survive on the Moon. Lunar soil may also be used as a source of propellants, metals, and carbon dioxide to support plant growth, resulting in an enormous savings in transportation costs [Mendell, 1985]. The Lunar Energy Enterprise Task Force [NASA, 1990] looked at three options for the use of the Moon to help provide energy to the Earth, including both solar power satellites in Earth orbit and a lunar based solar power system. It concluded that the Moon must play a role in the long term energy supply to Earth. Although one must account for the fact that the study was commissioned by NASA, and might therefore be biased towards large space programs, several good reasons for this conclusion were presented and a timeline was developed.

During the Apollo era we learned the detailed mineralogy and chemistry of lunar materials as well as the rock and soil compositions at various locations on the Moon. In addition to abundant oxygen, these materials also contain considerable silicon, iron, calcium, aluminum, magnesium, and titanium which can be extracted as metals, possibly as co-products of the same process which extracts oxygen [Sullivan and McKay, 1991]. The average composition is shown in Figure 9.13.

We also learned that lunar soil has trapped particles from the solar wind over the eons, and thus contains helium, hydrogen, nitrogen, and carbon from the sun. These elements can be extracted as gasses (as CH_4, CO, and CO_2 in the case of carbon) by heating the soil. These gasses are found in most lunar soils, but their concentration is low and varies from place to place.

The materials which are likely to be needed in large amounts in a space solar power program, and are likely to available from non-terrestrial sources, include structural materials such as metals and perhaps low tech ceramics such as fiberglass. By quickly quenching a melt of the lunar soil, glass fibers can be produced which might be useful in a variety of ways. Metallic silicon, for solar cell production, has already been produced in the lab from simulated lunar material. It remains to be seen if this process can be scaled up and whether the purity required for solar cell production can be met.

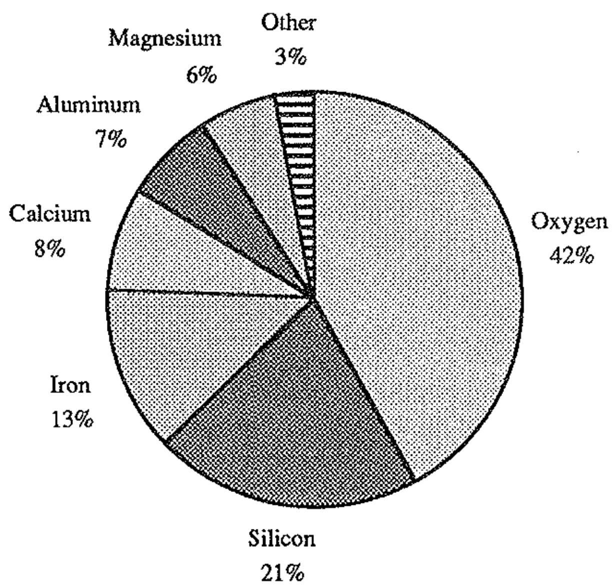

Figure 9.13 Average Lunar Soil Composition

An indigenous space resources utilization (ISRU) program has been envisaged which will be evolutionary, justify itself at each step, and demonstrate the necessary technologies for the next step. This can all be done with an eye towards developing technology to support an SPS construction project in the long term. This long-term vision will always be present, but should not be the sole purpose of the program. Of course, this approach only makes sense when a lunar base has been established.

A program which aims to produce large amounts of materials in the long term needs to start out first demonstrating the basic technologies. We need to consider which products can reasonably be made at a lunar outpost, considering the constraints both in the near term and in the long range capabilities. Obviously, as in an Earth-based marketplace, some products will have higher value and some will be easier to produce than others.

When considering the general operations that must be accomplished to produce materials from the regolith, several steps can be outlined. First, one must be able to obtain the soil by deconsolidating and hauling it from a mine site and placing it into the processing equipment. This is no minor task in the hostile environment encountered on the Moon. Next, there can be physical and/or chemical processing. This is often quite energy intensive and done in a pressure vessel. Both these requirements place a heavy burden on the infrastructure of the lunar outpost. The product must then be separated from any by-products, and both must be removed from the pressurized reactor. Finally, any processing aids, such as chemical reagents or consumable equipment, must be recycled as completed as possible to reduce any re-supply requirements from Earth. Many of the above tasks will involve the use of robotics or expert systems. By learning to accomplish these unit operations on the lunar surface, the experience should be applicable to many other products and processes, and will act as a knowledge base for future scale-up.

Lunar Oxygen

Lunar-derived liquid oxygen (LOX) is one of the products which is most likely to be worth producing at a lunar outpost. Its main use, from a mass viewpoint, will be as a propellant to power a lunar lander up to lunar orbit and back down to the lunar surface, or just for a direct return to Earth. The amount of oxygen required to make up for losses in the life support system is likely to be small, but represents another market - one that might be attainable before the larger production capacity is in place. Moderate amounts of oxygen might even make it practical to have less complex life support systems for habitats and extravehicular activities. The cost and maintenance of these systems may therefore be reduced.

The use of lunar-derived LOX is thus a high-leverage item because it frees space vehicles from the inefficient and costly exercise of shipping bulk propellants. The total mass shipped to the Moon will be reduced significantly. And for the remaining flights, instead of transporting large quantities of LOX to the Moon, more people, complex equipment, and scientific instruments can be shipped to provide additional capabilities at the lunar outpost.

Many chemical processes have been identified through studies and workshops sponsored by NASA and others which can potentially extract oxygen from lunar rocks and soils. NASA, universities, and industries throughout the world are all trying to understand these processes more fully to pick the best ones for plant design. It is important to note that the necessary co-products of oxygen production from the metal oxides in the lunar regolith are metals, especially iron and silicon. Aluminum and titanium are harder to produce, but would be useful for eventual satellite construction. This directly ties the initial production of oxygen with materials of value for construction of large space structures in the future.

Other Basic Processing Capabilities

Perhaps as important as propellant production will be the use of the regolith for the manufacture of basic material. While it is true that much of the cargo arriving on the Moon will be extremely complex equipment, there is a real need for simple, basic infrastructure; such as roads, rocket blast protection, and structures for habitats, storage, and equipment repair. If brought from Earth, the mass required for these uses would be enormous. For example, just for protection from solar particle radiation, the mass that would have to be brought to the Moon represents several Space Shuttle launches. The cost of transporting the hundreds of metric tons needed to protect an early habitat from this dangerous radiation would surpass a billion dollars at today's launch costs.

The general theme for all of the above is that the basic capabilities of mining, bulk material handling, and processing experience can be developed for production of material which is important for the initial customer, which is likely to be the first lunar outpost (a government or international agency.) These products will lower the cost of operating and expanding the base by providing a certain amount of self-sufficiency. The initial units for small production will themselves be small and thus have a short mass pay-back time. The lowered amount of mass that then must be shipped to the Moon will result in lower costs, but also enable the transport of larger units for the production of increasing amounts of products within the constraints of the existing flight rate and space transportation vehicle capabilities.

In their 1986 report, the National Commission on Space recommended the formation of "A continuing program to test, optimize, and demonstrate chemical engineering methods for separating materials found in space into pure elements suitable as raw materials for propellants and for manufacturing." This directive was based on lab results from preliminary tests of oxygen extraction using electrolytic and chemical processes. The Commission continued with the following recommendation: ..."Research to pioneer the use, in construction and manufacturing, of space materials that do not require chemical separation; for example, lunar glasses and metallic iron concentrated in the lunar fines." The development of many of the technologies in each of these disciplines will be synergistic. As in any development program, time and effort will be necessary to bring these possibilities to fruition. We have decades of experience to call on from the chemical processing, mining, energy, metallurgical, and manufacturing industries, however.

9.4.2 Other Non-terrestrial Resources

Long range, it might be reasonable to consider the use of asteroids to provide material for construction or propellant manufacture in space. The difficulties in developing a process to utilize a resource which has not yet been sampled and characterized are enormous, however. The logistics of obtaining and processing the material are also daunting. It is not likely that this will occur in the near future given the current space program plans. Nevertheless, missions to carry out initial reconnaissance, and perhaps sample return, of selected asteroids with desirable orbital parameters seems warranted from a scientific viewpoint. This has the feature of providing the knowledge on which to base future decisions concerning the use of asteroids for indigenous space resource utilization.

Man-made material which is currently discarded in space is another source of resources in space. An example of this is the use of external tanks from the U.S. Space Shuttle or the core of the Russian Energia. These are a large source of refined aluminum, mostly, which is currently allowed to re-enter the atmosphere and burn up. Other material which is in orbit, but is considered orbital debris and constitutes a hazard, could conceivably be recovered and reused. The amount of effort and expense involved is likely to be unreasonable for this latter scheme, but further study may be warranted.

No matter where the non-terrestrial resources come from - the Moon, discarded launch vehicles, or asteroids - there is an interaction of its use with the assembly node for the big satellites to come later in an space solar power program program. If we use non-terrestrial materials, then we probably don't want to transfer the material to LEO and assemble SPS's there since we'll be expending energy to get the material to LEO only to have to raise it to GEO again. This is not true, however, if we are going to use a constellation of satellites in LEO. Our choices for the assembly location are limited to either LEO or some high orbit, since assembly of structures in the Van Allen belts would be harmful to both the crew and the satellite. The cosmic radiation present in very high orbits can be tolerated by humans for a reasonable period, such as several weeks or even months. The safety implications of this are dealt with separately. Also, if we want to traverse the Van Allen belts with a large satellite being slowly accelerated and slowly spiraling out to GEO, we could damage the solar cells and electronics. Preliminary conclusions might be that, if non-terrestrial materials are used and GEO is selected as the orbit for the final product, then we should build it in GEO. If non-terrestrial materials and a constellation of satellites are used, then we should build them in LEO. If non-terrestrial materials are not used, then the assembly point can be decided on its own merits.

9.4.3 Non-terrestrial Resources Development Program Schedule

In reducing the above goals to a schedule, we must consider step by step tasks which will address the most important questions presented above. These appear in Figure 9.14. We should then consider how these relate to each other in time. There will also be relations of these tasks to other programs, both within the space solar power program and, more importantly, within other programs which justify their short-term existence. For instance, if lunar oxygen drives the initial study of lunar processing, a schedule focusing on how this will enhance and enable lunar outpost operations would be a driving factor. Such a lunar resource utilization program might entail several steps:

- Global orbital mapping of the Moon for resources should begin. This should be followed by robotic landers (such as the proposed Artemis program of NASA) to allow for further *in-situ* analysis of the chemical and mineralogical content of the regolith as well as, perhaps, some *in-situ* resource processing tests.
- Laboratory process development should continue, with engineering development to begin when appropriate. This should be integrated with the geochemical mapping missions.
- Supplying small amounts (500 kg) of oxygen which can re-supply boil-off to a Lunar Excursion Vehicle (LEV) and also be used for life support purposes. Perhaps small amounts of oxygen would obviate the need for closed loop systems on space suits, resulting in lighter weight portable life support systems (PLSS.)
- Then supply larger amounts of liquid oxygen. Current baseline designs for direct ascent and return to Earth (NASA's First Lunar Outpost Study) seem to require ~8 to 10 mt of LOX for return to the Earth (Joosten, 1992.)
- Eventually, larger amounts of LOX might be justified for round trip propellant use once reusable landers are phased in for trips to/from Low Lunar Orbit. Production of metals from the by-products of oxygen manufacture could begin.
- Supply small amounts of water, also for life support. This can be done by reacting the oxygen which is produced (above) with hydrogen in an 8:1 ratio, thus only 1/9 the mass of the product water needs to be brought from Earth. Even better, use the residual hydrogen in an LEV descent stage for this purpose. Eventually, this hydrogen might be supplied from the lunar surface solar wind volatiles, but a large mining capability must be in place for this to be possible.
- For solar-wind implanted volatiles extraction, begin a program which produces enough hydrogen to provide small amounts of water (see above.) Other gasses will be released in this process, such as nitrogen (for life support, pressurization, transport gas, etc.), carbon-based gasses (CO, CO_2, and CH_4), and helium (for pressurization, transport gases, and even preliminary experiments of He-3 extraction). Longer term, if it makes

sense at the time, larger amounts of hydrogen may be produced for propellant use, water manufacture, etc.

- Develop the ability to produce construction materials in support of an expanding lunar outpost. This can have a phased "marketing plan" similar to those above. Start with very low-tech, "dumb", non-critical uses, such as landing pads to minimize blast ejecta (dust) during landing and take-off, paved areas near the airlock for initial "brush off" of the suits for primary dust control, walls for non-pressurized structures to provide some thermal control, etc.
- Develop lunar mining capability so that real production can be done robotically. This digging unit should be tested very early, perhaps on the second visit to the lunar outpost. An early task might be to use it to provide radiation protection for the outpost by covering the module with regolith.
- Demonstrate the extraction of metals from oxygen-production by-products.
- The search for asteroids should continue to identify likely candidates for eventual resource evaluation and use. Spectroscopic analysis, and eventually a sample return mission, should be attempted if a likely candidate is found. A near-term mission to do reconnaissance of the Moon and an asteroid, entitled Clementine, is planned for a 1994 launch.

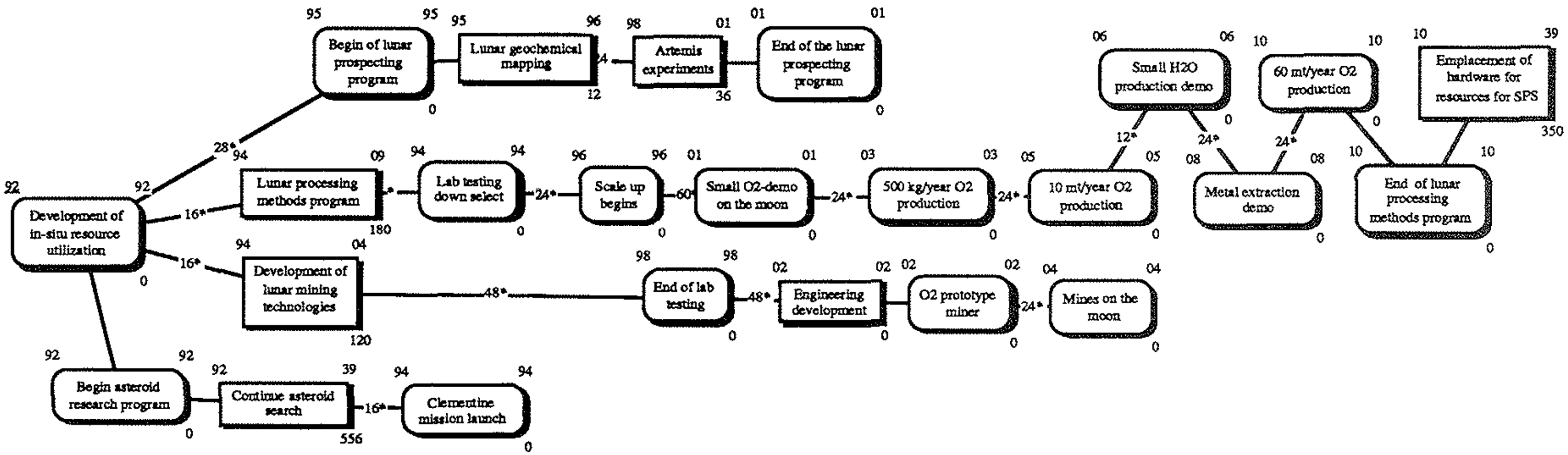

Figure 9.14 Non-Terrestrial Resource Utilization Task Schedule

9.5 In-Space Manufacturing

As presented in section 9.3 above, the time constraints of the near-term design examples make it difficult to propose a viable role for in-space manufacturing within the scope of the space solar power program. However, a plan for developing these technologies is necessary so that we have the capability to realize the larger systems which may follow. The near-term goals of this in-space manufacturing should justified themselves separate from a solar power program. The same studies which conclude that large scale SPS's require non-terrestrial material imply that there must be a manufacturing element present to transform these raw feedstocks into useful parts.

In considering where to focus our efforts for in-space manufacturing and construction, it is useful to consider what location makes the most sense for each operation. Realizing that lunar regolith must be chemically processed to provide the materials of interest to a Space Solar Power Program, it makes sense to consider doing chemical processing on the Moon, where gravity helps to hold material in place. Once the desired refined materials are in hand, the physical processing of these metals or other feedstocks might be best accomplished on orbit, where micro-gravity can aid in the production of large structures with unique properties. When the chemical processing is done on the Moon, the mass of the waste material which will be generated will not need to be lifted, thus saving a great deal of launch energy. We can launch refined ingots of Fe, Al, and Si to L2 or some other construction/assembly point very easily by using electromagnetic launchers. These metals will not need a "bucket" for launch and will not tend to disaggregate, simplifying both the launcher and catcher design. Physical processing of these metals at micro-gravity into foamed beams or thin films can utilize the properties of space to produce materials that could not be made on the Moon or would not survive launch.

9.5.1 Lunar Manufacturing

The availability of nonterrestrial resources is the key both to greatly increased energy on Earth and to large scale human exploration and exploitation of space [Energy Enterprise Task Force, 1990]. However, limitations in technology and the cost of sending and maintaining equipment and humans in space make manufacturing in space expensive. The cost of lifting one pound of material from the lunar surface into LEO is less than one-twentieth that of launching from Earth to LEO. Manufacturing materials on the Moon for use on the Moon is even more favorable. The use of the lunar regolith as a radiation shield for lunar outposts could be a first step in exploiting non-terrestrial resources for large scale projects in space. This could be followed by extraction of oxygen and metals for use as propellants and structural elements, production of cast basalt or glassy structural materials, fabrication of refractory materials, etc. Manufacture of composites in space, using glass fibers or metals produced on the Moon as both reinforcing material and matrix, could provide structural materials for future space structures [Goldsworthy, 1985].

We must ask if we can make all the necessary items for such a large project from the resources available on the Moon. The possible items required include solar cells, wires, microwave reflectors, and metal support structures. It has already been demonstrated in the laboratory that iron for structures and wires, fiberglass and iron for antennae and reflectors, and a variety of other individual products have been produced from simulated lunar materials. The vacuum and lower gravity present on the Moon or in space may actually make it easier to produce many of the articles we need(it should be noted that chemical processing may be difficult). It remains to be shown in a research and development program that large scale production of these materials and fabrication of such items can actually be carried out on the Moon, but a focused effort should be able to accomplish many, if not all, of these goals. The most "high tech" elements of such systems may still be imported from Earth, however.

An interesting self replicating, expanding lunar factory design was proposed by Freitas and Zachary [Freitas and Zachary, 1981]. The idea was to let the system grow itself from a 100 ton "seed" into a much larger, more capable system. The latest developments in artificial intelligence and robotics may provide alternative solutions to the problems such a system would face.

Whether the space solar power program considers the use of large satellites or the lunar surface as the platform for the collection and transmission of energy manufacturing capabilities at the lunar base would need to be developed. An evolutionary plan to provide a facility which can use metal stock to manufacture spare parts for the lunar base is one approach. At first, this metal stock (bar, plate, rod) would be provided from Earth and would use computer controlled tools to manufacture spare parts for the base. Thus, only the information of how to make spare parts for all of the relevant equipment (at least those which are non-critical) would need to be provided, and not the parts themselves. Statistically, not all of the items for which you might want spares are going to break. Also, the volume constraints and warehousing of these parts would be relaxed. Eventually, the stock could be supplied by the ISRU facilities, at least for low-tech needs where low grade alloys would serve. Eventually, the manufacturing plant could develop solar power generating capabilities for the outpost, even bootstrapping the productivity of the base. This near-term benefit would perhaps justify the effort in itself, and would act as the technology demonstration for larger scale manufacture for a space solar power program.

9.5.2 In-Space Manufacturing

The potential for in-space manufacturing is enormous. However, a number of scientific, economic, political, and structural problems need to be solved before this potential can be realized in commercial projects. Manufacturing operations in the micro-gravity conditions of space, rather than in the low gravity of the Moon, will provide their own set of benefits and limitations. Containerless processing of samples in space for the production of unique glasses from materials that are reluctant glass formers, fabrication of unique shapes and configurations without sagging or physical contact, such as concentric glass shells, production of ultrapure glass for use in optical wave guides, etc are good examples or advantages of micro-gravity manufacturing. Perhaps only physical processing of materials should be planned for the in-space segment, leaving the chemical processing to a lunar base, asteroid base, or even Earth. The wake shield facility, to be launched later this year or early in 1993, is an example of an experiment which will utilize the near-perfect vacuum of space to advantage. It will investigate the use of molecular beam epitaxy to produce semi-conductor devices such as solar cells. GaAs is one of the most important III-V semiconductors, with uses ranging from microwave devices to solid state lasers. In microgravity the role of thermocapillary convection becomes appreciable therefore the deposition method considerably improves the performance of the materials. Superconducting compounds represent another class of materials with a potential for space manufacturing. These may open up new technological possibilities and would find numerous applications in various types of space power systems.

Other products which might be enabled by the conditions found in space might include large, thin film structures and foamed metal beams. By manufacturing satellite structural components in orbit, it is possible that less mass will be required because launch loads and the special mechanisms required for deployment will be obviated. However, manned deployement of these structures could be a problem. On-orbit microgravity manufacturing tests which aim at developing technologies useful in the manufacturing of useful materials or devices and assembly studies should continue on Shuttle, Mir, and unmanned flights. Of course, all of the processes described above need to be traded against the mass required in-space and on the lunar surface to produce these products.

9.5.3 Schedule Issues for Space Manufacturing Technology

Certainly, our ability to manufacture items in space is very low. This program is not the only one in which manufacturing methods will be required, though. The maintenance and expansion of a lunar outpost will be enabled by the use of these technologies. The possibility of producing specialty electronics, nanotechnology products, or biotechnology products is another avenue for expansion of these technologies into space. One can see, then, that the development of technologies related to space manufacturing has a schedule of its own which interacts with many programs. Figures 9.15 portrays the tasks for in-space manufacturing programs related to space solar power program and other projects. Milestones in any one of these programs can be considered advances for all of them.

Programs and milestones which are important for manufacturing in space include:

- First flight of the Wake Shield Facility (WSF) to explore semiconductor growth by molecular beam epitaxial growth. Aim at specialty electronic market, but demonstrate photovoltaic device manufacture.
- Perform trade studies to explore the optimum location to perform each manufacturing step for SPS construction, and begin paper studies related to these manufacturing techniques.

- Consider CVD production of large optics using the free vacuum in space. This could, perhaps, be part of a solar dynamic generator production demonstration.
- Demonstrate the ability to produce parts useful for the construction of large structures in space. Perhaps foamed metal beam manufacture could be demonstrated. This would potentially extend the length of a beam which can be produced from a given mass of metal, providing less massive structures and lowering launch requirements. This can be done with ingots of Al or Fe from Earth, or it could be coupled with the next milestone to recycle external tanks.
- Demonstrate "cannibalization" of an external tank on-orbit. Perhaps use the metal in the production of beams as per above.
- Consider the manufacture of an integrated solar panel on-orbit, using microgravity-enabled thin film substrate manufacturing, structural beam production, and thin film photovoltaic deposition (perhaps with terrestrial semiconductor material) using the space environment to advantage. Can this be provided to a Spacecraft?
- Demonstrate the extraction of metal(s) from the waste material of oxygen production, first in the laboratory, then at a lunar outpost.
- Demonstrate glass fabrication technology from lunar regolith. Since this is a physical processing method only, it should be fairly simple. Use this to provide low tech material for a lunar outpost, such as thermal blankets, sun shades, blast protection walls, foundations for structures, etc.
- Develop lunar metal-based manufacturing capabilities once lunar chemical processing has proven to be feasible. At first, this can help maintain the outpost. Later, it can provide refined material to a Solar power satellite manufacturing site.

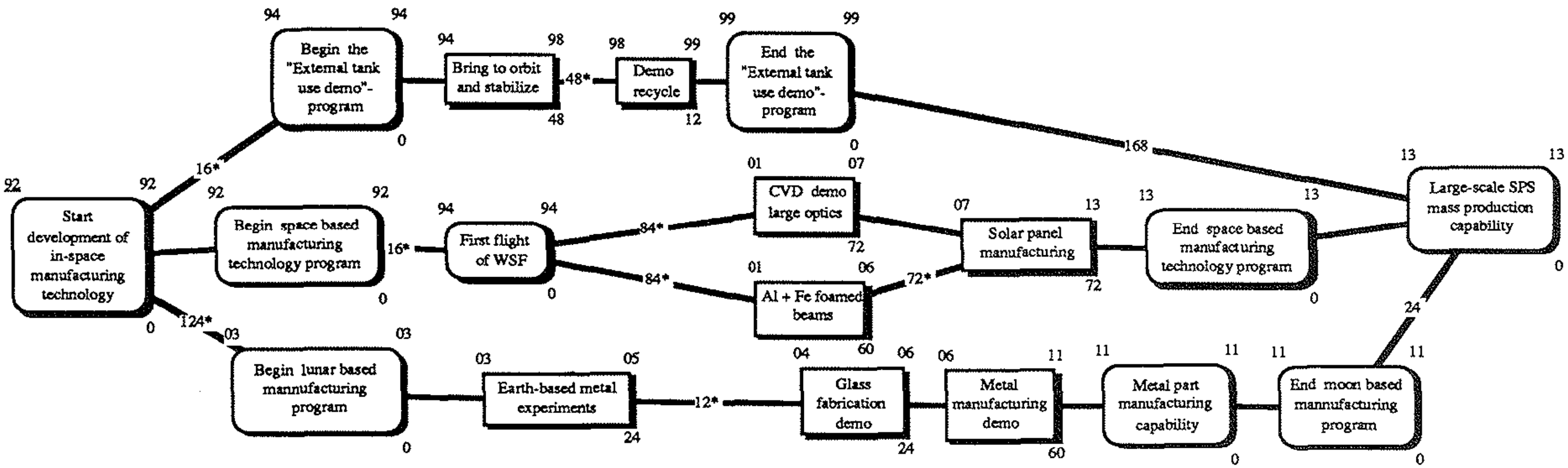

Figure 9.15 In-Space Manufacturing Task Schedule

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10 Design Examples

10.1 Near-Term Earth to Space

When designing a program to systematically demonstrate the technologies necessary to achieve a long-term, expensive goal, a variety of different factors must be taken into account. Each step should be achievable with a minimum of new or untested equipment, should be as cheap as possible while still demonstrating the necessary technologies, and should point the way to the next step in the program. To design such a program, two essential questions must first be answered. First of all, what technologies need to be demonstrated? Secondly, where do we start?

As has been described in earlier sections, there are several technologies which need to be demonstrated at the present time. Point to point microwave power transmission and reception has already been demonstrated on Earth several times. Solar collection technologies in space have been demonstrated on the tens of kilowatts scale, and early demonstrations of assembly in space will be demonstrated in the next few years with the assembly of Space Station Freedom. Things which have *not* been demonstrated include microwave power transmission over large distances, microwave transmission at high power levels, reception of microwave power in space, and transmission effects of beaming microwaves *through* the atmosphere at high power levels. The first steps toward demonstrating several of these essential technologies were taken in 1977 with the Ionosphere/Microwave Beam Interaction Study undertaken using the facilities available at the Arecibo Observatory in Puerto Rico. In the 1977 study, however, power was beamed into the ionosphere, not through it. [Duncan, 77]

The above considerations, when taken together, suggest a possible demonstration which could be conducted within the next few years. By using the Arecibo facilities or one of the large military phased-array radars to beam power to an orbiting receiver, several technologies essential to the continued progress of the space solar power concept could be tested at minimum expense and on a rapid schedule. Such a test would demonstrate both the atmospheric penetration necessary for space to Earth power beaming and the rectenna technologies necessary for space to space beaming, as well as providing an example of microwave beaming at high power levels. See section 10.3.5 for a detailed discussion of microwave beaming effects on the atmosphere.

10.1.1 Facilities

As a first iteration, let's take a look at what could be done with the Arecibo facilities. Arecibo has two large radar systems suitable for transmission demonstrations and a third system which can be used for ionospheric heating experiments but is not really suitable for power transmission. The first of these radars, which transmits at 430 MHz, has a peak power of about 2 MW, a 6% duty cycle, and an antenna with 61.5 db of gain. The second radar, which may be of most interest for power beaming, transmits at a frequency of 2.38 GHz with a power level of 400 kW and a continuous duty cycle. It's antenna has a gain of 71 db. The 2.38 GHz radar at Arecibo is probably the most powerful continuous wave system on Earth at the present time. [Sulzer, 92]

The facility itself centers around a dish 300 meters in diameter, again the largest in the world. Due to its large size, and the fact that it takes up an entire small valley as shown in Figure 10.1.1, the dish itself cannot be moved. Pointing is achieved by moving around the transmitting antenna which hangs over the dish suspended by a network of cables. This imposes two effective limitations on the use of the facility. The first of these is that it can point at most 20° from zenith. This is a fairly restrictive limit on the area of the sky which can be covered, but it should be noted that Arecibo itself lies at a lattitude of 18.3°, and so some equatorial orbits could be covered. The transmitters focus at infinity only, but this should not be a significant problem. The major problem with the facility is that it cannot track quickly; it can follow planets but not satellites.

All of these numbers would seem to make Arecibo an ideal facility for the type of demonstration envisioned except for the problem with tracking. Figure 10.1.2 shows the tracking rate necessary for satellites at various orbital heights. For a satellite moving at about 8 km/sec in a 1000 km orbit, tracking on the order of 0.5°/sec is necessary. Two ideas were considered for ways of getting around Arecibo's tracking limitations.

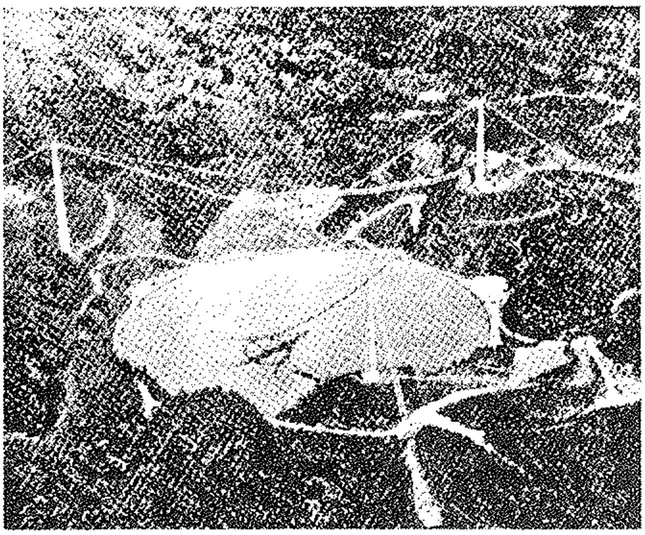

Figure 10.1.1 Arecibo Observatory, Puerto Rico [Ostro, 89]

The first of these makes use of the fact that Arecibo has two different modes in which it can move its antenna. One of these is a finely controlled motion used for tracking planets and other celestial bodies; the other is a slew mode used for rapid, but less controlled adjustments to pointing. By transmitting while in the slew mode, it was thought that it might be possible to track the satellite as it went overhead. Unfortunately, the slew rate is not very fast: it can move at 24°/min in the azimuthal direction but only 1.85°/min across the zenith. This is not fast enough to track a satellite in a low orbit, though it might be used for one at 5,000 km altitude or greater.

Another possibility suggested by Brian Tillotson is to set up a dual mirror system underneath the transmitting antenna. By sliding the second mirror along in a preset direction it might be possible to shift the effective pointing angle rapidly and track a satellite's orbit. It must be emphasized here that both of the above ideas are very rough; it could be that neither of them is feasible, or that some other method could be used to allow Arecibo to track more quickly, or that it can't be done at all.

In the latter, not unlikely case, one immediately thinks of the large military phased array radar systems which have been used over the last decade to track ballistic missiles for an early warning system. Though not as large as the Arecibo facilities, these radar systems have excellent tracking capabilities and can operate (perhaps only in pulsed mode) at high power levels. Examples of this kind of military radar include the Alaskan PAVE-PAWS, the Thule system in Greenland, the installation at Kwajelein in the Pacific, and new NATO radar systems in Portugal, Italy, and Greece. Corresponding Russian radar systems exist, though the most promising candidate seems to have been destroyed as part of a US-FSU arms reduction pact just a few months ago. Unfortunately much of the information necessary to evaluate these systems is still classified, so it is hard to judge their suitability for this particular activity.

One major advantage of using such a system for this sort of experiment is political in nature. The public relations potential of using a military facility to conduct experiments aimed at providing clean, safe future power sources for the developed and developing countries would be immense. Leaders around the world are looking for ways to turn their military investments to peaceful uses, in effect turning swords into plowshares. The military would likely be enthusiastic about such an idea as well, if only as a way of justifying the maintenance of their own facilities. Finally, the people of many Western countries who have long been looking for some sort of peace dividend from the end of the Cold War would probably embrace the project as well. Any positive publicity associated with the demonstration would improve the overall image of space solar power, and that is certainly worth pursuing.

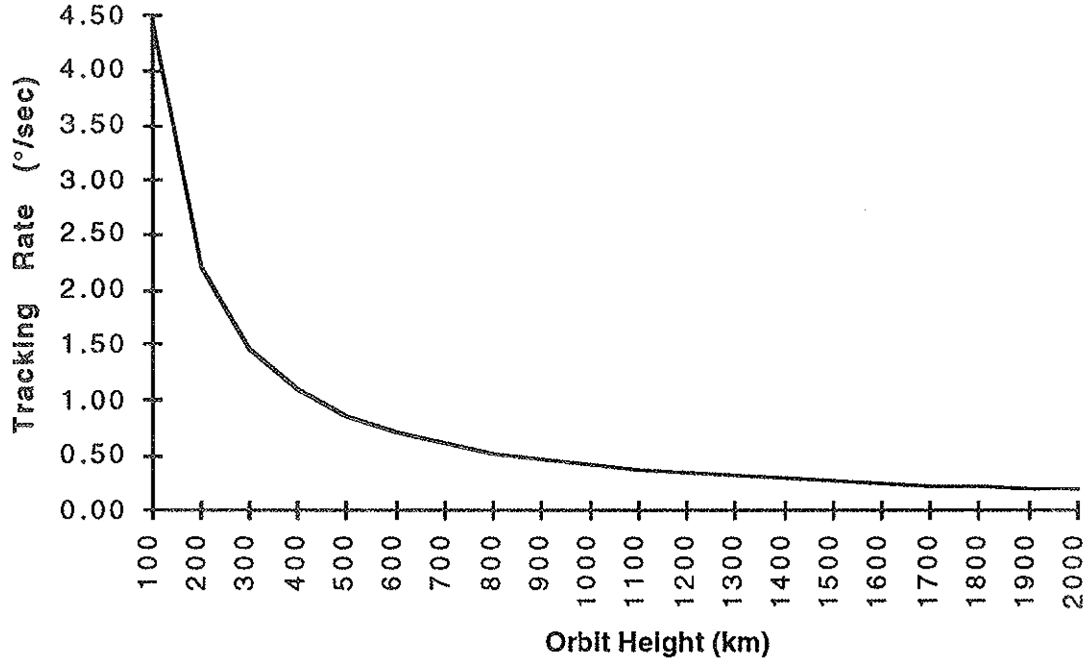

Figure 10.1.2 Tracking rates for Satellites at Various Altitudes.

10.1.2 Orbital Considerations

Totally aside from the issue of what ground facilities could be used, it is worth remembering that there are two parts to any beamed power demonstration: a transmitter and a receiver. In the case of this particular demonstration, the receiver will be a small spacecraft. To determine the characteristics and cost of the receiving spacecraft, it is necessary to take a look at the impact of various orbit choices on the mission.

On first consideration, taking into account the limitations of the Arecibo facilities, one would desire either a geostationary, sun-synchronous, or low equatorial orbit. A GEO orbit would overcome the tracking problem, but unfortunately the power received by any reasonably sized rectenna would only be on the order of milliwatts. As this power level is probably inadequate for the demonstration purposes, a lower orbit, likely under 2000 km, is called for. The advantage of a sun-synchronous orbit would be that it would pass over Arecibo at the same time each day, and so ground station scheduling would be much easier. The difficulty involved in this type of orbit would be the high ΔV required for orbital insertion, which would result in higher costs. An equatorial orbit might then be considered, which would allow a satellite to pass within Arecibo's arc several times a day. Figure 10.1.3 shows the problem with this concept. Arecibo can only see objects along the equator if they are over 70,000 km away.

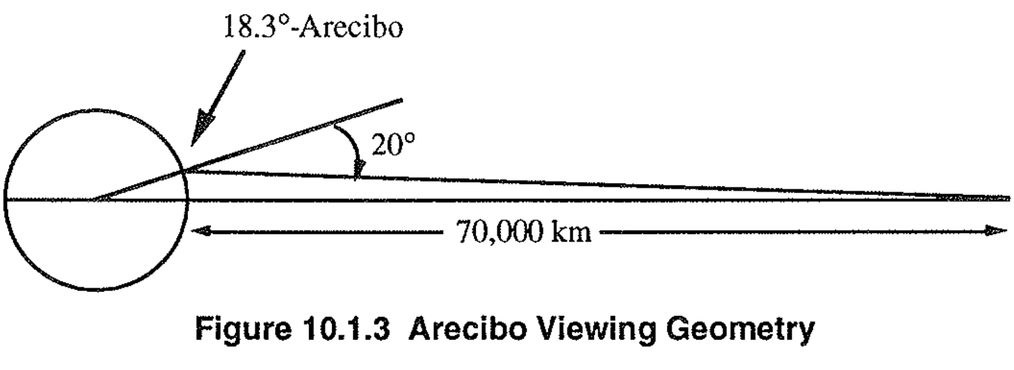

Figure 10.1.3 Arecibo Viewing Geometry

Knowing that low equatorial orbits are inaccessible to the ground station and sun-synchronous orbits are relatively inaccessible to the spacecraft, a compromise must be made. A synchronous orbit is desired with inclination over Arecibo's 18.3°and accessible with minimal ΔV. The orbit need not pass over Arecibo every day, but as much regularity as possible would be desired for scheduling and facilitating possible tracking systems modifications. The exact orbital parameters depend on the results of several trade-offs described below, but the height and inclination should be calculated with a view toward making the orbital period (taking into account the regression of the nodes) an integral fraction of a sidereal day.

The choice of orbital height of the satellite is closely related to the specific objectives of its mission. As shown in Figure 10.1.4 below, as orbital height increases, the time that the satellite will be in the beam increases while the power received decreases. A second trade-off is made when deciding which of the radars to use. If the 430 MHz beam is used instead of the 2.38 GHz beam, the amount of power received decreases, but the time in the beam is increased by about a factor of six.

Several other factors help to determine the orbit selected for the spacecraft. For a small satellite with a relatively large collector such as the one considered here, drag is a very significant parameter in choosing the orbital height. In an orbit up to 400-500 km, the satellite's lifetime would be relatively short. For altitudes of 700 km or greater, drag ceases to be a significant problem and larger structures can be employed. Another factor which tends to push the satellite toward a higher altitude is that drag forces at lower altitudes tend to disrupt attempts to use gravity gradient stabilization. Finally, one must consider the issue of orbital debris, perhaps a serious problem if an inflatable reflector as described below is to be used. From this perspective, an orbit of around 1200 km would be desirable. A high altitude such as this would also giving the advantage of having a transit time through the 430 MHz beam of a little over a second.

Unfortunately, the trade-offs described above are largely irrelevant for a satellite demonstration aimed at costs under $10 million. To launch a satellite within that sort of budget there are only a few possible vehicles. Of these, the major candidates would appear to be the space shuttle launched "Get-Away Special" (GAS) and the Ariane ASAP ring. As both of these systems are severely weight and volume restrained, carrying any sort of propulsion system would drastically reduce the usable system mass.

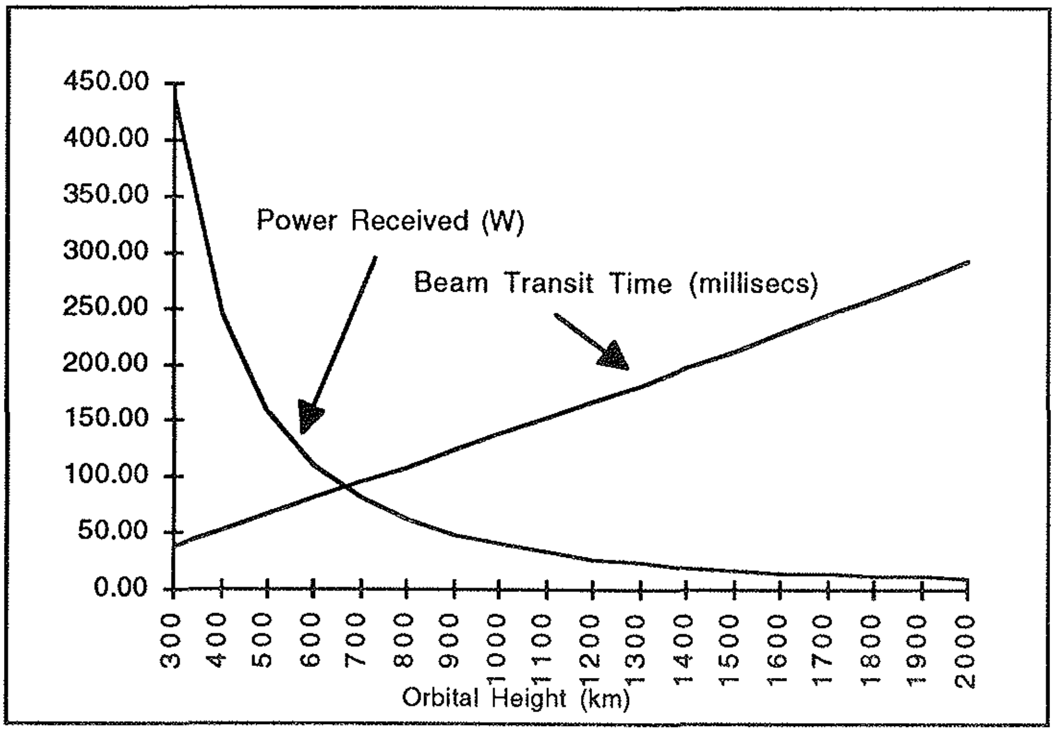

Figure 10.1.4 Trade-offs for Altitude Choice. Calculations are for 10 m Diameter Collecting Area Receiving Power from the 2.38 GHz, 400 KW Arecibo Radar

Thus the orbit of the satellite is basically determined by the orbit of the main spacecraft. In the case of Ariane ASAP launches, it is possible to achieve a sun-synchronous orbit, but not one tailored to pass over the proper spot each day. In the case of the GAS, the orbit is that of the shuttle. Figure 10.1.5 shows the frequency with which typical orbits of these types might pass within view of

Arecibo. Looking at these orbits, and noting that for the ERS-1 example the altitude is 785 km, it seems clear that an ASAP launch is dictated.

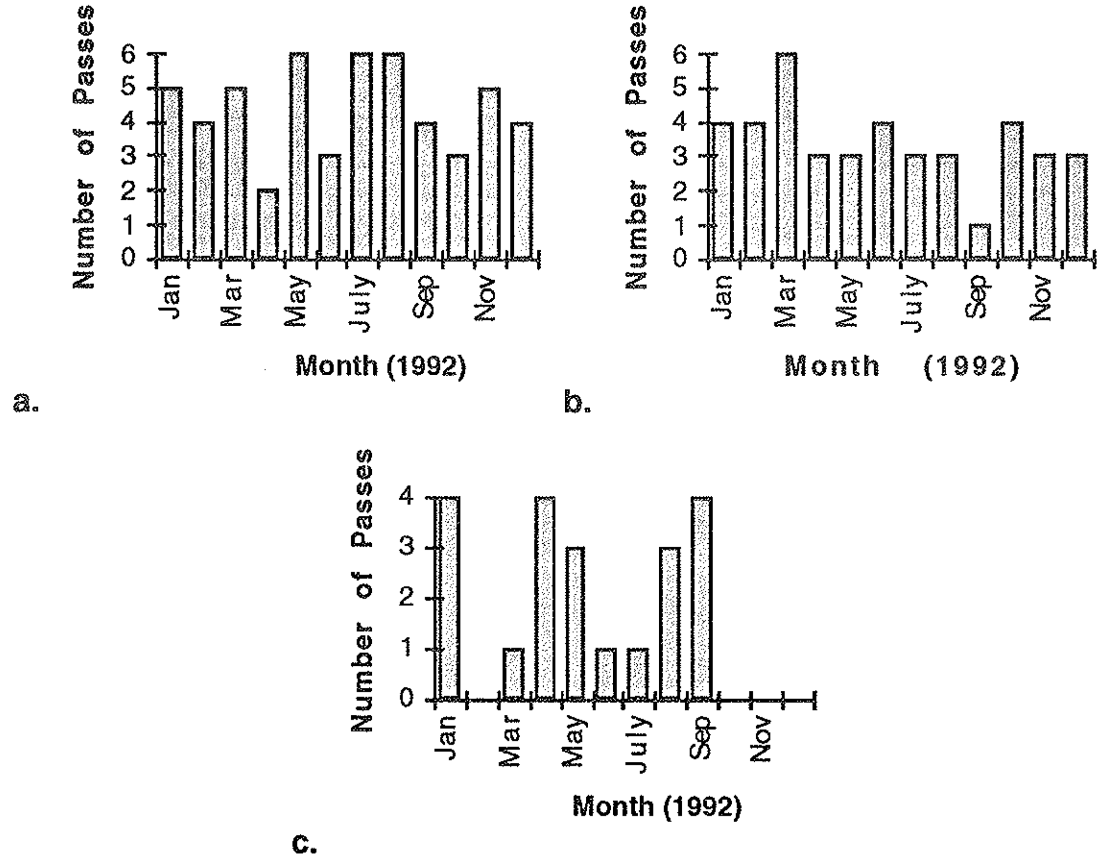

Figure 10.1.5 Number of Passages over Arecibo by a) ERS-1 b) Spot-2 and c) STS-46 Orbits During the Year 1992

10.1.3 Mission Objectives

To choose a proper vehicle configuration within the limitations of such a small spacecraft we must decide exactly what we want the demonstration to achieve within the overall space solar power program. To do this, three major aspects of the demonstration's mission must be considered: collection of scientific data, demonstration of the reception of useful power levels, and publicity.

One possibility is to orient the mission mainly around low cost and collection of scientific data. Here the main objective would be collection of data regarding beam scattering, sidelobe strength, frequency dispersion, atmospheric absorption under different weather conditions, and rectenna efficiency. Excellent information would be gained about rectenna performance over an extended period in the hostile conditions of the orbital environment. This type of mission could probably generate useful scientific data even if the power received was only on the order of milliwatts. However, the lower the power levels received, the more tenuous the connection with the high level power beaming which the experiment is supposed to model.

Keeping that in mind, one should remember the objective of showing that beamed power can be received over long distances at power levels high enough to be useful. The problems with reception of milliwatt power levels have already been discussed. With a single watt of received power, a small transponder could be powered. If power on the order of 10-100 watts was received, a small light bulb could be operated for the time the satellite was in the beam. If the step was made to kilowatts of power received, almost any space system now in operation could be powered.

The third mission objective that must be addressed is that of publicity. Engineering and science aside, people around the world love a good show. The more visible a demonstration is, the more likely it is to be funded. For instance, on the single watt level, a transponder could send out a "beep" or Morse-code signal saying something like "power received" strongly enough for amateur radio operators to pick up. On the next higher power level, a light bulb could serve as a flash bright enough to illuminate a printed logo ("Eat at Joe's," for instance) long enough for a photograph to be taken. At the kilowatt level (using a dish about 100 m in diameter), given a capacitor and some fluorescent paint, a sign could be set to flash on and off and could even be visible from Earth with a small telescope. It has been pointed out that such a demonstration might not be popular with astronomers, but the controversy itself would be great publicity.

The determining factor for all of these possibilities is, of course, the area of the microwave collector. The equipment necessary for monitoring received power levels and frequency dispersion and transmitting the data back to Earth can easily be made compact and lightweight, so it does not impose any real restrictions. For reception on the order of a single watt, the rectenna need only be about 2 meters square. Such a rectenna could be easily deployed from a small package using existing, well-tested technology. To get the larger receiver area necessary for demonstrations on the 100 watt scale, inflatable technology as described below would probably have to be used. Reception on the kilowatt scale with a collector deployed from a microsatellite is probably not feasible in the near term.

10.1.4 Vehicle Configuration

The choice of an ASAP platform puts rather stringent restrictions on the size, shape, and mass of the receiving satellite. The ASAP ring lies on top of the Ariane H10 upper stage, and is capable of carrying up to six separate payloads of up to 50 kg each. Each of these positions can accommodate a payload with dimensions equal to a 45 cm cube, though exceptions are sometimes made allowing the payload's height to be up to 60 cm or so depending on the nature of the mission's main payload. Individual payloads on the ring can be connected to each other by wires. [Arianespace, 1990]

Working within the above constraints, a two-section spacecraft is envisioned. One position on the ASAP platform would be taken up by the main satellite equipment, including sensors, data-handling equipment, a transponder for data transmission, an independent power supply, and whatever else is necessary for the demonstration chosen. The other position would be connected to the main satellite bus by wires strung along the ASAP ring, and would be used to house an inflatable reflector. When deployed, the spacecraft would look something like the one shown in Figure 10.1.6 below.

The inflatable reflector would be transparent on one side with a reflective parabolic inner surface. Connected by wires to the main bus about 12 m away, the satellite would be gravity-gradient stabilized and would always point toward Earth. Microwaves transmitted from Arecibo would be collected by the reflector and focused on a small rectenna on the surface of the main spacecraft.

Mass restrictions should not be a problem; designs for a 10 m inflatable rigidizable antenna for the QUSAT VLBI mission quote a total mass of 42.05 kg, which includes the main chamber torus, pressurization subsystem, and stowage elements. [Bernasconi, 1984] As the reflector is being used to collect rather than transmit, much less accuracy is needed in manufacturing, so it may be possible to fit an even larger structure within the mass limits. However, detailed numbers are not available for the volume of such an inflatable in its stowed configuration. A simple calculation does indicate that at the density of mylar, a 42 kg payload would take up only about a third of the volume of an ASAP fairing. This would leave two-thirds of the volume for consideration of packing constraints and storage of the inflation gas.

Table 10.1.1 shows the relevant statistics for such a satellite flying in the 785 km ERS-1 orbit. Power received averages about 60 W over the diameter of the first minimum for 2.38 GHz, and 10 W for the 430 MHz radar. Times of passage are about 0.1 and 0.6 seconds, respectively. The 60 W figure is enough to power a photograph flash as described above. A corporate or departmental logo could be printed on the outer surface of the reflector with something transparent to microwaves but opaque to visible light, and bids could be taken from various companies to use their logo. It is conceivable that the entire project could be funded by taking the offer of the highest bidder; it might even make a profit.

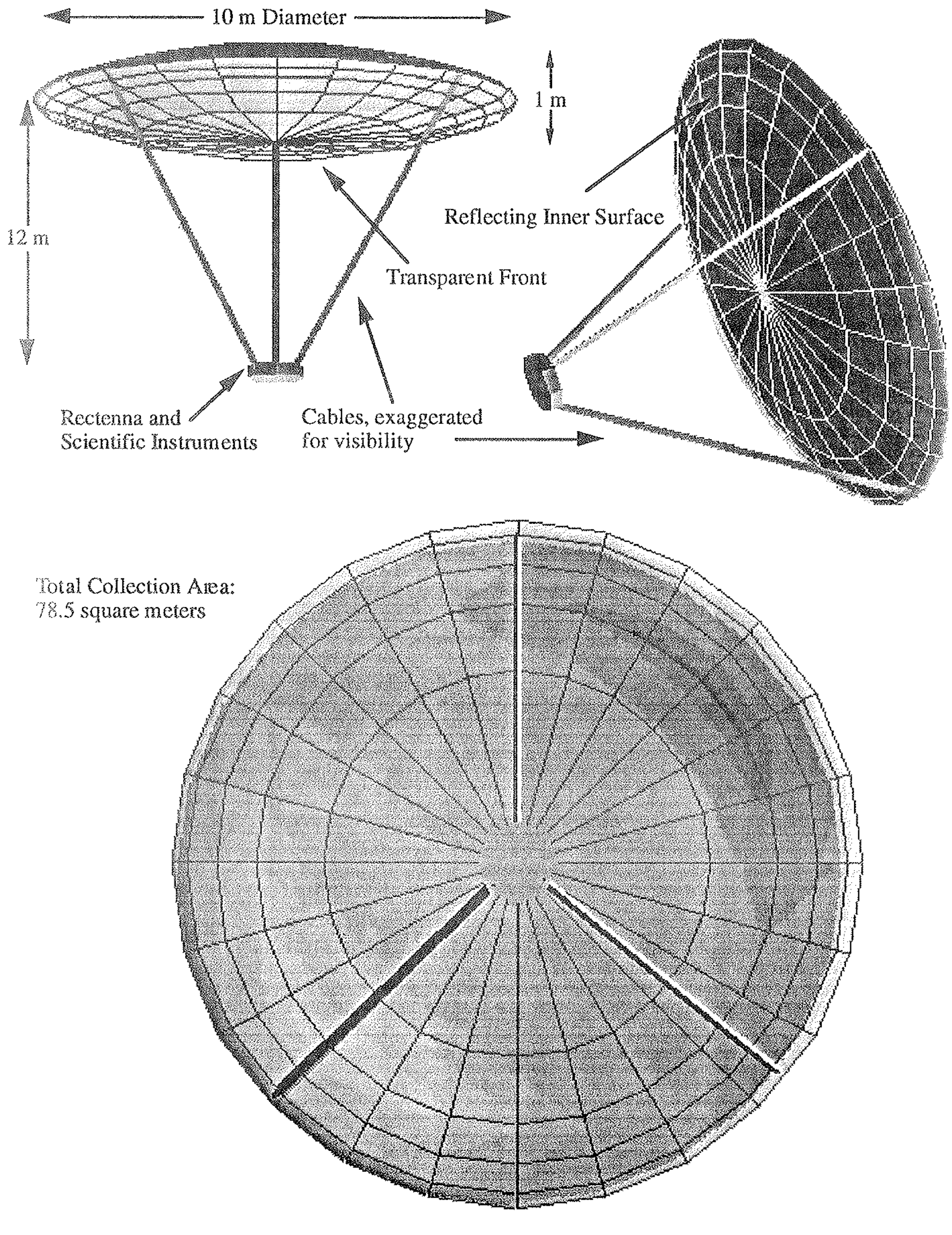

Figure 10.1.6 Inflatable Receiver Concept

Table 10.1.1 Preliminary Calculations for Transmission from Arecibo

Beaming Power from Arecibo		
Power (W)	4.00E+05	2.00E+06
Frequency (Hz)	2.38E+09	4.30E+08
Wavelength (m)	0.126	0.698
Aperture Diameter (m)	305	305
Orbit Height (km)	785	785
Diameter of First Min	792	4381
Power Density (W/m2)	0.81	0.13
Diameter of Reflector (m)	10	10
Area of Reflector (sq m)	78.5	78.5
Power Out (W)	64	10
Time to pass through (s)	0.11	0.59
Tracking Speed (deg/s)	0.54	0.54

10.1.4 Program Costs

The objective of this design example was to demonstrate trans-atmospheric microwave beaming as well as microwave reception in space for under $10 million. Program costs would include satellite construction, satellite launch, use of the Arecibo radar systems, ground station operations for data reception from the satellite, data processing costs, and staff salaries.

The launch costs for an entire ASAP ring of 4 or 6 microsats have been quoted at around $1 million. These costs are somewhat variable, and universities have been known to get much cheaper launches. So one can guess that the cost to launch a satellite taking two of the positions would be around $500,000. Fabrication of the inflatable should not be too expensive as it need not be manufactured to the high degree of accuracy necessary for a transmitting antenna. Actual satellite equipment is minimal: just a small rectenna, power measuring equipment, a small transmitter, camera, and power supply. It would not seem unreasonable to place the total satellite costs at about $1 million.

Operating costs for Arecibo would not be high; with an average encounter rate of 5 passes/month over a mission life of about two years, 120 data collection runs could be made. According to Mike Sulzer of Arecibo Observatory,

> It is a little hard for me to estimate the cost of running the 2380 MHz radar since we do not charge (although we do recover certain exceptional expenses in infrequent cases). I would guess several thousand dollars an hour if you need a number. It really is not practical to think about seconds or minutes; you need at least two hours to get things set up and going. [Sulzer, 92]

So at $5,000/hour for 120 2-hour periods, the cost would be at worst $1.2 million, and at best would be free. With another couple of hundred thousand dollars thrown in for ground station costs and staff salaries, the total program cost might come to about $3.5-4 million. So even if the cost of producing the inflatable was *much* greater than expected, the program would still be well under the $10 million target.

10.1.5 Time-Table

It is believed that this program could be carried out over a period of 5 years starting in 1993. Design would begin in 1993 and last throughout the year. Hardware development would begin in 1994, and actual manufacturing of the satellite systems in early 1995. Testing would be started in mid-1996 while manufacturing of some of the subsystems was still in progress. Transport of the finished spacecraft to the launch site would take place in the end of 1997 for a launch in early 1998.

Most of the necessary equipment would be fairly easy to design and construct; the only question would be procurement of the inflatable reflector. Meeting the 1998 or 1999 launch date would be essential, as the last ASAP launches are planned for sometime in that period, after which launches will be carried out mainly by the Ariane-5. An Ariane-5 ASAP ring is planned and could probably be used to launch a similar experiment (possibly even one of greater size), but the time-frame for such a system is presently unknown.

10.1.6 Alternative Possibilities

The design presented above was oriented at producing a quick and dirty solution to the problem. As an alternative, or a follow-up mission, it might be interesting to try a similar experiment on a slightly larger scale. Using a dedicated launch of a Pegasus or Delta to put a larger inflatable collector into a higher orbit has very interesting possibilities. Echo 1 and 2, launched in 1960 and 1964, were aluminized mylar balloons of 30.5 and 40 meters in diameter, respectively. They were the first man-made objects in space to be visible from Earth with the naked eye. Over thirty years later it should surely be possible to produce an inflatable reflector of 100 or even 200 meters in diameter. Such a large satellite could receive tens of kilowatts of power in a relatively debris-free 1200 km polar orbit and really give some interesting insights into the problems of transmitting and receiving high power levels. Or if distance beaming or long-term continuous beaming was the technology that needed to be demonstrated, such a large structure could receive significant levels of power even at GEO.

At the other end of the spectrum, it has been suggested that some military satellites for electronic eavesdropping may already have on board equipment capable of receiving and measuring incident microwave power. A program using these existing assets instead of requiring launch of a separate satellite would cost next to nothing.

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10.2 Space to Space Demonstration

Several concepts have been proposed for a near-term demonstration project within a total budget of $80 million (see Appendix A for a complete list). The work group for this near term demonstrator had a closer look at the following projects:

A. Beaming microwave power from a Viking-like platform (Swedish platform designed into the Ariane 3rd stage adapter). In a first experiment the target could be a rectenna mounted on the Ariane 3rd stage (Ariane Structure for Auxiliary Payloads (ASAP). The second experiment could be to beam power from the Viking platform to an experimental rectenna in Arctic regions.

B. Beaming laser power to a satellite in GEO or the GEO graveyard orbit.

C. Beaming microwave power from the Mir space station to the Progress service vehicle.

Proposal A has been discarded due to severe orbital mechanics constraints when using the spinning Viking platform. The experiments would have needed a permanent change of the spacecraft's spin axis thus leading to an excessive propellant demand. A redesign of the Viking platform to use another stabilization principle was not considered to fit into the cost and schedule constraints. The quite short experiment duration in the range of some minutes due to visibility constraints would have been another problem.

The second proposal (B) has been abandoned mainly because of technology and budget problems: no laser has been found to be compatible with the requirements of wavelength (to use existing solar arrays for back conversion), power and system mass. All considered laser design options exceeded the given envelopes in cost and project time significantly (see Appendix E for more details).

Proposal C has been retained. The experiment demonstrates key issues of microwave power beaming. It represents a major milestone with respect to the other design examples as well as to a long term power beaming application. This proposal will be presented and discussed in the following sections.

10.2.1 Mission Objectives

The aim of the mission is to beam power from the Mir space station to a Progress transport spacecraft over a distance of about 80 m and more. With this mission the following objectives can be met.

First of all the experiment would demonstrate in space a complete microwave power beaming system transmitting a significant amount of power. Since the mission time would be in the range of several days, the experiment would be the next step in terms of beaming time and transmission power compared with the proposed sub orbital demonstration projects such as the METS experiment. [Kaya, 1991] So an important feature of the experiment would be to demonstrate target acquisition and locking the beam on the target in various emitter/receptor constellations over a longer period of time.

The Progress vehicle as a beaming target could demonstrate the potential of a future microgravity laboratory without solar panels. This would possibly reduce weight and aerodynamic drag effects.

The mission incorporates a range of scientific experiments, chiefly investigating the nature of the interactions between the microwave beam and plasmas. This interaction is in two regimes. Firstly, the ambient medium through which the spacecraft travels will be analyzed and its characteristics when heated examined—ionization, thermal profile, energy distribution, charge and composition analysis. Secondly, the lower levels of the atmosphere will be examined while the beam is pointing toward the Earth. These experiments will be performed by an on-board mass (retarding potential) spectrometer and a multi-band electromagnetic spectrometer. Finally, a matrix of probes will investigate the surface charge characteristics of the beaming antenna, both while beaming and during passive phases of operation.

10.2.2 Mission Scenario

The Progress spacecraft will be launched by a Soyuz launcher into a circular orbit with a 51.6 degrees inclination, at an altitude of approximately 320 km. The Progress transport vehicle will carry all equipment for the experiment to the Mir station. The transmitting and receiving antennae are initially fitted to the Progress vehicle at the attachment points of the solar arrays, as shown in Figure 10.2.1.

When Progress has finished its servicing mission, it separates from the station and will be the beaming target for the experiment.

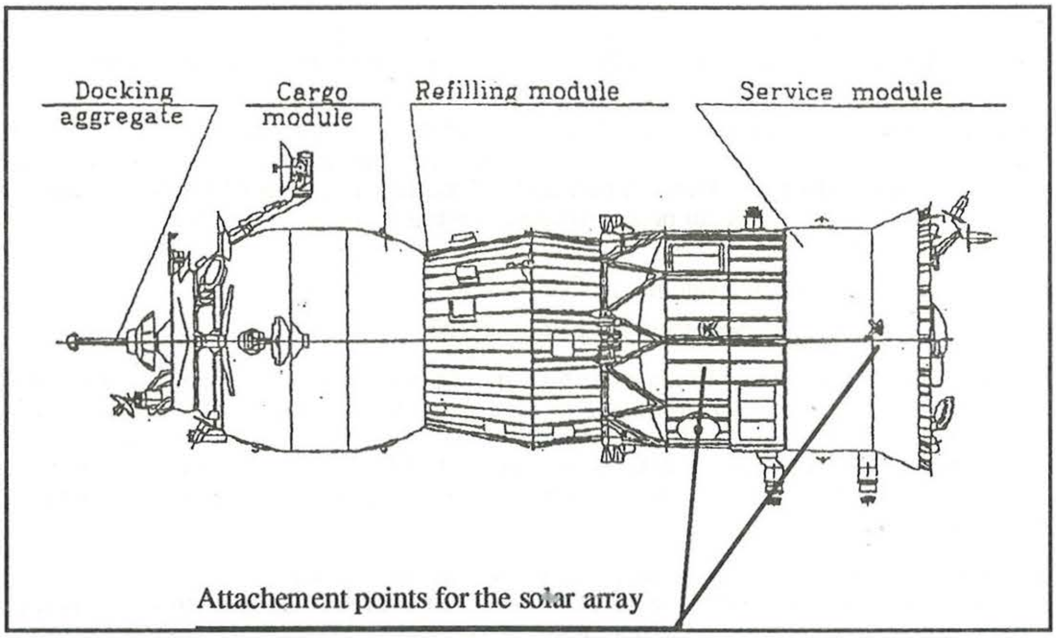

Figure 10.2.1 Progress Configuration

Prior to docking, the Mir Space Station is in a gravity gradient mode, with its minimum axis of inertia aligned with the local vertical in the nominal attitude orientation. During rendezvous, the Mir Station is rotated approximately 90 degrees in order to align its docking port with the approach velocity vector of the Progress spacecraft as shown in Figure 10.2.2. Upon completion of the docking operation, the station is rotated back to its nominal orientation. Subsequently, the nominal logistics mission is carried out over a period of approximately two weeks.

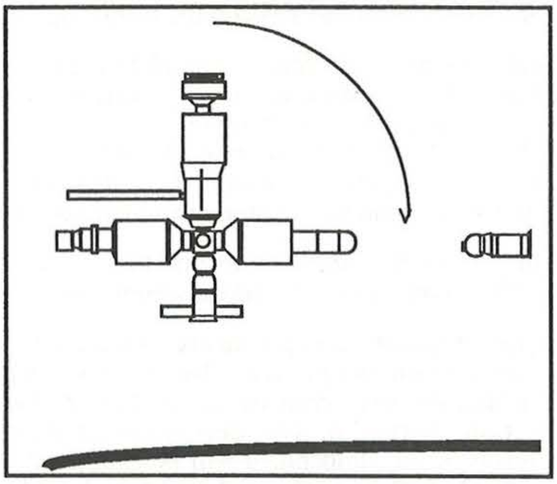

Figure 10.2.2 Mir-Progress Docking Configuration

During the docked phase, a set of procedures are adopted in order to prepare both spacecraft for the proposed experiment:

1. The transmitting antenna will be removed from Progress and installed on the Mir station. This operation will be performed by a cosmonaut EVA. It is proposed to attach the antenna to existing mechanical interfaces foreseen for additional solar arrays. This will ensure simple interfacing and shorter EVA times. It is envisaged that the overall task should be accomplished within one EVA working day.
2. The rectenna on the Progress will be deployed automatically.

3. Upon completion of the construction and deployment maneuvers, all the systems should be checked and tested.

In this preliminary design phase, two operational scenarios are considered:

1. In the flight vector configuration, the Mir Station is rotated 90° to align its minimum axis of inertia along the local horizontal. The Progress vehicle is released, and the relative position between the Mir Station and the Progress chaser vehicle is controlled by the rendezvous sensors shown in Figure 10.2.3. The Mir station should stay in this position for the time necessary to carry out the beaming experiments.
2. In the gravity gradient stabilized orientation, the Mir station returns to its nominal orientation after releasing the Progress vehicle on the same orbit.

In scenario 2 the demand on attitude control fuel would be minimal, however, there would be a need of a dedicated localization and pointing system. The advantage of scenario 1 is the permanent communication link between the two spacecraft, which allows for a high pointing accuracy and operational safety. That is why scenario 1 has been selected, however at a higher cost in fuel to maintain the Mir station in an unstable orientation.

After finishing the experiment and evaluating the data the Progress spacecraft will perform a controlled reentry and burn up in the atmosphere. Figure 10.2.4 shows the preliminary flight operation sequence.

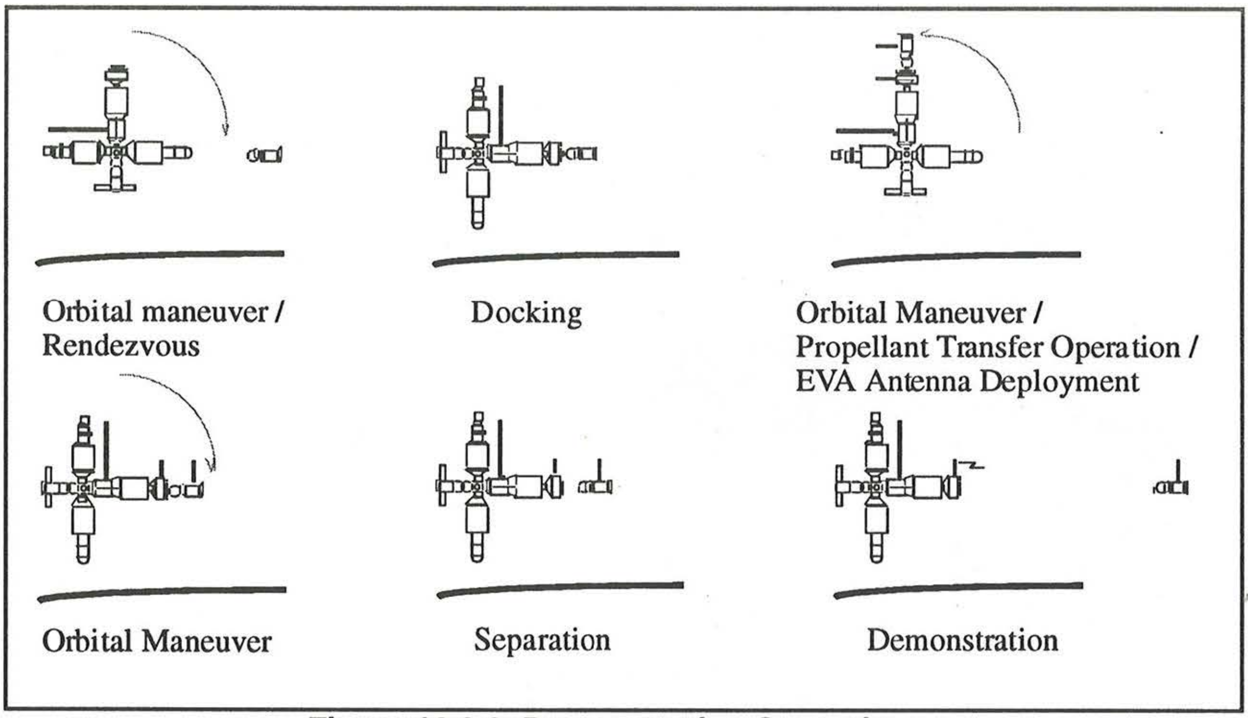

Figure 10.2.3 Demonstration Scenario

10.2.3 System Level Design

In this section the system elements are designed to estimate system performance, system budgets (mass, power and cost) and to reveal critical points in design, development and operations. Due to the limited amount of time the design studies were limited to a general system level. However, it is possible to conclude on the feasibility and to show some potential benefits of the proposed experiment.

The following mission constraints on the system have been identified:

1. *Payload mass and volume.* The Progress cargo module has a maximum usable volume of 7 m^3 and a maximum total payload mass of 1500 kg. As explained in the previous section, servicing Mir shall continue to be the main mission of the Progress spacecraft. Consequently the total mass and volume of the equipment for the power beaming experiment have to be significantly below the limits specified above.

2. *Payload dimensions.* If parts of the equipment are to be transported in the interior of the cargo module, their size will be limited by the cargo bay dimensions (2 m in diameter and 2.3 m in length) and the hatch diameter (0.8 m). For equipment mounted on the outside of the Progress spacecraft, the dynamic envelope of the Soyuz rocket fairing with 3 m diameter has to be respected.
3. *Orbit constraints.* The Mir space station is in an orbit of 320-330 km altitude with an inclination of 51.6 degrees.
4. *Power constraints.* The total solar array power of the Mir space station is about 25 kW. About 10 kW of power could be available continuously for periods of 1 hour. [Burgasov, 1992] The frequency of these periods will depend on the power load of the Mir station but several beaming experiments per day should be possible. The on-board voltage on Mir is 27 V.
5. *Time constraints.* The Mir space station will probably be operational only until 1996. Therefore the experiment should be carried out by mid 1996.
6. *Cost constraints.* The overall project budget should not exceed $80M.

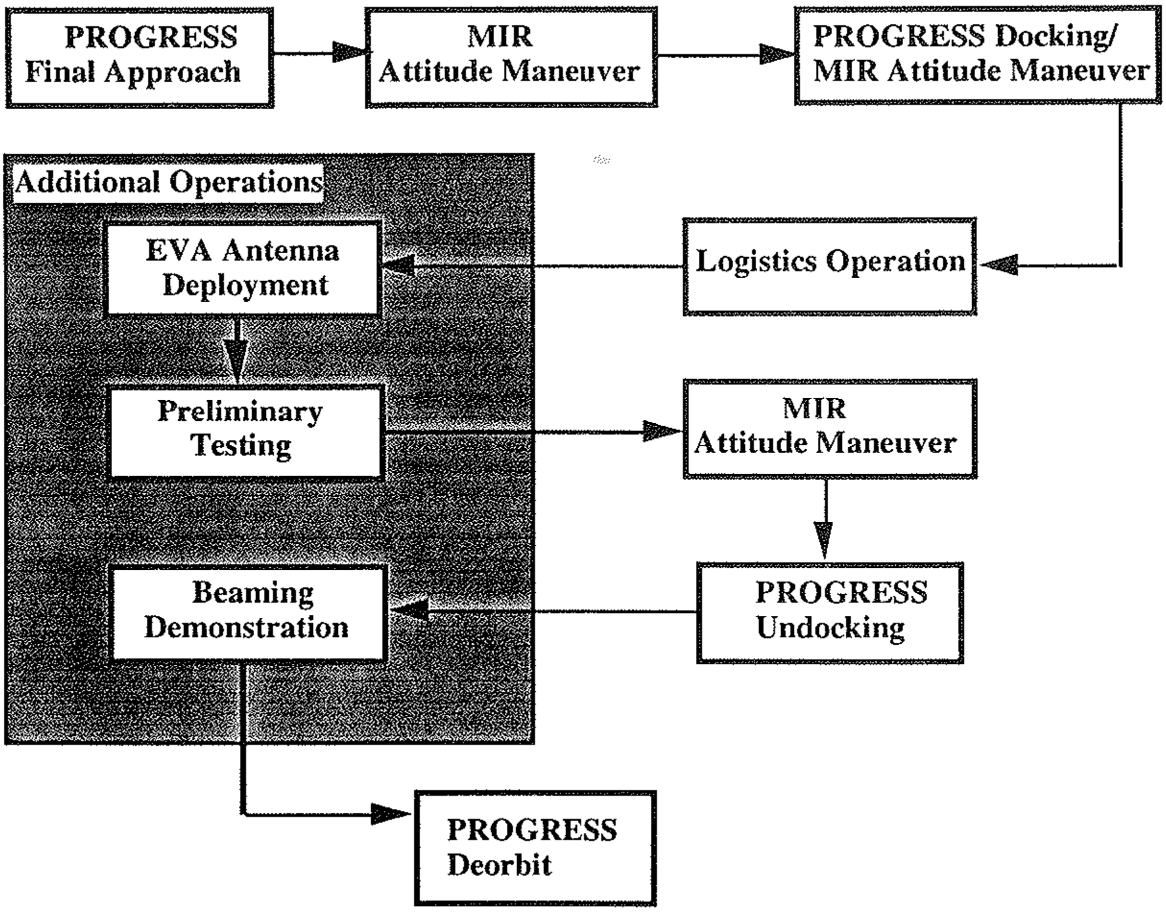

Figure 10.2.4 Flight Operation Flowchart

Power Beaming

Cost and time constraints are the main drivers for the choice of the power beaming system. It must be cheap, which means that only well-known technology should be used, and it must be realized in a very short time, which means that we may not undertake any long design study.

The proposed scenario should provide new knowledge about phased array technology, which probably will be used for future demonstrations of space to ground power transmission. A "large scale", "low" frequency phased array of 2.45 GHz has been chosen. In this frequency range, solid-state components are readily available. The size of the array is a trade-off between cost, feasibility and demonstration value. It must be large enough to be innovative without exceeding the cost and time constraints mentioned above. Some additional mechanical constraints arise as this payload must fit in a proper way in the available space of the launching vehicle. A square array with a size of 2 m by 2 m seems to fulfill all these requirements.

Phased arrays of this order of size already exist for these low frequency bands. INMARSAT-2 is equipped with a 1 m phased array operating at 1.6 GHz. Matra Marconi is developing an Advanced Synthetic Aperture Radar (ASAR) using a 10 m by 1 m phased array operating at 5.3 GHz. The power per element in these two applications is only slightly lower than the proposed power beaming experiment, giving confidence about the availability of solid-state components. As their linearity constraints could be relaxed for this application, it should be possible to slightly increase power.

The innovation of this proposed demonstration is the increased number of independently controlled elements, and the steerability of the beam. The accuracy requirements for the two applications mentioned above are not critical, so that a quite rough steerability was acceptable. On the contrary, the pointing accuracy requirements for future power beaming systems are very important, so that this demonstration should proceed in that direction. One of the results of this experiment will then be a better understanding of the technological requirements, trade-offs and limitations for future systems.

Phased Array General Characteristics

The following phased array general characteristics are shown in Figure 10.2.5:

- Frequency: 2.45 GHz
- Wavelength: 0.1224 m
- Array size: 2 m by 2 m
- Elements spacing: $\lambda/2 = 0.0612$ m
- Number of elements: 32 x 32 = 1024

A pointing accuracy better than 0.5 degrees, and a beam deflection angle of more than +/- 30 degrees should be achieved. The array will be designed in two parts for mechanical assembly purposes.

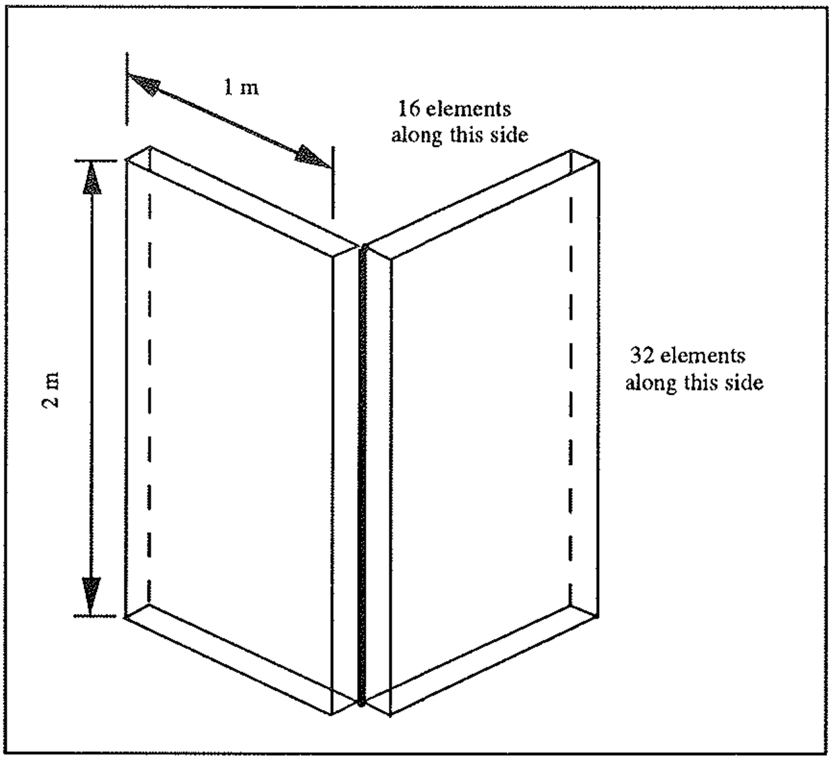

Figure 10.2.5 Phased Array Antenna.

Phased Array Electrical Characteristics

The word "segment" will refer to an element of the array and its electronic device which generates its signal. The generation of these various signals can be broken down in the following operations:

- A microwave reference signal must be generated with its phase considered to be the 0 degree reference phase.
- This signal must be split and distributed to each segment of the array.
- The phase of the signal arriving at each segment must be independently controlled over a range large enough to fulfill the steering requirements, and with an accuracy compatible with the tracking accuracy requirements.
- Each signal must be amplified to provide the overall power. The amplitude of each signal could also be controlled in order to maximize the flexibility in shaping the beam.

The usual structure of a segment is illustrated in Figure 10.2.6. A local oscillator (LO) provides the reference signal. This signal is split and distributed to each segment. A digital phase shifter is then used to adjust the phase of the signal. As a reference, the METS experiment [Kaya , 1991] provides a 4-bit phase control, and a pointing accuracy better than one degree. Hence, it seems feasible to improve these specifications as this technology is already a few years old.

After having adjusted the phase, the signal must be amplified up to a power of a few watts. This requires high efficiency and high power solid-state amplifiers. The phased array being developed by Matra Marconi uses solid-state amplifiers delivering the same order of power at 5.3 GHz, and with an efficiency of about 80%. To find an amplifier at 2.45 GHz should therefore not be a problem. The energy which is not transmitted is converted to heat, and raises the operating temperature. This problem must be addressed, and a key issue of the design would be to thermally connect all the power amplifiers to a sufficiently large radiating surface.

Finally, each element of the array is fed with the controlled signal coming from the amplifier. These elements are simple antennae. Various kinds of antennae could be used: dipoles, crossed dipoles, helices, micro strip antennae etc. The latter has been chosen for its low thickness. Some special micro strip (MS) designs have very interesting properties. For instance, some circular MS antennae radiate the fundamental frequency very efficiently, while the harmonics are blocked. Further investigations should be done before the choice of antenna is made.

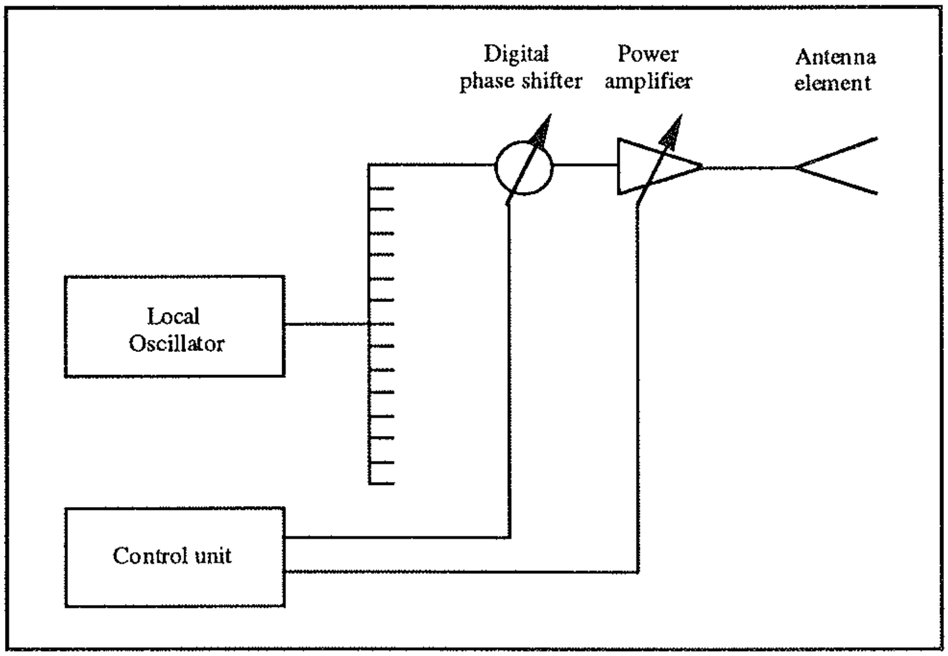

Figure 10.2.6 Phased Array Functional Diagram

Beam Control

The beam emitted by the phased array could be controlled using retrodirective phase control technique. [Kaya, 1991; Chernoff, 1980] The rectenna emits a pilot signal to the power transmission antenna. By measuring the phase of the incoming signal on the different elements, it is possible to determine with very high accuracy the necessary phase shifts to steer the power beam on the target. This technology has already been validated on a small scale phased array. [Dickinson, 1978] Unfortunately, this requires a quite complex microwave circuitry, which would probably be too expensive and would take too much time to develop, so that this pointing technique must be discarded for this demonstration. As this pointing technique is the only one able to correct any flatness inaccuracies, the mechanical flatness requirements are tightened.

An alternative would be to use a simpler design composed of four isotropic antennae placed in the center of the phased array as shown in Figure 10.2.7. An isotropic antenna placed in the center of the rectenna emits a pilot signal at a different frequency than the power transmission. The four elements of the rectenna receive signals which are carrying the direction information of the incoming signal in their relative phases. This information could be extracted by multiplying the signals two by two, in order to get the sine of their phase difference, as shown in Figure 10.2.8. The results are sent to the control unit, which computes the direction of the incoming signal. The correct phase commands can then be determined and applied to the phase shifters. Further investigations should be done to specify this system. In order to avoid any phase ambiguity, the antennae of the interferometer should be spaced less than half a wavelength apart (of the pilot signal). No distance measurement is needed as the pilot signal transmitter is in the far field of the interferometer.

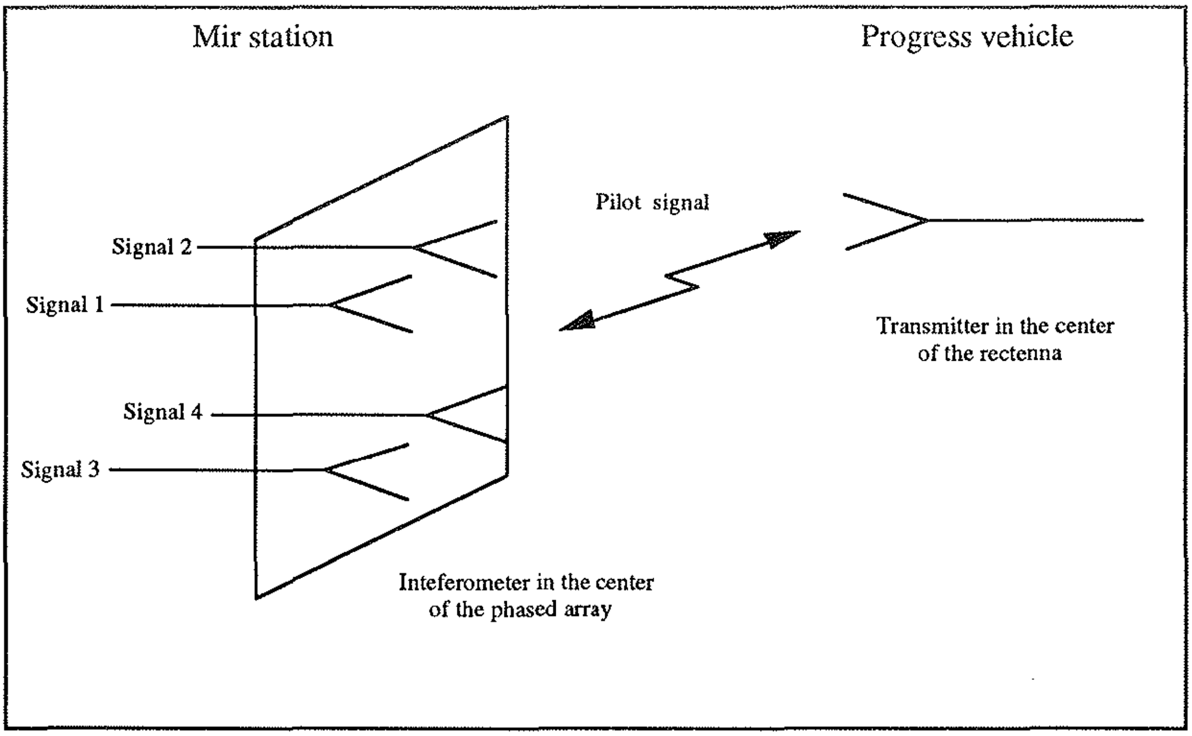

Figure 10.2.7 Antenna System Layout

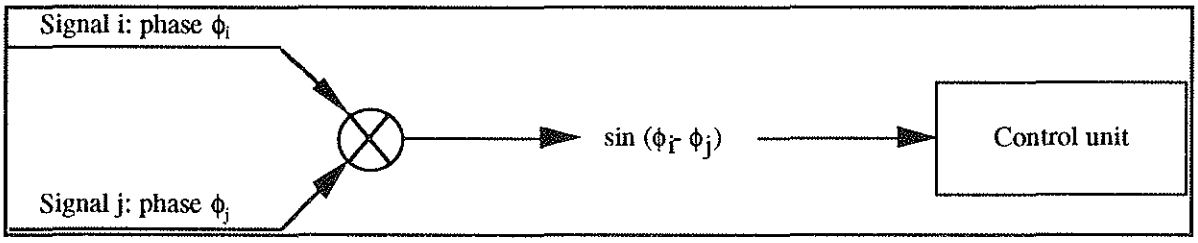

Figure 10.2.8 Phase Control

Transmission Efficiency

The transmission efficiency has been estimated as follows. It is assumed that the rectenna always stays within a cone with an opening angle which corresponds to the half gain angle θ_{3dB} as shown in Figure 10.2.9.

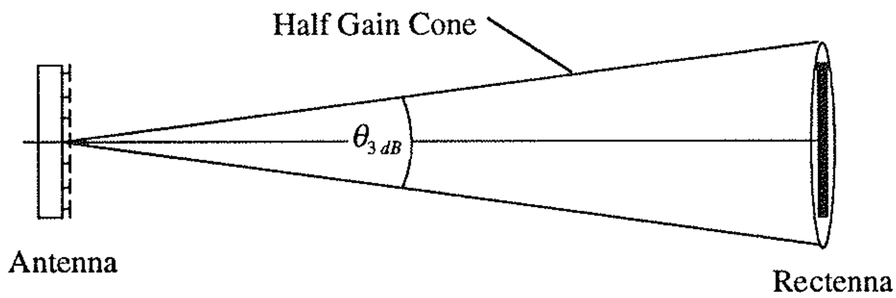

Figure 10.2.9 Beam Cone

This angle can be estimated using the following equation:

$$\theta_{3dB} = 0.88\frac{\lambda}{a} = 0.0539$$

where

θ_{3dB} is the half gain angle [rad],

λ is the wavelength = 0.122 m, and

a is the diameter of the transmission antenna = 2 m.

Thus the diameter of the spot on the target will be $D_s \approx \theta_{3dB}R$ = 4.3 m when taking into account the beaming distance R = 80 m.

We obtain the directivity D and the gain G of the antenna using the following relations:

$$D \approx \frac{4\pi}{\theta_{3dB}^{\ 2}} = 4330$$

$$G = \eta_a D = 3030$$

where antenna efficiency is η_a = 0.7.

According to the definition of the angle θ_{3dB}, the antenna gain on the surface of the half gain cone will be:

$$G_{3dB} = G/2 = 1515$$

Then we consider the power density on a rectenna within the distance R of an isotropic transmission antenna to be:

$$N_{is} = \frac{P_a}{4\pi R^2} = 1.24 \cdot 10^{-5}\, W/m^2,$$

where P_a is emitted power [W], here = 1 W

Finally we obtain the power density of the real antenna on the target on the surface of the half gain cone:

$$N_{3dB} = G_{3dB} N_{is} = 0.0188\, W/m^2$$

Considering a rectenna of 2x2 m^2, a power density of 0.0752 W/m^2 is obtained on the rectenna for each watt emitted by the transmitting antenna, which can be interpreted as a beaming efficiency of 7.5%. This value represents a worst case estimate. Actually the power density on the rectenna will be higher than on the surface of the half gain cone. Thus for an ideally pointed beam the efficiency could go up to about 15%.

Rectenna Characteristics

The main constraints placed upon the rectenna are a light weight, low cost (which will probably be very low compared to the cost of the phased array) and good efficiency. As the distance between the transmitter and the receiver will be kept small, the power density would probably be high enough to provide a high rectenna conversion efficiency. Hence, no special care has to be taken, and a low cost, low profile and light weight thin-film technology rectenna can be used. Such rectennae have already reached conversion efficiencies over 85%. [Chang, 1991] A conversion efficiency of 70% may be assumed as a worst case.

Thermal Control

This section addresses the thermal control of the antenna system. A worst case scenario is assumed. In this scenario the antenna is in operational mode and fully faces the Sun at some point during its orbit. This is a conservative assumption since the orbit inclination is 51.6° (if the solar vector is not normal to the surface the solar flux is multiplied by $\cos\theta$ where θ is the incidence angle between the surface normal and the Sun). This implies that the antenna should not only radiate heat received from the Sun, but also the heat generated by the electronic equipment. In the chosen configuration the antenna does not receive albedo radiation nor Earth infrared radiation while in operation.

The two main design criteria for the thermal control are:

1. low costs
2. no thermal interface to the Mir space station

Passive thermal control is therefore a good choice. This implies the use of thermal control coatings. These are surfaces with special radiation properties that provide the desired thermal performance of the surface.

The antenna will be side mounted on the Mir station. The antenna should be thermally decoupled from the Mir station as much as possible so as to limit heat exchange. This requirement arises from the fact that Mir only has a limited thermal control capacity. As a consequence the back of the antenna must be shielded and heat can only be radiated from the front of the antenna. For the thermal analysis several parameters need to be chosen, the absorptivity and emissivity of the antenna surface being the main ones. As a surface coating white epoxy (Al substrate) is considered. This coating has a small ratio of solar absorptivity (α_s = 0.248) to infrared emissivity (ε_{IR} = 0.924) and a low equilibrium temperature. Degradation of the surface does not play a significant role since the mission duration is not more than one month.

The power transmitted is taken to be 5000 W. A margin of 10% is assumed. The efficiency of the antenna system is 70%. Therefore 1650 W must be dissipated as heat. At thermal equilibrium the heat input to the antenna should equal the heat emitted. The heat input is due to the direct solar flux and the loss in the amplifiers (system efficiency = 70%). The general equation can be written as:

$$q_{emitted} = q_{absorbed} + q_{dissipated}$$

Rewriting this equation for the antenna (per unit time per unit surface) yields:

$$\sigma \varepsilon_{IR} T_A^{\ 4} = \alpha_s q_s + p_{dissipated}$$

where σ is the Stefan-Boltzman constant (5.67 x 10^{-8} $Wm^{-2}K^{-4}$), T_A is the absolute antenna temperature in Kelvin, q_s is the solar flux (1358 Wm^{-2}) and $p_{dissipated}$ (Wm^{-2}) is the power that needs to be dissipated. In the above equation it has been assumed for simplicity that the free space temperature is very small with respect to the antenna temperature. Solving for T_A an equilibrium temperature of approximately 75 °C is found.

The operating temperature of the chosen antenna as well as the amplifiers mounted on the back of the antenna function well at high operating temperatures in the range up to 100°C. [Matra Marconi ,1992] Since structures are allowed to operate up to 65°C or more, the equilibrium temperature of 75°C is acceptable.

Another important aspect to consider is the other temperature extreme, the cold case. The lowest temperature the antenna will experience is in eclipse when the antenna does not transmit power and does not receive any sunlight radiation (direct or indirect). The computation of this minimum temperature is not straightforward. The thermal heat capacity of the overall antenna system must be known. The eclipse time is on the order of half an hour. Although the thermal analysis is not presented here, a detailed analysis is necessary.

The above discussion briefly covered both extreme thermal cases, hot and cold. The final aspect to be considered concerning the antenna is the thermal cycling. The components used in the antenna structure must all have thermal expansion characteristics which are similar. This is to assure that components do not crack (worst case) and that structural distortion is limited. Thermal cycling needs to be studied in a detailed simulation. The allowable thermal distortion is determined by the required pointing accuracy of the whole beam and by the beam dispersion caused by surface errors. As a general rule the thermal deformation should not exceed wavelength divided by 10 (= 12 mm) from the nominal flat antenna surface. Taking into account the surface defects, this a severe constraint, because with the beam pointing system selected the defects cannot be compensated by phase shifting. This compensation is only possible if a retrodirective system is used.

The 20 m long cable used to connect the Mir power source to the antenna (see section on "Power interfaces") is a potential source of concern. The cable has a diameter of 17.2 mm and is assumed to be made of copper for preliminary dimensioning purposes. The power loss is assumed to be 1%. For this case the equilibrium temperature of the cable was computed by the same method used above. It was furthermore assumed that the cable can only radiate heat directly to space, i.e. the angle of view was taken to be 180°. The surface of the cable was assumed to be covered with white epoxy (data: see above text). The results are summarized in Table 10.2.1:

Table 10.2.1 Power Cable Equilibrium Temperature

Cable losses	1%
Dissipated power	50 W
Equilibrium temp.	28 °C

The results are not very conservative because the cable insulation has not been taken into account. The insulation would lead to a much higher core temperature of the wire since the thermal conductivity of electrical insulators is generally very low.

Mechanisms and Structures

The phased array antenna and rectenna are to be folded in half. They are mounted on the outside of Progress. Care must be taken that they are inside the dynamic envelope of the Soyuz launch vehicle fairing as can be seen in Figure 10.2.10.

In order not to modify the Progress spacecraft too much, use is made of existing mechanical interfaces. These interface points are normally used to support solar arrays. The distance between these two mechanical interfaces is approximately two meters in the longitudinal direction of Progress. The support points of both the antenna and rectenna must be strong enough to withstand the axial launch loads.

To avoid damage to the antenna and rectenna they must be secured safely during the launch. State of the art thermal knife technology as developed by Fokker Space Systems will be used to hold down the panels. This thermal knife system is based on Kevlar cables being degraded by two heating elements and has numerous advantages over using pyrotechnic devices. The thermal knife has already been used in 1988 on a CNES antenna release experiment at the Soviet space station Mir.

Both the antenna and rectenna panels should have sufficient lateral stiffness so as to fulfill the dynamic requirements of the Soyuz launcher (Ariane 4 requires that the first lateral mode is more than 55 Hz). No information was available about the thermal and acoustic levels in the Soyuz fairing.

The deployment mechanism of the rectenna must be simple and automated, so that there is no need for additional EVA. The thermal knives will release the panels. Torque springs will guarantee that the deployment of the rectenna is successful. This operation is followed by a rotation of 90 degrees about the hinge of the rectenna as shown in Figure 10.2.11. The antenna will be supported in a similar manner. However it should be easily removable by cosmonauts performing EVA. During EVA the cosmonaut(s) will first remove the antenna from Progress. After that they will remount it on the outside of Mir. Use will be made of either the additional solar array mounting points or the additional working platform.

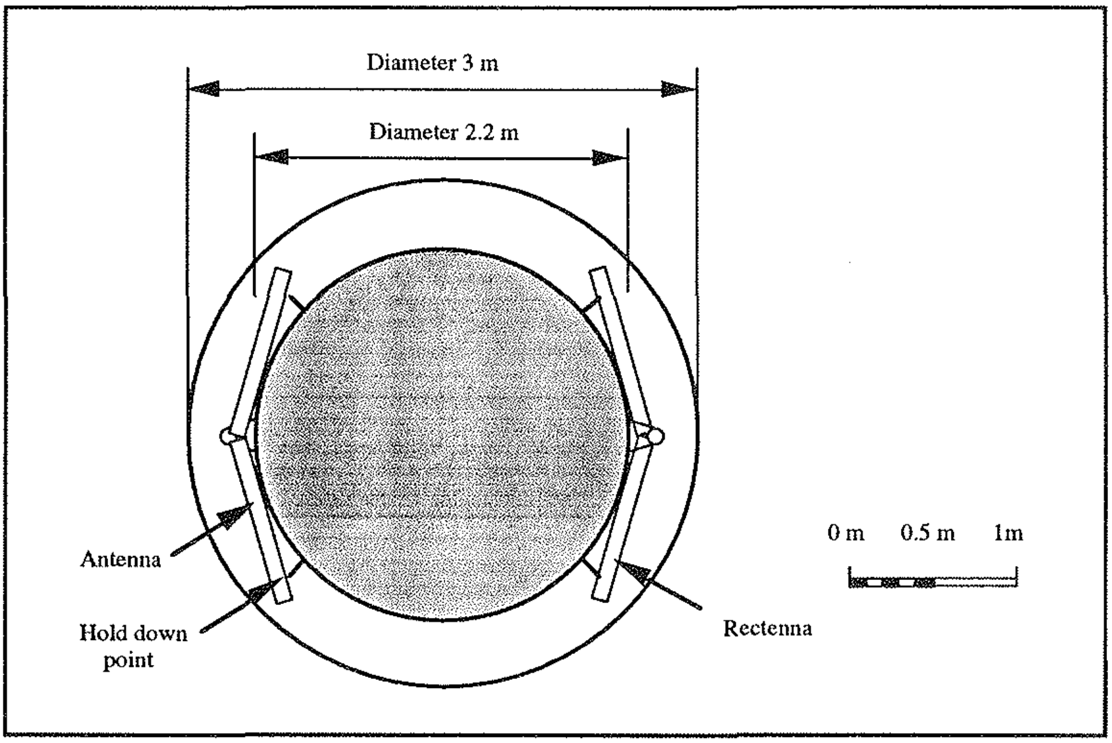

Figure 10.2.10 Positioning of Antenna and Rectenna

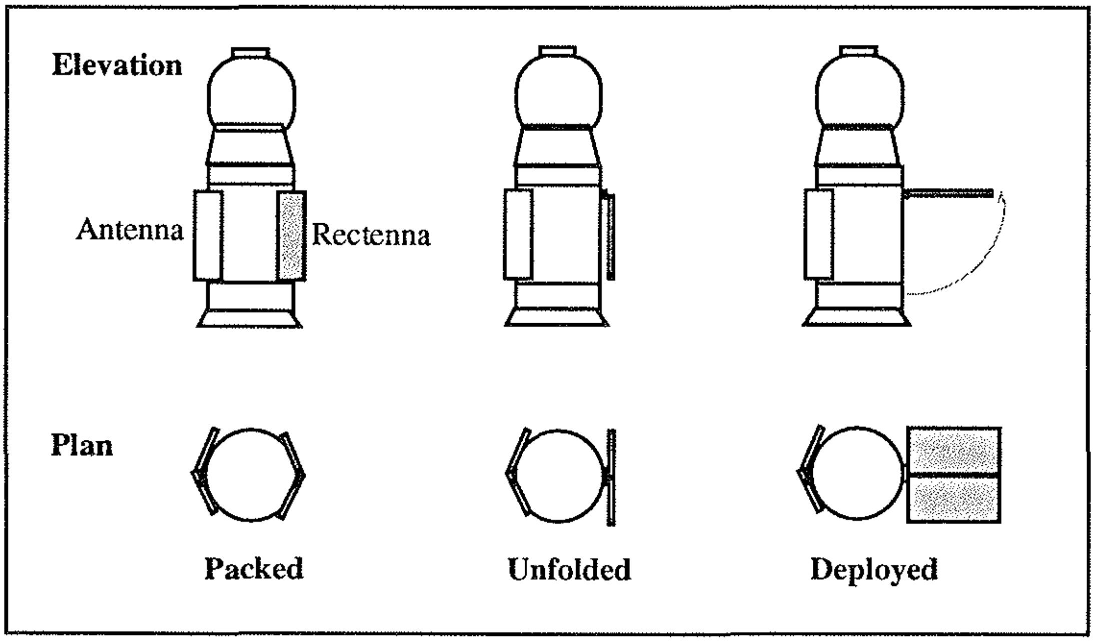

Figure 10.2.11 Deployment of the Rectenna

The pointing accuracy of the antenna imposes strict requirements on the surface stability. As a general rule the surface should not deviate more than one tenth of the wavelength ($\lambda = 12$ cm) from its nominal flat position. However this is not necessarily true for the rectenna—it can be moderately warped and still function.

Electrical Interfaces to Mir and Progress

The microwave power transmission experiment requires two power connections to be made between the microwave equipment and the Mir/Progress spacecraft. One connection has to be made from the

Mir station power bus to the microwave generator input. The other is needed at the receiving side to connect the rectenna power output to the power input of the Progress spacecraft.

The Mir station electrical power subsystem has the capability of supplying an external user with 27 V DC of electrical power at a maximum load of 10 kW for 1 hour periods. The experiment uses a nominal power of 5 kW. The Mir electrical power is available via the connectors that are used for supplying power to modules docked to the Mir. They are situated at the corresponding docking aggregates. The power is fed to the microwave generator via a 20 m long cable. The power cable is laid along the outside of Mir and the connection is made by a cosmonaut during EVA .

Assuming a power loss of less than 50 W (1%) by the interface cable, application of Ohm's law gives us a maximum resistance of 0.0015 Ω for the cable. Considering that the resistivity for copper is 1.7 x 10^{-8} Ω/m, a cable diameter of 17.2 mm is obtained. The mass of this cable is about 36 kg.

The power system on the transmitting side has the main functions of switching and regulating power for the microwave generator.

On the receiving side, the rectenna power output is connected to the power connector located at the auxiliary solar panel attachment of the Progress spacecraft. The required power connection and cabling is made on-ground.

As a minimum, the following data interfaces to the Mir and Progress spacecraft would be needed:

- serial telemetry/telecommand interface between the phased array control computer and the data handling equipment inside the Mir spacecraft
- serial telemetry interface between the rectenna power measurement equipment and the data acquisition equipment inside the Progress vehicle
- serial telemetry interfaces between the scientific experiment and the data acquisition equipment inside the Progress vehicle

The Mir station provides auxiliary electrical signal interfaces via connectors which are nominally used for data transmission between the Mir and any docked modules. They are situated at the corresponding docking aggregates. The control computer of the phased array antenna could be connected to this point using a 20 m long shielded twisted pair cable. During EVA the cable is installed and the connections are made by a cosmonaut.

The signal interface connections between the rectenna side equipment and the Progress vehicle are made on-ground. This also requires some minor modifications to be made to the existing Progress vehicle.

An estimated data budget for the mission is given in the section "Command and data handling".

Guidance and Control

The problem of guidance and control can be split into three fields:

- attitude control of the Mir space station
- orbit and attitude control of the Progress vehicle
- beam control

As described in the section "Mission Scenario", the Mir space station will be oriented with the docking port in the local horizontal plane pointing opposite to the orbital velocity vector. The Mir attitude control loop is independent from the experiment and pointing accuracy is better than 1 degree.

The attitude of Progress will be controlled by the automatic docking system so that the longitudinal axis of the Progress vehicle will be pointing to the docking port with an accuracy of better than ten arc minutes. A critical parameter will be the orbital position relative to Mir. Besides the distance requirement of 80 m the variation in orbit radius should not result in excessive shading of the beam on the antenna or the rectenna side as shown in Figure 10.2.12. Since the precise mounting positions of the antenna and the rectenna are not yet decided, this requirement could not be translated into a precise orbital position requirement. However the effect of shading on the beam control loop would be an important feature of the demonstration mission.

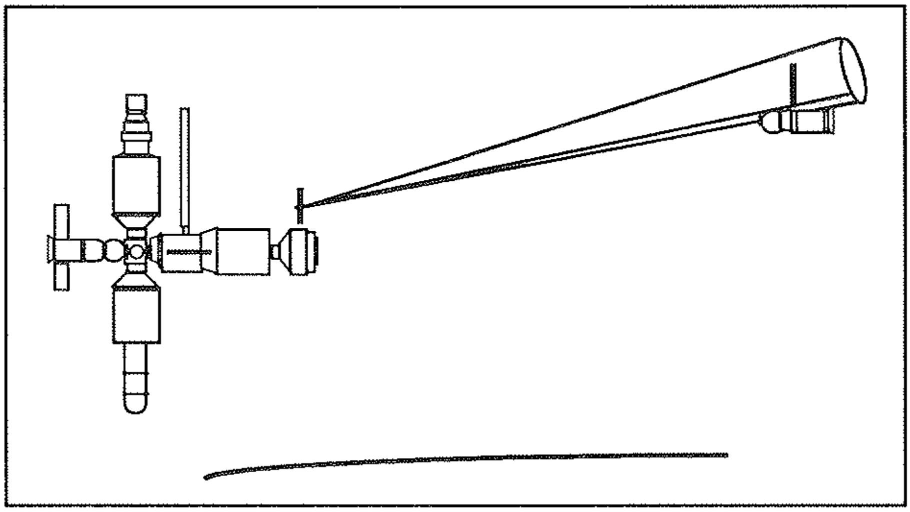

Figure 10.2.12 Shading of the Rectenna

The beam will be controlled by a simplified retrodirective phase control system as explained in the section on beam control. The range of steerability of the beam will be dependent on the orbit position limitations and thus on the shadowing problem as described in the previous section. It is supposed to be in the range of +/- 30 degrees measured from the maximum antenna gain direction.

Command and Data Handling

A functional block diagram of the microwave experiment, showing also the power and data flows, is illustrated in Figure 10.2.13 The data handling system has the following main functions:

- overall command and control of the mission equipment
- data acquisition from the phased array antenna controller
- acquisition of power readings from rectenna elements
- acquisition of measurement data from scientific sensors
- adaptation to Mir and Progress data handling subsystem in terms of electrical connections and data protocols

All science data from the scientific experiments as well as all power readings from the rectenna side are acquired by a computer residing in the Progress spacecraft, and transmitted to the Mir station via a telecommunication link. This data will be received by a computer inside the Mir and stored temporarily. The computer inside Mir also acquires data from the phased array controller and combines it with the data received from the Progress. All stored data are then transmitted to Earth via a communication channel allocated to the mission.

Now let us consider the data budget of our mission in order to predict its net impact on Mir operations. We can have a very rough order of magnitude estimate of the total data rate by assuming a 1 Hz sampling frequency for all acquisitions. Table 10.2.2 summarizes the estimated data budget for the mission. The total data rate is considered to be relatively low and therefore would be easily adapted to the existing Mir data handling subsystem.

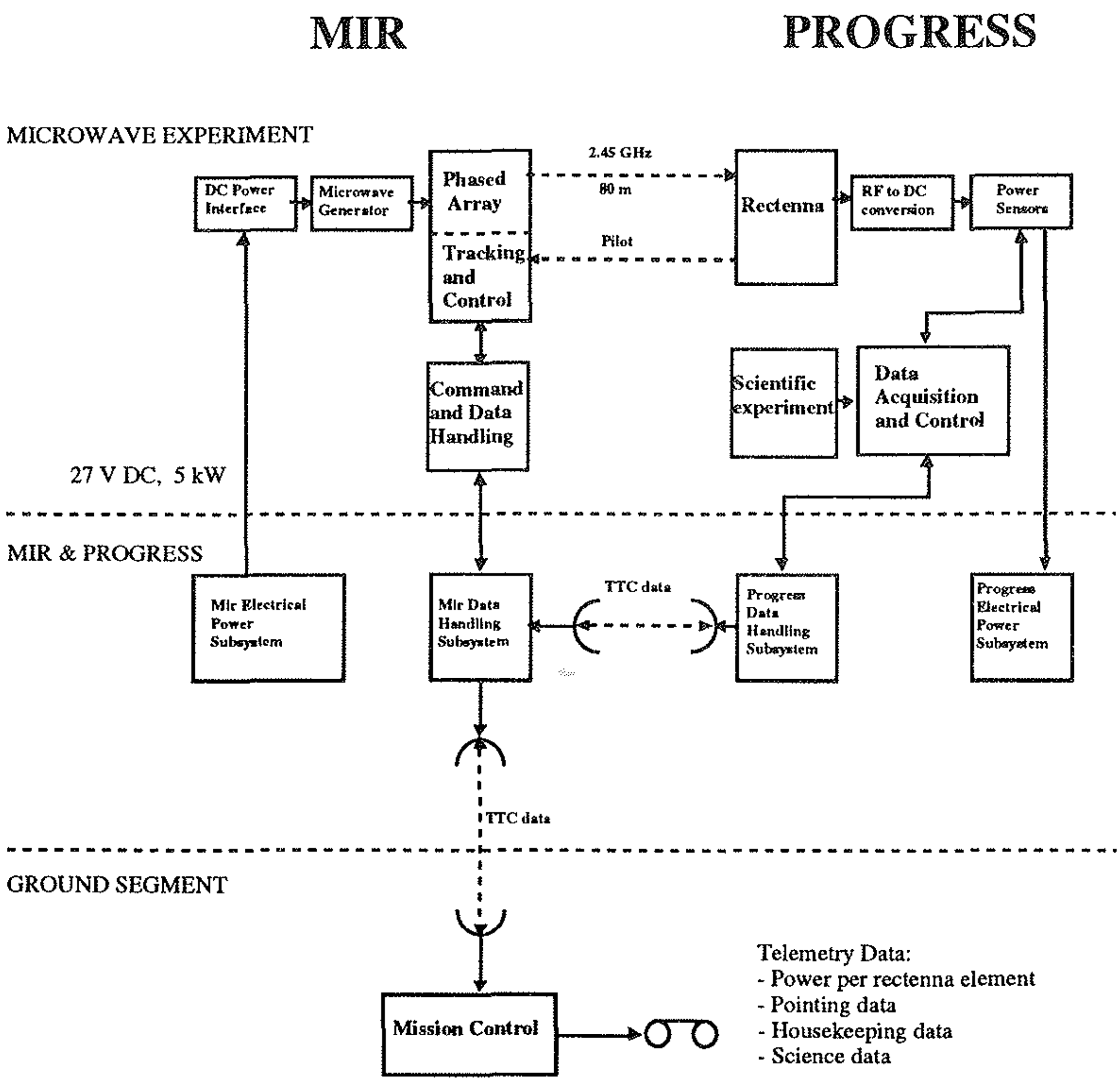

Figure 10.2.13 Experiment Functional Block Diagram

Table 10.2.2 Data Budget Estimation

Data source	Data rate (TBC)
Antenna pointing data	10 kb/s
Rectenna power readings	10 kb/s
Housekeeping telemetry	1 kb/s
Total downlink	21 kb/s
Total uplink	< 1 kb/s

Environment and Safety Issues

During all the experiment phases (including EVA activity) the normal safety regulations of Mir shall apply. Furthermore the question of microwave radiation effects on the Mir crew has to be examined. As mentioned before, the experiment will be carried out at 2.45 GHz. In this frequency range the penetration depth through metals is in the range of some micro meters. Therefore the Mir modules with their metal structure offers a perfect shielding. Only the Mir windows could be potential points of radiation leakage, but since these windows are commonly protected by a thin gold layer, they should provide a sufficient radiation shielding. However, a more detailed study on this subject has to be carried out during the concept development phase.

Electromagnetic Interference and Compatibility

The transmission frequency of 2.45 GHz has been reserved for industrial use. Therefore there should be no interference problem with existing telecommunication links. A first assessment has identified a potential filtering problem in the Mir-Progress docking system. It could be solved using a pulsed power transmission to be triggered by the docking system. However in the concept development phase, all Mir and Progress subsystems have to be examined systematically to ensure that there is no harmful impact of the power beaming experiment.

10.2.4 System Budgets and Scheduling

The system budgets presented in this section describe the overall power budget of the beaming experiment, the mass estimates for the experiment hardware, a cost estimate and a preliminary schedule to realize the project.

Power Budget

Figure 10.2.14 shows the power budget for the beaming experiment in a worst case estimate. The conversion and beaming efficiencies are discussed in detail in the section on power beaming.

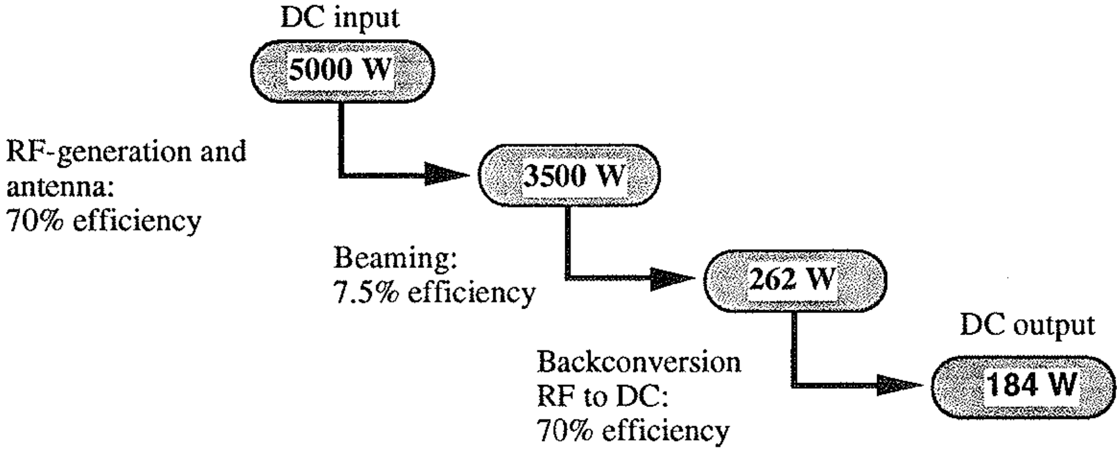

Figure 10.2.14 Power Budget

Mass Budget and Schedule

The masses of the phased array and the rectenna have been estimated using the technology value of 22 kg/m^2 (including power amplifiers and phase shifters) of the Matra Marconi Advanced SAR antenna. In case of the transmitting antenna a factor of 1.75 has been applied to take into account the down scaling effects (structural parts with non-scalable weight) as well as the higher thermal loads on the array. Since the rectenna is much less complex in technology and structure, the value of 22 kg/m^2 has been adjusted by a factor of 0.75. The relatively high mass margin of 25% represents the high uncertainty level of this estimate due to the small amount of available reference data. The scientific equipment has not been considered in this mass budget.

Table 10.2.3 Mass Budget

Phased array antenna	154 kg
Phased array support structure for mounting on Mir: (10% of phased array mass)	18 kg
mechanical interface for launch on Progress:	5 kg
Phased array control computer	10 kg
Interfaces to phased array	10 kg
Rectenna	66 kg
Rectenna support structure: (10% of rectenna mass)	7 kg
Rectenna data acquisition and control computer	5 kg
Interfaces to rectenna:	5 kg
Harness	45 kg
Margin 25 %	82 kg
Total	**407 kg**

The schedule for the demonstration is shown in Figure 10.2.15. Figure 10.2.16 shows the same information in timeline form. See section 4.4 for further information.

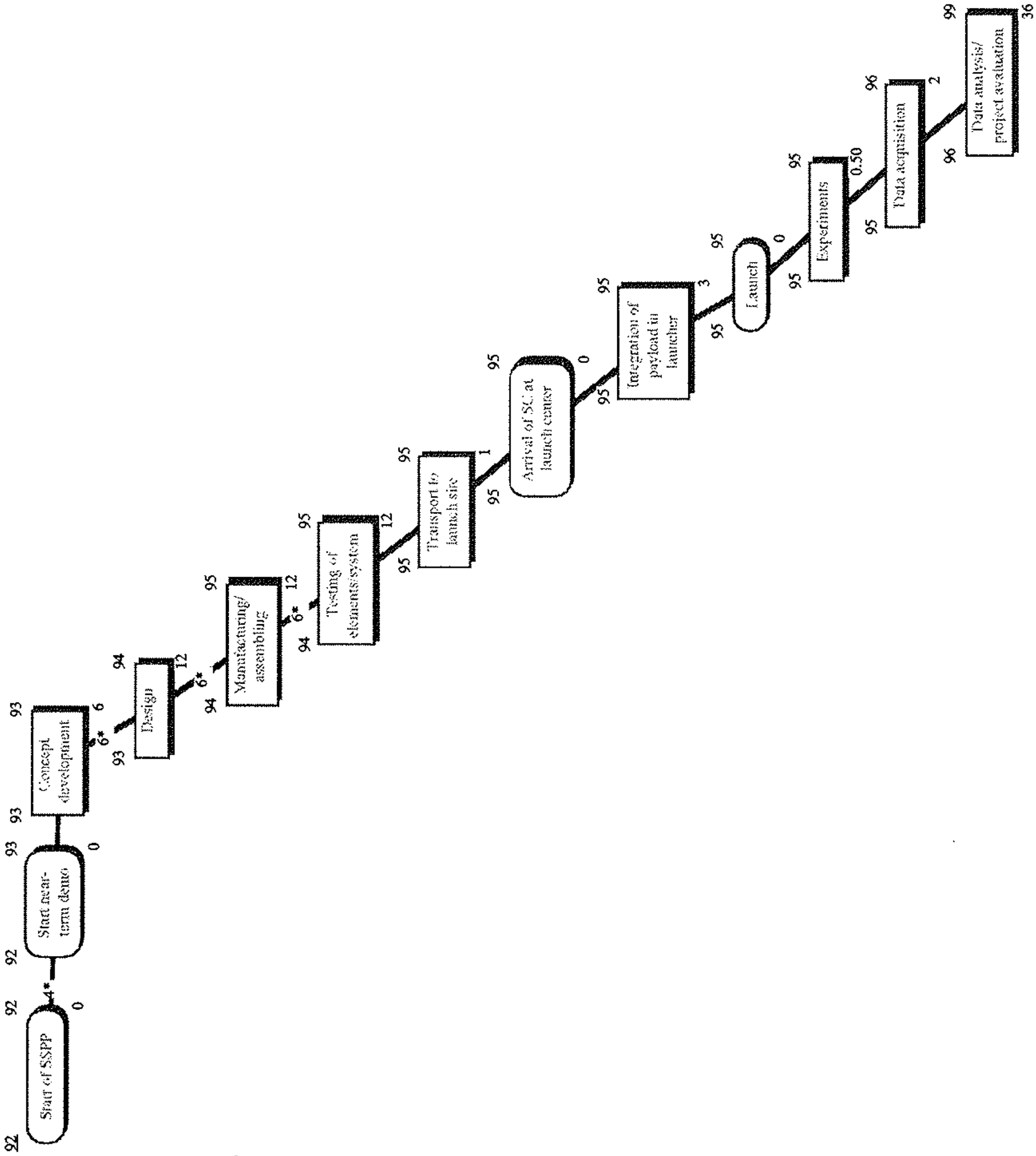

Figure 10.2.15 Task Chart (Demonstration 1)

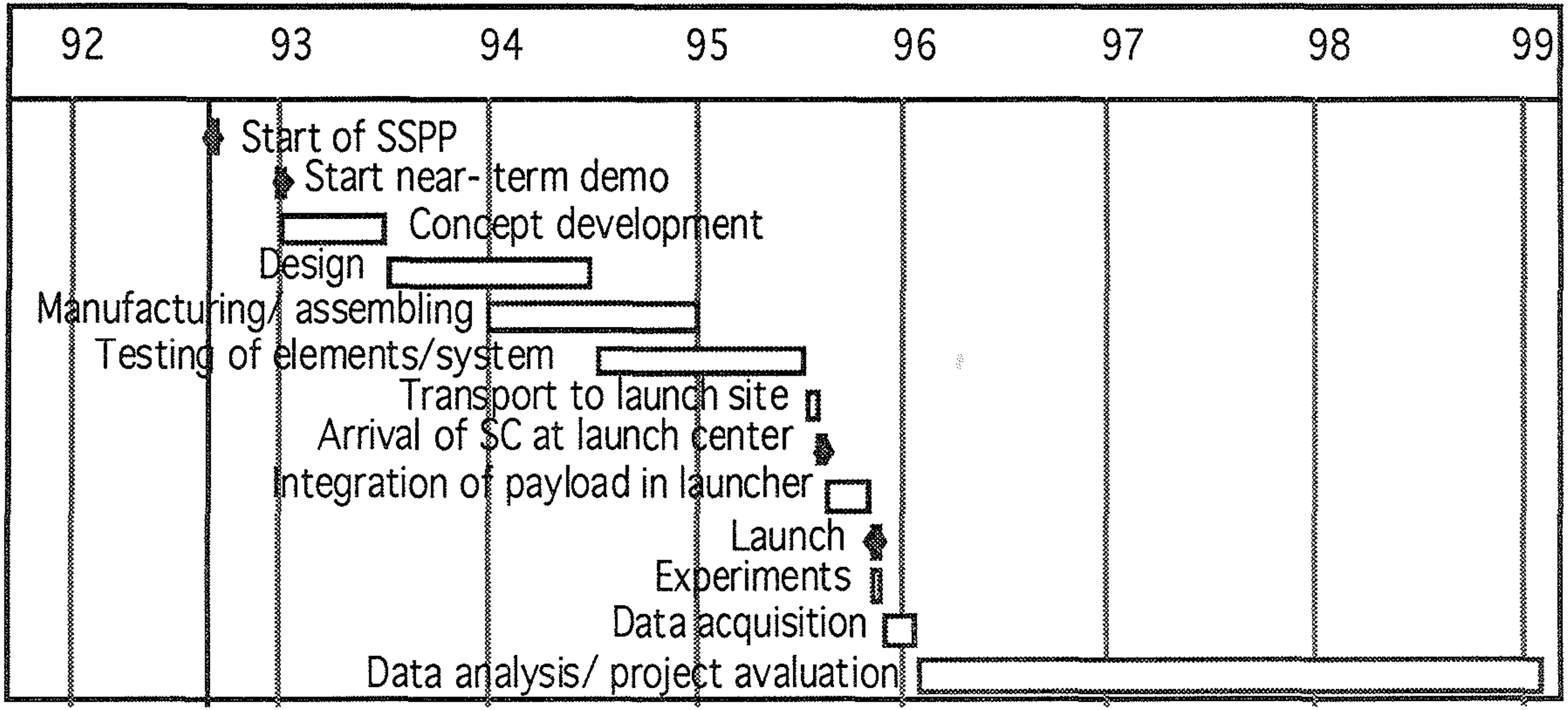

Figure 10.2.16 Timeline (Demonstration 1)

Cost Estimation

This section describes the first cost estimation based on the present system design. To calculate the cost the break down structure shown in Figure 10.2.17 was used.

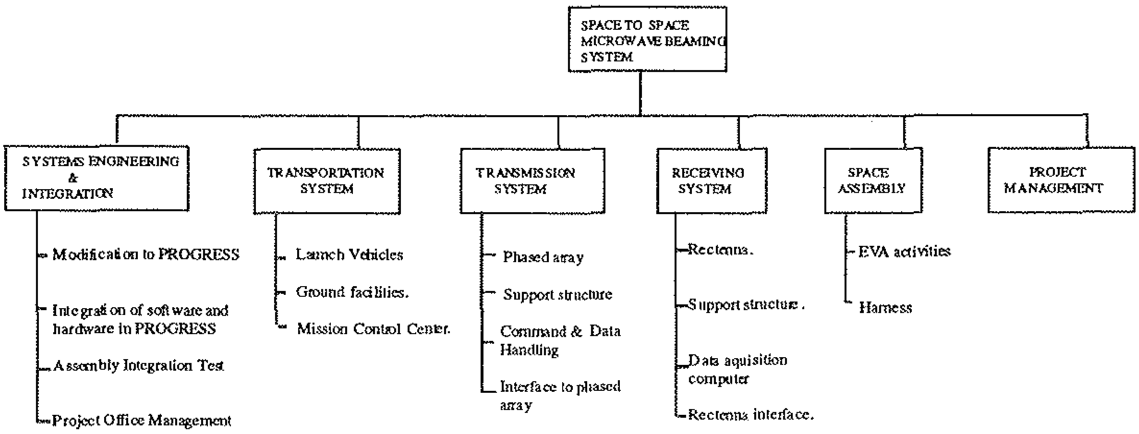

Figure 10.2.17 Work Break Down Structure

The actual costs are derived using the ESA PCM-software, which estimates the development and production hardware costs. [Wnuk, 1992] EVA activities and Mission Control Center costs are taken from US-standard values. [Comstock, 1990] The launch and modification costs have been determined in discussion with Russian engineers. The experiment takes 1/5 of the total payload mass and the launch costs will be shared accordingly. For the cost estimation only one development and one flight model is built. The reason for this is that high reliability of 99% is rather costly and not really needed (the failure of one single phased array component does not affect the experiment). The overall cost result is presented in Table 10.2.4.

The phased array and the management office are the main identified cost drivers. New electrical equipment needs to be developed and is therefore very expensive. Additionally a rather complicated management interface with the Russians has to be taken into account and raises costs.

This first estimation is conservative and the realization of this demonstration seems to be feasible within the budget limitations.

10.2.5 Conclusions

The Mir/Progress mission scenario to carry out microwave power beaming in a space to space application has been examined. This scenario seems to be very promising to meet the constraints of cost and schedule. It uses only existing hardware to carry out the experiment and the launch costs can be shared with the Mir servicing mission. Logistics are intended to stay the main mission of the Progress vehicle.

The main objective of this study is to demonstrate power beaming technology for a longer period of time and at a higher power level than previous proposals. Other objectives are to show the potential of a free flying microgravity laboratory and to perform scientific experiments on microwave-plasma interactions. The critical mission constraints have been identified and used for further evaluation of the various subsystem level designs. The main subsystems (power beaming, thermal control, structures and mechanisms, electrical interfaces, guidance and control and data handling) have been analyzed in more detail. At this preliminary stage, all identified parameters have satisfied the design constraints. The total cost of this mission has been budgeted at around US$ 78 M.

Given the Mir space station life span, this experiment is envisaged to be carried out by mid-1996. However, should this scheduling not be met, the same experiment could be flown on the Mir-2 Space Station.

Table 10.2.4 Cost Estimation

	Development Cost (US $M)	Production Cost (US $M)	Total Cost (US $M)
System Engineering & Integration	6.62	19.78	**26.39**
Integration in Progress	0.00	2.00	2.00
Modification of Progress	0.00	6.00	6.00
Assembly Integration & Testing	0.00	1.56	1.56
Project Office Management	6.62	6.62	13.23
High reliability parts	0.00	3.60	3.60
Transportation system	0.00	3.18	**3.18**
Launch vehicle costs	0.00	3.00	3.00
Ground facilities	0.00	0.09	0.09
Mission Control Center	0.00	0.09	0.09
Transmission	20.87	13.07	**33.94**
Phased array	17.16	10.68	27.84
Support structure	0.53	0.13	0.66
Command & Data Handling	1.56	1.44	3.00
Interfaces to phased array	1.63	0.82	2.44
Receiving	3.78	2.46	**6.24**
Rectenna	1.68	1.32	3.00
Support structure	0.24	0.12	0.36
Data acquisition computer	0.96	0.84	1.80
Rectenna interface	0.90	0.18	1.08
Space Assembly	0.60	0.86	**1.46**
EVA activities on Mir	0.00	0.31	0.31
EVA activities on Progress	0.00	0.31	0.31
Harness	0.60	0.24	0.84
Project Management	2.21	2.21	**4.41**
Total Cost	34.07	41.55	**75.62**
Miscellaneous (3%)			**2.27**
Total			**77.89**

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10.3 Space to Earth Demonstration

10.3.1 Project Description

The general objective of a space to Earth demonstration is four-fold. Firstly, to demonstrate an environmentally safe energy source for future use. Secondly to explore the atmosphere and ecological effects of power beaming. Thirdly, to see if solar power beaming can be cost-effective and fourthly to demonstrate the technical feasibility of collecting beamed power. The goal of the following sections is to highlight how a demonstration program for beaming power from space to ground can be achieved, with the following high level planning and cost constraints:

- Time scale: the program should be realized within 10 years.
- Cost target: the overall cost of such a program should not exceed $800 M.

These two constraints will mainly drive the system. Indeed, the near-term schedule means for instance that no revolutionary technological step nor spectacular improvement is expected to occur. In addition, the cost allocation quickly leads to the conclusion that a simple system shall be built, i.e one automatic satellite launched by a single launcher (no assembly in space, no maintenance nor astronaut intervention). A spacecraft in a sun-synchronous orbit beaming at 35 GHz will be the major cost, using $600 M plus $80 M for the launch. The advantage of this concept is that once the spacecraft is in orbit, we have the flexibility to beam to any part of the world and the services of the spacecraft could be used by any buyer. The mission envisioned was to supply isolate areas for local use, staying within the limits of a reduced scale demonstration.

Antarctica was originally discussed as a possible site for space to Earth power beaming. A closer look presented here and in the following technical sections reveal several doubts about using this location at this time except for scientific experimentation. One possibility would be to use Antarctica as the primary site for atmosphere testing and possible ground-based testing with sensors placed to measure the amount of energy reaching the ground to theoretically determine how much energy could be collected if a rectenna was built. A small-scale rectenna could be built instead in a desert to show that we can collect the power even though at this stage it will be very little, a maximum of 200 mW/m^2. This location could also serve as another data point for atmospheric testing.

The following sections further explore the possibility of using Antarctica as a site for a space to Earth power beaming demonstration. The motivation is primarily an environmental one and so the issues will be discussed from that perspective. Regardless of final location, a similar analysis applies.

Problems With Current Energy Sources

Increasingly there have been efforts by many national and international organizations to move away from non-renewable energy sources such as fossil fuels. This includes the effort to reduce the CO_2 levels being released into the atmosphere contributing to depletion of the ozone layer and the greenhouse effect. One problem is that many developing countries and remote locations do not have the resources to meet their increasing energy needs as well as develop new energy sources, thus compromising global energy goals. A major hope of a space solar power program would be to provide power to areas in need of more energy and also an alternative source of energy.

Antarctica was considered as one possible site for a space to Earth demonstration because of the great need there for an alternative energy source. Currently, oil is brought into Antarctica by tankers at high delivery cost. Low temperatures reaching -85°C make biodegradation nearly impossible, raising a problem with normal waste management and increasing the environmental damage through oil spill pollution. Another possible location would be a desert such as the Arizona desert in the U.S. This would serve as a model for a remote location where energy is also expensive, or a developing country, where wood and coal may be the primary sources of energy contributing to environmental pollution and deforestation

Because of its sparse ecology and relatively low power needs, Antarctica seems to be a prime candidate for a space to Earth power beaming demonstration. It is considered to be a huge desert as its non coastal regions receive only 5 cm of precipitation per year. The terrestrial living organisms consist mostly of birds and some plants, mostly lichens. Seven countries have territorial claims in Antarctica and the power loads are some of the poorest economically in addition to being orders of magnitude lower than almost any other populated areas [Dalby, 1992].

Alternative Energy Sources

The issues involved in determining alternative energy sources for Antarctica are integrally related to the energy needs. During the Antarctic summer the population is the highest and the power needs are the greatest. During the six months of winter the power needs are at their lowest. Thus the alternatives to be considered at these two different times may also be different.

Alternative energy sources usually include not only solar, but also biomass, water, geothermal and wind. Ground-based solar energy using photovoltaic technologies is under current consideration for the summer months [Wills, 1988], but during the winters, solar cells are not feasible so other renewable sources should be considered. Water and biomass can be eliminated quickly as ice is the primary characteristic of Antarctica; biomass not only has a low energy density but would also have to be shipped in at a cost and disposed of. [Broman, 1991] suggests that the McMurdo sound area could use geothermal energy but only during the winter. This is expensive to realize with drilling and machinery to deal with the high ionization of resulting water. Wind may be the most viable option as velocities can reach up to 320 km per hour [Dalby, 1992] and this option will probably be explored in the next few years. The current concern with wind power is the durability of the equipment, especially in the extreme conditions of Antarctica. If it proves to be a cost-efficient environmental alternative it may overtake solar power as the preferred option. Though currently, solar power seems to have the most potential for the summer months.

Effects of Beamed Power (Scientific Measurement)

In our study we are interested in the transmission from a space platform to the Earth's surface. Problems dealt with in this demonstration allow the correlation between the microwave beam and the atmospheric gases which govern the radiation exchanges in the terrestrial atmosphere (mainly CO_2, CH_4, etc.,) with important consequences in determining any change in the greenhouse effect. Therefore it is essential to monitor the following parameters in the presence and the in absence of the microwave beam: Column content and vertical profile of ozone, temperature, vertical profiles of ClO and H_2O, profiles and vertical distribution of aerosols and NO_2, column contents of NO_2 and HCl, vertical profiles of CH_4, N_2O, HNO_3, ClO, NO_2, OH, and the electron density in the ionosphere. These measurements are being done on a continuing basis by a number of satellites, and good baseline figures are now available. One example is the Upper Atmosphere Research Satellite (UARS) which was launched on the September 12, 1991. Its main objectives are to perform simultaneous, comprehensive measurements of the Earth's stratosphere, mesosphere and lower thermosphere, for investigations of energetic, chemical composition and dynamics. In Table 10.3.1 is a list of the detectors that are onboard the UARS and their main functions.

From a scientific standpoint, Antarctica is an appropriate place for a space to Earth microwave beam since there are similar ground-based measurements already in place. Another reason is based on a sun-synchronous orbit for the spacecraft which would enable measurements about every two hours over Antarctica. Finally, the fact that there is darkness six months of the year over Antarctica makes the measurements during this time period a more reliable estimate of beam effects on the atmosphere as sunlight would be absent.

Table 10.3.1 Detectors on UARS and their main function [Kendall, 1992].

INSTRUMENT	DESCRIPTION	MEASUREMENT OBJECTIVES
CLAES: Cryogenic Limb Array Etalon Spectrometer	Scanning spectrometer sensing atmospheric infrared emissions in the spectral range 3.5-12.7 microns	Concentrations of members of the N and Cl families, O_3, H_2O, CH_4, and CO_2 at altitudes of 10-60 km: atmospheric temperature profiles for indirect wind measurements
ISLAMS: Improved Stratospheric and Mesospheric Sounder	Radiometer sensing atmospheric infrared emissions in the spectral range 4.6-16.6 microns	Atmospheric temperature structure and variability: minor constituent distributions including the N family, water, methane carbon monoxide, and ozone

MLS: Microwave Limb Sounder	Radiometer sensing atmospheric microwave emissions at frequencies of 63, 183, and 205 GHz	Concentrations of ClO, H_2O, O_3, and atmospheric pressure at various altitudes from 5 to 85 km
HALOE: Halogen Occultation Experiment	Radiometer sensing atmospheric infrared absorption from occulted sunlight in the spectral range 2.43-10.25 microns	Vertical distributions of HCl, HF, CH_4, CO_2, O_3, H_2O and members of the N family.
HRDI: High Resolution Doppler Imager	Fabry-Perot interferometer sensing atmospheric emission and absorption in visible and near-infrared spectral ranges	Velocity of upper-atmosphere wind field through measurement of Doppler shifts of molecular absorption lines (below 45 km, daytime only) and atomic emission lines (above 60 km, day/night)
WINDII: Wind Imaging Interferometer	Michelson interferometer sensing atmospheric emissions in visible and near-infrared spectral ranges	Velocity of upper-atmosphere wind field through measurement of Doppler shifts of molecular and atomic emission lines above 80 km
SUSIM: Solar Ultraviolet Spectral Irradiance Monitor	Full-disk solar ultraviolet irradiance spectrometer	Spectrum of solar ultraviolet radiation from 120 to 400 nm, with resolution of 0.1 nm
SOLSTICE: Solar/Stellar Irradiance Comparison Experiment	Full-disk solar ultraviolet irradiance spectrometer	Spectrum of solar ultraviolet radiation from 115 to 430 nm, with resolution of 0.12 nm
PEM: Particle Environment Monitor	Electron, proton, and imaging X-ray spectrometer	Energy spectrum of electrons (1 eV-5 MeV), protons (1 eV-150 MeV) and X-rays (2-50 keV)

As the beam that we are currently able to transmit is rather weak, it would be better to place detectors directly on the spacecraft where the beam is being transmitted from. In this mission it will be possible meaningfully to measure composition, charge and temperature variations caused directly by the beamed microwaves using instruments located on the platform itself. These results are interesting both for the lower layers of the terrestrial atmosphere and for the ambient medium through which the satellite travels.

A retarding potential spectrometer will measure the composition of the local medium, measuring mass, charge and temperature composition. This is important for the study of various factors affecting the lifetime of any future solar power satellite.

A multi-band electromagnetic spectrometer will measure absorption of the microwaves by the various components of the atmosphere and their re-emission. This will give important information on the temperature variations caused by the beam, as well as pointing up any anomalous acceleration effects that might be caused. While unimportant for low power systems, as beam concentrations increase this could prove important.

Finally, series of static charge sensors on the transmitting antenna will enable a cull dynamic charge profile to be built up. As the outer atmosphere has significant ionization, any significant charge differential across the surface may set up electric currents, shorting the transmission circuit and preventing any power beaming at all, This could also cause significant heating problems for the spacecraft.

Effects of Beamed Power (Living Organisms)

Power leakage at the rectenna site can promote changes in the local climate. The amount of leakage would increase with increasing frequency but affect a smaller area as the rectenna size decreases. This excess energy emission from the rectenna site is released as heat into the atmosphere potentially causing changes in the ecological system.

The inner part of the Antarctic continent is said to be the most sterilized land on the Earth though there are various biota and an eccentric biome on the shore line around it. Therefore, we have two subjects to be considered on the usage of the Antarctic Continent for the rectenna site. One is the preservation of the ecological system on the shore line, the other is the preservation of the sterility of the inner continent.

The biome on the shore line of the Antarctic continent is composed of many peculiar living organisms which have evolved to adapt to the low temperature of the region. For them, even a slight additional energy such as excess heat flow from the rectenna site may be destructive because of the constant poverty of energy there. Therefore, large scale rectenna facility should not be constructed near to the shore line of the Antarctic Continent.

There may not be any biological problem in the inner part of the Antarctic continent if we consider only the existing biome. But the possibility exists that the energy spill from the rectenna site may make and grow a new biome there. Even one short period of high temperature (as warm as the shore line region) per year would be enough for an ecological system composed of the blue algae and molds to destroy the sterility of the inner region. Because of this sterility, the Antarctic is now a treasure house of envrionmentally clean scientific specimens such as meteorites, ancient rock samples, and ancient air trapped in the ice. In some research, the organic pollution caused by the biological activity is very serious. Therefore, large scale rectenna facility which can make the partial climate as warm as the shore area should not be made in the inner part of the Antarctic Continent.

Regulatory Considerations

Normal obstacles for a project of this caliber such as air, construction, and town permits or community outreach and health and safety issues will be minimal. Environmental monitoring requirements and national or international permits will still have an impact on the project, but again should be minimal. The primary regulatory factor in Antarctica is the Antarctic Treaty of December 1, 1959 which declared the area south of 60° latitude an international preserve for science. Thirteen countries participated, prohibiting mining and resource exploitation; these include Argentina, Australia, Belgium, Chile, the French Republic, Japan, New Zealand, Norway, the Union of South Africa, the Union of Soviet Socialist Republics, the United Kingdom of Great Britain, and Northern Ireland, and the United States of America.

Plans regarding scientific programs in Antarctica working toward the preservation and conservation of living resources need to be exchanged amongst the participants. Each country involved has the right to designate observers to carry out an inspection of all areas of Antarctica to ensure compliance with this treaty. Representatives meet at suitable intervals to exchange information, consult together on matters of common interest to Antarctica, and formulate recommendations to their Governments to further the principles and objectives of the treaty. Cooperative working relationships between specialized agencies of the United National and other international organizations having scientific or technical interest in Antarctica is encouraged.

Disputes between countries with regard to the exercise of jurisdiction in Antarctica shall immediately consult together with a view to reaching a mutually acceptable solution. This can be done through negotiation, inquiry, mediation, conciliation, arbitration, judicial settlement or other peaceful means of choice. If these methods prove unsuccessful, the dispute will be referred to the International Court of Justice for settlement.

Market Value

The section on remote locations in Chapter 3.1.1 contains a preliminary market analysis of solar power in Antarctica versus other locations such as the Arizona desert. Of course, a demonstration is not expected to be of market value and the primary motivation of providing Antarctica with an alternative energy source is an environmental one, but it is still useful to compare costs.

The cost of providing electricity to Antarctica works out to approximately 0.60$/kWh. [Wills, 1988] The cost of using a portable generator to convert diesel to electricity is 0.10$/kWh for a 250kW generator size. [Leonard, 1991, in Ch. 3] For comparison photovoltaics are thought to produce electricity at 0.22$/kWh. [Wills, 1988]

This figure of 0.60$/kWh does not take into account the cost of environmental damage caused by oil spills or the greater damage caused globally by CO_2 and the depletion of fossil fuels. Thus a maximum cost of electricity must be determined by the customer realizing that the motivation for providing this alternative energy source is calculated from indirect costs. As an arbitrary figure lets choose $10.0/kWh which because of costs of building actually means charging $4.0/kWh. The calculations described below indicate that $1 billion is the most that could be returned on a project costing around $3 billion. This in and of itself may challenge the feasibility of the program.

Assume a large base in a dry valley would receive 1.5 MW of power. To optimize the kW hours we would need a Molniya orbit (with the average distance taken to be 25,000 km) transmitting 35 GHz from an antenna with a 100 m diameter.To receive 98% of the power would take a rectenna of a 5.2 km diameter. For cost reasons we choose a rectenna of 520 m to receive about 20% of the power although in actuality we would only receive about 10% of the beamed power due to the elliptical path of the satellite. Assume coverage of 16 hours per day which results in 58,400 hours over the ten year life span of the satellite (365 x 10 x 16 = 58,400). Therefore the satellite must beam 15 MW in order to produce 1.5 MW on the ground at 5.66 W/m^2 excluding weather interference and other losses.

Assuming a highly inflated cost of fuel at $10/kWh, the total market would be 58,400 hours times $10 times 1500 kW or $876 million (1500 x 58,400 x 10 = 876,000,000). But it will cost at least $3 billion to put a 15 GW satellite in orbit. Thus we have demonstrated the ability to lose over $2 billion. Clearly some advances need to be made before space solar power for Antarctica is a viable energy alternative. In the subsequent chapters, several technologies will be identified that need to be advanced in order to beam power from space to Earth more efficiently and cheaply.

10.3.2 Mission Analysis

Altitude Selection

The altitude selected for this design example is mainly driven by the system cost constraint. Indeed, this constraint limits the size of the spacecraft, which in turn limits the power available from the satellite. Therefore, the altitude will be imposed by the amount of power the system can get back on ground. The results of a trade-off analysis are presented in Table 10.3.2 showing that, assuming a given system (power output, rectenna), the lower the altitude is, the more energy there is that can be stored on the ground. A typical 1000 km altitude orbit can be selected for the purpose of this exercise. Actually the precise value shall be the result of a compromise between the performance and the air drag problems which appear at lower altitudes (attitude and orbit control).

Table 10.3.2 Altitude Selction Trade-Off Analysis

Orbit	Visibility time	Power on a 1 km^2 rectenna	Stored Energy on ground for 140 kW S/L output and 1 km^2 rectenna
1000 km	5%	50 kW	60 kWh per day
10000 km	20%	750 W	3.6 kWh per day
36000 km	100%	80 W	1.9 kWh per day

Orbit Selection

According to the previous point, two typical orbits are discussed here: the sun-synchronous orbit (SSO, inclination ≈ 100°) and the " near" equatorial orbit (inclination < 30°), both in low Earth orbits (typically in the range of 1000 to 1500 km). Table 10.3.3 makes a tentative comparison of the orbits. The SSO orbit can be used as a baseline for this design example because it allows more power, and has the potential to supply any location in the world.

Table 10.3.3 Orbital Selection Trade-Off Analysis

Item	SSO orbit (6:00 - 18:00)	near equatorial orbit
Solar array efficiency	+ (Angle between SA and sun constant at first order)	- (Orbit plane drift)
Eclipses	+ (Very few, possibly none around 1400 km)	- Every orbit
Access	+ (Potentially any)	- (only equatorial belt)
Rectenna visibility	- (A few minutes per orbit)	- (A few minutes per orbit)
Mass into orbit	-	+
Beam pointing	- (pointing needed because rectenna seen with various angles)	- (pointing needed because rectenna seen with various angles)
Losses	- , for polar regions (Snow, ice)	+
Maintenance	-	+ (In principle possible with the shuttle)

Launcher

For this mission, the largest launchers will be considered (HII, Ariane V, Titan IV, Energia) with the basic assumption that no other launcher will be available within the next ten years. The preferred one is Energia because of its volume and mass into orbit capabilities.

Power Generation and Beaming Analysis

Power Generation

The baseline solar generation system for this mission is the use of classical flight proven solar panels covered with silicon cells. An alternative using solar concentrators focusing sunlight on gyroreactors is proposed as an option (see space segment section). Due to the fairing constraints (Energia), the solar array size is limited to 1000 m^2. This gives a power output in the range of 100 kW after 5 years in orbit. Since the useful transmitting part of the orbit is in the order of only 5% only, batteries can be used on board to store energy during the non visibility periods. This typically increases the power up to 200 kW at the power subsystem output.

Power Conversion and Transmission

Here again, for fairing and cost constraints, the size of the transmitting antenna is limited (practically lower than 100 m^2). In addition, in order not to have the whole power concentrated on one point, an active phased array antenna is proposed as a baseline. Taking into account the conversion efficiency, the power which actually leaves the satellite is in the order of 150 kW. Another point concerns the selected frequency, which has to be compatible with the state of the art extrapolated to the next ten years. Two frequencies are possible here: 2.45 GHz which is well known but leads to a very spread out beam, and 35 GHz for which the technology is only now under development but with a more concentrated beam. Both take advantage of atmospheric windows, with less rain losses for 2.45 GHz. The proposed baseline however is 35 GHz for ground power flux density reasons. This is reflected in the Table 10.3.4:

Table 10.3.4 Frequency Trade-Off Analysis

	2.45 GHz	35 GHz
Satellite power output	150 kW	150 kW
Maximal antenna gain	49 dB	72 dB
size of half power beam intercepted on ground	12 km	1 km
Atmospheric losses in dry conditions	0	5%
Peak power collected on 1 km x 1 km rectenna	700 W	70 kW

Power Flux Density

In order to allow the power to be received (i.e higher than thermal noise for instance) and efficiently converted back to DC, this density has to be higher than a certain threshold per rectenna converting element. This threshold is assumed to be 100 mW, which should be achievable within ten years. In our case, the power flux density ranges from a few tens of mW to a few mW per square meter ; this means that the power shall be concentrated prior to the rectification. This can be achieved with parabolic dishes. Then, as the satellite passes over the rectenna with high velocity and with different azimuths from one orbit to the other, the concentrating dishes shall track the satellite.

Conclusion

The system power budget is given on the Table 10.3.5. The power flux density constraint appears to be a very limiting factor which reduces the extent of the demonstration to experimental purposes only, as significant amount of power will also be needed to move the concentrating dishes. Given these conditions, the location of the rectenna in Antarctica is questionable. A suitable location would be near the equator or even in a high latitude (but accessible) area in order to take advantage of more visibility periods.

Table 10.3.5 System Power Budget

	Beginning of visibility	Middle of visibility
Satellite elevation (°)	35	90
Solar Array surface (m2)	1000	1000
Cells efficiency (EOL)	10%	10%
Cell layout ratio	90%	90%
SA power output (W)	1.22E+05	1.22E+05
Power conversion & distribution ratio	90%	90%
Battery subsystem output	1.00E+05	1.00E+05
Power subsystem output	2.09E+05	2.09E+05
frequency	3.50E+10	3.50E+10
wavelength	8.57E-03	8.57E-03
Transmitting Antenna surface (m2)	100	100
Array Efficiency	0.7	0.7
Gain	1.71E+07	1.71E+07
Satellite output power (W)	1.47E+05	1.47E+05
Satellite distance to rectenna (m)	1.55E+06	1.00E+06
Beam width (rad)	8.57E-04	8.57E-04
Rectenna size for 3 dB power collection (m)	2316	857
Chosen rectenna size (m)	857	857
Atmospheric losses (dry conditions)	5%	5%
Power flux density at center of beam (W/m2)	8.30E-02	1.99E-01
Collected power on rectenna (W)	3.32E+04	6.96E+04
DC conversion efficiency	80%	80%
Power at system output (W)	2.66E+04	5.57E+04
ground concentrator diameter (m) for 0.1W power (at center of the beam)	1.21	0.50

10.3.3 Space Segment

Baseline Design (Photovoltaic Power Generation)

Satellite General Architecture

The proposed spacecraft architecture for the sun-synchronous orbit mission is presented in Figure 10.3.1 in stowed and deployed configurations. This configuration has been determined using the following constraints and guidelines:

- Launcher capabilities: mass and volume offered by Energia
- Symmetric configuration to minimize disturbance torque's
- Minimize the air drag
- Take into account sufficient allocation for solar array and antenna accommodation.

The basic satellite configuration includes a 1000 m^2 Silicon solar array for power generation (120 kW), a 100 m^2 transmitting antenna, deployed from the platform which provides the basic services (power storage, attitude and orbit control, structure and thermal control, and data handling facilities). The satellite as presently designed is in the range 10 to 15 tons, and provides 150 kW at its output (solar array, batteries, antenna).

Solar Array Sizing

Considering the trade off between price, efficiency and mass, Silicon solar cells were chosen for the solar array SGD-1 subsystem (Space to Ground Demonstrator-1). Table 10.3.6 shows a quick comparison between Ga-As and Si solar cells.

Table 10.3.6 Solar Cell Trade-Off Analysis

Topic	Si Solar Cell	Ga-As Solar Cell
Efficiency	14.6% at 28°C BOL	19% BOL
Weight	1.8 W/g	0.9 W/g
Cell size	8cm / 8cm	4.4cm / 4.4cm
Cost per cell (c)	$ 128.00	$ 387.00< c < $968.00
Reliability	high	low

Based on the above data, calculations on total power and mass for the SGD-1 were done and are shown in Figure 10.3.2.

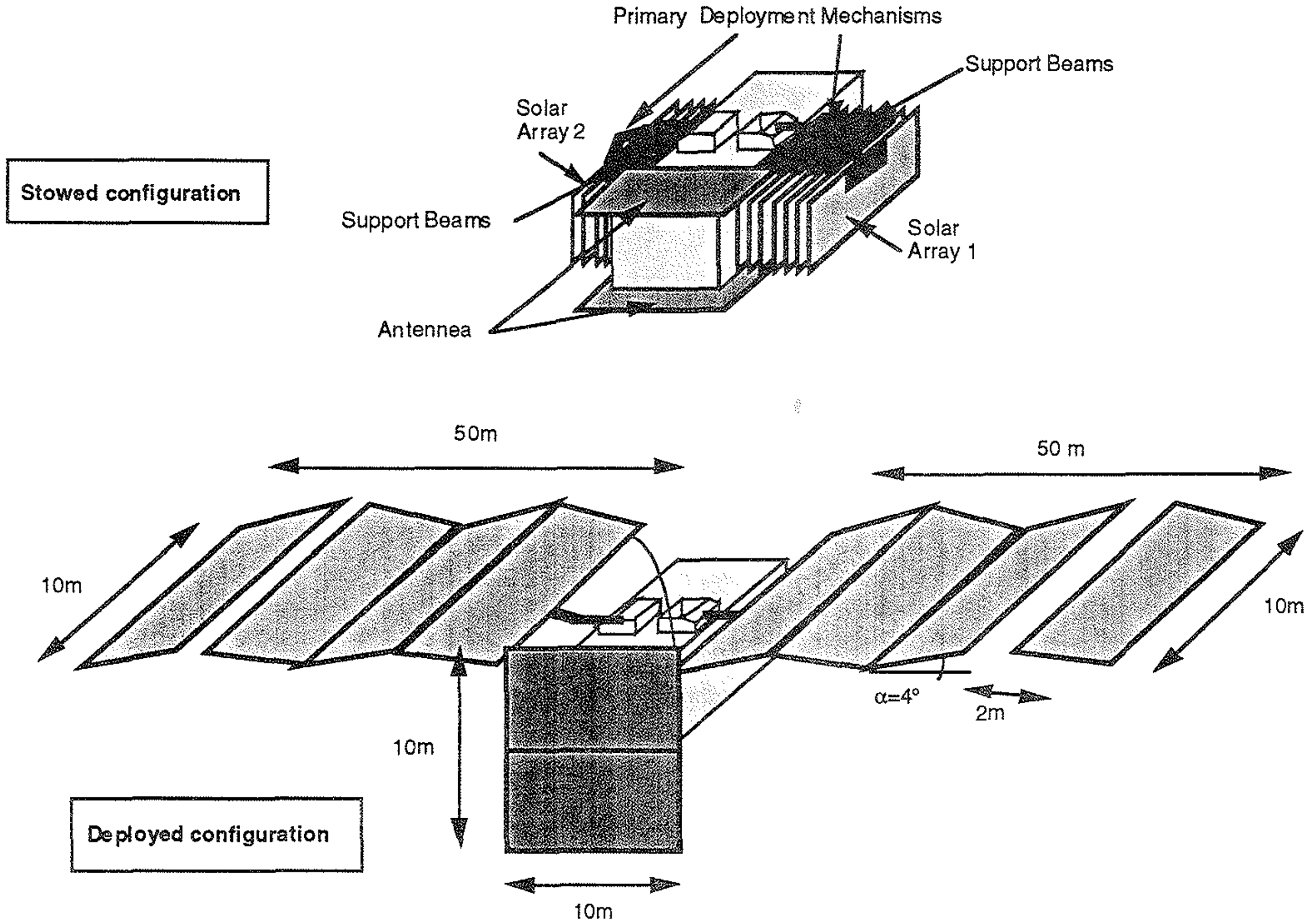


Figure 10.3.1 Satellite Configuration

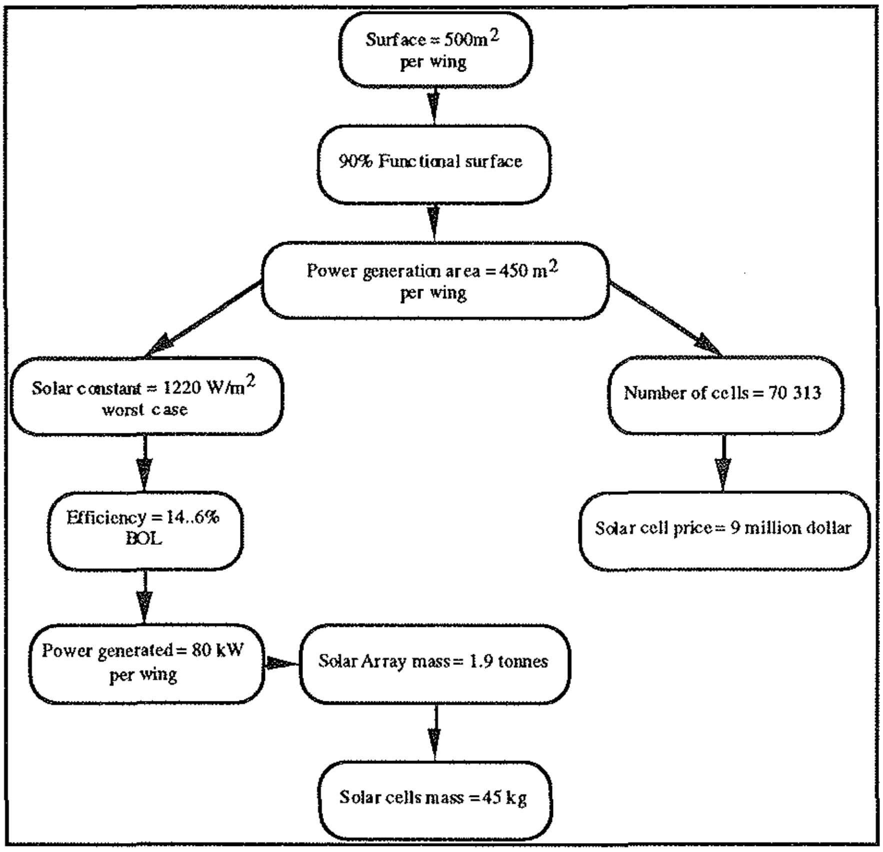


Figure 10.3.2 Calculations for SGD-1

Phased Array Antenna

The operating frequency of the microwave transmission system is 35 GHz. This choice was driven by the need to reduce the size of the transmitting antenna. Phased array antennas already exist for S- and C-band applications and similar technology is expected to be developed for higher frequencies in the near future. For a 35 GHz array the inter-element distance will be roughly 5 mm, corresponding to one half of the wavelength. A 10x10 meter antenna will have approximately 10^6 elements discrete elements. It is unlikely that this could be designed using hybrid solid state technology. Monolithic microwave integrated circuits (MMIC) is a promising technology that could make large scale integration of discrete circuits possible.

The antenna is a phased array with a retrodirective pointing system. The antenna receives a pilot signal from the rectenna and the phase information of this signal is used to direct the beam in the desired direction. The pilot signal is shifted in frequency with respect to the carrier to provide sufficient isolation between the two. The frequency shift should be as small as possible so that the individual phasing of the elements compensates for any atmospheric perturbation of the wave front. The proposed phase control scheme for the METS rocket experiment is an example of a retrodirective system using a two tone pilot signal. [Kaya, 1991]

Modular antenna design is required to be able to integrate the large number of elements necessary for the antenna. Each element would have phase conjugating circuitry for the retrodirective system and a local oscillator to drive the antenna elements. To reduce complexity, several elements could be connected in parallel. This will introduce beam focusing which will reduce the maximum angular area in which the main lobe can be steered.

A suggested block diagram for the cell architecture is shown in Figure 10.3.3. The received pilot signal is a low power signal compared to the emitted power carrier. This implies that very high isolation is required between receiving and transmitting channels. The received pilot signal is processed to extract phase information for the carrier. A phase locking circuit drives a low power oscillator feeding a power amplifier which in turn is connected to the antenna element. Every cell of the antenna is thus a stand alone device only requiring DC power supply. This eliminates the need for RF-power distribution within the antenna structure.

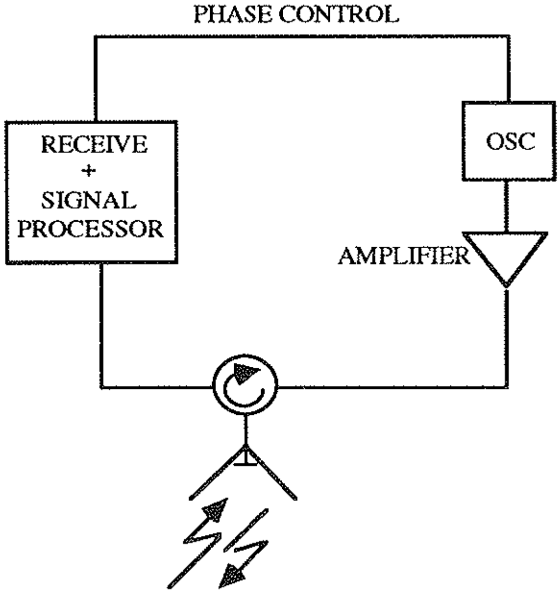


Figure 10.3.3 Retrodirective Cell Architecture.

Subsystem assessment

Structure

The structure of the SGD-1 (Space to Ground Demonstrator -1) satellite will not be describe in detail because of design decisions yet to be made. The general structure is illustrated in Figures 10.3.4 and 10.3.5.

Platform: The structure makes maximum use of light-weight carbon fiber/honeycomb technology. A modular design simplifies satellite assembly and integration activities. A central tube in carbon fiber provides the necessary rigidity for both, and houses the tanks for the propulsion systems.

Solar Array: The solar array is composed of two wings. Each wing has a surface of 500 m^2. The solar array is deployed by a Deployment Mechanism that is composed of three main parts: driver mechanism, a boom and an interface plate, which connects the deployment mechanism.

Thermal Control

The thermal control subsystem is important in the design phase because of the harsh thermal environment of the spacecraft in space:

- Extremely cold space (4K)
- Intense solar illumination (1350 W/m^2 in low Earth orbit)
- Eclipses (even in sun-synchronous orbit, none at 1500 km)
- Vacuum: absence of convective heat transfer

The narrow temperatures ranges for spacecraft components, the limited thermal gradients allowed and the thermal stability required makes the thermal mathematical model essential for the validation of the global mathematical model of the spacecraft.

The SGD-1 thermal control subsystem goals are globally to keep the temperature of the spacecraft components within allowable ranges for all mission phases by controlling the following parameters:

- Net thermal energy exchange with space environment (by selecting coatings and finishes and by heating)
- Heat exchange between interior and external surfaces
- Heat exchanges between internal satellite components mutually
- Local time constants (temperature excursions, selection of materials and their masses)

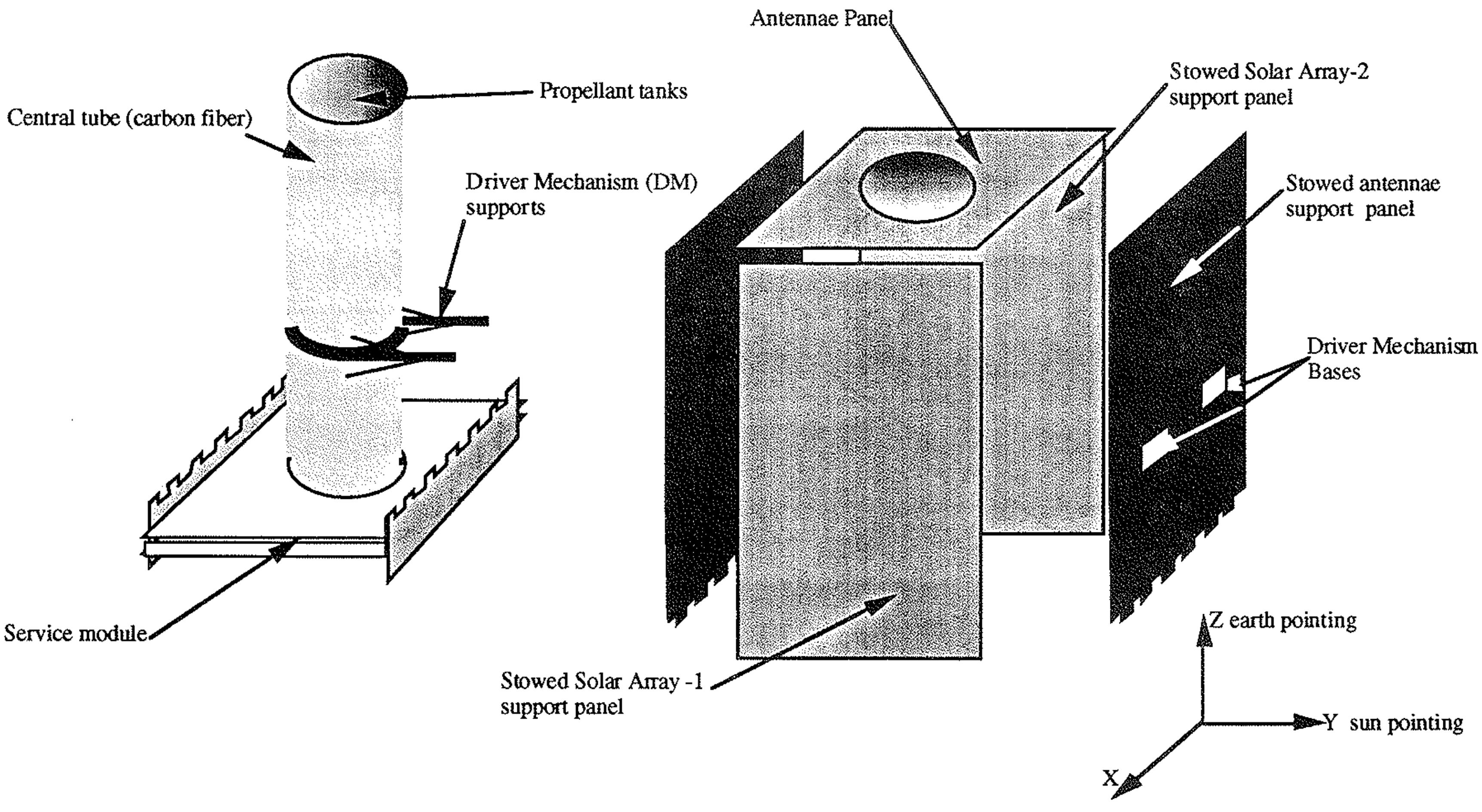


Figure 10.3.4 SGD-1 Platform Structure Design

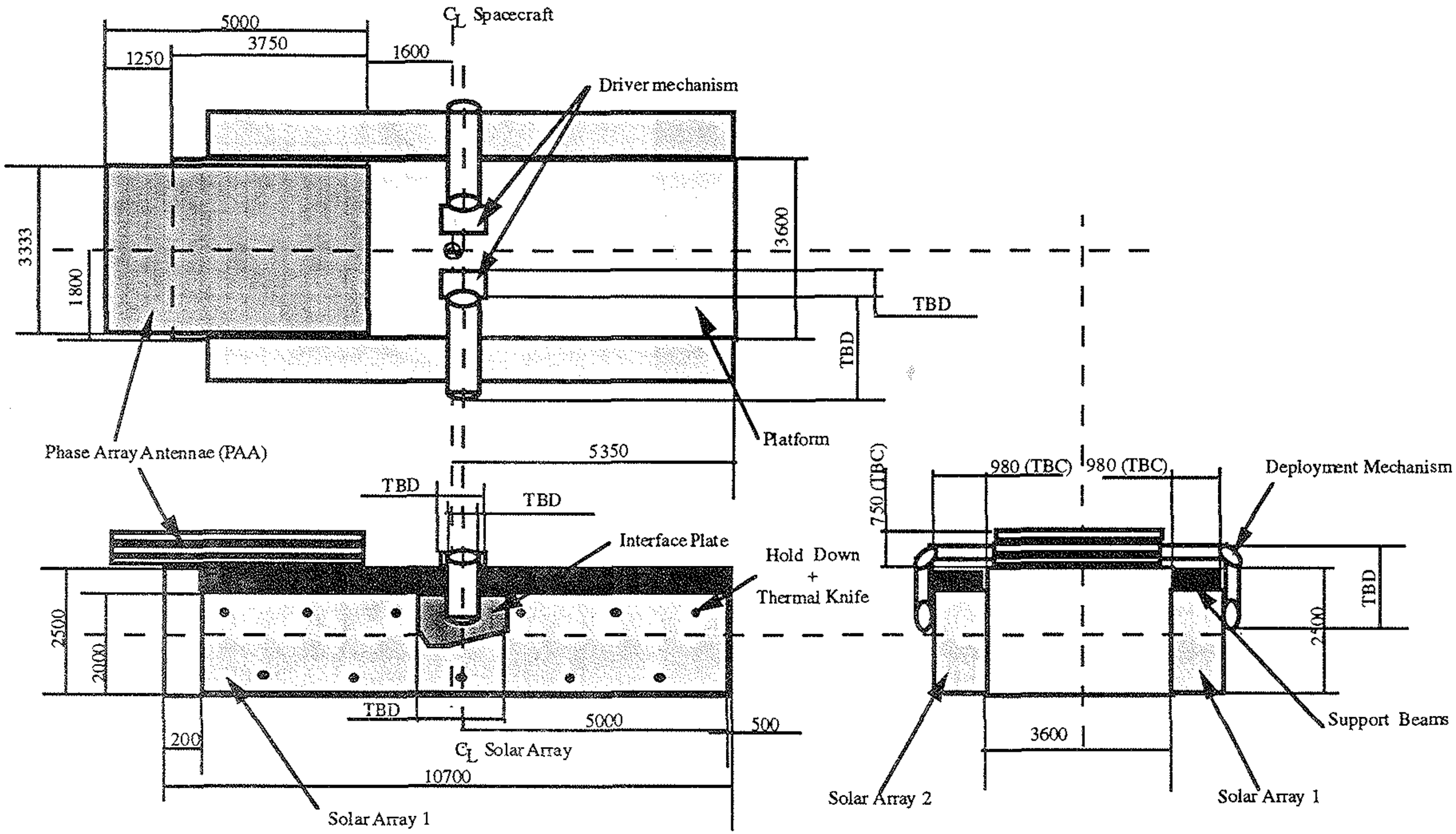


Figure 10.3.5 SGD-1 Satellite Stowed Configuration

The SGD-1 basic thermal control design will be based on:

- Thermal space environment as shown in Figure 10.3.6
- Thermal balance of external surface in space
- Internal heat transfer by radiation
- Internal heat transfer by conduction
- Eventually internal heat transfer by evaporation or melting

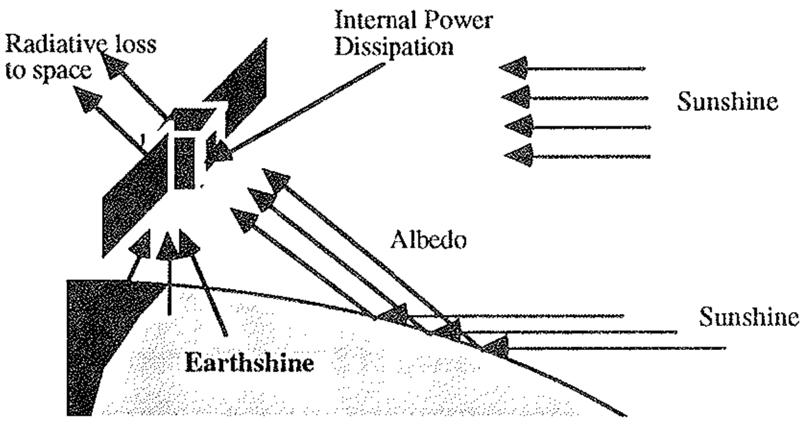


Figure 10.3.6 Thermal Space Environment

The thermal space environment of SGD-1 is shown in Table 10.3.7 and the environmental fluxes are indicated in Table 10.3.8.

Table 10.3.7 SGD-1 Orbit

Orbit	Sun-synchronous
	descending node at 18:00h
Altitude	1000km
Inclination	100°

Table 10.3.8 Environmental Fluxes

Sun	1350 W/m^2
Albedo	350 W/m^2
Earth shine	200 W/m^2

In order to calculate the heat balance of the spacecraft three kind of data are needed:

INTERNAL DISSIPATION + ABSORBED FLUXES = RE-RADIATED HEAT TO SPACE

The Thermal Control Subsystem (TCS) of the SGD-1 will not be designed in detail but globally described because of the high level spacecraft design.

Two kinds of internal dissipation can occur in the spacecraft: radiative and conductive. The first one can be controlled quite reliably when modeling and computing data. However, the conductive heat transfer is difficult to estimate because of the high uncertainties.

Two kinds of Thermal Control Methods can be use in the SGD-1: Passive and Active.

Passive systems

These kind of systems are the basis for all thermal control subsystems. No mechanical moving parts and no power consumption are required. The characteristics of these systems are the low mass, low cost, high reliability and low heat transfer capabilities. Passive control will be predominant in the SGD-1 TCS.

A critical item in the SGD-1 is its Phase Array Antenna (PAA) 10m x 10m. In fact, considering that the PAA will have an efficiency of 70% transmitting at 35GHz (technology to be developed) the internal radiation of the PAA electronics raises to 60kW as shown in Figure 10.3.7.

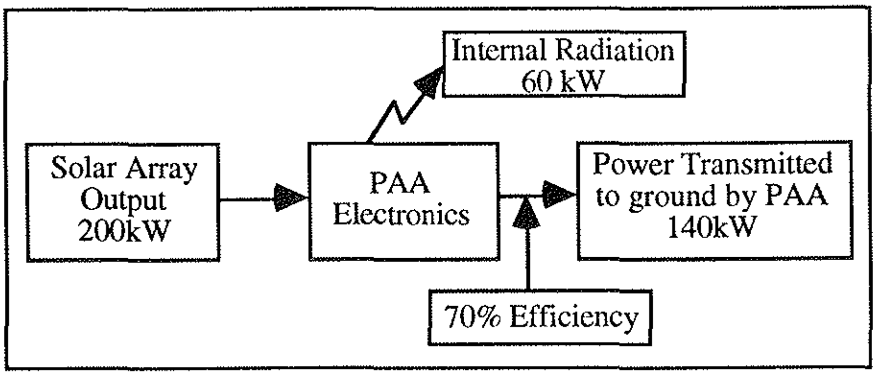


Figure 10.3.7 Efficiency of SDG-1 Phased Array

The next calculation gives an idea of heat radiated from the rear side of the PAA.

$$Q = A\varepsilon\sigma T^4$$

Where	Q	heat to be dissipated
	$A = 90\ m^2$	total surface
	$\varepsilon = 0.85$	surface emmisivity
	$\sigma = 5.67\ 10^{-8}$	Boltzman constant
	$T = 353\ °K$	nominal PAA functioning temperature (80°C)

$$Q = 90 x 0.85 x 5.67.10^{-8} x (353)^4 = \underline{67\,000\ W}$$

This means that it would be possible to radiate to deep space 67 kW from the transmitting antenna. White painting could be used as the thermal coating (α_s=0.15, ε_H=0.9)[1].

The general thermal control tools of the SGD-1 are listed in Table 10.3.9.

Table 10.3.9 SDG-1 Passive Thermal Control

Topic	Passive thermal control tools
External radiation	-Location and sizes of surfaces which have to reject internally dissipated heat: RADIATORS -Insulation: Multi-Layer Insulation (MLI)
Internal radiation	-Layout of units -Thermo-optical properties (a and e) absorption and emissivity - Insulation (MLI)
Local time constant t	-Components heat capacity
Conduction through structural parts	-Thermal doublers -Material selection (i.e. aluminum conductivity =150 W/mK, Titanium and steel conductivity =10W/mK) -Insulation
Conductance over interfaces	-Contact areas -Interface filler (high conductance) -Thermal washers and stand-offs

[1] α_S = solar absorptivity ε_H =hemispherical emmisivity

Active systems

Active systems are always a supplement to the passive ones. It involves mechanical moving parts and/or power consumption. They are characterized by their high mass, cost and power, lifetime constraints considerations, and their high heat transfer capabilities and adaptability in flight. The SGD-1 platform might have a network of heat pipes, which transfers heat generated by the internal subsystems to outer space. Some examples are given in Table 10.3.10:

Table 10.3.10 Active Thermal Control

Active thermal control tools
Heaters: Thin foil heater mats Resistors
Fluid loops
Louvers

Thermal Control Subsystem Interfaces

The Thermal Control Subsystem (TCS) interfaces with nearly all other subsystems in the spacecraft. Some requirements and design drivers to TCS are listed in Table 10.3.11. This implies that:

- TCS **has many requirements** and design drivers from other subsystems
- TCS **needs** thermal information about other subsystems
- TCS **imposes** thermal requirements and thermal interfaces to other subsystems
- TCS is **affected** by most of the changes in other subsystems

In fact, the thermal information of other subsystems is required for spacecraft analysis and hardware design. For example:

- Mission phases (launch, maneuvers, deep space)
- Configuration, geometry, dimensions
- Mass
- Material data (solar absorptivity, conductivity)
- Instrument and subsystem equipment operation
- Unit dissipations
- Launcher interface
- Heater bus(es) supply voltages

Table 10.3.11 Requirements and Design Drivers

Requirements	Design Driver
Temperature limits	
Temperatures gradients	Thermal Distortion
Heater power budget	
Mass budget	
Data-handling and telemetry budget	Thermistor conditioning
Dimensional limitations	Field-of-view of instrument apertures
Lifetime	Degradation of optical properties
Autonomy	Limited ground coverage
Electromagnetic compatibility	Electrically conductive external surfaces
Mission	Operations (dissipations) Attitude, orbit (space environmental inputs)

Attitude and Orbit Control

The attitude of a spacecraft is its orientation space with respect to some reference system, i.e. how the spacecraft body axes are oriented relative to an inertial or rotating coordinate system.

The motion of a spacecraft is described by four aspects: position, velocity, attitude and attitude motion. Position and velocity are the subject of orbital mechanics. Attitude and attitude motion, however, are the subjects of attitude dynamics, i.e. the motion of the spacecraft about its center of mass. Since environmental torques (such as gravity gradient torques, solar wind torques and atmospheric torques) are a function spacecraft position and velocity, spacecraft orbit and attitude are coupled. That is, the spacecraft position and velocity determine the environmental torques which affect the attitude motion, where as in low attitude Earth orbits, the attitude of the spacecraft affects the atmospheric drag on the spacecraft and therefore its orbit.

Attitude control is the process of orienting the spacecraft in a desired direction. This includes attitude stabilization (maintaining the attitude in a desired state) and attitude maneuver control (changing the attitude from one orientation, old state to another orientation, new state). This process involves the use of attitude control hardware (actuator such as reaction wheels or jets), on-board or remote computers to generate the commands and relevant software. The concept selected is given in Figure 10.3.8.

Any spacecraft requires a type of attitude control and determination. For engineering purpose, attitude control is needed for functions such as pointing the antennas in a desired direction, pointing solar panels in a desired direction, pointing the control jets in the desired direction to accomplish efficient maneuvers.

Assumptions

The satellite is in a sun-synchronous orbit, the inclination selected is 100 degrees for an altitude of 1000 km. The antenna is a phased array type, the solar array does not need fine pointing so the body will be piloted with an accuracy of 0.5 degree which is classical for a LEO satellite. Looking at the assigned mission the satellite will be three axis stabilized.

Sensors

- Four gyroscope blocks of two axes (two redundant)
- Two Earth sensors of Infrared scanning type (one) on +Z side
- Two sun sensors on +Y side (one redundant)
- Two sun acquisition sensors. One on +Y side , one on -Y side

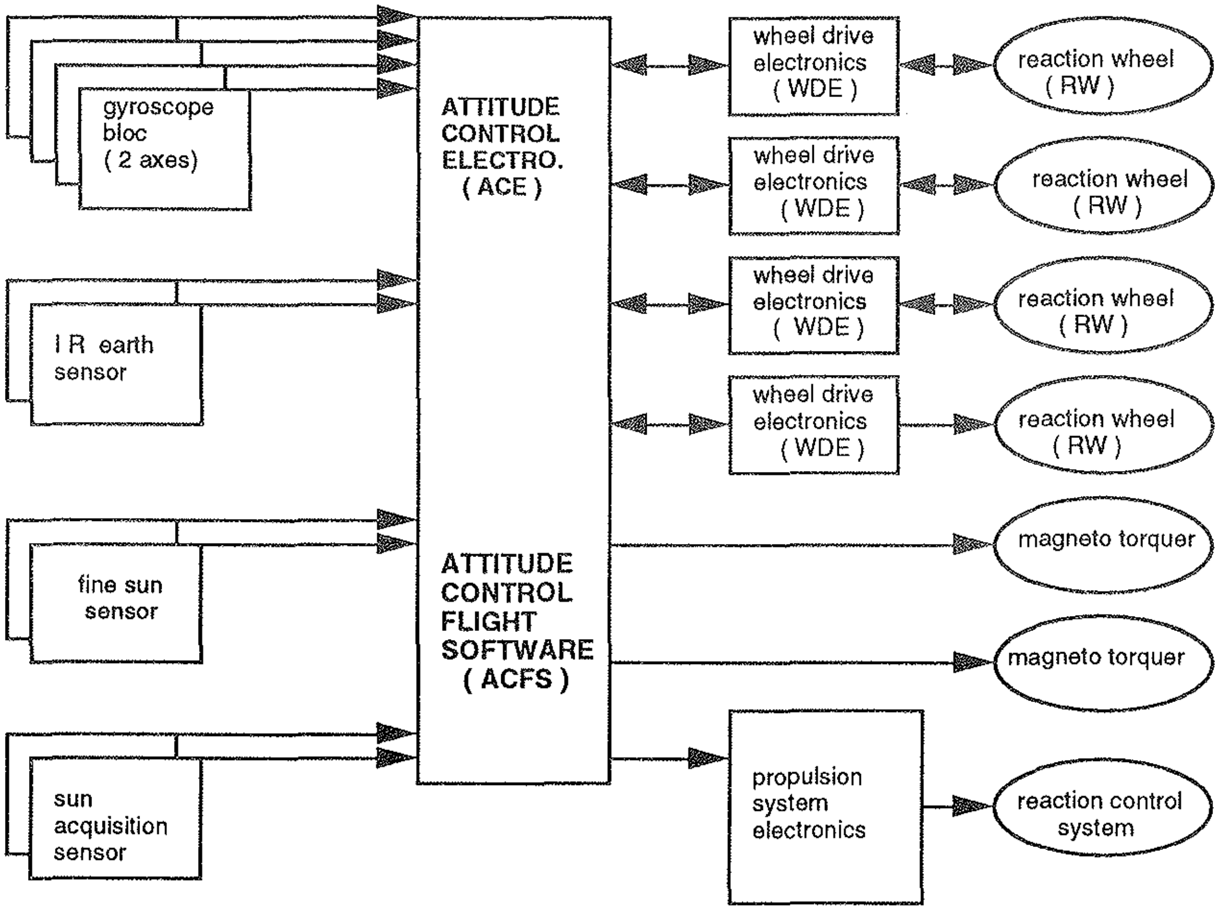
Actuators

- Three reaction wheels, one in each satellite axis
- Two magneto torquers
- A propulsion system
- For simplicity it is chosen to have a monopropelant system using Mono Methyl Hydrazine; the tanks will be pressurized on the upper part by nitrogen (ystem used on Spot). We estimate the consumption for 5 years to 110 kg ; the satellite will be equipped of two tanks of 75 kg MMH each.
- Twelve 15N thrusters on the sides +,- Y and +,- X
- Pressure transducers, electrical and pyromecanical valves, fill and drain valves, filters

Platform Electrical Architecture

The Electric Control System receives a small part of the electric power from the large solar array that generates the power to be transmitted to the Earth and distributes the appropriate power to each onboard equipment, thermal control system, attitude control system, orbit control system, TT&C (Telemetry, Command, and Control system) and so on, of the spacecraft. The total power for all the systems would be much smaller than the power transmitted to the Earth as the mission. If the orbit would be selected to have eclipses, batteries to provide power for all systems should be required. The electric power needed would then be collected by the electric control system directly from the solar array, or from the battery connected to the solar array. The electrical architecture is shown in Figure 10.3.9.

SGD1 AOCS BLOCK DIAGRAM



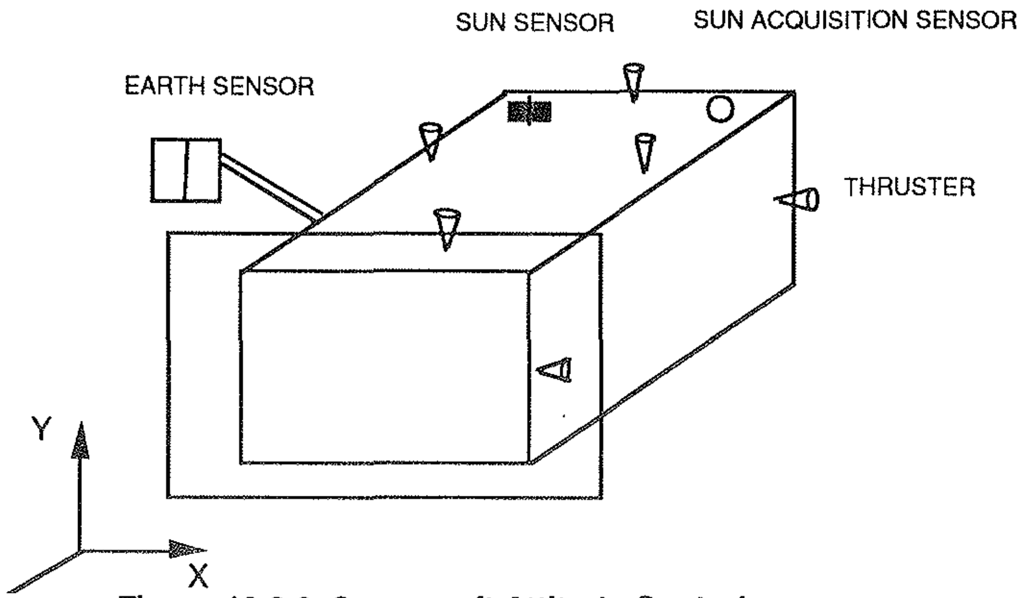
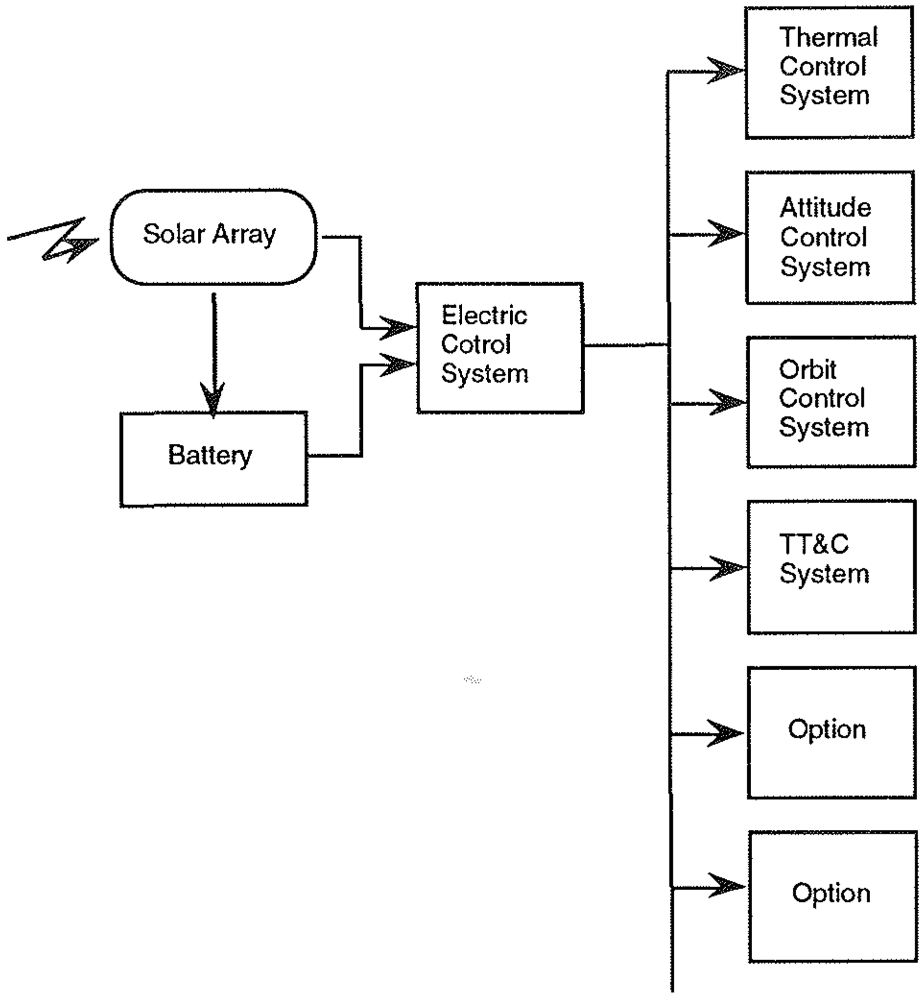


Figure 10.3.8 Spacecraft Attitude Control



Distribution of Electric Power for each system

Figure 10.3.9 Electrical Architecture

TT&C and Data Handling

This subsystem performs the telemetry, command, ranging, and control functions. The telemetry subsystem generates the telemetry signals, which monitor the status of spacecraft, and transmits them to the Earth. The command subsystem receives the command signals from the ground station and interprets them. The ranging subsystem receives the ranging signals for orbit determination from the ground station and then returns them to the Earth.

The radio frequency (RF) communication equipment required for the telemetry, command, and ranging signals consists of a telemetry transmitter, command receiver, wave guide, coax and so on.

S-Band, C-Band, and Ku-Band frequency could be used for signal transmission. S-Band frequency is generally recommended for spacecraft use. If 2.45 GHz is assigned as the requency to transmit electric power, a frequency other than 2.45 GHz should be assigned to communicate for telemetry and command.

The control electronics or computer (microprocessor or CPU), integrates the distributed set of the other electronics performing TT&C functions. This computer includes software programs installed in its memory beforehand to perform attitude determination and control as well as TT&C functions.

Input/Output (I/O) Requirement for Telemetry and Command

Since hardware mass and power are subject to the I/O points for telemetry and command, I/O requirement would be set up early in the design phase. A standard communication satellite, which is intermediate between simple satellite and complex satellite, has 400 to 700 I/O points for telemetry and 600 to 800 I/O points for command. Considering the space solar power mission to be neither simple nor complex, the solar power satellite would have the same scale of I/O points as the standard communication satellite. The total number of I/O points for both telemetry and command would amount to about 2000 points. In this case, the hardware mass including telemetry equipment and command equipment would range from 50 kg to 60 kg and the power required for those two equipment's would be about 135 W.

RF Communication Equipment and Antenna

For RF communication equipment, an antenna (omni antenna for S-Band), telemetry transmitter, command receiver, wave guide, coax, and so on would be needed. If the same band frequency would be used for both uplink and down link, only one antenna could be used commonly for transmitting and receiving. However, the diplexer should be put behind the antenna to divide between the transmitted signals and the received signals. Data rate, which is a very important factor to hardware, would be selected to be 250 bits per second for the command receiver, and 1000 to 5000 bits per second for the telemetry transmitter. If the sun-synchronous orbit is selected, the ground station for TT&C on the high latitude should be better for RF communication, because the higher latitude the ground station is located on, the more time it can see the spacecraft.

Budgets

Spacecraft Power estimation

Power level is the most important parameter in designing the space solar power satellite. The key driver is the ground power level requirement, and we must take into account of other requirements such as the constraints of cost, launcher capability, time limitation and the current technology availability.

Given the power level on ground and the microwave transmitting frequency, we can decide the power level in space and the size of transmitting antennae. For most large satellites, the power consumed by payload is about 40 to 80% of the total energy generated by the solar array. For the case of solar power satellite, the percentage will be in the range of 80%, since the only purpose of solar power satellite is to beam power to the Earth.

With the basic assumption that the spacecraft is made of two solar arrays of 50 m by 10 m each, a service module and a phase array transmitting antennae using 35 GHz frequency, we can make our power budget with the current technology available to us.

We assume that the current Si solar array's conversion efficiency is 9% for space use, the specific performance is 120 W per square meter and the mass specific is 30 W/kg. The life span is five years and the degradation rate is 20% at the end of life.

At the beginning of life ,the total power generated by the Power Subsystem is 200 kW among which 150 kW could be converted to micro wave to the Earth and 50 kW is used by the spacecraft bus for thermal control, TTC, GNC and other subsystems.

As for the power used by the spacecraft bus, GNC subsystem takes about 50% of it. Since the solar power satellite is much larger than the existing satellites, its increased solar panel and transmitting antennae are the major sources of disturbance for both attitude and orbit control because of the atmospheric drag and solar radiation. So larger magnetic torquers with more power input are needed to compensate the attitude drifts. As for the power subsystem, no batteries or only limited amounts are needed, because the solar power satellite demo is in a sun-synchronous Orbit of 1000 km high which means that there will be almost no eclipse during the whole operational period. So we do not need to store energy for using in the shadow area.

Spacecraft Mass Estimation

First we try to use the maximum capability of the current launch vehicles available to put satellites into sun-synchronous orbit with height of 1000 km. Again the satellite's mass is divided into four different parts: solar panels, phase array transmitting antennae, service module and propellant.

With the current solar array available for space use,the specific mass is about 30 W/kg, and the highest specific mass of solar array is 66 W/kg at the beginning of life. Given the solar array's size by 50 m by 10 m, the mass of two solar arrays are 4000 kg and 2000 kg respectively.

The service module is almost the same as the existing 3-axis stabilized satellites except that the sizes of solar arrays and antennae of solar power satellite are larger than the current ones. So the solar power satellite service module will be heavier, larger and more complicated. If we take into account the solar power satellite's high reliability requirement because of the uniqueness of its application, more redundant parts are needed in order to achieve the goal. So the mass of the service module could be 2 or 3 times of the mass of the existing satellites which means that the mass of the service module of the solar power satellite is in the order of 2 or 3 tons.

In order to minimize the ground rectenna and keep the ground power output at a given level, the size of the transmitting antennae should be as large as possible but within the constraints of spacecraft structure. Assuming that the phase array transmitting antennae is 10 m by 10 m when deployed, and the materials used to build the antennae has a density which is two times of the density of the solar array (4 kg/m^2), we get a mass of 0.8 ton for the service module of the solar power satellite.

Since the sun-synchronous orbit of 1000 km is chosen,there will be almost no eclipse during the whole operational period, we do not need batteries to store energies for eclipse usage which saves quite a lot of mass. So the total dry mass of the solar power satellite is about 4+3+0.8=7.8 tons. If we add the mass of propellant which is about 20% of the total mass of the spacecraft, then the mass of the $800 M solar power satellite will in the order of 10 tons.

One thing we should remember is that the mass of transmitting antennae is mainly determined by the current technology available and its size which is closely dependent on the ground segment and the frequency selected for beaming the power.

Spacecraft Cost Estimation

The only cost assumption is that the space solar power program is a $800 M space to ground demonstration project. Now the question is how to spend the amount of money in the three main parts, the launch, the ground segment, and the satellite.

With a satellite of 10 tons and a sun-synchronous orbit of 1000 km, the only launcher which is capable to put it into such orbit is the Energia. The cost of one Energia launch is about $80 M.

The second part is the ground segment cost of about $120 M which includes the cost for building the huge rectenna, power distribution system and also the cost for operation maintenance. At the early phase of the demonstration within the budget constraint, only one ground rectenna could be built. Hopefully this is enough to demonstrate the technologies we have achieved to beam power from space to ground and to find some true usage of power from space. With the development of the full scale space solar power system, the cost for ground segment will grow at a moderate rate as the case of telecommunication satellite ground receivers which may be far beyond the $800 M budget constraint. So we can only build a ground rectenna with this limited amount of money under the extremely favorable conditions.

The most expensive part is the satellite which is estimated to cost $500 M to $600 M for research and development, manufacturing and testing. As for the detailed cost of every part of the satellite, much more data are needed in order to proceed the cost estimation. So now only the analogous method is used here. We feel that the spacecraft can only be built within the given budget by assuming satellites cost will be lowered by 50% of the current cost during the next 10 years.

Based on the current satellite cost, 20 to 25% of the cost is for the Control Subsystem, 25% is for Payload Subsystem, 5 to 10% for the Power subsystem. But for the solar power satellite the situation may be a little different because of the unique purpose of the solar power satellite. So the portion for the Payload Subsystem and Power Subsystem will be relatively higher than those for the existing satellites, may be 30 to 35% for the Payload Subsystem and 20% for the Power Subsystem. If GaAs solar cells , which are three times expensive than the silicon cells ,must be used in order to reduce the size of solar panel for the same amount of power level, the percentage for the Power Subsystem may be even higher. But with the current budget constraint of $800 M, Silicon cell is the only choice for us. For the two solar panels the cost will be about $200 M which leaves some money to us for the other parts of the spacecraft. Since the phase array is used to beam the microwave to the target on the ground, it will cost much money to develop the technology used in space, so the cost of the phase array transmitting antennae will be in the order of $150 M. We still have about $250 M for the other subsystems which are more technologically developed now. Since up to now no detail subsystem designs are available, the only thing we can do is to break down the cost of solar power satellite according to the relative technique readiness level by subsystems as shown in Table 10.3.12.

Table 10.3.12 Spacecraft Cost Breakdown

subsystems or parts	percentage of total cost	technology readiness level
solar panels	20%	S.U
phase array	15-25%	S.U
power conversion	10%	S.U
AOCS/propulsion	10-15%	S.U
structures	5%	S.Q
thermal control	5%	S.Q
data handling	5%	S.Q
TTC	5%	S.Q
integration and test	10%	S.Q

S.U -- full scale space qualified hardware not available, development work needed.

S.Q -- full scale space qualified hardware could be built with the existing technology.

The phased array is the most complex part of the satellite, because of the large size (10 m by 10 m) and the simultaneous phase control requirement of every element for pointing the beam to the given target. For the power conversion of 100 kW level, problems such as power balance, thermal control and high efficiency conversion hardware in space need to be solved. Finally for the AOCS,the satellite's large flexible solar array will cause attitude stabilization problem which may need new control techniques to be developed. And the magnetic torquers for attitude control may also cause EMC problems which can only be solved by system test. Based on the above facts, we come to the cost breakdown which needs to be improved with the progress of this demonstration project.

In conclusion, much more research and development work need to be done with regard to the technology we need to put solar power satellite into orbit. We need to convert the lab scaled technology to the space qualified full scale technology with low cost. It is found that the budget of $800M is a strong constraint to this demo project. And a new concept for the $800 M demonstration may be needed.

Alternative Design (Solar Dynamic Power generation)

In Chapter 7, it was shown that over a long period, a solar dynamics system was potentially less expensive (almost 1/2 the cost) than an equivalent photovoltaic system. Therefore, in this section an alternative design based on a solar dynamic system will be discussed. At the moment no solar dynamic systems have flown in space. Therefore it was not thought feasible to baseline the SDS as the power source of the satellite, since this is aimed for a near term launch (in ten years time). However if technology became sufficiently developed and systems were space qualified then it may be possible to integrate this system on the satellite in time for the launch on this mission, thereby taking advantage of the benefits of a SDS demonstration. Even if this was not feasible then it would be possible to plan a follow on mission, which could be used to demonstrate solar dynamic systems before expanding the scale to much larger systems. It should also be noted that for a launch on Energia the photovoltaic baseline is also power limited, the SDS mission would allow the power generated to be increased over the present level.

In this section various design concepts are discussed before the proposed design is detailed.

System Requirements

The Solar dynamic system (SDS) has been designed to supply the same power as the photovoltaic baseline system, i.e. 120 kW electrical output to the satellite. This will allow a direct comparison to be made between the two competing technologies in terms of mass and cost. In addition, the use of the Energia launch vehicle imposes a number of requirements which are as follows:

- Payload mass = 15 tons into polar orbit
- Fairing diameter = 6.1 m (dynamic envelope), 6.7 m (static envelope)
- Fairing height = 42 m
- The spacecraft must be designed to fit within these constraints.

It has been attempted to use the same platform design as the baseline option whenever this does not comprise the performance of this option. For example the SDS satellite needs a large radiator to reject waste heat and the photovoltaic system does not. This approach allows a reduction in the overall cost and in the schedule risk and possibly would allow for the integration of the SDS as the power source for the baseline option.

Spacecraft Configuration Options

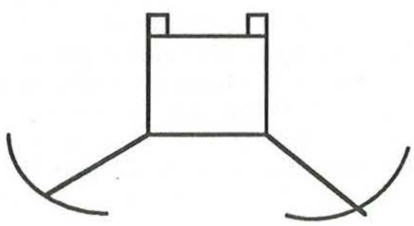
The spacecraft design is determined by the selected Solar Dynamic System and in particular the large solar concentrators required to collect the necessary heat energy. An area of approximately 280 m^2 is required for the solar concentrators to generate 120 kW to the satellite. An important consideration is that the satellite should be symmetrical. Otherwise the spacecraft would have to compensate for external torque's that are created by solar radiation.

Solar Concentrators Concepts

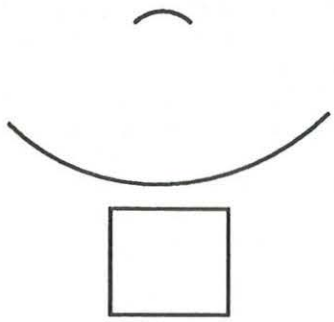
Since clearly the concentrator is too large to be launched in its deployed state, some method of stowing the concentrator is required. Unfurl able mesh antennas as used on the NASA's TDRSS satellite cannot be used to collect solar radiation since they work at a different wavelength. Inflatable space rigidified antennas are a possible candidate as concentrators, using an aluminized coating to reflect the solar power to the receiver. However, it will be very difficult to achieve the very high precision surfaces which are required to obtain the high concentration ratio necessary. Therefore inflatable space rigidified antennas have been rejected as a solar concentrator for this design. Another possibility is to have folded mirrors which unfold in orbit but these require a mechanism to unfold the mirror which may increase the risk. Therefore, for the solar concentrators, a petal fold out design is proposed using overlapping mirrors which are deployed in orbit. Small stub separators are used to keep the mirrors separate in the stowed configuration thereby protecting the mirrors.

Possible Spacecraft Configurations

A number of schemes are possible to mount the concentrators onto the platform. One scheme is to mount two reflectors, each with a diameter of ~13 m to the platform on opposite walls of the spacecraft, i.e., side-mounted, each supplying one receiver. Another is to mount a single concentrator on top of the platform. This scheme would use an Cassegrain type configuration with a primary and secondary mirror to focus solar heat back onto the receivers. Since only one concentrator is used the diameter of the dish is larger at ~19 m. Schematics of both designs are shown in Figure 10.3.10.



Satellite with two concentrators side-mounted



Satellite with single concentrator top mounted

Figure 10.3.10 Possible Deployed Satellite Configurations

With a single concentrator mounted on top of the satellite, the deployment can be very simple, essentially this design only requires the petals of the solar concentrator to be deployed. On the other hand, using side mounted concentrators requires first, that each of the concentrators be deployed away from the satellite before the petals are deployed.

Mounting the concentrator on top of the satellite is also very efficient at utilizing the Energia fairing volume, since the payload height is much higher than the payload width. In the side mounted configuration, the envelope of the stowed concentrators restricts the size of the satellite (a 6 m diameter) although it may be possible to accommodate the stowed containers on top of the platform. Therefore the selected concept is to have a single concentrator mounted on top of the satellite as shown in Figure 10.3.11.

With this configuration, the solar concentrators must always be pointed at the Sun with the phased array pointed at the Earth. To do this it is required that the attitude control subsystem be used to rotate the spacecraft around the orbit so that the phased array remains Earth facing. The concentrator requires a 0.1° pointing accuracy towards the Sun from the spacecraft to maintain a good efficiency.

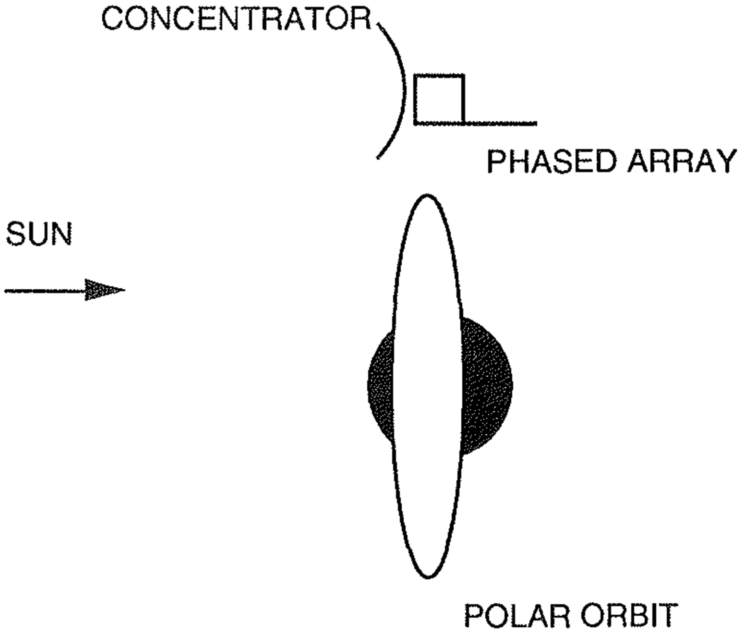


Figure 10.3.11 Schematic of a Satellite in Orbit

Radiator Configuration

To reject the waste heat, a large area of radiators is required. Conventional flat radiators have been proposed which directly radiate the heat into space. Heat pipe radiators being very efficient and a relatively mature technology have been baselined. It has been estimated that -120 m^2 of area will be required. This may be achieved by using three of the four large sidewalls of the platform. The fourth side is taken up with the phased array assembly. Using this approach each of the sidewalls will be 4m wide by 10m long retaining partial commonalty with the baseline design. This gives the required 120 m^2 of radiator area.

The large concentrator will also act as a shield for the radiators from the solar radiation, maintaining the high efficiency of the heat rejection process.

Phased Array Configuration

The phased array design is different from the baseline since for this option, the phased array is mounted on the sidewall rather than the top floor of the satellite. It is proposed that the basic structure of the satellite by a rectangular box with each side being 4 m by 10 m. This would easily fit within the Energia fairing. It is proposed that the phased array would consist of three sections, each 3.3 m by 10m, two of which are deployed in-orbit so that the required total size of 10m by 10m is achieved. These phased arrays sections could be stowed against the radiator walls during launch since they of similar size. This has the advantage of providing thermal insulation of the radiator walls during the transfer orbit.

SDS Power Generation Design

Solar Concentrator

The concentrator is needed to provide a solar flux density of 1.7 MW/m^2 on the collecting area of the generator. The solar flux of 1.3 kW/m^2 is focused using a parabolic shape concentrator. The architecture of this component uses 24 petals, each 8.5 meters high, covering an area of 282 m^2. Calculations have been performed using formulas presented in [Hedgepeth, 1978]. This system is coupled with a smaller concentrator in a cassegrain configuration to direct the solar flux back to the spacecraft. An additional reflector is needed to reflect the flux directly onto the collecting area. This design is presented in Figure 10.3.12.

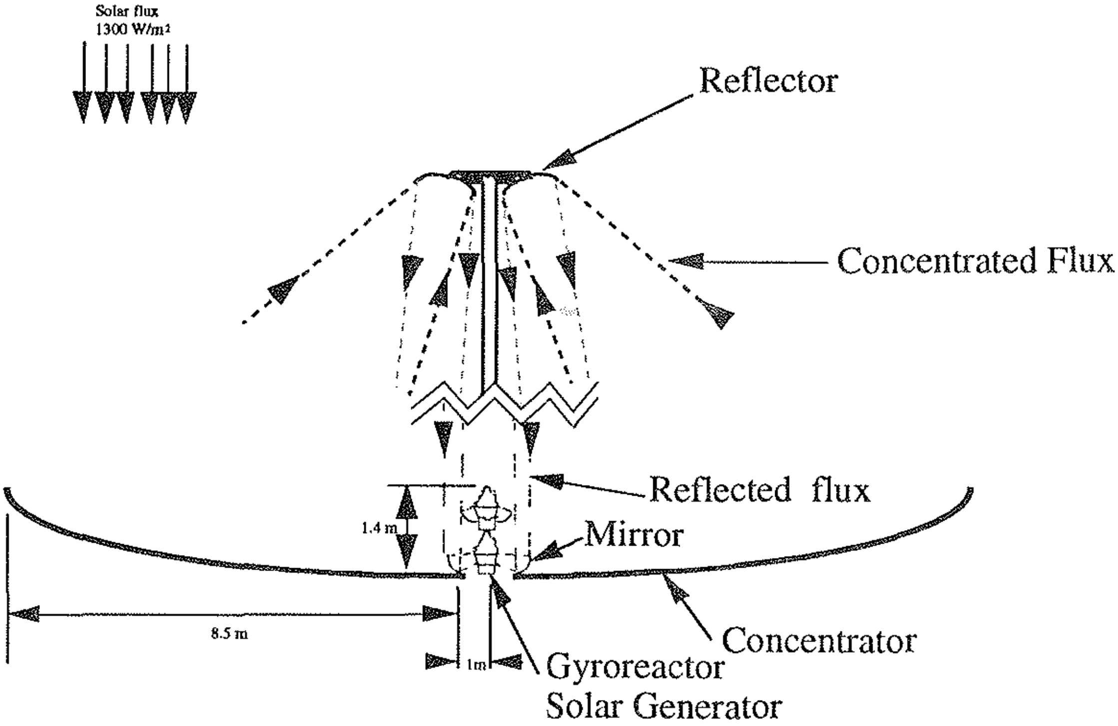


Figure 10.3.12 Solar Dynamic Concepts Using a Gyroreactor

Currently, the reflecting surfaces have an efficiency of 0.9. This type of configuration using three reflectors imposes a mass penalty of ~200 kg compared to having only one concentrator which reflects the energy directly onto the collecting area. However for structural reasons during launch, it is important to minimize the weight of equipment's that are located on the tower, ie.,., the receivers are heavier than a secondary mirror. This approach also minimizes the vibration on the spacecraft, due to the high rotation speed of the engine.

To estimate the mass of this subsystem we have considered a unit mass of the collector of 2.5 kg/m^2. This weight is equivalent to 375 μm thick sheet of aluminum. It seems very low but it is a very conservative number. Some ultra lightweight concentrators are designed with a unit weight of only 0.1 kg/m^2. So, this concentrator has a total mass of 700 kg to which has been added 50 kg due to additional support structures.

Gyroreactor Engine

The gyroreactor briefly presented in the section 7.1.4 New Technologies and described in [Baillly du Bois, 1991] has been chosen as the power generator to be used in this concept. This engine is a new technology which has not yet been demonstrated. We are fully aware of the risk taken by using this technology but it is so promising that additional investigation is necessary. As mentioned earlier this reactor offers an 45% thermal efficiency which directly lowers the radiator and the concentrator mass.

The proposed gyroreactor prototype weighs 50 kg and could generate 50 kW with an energy density of 136 kW/m^2 on the collecting area. The temperature on this surface is 850 K. In order to produce

120 kW, two engines are required with slightly improved performances than the actual proposed design. An gyroreactor of 80 kg generating 60 kW, using a solar flux density of 170 kW/m^2 is the design selected for this solar dynamic system. The estimate for the mass of each gyroreactor has been increased from 50 to 80 kg to incorporate a mass margin in the development of this equipment.

Compare this generator to more classical types such as the solar Brayton cycle the following operating characteristics should be high-lighted. Concerning the rotors speed, the gyroreactor rotates at about 20 000 RPM which is two times less than the Brayton's turbine. The turbine inlet temperature is also two times less in the gyroreactor (476 K vs 1013 K). These numbers suggest that the life time of the gyroreactor will be longer. For the space station freedom design solar dynamic system a 7 to 10 years life was planned. The Space Station Brayton generator has been designed to weigh 145 kg and to generate 40 kW. Compared to the system discussed here it is twice as massive.

Radiator

A radiator is needed to get rid of the excess heat. A classical heat pipe radiator is used in this system to radiate 73.3 kW. For the radiator's material, we have considered an emissivity factor of 0.8. The radiation temperature will be 400 K. So, the area needed in this case is -126 m^2. Using the assumption that the radiator mass is proportional to its area and that the unit mass for such heat pipe radiator is 5 kg/m^2 this gives a total radiator's mass of 630 kg. This consists of 3 radiator panels, each one weighs 210 kg

Costing

The estimation of such system is very hard to evaluate because this is new technology which is yet to be developed. Additional cost margins have been applied. The cost evaluation is based on the mass of each subsystem. Development, production and management costs have been considered. We considered two gyroreactor engine of 80 kg each, 24 petals concentrator and 3 panel radiators. The cost summary is seen in Table 10.3.13. This high cost is mainly due to the development of new technologies and will be reduced for follow on satellites. However this price is competitive with the Si solar array described in the baseline. The total mass of the system is also less at -1.5 tons and so for launchers which are dependent on the mass of the payload, further cost advantages are possible.

Table 10.3.13 Cost Summary of the Solar Dynamic Subsystem

Engines	\$90M
Concentrator	\$35M
Radiator	\$5M
Sub-system integration	\$10M
TOTAL	\$140M

10.3.4 Ground Segment

The power density on the ground is very low, and some sort of collection scheme must be employed in order to rectify the RF signal. The focusing of the incident power can be performed by connecting individual elements in parallel. The focusing is performed in order to operate the rectifying elements at a reasonable efficiency level. In any case, focusing of the received power will introduce beam focusing in the rectenna system. When the satellite elevation is 35° the incident power density is estimated to be 0.02 W/m^2. Assuming a 100 mW threshold for the rectifying diode, a 5 m^2 antenna is required for each diode. A 5m^2 antenna corresponds to a 1.3 m parabolic dish antenna. The beam width of a 1.3 m dish is approximately 0.4°. The footprint of the antenna along the satellite track is roughly 7 km. Since the satellite is moving at 7 km/s, the visibility-time as seen from the ground station will be only one second. Clearly this is not desirable for a power beaming experiment.

To able to monitor the incident field for the entire period of visibility, it was decided to use several small tracking dish antennas for the receiver. The configuration is shown in Figure 10.3.13. The central antenna serves as the master station transmitting the pilot beam to the satellite. The slave stations are distributed over an area corresponding to the satellite-beam footprint size. The network of receiving stations will be able to monitor the beam from the satellite and verify the performance of the phased array.

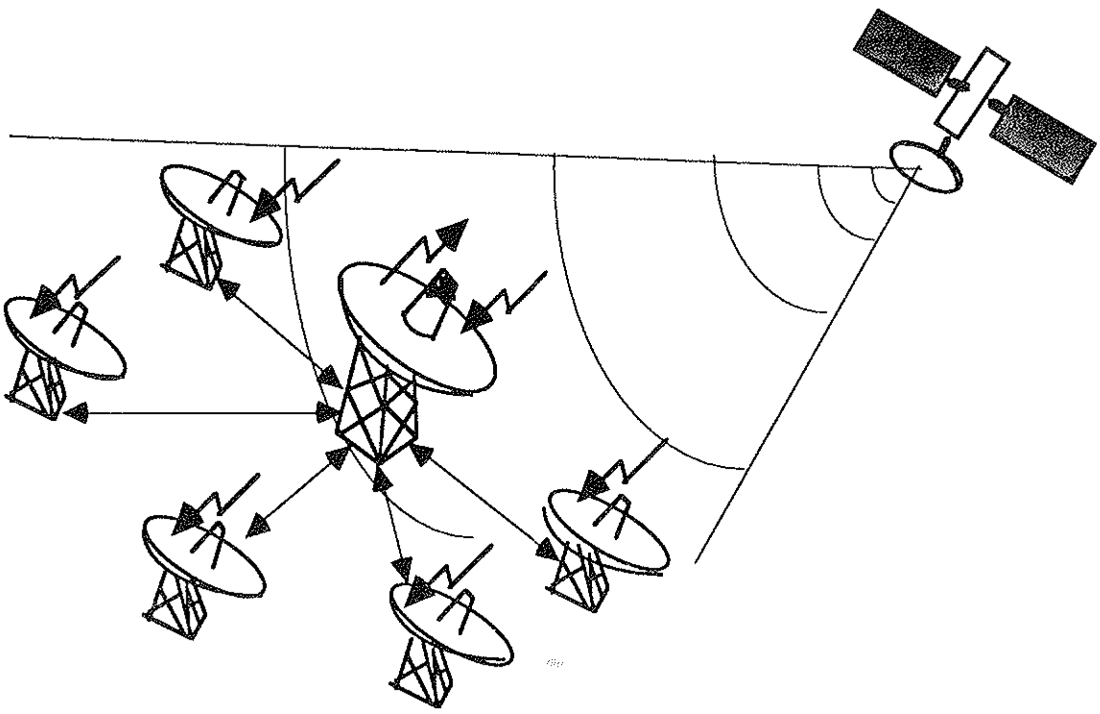


Figure 10.3.13 Ground Segment Configuration

The tracking of the satellite could be done by orbit prediction, but the master station could in addition have an automatic beam locking function which assures rectenna pointing in the direction of the satellite. The satellite velocity is 7 km/s corresponding to a ground measured angular velocity of approximately 0.4 °/sec.

Because the rectenna must have a tracking function to absorb the incident power, beaming to the Antarctica was not found to be feasible. In fact, microwave beaming from a polar orbit is only efficient when the power density on the ground is sufficient to eliminate the need for focusing before rectification. The location of the receiving station should be at a high latitude in order to obtain maximum temporal resolution.

10.3.5 Scheduling

The following can be thought of as a template for planning the space to Earth demonstration phase. This schedule is not dependent on the specific design or concept chosen and even does not restrict us to one example. For instance the entire schedule could represent the proposed 800 M US$ project or this project could be a small section contained under the "Power generation and transmission research and development program" block.

A space to Earth demonstration phase begins with the research and development program for power generation and transmission. A proposed starting date and duration is 01/01/1995, lasting 42 months. The actual start of Demo 2 begins on the 1st January 1997 starting with a concept development, lasting 18 months and followed by 18 months design phase. Schedule chart for Demo 2 is seen in Figure 10.3.14 and the task timeline chart in Figure 10.3.15.

The actual development phase (phase C/D study) begins in mid 1999. It is divided into the three typical models: structural & thermal models (checks if the structure and thermal control is good, no flight equipment), engineering models (like real satellite, but with standard components), and protoflight models (real satellite with space qualified electronic components). Each of these models are divided into three phases. The first two phases run parallel and are the payload - power generation and transmission (P/L) and the rest of the spacecraft (platform) excluding the payload (P/F). The third phase is the assembly integration testing (S/L) and integrates these first two phases. Table 10.3.14 gives a detailed explanation of the schedule of the actual proposed development phase.

The four months (July-October) of 1994 will be used for the transport and integration of the satellite. Actual launch is expected to be on the 1st November 2004. It is expected that the lifetime of the beaming experiment be one year with evaluation of results running continuously.

Table 10.3.14 Demo 2 Schedule

TYPE OF MODEL		START DATE	PHASE LENGTH (months)
STRUCTURAL AND THERMAL MODELS	P/L	01-07-1999	18
	P/F	01-07-1999	18
	S/L	01-07-2000	12
ENGINEERING MODELS	P/L	01-01-2001	18
	P/F	01-01-2001	18
	S/L	01-01-2002	12
PROTOFLIGHT MODELS	P/L	01-07-2002	18
	P/F	01-07-2002	18
	S/L	01-07-2003	12

10.3.6 Summary and Conclusions

Conclusion of the $800 M Design Example

Our conclusion is that a space to space demonstration with the current technology would cost billions. Thus our recommendation to the client is that the stage following the $80 M demo is a research and development phase resulting in a proposal for a space to Earth demonstration. We have identified several areas of technology development that would facilitate a space solar power program; the primary develop needs are in the areas of 35 GHz phased arrays, high power generation devices, and the rectenna receiving elements.

Within the context of a Space Solar Power Program, the space to Earth demonstration falls in a crucial position that could be a turning point in determining the future of solar power for Earth. This stage of the program occurs after small-scale demonstrations showing the feasibility of beaming Earth to space and from space to space, but before we have reached the technical or commercial viability of a large-scale space to Earth power use. To bridge this gap between theoretical and practical feasibility may take more than one step. Thus we conclude with an additional demonstration that will lead us into the demo 3 phase (1 MW) of a space solar power program.

100 kW Early Commercial Design Example

High Level Requirements

The purpose of this design example is to demonstrate technical and commercial suitability of solar power satellite concept. Indeed, it has been demonstrated that the previous step ($800 M design example) was limited to technological experiments only, regardless any further use of the power supplied. Therefore, the goal of the present step is to prove that energy available in space can be beamed down to Earth for actual use afterwards. In addition, this step shall prove also the commercial potential of the solar power satellite concept.

Consequently this step shall be a logical follow-on to the previous one in the sense that it assumes that the basic technical problems such as large power generation systems and large phased array antennas would have been (at least partly) solved in the $800 M space to Earth demonstrator. Thus, this section focuses on beaming significant amount of power to ground rather than technological experiments, with relaxed cost constraints. Furthermore, this example stands before the 1 MW precursor. The following set of high level requirements can be proposed:

- 100 kW delivered on ground ;
- 10 years lifetime
- Time schedule: within 15 years
- Cost: to be defined, in the order of a few billions.

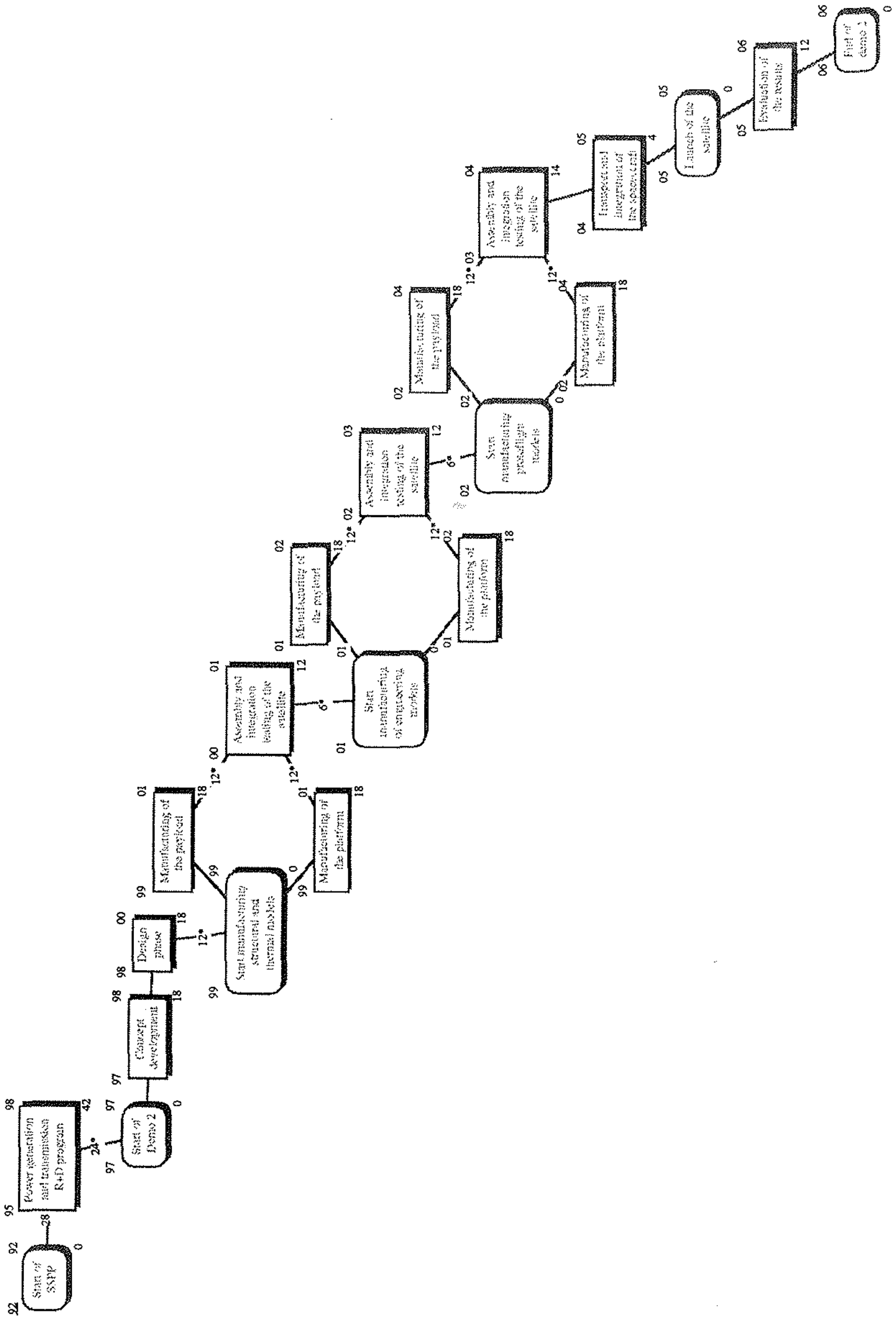


Figure 10.3.14 Demo 2 Tasks

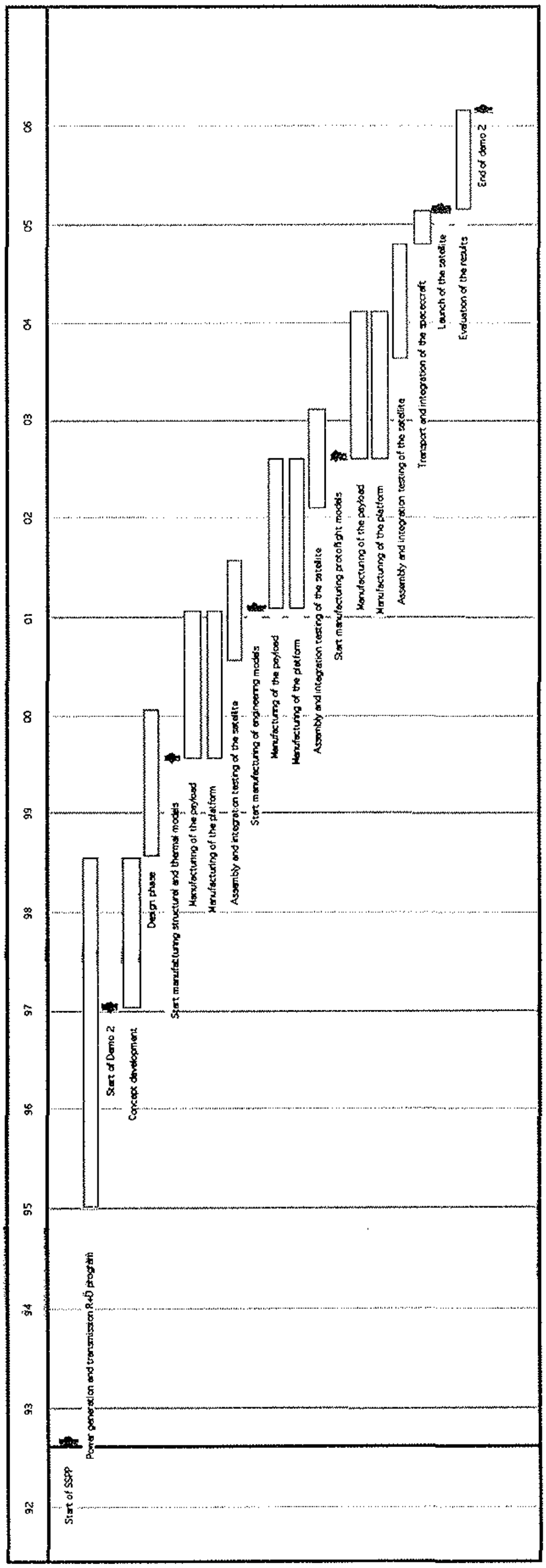


Figure 10.3.15 Demo 2 Schedule Timeline

Mission Analysis

Orbit Selection

In order to have a valuable commercial demonstration, the amount of power delivered to users shall be supplied during a relatively high percentage of the time. In this respect the geostationary orbit can be selected, associated to a near equatorial rectenna.

Launcher

Since the spacecraft will weight a few tens of tons (typically 20 to 50), an heavy lift launcher will be necessary. On the other hand, this demonstration is relatively short term, so that the launcher that might be used will stay in the range of present ones. Consequently, Energia can here be envisaged: with a mass of 18 tons into geostationary orbit, the spacecraft could be launched in two or 3 times and robotically assembled into GEO.

Beaming Analysis

A preliminary analysis have been performed, using the following differences with respect to the $800 M demonstration:

- solar array: 2500 m^2 instead of 1000 m^2
- No batteries used for additional power generation
- Use of Gallium Arsenide solar cells with 15% after 10 years instead of 10% efficiency Silicon cells after 5 years
- Phased array: use of a 50m x 50m array instead of 10m x 10m
- Altitude: 36000 km instead of 1000 km
- Rectenna: 6 km x 6 km instead of 1 km x 1 km, based on 30 m diameter motionless concentrators prior to rectification.

This gives in the range of 100 kW on ground. The detailed power budget is indicated in Table 10.3.15.

Table 10.3.15 Power Budget

	Polar	Design	Geostationary design
	Beginning of visibility	Middle of visibility	Permanent visibility
Satellite elevation (°)	35	90	90
Solar Array surface (m2)	1000	1000	2500
Cells efficiency (EOL)	10%	10%	15%
Cell layout ratio	90%	90%	90%
SA power output (W)	1.22E+05	1.22E+05	4.56E+05
Power conversion & distribution ratio	90%	90%	90%
Battery subsystem output	1.00E+05	1.00E+05	
Power subsystem output	2.09E+05	2.09E+05	4.10E+05
frequency	3.50E+10	3.50E+10	3.50E+10
wavelength	8.57E-03	8.57E-03	8.57E-03
Transmitting Antenna surface (m2)	100	100	2500
Array Efficiency	0.7	0.7	0.7
Gain	1.71E+07	1.71E+07	4.28E+08

Satellite output power (W)	1.47E+05	1.47E+05	2.87E+05
Satellite distance to rectenna (m)	1.55E+06	1.00E+06	3.60E+07
Beam width (rad)	8.57E-04	8.57E-04	1.71E-04
Rectenna size for 3dB power collection (m)	2316	857	6171
Chosen rectenna size (m)	857	857	6171
Atmospheric losses (dry conditions)	5%	5%	5%
Power flux density at center of beam (W/m2)	8.30E-02	1.99E-01	7.54E-03
Collected power on rectenna (W)	3.32E+04	6.96E+04	1.36E+05
DC conversion efficiency	80%	80%	80%
Power at system output (W)	2.66E+04	5.57E+04	1.09E+05
ground concentrator diameter (m) for 0.1W power (at center of the beam)	1.21	0.50	13.27

Cost

Table 10.3.16 Preliminary Allocation

Launch (3 Energia launches):	$250 M
Space segment:	$3000 M
Ground segment (rectenna, control center operations):	$1000 M
Robotic hardware development for space assembly:	$250 M
Total:	$4500 M

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10.4 Megawatt Class Demonstration

This chapter presents a conceptual design for a 1 MW commercial precursor. The main departure from similar concepts is the LEO assembly using EVA operations rather relying on robotics. The structure retained is a prismatic one with 100 m vertices. The sun energy conversion process is based upon GaAs solar cells and the energy is beamed at 35 GHz to keep the antenna dimensions in the frame of the spacecraft. It is also shown that high orbits, GSO or near GSO are of interest for operational reasons. The mass assumption shows clearly the dominant importance of the antenna subsystem.

To put the design in perspective, a set of alternative concepts are summarized: the Japanese SPS 2000 project, and two generic planar platforms in addition to the precursor concept. In addition, future trends are mentioned showing that integrated array technology could drastically change the structure of the spacecraft. A schedule for realization is provided, consistent with the rest of the project. The study is closed by a summary and a set of conclusions.

10.4.1 Constraints

The following constraints are imposed on the system design in order to limit the cost. One or two Energia class launchers and one space transportation system (STS) assembly flight using manned operations are used to perform assembly in low earth orbit. Those classes of launchers have been retained as basis for launch and assembly. It does not mean necessarily that they will still be available or be the cheapest launchers in the future. The main point is to have payload and trajectory references with which to develop the design. As much as possible, proven or highly cost effective technology should be selected. The total target cost of the system is to be in the range of 1 billion US dollars, and is erected by an international crew.

Orbit Choice

The choice of an orbit is driven primarily by mission considerations. In this case the main mission is to beam power to ground at optimum efficiency in order to provide a practical quantity of energy for use on Earth. This means that orbits leading to a short time of visibility are not well suited as long as no efficient device is available to store large amount of energy provided in a short time. Since this kind of technology is not foreseen in the near future, it is then necessary to consider higher orbits, up to GEO, to allow for an adequate time of visibility. A brief listing of the approximate visibility windows is given in Table 10.4.1.

Table 10.4.1 Visibility for Various Orbits

Orbit Type	Frequency of Pass	Duration
LEO 300 km 28.5	Infrequent	Minutes
Sun Synchronous	Infrequent	Minutes
Equatorial 1000 km	90 minutes	Minutes*
Equatorial 20000 km	12 hours	6.4 hours*
Equatorial 36000 km	Continuous	Continuous

*Assumes rectenna can receive a signal from 30 degrees above horizon.

Since the system is required to be operational for many years, and in order not to use many spacecraft (at least in early phases), it is of some interest to avoid any seasonal effect. This consideration precludes elliptical orbits for a single spacecraft, that is, the orbit should be circular.

The visibility time does not favor polar orbits for which the combined rotations of spacecraft and earth result in a very short time over the same point. Furthermore, as long as the elements are launched from the ground, plane changes to the polar orbit are costly. In addition, direct launch is not feasible within the study constraints due to the manned assembly limitation.

It is also important to consider the location for the power delivery. For the precursor system it is believed that the first use will be in a subsidized context such as developing countries. The equatorial plane looks like a preferential location in this respect. As will be explained in the rectenna section, the equatorial plane also allows simple solution to the problem of rectenna configuration.

From the discussion above, it can be concluded that GEO orbit could be a good choice, but this orbit is already a scarce resource and so it may be difficult to obtain a desired slot. There are also potential Electromagnetic Interference (EMI) problems to consider. In these respects, a lower orbit may be preferable. However, lower orbits have significant drawbacks in terms of period, visibility time from ground, aerodynamic drag and gravitational perturbations, hence precluding these orbits from efficient long term usage.

An alternative to GEO could be: circular equatorial altitude: 20309 km, 12 h period, with satellite and ground station aligned at the local zenith sun transit, West to East rotation

This orbit allows a visibility time from ground station of 6h24' per orbit, and is compatible with conservative energy storage devices such as fly wheels. This has the advantage of not requiring exotic high energy storage devices on the ground which store MW in a few minutes. An equatorial orbit coupled with a 12 h period allows flight over the receiving station each day at the same solar time. This will facilitate the ground operations. Furthermore, the orbits in the range of 20000 km present a very clean environment in terms of orbital debris and solar particles.

All of these concepts require assembly in LEO (typically 350 km 28.5°) to allow heavy launchers to deliver the spacecraft to the assembly altitude with one launch. It is also necessary that the Shuttle be able to reach the same location in order to perform assembly operations and provide a life support base for the crew. The Extended Duration Orbiter will have an ability to remain in orbit for 30 days. At an inclination of 28.5°, the STS is capable of 40,000 lb_m, or 18 Mt, which diminishes for a different orbital plane. There is thus a trade off between orbit (altitude and inclination) and the cargo capacity of the two vehicles launched from different launch sites. As an example, the Energia is capable of placing 70 Mt into a 28.5° orbit, which allows the Shuttle to remain for the maximum duration. A second Shuttle mission may be required for the larger, more advanced platform. The feasibility of construction of the system is discussed hereafter.

10.4.2 Platform Design/Sizing

Manned vs. Automated Deployment

The literature regarding structures of interest to the 1 MW-class was reviewed and the following points were developed. First, a large-scale platform of the 1 hectare area (100 m x 100 m) or larger would be very difficult to deploy automatically and yet exhibit the desired stiffness. It was noted that in 1990, the design of the Space Station Freedom was significantly changed (Fisher-Price Study) due in part to the perceived difficulty of orbital assembly of large structures. To attempt to construct a large platform automatically at this juncture would require a development and testing program, particularly with the use of a robotic-assisted system. This was thought to exceed the cost and scope of the current effort (other smaller platforms with deployable structures are covered in other sections of the report). The initial concept sizing used Space Station Freedom 5m truss which were originally intended for the full dual-keel design (an actual design would optimize the truss design and save, for example, 80% of the original development costs). The advantage of using this was that the mass, stiffness, construction times (the result of many hours of buoyancy tank experiments as well as an STS flight during which a large element was constructed) and other parameters are well understood. From this, it was argued that the most conservative and therefore least expensive large structure suitable for an early Solar Power Satellite demonstration was the result of this strategy.

Examples of erectable structures are given in the following Figures (Mikulas, 1988; Katzberg, et al, 1990). Figure 10.4.1 shows 5 m truss assembly inside of the Shuttle cargo bay. In Figure 10.4.2, the truss assembly can be seen as it is gradually constructed. In previous on-orbit experiments (which correlated well to buoyancy tank experiments), the truss elements could be assembled at approximately 1 tubular element per minute. This results in truss assembly of 30 m per hour, such that a 100 m section requires over 3 hours for assembly. Examples of large scale elements are given in Figure 10.4.3, showing the 'dual-keel' space station in which the dimensions of the dual keel are close to 100 m x 100 m. The last example, shown in Figure 10.4.4, illustrates the construction of a Mars mission construction bay. Large truss segments, once assembled, are then moved into place with a manipulator arm system. The 100 m x 100 m planar array, followed by the prismatic array, are regarded to be simpler to construct and integrate than both the dual keel Space Station (for example, there are no fluid lines, radiators or alpha joints) and the Mars mission construction bay.

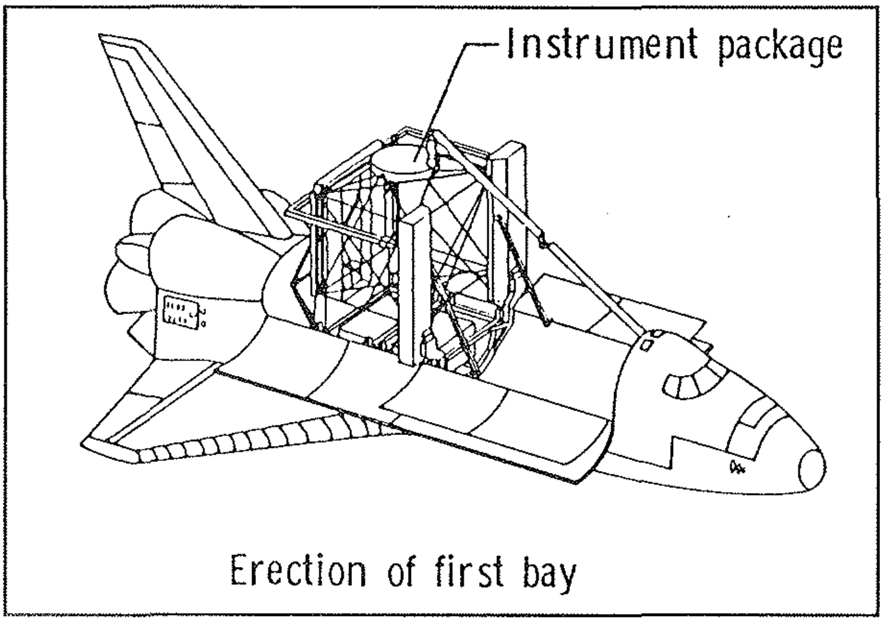


Figure 10.4.1 Erection of the First 5m Segment.

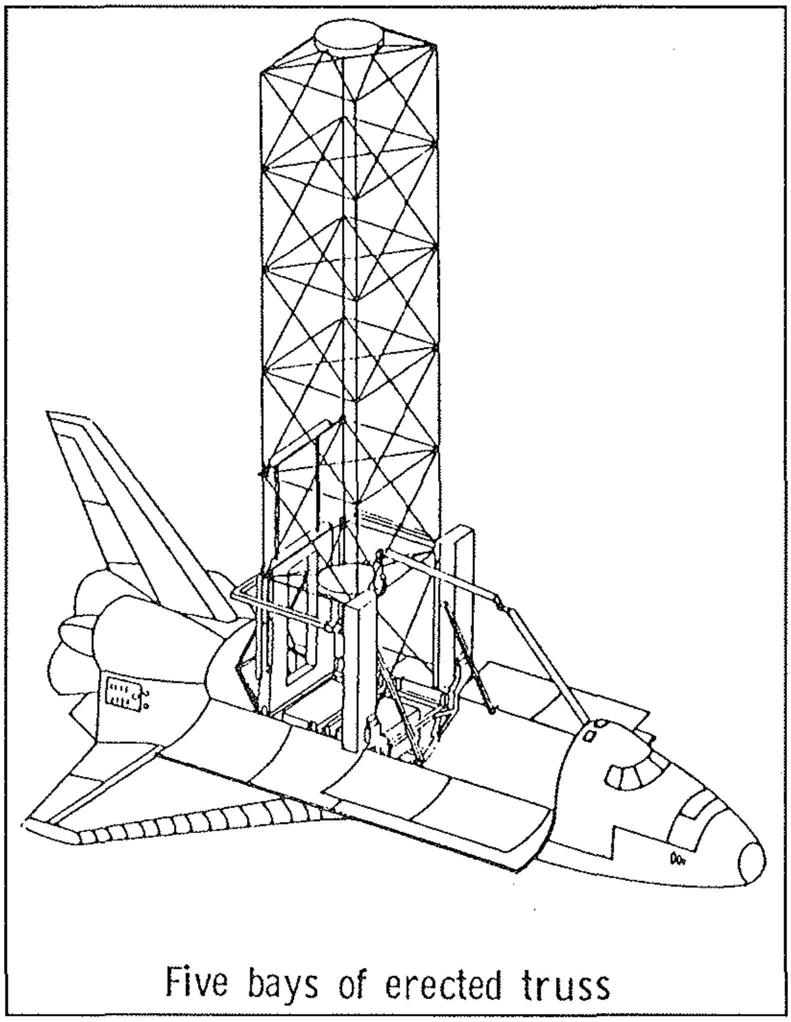


Figure 10.4.2 First 5 Segments from the Shuttle Cargo Bay.

Assembly and Construction

The construction and assembly is a critical element of the designs considered in this section. As indicated previously, the primary assumption is that the current and mid-term state of the art regarding robotic assembly is such that erectable structures are more desirable.

The assembly sequence is briefly outlined as follows. For the simple planar array, a single STS flight is assumed. The construction would occur in the STS cargo bay. For the prismatic pre-commercial demonstrator, the following is assumed. An Energia is launched from the Baikonur launch site and carries 70 t of cargo into a 350 km, 28.5° orbit. Shortly thereafter, a Shuttle is launched into the same orbit and performs the required proximity operations. The assembly is then completed in LEO. This is then followed by a verification and operation period before the transfer into the operational high equatorial orbit.

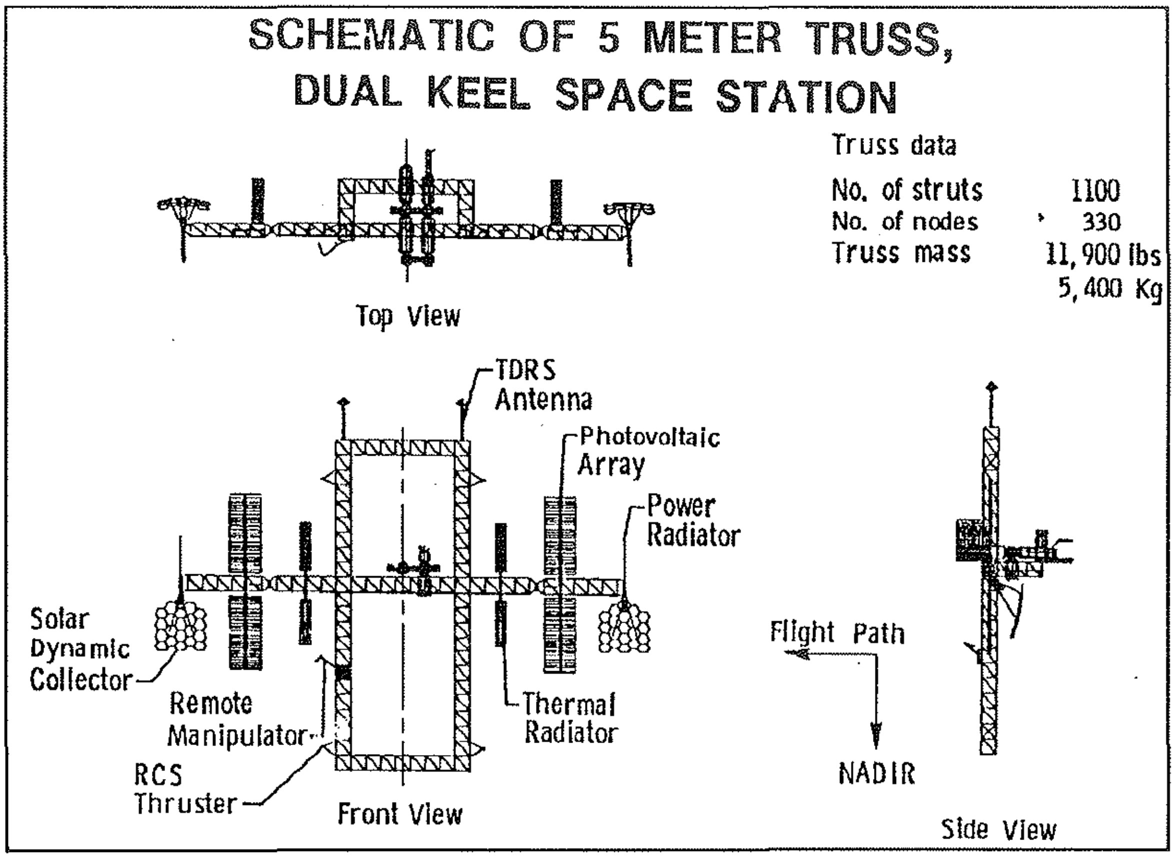


Figure 10.4.3 The Dual-keel Space Station.

MARS ASSEMBLY DEPOT

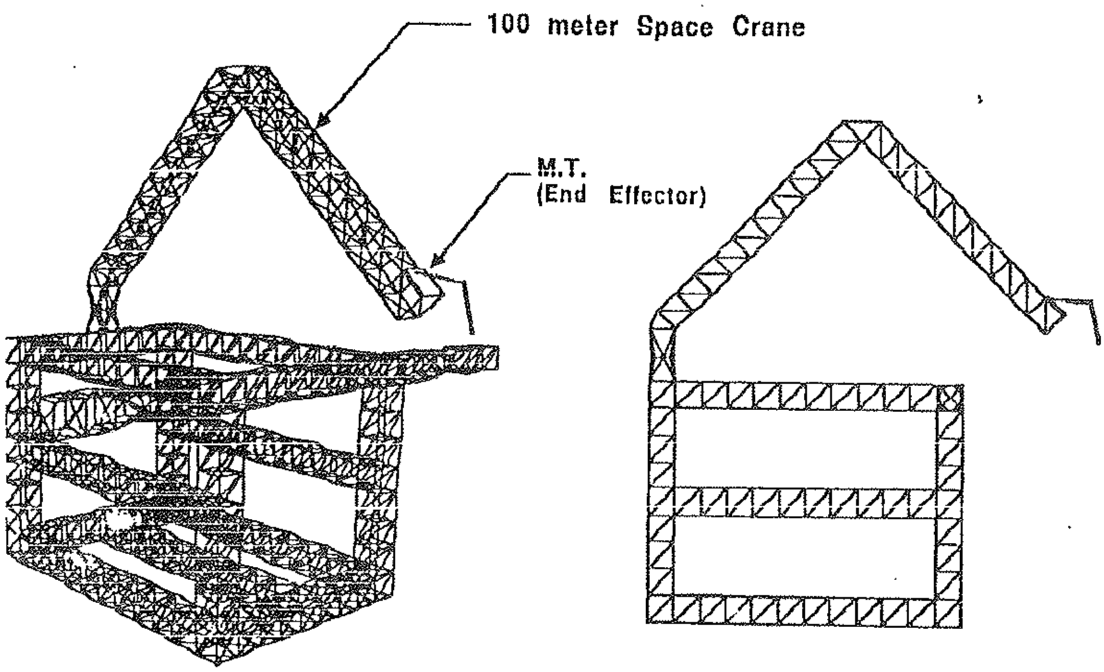


Figure 10.4.4 Erection of the Mars Mission Vehicle Assembly Bay

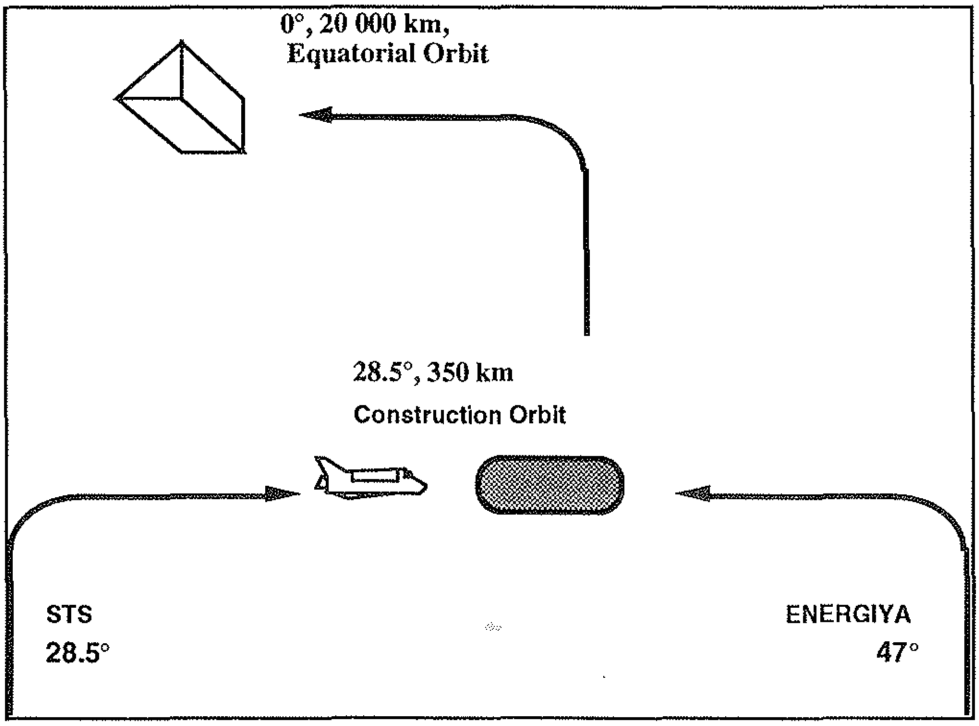


Figure 10.4.5 Outline of Launch and Assembly Sequence.

Basic Topology Trades

There were a variety of basic platform topologies that were considered. A large planar array with gravity gradient stabilization; a conventional planar array in which the long dimension of the array flies parallel to the flight vector; and finally, the current choice is a large prismatic concept. Recent studies (SPS-2000) have explored the prismatic concept and it appears attractive because of the greater stiffness that results. However, the planar array that is currently explored in this section utilizes space station derived trusses combined with a tension wire concept which may result in sufficient stiffness. The choice for the current prismatic demonstrator concept is not necessarily due to the stiffness but to the flight stability which results The planar structure in LEO would be sensitive to gravity gradient perturbations and would therefore require an active control system or an extremely large boom to ensure passive stability control. Furthermore, the low time of visibility from the receiving rectenna to the transmitting antenna (less than 10 min for 1000 km) precludes, or at least significantly penalizes, the use of the delivered power at ground level or requires the use of huge movable solar arrays or antenna..

Power Collection: Photovoltaic vs. Solar Dynamic

The solar dynamic concepts have certain advantages, in particular relating to increased efficiency which reduces the size of the overall collecting surface (that is, for a similar power output). In addition, the cost per unit area of the reflecting material is lightweight and much of the array does not have electrical subsystem connections. However, past studies have indicated that the radiator subsystem required to dissipate the thermal energy of any particular thermodynamic cycle is relatively massive, and perhaps requires an actively pumped coolant in order to adequately transport the thermal energy. In contrast, the steady development of lightweight and easily deployable photovoltaic arrays over the last decade has resulted in systems which are ready to be used for large scale space applications. For these reasons, the selection of a photovoltaic array for this class of demonstration platform seemed the most conservative and appropriate choice given the budget constraints associated with this project.

This choice allows the sizing of the solar arrays and then the sizing of the spacecraft:

- The total efficiency of the chain is the product of each element efficiency: solar array, power converter, antenna, and atmospheric path. The rectenna efficiency is taken into account at the ground level.

- Solar array is assumed to have an efficiency of 0.15 (GaAs technology),
- Power conversion efficiency (vacuum tubes and HT power) is assumed to be 0.8
- Antenna efficiency (including phase shifter) is assumed to be 0.9
- Frequency selected is 35 GHz (see transmitter discussion section)
- Atmospheric path coefficient is assumed to be 0.8 (reference. value at 35 GHz for dry air is 0.95)
- Total efficiency is 0.0864. With this value it is easy to determine the area The solar array area is: A= 10^6 / (1300 x 2 x 0.0864) = 8903 m^2.
- We have selected a basic structure of 100 m x 100 m per face in order to have some margins.

PV Material Selection/Suitability

Much progress has been made in the past decade regarding the development of efficient photovoltaic arrays which are applicable for use in the space environment. Of these, InP (Indium Phosphide) appears attractive because there is little degradation of an already high efficiency in the space environment. However, the development of 1 hectare or more of these cells might be costly at this point since the manufacturing process is difficult. Another choice includes the thin film a-Si (amorphous silicon) arrays, which can be produced in large quantity. The disadvantages seem to include a relatively low efficiency which has been found to degrade further in a space environment. Whereas a long term commercial project may choose the more expensive option of InP, the choice for a constrained demonstration project might be the thin film a-Si technology or GaAs if economically viable, with the potential of future upgrades if other mission factors (that is, the final orbit selection) permit. It should be noted that the selected orbits are extremely clean. As previously stated, for the 20309 km orbit, the electrons are the main concern even if ions can not be totally neglected.

Power Subsystem

As the solar arrays are fixed (that is, not mechanically movable with respect to the spacecraft structure), the amount of solar energy collected varies with the relative orientation of the spacecraft toward the sun. The energy goes from zero to a first relative maximum, then reaches a second maximum and repeats symmetrically to the end by reaching zero when the arrays are masked by the antenna as shown in Figures 10.4.6 and 10.4.7.

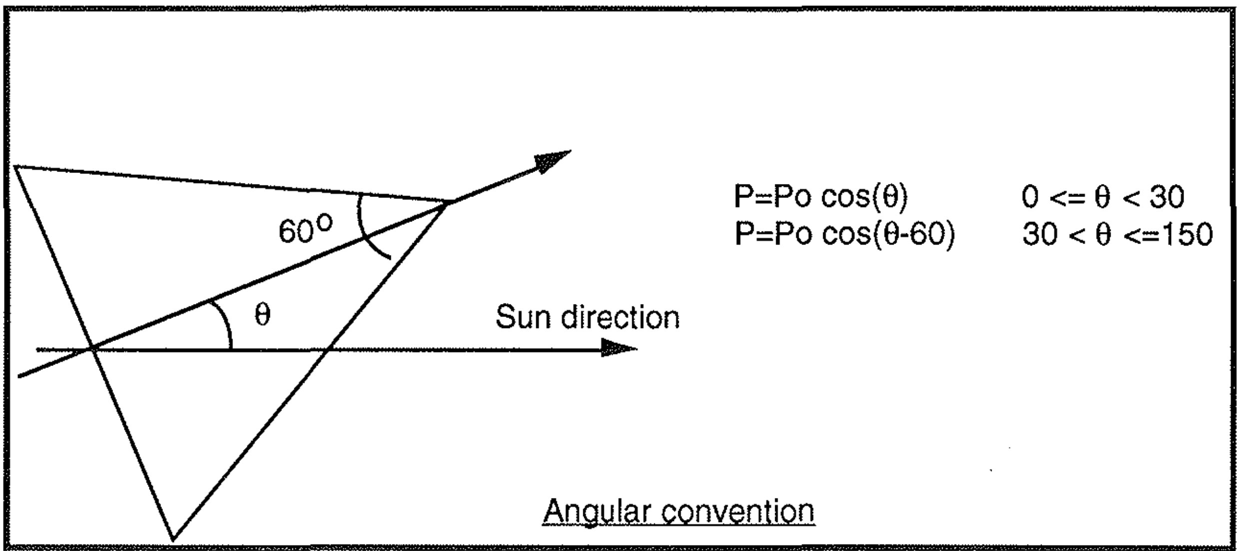
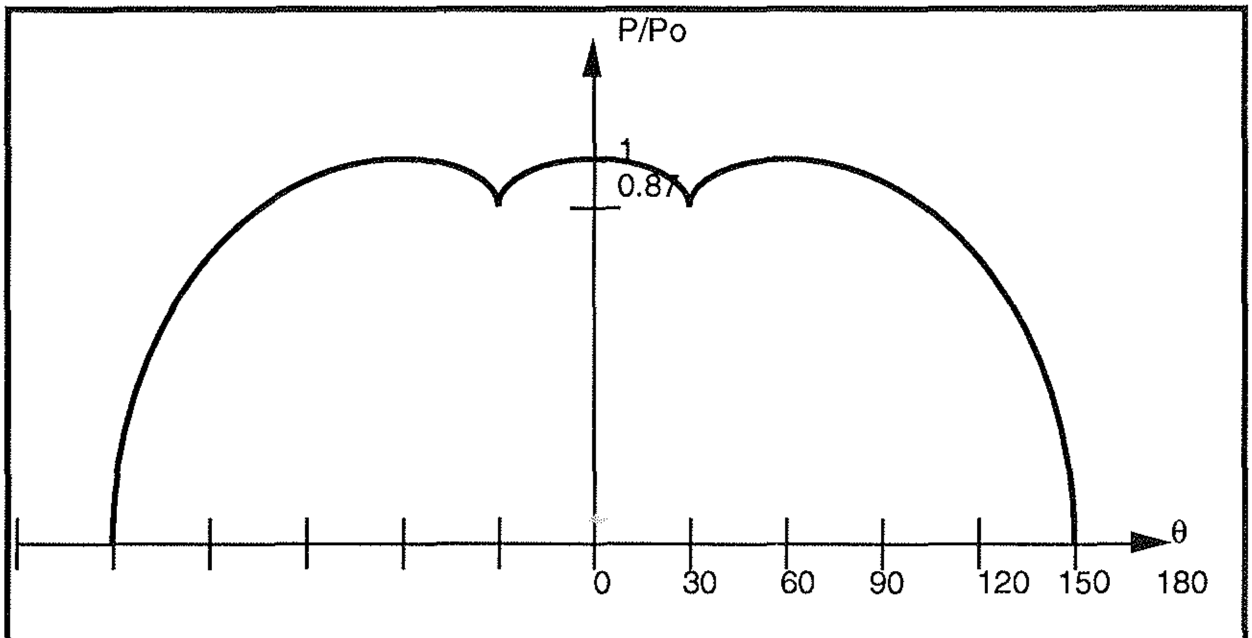


Figure 10.4.6 Angular Convention

To simplify the system design, it is chosen to cease the power transmission when the level drops below 80% of the maximum (when the two arrays are symmetrically facing the sun). The fraction of energy lost due to this choice is 21% of the total collected amount. This fraction does not justify increasing significantly the mass of the spacecraft by a battery or a fly wheel so as to store energy on board. Furthermore, this storage would occur during a phase when the power collected varies strongly, thus complicating the power subsystem. With the figures of efficiency used above, the energy transmitted during one pass is: 13 MWh beamed in 13 h for GEO and half of that for 20309 km orbit. It should be pointed out that the GEO option allows only to feed one receiving station when the 20309 km option could feed two stations and reach the same efficiency. The average level of delivered power during one pass is 1 MW for both cases.



Sun power received as a function of the angular position

Figure 10.4.7 Sun Power Vs. Angular Position

The power subsystem is organized around a set of solar cells arranged in a series-parallel cluster; the series structure is long enough to provide a voltage level allowing an easy DC/DC conversion (that is, over 160 V); the parallel structure is dimensioned by the current requirements. Except for the levels of voltage and current used, the architecture of the power subsystem is classical and could be similar to existing architecture, the input level varying only by 20%. It should be noticed that the variation of power level is directly related to the angle between the solar arrays. The analysis shows that an angle of 60° minimizes the difference between the maximum and first minimum of received power. In addition, this angle allows a symmetrical construction with elements of the same dimensions for each face.

Transmitter Design

The considerations for transmitter design are frequency selection, solid state or vacuum tubes, and a mechanically mobile structure or phased array.

The frequency choice is driven by two considerations: size of the antenna and atmospheric attenuation. The size of the antenna is limited by the size of the spacecraft, to say: 100 m x 100 m. The atmospheric attenuation is an external factor implying a higher level of power at emission.

With the antenna, assumed to be a phased array, it is difficult to predict accurately the transmission characteristics without using a simulation program. For any antenna a relationship exists between the diameter of emitter De and receiver Dr , the wavelength λ and the distance of transmission D. The general shape of this relation is:

$$Dr.De = K\lambda D$$

With K= 1.27 for parabola (considering power density at the receiving site) and 2.44 for a plane aperture (receiver in the first minimum of the diffraction pattern). The most stringent requirement is for the plane aperture that we would select for dimensioning purposes. With Dr=10 km, De=100 m, D=36000 km, λ should be less than 2 cm. This is a higher frequency than 15 GHz. To avoid interference with X band communications and to be in the first window of transmission , a possible frequency could be 35 GHz. A short survey of existing technologies shows that this frequency is the domain of the so called 'Fast Wave Tubes'. In this family, the most promising device is the gyrotron

(Firmain, et al, 1991). This frequency is selected for the system design, whether in GEO or at a 20000 km altitude.

The following discussion on solid states versus tubes will consider the robustness of this latter choice (for the near term). Solid state usually saves a lot of mass and electrical power consumption, especially at low power levels and for frequencies in the range of a few GHz. Nevertheless, at higher frequency and power level, solid state devices are inefficient, unreliable and even not demonstrated at present. Furthermore, considering a power in the range of 1 MW, and solid state devices handling a reasonable 100 W (not existing today above a few MHz), the antenna would require 10000 of those elements. Even without taking into consideration the problem of control of such an amount of active devices, the foreseen efficiency (based on actual designs) is in the range of 50% at 2.45 GHz (Kaya, Matsumoto and Akiba, 1991). At 35 GHz, a better efficiency is not expected in the near term. The amount of heat dissipation is a serious limitation, especially effecting long term reliability. On the other hand, vacuum tubes have already demonstrated performances above 100 GHz (Firmain, et al, 1991) with power levels higher than 10 kW and an efficiency in the range of 80%. The main weak point is the lifetime of the cathode, limited to around one year for ground systems in the range of 1 MW. Nevertheless, low power tubes in X band are already space qualified and show a lifetime in the range of 10 years. It is therefore expectable that those performances will be reached by gyrotrons in the near future. Furthermore, the need to use many amplifiers would suggest a power level per tube from 1 to 10 kW, extending simultaneously the expected lifetime. For the long term it can not be excluded that solid state devices will become an attractive option.

The geostationary orbit allows a fixed pointing antenna as long as the spacecraft is able to provide a stable platform. The antenna could be either parabolic or planar. For big dimensions the parabolic type is difficult to use due to size limitation by the launcher. A deployable technology could be considered but the uncertainty to obtain this kind of structure in a size large enough is high. Hence it seems more plausible to select a planar model, assembled in orbit. If a single micro waves source is used, this kind of antenna is relatively easy to obtain by distributing a set of slots on a planar surface. In our design a set of sources is used. These sources have to be phased with each other. The simplest way to obtain this result is to use a unique low power source feeding a set of amplifiers adjustable in phase to compensate for the relative discrepancies and aging effects. The design will then use 100 gyrotrons at 35 GHz. The antenna will be made of a set of 100 panels assembled in LEO and adjusted by optical interferometric means. The remaining deformations would be compensated by phase tuning of the amplifiers.

The previous design is suitable for GEO where there is no strong requirement on beam mobility. For the 20309 km design it is necessary to have a total beam deflection over 24° side-to side. It is then expected that the previous design will no longer be usable as the fixed pattern of slots does not allow for the steering of the beam in various directions. Hence, a design using 1000 sources of 1 KW each will be used, these sources being fitted on a composite structure to ensure rigidity, each one having its own radiator. The need for an individual phase shifter drives the design to that of one gyrotron directly feeding one phase shifter connected to the antenna face. The considerations regarding solid state versus vacuum tubes remain unchanged. It is then necessary to assess the required pointing accuracy. The accuracy could be considered as the minimum angle variation of the beam that allow it to remain in a given area at ground level. Assuming that a safety ring of 1 km exists around the rectenna and considering the altitude of flight the angular accuracy should be in the order of magnitude of (1 / 20309) x 57.3°= 0.03°. This accuracy is directly related to pointing accuracy of the platform for the GEO design. For the mobile concept this becomes a dynamic accuracy to be provided at beam deflection level. Considering a total range of beaming over the rectenna of 24°, the number of steps for the phased array would be: 24 / 0.03 = 800. Assuming a binary coding of the angle for the phase shifters, the closest power of 2 is 1024. Then the phase shifter should be capable of 1024 steps of 3 / 100°. This requirement would be probably relaxed if it is possible to show that the beam is not sensitive to phase quantum changes of individual elements, allowing a reduced number of individual phase patterns. On the other hand, a graceful degradation of the antenna array would require an individual control of each phase shifter to be able to face any configuration occurring in flight.

The speed for updating the array (assuming a central computer unit to control the phase) is determined by the time needed by the satellite to fly from one horizon of the rectenna to the opposite one. With the chosen orbit, and taking into account the Earth rotation, the time needed to deflect the beam 24° is 6h24' (to deflect it of 3 / 100° takes 28.8 s). Even with 10 bit phase shifters and 1000 devices, the data flow is well within the range of the present technology. This requirement could be increased if random phase shifts of each oscillator have to be compensated. However, even with 100 time more data this rate is still in the range of any network as a MIL STD 1553B already used in space.

Phase computation could be done in two ways: using the global attitude of the total array, (through an interferometric measurement of a reference signal beamed from the ground), or using an individual measurement at each element level. The latter solution allows a better shaping of the beam, and takes into account the effect of the atmospheric path on each individual element. However, it will also need some additional hardware and a heavier computational load. There is not enough information available to select one of these two solutions. It is probable that an intermediate approach could be successful, relying on interferometric measurement to determine the attitude and measuring at ground level the shape of the beam, then sending back to the spacecraft a correction signal.

Rectenna Considerations

Strictly speaking, the rectenna is not a part of the spacecraft design. Nevertheless it is impossible to ignore the constraints induced on (and by) this part of the system. Using a high altitude orbit and a relatively low size transmitting antenna presents the main problem of the energy density at ground level. The rectifying elements, based on diodes, exhibits an energy level threshold under which their efficiency decreases quickly. Using high orbits and transmitting continuously does not allow to provide an energy density above the threshold level. To overcome this point, one can emit in bursts to increase the instantaneous value or use concentrators.

Using bursts seems very attractive but requires a specific electronics on board, very similar to the power stages of the emitting part of a radar. This kind of technology is not yet mature at this power level for space applications and could be a possible threat in GEO due to EMI induced perturbations. Moreover it would need an extra mass to be flown leading to an increase in cost for launch and development.

The concentrators, usually of parabolic shape, should track the spacecraft. For kilometers wide rectenna this solution seems barely practical due to the number or size of these concentrators. The concentrators used should thus be mechanically static. This requirement is easy to fulfill for the GEO orbit.

For a different orbit it is generally not feasible. However, for an equatorial orbit, a set of parabolic cylinders with their axis parallel to the ground track of the spacecraft and featuring rectifying elements on their focus plane could be used. Due to the short wavelength, a metallic lattice would act as a good reflector allowing to keep the cost of the system to be lower.

Thermal Control During Mission Phases

The basic assumption is to use the structure of the antenna itself to dissipate the excessive heat. With the solid angle of view of the earth being low compare to that of the black space, it can be considered for a first estimation that the antenna radiates as a black body at 4 K. The total amount of heat radiated is given by the black body equation weighted by the emissivity of the radiators. $Q = RsE(Tr^4 - Ts^4)$, where Q is the amount of energy, R the area of the radiators, s the Planck constant , E the emissivity, Tr the absolute temperature of the radiator, and Ts the absolute temperature of the surrounding space. Considering that the solar arrays are self radiating, the energy lost in the chain from the solar arrays to the rectenna is at the maximum 13 MW x 0.15 x (1-0.0864/.15) = 8.27 kW. Assuming a temperature of the radiator of 373 K, which is a low value when considering tube temperature, and a space temperature of 4 K, the area needed is well below the total surface of the antenna. When the antenna is facing the sun, the solar arrays are no longer powered, drastically reducing the cooling requirement. The other elements of the spacecraft not in contact with this face could radiate directly to space through the triangular open surfaces in the north and south directions.

Propulsion Subsystem

This section discusses the propulsion needs of the MW class demonstrator. From these requirements, an outline of the optimal system is devised, accounting for all stages of the mission. The two main mission phases are the construction phase in LEO and the operational phase in 36000 km orbit (worst case scenario compare to 20309 km). Of these, the latter must be the design driver, that is, the phase of the mission which the propulsion subsystem design is biased towards. The tasks of the propulsion system for the mission phases identified above are:

LEO construction phase: attitude control and drag compensation

36000 km operational phase: orbit transfer , plane changes and attitude control

Attitude control is discussed in a later section. This section will concentrate on the propulsion system designs for drag compensation and orbit raising.

Drag Compensation

Propulsion system design is driven by the 'delta V' (velocity increment) required to be imparted on the spacecraft to perform a given maneuver. The delta V required for drag compensation for a spacecraft of this size is considerable. This is illustrated in Figure 10.4.8. As can be seen, the drag is very sensitive to temperature and hence solar activity. The propulsion system must therefore be designed for the worst case (highest delta V) and will be understressed for most of the time. When chemical thrusters are used for this function, they induce undesirable attitude disturbances. Some degree of 'throttle-ability' is required, and this is more easily achieved through the use of electric propulsion.

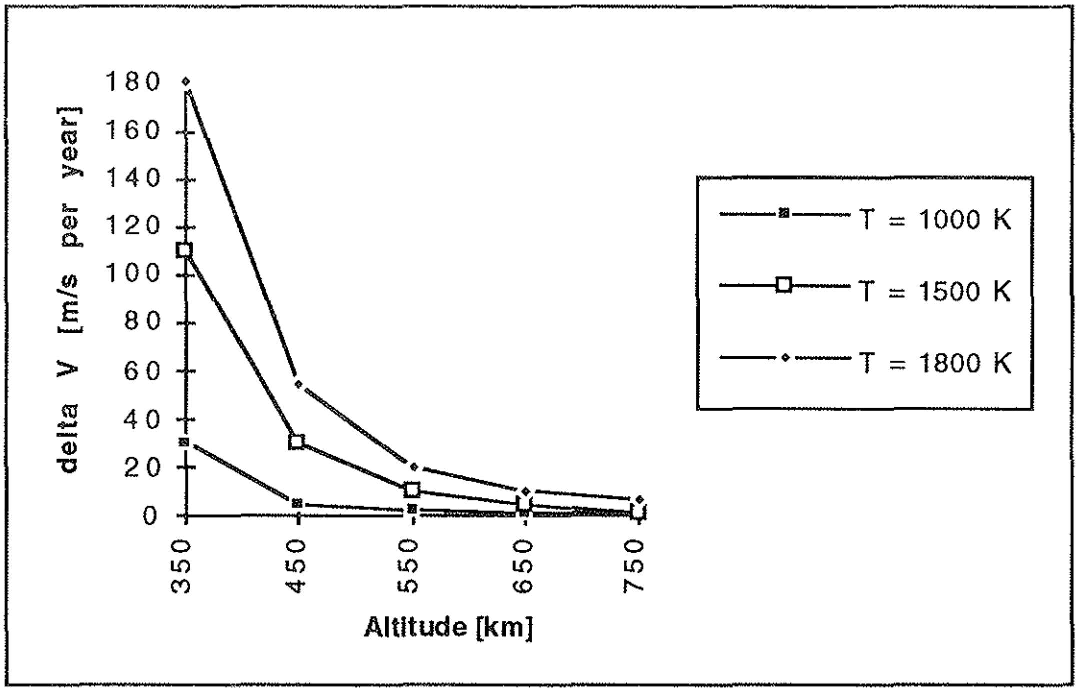


Figure 10.4.8 Drag Compensation Velocity Increment Requirements

Electric propulsion systems are, however, very low thrust devices. At this altitude (350 km) and with this size of structure, the drag force is very high. For 'worst case', the drag force is calculated below.

$$Drag, D = \frac{1}{2}\rho v_a^2 S C_D$$

where:

ρ = air density

v_a = velocity of spacecraft relative to atmosphere

S = reference area (area perpendicular to spacecraft direction)

C_D= drag coefficient

For LEO spacecraft, particles are incident at $v_a \approx 7 km/s$ (see later), accommodated for a short period of time on the spacecraft surface and e-emitted at the 'thermal velocity', $v_{th} < 2km/s$ (corresponding to the spacecraft surface temperature).If we assume very low specular reflection (low thermal velocity) then it can be shown that:

$$C_D \approx 2.2$$

Velocity: $v_a = \sqrt{\frac{\mu}{r}} = 7.7$ km/s

Area: $S = 100 \times 100 = 10000 m^2$ - worst case

Density: let $\rho \approx 10^{-12} kg/m^3$

Hence, $D = 0.65N$

Thus, it can be concluded that either chemical or electric propulsion systems can be chosen to fulfill the task of drag compensation, since both can easily supply a countering force of this magnitude.

Now, from Figure 10.4.8, the worst case scenario provides a delta V of about 180 m/s per year. Using this value, a comparison of chemical and electric propulsion for this task can be performed. This analysis uses the 'Rocket equation', as shown below:

$$\Delta V = I_{SP} . g_o \ln\left(\frac{M_o}{M_f}\right)$$

where:

I_{SP} = specific impulse

g_o = acceleration due to gravity

M_o = initial spacecraft mass

M_f = final spacecraft mass

In this case, we assume an initial mass of 70000 kg. The resulting mass data is tabulated below (note: Mp = mass of propellant used):

Table 10.4.2 Propellant Masses

propulsion	specific impulse	Mo/Mf	Mf	Mp
chemical	320 s	1.059	66099 kg	3900 kg
electric	5000 s	1.004	69744 kg	256 kg

From the table, it can be seen that the electric propulsion system offers a significant propellant mass saving. It also has the property that it is compatible with the propulsion system for orbit raising although the thrust requirements for each operation are different. The main disadvantage of using electric propulsion at this phase of the mission is the power requirement.

The specific power for an electric propulsion system is typically 35 kW/N for the magnitude of specific impulse given. Hence, to supply 0.65 N to counteract the drag, the electrical power required is 22.75 kW. This may be a problem in the initial stages of construction of the spacecraft, when the solar arrays are not operational (although the drag will be lower when the solar arrays are not in place). However, it has been decided that this disadvantage is outweighed by the advantages of fuel mass saving, system compatibility and avoidance of attitude disturbances (as mentioned at the beginning of this section).

Orbit Raising

Here, the maneuver considered is that of raising the spacecraft's orbit from LEO (350 km) to GEO coupled with a 28.5^o plane change. There are two basic options:

- a Hohmann transfer using high thrust, chemical propulsion, coupled with a plane change at apogee (minimum energy case)
- a spiral transfer using electrical propulsion, with the plane change integrated optimally in the maneuver.

These two options are illustrated schematically in Figure 10.4.9 and are analyzed below.

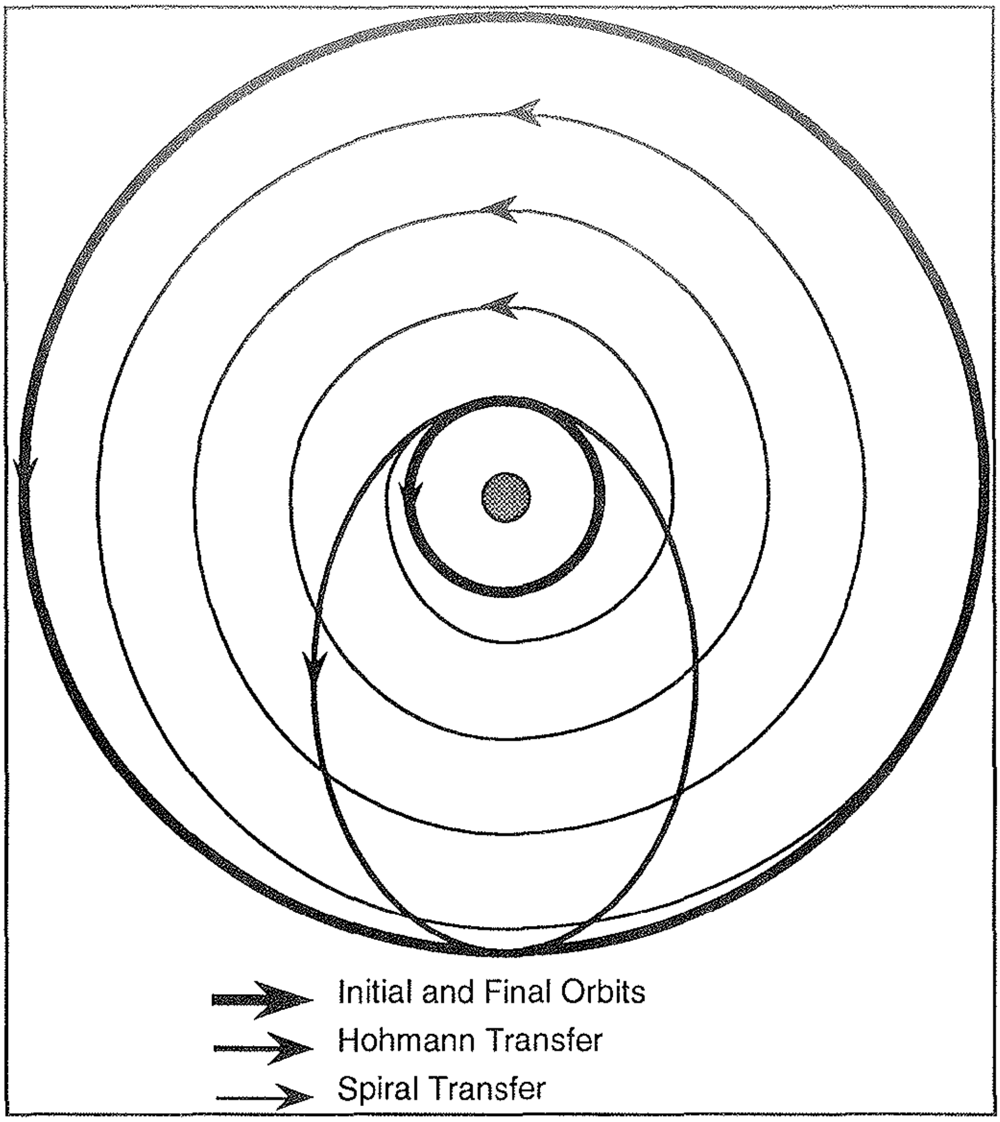


Figure 10.4.9 Basic Orbit Transfer Options

The Hohmann transfer coupled with a plane change at apogee requires a total ΔV of 4330 m/s. Using a cryogenic chemical propulsion system with a specific impulse of 450 m/s leads to the following ratio of fuel mass to spacecraft dry mass: 1.65.

Hence, for a spacecraft dry mass of, say, 50 t, the mass of propellant is 82.5 t giving a total mass of 132.5t. This is very large and would necessitate, for instance, two Energia launches. Similar analysis can be done for the purely electric propelled spacecraft in a spiraling trajectory, with optimal plane changing as an integral part of the maneuver. This calculation is summarized below:

propulsion system:	specific impulse	5000 s
	thrust	29 N
	power required	1 MW
ratio of fuel mass to dry mass:		0.09
total transfer time:		200 days

So, electric propulsion has considerable advantages in terms of propellant mass saving, but significant disadvantages in terms of the power requirement and the transfer time. The latter suggests that long durations are spent traversing the Van Allen radiation belts with the serious risk of solar cell degradation (specially for a low cost a Si option) and damage to on board electronics.

To account for the advantages and disadvantages of both propulsion systems, a possible solution may be to design a 'hybrid propulsion system for the maneuver. This would use chemical propulsion for the first stage of the maneuver in order to

- shorten the total transfer time
- shorten the duration spent in the radiation belts.

An electric propulsion system would then take over so that the system is far more fuel efficient than the purely chemical case. This maneuver is shown schematically in Figure 10.4.10.

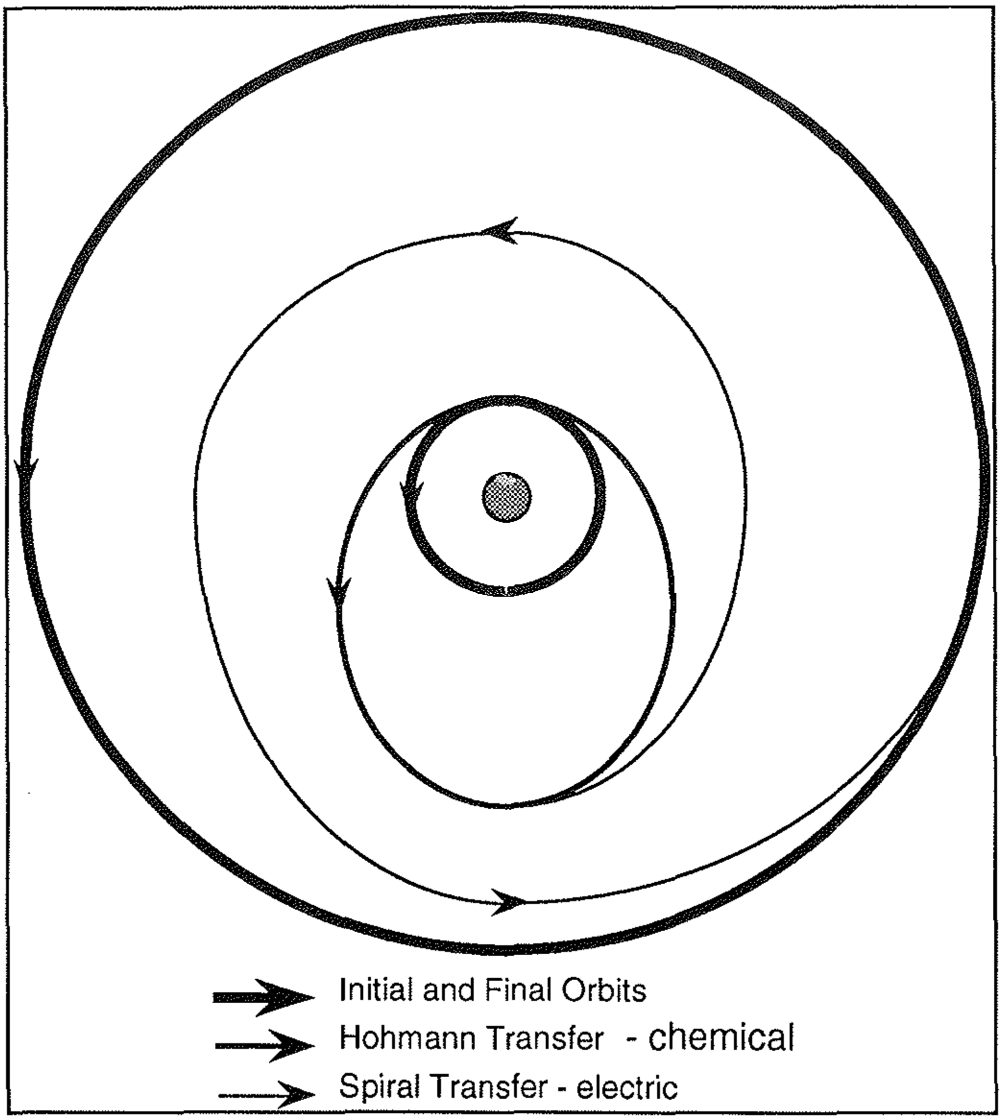


Figure 10.4.10 Hybrid Orbit Transfer

Considering a 8000 km point as the altitude of propulsion system change, and using a Hohmann transfer for the first phase of the maneuver, the ΔV required are:

From LEO to 8000 km; ΔV=1285 m/s, with the same hypotheses than above the mass ratio is then 0.34, considerably less than previously.

From 8000 km to GEO, including the same plane change the optimal ΔV is 3815 m/s. It can be roughly considered that an electrical transfer will use 1.4 times this ΔV. Considering a specific impulse of 5000 s ,the mass ratio for fuel is then 0.09.

With this method the total amount of fuel is: 21.5 t. Even with an overhead of 10 t for the electrical thruster the gain is still 51 t. The launch could then be made with only one Energia, greatly decreasing the costs, even when taking in consideration the electric propulsion system cost.

Attitude Control

The problems of station keeping are similar to the ones experienced by telecommunications satellites, though on a different scale (assuming that the structure is sufficiently rigid). Due to the high level of accuracy required, a three axis electrical propulsion stabilization system is recommended. The use of electric propulsion not only minimizes the overall mass but reduces the number of different subsystems. Fly wheels could also be considered mainly because of their clean properties but the mass of these devices for a huge spacecraft and the necessity to desaturate them periodically are significant drawbacks. Furthermore, assembly considerations favor a set of modular independent propulsion packages requiring only electrical connections. For guidance considerations, the high accuracy required favors the use of a star tracker for the GEO case. For the lower orbits, pointing requirements are relaxed as the beam steering can compensate for the pointing variations of the platform.

Mass Assumption

The mass of the basic framed structure is comprised of the truss elements and a wire frame which further stiffens the structure. The mass of the trusses is obtained from Space Station Freedom dual-keel 5 m design. The basic mass is 18 kg/m leading to 1800 kg for the basic 100 m segment. For the planar array, the result for the exterior frame is 7200 kg. For the prismatic geometry, the total length is 900 m resulting in 16200 kg. To rigidify this frame and tallow attachment for the solar cells array, a 5 m x 5 m squared lattice of kevlar wires is used. The linear mass of the kevlar wire is 5 g/m resulting in a negligible mass contribution to the total structural mass.

The antenna subsystem dominates the mass of the array. Both conventional tube derived and solid state transmission arrays are considered. The conventional array is comprised of heavy elements such as tubes, phase shifters and wave guides. The design assumes an array of 100 radiative elements. The mass of each set (amplifier and wave guide) is of the order of 10 kg, and the coupled radiative elements sum to 1000 kg. The slot antenna is constructed of 100 square panels (10 m x 10 m) made with 0.5 mm thick aluminum. The average weight for a panel is then 270 kg, with a total element mass of 27 t.

The phased array antenna assumes 1000 radiative elements, the power per element is lower but it should be pointed out that, basically, the gyrotrons are identical to the previous case, and that the phase shifters are much more demanding than for the slot antenna. Phase shifter being usually made by using a set of different wave guides switched on and off their weight is very rapidly increasing with the number of phase step they provide. Then, an optimistic evaluation (within the frame of the present technology) leads to account each radiative element at 20 kg. The total amount is then 20 t. The constraints of rigidity and of thermal conduction on the antenna are roughly comparable to ones for the slot antenna. The total mass of the phased array, using conservative technology is 47 t (based on 4.7 kg/m^2).

Various types of solar arrays were considered. GaAs were assumed with both a near term and mid-term (10+ year) values. In the near term, Si cells with an efficiency of 14% had mass figure of 2.15 kg/m^2. This figure would include some rigid structure for deployment. For the mid-term, the use of 20% GaAs cells at 0.3 kg/m2 was regarded as attractive. A basic 100 m x 100 m area results in a mass of approximately 20 t for the near term and 3 t for the mid-term (flexible array). For the prism structure, the mass is double this at 40 t for the rigid (near-term) and 6 t for the flexible (mid-term) array.

The mass of the electrical subsystems, including attitude control electronics, is given as 2000 kg,. This is regarded as a very conservative as this mass is loosely coupled to the size of the spacecraft. The electrical subsystem architecture would be similar to the one used for a telecommunication satellite, (the power subsystem excepted.) It would then be meaningful to use systems already developed. The power conversion subsystem mass is largely taken into account in the mass of the antenna, as it includes the full set of tubes and phase shifters (which represent the most significant. contribution).

The fuel consumption is given as follows:

Orbit transfer: from LEO 350 km to 36000 km, assumed to be hybrid: chemical and electrical propulsion using a LOX, LH2 engine with a 450 s ISP, and an 4500 s ISP for the electrical thruster.
Station keeping ; Electric propulsion system only. 10 years of operation.

The cryogenic fuels needed are:	LOX/LH2	44 t worst case, 23 t best case.
	E.Fuel	9 t worst case, 7.2 t best case

It should be pointed out that this hybrid transfer does not include a ΔV for circularization at the apogee of the Hohmann transfer. The spacecraft will thus cross the Van Allen Belts at perigee. It is assumed that the speed at this location will be high enough to avoid any serious degradation. Furthermore the on board electronics could be, to a large extend, be powered off to avoid potential latch-up phenomena.

Structural weight for tanks and main engine is rounded to 4% (usual value is 10%, but we assume a better value due to use of Al Li alloy) of the total mass of fuel. The result of these mass estimates are provided in the summary below. Under those assumptions the total mass for 20309 km is: 172.3 t (near term) and 77.6 t (far term).

Table 10.4.3 Mass Summary For Structure

Mass Summary	Near Term (t)	Far Term (t)
Truss/structure	16.2	16.2
Array	40	6
Antenna	47	10
Electrical/power	2	2
Tanks	2.1	1.2
Cryogenic engine	2	2
Electric Thrusters	10	10
Dry mass	119.3	47.4
E-Fuel (transfer + St. Kp. 10 y.) 9	7. 2 *	
LOX/LH2(transfer)	44	23*
Total		
LEO	172.3	77.6
Operational	128.3	54.6

* assuming hybrid transfer

Those masses are worst case values, as for GEO the lighter antenna save around 10 t. Nevertheless, due to the uncertainties for this estimate, a gain of 10 t does not change dramatically any conclusion.

Recommendations and Critical Issues

Engineering Aspects

After this brief description the main critical issues from a technological standpoint are;

- Development of efficient amplifiers and oscillators at 35 GHz, mainly in the family of gyrotrons.
- Development of phase shifters for phase adjustment of the sources of the antenna.
- Development of electrical thrusters, both for attitude control and transfer.

It is then recommended to include those developments in a more general plan to share the costs with other projects. The field of telecommunications could be used to get a first return on investment from those technologies.

A point of great interest, even if not directly linked with the spacecraft itself, is the energy density at rectenna level. This density has a direct impact on the efficiency of the system and depends highly on the performances of diodes at GHz frequencies. Within the present technology frame and for high orbits, it is necessary to consider either extremely large spacecraft (kilometers wide) to allow a large amount of energy to be beamed to relatively small areas on the ground, or to use concentrators. Neither of those solutions are perfectly satisfactory. It is therefore recommended to push the development of better diodes or rectifying devices and to investigate the possibility of 'burst' transmission. This later point could be a joint development with radar users. Finally it should be pointed out that this study does not cover the reliability and safety aspects, which should be carefully detailed for an operational usage.

Energy Aspects

It is of some interest to compare the amount of energy collected in this design with what could be collected at ground level with solar cells. The average solar power at ground level over the US is pessimistically one quarter (Whitehouse, 1989) of the value in space. Considering that the rectenna complex is roughly 10 km x 10 km wide, and even considering a very inefficient conversion ratio of the ground system, that is, 5%, the available energy would be 10000 m x 10000 m x 0.25 kW/m^2 x 0.05 = **1.25 GW**. This figure would improve still further using a thermodynamic system. It seems then, that even not mentioning the tremendous difference in cost between space and earth operations, covering the area of the rectenna with solar cells could warrant further investigation!

10.4.3 Concept Summary

The following concepts are discussed in the previous text, they both fill potentially the requirements expressed in item 0 and allow consideration of alternatives. The baseline is concept number 2.

Table 10.4.4 Concept 0 SPS-2000

Power	17.9 MW (10 MW from platform)
Construction/Launch:	10 ArianeV; Robotic Assembly
Array Topology:	Prism
Array dimensions/type:	336 x 336 x 333 m; a-Si
Truss:	Part deployable; robotic assembly
Assembly time:	unknown (1-2 years)
Orbit:	Equatorial LEO
Stability:	Gravity gradient
Transmitter:	Solid state; phased array; 2.45 GHz; (10 MW beam)
Mass Statement	
Truss:	2.71
Panel:	39.16
Transmitter:	60
Cable:	18
TOTAL:	120 Mt

Cost

The projected cost of the project is 30 billion ¥ (200 million US dollars), which does not include the launch costs for 10 Ariane V vehicles (approximately 1.1 billion US dollars). With that, the total projected cost approaches 1.5 billion US dollars. If current prices are used for the transmitting antenna (which dominate the cost of the array in current prices, as illustrated in the next section), then the current costs would be 10-100 times larger. In the mid-term, these costs of the critical components must decrease by those factors in order to achieve even 1 billion US dollars platform cost. Finally, the considerable cost of the rectenna is not included in the above project costs.

Notes

The first major areas of concern include the difficulty of constructing a platform of this size given the current state of the art as well as the extremely difficult environment. The DDTE costs of a telerobotic servicer is of the order of several 100 million $US. Also, the proximity operations, rendezvous/docking, transfer of the cargo elements into the structure is thought to be demanding. Other points of difficulty include the low mass of the truss structure, the high mass of the cable (though some power conditioning equipment is thought to be included), the 50% mass figure for the transmitting antenna (the heat transfer limit of 798 W/m^2 is a factor of five larger), the stability of the platform during construction, and, the cost of maintaining ground control people during the year-plus construction period. In order to reduce costs, conventional a-Si arrays are used. However, these are known to degrade very rapidly in the space environment (by a dramatic drop in efficiency) and may not perform as well as other array choices. Finally, the overall rigidity of the structure is a problem if there are perturbations of the transmitting antenna which cause inaccurate pointing of the microwave beam.

General

The SPS-2000 study is particularly useful, in that it offers a strawman design of a 1-10 MW class facility with all of the attendant problems. As a quick comparison, though, if one takes the approximate figure of 1 billion US dollars for the cost of a 10MW SPS platform, the result is 100 million US dollars/MW. A fast breeder reactor of the class contemplated by Japan provides 700 MW for 1 trillion ¥ (7 billion US dollars) which results in 10 million US dollars/MW. The price for the SPS-2000 (again, not including DDTE costs) is approximately 10 times the cost of the fast breeder reactor per MW. In addition, the platform would not continuously beam 10 MW to a particular location, but only some fraction of the orbit. This also would increase the cost per continuous MW *delivered*. On the other hand, once the basic DDTE is complete, the unit costs per SPS platform would diminish. In addition, the same rectenna could be used for several SPS platforms operating in a constellation, further reducing the cost per MW delivered.

Table 10.4.5 Concept 1 1MW Mid term Demonstrator

Description	
Power:	1 MW class
Construction/launch:	1 STS
Array Topology:	Planar
Array Area/type:	65 x 65 m ; Si cell.
Truss:	Modified SSF 5 m truss; tension wire assisted (Kevlar)
Assembly time:	10.3 h (truss)
Orbit:	28.5 LEO; potential higher orbit later
Stability:	Active
Total mass:	65 t (22 10+ years)

Table 10.4.6 Mass and Cost Summaries for Concept 1

Mass Summary	Near Term (t)	Far Term (t)
Truss/structure	5.6	5.6
Array	8.3	3.8
GNC	2	2
Antenna	48	9.6
Power	1	1
Total	64.8	22
Cost Summary		
	Far Term ($M)	Near Term
Truss/structure	50	50
Array	342	50
GNC	100	100
Antenna	12500	12.5
Power	75	1
Total	64.8	22

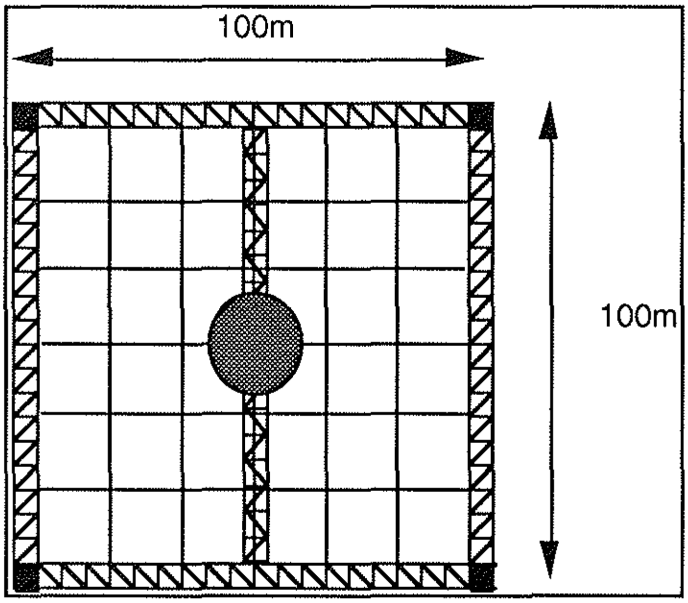


Figure 10.4.11 1 MW Planar Array Demonstrator.

The planar shape, constructed with Space Station Freedom derived trusses, is provided as a strawman mid-term demonstrator in the 1 MW class. It is constructed in LEO with the STS and, if necessary, an additional Energia launcher. The assembly time for the truss is approximately 10.3 h (based on 1 element/minute as previously discussed), and the additional assembly is estimated at 20 h for a total of 30 h. The mass of the platform is dominated by the mass of the phased array, based on current mass and power estimates. This, however, drops dramatically as the transmitter technology advances (currently, the arrays are heat transfer limited, so that active cooling may be required in the future to make the array less massive.

In the cost summary, as in the mass, the figures are completely dominated by the cost of the transmitter array. The shown cost of 12.5 billion US dollars is based on the price used to estimate the cost of the array used in the near term demonstration discussed in another section of the report. The other dominant figure is the cost of the solar cells, which also should show dramatic improvement in the next decade. If a 100-fold improvement in phased array technology, and a 10-fold improvement is the array cost (and thus, the efficiency), then a much more modestly priced platform results. If the cost of the solid state phased array technology drops by only a factor of 20, then the cost of the array will be in the billion dollar class.

From a mission perspective, the primary difficulty of the concept is related to the 28.5° low earth orbit. As previously discussed, this limits the time during which the a useful beam is received.

Table 10.4.7 Concept 2 1 MW Commercial Pre-cursor

Description	
Power:	1 MW class
Construction/launch:	1 STS/1 Energia
Array Topology:	Prism
Array Area/type:	100 x 100 x 100; Ga-As.
Truss:	Modified SSF 5 m truss; tension wire assisted (Kevlar)
Assembly time:	30 h (truss only)
Orbit:	28.5 LEO construction; raised to 20000 km equatorial
Attitude Control:	Active 3 axis control
Total mass:	172.3 t (78 t in 10+ years))

Table 10.4.8 Mass and Cost Summaries for Concept 2

Mass Summary	Near Term (t)	Far Term (t)
Truss/structure	16.2	16.2
Array	40	6
Antenna	47	10
Electrical/power	2	2
Tanks	2.1	1.2
Cryogenic engine	2	2
Electric Thrusters	10	10
Dry mass	119.3	47.4
E-Fuel (transfer + St. Kp. 10 y.)	9	7.2*
LOX/LH2(transfer)	44	23 *
Total LEO	172.3	77.6
Operational	128.3	54.6

Cost Summary	Near Term($M)	Far Term ($M)
Truss/structure	50	50
Array	553	53.3
Antenna	12500	12.5
Power	75	75
Propulsion	500	500
Total	13678	690.8

* assuming hybrid transfer

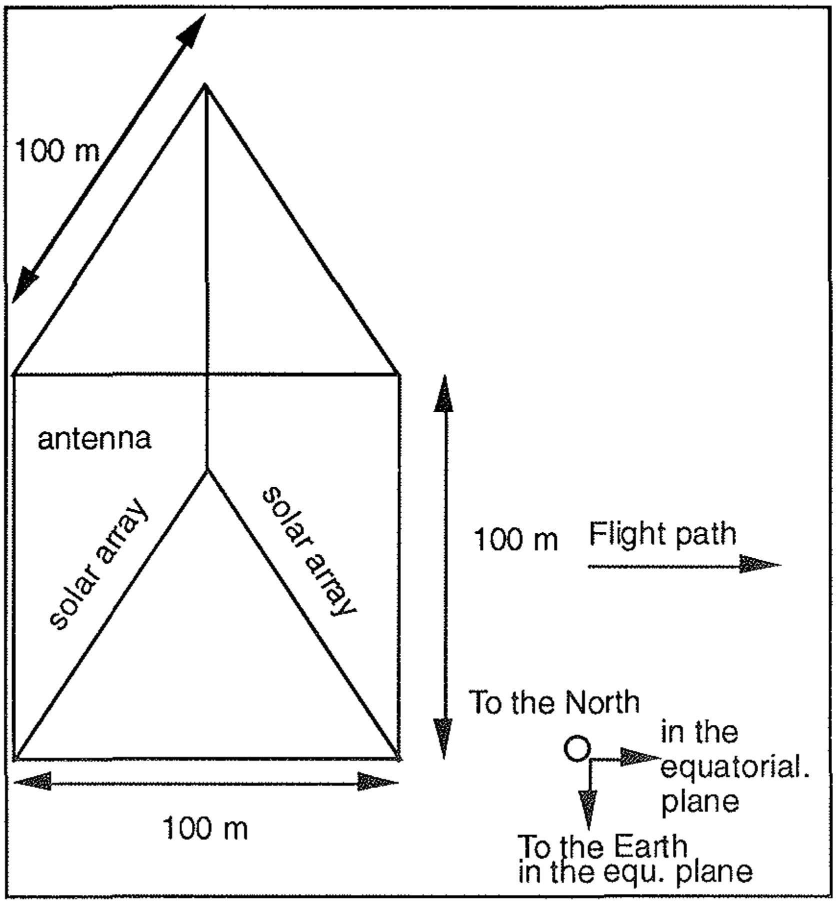


Figure 10.4.12 1 MW Pre-commercial Demonstrator

As previously discussed, this concept is constructed in 28.5° LEO with one STS and one Energia (far term assumption) flights. The platform is a symmetric prism and is shown in Figure 10.4.12. Once the vehicle is constructed, it is raised to the final orbit with electrical and chemical propulsion. The selection of the 20000 km orbit provides approximately 6 hours of continuous beaming to the collecting antenna site per day. The low power density of the beam, however, requires a concentrator on the receiving site. As before, the mass of the platform is completely dominated by the transmitting antenna in the near term, with strong potential improvements in the mid to far-term. In the far term, with the expected advances in electric propulsion, the orbiting platform may be of the one billion US dollar class.

Table 10.4.9 Concept 3 Futuristic Large Planar Array

Power:	20 MW class
Construction/launch:	1 STS, 1Energia
Array topology:	Planar
Array area/type:	200 m x 200 m Integrated solar array/ transmitter.
Truss:	Modified SSF 5 m truss; tension wire assisted
Assembly time:	95 (truss)
Orbit:	28.5 LEO assembly; potential higher orbit later
Stability:	Active
Total mass:	70 t (Based on future mass reductions)

Table 10.4.10 Mass and Cost Summaries for Concept 3

Mass Summary	Far Term (t)
Truss/structure	15
Integrated Array	40
GNC	5
Power (Electric Prop).	3
Total	78
Cost Summary	Far Term ($M)
Truss/structure	50
Integrated Array	300
GNC	100
Power (Electric Prop.)	10
Total	460

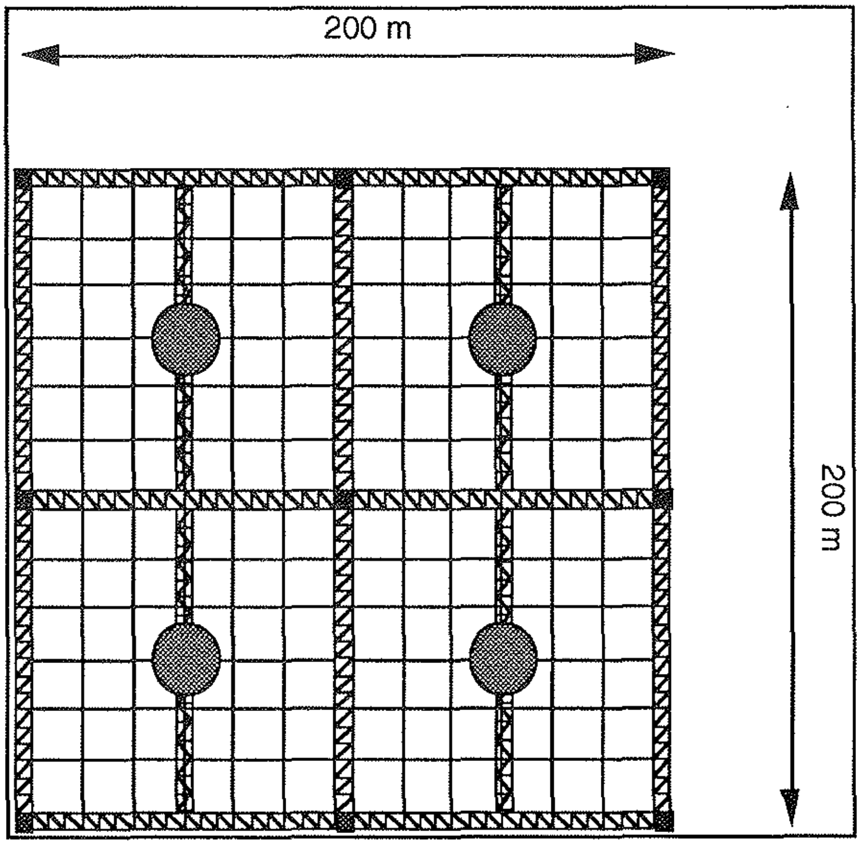


Figure 10.4.13 Large Planar Array

The concept is briefly presented to indicate both the direction of future technologies as ell as the potential for modular designs. This concept is a scaling of the concept 1 planar array by a factor of 4. As in concept 2, a suitable equatorial orbit would be desirable. It is assumed possible only by the projected drop in mass and cost of the of the antenna (factor of 10-100) as well as the solar array technology (factor of 10). For a more futuristic option, the planar array is regarded as attractive particularly for the integrated solar-array /transmitter technology (see section 7.2.1.5 of the report) which eliminates the large transmitter as a separate subsystem. There are also potential improvements in the efficiency by closely coupling the elements, as well as having a transmitting antenna of large dimension (that is, the area of the solar array would equal the area of the transmitting antenna). In addition, the concept works at 2.45 GHz and there is no dependence on the more difficult 35 GHz technology. This concept requires an open structure (since the solar array is on one side and the transmitting antenna on the other) and the size would be limited by the structural rigidity of the large array. In the above example, the central transmitting antennas would disappear. For such a platform, the power to the transmitting antenna would be of the order of 10 MW. With launch costs, the cost for such an integrated array platform may approach 1 billion US dollars for a 10 MW platform. As unit costs decrease, these integrated array platforms could assembled modularly or flown in constellations.

Launch Cost Notes

The STS launch was either a NASA sponsored project or a NASA Joint Venture Agreement which would not cost the project directly. One Energia launch was assumed to be 100 million US dollars.

10.4.4 Scheduling

This section illustrates a feasible schedule in which the MW class demonstration is developed, constructed and made operational. The total time between conceptual studies and fully operational status is envisaged to be about 7 years. This includes approximately six months for the orbit raising and plane changing maneuver using the hybrid propulsion system discussed earlier.

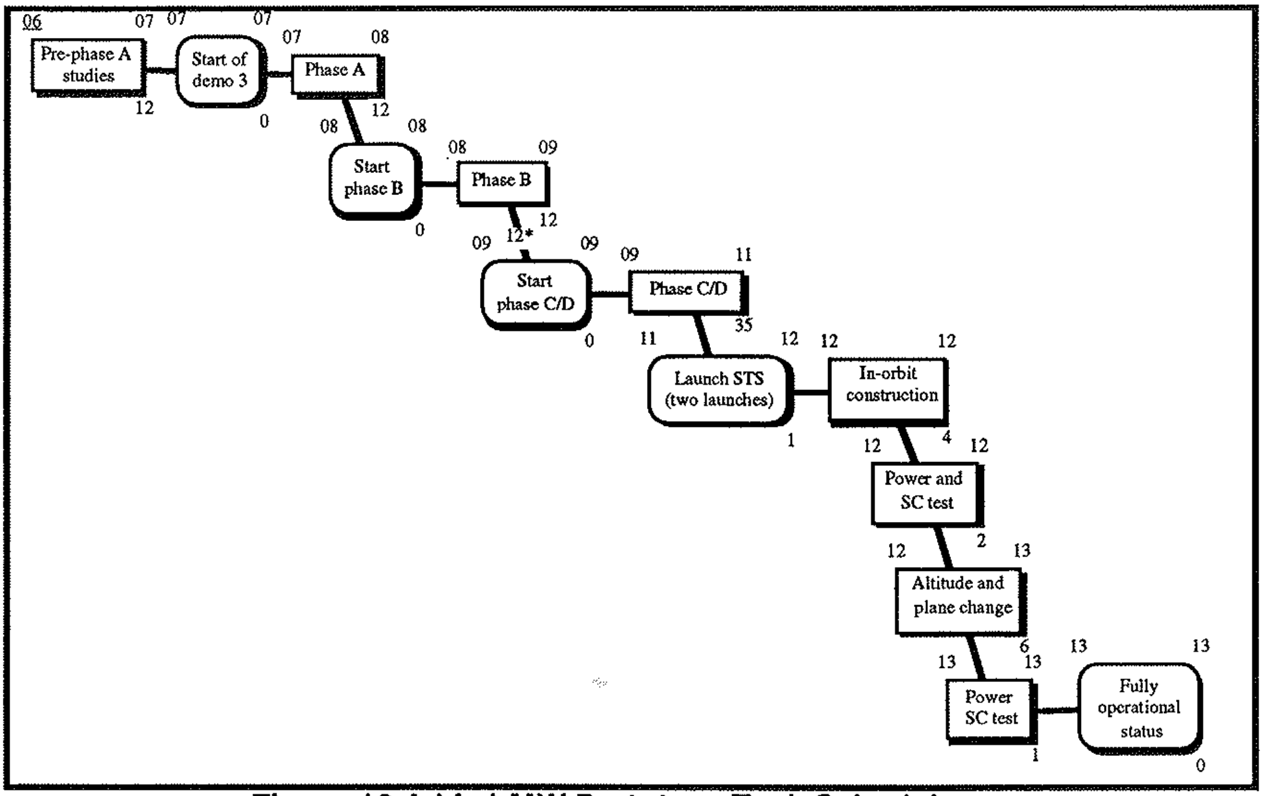


Figure 10.4.14 1 MW Prototype Task Schedule

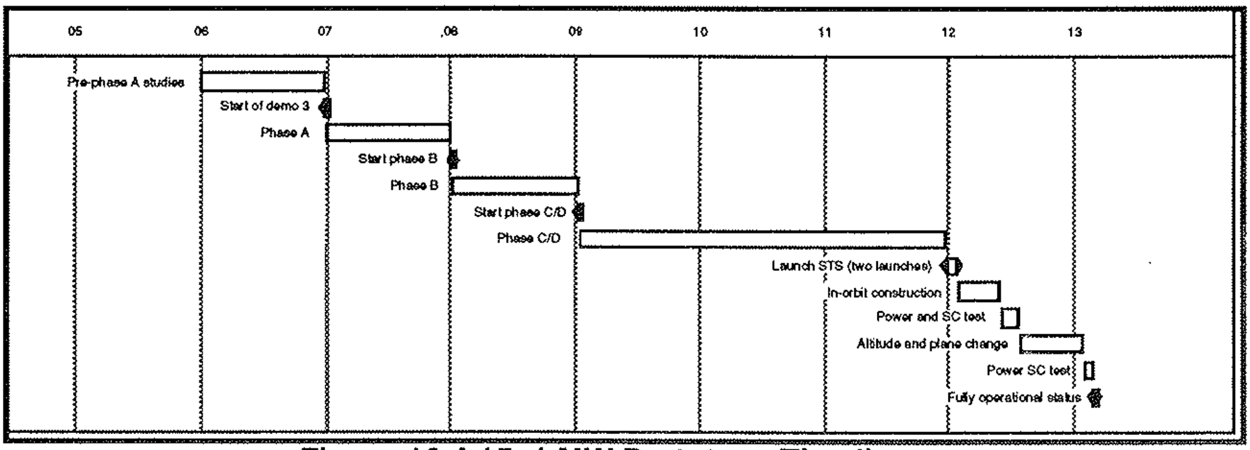


Figure 10.4.15 1 MW Prototype Timeline

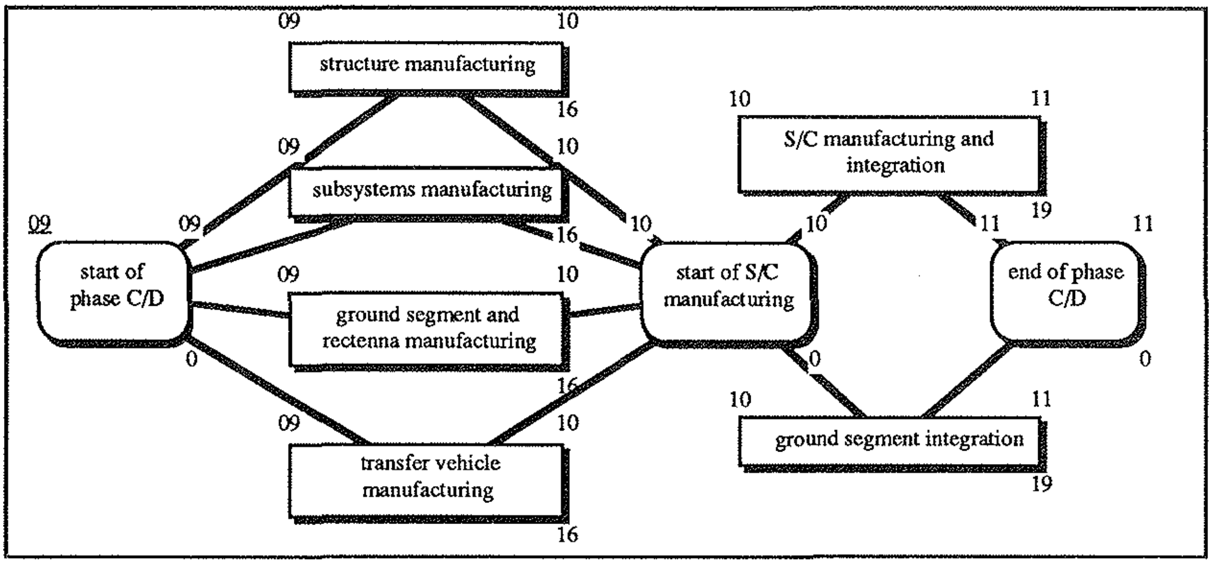


Figure 10.4.16 Phase C/D Task Schedule

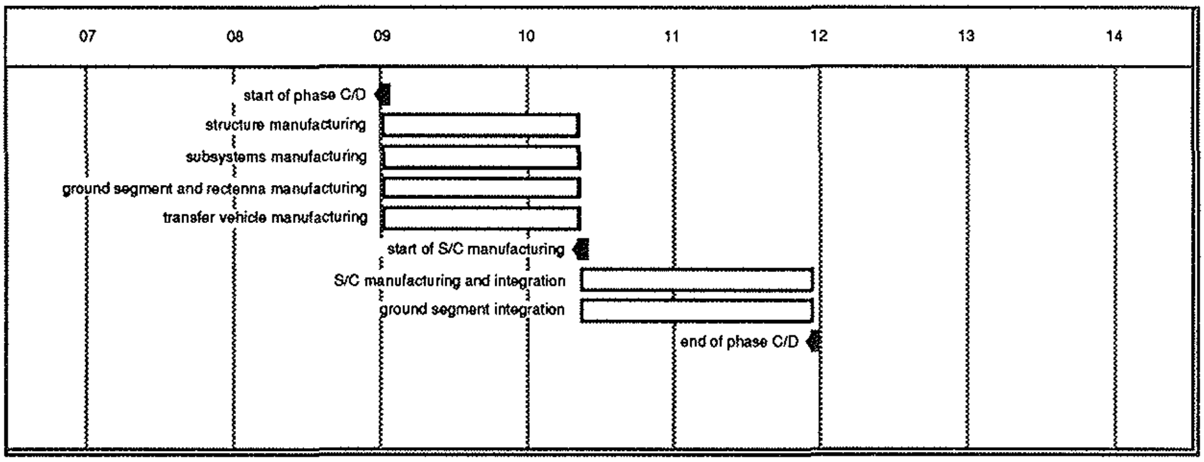


Figure 10.4.17 Phase C/D Timeline

10.4.5 Summary and Conclusions

The results from the design examples developed in this section are as follows.

1. Manned construction is (perhaps unfortunately) the most conservative and direct means of assembling 100 m scale space structures in the near to mid-term.
2. Combination of STS and Energia results in potentially large -scale structures that can be constructed within short time scales (that is, this decade).
3. A simple 1 MW demonstration platform ('trampoline') can be constructed at 28.5 degree orbit and can be later incorporated into a more practical equatorial orbit platform.
4. Platforms constructed in low earth orbits which are non-equatorial have poor ground traces and may serve primarily as technology demonstrators. For these low orbits, it is not even economical to construct a full rectenna for demonstration purposes. If demonstration rectennas or concentrating antennas are built, it should be done at an equatorial location that could later be expanded for the more practical equatorial orbit platforms. Sufficiently high orbits which have some inclination (with antenna angle compensation) may suffice but were not examined in detail.
5. Equatorial orbits are perhaps the most desirable though low equatorial orbits limit coverage. GEO is attractive though problems may result from competition for orbit slots as well as potential interference with other satellites. With the construction and assembly described, an additional cost is required in terms of a propulsion system to place the platform in the desired orbits.

6. An attractive equatorial orbit may be a t 20000 km which provides good ground coverage (6 hr/day) and not have disadvantages of either LEO or GEO. Further, the rectenna size is smaller than that required for a GEO platform.
7. Platforms of the 10 MW class or smaller have a general problem of energy flux at the rectenna location. For the previous orbit (20000 km), it was found necessary to construct a concentrating antenna aligned parallel with the velocity vector of the orbiting equatorial platform.
8. Platform topology is dependent on at least orbit and application. For the small demonstrators, a rigid planar array is adequate. For the higher orbit, a prismatic structure has advantages. With advanced integrated technologies in which both solar array and the phased array transmitting antenna are in plane, large rigid planar arrays may be desirable. This will also reduce the size of the receiving rectenna and increase the energy flux.
9. For orbit raising and plane change, the available high power levels make electric propulsion (for example, xenon ion with an Isp of 5000 s) attractive. The dV in order to make the maneuvers is of the same order as going to low lunar orbit, though in both cases requires relatively little propellant (order of 10 t). This may be combined with an initial chemical propulsion stage to avoid excessive radiation degradation of the solar cells during the transit through the Van Allen belts.
10. In the near term, both the mass and particularly the costs of the phased array transmitting antenna utterly ***dominate*** the platform design. In order to construct the first platform element for 1 billion US dollars, it is necessary for the antenna costs to drop between 10-100 times. Using technologies which are developing rapidly this decade, this may be possible. To a lesser extent, the solar arrays also contribute significantly to the cost, but a decrease of 10 times with the expected improvements in efficiency will also contribute to the feasibility of the platforms. Finally, advances in electric propulsion technology should reduce the unit costs of each element by a significant amount.
11. Eventual technology growth expected in this decade is expected to significantly reduce the size and cost of critical subsystems, making the overall concept more commercial viable (though launch costs will remain as the dominant economic factor for true commercial applications). The costs for the platforms may drop to a 1 MW, 1-billion class facility, which would be capable of beaming 6 MW-h/day of electricity to an equatorial ground station.

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11 Finance

Finance and cost constraints play a key role in the realization of any project. The high costs and low revenues of existing space programs mean that few commercially viable space enterprises exist today. These facts mean that most of the worlds space projects are funded from governmental funds on political, science, or military grounds.

Power generation produces huge revenues and could be one area where space projects could justify their high costs. This chapter aims to analyze the financial scenarios whereby a space solar power program could be a viable business proposition, and outlines how this scenario could be achieved. The main questions answered by this chapter are:

1. What are the costs of such programs? What are the cost sensitivities ?
2. Are there commercially viable markets for space solar power? What is their estimated value?
3. Where can we get the funds necessary to realize such programs?
4. What are the major factors which could affect investors confidence in the Internatioanal Solar Power Organization?
5. How can ISPO stage programs to maximize investor confidence?
6. Which sources of finance should we utilize for different ISPO programs?
7. Under what conditions will space solar power be economically viable?
8. When will such programs break-even?

11.1 Costing and Economic Analysis

This section undertakes a general costing and sensitivity analysis of first: early commercial uses of beamed power for space based users, and second: full scale space to Earth commercial power beaming. Since space based early commercial uses stand alone as a beamed power application independent of space to ground power programs, the analysis extends to consider the commercial viability of such projects, and the factors important to such viability.

11.1.1 Space Based Early Commercial Uses - Costing and Viability

This section analyses in broad terms the cost and viability of some of the early commercial uses of beamed power identified in Chapter 3. Unfortunately, the scope of this summer's study does not allow a detailed analysis of each market identified, its value and possible revenue flows, since the market research data is broad based and largely conceptual. Therefore, this section assumes a baseline value for power delivered to a customer in space of \$1000 per kilowatt-hour (kWh) in the near term, at present day US dollars. This is based on an analysis by Karl Faymon of NASA Lewis Research Center, giving the cost of power in 1989 dollars at just under \$800 per kWh, and other higher figures quoted in the literature (up to \$4000 per kWh has been quoted). Where further analysis of a given space power market is warranted or possible, this baseline figure is adjusted: it is certain that some power markets are considerably more valuable than others. In particular, the value for the mid to long term is taken to be \$200 per kWh, since technological advance will undoubtedly lessen the cost of space power subsystems.

The Net Present Value

Given the uncertainty inherent in the market data and other cost parameters to be used for this analysis, the absolute value of the output or Net Present Values (NPV) of any costing model will be unreliable. Moreover, the models we will use will necessarily be simplistic due to the potential scope of this summer's study. However, we may usefully use these models and approximate figures to calculate the NPV for the commercial project, and then to analyze the results in terms of sensitivity of the output NPV versus the various input parameters, in order to draw general conclusions about commercial viability.

Net Present Value method calculates the total monetary value of a project over its duration in present day dollars [Oxnevad 91], by incorporating the effects of inflation and the value of money (interest or discount rate). Baseline values for these variables are: discount rate 20% (typical for high risk investment) and inflation 5%. The Net Present value is the sum of the Present Values (PV's) for the pertinent variables. Those that we consider are:

- PV_{inv}: PV of Investment (cost of money)
- PV_{oi}: PV of Operating Revenues less Operating Costs (net operating income)

We may express these as follows:

$$PV_{inv} = \sum_{y=0}^{PE} \frac{Inv_y}{(1+r)^y}$$

where y is the number of years after program start, PE is the program end in years, r is the discount rate, and Inv_y is the investment required in that year.

$$PV_{oi} = \sum_{y=1}^{PE} (R_y - OC_y)\left(\frac{1+i}{1+r}\right)^y$$

where R_y is the revenue in year y, OC_y is the operating costs in year y, and i is the inflation rate. The Net Present Value is then PV_{oi} minus PV_{inv}.

$$NPV = PV_{oi} - PV_{inv}$$

It should be noted that although inflation is included explicitly, it is meaningless to consider it in isolation as it directly affects the real discount rate: higher inflation is equivalent to a lower discount rate. Many NPV models include the effects of inflation implicitly through its effect on the net discount rate.

Ground to Space Power Beaming

Ground to space laser transmission is the only application technically viable in the near term (see 7.2.2), and this time-scale is considered initially. We therefore consider here the cost of ground based lasers, of which the free electron laser provides the only near term solution to the power and frequency requirements (optical frequencies for solar array conversion).

Project Selene studies, under way at NASA Marshall Space Flight Center, have costed a 10MW ground based free electron laser at $500M for the first ground station, and $225M for subsequent stations. Many consider these figures optimistic, however, and we will assume technology development costs to be of order $1B, with subsequent construction costs of $250M per station. While most near term uses will require much lower power levels than those considered for Selene, we will use these cost figures for a generic ground based laser system. Maintenance costs per year, again based on Project Selene figures, will be taken as $40M.

Since a ground based laser is limited in practice to beaming within 60° of zenith [Landis 92], to provide coverage over 360° of longitude a ring of at least 3 ground laser stations would be required (4 would be required for coverage of geostationary orbit). For low Earth orbit customers, however, many more stations would be required along the orbital ground track, since a 500 km altitude orbit reaches an azimuth of 60° with respect to a point on the ground when only 900 km away laterally. Approximately 40 ground stations would be required for continuous illumination of a satellite at 500 km altitude and equatorial inclination. For the space station orbit, however, the 28° orbital inclination renders totally impractical the supply of ground based laser energy on a continuous basis as discussed in section 3.1.1.

Moreover, even the areas with the world's best weather factors have 50 to 100 cloudy days per year. If more continuous power must be supplied in order to attract customers, as many as double the number of ground stations may be required, separated by sufficient distance as to be in different weather systems, and this would increase power availability to nearly 98%, assuming no correlation between the weather at different ground station locations.

For the purpose of this analysis, we will make a baseline assumption of 10 ground stations required, noting that this does not allow for continuous coverage of low orbiting spacecraft.

Near Term

The principle unknown here is the amount of available power that will be purchased. In the near term this may be very limited (but may grow in the mid term as satellite and station designs take advantage of the service that is offered). On the basis of the market analysis in section 3.1.1, we will assume (optimistically) in the near term a total baseload demand for the system as a whole at 50kW delivered continuously. This would therefore generate revenue at a cost of \$1000/kWh of \$438M per year.

The accuracy of these figures is quite clearly far from reliable in absolute terms. However, we may use these approximate figures to calculate the Net Present Values for the commercial project. We will consider a 5 year development and construction period, followed by a 15 year operational period, but for simplicity's sake assume all investment costs arise in equal proportions in the first 5 years. Investment costs are development cost and ground station cost times number of ground stations. First revenue occurs at the end of the 5th year of the program The formula used for NPV in full is:

$$NPV = \sum_{y=5}^{20}\left(R_y - OC_y\right)\left(\frac{1+i}{1+r}\right)^y - \sum_{y=0}^{4}\frac{\left(DC + GC_{tot}\right)}{5}\frac{1}{(1+r)^y}$$

where DC is the development cost and GC_{tot} is the total ground stations cost. The NPV of this project, and its sensitivity to variation of the assumed parameters between low and high values, is shown in Table 11.1 below.

Table 11.1 NPV - Near Term Ground to Space Laser

Baseline Values	PV_{inv}	PV_{oi}	NPV
(\$M)	-2512	1440	-1072

Variable:	Baseline Value	High Value	Low Value	NPV low	NPV high
Development Costs	1000	2000	500	-1790	-713
Ground Station Cost \$M	250	500	100	-2866	5
Number of Stations	10	40	3	-6455	184
Operating Costs/Station \$M	40	60	20	-1144	-999
Amount of Power Sold (kW)	50	200	10	-2340	3683
Price Charged \$/kWh	1000	2000	500	-1864	513

It can be seen that with the baseline parameters chosen, the project makes a \$1B loss over its 20 year duration. As noted above, however, less emphasis should be given to the absolute value of this model's output NPV, than to a consideration of the sensitivities it highlights.

First of all, it can be noted that the most sensitive parameter is amount of power sold in kilowatts (average). If as much as 200kW could be sold the venture goes into net value of \$3.6B. If only 10kW are sold, the net loss increases from \$1B to nearly \$2B.

The near term markets, as identified in section 3.1.1, are extremely limited. The sale of 50kW on a continuous average basis is itself extremely optimistic, noting that all inclined low altitude orbits are in line of sight of equatorial ground stations for only a small amount of time. On this basis, it is considered justifiable to state that near term implementation of ground to space lasing is not commercially viable.

Mid to Long Term

For mid and long term markets we may use the same model, adjusting the values of our baseline parameters as necessary. Most importantly, we can increase the amount of power sold, but technological advances will undoubtedly lessen the cost of power in orbit as power sub-systems become cheaper, lighter, and longer lived. Moreover, the large scale power users, such as 1MW orbital transfer vehicles (OTV's), will pay still less for beamed power because of the economies of scale in large power systems (i.e. if customers using high power levels had to generate their own supply, they could do so more cheaply than the typical cost of power in space and therefore will be prepared to pay less for a supply of beamed energy).

We will estimate the future cost of power in space to be just \$200/kWh. An analysis of OTV costs and revenues indicates that an operator of an OTV service would pay at most a few tens of US dollars per kWh (based on a 1MW power supply over a round trip of 120 days and revenues from the mission of order \$60M). In order to most conveniently account for this reduced price for bulk users of power in the NPV model, we will assume that such bulk users will pay 10% of the price of power to small users. Therefore, we can generate the same effect on the revenues (or PV_{oi}) in the model by entering just 10% of the amount of power sold to such customers, but at the baseline price. If the amount of power sold, excluding transportation (e.g. OTV) customers, is taken to be 1000kW, and the amount sold to a fleet of 10 OTV's or other bulk power users is 10,000kW, we enter into the model an amount of power sold at 2000kW equivalent at the baseline price. The model then yields the results shown here in Table 11.2.

Table 11.2 NPV - Mid Term Ground to Space Laser

Baseline Values	PV_{inv}	PV_{oi}	NPV
(\$M)	-2512	12536	10023

Variable:	Baseline Value	High Value	Low Value	NPV low	NPV high
Development Costs	1000	2000	500	9306	10382
Ground Station Cost \$M	250	500	100	8229	11100
Number of Stations	10	40	3	4640	11280
Operating Costs/Station \$M	40	100	20	9806	10096
Amount of Power Sold (kW)	2000	3000	500	513	16364
Price Charged \$/kWh	200	400	50	513	22704

It can be seen that with the baseline values we have assumed, the venture generates large profits over its lifetime. The possible "high value" of the operating costs in the table has been increased to account for the potential cost of increased power beaming from the ground stations, but the sensitivity of NPV to this variable is in any case small. The high sensitivity parameters remain those that effect revenue: power sold and baseline price charged. It should be noted that if the "low values" of both power sold and price charged are taken together, the NPV becomes a \$1.8B loss, but remains in profit as shown in the table above if either low value is taken independently.

Power could be provided to the lunar surface by ground based lasers, but this market has not been considered, principally because it is a specific power requirement which depends on a decision to place permanent human settlements on the lunar surface. The power requirements of such a base and the value of power have been assessed as up to 5MW and \$500 per kWh on the basis of the cost of launch of a solar power/regenerative fuel cell system [based on data from Bozek, Lewis Research Center]. Lower estimates put the requirement at a more modest 200 to 500kW at \$100 to \$200/kWh. Clearly the former estimate would drive a power beaming venture into large profit even in the absence of other customers, on the basis of the model presented here.

Ground to Space Power Beaming using Microwaves

The Net Present Value model described above is not specific to laser power beaming techniques. However development and ground station costs for microwave technology are likely to be lower, since the technology is more mature, and the implementation simpler. Development costs may be of order a few hundred million dollars, and ground station cost perhaps $100M (since large antenna areas are required). Maintenance and operating costs are unchanged, since the size of the microwave facility will in part balance the reduced complexity over the laser ground station.

In the tables above, however, it was shown that the sensitivity of the project NPV to development and ground station costs was small. Indeed, the inaccuracies in other baseline assumptions will clearly swamp the small variations in development and ground station costs considered.

However, space based receivers will be considerably more expensive, since even with large ground antennas microwave beam spreading is significant and the spacecraft rectenna must be some hundred meters in dimensions (see section 7.2 for an explanation of transmission fundamentals). This may limit the market size for microwave beaming to space users, but since in the mid to long term we expect power customers' spacecraft design to reflect the available beamed power source, we make the baseline assumption that the market would be the same irrespective of whether a laser or microwave ground to space power system is implemented. This assumption is optimistic in the case of microwave beaming.

In the near term, the potential of microwave beaming is further limited since power delivery to existing spacecraft and solar arrays is not possible: a dedicated rectenna is required. There can be little doubt therefore, in view of the results of the near term laser analysis above, that near term microwave is also not commercially viable.

For completeness, however, the mid term case with modified development and ground station costs for the microwave case is shown in Table 11.3 as follows:

Table 11.3 NPV - Mid term Ground to Space Microwave

Baseline Values	PV_{inv}	PV_{oi}	NPV
($M)	-1077	12536	11459

Variable:	Baseline Value	High Value	Low Value	NPV low	NPV high
Development Costs	500	1000	100	11100	11746
Ground Station Cost $M	100	250	50	10382	11818
Number of Stations	10	40	3	9306	11961
Operating Costs/Station $M	40	100	20	11242	11531
Amount of Power Sold (kW)	2000	3000	500	1949	17799
Price Charged $/kWh	200	400	50	1949	24139

Though this shows a slightly increased profit over the laser beaming model, this is of a sufficiently small value that the conclusions that may be drawn are the same as those in the discussion of the laser beaming venture. The lunar market, however, may not extend to microwave beaming simply because of the transmission distances and beam spreading: a 20km circular antenna on Earth would still require a 5km square rectenna on the lunar surface at 2.45 GHz transmission frequency. With the exception of the lunar market, we may consider the results of the analyses relating to laser transmission above to apply equally to microwave beaming in all time-scales considered.

Conclusion

We conclude that the near term market is not a viable one for laser or microwave Earth to space power beaming. Both transmission technologies look promising in the mid and long term, and may generate multi-billion dollar profit over a 20 year period, but this depends crucially on the price and amount of power sold. Pessimistic assessments of both of these parameters drives the program into net loss, but a large lunar market would seem likely to ensure the viability of the laser power beaming project.

The decision on whether to implement a microwave or laser system in the medium term must depend on a more thorough economic and technical analysis in the future, when developments in laser and microwave technology can be re-assessed, along with the potential market size and value. This latter consideration depends largely on the degree to which large scale space development and exploration will occur in the time-scales considered.

Space to Space Power Beaming

Space to space power beaming offers an alternative approach to servicing the same markets examined for ground to space beaming above. As discussed in Section 3.1.1, however, space to space power beaming is essentially a mid term market (because space based laser systems are not yet technically feasible), that is a market that involves delivery of power to satellites that are designed specifically for such power reception.

The Mid to Long Term Market

In general, the transmission medium may be either laser or microwave, and while the former has advantages in terms of antenna and receiver size, microwave techniques offer greater efficiency and generally cheaper technology (see section 7.2 for an explanation of transmission fundamentals). It terms of cost, the size of the required antenna for microwave transmission will offset the savings of cheaper technology and greater efficiency. Accurate costing of power satellites is not practical at this stage without some details of satellite design, so for the purposes of a NPV model and sensitivity analysis for a commercial space to space power project, we will assume the cost of microwave and laser stations to be the same.

In determining the cost of a 1MW class space power station, we note that Brauch [Brauch et al 91] estimates the mass of a 1MW diode array laser system in the 100 ton class (187 tons). In section 10.4 of this report, the 1MW class microwave beaming station is also in the 100 ton class (130 tons), with estimated costs of order $1B ($1.3B). This cost corresponds to a widely accepted rule of thumb for space hardware costs of $100M per ton. A baseline cost of development and launch of a 1MW power satellite of $10B is assumed for this analysis. Recurring costs for subsequent stations is taken to be 60% of this value, so that we can designate development costs at $4B and construction costs at $6B per satellite. The development and construction costs are assumed to be evenly spread over a 5 year period, with revenue operations beginning in year 5.

Operational costs can be taken at $50M per year per satellite, since fairly complex monitoring, tracking and control will be required for such a system. However, we will take an optimistic baseline assumption that regular space based maintenance of the satellites would not be required, and that the power station design life is 10 years. Inflation and discount rate are 5 and 20% as before.

Three power stations in 7500km equatorial Earth orbit can supply power to both low and high equatorial orbits, with only minimal and short term gaps in coverage [Eurospace, 92]. To cover polar orbiting spacecraft, we assume a further 3 power satellites are required. Though power hungry OTV clients might not be serviced by such a system (each OTV would require a dedicated power satellite at 1MW), the high inclination LEO customer base which cannot be supplied by the ground based power system should more than compensate for this. We therefore assume the market related parameters (price and amount sold) used in the mid term analysis above. These give the results shown in Table 11.4.

Table 11.4 NPV - Mid Term Space to Space

Baseline Values	PV_{inv}	PV_{oi}	NPV
($M)	-28710	10444	-18266

Variable:	Baseline Value	High Value	Low Value	NPV low	NPV high
Development Costs $M	4000	6000	1000	-19701	-16112
Power Station Cost $M	6000	8000	1000	-26879	3267
Number of Stations	6	8	3	-26879	-5346
Operating Costs/Station $M	50	100	20	-18417	-18175
Amount of Power Sold (kW)	2000	4000	500	-26212	-7670
Price Charged $/kWh	200	1000	50	-26212	24116

It can be seen immediately that such a scheme is far from economically viable on the basis of the baseline parameters assumed. In this analysis, however, in contrast to the ground based power station case above, power station cost crucially impacts NPV. Amount of power sold and price of power are, as before, also sensitive parameters.

However, while we have assumed a baseline cost of space hardware which is typical of today's technology ($10M per ton), we have also assumed a price of energy in space based on extrapolation of technology into the future. This is perhaps inconsistent. If we take the present day cost of energy in space ($1000 per kWh, as discussed above) it can be seen from the table that the NPV becomes large and positive at $24B. A more appropriate analysis might be to assume both a reduction in energy cost in space to $200 per kWh, and a comparable reduction in space systems cost in general: this would bring the price of a power station to of order $1B. In this case, the NPV shown in the table is $3B profit over 10 years. If these values are input as baseline parameters, the NPV model yields the results of Table 11.5.

Conclusion

The NPV sensitivity remains greatest with respect to amount of power sold, price charged and power station construction costs. However, provided the power sold approaches the 2000kW baseline, the project NPV is large and positive. It is worth re-emphasizing at this point that a lunar base would be an important user of power and if one is established it might render the power beaming venture successful even in the absence of other significant customers.

Summary of Space Based Early Commercial Uses

The Net Present Value cost models used for the analysis of space based early commercial uses are only as accurate as the associated baseline assumptions and input variables. The models of this section have therefore been necessarily simplistic, and no particular confidence can be attached to their absolute NPV outputs. The sensitivities that the models display, however, offer some guidance as to the broad requirements for successfully commercial ventures. These revolve crucially around the amount of power sold and the price that can be charged for this power. In the case of space based power stations, a greatly reduced cost of space hardware over present day values seems essential for commercial viability.

Table 11.5 NPV - Mid Term Space to Space Baseline 2

Baseline Values	PV_{inv} \$M	PV_{oi} \$M	NPV \$M
(\$M)	-7177	10444	3267

Variable:	Baseline Value	High Value	Low Value	NPV low	NPV high
Development Costs \$M	4000	6000	1000	1831	5420
Power Station Cost \$M	1000	2000	800	-1040	4128
Number of Stations	6	8	3	1831	5420
Operating Costs/Station \$M	50	100	20	3116	3357
Amount of Power Sold (kW)	2000	4000	500	-4680	13862
Price Charged \$/kWh	200	1000	50	-4680	45648

The indications of the models are that with baseline parameters, multi billion dollar profits might be realized in the medium to long term. A continued and thorough review of the sensitive parameters (price of power and market size) would indicate the point at which a successful program might be initiated.

The decision on whether to implement a ground or space based microwave or laser system in the medium term must depend on a more thorough economic and technical analysis in the future, when developments in laser, microwave and space technology can be re-assessed, along with the potential market size and value. If space development is paced by commercial programs it is possible that technological advance will render space based power beaming stations more attractive than the ground alternative by the time the market for space power matures. If, however, space development and exploration activity accelerates in the mid term ahead of technological advance (for example as a result of a government backed space exploration initiative), a ground based power system may provide the first viable commercial implementation of space beamed power.

11.1.2 Space to Earth

The object of this chapter is to assess the cost for the long term full scale Solar Power Satellite from Space to Earth. During this study it was decided not to develop a point design for the long term solar power satellite system. In order to examine the costs of a full scale system a sensitivity analysis of the SPS cost study done by NASA/DOE [Satellite Power Systems (SPS) Concept Definition, 1980] has been performed.

The main guidelines and assumptions considered in this study are the following:

1. Cost estimates are in 1992 US dollars, scaled with a rate of 5% per year. All the units are \$M.
2. The scenario considered is 120 satellites with a power of 5 Gw each.
3. Overall SPS lifetime will be 30 years.
4. The satellite components will be launched from Earth and complete construction and assembly will occur in Earth orbit.

The Satellite Power System concept of NASA/DOE is based upon a large photovoltaic power collection satellite located in a geosynchronous Earth Orbit (GEO) utilizing microwave power to transmit the collected energy to Ground Receiving Stations (GRS) located at selected sites. The ground receiving sites then convert the energy received to a form compatible with local utility power networks. The available energy contributes to the base load power capability of the network.

This cost estimation includes all elements of hardware, software, and activities required for the design, development, production, assembly, transportation, operations, and maintenance of the space solar power full scale program. Included are the satellite and ground receiving station systems, as well as the necessary support systems such as space construction, support and transportation.

The costs are split in five major areas:

- Design, Development, Test and Evaluation costs (DDT&E).
- Theoretical First Unit cost (TFU).
- Cost of each new unit built within the project (New Unit).
- Costs of Maintenance and Spares per year and satellite (M/S).
- Daily Operations cost per year and satellite (Daily Ops).

The cost of the total space solar power program can be split into non-recurring cost and recurring costs. Non-recurring costs will be done just at one time. Recurring costs will be for each satellite and during each year. Non recurring costs include DDT&E and TFU. New unit costs, maintenance/spares and daily operations are recurring costs. Along the life of the project non recurring costs are affected by multiples factors that can not be completely defined or estimated at the beginning of the project. The aim of the sensitivity analysis is to quantify these possible effects in the entire project.

The sensitivity analysis has been performed considering possible fluctuations in the cost of the most important elements in each segment. The sensitivity analysis considers a range from the worst case when the cost increases to twice and the best case when the cost decreases to half (this is represented in the graph by the factor costs which multiply the assumed cost). The results are presented in graphs in which the central points corresponds the assumed value for each parameter.

The main segments in the space power system are the following:

1. Spacecraft.
2. Space Construction and Support.
3. Transportation.
4. Ground Receiving Station (GRS).
5. Management and Integration.

The total cost of an SPS Program ($M), split in the main subsystems according to the NASA/DOE study mentioned before are shown in Table 11.6.

Table 11.6 Full Scale SPS Program Costs

	DDT&E	TFU	New Unit	M/S	Daily Ops
Spacecraft	7799	9811	4978	33	0.7
Constr&Supp	8564	10757	210	19.5	19.3
Transportation	13154	23334	1990	78.5	17.4
Ground Stat.	135	4248	4217	0.3	33.2
Mngmt&Integ.	1482	2408	570	6.6	3.5
Totals	31134	50558	11965	137.9	74.1

The complete system with all the satellites (120) must be considered, as well as the lifetime of the system, to understand the weight that recurring costs have in the total cost of the project . These recurring costs are shown in the Table 11.7 as well as the fraction of the total project cost.

The recurring costs are almost 96.5% of the total project cost. Therefore the evolution of these costs along the life of the project has to be examined very carefully because small deviations of the predictions can have dramatic influence on the viability of the long term program.

Table 11.7 Full Scale SPS Recurring Costs

	Recurring cost	% total cost
Spacecraft	718680	31.51%
Constr&Supp	164880	7.23%
Transportation	584040	25.61%
Ground Stat.	626640	27.48%
Mngmt&Integ.	104760	4.59%
Totals	2199000	96.42%

Observing Table 11.7, it can be seen that Spacecraft, Transportation and Ground Station are respectively the main segment drivers of these costs. Sensitivity analysis of these three segments is very useful in order to evaluate what will be the influence of changes in these cost in the lifetime of space power program. The results of these analyses are shown in Figure 11.1.

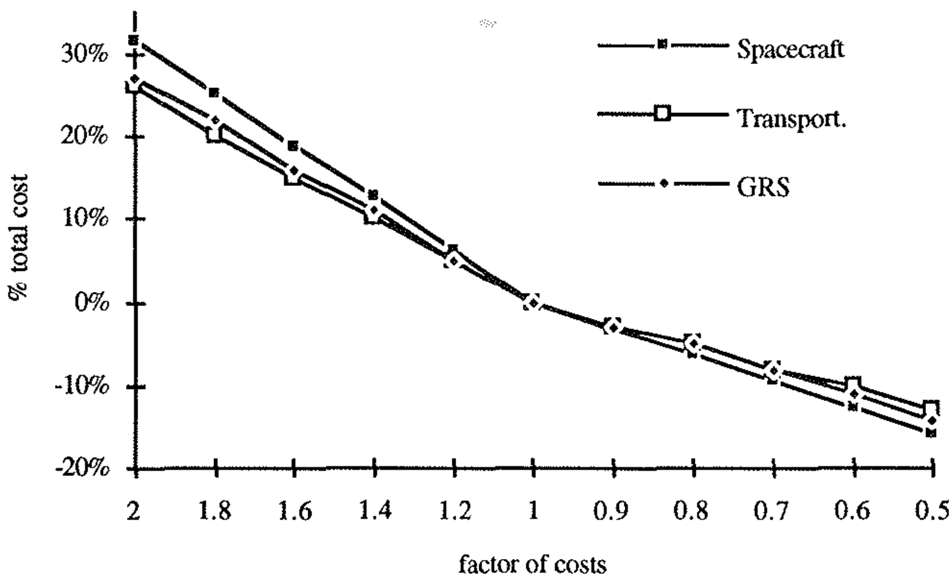

Figure 11.1 Sensitivity Analysis Curve of Spacecraft, Transportation and Ground Segments

The main feature of this results is that changes in the cost of one of this parameters cannot dramatically alter the cost of the program. A combination of cost reduction in each of these segments would be required.

In these results shown above, it has to be considered that the change in one of the segment costs may produce effects in the other two and this has to be considered in the final effect on the total cost. The links among these segment costs should be derived from the technical design and it is not the purpose of this chapter to go far into details.

This sensitivity analysis can examine changes from this reference design but no alternative designs effects have been considered. Launch and construction of satellites from the Moon, as well as the use of Moon resources, may result in significantly different allocations of costs. Chapter 9 describes some of the alternatives involving the use of non-terrestrial resources.

A brief summary of the costs of each segment (according to NASA/DOE reports) is presented as well as the sensitivity analysis of the most important factors in the rest of the chapter.

Spacecraft

This segment includes all elements that integrate the satellite formation placed in GEO. All the hardware and software use for these satellites to get solar energy and to transmit it to the Earth are mainly the components of this segment. Also is included the required ground support for these satellites.

According to NASA/DOE study the main split costs are shown in Table 11.8.

Table 11.8 Spacecraft Segment Cost

	DDT&E	TFU	New Unit	M/S	Daily Ops
Energy Conversion	247	4597	4228	13.52	0.05
Power Transmission	1874	4623	4147	43.98	1.02
Information Mngmt. and Control	176	414	341	1.14	0.00
Attitude control and Station keeping	27	146	130	0.42	0.00
Interface Energy Conversion/Power transmission	53	133	115	0.39	0.05
System Test	8961	0	0	0.00	0.00
Ground Supp. Equipt.	1134	0	0	0.00	0.00
Pilot Plant/Test	1566	7747	0	0.00	0.00
Totals	14038	17660	8961	59.45	1.11

Looking at Table 11.8 it can be noticed that the main cost driver subsystems are energy conversion (from solar power to microwave beams) and power transmission (from space to the Earth). Considering the 120 satellites and a lifetime of 30 years the recurring costs of energy conversion are $556B which is the 42% of the cost of the spacecraft system. The power transmission costs are $660B which is 50% of all cost of the segment.

The results of the sensitivity analysis of these subsystems are shown in Figure 11.2.

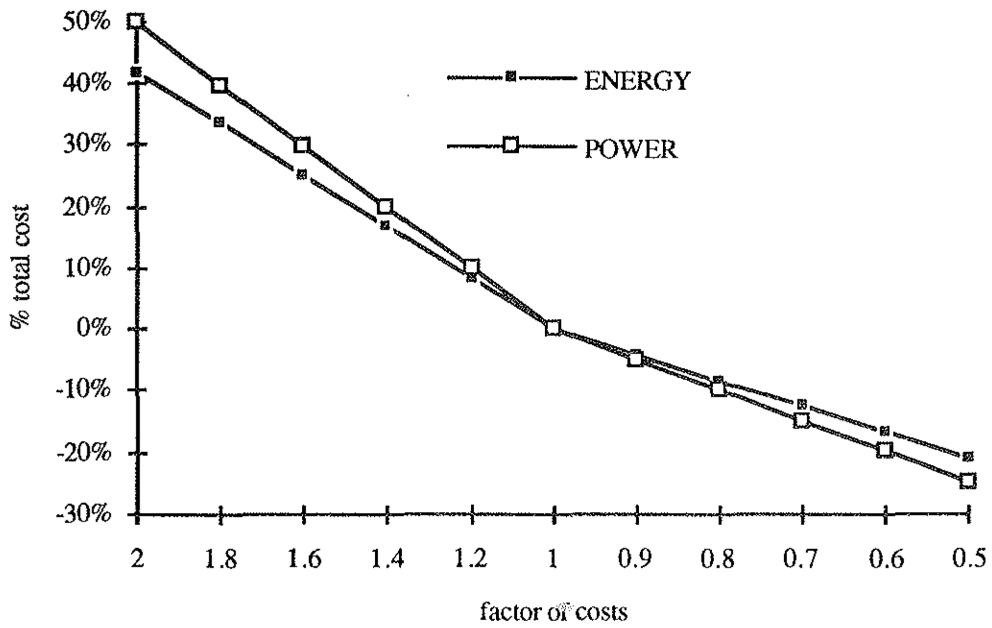

Figure 11.2 Sensitivity Analysis of Energy Conversion and Power Transmission Systems Costs in Spacecraft Segment

Space Construction and Support

This segment includes all the facilities and activities required to build the satellite formation. This include space stations, construction facilities, manpower operations, logistics, etc...

According to NASA/DOE study, the main costs are shown in Table 11.9.

Table 11.9 Space Construction and Support Segment Costs

	DDT&E	TFU	New Unit	M/S	Daily Ops
Construction Facilities	7670	14090	350	6.94	6.68
Logistics Support Facilities	7746	1964	28	1.07	0.23
O&M Facilities	0	3310	0	27.03	27.89
Totals	15415	19364	378	35.04	34.81

The recurring costs for this segment are $165B and represent 7% of the total space power program cost. Due to this small percentage the total cost is not sensitive to this segment so no further sensitivity assessment was performed.

Transportation

This segment includes all the vehicles needed to built the satellite configuration. It includes Heavy-Lift Launch Vehicles (HLLV), Cargo Orbital Transfer Vehicles (COTV), Personnel Launch Vehicles (PLV), Personnel Orbital Transfer Vehicles (POTV), Personnel Modules (PM), Intra Orbital Transfer Vehicles (IOTV) and Ground Support Facilities.

The main costs according with NASA/DOE study are shown in the Table 11.10.

Table 11.10 Transportation Segment Costs

	DDT&E	TFU	New Unit	M/S	Daily Ops
HLLV	18112	20938	2993	125.74	23.67
COTV	231	8242	204	5.45	3.80
PLV	516	5441	91	0.00	0.00
POTV	737	122	7	0.43	0.06
PM	249	518	7	0.75	0.06
IOTV	211	12	4	0.34	0.03
Ground Supp. Facilities	3622	6729	112	7.48	3.74
Totals	23677	42002	3418	140.18	31.36

Looking at Table 11.10 it can be noticed that the most important costs are those related to HLLV. The total recurring costs of HLLV subsystem are $897B which is the 82% of all the cost of the Transportation segment. This shows that for this solar power design the HLLV costs are the principle costs in this segment. Reduction in costs for this vehicles would have a significant effect on the total segment cost. However the Space transportation chapter has described that the HLLV costs are already estimated to be very low in the NASA/DOE study. The results of the sensitivity analysis of the HLLV system are shown in Figure 11.3.

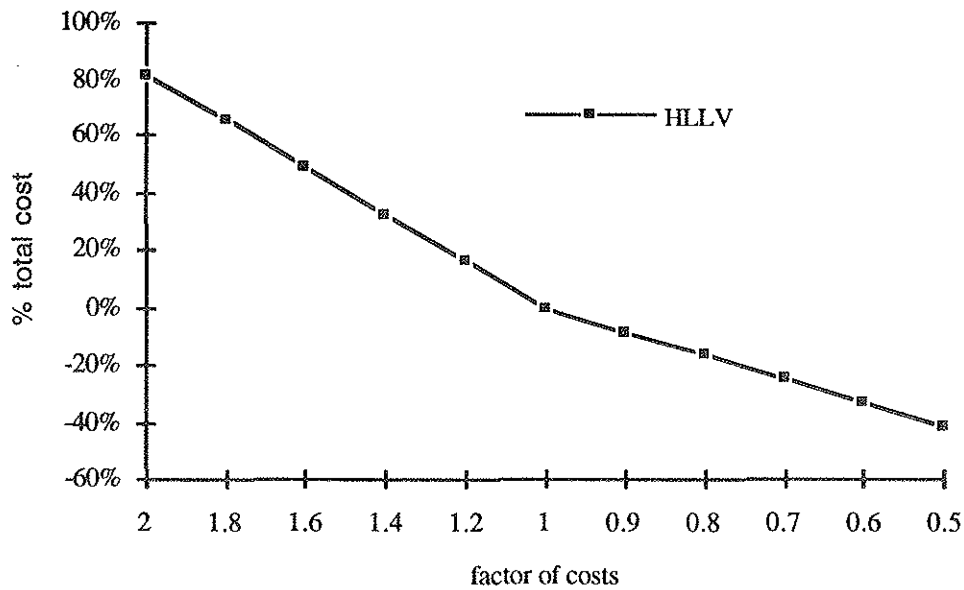

Figure 11.3 Sensitivity Analysis of HLLV Costs in Transportation Segment.

Ground Receiving Station

This segment includes all the facilities and equipment required to convert the microwave beam received from Space into power, which is ready to connect into the power grid.

According to NASA/DOE study the split costs of this element are shown in Table 11.11. It can be noticed that the main cost drivers are the rectenna structure and power collection. The recurring costs of the rectenna are $473B which is 41% of the cost of the ground receiving system. The recurring costs of power collection are $342B which is 30% of the ground segment cost.

Table 11.11 Ground Receiving Station Costs

	DDT&E	TFU	New Unit	M/S	Daily Ops
Site and Facilities	2.11	411	398	0.42	0.17
Rectenna Structure	4.21	3896	3850	0.16	2.83
Power Collection	6.32	2850	2850	0.00	0.00
Control	21.06	158	158	0.00	0.00
Grid Interface	209.97	335	335	0.00	0.00
Operations	0.00	0	0	0.00	56.81
Totals	243.67	7650	7591	0.58	59.81

The results of the sensitivity analysis of these subsystems are shown in Figure 11.4.

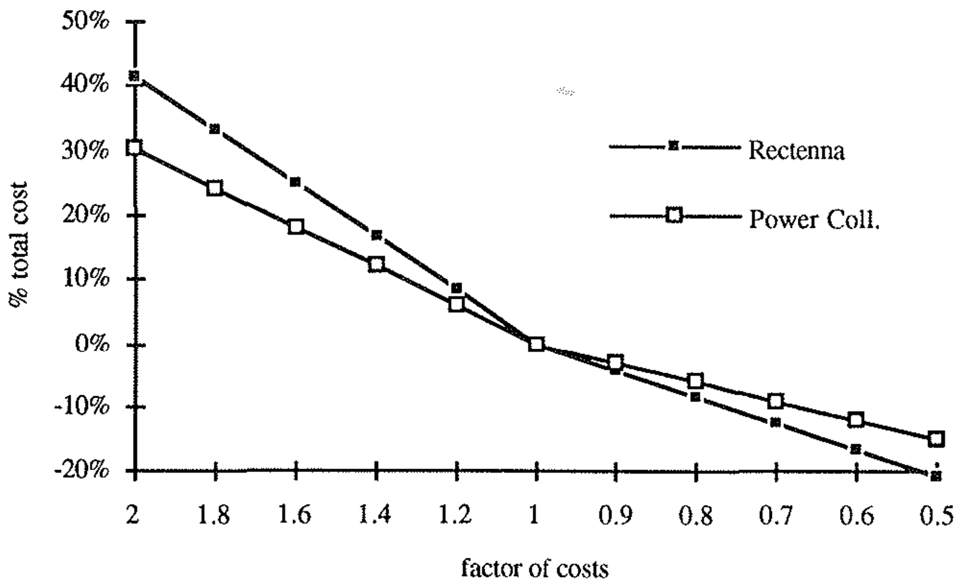

Figure 11.4 Sensitivity Analysis of Rectenna Structure and Power Collection on Ground Receiving Station Segment

Management and Integration

This segment includes the Program Administration, Planning and Control, Quality Control, Scheduling, Contracts, System Engineering and Integration.

According to NASA/DOE study the cost are shown in Table 11.12.

Table 11.12 Management and Integration Costs

	DDT&E	TFU	New Unit	M/S	Daily Ops
Management and Integration	2669	4334	1026	12.00	6.36

The recurring cost of this system are $105M represented only 4.6% of the total full scale project. Due to this small percentage, the total cost is not sensitive to this system so no further sensitivity assessment was performed.

Conclusions

The cost sensitivity of the NASA/DOE SPS design was examined. The recurring costs for the 120 satellite system constitute more than 96% of the total project cost. The most significant segment of the recurring cost are the spacecraft, the ground station and space transportation. The principle cost drivers in the spacecraft is the energy conversion and power transmission systems. The ground station cost drivers are the rectenna structure and power collection systems. The principle transportation cost driver in the NASA/DOE design is the heavy lift launch vehicle. A significant cost reduction in all these systems would be required to reduce the overall project cost by a large amount.

11.2.1 Financial Sources Overview

> "Engineers and physicists tell us that it is through celestial mechanisms and physics that the earth revolves around the sun and rotates on the axis, but we all know that it is really money that MAKES THE WORLD GO ROUND"...**Dr. Donald Bunker, ISU '92**

As Mr. Bunker told us, whatever you decide to do, it is with the purchasing power of money that you will be able to acheive it. Money is a unit of exchange and a store of value. Unfortunately, a limited amount of money's available in the World. As any industry, the aerospace sector participates to the quest for those scarce dollars, pounds, guilders or rubles in ways to achieve different projects. [Bunker, 92]

A few characteristics can be presented to depict the nature of space business:

- Huge investment requirements
- Market usually undeveloped
- Low level of products sold per year
- High risk of technical failure
- Risk of political interference
- No investors (except communications) have as yet received substantial return from their investment, and
- Few product have ever made it to the market on cost and/or on schedule [Diamandis, 92]

Considering those factors, one can have a good idea of how difficult it is to find funds for the realization of a project related to the space sector. The purpose of this section is to present the different sources of money existing in the world and to explain what type of commitment a company or agency should expect to putforth to obtain those scarce resources.

Sources of Funds

There are various sources of funding the space project financing, which can be considered available.

Capital Market

The capital market is principally composed of banks, trust companies, insurance companies, pension funds, venture capital and the general public (private placement & public offering). The purpose to invest varies widely from one investor to another. The purpose of a financial participation in a project can be to vary the degree of diversification of a financial portfolio, to get more security, or for the trill of a great adventure. The characteristics of space projects (described in the introduction of this section) tend to limit capital market investors to senior financial institutions like banks or insurance companies.

Of the big market players, three participants can be considered in more detail: financial institutions, venture capital, and the general public.

Financial Institutions

Banks have to be considered as really conservative lenders. The main goal of the business of commercial banks is to make profits for the shareholders in satisfying the credit needs in the market (businesses and individuals). Banks face a conflicting situation. They must respond to safety and liquidity needs holding cash or near-cash, and they have to maximize return of investments putting funds in long-term loans with a greater risk premium. However, a bank's own risk capital represents a small part of the total amount of money loaned. Consequently, they will become interesting lenders when projects will be considered relatively safe and show short term (5 years) profit possibilities.

Venture Capital

Venture capital will become a source of funds for companies starting new operations or reaching a new market. High-technology companies can be considered as interesting targets for venture capital. Generally, venture capital funds come from rich individuals and financial institutions. Venture capital is more likely to fund relatively small companies considered too risky for more conventional financial organizations.

The General Public

Underwriting houses which present potential investments which present potential investments to the public through an offering secured by a commission. This gives companies access to the investments funds of the general public. The essential function of a stock exchange is to provide a central market for purchase of shares, regardless of the size or location of the broker's firm. Securities are bought or sold on the same terms. Accepted securities (following securities rules) only can be traded on the stock exchange floor.

A stock exchange cannot supply equity capital to industrial companies directly, but it offers services and facilities to permit the exchange. Once a company goes public it has to fulfill specific commitments like continuous reporting and opening of its financial statements to public scrutiny. Its shareholders become business partners.

Governmental Support

The comparative advantage economical concept states that no two countries are alike in business specialities and that they will try to maximize the profit that they can obtain over others countries using that advantage. An international space speciality for a specific country can be proposed as an example.

Most industrialized countries have specific governmental agencies to support companies and industries to exploit that comparative advantage in providing legal, political and financial support to attain fixed goals. Help can specifically be provided through insurance, loans and guarantied programs.

Manufacturer's Assistance

This source of funds may seem to be unusual, but in the aerospace industry, financial burden insured by the manufacturers for research and development can be considered as a source of pressure for cooperation. Added to the high level of competition between the manufacturers in the space sector, and the selection process used by the space clients, the manufacturer has to take a variety of financial risks.

Interest

Interest can be defined as the cost of using other people's money to complete a transaction or a product. When a borrower accepts a fixed interest rate from a lender, the risk he encounters relates to a fall in the market interest rate, which would put him is a disadvantageous position with respect to competition.

To protect themselves from those risks of rates variation, borrowers and lenders use various means, such as interest rate swap: this changes the nature of the risk by limiting interest payments from floating to fixed rate or from fixed to floating rate.

Currency Risk

When a corporation proposes taking on obligations in a currency other than that of its base country, a risk occurs. Three situations can be observed about currency risks:

- Assets financed outside of the base country
- Assets financed locally by a lender without sufficient source of local funds

- Lender from inside or outside of the base country proposes a foreign dominated loan

Foreign exchange risk should be avoided using an adequate currency risk protection. [Bunker, 92].

We consider three opportunities for Space Solar Power Program financial sources. They are international, government and private financing. Government sources can be acquired from space funds, public energy companies and by selling bonds. In case of United States government funding can be also held through the Environmental Protection Agency (EPA). European countries can obtain their contributions or a part of them through European Space Agency (ESA) and European Environmental Agency. International sources include money going from the IRDB (World Bank) and International Monetary Fund. Private money can come from oil companies, venture capital and shares sales.

Government Funding

Government funding can be obtained in 2 ways, as shown in Table 11.13 below :

Table 11.13 Government Funding Opportunities.

WAY	ADVANTAGE	DISADVANTAGE
Govts. are paying upon projected use	Charges based upon usage of energy	Developing nations can prefer more dirty but cheap sources
Govts. contribute upon % of GDP.	Developing nations can use more energy	Less feedback, than in a first case.

The first method is an Intelsat- type funding scheme. But since we need ecologicaly pure energy sources with worldwide coverage, it is vital to encourage developing nations to use space solar power instead of coal and oil energy. The way to do that is to contribute a percentage of Gross Domestic Product (GDP). In this case a part of payments for space solar power usage by developing nations is made, in fact, by developed countries, but in such a way developed nations will pay also for the insurance that almost the whole World is using ecologicaly pure energy sources. It will also provide developing nations possibilities for future economic growth.

This way is similar to funding European Space Agency mandatory programs, but without mandatory obligations for allocation of the contracts. It is suggested to ask for a percent of GDP up to 0.4. This is based on the amount of money approximately spent by developed countries for their space programs and it will cover an Space Solar Power Program costs. This can give us up to 48 Billion current US dollars yearly, as shown in Table 11.14.

Table 11.14 Demo Stages Of The Project.

Total World GDP	$12,000,000 mln.
Developed nations	80%
Developing nations	20%
Yearly Percentage	0.4%
Yearly funding	$48 Billion

To find money for such an expensive project governments, as was mentioned, can use three ways (space funds, public energy companies and bonds). But this scheme is available for the developed countries, while developing countries could have difficulties in finding cash for such a project.

In ths case, it's offered to use a payment-in-kind option . A poor country which may not be able to contribute hard currency will contribute either data, scientists, hardware or software. In this way, loans could be obtained from the regional development banks (World Bank) for use in the developing country. Another way to do that is to get some software or hardware from the developing nation instead of hard currency input [IPEO, 90].

A special case is Russia, which can't be considered developing as well, as developed. Taking into account that it is almost impossible for Russia to contribute hard currency, payment-in-kind can be the only possible way to participate. In fact, Western governments now show an increasing anxiety about an instable group of first-rate Russian space and nuclear engineers who are out of job and money and eager to work on any weapon for anybody who will pay. These engineers and scientists can be used in Space Solar Power Program, going directly from developed countries governments, in order to prevent "Russian brain drain" to the Third World countries.There are a lot of projects to create such funds and Space Solar Power Program can be a good allocation for that money.

The abive mentioned scenario is an ideal case, when all the countries will join the project and contribute as much money as required.In reality we can expect, for instance, strong opposition from the side of oil companies and Arab nations which will be reluctant to stop oil usage. On the other hand there is an existing possibility that Arabs may wish to use some of their present wealth to "buy-in" to future energy production systems.

International Funding

A program such as SSPP can also be funded by some international bodies. There are two major debt possibilities :

- UN, through International Resources and Development Bank (IRDB). In this way it we can also partly cover a share of developing countries.
- International Monetary Fund

Private Funding

Private funding sources are not an object of serious studies at the first demo stages of the project. In future, money could come from :

- Oil companies - in case they will face with threat to be out of resources and will look for possible alternative business.
- Venture capital - in case of fast and profitable payback.
- Advertising companies-as far as governments will allow them to use SSPP for advertising.
- Shares sales
- Commercial banks - if they will see that project is stable or guaranteed by governments.
- Pension funds - also in case of stable and evidently safe project.

11.2.2 Financial Risk Analysis

"Sometimes I only find where I should be by going somewhere I don't want to be"
-- Buckminster Fuller

This idea presented by Dr. Peter H. Diamandis during the ISU summer session 1992, gives a good indication about the nature of risk. With enormous investments required, markets which are underdeveloped, technical failures which are probable, and weak possibilities of return from investments expected, the space industry can be considered as risky. [Diamandis, 1992]

Financial risk associated with a specific project can be defined in several ways. This type of risk can be assessed as the variation of possible returns emanating from the project considered. It is frequently measured by developing cash flows based upon pessimistic cost or revenue assumptions. The longer the time period involved in realizing the project, the less certain the investor is of a return. This is because the events and conditions influencing the return in the distant future are not foreseeable.

To minimize the financial risk identified by potential investors, a sequence of actions has to be implemented at all levels of the project. The process of managing risks as a means of planning how best to survive setbacks is called Risk Management and can be identified through four basic steps:

- Identify the risk
- Evaluate the risk
- Determine the best way of dealing with the risk

• Implement the chosen method.

While that management technique [Higginbotham, 1990] may appear simple, for the high technology, international, legal, innovative aspects and global political environment of a program like the Space Solar Power Program, risk management can be very complex. The first step, identification of potential risks, should be done at all levels of the relating project. Quantitative actuarial approaches have been developed by insurance companies to evaluate risks associated with different events, but these do not necessarily apply in the case of the Space Solar Power Program because it is unique (no statistics available), it has little homogeneity with past projects and the concepts have not been tried yet. Considering those factors, assessments based on combination of empirical results, engineering and projects management experience, past tests results and margins of performance and errors will be established [Barret, 1990]. There are essentially four basic ways to handle risks: avoidance, reduction, transfer, or retention.

Unique to the Space Solar Power Program, the approach selected is to identify elements which could be considered as major "show-stoppers" from a financial point of view. Considering those identified risks, the sections 11.2.4 and 11.2.5 of this chapter will try to observe their potential impacts on the expected results for the Space Solar Power Program . The assumptions under which that analysis have been formulated are as follows:

• The Space to Earth demonstration case and what follows will be considered;

• The Space to Earth demonstration is a platform beaming electricity to Antarctica;

• Private funds will not be utilized before a successful space to Earth demonstration.

Based on discussions with members of specific specialist groups associated with this report, six categories of risks have been identified and will have to be considered seriously to give confidence to the potential investors. Risks relative to market, management, political, environmental aspects, technical aspects, and general risks have been identified, and will now be detailed.

Market Risk

The first risk category relates to the energy market. Relating to the energy which will be produced by the Space Solar Power Program systems, two major factors can be identified as major risks to be considered by potential investors: cost of production and the state of the technology necessary.

As discussed in Chapter 3, the energy market can be analyzed in three phases: near, mid and long term. Considering the near term market, space to space activities studied relates essentially to beaming of energy to communication and Earth observation satellites. Potential market related to that type of activity seems to be limited in the near term considering the unreliability and maturity of the existing technology (assuming that no modification would be made to the satellite) and the high cost associated with this type of activity. Other near term activities for which market seem to be highly risky and difficult to evaluate consist of new fields of space to space activities such as beaming of energy to space stations or free flyer platforms.

Relative to the mid term market, major space to earth applications to which a risk factor can be associated relate to four specific aspects: remote locations with developed energy demand, locations with little power capacity, power relay from one place to another and supply of electricity to networks to fulfill peak power demand. Each of those markets contains a potential show-stopper . Reliability of energy availability to remote locations market will have to be demonstrated. The developing countries market shows a serious lack of funds to purchase the supplied electricity. Technology associated with the power relay has to be developed, and reliability of beaming to electrical networks during peak power demand has yet to be proven. So, mid term potential market risks are essentially associated with a lack of reliable technology.

Finally, the long term market shows a lot of possibilities for the Space Solar Power Program energy to be utilized. However, regarding the industrial and domestic energy demand, it is again dependent on the cost effectiveness of power supply and reliability of supply systems. The improvement of technology is a milestone that must be reached in order to realize a cost competitive systems These issues will be discussed in the technical section.

Management Risk

Two major potential risks associated with international space project management are considered by investors, delay in project completion and underestimation of realization costs.

Major international projects can be characterized by large budgets (billions of dollars magnitude) and long duration for completion as well as complex planning procedures. These factors, combined with the large number of individuals associated with the realization of the project complicates its management. The most likely risk associated with this combination of factors is a realization delay. The impact of a realization delay on an investment is to be considered seriously by any potential investor. A Space Solar Power Program would likely be associated with that risk.

The second risk, underestimation of realization costs, is a generalized characteristic also observed in major international projects. Basically, the delay factors presented in this chapter apply to cost estimation. Considering the complex planning procedure, budget underestimation is to be seriously considered. Space transportation, solar cells and orbital construction, and rectenna building can be pinpointed as the major sources of costs relative to the realization of the Space Solar Power Program Considering the magnitude of the potential investment associated with the realization of those elements, a financial risk factor should also be considered.

Political Risk

Political risk addresses the complexity of large international cooperation and the potential for disagreement in the realization of a long term, high cost, and highly innovative program.

The first element to be considered is the site selected for the beaming experiment. Antarctica represents a protected land. A treaty and moratorium signed by over thirty countries dictates that any exploitation of resources present in Antarctica is forbidden. Any activity which would threaten the Antarctic environment or which could be considered as "for-profit" cannot be initiated.

A second aspect which should be considered on a political point of view is the political threat or public unpopularity associated with the building of the rectenna in Antarctica. Considering the fact that Antarctica has been specifically selected as the site for the space to Earth demonstration to prove its environmental safety, the potential damaging effects of rectenna construction could generate public disapproval and compromise Space Solar Power Program long term public support. Public funds are essential in the first phase of the program, and therefore public support is essential to the Space Solar Power Program development.

In a more long term perspective, political concerns could be addressed regarding the considerable amount of public money needed in the different phases of the SPS program. The difficulty encountered by space programs in obtaining public funds, and general trends to diminish public expenses for space program realization indicate that obtaining government funding for another major space program will be difficult.

Another element associated with public funding and which should be considered as a risk source is the yearly space project budget allocation. In countries like the United States of America, which can be considered as a major partner in international project realization, the public budget allocated to agencies' projects participation is reevaluated and allocated yearly on the base of political and social priorities. So, each year the project's budget must be defended and it is possible that the project could be canceled at any moment. Withdrawal of a major partner like the USA can compromise the achievement of the Space Solar Power Program.

Considering the international nature of the Space Solar Power Program, withdrawal of one of the participating countries can be considered as a threat for the realization of the program or at least as a source of delay. Considering that in an international program realization each country's contribution is often related to its unique field of specialty, withdrawal can mean that a specific technology or system will not be available.

Technology transfer could be another source of financial risk. Considering again the international nature of the Space Solar Power Program and the necessity to share some technology, restrictive policies to the exchange of industrial techniques could become a threat to the participation of certain countries in that specific project.

Finally, a less considerable but still possible source of risk is the possibility of a terrorist attack against the Space Solar Power Program structures. The large size of the predicted Space Solar Power Program systems in terrestrial orbit and their proximity to Earth could turn out to be a risk source.

Environmental Risk

Environment financial risk is associated with public support. Today the environmental factor is one of the most important social concerns. Related to that observation, any aspect of the Space Solar

Power Program which could be considered as a threat to the environment has to be seriously analyzed. Lack of public support will mean withdrawal of public funds.

Environmental risk can be associated with three major elements of the Space Solar Power Program: beaming of energy, satellite construction and rectenna construction.

First, concerning beaming of energy from space to earth, several studies will be initiated concerning to the effects of beaming of large quantities of energy from space to earth . The potential financial threat could be related to two specific elements: negative impact of the environmental studies concerning the effects of beaming and the relatively large costs associated with the completion of the environmental impact studies. There exists a general concern that doubt is present in public opinion concerning the security of beaming energy to earth. Any conclusion supporting harmful beam effects could seriously threat public acceptance of the Space Solar Power Program. The important costs of realizing those studies could also turn to be a show-stopper as public funds will possibly be used to achieve them. A long and expensive process of information acquisition and analysis could be politically unpopular.

The second concern relates to the construction of the satellite or space structures. If the Space Solar Power Program goal is to produce large structures in terrestrial orbit, many launches will be necessary in order to put the necessary building material in orbit. A major environmental concern relates to the atmospheric pollution associated with those launches. If it is proven that intensive launching activity causes environmental damage, Space Solar Power Program could lose public and Government support.

Third, a risk associated with the construction of the rectenna could also be considered as a major show-stopper. Considering the innovative aspect of building gigantic rectennas on Earth, a great numbers of environmental impact studies will be performed. Again, negative conclusions concerning the effects of beaming energy on those rectennas and the size of the budget allocated to the realization of those studies could compromise the accomplishment of the Space Solar Power Program.

Finally, the safety of workers in space also requires additional study. If building of large structures in space means that many long duration manned missions have to be performed, status of human life protection will have to be seriously considered for two reasons. First, the Space Solar Power Program could be unpopular if it is proven that human life would be endangered by the nature of the activities to construct it, and might be postponed until it was shown as a safe activity. On the other hand, a too expensive life protection system (shuttle to come immediately back to earth in case of emergency) could threaten realization of the program due to high costs.

Technical Risk

Clearly identified as the critical issue for the Space Solar Power Program potential completion, technical risks relate essentially to the state of the art and the cost associated with the development or realization of identified technologies.

Specific show-stoppers which can be identified as risky from an investor point of view can be associated with the necessity to develop specific technologies necessary to achieve the Space Solar Power Program from an economical point of view. Delay in the development of some of the identified components of the system could contribute to the financial risk.

First,. the necessity to lower launch costs has to be addressed. The importance of developing a low cost heavy launch vehicle has been identified as the element which could permit building of large structures in space. Transportation is considered as the major cost driver in a program like the Space Solar Power Program. As such this element can be pin-pointed as a source of cost increase or as a major source of delay in the realization of this program. Another cost factor of major concern is system maintenance: beam generators, satellite solar cells and structures being considered as the sources of risks.

Other concerns which have to be addressed as financial risks based on the present status of the associated technology are the following: Attitude control of large platforms, the capacity to develop large flexible antennas, improvement of the efficiency of solar cells and the necessity to develop accurate and effective robotic systems to assemble large structures in space if an unmanned program is chosen.

Other Risks

More general concerns have to be addressed to conclude this section and complete the financial risk analysis.

An element which will be out of the control of the Space Solar Power Program leaders is competition. Development of a cost-competitive alternative source of energy could mean that the Space Solar Power Program would be abandoned. Considering research efforts involved in fields like solar and fusion energy, the market impact of alternative sources has to be studied seriously. In the same way, research resulting in the development of a process to extract CO2 from the atmosphere would calm environmentalists concerns of effects of coal and oil burning on the atmosphere.

Considering all the elements presented in this section as potential risks from a financial point of view, financial investor confidence should be a major concern of the management. This concern should be seriously addressed during the development of the International Solar Power Organization. Potential success of the Space Solar Power Program will strongly depend on the efforts to reduce those risks.

11.2.3 Staged Plan for Financing

The purpose of this section is not to define a detailed plan for the coming hundred years, but to outline the global plan and to describe principles that can be used to refine and adapt the plan as the program develops. Our approach is to implement space solar power step by step, gradually taking it from a development stage to a fully operational one. Each step should have clear objectives from the business point of view and reaching these objectives will build up confidence for starting the next step of the plan. The project size gradually increases from stage to stage, mainly indicated by the power levels generated. The plans for each of the two main markets (space based and Earth based) are separated, though there is commonality in approach and they can share technology. At first, the overall planning is discussed from a business point of view, followed by the approach to commercialization of space solar power activities common to both Earth and space markets.

Plan for Space to Earth Solar Power

The plan for space to Earth solar power shows technical demonstrations with low powers at first, in order to build up general credibility and facilitate the large scale deployment. Hereafter the operational capability will be installed, in 3 steps increasing the power delivered to full scale (5 GW). The steps proposed are:

- Demonstration 1; space to space power beaming
- Demonstration 2; space to Earth power beaming
- 1 MW solar power satellite; revenues will allow for first business application
- 500 MW solar power satellite; intermediate step
- installation of 5 GW solar power satellites, evolution to full deployment

This is depicted in Figure 11.5 below.

Demonstration 1

For Demonstration 1 a few hundred watts will be delivered from space to a space located experiment, for a total project cost of $80 million. The main technical objectives are to demonstrate technical performance of power beaming by microwave from an antenna to a rectenna. Although the power delivered values $1000 per kWh, the short delivery time gives small revenues and it is not expected that they can be used in a commercial sense. More important from a business point of view is that the International Solar Power Organization (ISPO) should manage the project in order to show that the approach followed by ISPO delivers projects that deliver performance on time and within budget. This should build credibility for the ISPO, and giving confidence that it is capable of successfully managing all aspects of the next stage.

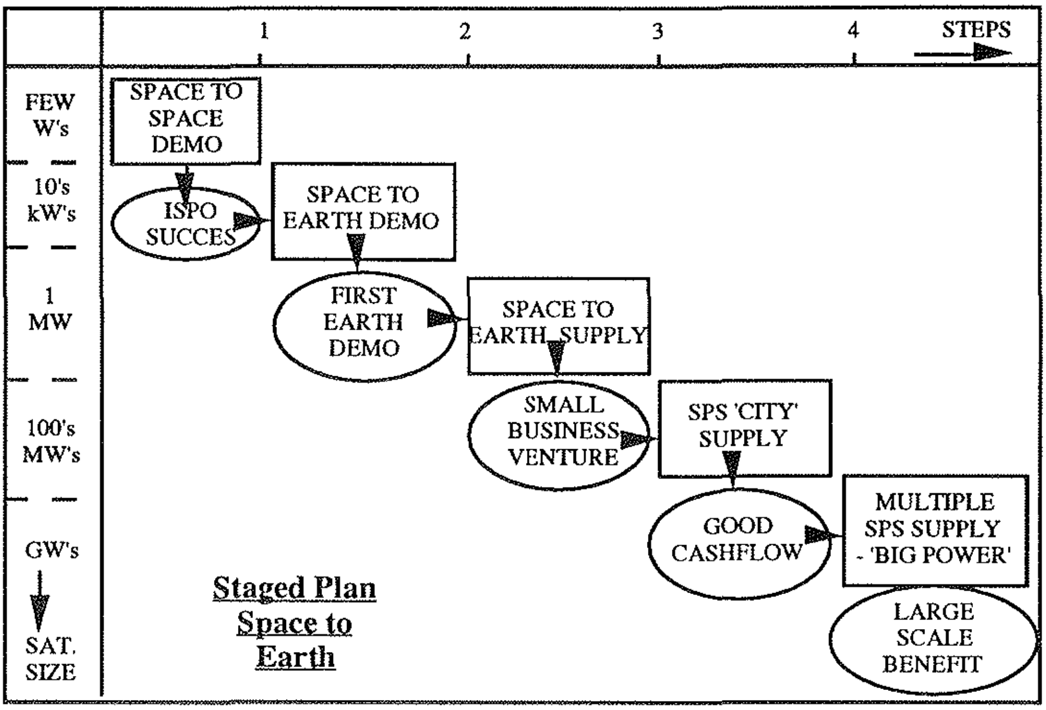

Figure 11.5 Staged Plan for ISPO Financing

In the same time frame an experiment of beaming through the Earth atmosphere would be advisable to develop the knowledge in that area and reduce risks of the next step of the program. This can be done on quite a small scale, for instance the Arecibo Earth to space demonstration as described in Chapter 10.

Demonstration 2

Some 40 kW of power will be beamed from sun synchronous orbit to a remote location for Demonstration 2, typically providing at a 100 minutes interval 10 kW of power average for a 7 minutes time span to each location. Duration of this experiment is 2 years, and the yearly revenues are $4600 at $0.22 /kWh for each location serviced. In case of servicing Antarctica, the value of $0.58 /kWh is applicable, giving yearly revenues of $12,000. Project costs are budgeted at $800 million, but in order to reduce direct costs the platform could be shared with other smaller payloads. This next step of the staged plan is to support the concept of solar power from space and to show that tangible benefits can be generated to humans by using space solar power, both direct benefits in the form of providing a safe and reliable energy source, and more indirect benefits as the substantially lower impact on the environment than conventional energy resources. It will also show technical performance and limitations of beamed power. From a business point of view, this step would demonstrate the market for this kind of power generation, and give confidence that the Space Solar Power Program is worthwhile to pursue.

First business application

The next step is to go to a size that delivers some substantial revenues, and a size of 1 MW is regarded as lower limit for this. Chapter 10 describes a suitable design example. One megawatt of power will be delivered to Earth for several years, typically 10, and since this application does not use a geosynchronous orbit, power delivery is at 12 hours interval for 6 hours 24 minutes per day. It is assumed that this satellite can deliver power during morning and evening peak loads as to maximize the revenues. At a price of $0.10 /kWh the yearly income is about $230,000, with a total revenue over its lifetime of $2.3 million; the cost of design and development (including first unit) are estimated at $2.5 billion. The yearly revenues could mainly be used to demonstrate that the business as organization works well, and the generated cash can give some return on the initial investment.

Intermediate business application

As a step towards the large scale gigawatt size power satellites an intermediate size typically of 500 MW delivered to the grid is proposed. Lifetime should be 15 years. If a similar orbit is selected for this than for the previous one, hence about 6.4 hours delivery per day during peak load times, yearly revenues would be about $116 million. Over its 15 years lifetime this could total to $1750 million. Project costs are not specifically estimated, and are probably $23 billion for recurring costs (development not included). With a constant and predictable cash flow the full deployment of space solar power could be further driven.

Large scale implementation

Next step would be transition into full scale power delivery with a typical size of the power stations of 5 GW in geosynchronous orbit. This orbit maximizes the revenues, compare delivering about 23 hours power for $0.05 /kWh instead of 6 hours of power for $0.10 /kWh a day. Revenues for ISPO are substantial now, about $2 billion per year per operating satellite. Lifetime for each power station is assumed 30 years, with replacement costs of a satellite being estimate at $30 billion. For more detailed estimation of project costs please refer to section 11.1 of this report.

Plan for space to space solar power

For space to space solar power, section 11.1.1 of this report concludes that near term commercial uses do not seem viable at this moment. It is found that the size of a power satellite should be large compared to the small market identified on the near term, with large scale use giving better opportunities for a space to space application, for instance a lunar settlement. Consequently, we give below the typical plan that could be adopted to develop this market if a near or mid term use is foreseen. For this market we propose the same basic steps as for the space to Earth market, but in general the amount of power delivered will be smaller than for Earth with the following milestones:

- technical demonstration
- first business application
- first full scale power generation
- evolution to full deployment.

The demonstration case 1 discussed before, space to space power beaming, could be used as a stepping stone for the development of this market. For next steps the upscaling of the satellites as the delivered power increases should be done in steps, like for the space to Earth market. We assume that since the price of power is high, $200 /kWh for the mid term, some business application could be found for the early steps, before reaching full size. Only if the satellite size can grow to MW size, commercial use is deemed viable (see section 11.1.1).

Commercialization of ISPO

By commercializing ISPO the stimulus of possible profit will spread the participation over a larger number of industries and countries, and will drive to lower general cost levels. Thus, benefit to all players is expected. The approach to commercialization of ISPO is similar for space and Earth markets and is described hereafter.

Since investments are large and substantial revenues are not expected in the first 10 years of the space solar power program, we assumed that it starts as a mainly public funded program. The principle we use to commercialize the solar power activities is that, as soon as actual revenues of selling solar power cover certain activities of power delivery in a profitable sense, these are transferred to a commercial organization. For example, rectenna operations could be transferred to ISPO as first commercially run activity. In this way, reasonable returns on related investments can be made, and private funding sources could also be used to finance these activities, instead of public funding only. In general terms, the operations could be privatized as first step due to their relatively low cost. Next, launch and manufacturing costs could be covered, with as most costly part the development activities being the last step. Hence, ISPO as organization could step by step be responsible for a larger part of the cycle of development, manufacturing, maintenance and operation activities of the total system, starting at the end of the sequence and working backwards. This should be a dynamic process based on the actual costs and economic conditions.

Based on the estimates of the cost for the various parts as presented earlier in this chapter (section 11.1.2) we have drawn a typical plan. Below this plan is given by a sequence of activities which are proposed to be privatized step by step. The steps show roughly an increasing monetary size, with typical cost estimates being indicated as below:

- maintenance of the ground facilities: $0.5 million/year

- ground operations of the space segment: $1.3 million per satellite per year
- running cost of ISPO: $18 million per satellite per year
- operations of the ground facilities: $60 million per rectenna site per year
- maintenance of a spacecraft: $60 million per satellite per year
- operations of space transportation and assembly: $66 per satellite per year
- maintenance of space transportation and assembly: $176 million per satellite per year
- cost of in-orbit assembly: $380 million per satellite
- cost of launch for a new satellite: $3580 million
- cost of a new ground station: $7600 million
- cost of management and manufacturing of a new satellite: $10 billion
- development cost of the first 5 GW SPS are estimated at $210 billion; as a intermediate step improvements of the SPS design could be covered by ISPO itself (no cost estimates available)

The above cost figures are based on the data of section 11.1.2, with a contingency added of 15%. All values are in '92 dollars.

It is clear that the demonstration cases do not generate revenues of real financial importance, and only the intermediate size satellite, giving about $116 million revenues per year, can be used to privatize the first four steps of the above sequence. The other steps should be gradually split-off to the private sector as the full scale implementation of space solar power proceeds.

Similar approaches are followed for instance by Ariane Espace, covering manufacturing and all operations of launchers, or by Spot Image, covering operations of remote sensing satellites. The other parts of the total cycle (design, development, etc.) are still covered by public funding sources for these cases.

11.2.4 Financial Options for the SSPP staged plan

This section will address the financial sources which will be required during the evolution of the SSPP staged plan. The information provided will draw on the overview of financial sources presented in section 11.2.1 of this report, and selection of financial sources will be based on the financial risk analysis presented in section 11.2.2.

An outline plan showing the steps required to introduce full scale space solar power to the energy utilities market was presented in section 11.2.3 of this report. This provided the reader with a financial overview of the growth of SSPP and the steps required to secure continued financial suport for the program. The financing requirements for each stage of this plan are different, and hence the funding sources utilised to implement each stage will differ.

There are three options for the financial structure of each stage of SSPP. These are:

1. Purely private funding with fully commercial activities.
2. Hybrid structure - government funding with commercial activities to private funding without commercial activities.
3. Purely government funding with no commercial activities.

The relevance of each of these scenarios is governed finacially by the potential private sourcing of space projects. Conventional space projects are often too high a risk and offer too little return on investment to be considered by financial institutions.

It is considered that SSPP too would not provide sufficient return on investment in the early stages of the program. It is therefore not viable to begin with a purely private financed institution. As SSPP develops and increases it's market over the next twenty plus years, the potential for private financing will progress as more of the individual sub-sections of the SSPP stages are able to be run on a commercial basis. This assumption is based purely on financial considerations and does not integrate political and institutional ideals.

There is a wide spectrum of possible hybrid financial structures based on the percentage of government funding, coupled with the percentage of revenue generating activities. In order for SSPP to be as successful as possible, It will be necessary to maximise the program's revenue generating activities.

Space to Earth solar power

Demonstration 1 (Budgetry requirement $80M)

The early SSPP demonstration projects are really technology validation exercises. These proof-of-concept demonstrations will show the feasibility of space beamed power and it's potential applications. The budgets for these projects will be very low, $80M targets, and will be funded from existing space agency budgets. These programs have significant science content and the use of space agency budgets can be justified on these grounds.

Demonstration 2 (Budgetry requirement $800M)

The second demonstration space to Earth solar power systems will build on the phase A demonstration systems by providing the consumer with a demonstration service as well as demonstrating the feasibility of larger systems. These systems should expand the market size for space solar power and provide a limited service of the market, by demonstrating the business mechanism of supply and demand.

Financial sources for each project will depend on the market addressed For all areas the financial sources identified for phase A are still valid. However, the larger budget requirements point to international cooporation as a means of distributing cost.

Other novel ways of obtaining limited sources of funds could be to tap institutions such as the European Environmental Agency (EEA) or the US Environmental Protection Agency (EPA) who would be interested in promoting clean power sources. These programs should be actively 'sold' to the public. This promotion could lead to limited sponsorship by companies intesested in promoting a 'green' image.

First Business applications (Budgetry requirement $2.3BN)

First commercial applications of space to Earth solar power will address and expand the markets generated by phase B of the program. There will be limited limited cash flow in this phase, but institutional support will still be required to implement space infrastructure unless the cost of launching space hardware has reduced dramatically.

First space to Earth applications will provide power on the scale of village level supply (1-10 MW). Sources of capitalfunds could come from a rights issue of shares in a young SSPP company. Interested parties could include existing space and aerospace companies, governments, public energy companies, and even existing private energy suppliers. Support could come from an energy futures market if established.

Income for power supply would be in two forms. The first would be the paying user, who pays for energy as they consume it. Other larger users of energy could provide capital support for the company while enjoying discounted, or free, use of space solar power. In this way, the ISPO company could utilize institutional budgets on an international basis.

A potential source of funds for space to Earth demonstrations will be world aid funds to developing countries. This can be justified if the countries supplying the aid build the spacecraft and supply power free of charge to developing countries. This option is attractive politically as it advances national technology programs and also furthers foreign and national policies. Other interested parties could be the UN and the IMF if the global environmental benefits of such space solar power systems can be 'sold' effectively.

Large scale applications (Budgetry requirements $15BN)

Large scale applications of space solar power will involve a satellite providing 500MW of power direct to an electricity grid for a period of fifteen years. This will provide a constant and predictable source of cash flow.

The space infrastructure will have to be provided by institutional funding as revenues from this venture will not be able to pay for this. Initially, the ground segment will have to be provided by the same funding source, but the revenues produced from the sale of power will lead to the privatisation of operation and hence the use of private funding sources.

Full Scale Power Delivery (Budgetry Requirement $23BN)

The next step in the evolution of the SSPP staged plan is the transition to 5GW space power stations providing constant power to the electricity grids. The revenues generated by such ventures are substantial and will present a major source of stable cash inflow to the company.

The funding for space infrastructure is large and will require either full or partial institutional financial backing. However the large cash inflows during operation allow more of the program to become commercially viable and hence able to use private sources of finance.

The hybrid funding structure of the ISPO organisation will move away from institutional backing to private sources of finance as hte operation expands and revenues continue to increase.

Space to Space Solar Power

Demonstration 1

The first demonstration of space to space beamed power is a technology demonstration exercise . This demonstration has a very low budget ($ 80M) and would be funded by any one of the major space agency budgets. International cooperation could be applied to show this approach to program management and build the future for further cooperation on larger scale projects.

Demonstration 2

The second space to space power beaming demonstration is to beam laser power to existing solar arrays in space. This is again a technology demonstration in the first instance, but a market for this kind of power supply has been identified so commercial benefits could be envisaged at this stage.

These early commercial benefits from space to space beamed power, for eample extending the life of commercial communications satellite, could provide a limited source of cash inflow. However these demonstration exercises will probably be supplied free of charge to cultivate this market area and generate a potential customer base.

First-Business Applications

First business applications of space to space beamed power use a custom satellite to beam several kilo watts of power to multiple satellites. This could generate significant cash inflows from this venture.

Initially a government supported venture such as the french SPOT company would be required to establish the required space infrastructure. This project could become commercially viable if potential customers can be convinced of the reliability and cost effectiveness of this method of power supply.

Mid-term Applications

Mid term applications of space to space solar power will use several custom satellites to povide several tens of kilo watts to users in all orbits and on the moon. As the market has been developed there will be significatnt cash inflows from this venture which should be commercially viable. This will allow all space infrastructure and operating costs to be met by private sector financing.

11.3.1 Financial Revenue Forecasts

This section of the SSPP report will consider the possible revenues that will be obtained during the implementation of the SSPP financial plan and show how these vary with the global cost of energy. The correlation between the system costs and revenues obtained, along with a simple sensitivity analysis, will also be presented. Results of space to space revenue forecasts are given in section 11.1.2.

The revenues obtained from any space beamed power program will depend on the price that this energy can be sold to an end user. As discussed in Chapter 2, the average current price of energy supplied to electricity grids today is:

$0.05 per kWh - Baseload

$0.22 per kWh - Peakload

The price of energy will dictate which supplies the energy companies will utilize for a given demand (time of day).

Due to the high capital costs of large space to earth solar power satellites, they should be utilized as fully as possible. This will mean supplying baseload power demand, which will generate maximum cash inflow to the project.

Demonstrations 1 and 2

The first and second space to earth space solar power demonstrations will be technology demonstration and validation missions. These will not be revenue generating ventures but will provide the business confidence needed for further investment in energy generation from space beamed power.

Any useful energy produced by demonstration 2 will be provided free of charge for other institutions to provide application demonstrations for uses of space beamed power, such as remote African villages for example.

Financial Viability Analysis for Different Sizes of Space Solar Power Satellites

The first financial analysis performed for different size space solar power satellites was to compare the required satellite output in millions of kWhrs to the grid vs. satellite size for the satellite system to become viable. This gives values for the satellite replacement costs in kWhr, and US dollars at current energy costs of $0.05 per kWh. Example results are shown in Tables 11.15 and 11.16 below.

Table 11.15 Example of Satellite Repleacement Cost for Viable Operation for 0% Return on Invested Capital

SPS power	1	500	5000	Megawatts
SPS power delivery hrs/day	23	23	23	hrs/day
SPS life	30	30	30	years
SPS yearly maint.ops cost	3	3	3	% of replacement cost
SPS yearly capital cost	3	3	3	% of invested capital
SPS total energy delivered	252	125925	1259250	Million kWh
Total costs to be paid:	3.33	3.33	3.33	In recurring SPSs
Recurring value of 1 SPS	76	37846	378464	Million kWh
SPS selling price	0.05	0.05	0.05	$/kWh
Value of 1 SPS at current $/kWh	0.004	1.892	18.923	billion U$'92

Table 11.16 Examples of Satellite Repleacement Cost for Viable Operation, 3% Return on Invested Capital

SPS power	1	500	5000	Megawatts
SPS power delivery hrs/day	23	23	23	hrs/day
SPS life	30	30	30	years
SPS yearly maint.ops cost	3	3	3	% of replacement cost
SPS yearly capital cost	0	0	0	% of invested capital
SPS total energy delivered	252	125925	1259250	Million kWh
Total costs to be paid:	1.90	1.90	1.90	In recurring SPSs
Recurring value of 1 SPS	133	66276	662763	Million kWh
SPS selling price	0.05	0.05	0.05	$/kWh
Value of 1 SPS at current $/kWh	0.007	3.314	33.138	billion U$'92

The main assumption associated with the scenarios presented in this analysis are :

1. 30 year satellite lifetime.
2. Power delivery will average 23 hours per day and all energy is sold.
3. Energy selling price is based on current estimates of $0.05 per kWh.

For our three larger scale steps of the ISPO operation we see that the satellite replacement costs for continuous operations must not exceed:

Table 11.17 Satellite Replacement Costs

	1MW	500MW	5000MW
3% ROI	$0.004 BN	$1.892 BN	$18.923 BN
0% ROI	$0.007 BN	$3.314 BN	$33.138 BN

These figures clearly show that for the one and 500MW systems operation will not be a profitable venture.

Figure 11.6 shows the viability of space solar power satellite operation based on $0.05 per kWh revenues. The assumptions used in this analysis are the same as above.

The figure shows that if the cost of satellite replacement falls into the shaded area then the power supply venture will be viable. The curve will move up with the increase in the cost of energy. Using this model it is possible to calculate the price at which energy must be sold to obtain a profitable venture, based on the replacement cost of each satellite.

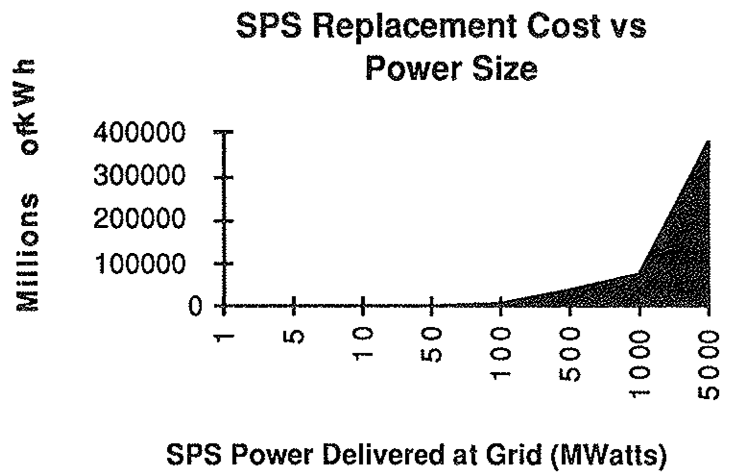

Figure 11.6 Viability of Space to Earth Solar Power Satellite Operation

The impact of this analysis will now be discussed with respect to the stage implementation of ISPO programs, introduced in section 11.2.3 of this report.

First Business Application

The first business application of space to earth beamed power will be step 3 of the SSPP staged plan. This will deliver 1MW of intermittent power to remote locations over a ten year lifetime. The cost of power delivery in these areas will be substantial so a high price can be charged for power delivery.

The utilization of world aid funds to developing countries to finance part of this project will mean that power delivery to some of these areas will be provided free of charge and will be used as a demonstration to expand the market for larger sources of power supply to other areas.

From the above analysis we see that the recurring cost of a multi-satellite system would have to produce 76 million kWhrs of energy to become viable. For a 1 MW system operated over 10 years the satellite will have to be operated for 7600 hours per year continuously. This will not be possible for this system which will supply remote locations and hence will not be financially viable. However this system will be needed to demonstrate the reliability of moderate scale power supply from space.

First large scale application

The first large scale application of space to earth power delivery will provide 500MW of continuous baseload power directly to the electricity grid for a period of 15 years. The revenue generated form this venture will be provided at three costs for power, to reflect the uncertain price of energy in the future. This system could generate total revenues of:

$ 16.425 BN @ $0.025 per kWh

$ 32.850 BN @ $0.05 per kWh

$ 65.700 BN @ $0.10 per kWh

The cost of this satellite would be of the order of $23 BN. However the development costs have been estimated to be of the order of $110 BN. The system could pay for it's own operation but as a true business venture it would be far from profitable. For a multi-satellite system the recurring cost of each satellite would have to drop to something approaching $3 BN, and the current state of technology does not allow for this.

This stage of the SSPP development plan would be important to show the production of space to earth solar power on a city scale and gain acceptance of this technology.
Full scale power delivery

The full scale power delivery system will use a constellation of satellites each generating 5 GW of baseload electrical power for direct connection to electricity grids. Each space power station has a lifetime of 30 years.

The recurring cost of these satellites has been calculated to be of the order of $30 BN at current technology levels. From the financial viability analysis we can see that these systems are the first that offer the potential for large scale profits, if the cost of capital for the program can be written off, i.e. 0% return on invested capital.

A financial analysis has been performed for this type of sytem. This is based on a NASA/Department of Environment (DOE) study conducted in 1978 which proposed a constellation of 120 5GW solar power satellites. A cost breakdown and sensitivity analysis for this program is shown in section 11.1.2. For this analysis the cost of satellite development has not been included. Results of the analysis and assumptions are presented below.

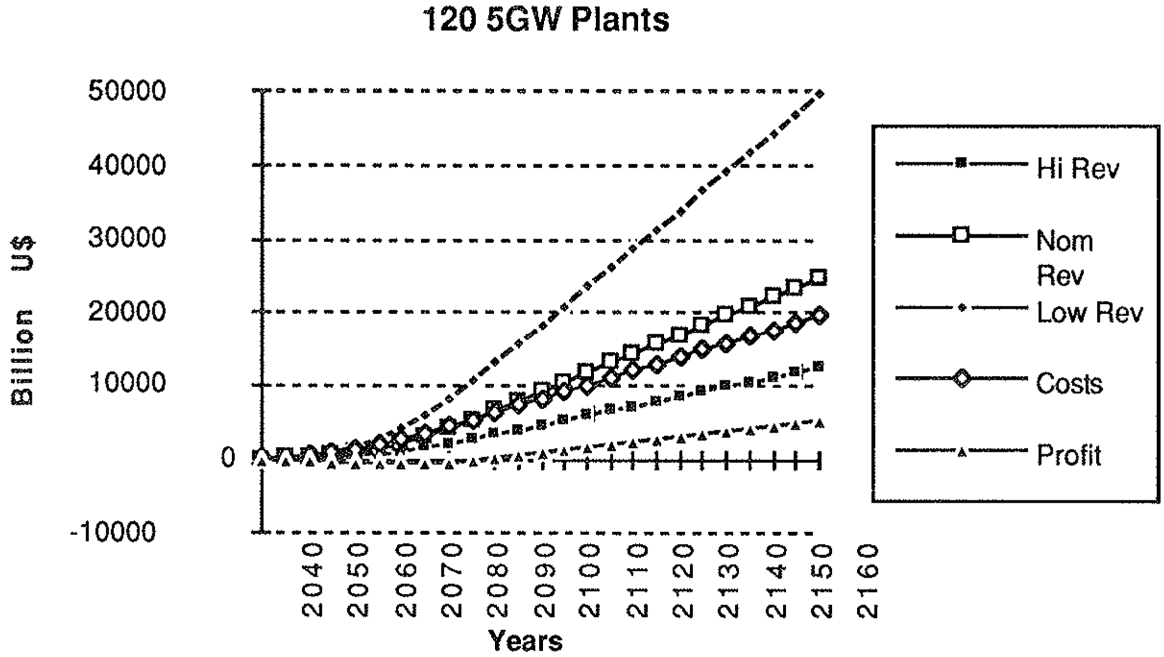

Figure 11.7 Revenues and Costs for 120 5GW SPS Operation

Figure 11.7 shows that for the nominal revenue case the project revenue will exceed project costs after about 40 years. The assumptions associated with this analysis are:

1. No interest is incurred on the cost of venture capital.
2. Satellites have a thirty year lifetime.
3. The cost of system maintenance and operations is $567 Million per year. This is based on the cost data presented in section 11.1.2 plus a 15% contingency.
4. The satellites are operated for 23 hours per day and all generated power is sold.
5. Hi Rev - Energy is sold at $0.10 per kWh

   Nom. Rev. - Energy is sold at $0.05 per kWh

   Low Rev. - Energy is sold at $0.025 per kWh
6. Recurring satellite costs are $ 30.7 BN. This is based on the cost data presented in section 11.1.2 plus a 15% contingency.
7. Satellite system development costs are not considered.

The most contentious issue associated with this analysis is the fact that the cost of capital is not included. This would be 3% for this sort of government venture, e.g. nuclear power station production.

The results of a financial analysis of 120 5GW power plants including a 3% return on cost of capitàl is shown below in Figure 11.8.

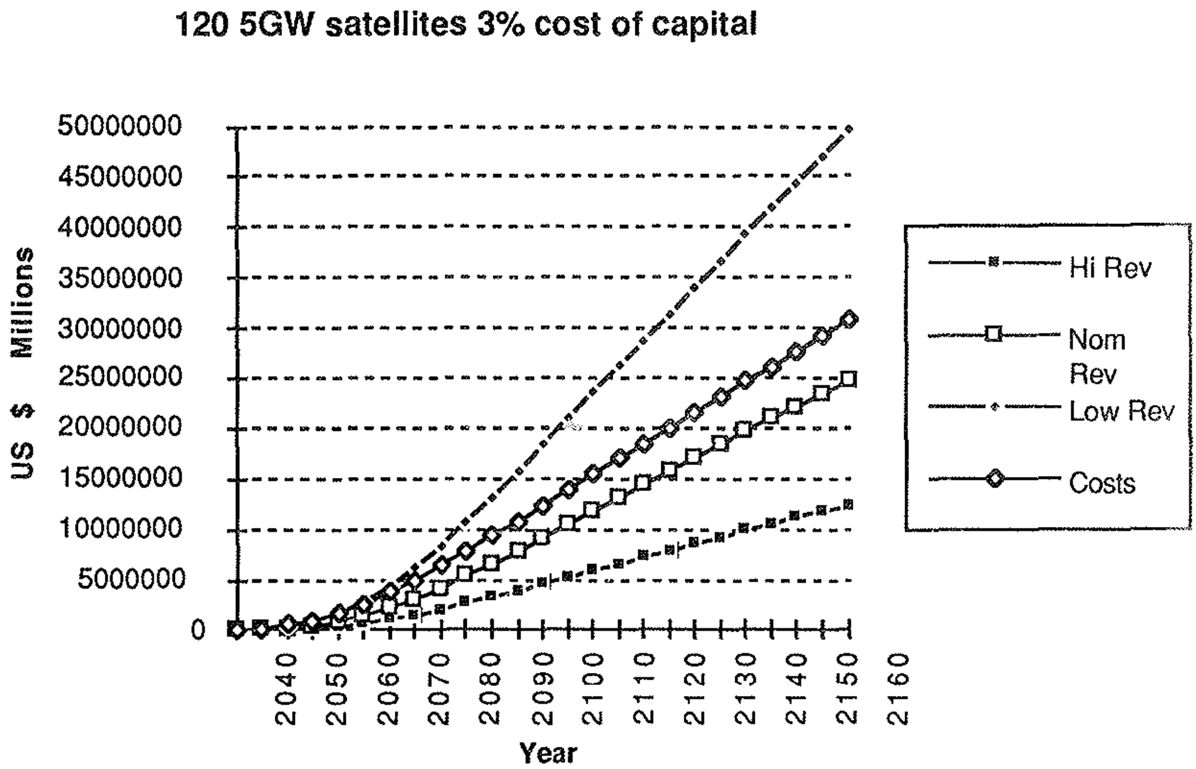

Figure 11.8 Revenues and Costs, including 3% Capital Cost, for 120 5 GW SPS Operation

This shows that the project revenues will only exceed costs if the cost of energy rises in the future, or the program costs can be reduced.

Chapter 2 of this report predicts the increased future demand for energy and it's cost in the future. Under these assumptions the energy prices will allow this space solar power system revenues to exceed current cost and thereby produce a profitable venture.

11.3.2 Conclusions

This section will present the conclusions obtained from the considerations of all financial aspects of ISPO. These will include conclusions about early commercial uses, program costs, and the funding of ISPO in terms of risk, sources, and financial viability.

Commercial aspects of beamed power supply for space applications

The short term there is a very small market for space to space solar power which renders this market non viable. Currently space platforms have the ability to supply all the energy needs of their payloads.

In the medium to long term (> 20 years) There is considerable potential for profitable ventures which supply space based energy demand. These opportunities will be crucially dependent on space power requirement growth. This is linked to the continued growth of space infrastructure and the price that can be charged for the supply of beamed power. An example of this potential growth is the US space exploration initiative. Lunar base power requirements alone could support a commercial beamed power venture.

Power supply could be anywhere form 500 kW to several MW. The price of this power would range from US \$ 400 per kWh to US \$ 50 per kWh respectively.

Cost sensitivity of ISPO programs

The main conclusions of the cost sensitivity analysis of ISPO programs are :

1. The most important cost factor is that of transportation of materials form earth to space. This accounts for more than 32% of total program costs. These transportation costs are very sensitive to the use of heavy launch vehicles which could seriously reduce the cost per kg launch into earth orbit.
2. The sensitivity of cost increases in the space and ground segments are roughly equal. The technical relarelation between these segments makes their total cost similar.
3. If cost changes are small (< 20%), then the effects on transportation, spacecraft, and ground segments are similar.

Financial risk analysis

The major financial risk areas that have been identified and will have to be managed effectively for successful ISPO programs are described below.

Market risks

The concept of space solar power supply for earth is not widely accepted. This will not be improved if the technology utilized does not show proven reliability at an acceptable cost.

To manage this, technology must be improved at competitive production costs.

Management risks

Steps must be taken to avoid delays in project completion and to avoid poor cost prediction. The large sums of venture capital involved will mean that only ten or twenty percent cost increases could seriously affect the projects financial viability and continued ability to raise funding.

These steps will be complicated by the complexities of international project management.

Political risks

A strong political commitment will be crucial to the success of ISPO due to the need for large international financing. Risk areas that could affect this commitment are:

1. Short term public fund allocation.
2. Delays introduced by the withdrawal of one partner.
3. Lack of technology transfer between countries.
4. The difficulties associated with obtaining public funds for space projects over short term terrestrial needs.

Environmental Risk

The environmental benefits of space solar power must be stressed to 'sell' the program effectively to the public. The risk areas that could affect this are:

1. Public acceptance of power beaming.
2. Pollution associated with the construction of space hardware and earth to space transportation.
3. Rectenna size and location.
4. Human safety while working in the space environment.
5. Environmental impact of large space structures which are visible from the earth.

Technical Risks

The main technical risks that need to be managed are those associated with program delays due to the development of new technologies and components.

All of the above risk areas will have to be managed carefully to ensure financial investor confidence in ISPO. These management issues need to be seriously addressed during the development of the ISPO, as its success will depend largely on it's ability to manage risk effectively.

Financial source utilization for ISPO development

A plan for the evolution of ISPO was presented in sections 11.2.3 and 11.2.4 of this report. This allows for a staged implementation of ISPO projects and effective use of financial sources an management of risk.

The first two demonstration programs will not produce financial revenue and will therefore require institutional support to finance them.

The 1 MW program will not generate substantial revenues compared to program cost but will show the business mechanism of supply and demand, while gaining public confidence in space to earth solar power supply. International institutional funds will have to utilized to finance this program.

The 500 MW power generation program will generate substantial revenues. These will allow ISPO operation to be paid for and will constitute the first business venture. However the cost of implementing the space infrastructure will still have to be met by international institutional finance. This step will be crucial in showing large scale beamed power applications, eg city level supply, and gain public and energy utility confidence for this form of energy suply.

The 5 GW power plants will provide revenues to pay for all operation and space infrastructure costs. The large amount of capital required for satellite development will have to be underwritten by international institutional finance, as the project will not break even for about 30 to 40 years, depending on the price for which energy can be sold. After this point large stable revenues are forecast, and operation will become highly profitable.

Financial Viability

The full financial viability of ISPO will only occur with the very large space power stations. This is due to the large revenues which must be generated to offset the high initial investment associated with establishing the required space infrastructure.

The cost of capital will be crucial to the viability of large ISPO programs. This cost of capital will relect the price at which energy is sold. This in turn will affect the utilization of the solar power spacecraft and the total amount of revenue generated.

It is estimated that a 3% ROI would provide a profitable venture if the cost of power increases to about US$ 0.07 per kWh and the initial satellite development costs are covered by government finance which can be written off.

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Appendix A

Summary of Proposed Design Examples

Early in the project a request for proposals for design examples was made. Twenty two proposals were submitted, and four of these evolved into the examples presented in Chapter 10. The other proposals and comments are briefly summarised below. In some cases similar ideas have been combined in a single entry

Laser/Space-to-Space

Using a laser tuned to near infre-red or optical frequencies, it would be possible to use a satellite to provide power to existing satellites near the end of their lifetime by beaming power onto their solar arrays.

Power for μ-gravity platform

Two small satellites, to be carried by Ariane, are launched. The mother satellite has solar arrays and a transmitter and the daughter a rectenna.. Energy beamed from the mother an be used to power microgravity experiments on the daughter. The rectenna is jetisoned before re-entry.

Peak power for Earth

To have a Solar Power Satellite in such an orbit (sun synchronous) that you selectively deliver peak-power to ground, because you can sell that for a higher price.

SPS constellation

Constellation of small SPS in LEO (rather than a few big SPS in GEO).

Deployable (Inflatable/Rigidizable)

Gravitationally stabilised inflatable platform. Thin film amorphous silicon photovoltaic arrays on the surface of the inflatable. The size of the satellite is easily scaled up for increased power.

Power beaming for Space Transportation

A satellite beaming energy to vehicles for use in powering electrical propulsion systems. A 10-100kW demo could be carried out for $800M

$800 M/Deployed/SPS 2000-class

Similar concept to SPS2000, but in sun synchronous polar orbit at 1000-1500km. Beaming 10-15kW to earth, probably to power experiments at the poles. Could be used as the first satellite in a constellation.

Space(Space Shattle)-to-Space

Power beamed from the Space Shuttle cargo bay to a dead satellite by laser.

Ground-to-Space-to-Space

A mother satellite to relay power transmitted from a ground antenna to a daughter satellite.

Antarctic Power Satellite Program (Space to Earth Demonstration)

Polar orbiting satellite to provide power to Antarctic bases. Possibly in Molniya orbit to maximise duty cycle.

1MW class SPS

Low earth orbit photovoltaic array beaming 1MW to earth. Launchable on a single Energiya launcher.

SPS for Peak power market

Platform in a high inclination, nearly polar orbit with high eccentricity deliver power to the recieving station. A non-eclipsing orbit would simplify design considerations.

Lunar SPS application

Lunar orbiting satellites relaying power from the illuminated side of the moon to observatories the dark side. The satellites have no large solar arrays in order to prevent disruption of astronomical observetions made on the surface.

Lunar Resources

Any large scale space power satellite should exploit lunar resources to the full.

Space Transportation Demo for SSPP transportation cost reduction

Before any large scale power satellite can be launched from earth, cheap transportation is essential. There are several candidates which could be developed.

Power Transmission Demonstration for satellites in GEO

Positioning a power satellite near geosynchronous orbit it could power satellites in eclipse by beaming a visible laser on their solar panels. Cost target $100M to $200M.

Observing satellite and high-altitude balloon

The effects on the atmosphere of power beamed from a satellite in low earth orbit could be examined eith by a companion satellite or by a high-altitude balloon.

Microwave power beaming using small airplane

To place a small photovoltaic array on a high-altitude airplane and beam the energy to the ground in the form of microwaves.

SPS for the scientific rover on the Lunar surface

To provide laser-beamed power from an space power satellite in lunar orbit to a scientific rover on the lunar surface. More specifically, it might be possible to provide power for one of the two anticipated rovers on the NASA Artemis mission, scheduled for 1997.

APPENDIX B

LUNAR ROVER

DESIGN EXAMPLE PROPOSAL

Demo: To provide laser-beamed power from an SPS in lunar orbit to a scientific rover on the lunar surface. More specifically, might be possible to provide power for one of the two anticipated rovers of the NASA Artemis mission, scheduled for launch in 1997.

Purpose: For SSPP, to demonstrate power beaming over large distances and to remote areas; also, another purpose would be to support a mission that may help to establish a lunar base, the resources from which could then be used to facilitate large-scale production of SSPP. For Artemis, to provide power for up to one full year, as opposed to current estimates of a power-limited rover lifetime of 1 or 2 lunar days. For other, larger rovers, the purpose may be to provide continuous power at lower weight for the rover and also to provide power to rovers operating in permanently shadowed craters near the poles.

Target Cost: $500 Million for Artemis, $1-2 Billion for larger rovers.

Timescale: 5 years for Artemis, 10 years for larger rovers.

Power Level: An Artemis rover is only planned to be about 30kg in size and would only need a few hundred watts of power, larger rovers may require a kilowatt or more. The satellite would probably have to transmit at least 2-3 times the power needed by the rover.

Orbit: We've come up with two scenarios: place the satellite at L_1, and restrict the rover to the near side, or place the satellite in low lunar orbit (at about 300 km), and have the rover zigzag across the projected path of the satellite. The former scenario allows for continuous beaming, while the latter requires the rover to be able to use bursts of laser energy to recharge its batteries for up to two hours or so.

Organization: The Artemis option would obviously have to be conducted with NASA, powering other rovers could be done through other agencies.

Appendix C

LEO Constellation of Small SPS

This section is intended to study technical and economical aspects of a LEO constellation of small solar power satellites (10 to 100 MW) as an alternative to provide a continuous and reliable source of energy to the Earth. Obviously, this concept is mainly suitable for a long term commercially oriented project. It could not be easily applied to a small scale profitable venture because such a constellation requires a complete fleet of satellites to feed many receiving sites (rectennas).

We should notice than this concept is in opposition to the "classic" SPS design, which usually consists of a big satellite (5 GW) in GEO and one rectenna on the ground. We will try to outline the advantages of the LEO constellation over the GEO approach, but we will also discuss the drawbacks of this option.

Analysis of a simple constellation (coplanar case)

We will do a quick analysis of the simplest case, a constellation of satellites on coplanar and circular orbits. This is the case, for instance, of the equatorial orbits. We will evaluate technical characteristics of the power transmission infrastructure required (antenna and rectenna) for many orbit altitudes. We will try to explain why a LEO constellation can be a good solution to ensure the economical viability of a long term SPS development.

The first factor which comes to mind is the amount of illumination of the spacecraft by the Sun. It is well known that high altitude orbits allow larger duration in sunlight. As shown in figure C.1, this visibility increases very rapidly with the altitude. But we observe a significant gap between LEO (65-75%) and GEO (96%).

We must also consider the total visibility of the rectenna from the spacecraft. Obviously, it will vary significantly with the altitude. GEO are chosen for many space applications because they allow a permanent link with the ground. On the other hand, LEO suffer from their very short visibility time. To illustrate this, we have plotted in figures C.2 and C.3 the total visibility angles and times for different altitudes. As these factors are strongly dependent on the maximum admissible elevation angle of the rectennas, we have evaluated 4 cases where the elevation angles range from 15° to 60° (in increments of 15° between each curve).

In figure C.2, the total visibility angle is a measure of the orbital arc where the spacecraft is able to beam power to one rectenna. These arcs are very small in LEO (10° to 50°), but they grow rapidly as we get closer to GEO (continuous visibility between the spacecraft and the receiving site). Similarly, the total visibility time (figure C.3) is very short for LEO (only a few minutes).

If we look at these curves, a LEO does not seem interesting since only a small amount of the total power produced by the spacecraft could be transmitted on a single rectenna (if no energy storage device is used aboard the satellite). But as we will see further, we can overcome this negative element by multiplying the rectennas along the trace of the orbit. Then we could collect all the energy beamed by the satellite at each revolution (but with reduced illumination of the spacecraft by the Sun as compared to higher orbits).

However, if we want to use many rectennas for an entire coverage of the orbit, we must be aware of some physical limitations such as the maximum elevation angle between the beam and the rectenna and also the deflection of the beam from the antenna (figures C.4, C.5 and C.6). Actual technical limitations are roughly 60° for the elevation angle and 30° for deflection (these parameters depend on the technology used, and they will certainly improve in the future). We can see that these constraints are almost impossible to meet in LEO with less than 10 rectennas.

The construction cost of the rectennas is a major part of total cost of the project and it needs to be carefully optimized in relation with the cost of the other main element of the infrastructure, i.e. the spacecraft. We do not possess enough information about the real costs of these elements and we will not be able to do a realistic and precise estimation. But we can assume, for instance, that for the rectennas, the construction cost will be nearly proportional to the total area. As we know how to estimate the area of the rectenna required to collect the power from a satellite at a given altitude, we can sum up the total receiving area needed for a entire coverage of the orbit vs altitude (figure C.7; figure C.8 is focusing on LEO).

In these figures, the equatorial orbit is covered by a complete network ranging from 5 to 30 rectennas (for each curve, the total number of rectennas is incremented by 5). For the calculations, we have chosen a reference antenna of 1 km (diameter), and a frequency of 2.45 GHz. To compare these results with other values, we just need to multiply the y-axis by the corresponding factor. For our rough estimations, we have use this well known approximate relation (with elevation and deflection corrections):

$$D_R = \frac{1}{\cos\beta.\cos\delta} \cdot \frac{2.44\,.\,\lambda}{D_A} \,.\, L$$

Then, we can estimate the rectenna area (assuming an elliptical shape):

$$A_R = \frac{\pi}{4} \cdot \frac{1}{\cos\beta.\cos^2\delta} \cdot \left(\frac{2.44\,.\,\lambda}{D_A}\right)^2 . L^2$$

According to these figures, the minimum total rectenna area (7 km^2) occurs with a network of 30 rectennas and an orbit of 900 km altitude. For comparison, the same spacecraft in GEO require 90 km^2 of rectenna. However, we must notice that many networks (10 to 25 rectennas, 1000 to 2200 km of altitude) offer similar small rectenna areas (8 up to 21 km^2). The total area grows rapidly with altitude, and this factor can be a major driver if the rectennas construction costs prove to be a main contribution to the total project cost.

Similarly, if the antenna cost is the major driver, we could compare antenna diameters required at each altitude to feed a network of rectennas with similar total area. For instance, if we use a network of 10 rectennas with a total area equal to the area required by the GEO option, and a circular orbit 2200 km high, our antenna will be 4 times smaller than for GEO ($D_A = 0.48\ D_{GEO}$ and $A_A = 0.23\ A_{GEO}$). This means a significant saving on construction costs. We should analyse and optimize all the parameters (antenna, rectenna, power) together, but in this case also, LEO proves to be very interesting.

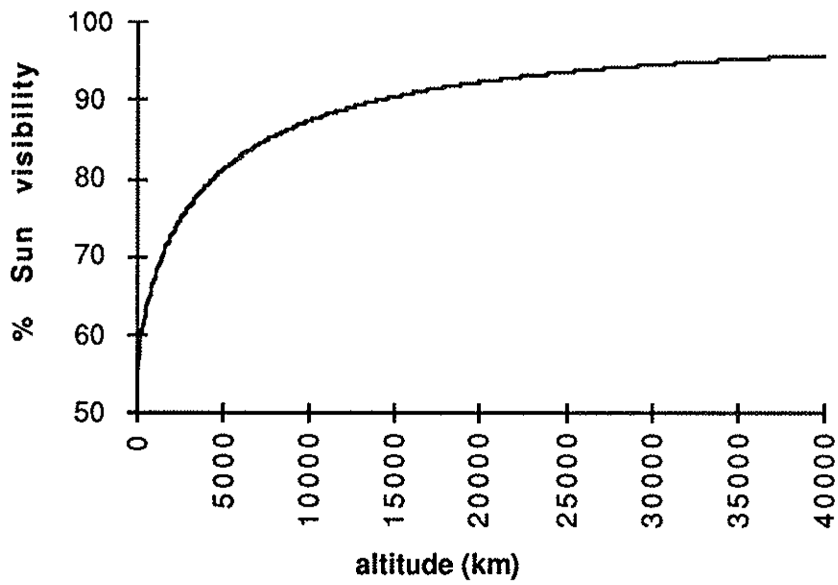


Figure C.1. Total illumination of the SPS by the Sun (per orbit)

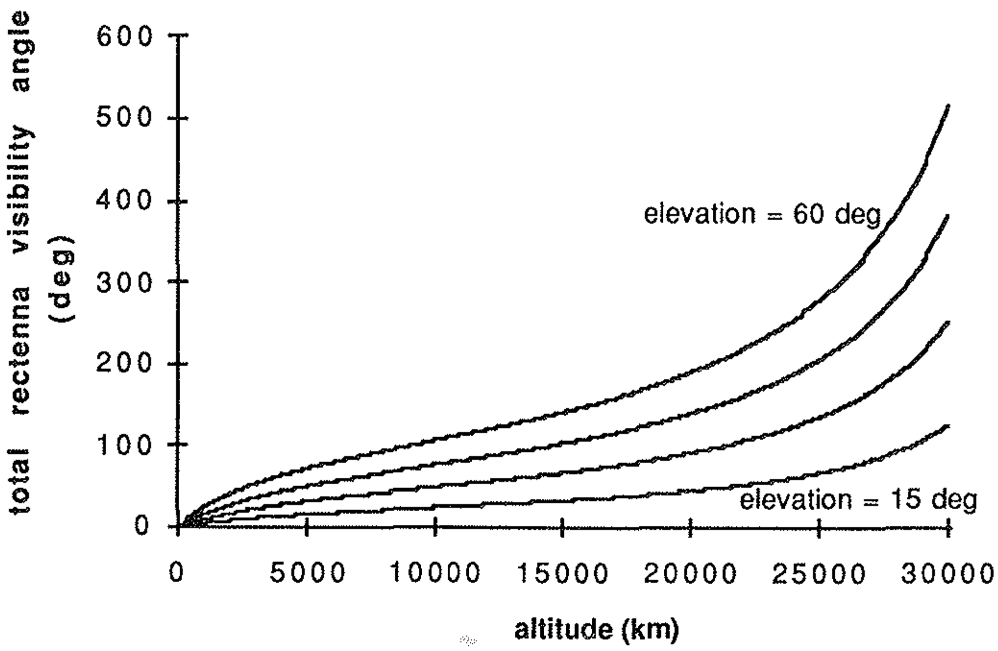


Figure C.2. SPS - rectenna visibility angle

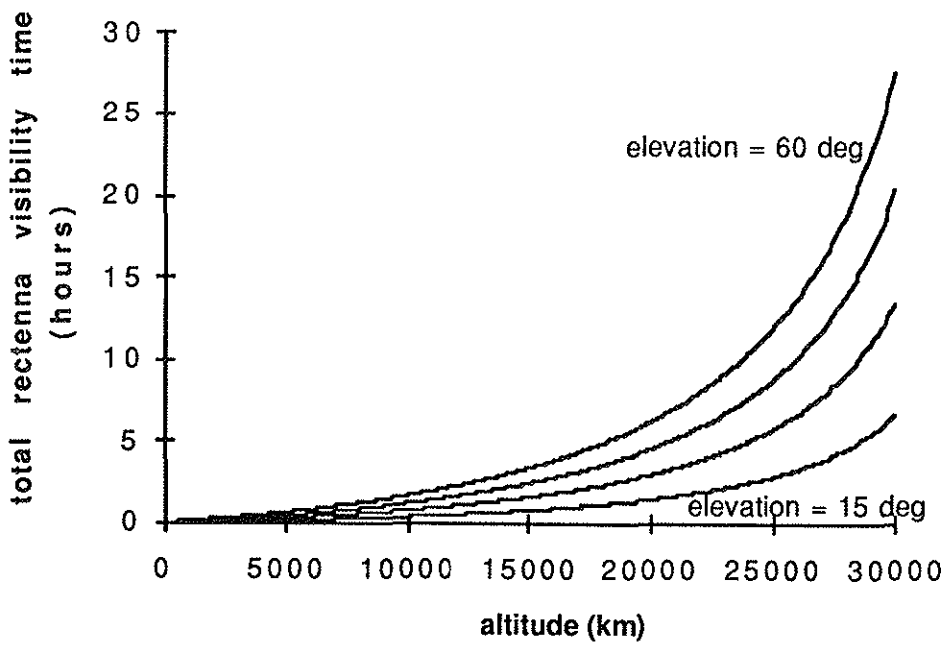


Figure C.3. SPS - rectenna visibility time

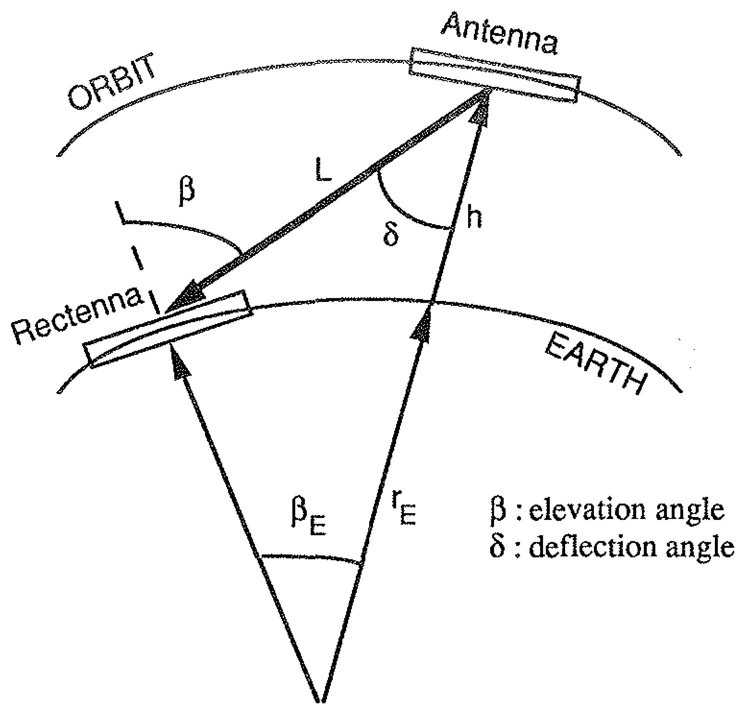


Figure C.4. Geometric relations between the antenna and the rectenna

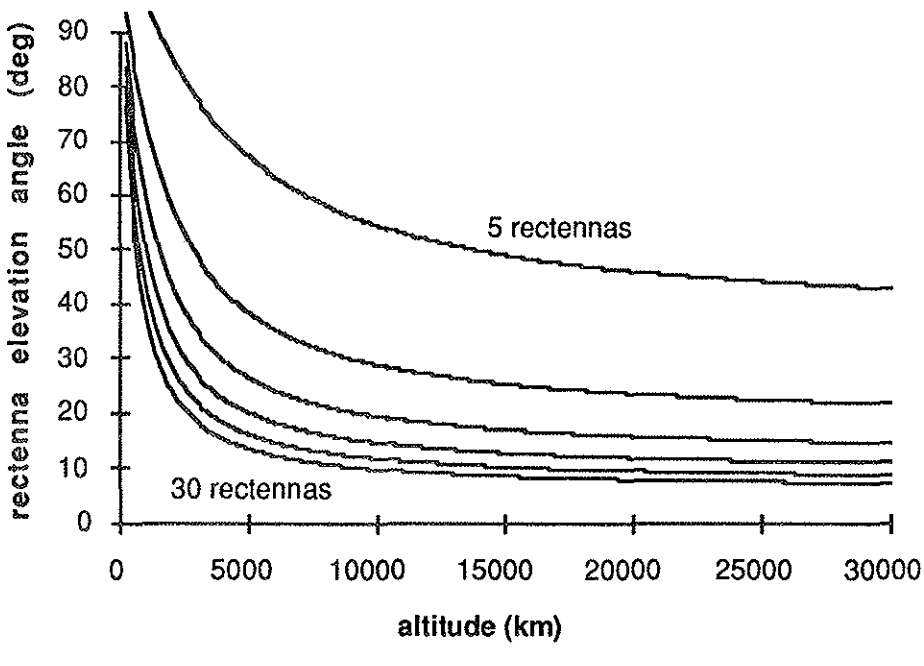


Figure C.5. Maximum rectenna elevation angle

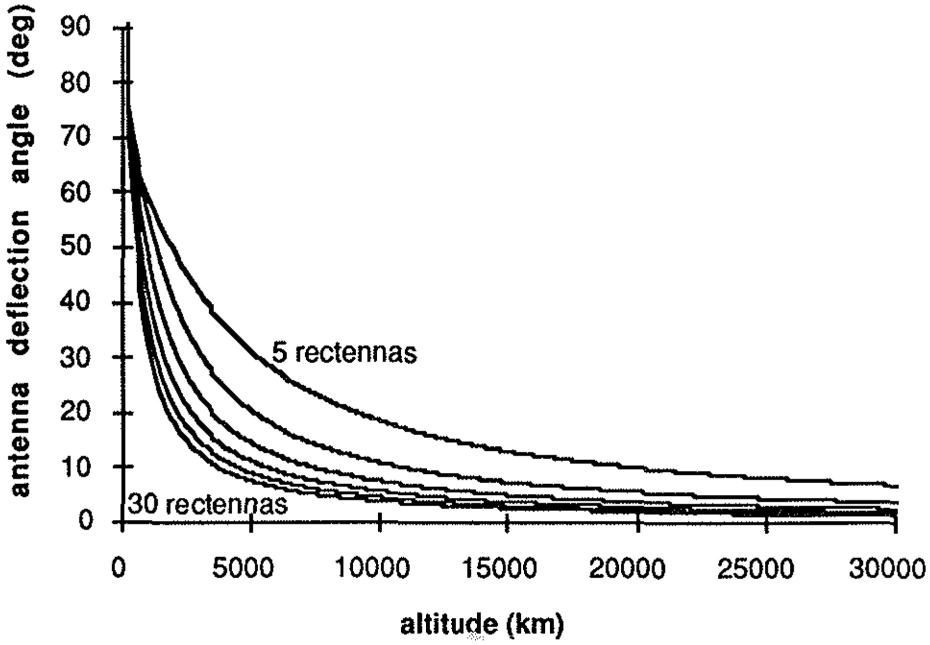


Figure C.6. Maximum antenna deflection angle

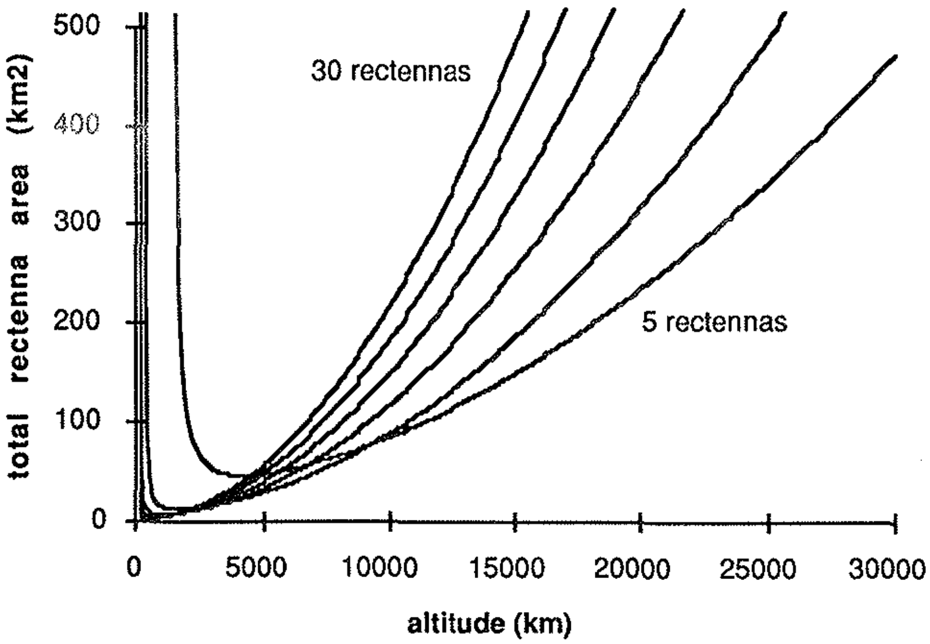


Figure C.7. Total rectenna area

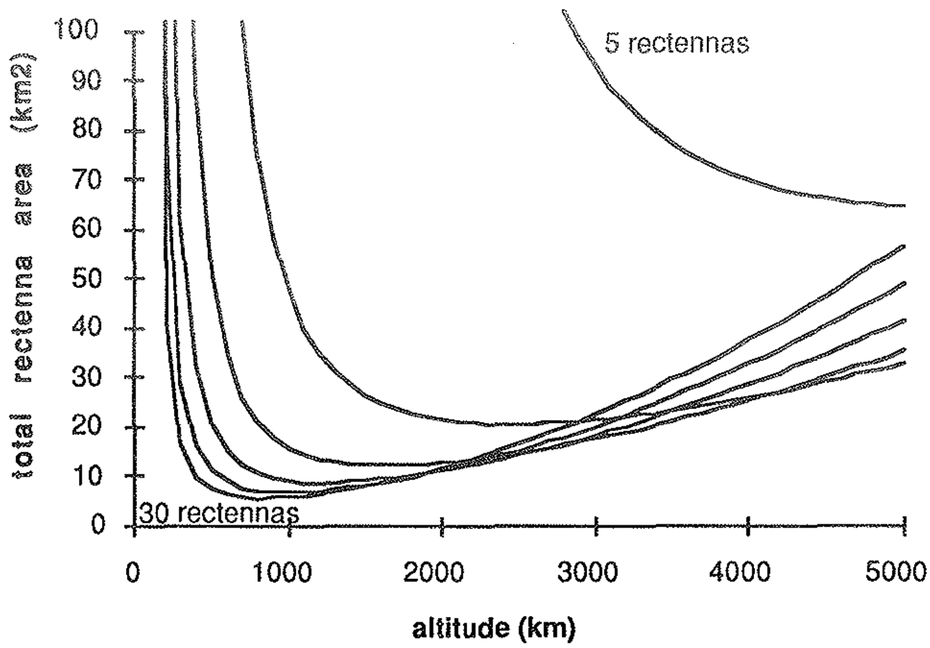


Figure C.8. Total rectenna area (LEO)

Factors driving the choice of the orbit

We will try to summarize the advantages and the drawbacks of a LEO constellation, per comparison mainly to the big GEO solar power satellite design:

Advantages

1- Design benefits:

- Smaller and lighter structures than for GEO SPS (moreover, we do not need fuel for a GEO transfer).
- We need fewer launches to bring one small SPS in LEO orbit (but it is also a trade-off with the number of spacecraft to launch).
- We need also less EVA and/or robotic operations in orbit (and even none, if we can ensure completely autonomous deployment in space).
- Therefore, a higher percentage of on-earth operations and verifications greatly improves reliability and cost.
- A constellation is more tolerant to failures (if one SPS breaks down, it will not affect too much the total power output of that constellation).
- In LEO, we can use passive gravity gradient stabilization of the spacecraft.
- We do not have the interference problems of GEO, and we can avoid slot allocation committees and politics.
- With the same antenna and microwave frequency, the receiving rectennas are smaller if the spacecraft is in LEO (if the power transmitted is similar, the average power density will be higher and we will be able to ensure more easily densities above the rectenna's threshold).

2- Commercial benefits:

- Shorter construction delays for small SPS and small rectennas.
- The power is available quicker from space, therefore early sales of electricity generates funds.
- Early demonstration of the validity of this commercial concept.
- A rapid success of the firsts phases makes it easier to raise funds to continue the project.

- Smaller capital and investments required to start the program. A complete ground and space infrastructure is not needed to produce electricity.
- Many identical and redundant elements of this infrastructure allow mass production cost reductions.
- The feeding of many rectennas enables the organization responsible for commercialization of SPS electricity to diversify and stabilize its revenues.
- A complete and well balanced geographical distribution of rectennas may allow shorter distances between the receiving sites and major consumer areas, and therefore, cheaper electricity transportation and distribution costs (this is mainly true for a high inclination constellation, the non coplanar case, where satellites fly over or near industrialized countries).

3- Adaptability:

- This concept enables technology improvements to be gradually integrated to the infrastructure over the years.
- Eventually, it can accommodate many design of SPS, as long as the power transmission subsystems are compatibles.
- And it is fully adaptable to the growth of electricity demand.

Disadvantages

1- Spacecraft:

- As we use LEO, atmospheric drag losses are not negligible, especially for large structures.
- Total illumination of the spacecraft by the Sun is significantly reduced in LEO compared to GEO.
- The selection of the best orbit can be constrained by space debris or Van-Allen belt considerations.
- This constellation will increase the number of spacecrafts in LEO orbit (and eventually, the density of space debris).
- In LEO, rectennas are in the visibility of the SPS only for a few minutes and the antennas need very good tracking and/or deflection capabilities.

2- Rectenna:

- With a constellation, we cannot always assure a continuous and stable feeding of all the rectennas (this can cause problems for basic energy supply, unless we use storage devices associated with the rectennas).
- The rectennas must be able to collect efficiently the microwave power from many directions.
- Because of the important number of receiving sites required by a LEO constellation of SPS, it could be difficult to meet the ideal geographical distribution since a large part of the Earth surface is not suitable for rectennas (oceans, mountains, etc.).

Conclusion

Many factors need to be taken into account when making orbit selection. We must optimize technical as well as economical parameters. The scope of this small study does not pretend to give the final and definitive answer to that question. Too many elements are missing, or are simply unknown. But, a constellation of SPS in equatorial LEO seems to be an attractive solution. Obviously, some technical problems, like the tracking of the beam, have to be solved. It should also be interesting to evaluate the more general problem of a 3D constellation (28.5° or polar orbits, for instance) to provide energy all over the world.

APPENDIX D

Atmospheric Tester

Design Example Proposal

Demo: To place a Solar Power Satellite in low Earth orbit with the capability to alter its beaming angle and power density. Two types of instruments would be used to monitor the effects of these variations on the atmosphere: a companion low Earth orbit atmosphere observing satellite and high-altitude balloons.

Purpose: For Space Solar Power, to determine the effects of a solar power satellite on the atmosphere, as well as of the atmosphere on the power beamed by a solar power satellite. For scientists, to learn more about the upper atmosphere and magnetosphere by using either lasers or microwaves to induce experiments and reactions at high altitudes.

Target Cost: $800 Milllion.

Timescale: 10 years.

Power Level: The amount of power-- as well as its type (microwave or laser)-- would strongly depend on various factors, such as the ability to alter the beaming angle, the likelihood of the power type's inclusion into the large-scale space solar power program, and the ability to induce reactions in the atmosphere of scientific interest.

Orbit: Low Earth Orbit.

Organization: Any space agency that has previously run Earth observing satellites (or a combination of agencies) should suffice.

Note: Costs might be significantly defrayed by placing the SPS near an Earth observing satellite already in orbit.

Appendix E

Feasibility Study of Laser Technology in the Space to Space Demonstration

As part of an early demonstration program it was opted to work out a concept using laser technology in space to space power beaming. The work presented is a summary of the work performed by a small task group.

1 Mission objectives

The mission objectives can be summarized as follows:

(1) Space to space power beaming
(2) Laser beaming to solar arrays of a receiving satellite
(3) Demonstrate a commercial potential or benefit
(4) Should be launched within 5 years
(5) Total cost $ 80 M

2 Assumptions

Assumptions	Justification
1. Use an **existing receiver satellite**, thus only one satellite (transmitter) has to be built/launched	- due to cost constraints
2. Launch a satellite into **GSO** to deliver power to a **graveyard satellite**	- distance between XMTR and RCVR must be within 500 km (TBC) - low relative drift velocity - graveyard satellites have functioning electrical systems - no interference with an existing, operational GSO satellite
3. Target satellite should be a **spinner**	- for conditions of pointing, attitude control and solar array illumination - spinning satellites tend to have a lower power demand, therefore the impact of beaming power is larger
4. Received beam should be 10% of target satellite's power	- power beaming should lead to a measurable amount of power on the satellite

3 Requirements

The laser has to emit in the visible light spectrum in order to be compatible with the existing solar arrays of the target satellite. This requires a beam with a wavelength of **about 580 nm** [S. Bailey lecture for Space Solar Power Project, Aug. 3, 1992)]

4 Mission Assessment

The concept presented serves as a dimensioning example/tool in order to obtain some representative data on a possible mission.

Laser power demand

To show a measurable effect on the target satellite, 10% of its solar array power should be generated by laser beaming. Therefore the flux power density on the target satellite due to the laser beam should be about 130 W/m^2 (1/10 sun solar power).

To cover the full solar array of the satellite the diameter of the laser SPOT should be about 5 m which is equivalent to about 20 m^2. Thus the total laser *output* power would have to be 2.6 kW.

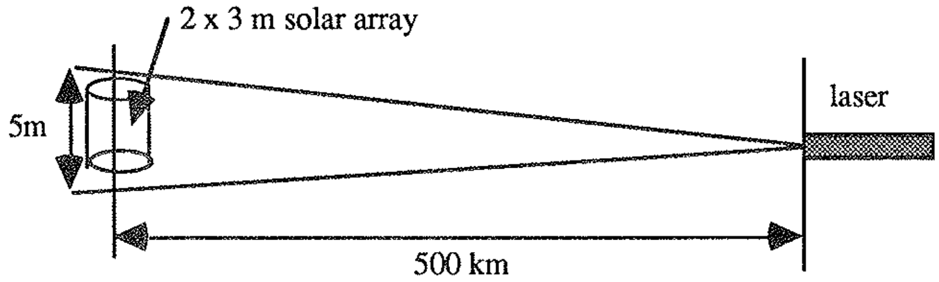


Figure 1 Beaming principle

Spacecraft subsystem dimensioning

Solar arrays:

Current laser technology has an input power to beamed power efficiency of less than 15% (in infrared). So the spacecraft would have to provide at least **17 kW** to the laser payload. The generation of this power requires a solar array area of 126 m^2 for the laser payload only (assuming 10% overall conversion efficiency for the solar arrays). Assuming that the mass to power ratio of the solar panels is 30 W/kg (TDRS satellite), the mass of the solar arrays becomes about 570 kg.

Laser:

The total required laser output power was found to be at least 2.6 kW. Unfortunately there are no commercially available high power lasers in the visible wavelength range. The development of a dedicated high power laser system for short term demonstration seems unlikely. Since gas lasers operating in the visible light range can not be scaled up to the required power level, potential lasers will have to be of the solid state type. A solid state visible light laser system for high power output will lead to significant cooling problems and therefore be quite large and heavy (some hundreds of kilograms).

Launch cost analysis

Due to cost constraints it was suggested to use ASAP (ARIANE Structure for Auxiliary Payloads) to carry the laser experiment. The laser equipment is considered to be the payload of a satellite. As a general rule of thumb, the payload mass is usually about 20% of the overall spacecraft mass. Assuming a laser system weighing 400 kg the resulting overall spacecraft mass totals 2 tons. ASAP is a structure to carry micro satellites with a maximum weight of 50 kg each. Therefore the proposed launch scenario with ASAP is not feasible. Considering a conventional prime passenger launch with 70000/kg to GEO the total launch cost is $140M!

Pointing accuracy and beam locking

To keep the laser beam pointed on the satellite over a distance of 500 km would require a pointing accuracy of better than 1 arc second. No previous experience exists on accurate beam pointing. The first problem is to precisely locate the target satellite in a graveyard orbit without a pilot beam. The use of the target satellite's RF signals does not provide the necessary accuracy of better than 1 arc second. It is not clear whether optical systems could perform this task.

Spacecraft concept

The VIKING platform is proposed as a platform for the laser experiment. The VIKING has the following advantages:

- it is an existing platform
- it is cheap
- it can be launched together with the French SPOT satellite. The combination of VIKING and SPOT has already flown before and is therefore qualified

However, after some evaluations it was found that the proposed VIKING platform is incompatible with this type of mission. As a spinner, this platform is designed to be oriented inertially. To point a laser beam this design would have to be changed completely in order to have a 3-axis stabilized spacecraft or a dual spin concept.

5 Conclusions

Most of the parameters identified in the technical discussion exceed significantly the given envelopes in terms of cost and schedule:

- the mission would require significant payload developments for a high power laser in the visible spectrum (time, money, risk)
- the technology for high precision pointing would have to be developed
- total spacecraft mass would be in the range of a big telecommunication satellite

Possibilities to improve feasibility	Comment
- scale down the experiment by two orders of magnitude to be able to use existing lasers in the visible spectrum	- this small amount of power would have no measurable effect on a target satellite
- launch directly into graveyard orbit	- shorter beaming distance to relax the pointing and aperture requirements
- decrease the beam diameter on the target to decrease the laser power needs	- higher pointing requirements for the emitter spacecraft - beam scattering

Despite these modifications it seems to be impossible to carry out a reasonable demonstration experiment with this small amount of money.

See also M. L. Gaillard: *Open questions in high power laser transmission systems* , proceedings of the Solar Power Satellites conference, C.3., p 95 ff, Gif sur Yvette, France, June 5/6, **1986.**
Team members in alphabetical order: R. Bertrand, M. Fahey, R. Häberli, D. Jacobs, P. Kumara, J. Laaksonen, A. Suleman, J. Vidqvist

Appendix F

The ASAP / Viking Near Term Demonstration

Purpose of "ad hoc" task group

The intention was to produce early information to other groups of what is feasible within 5 years and 80 M$. This group -coordinated by Dieter Kassing- the exchange of ideas of such possible missions. This task would logically be performed in a later phase of the overall development plan e. Never the less this task was felt important to do early, in order to save time later.

Given Mission Goals

- Demonstrate power beaming, space to space, space to ground, or both.
- Demonstrate the technical feasibility of power beaming in / from space within 5 years.

Selection Criteria

The Selected mission
- should form a milestone in an SPS exploration program
- should not be covered by any other space project
- should be realized within 5-8 years
- should cost less than 80 M$
- should, if possible, reflect the international character of ISU
- should not develop new technologies which are not essential to the prime mission , (instead should use existing hardware were possible.)
- should have a value in its own, i.g. commercial or scientific. (independent from later SPS implementations)

Proposed missions discussed by the "Ad hoc" group

1 Satellite beaming power to polar station.
2 Satellite beaming power from shuttle solar array to micro gravity experiment satellite.
3 Satellite beaming power to two earth observation scientific satellites.
4 Micro gravity capsule beaming power to other micro gravity capsule.
5 Satellite beaming power to SS Freedom.
6 Ground station beaming power to airplane carrying communication relay station.
7 Viking Satellite demonstrating power beam control by beaming to ASAP receiver.
8 Satellite beaming power to astronomy satellite up to thousand AU away.
9 Satellite beaming power to telecom satellite with ion engine in GEO for station keeping .

Selected mission

Based on the above given goals, selected criteria and the suggested missions, the group selected a combination of missions 1 and 7. The selected mission statement is :

" To demonstrate microwave power beaming in one mission both, space to space (1 km distance) and space to earth (LEO to polar cap)."

Mission description

The suggested mission #1 was to have a power receiving infrastructure in Antarctica. The reasons being, their is an existing scientific oriented community needing power. Therefore the satellite's orbit is polar. The selected mission could fly as piggyback on the Spot 4 mission in 1996/97 . On the Spot 1 the Viking scientific satellite was a piggyback.

The idea is to again use the Swedish Viking satellite as a platform to mount the beaming experiment on. The beam would first be directed in space 1 km away to a rectenna mounted on the Ariane third stage Ariane Structure for Auxiliary Payloads (ASAP) . The solar panels will be body mounted on the Viking bus. A preliminary drawing of the concept is given below.

The Viking platform as well as the Ariane third stage need to be in the same orbit. Both the Viking satellite and the Ariane third stage are spin stabilized. Because of the changing nature of the orientation of the spinning satellites with respect to each other, the rotation axis of both satellites have to be aligned at the moment of power beaming. This implies that the beaming can only be performed for a short duration, once per orbit.

After the space to space experiment the beam could be pointed to earth. This means that at the moment of power beaming the satellite's rotational axis must be perpendicular to the equatorial plane. This should happen at the south pole (rectenna site). In this case the power beaming can also be performed for a limited duration.

A initial calculation was made to look at the order of magnitude of power beaming possible. This preliminary example shows that about 250 W are available for power beaming from the Viking spacecraft. Given a 1 km distance, 35 GHz frequency and the antenna diameter of 1.2 m (Viking) and 4 meters (ASAP) the efficiency is 20%. The resulting maximum received power is approximately 50 watts. Due to power conversion the actual received power would be 10 watts. For the space to ground the received power is unclear.

The main purpose is to make a technology demonstration of beaming efficiency at different angles and distances over a time of some months at a power level substantially above previous experiments. Another technical issue in pointing accuracy. Objectives of second order priority are scientific issues. An example of on such issue is the 35 GHz influence on upper atmosphere

The Viking platform

The size of the Viking satellite bus is 2 meter octagonal and has a height of 1 meter. The mass is 550 kg, with a possibility of being increased. The load carrying structure is a 1.2 meter aluminum cylinder 6.1 mm thick and 0.525 meters high.

The ASAP

ASAP is a ring with an internal diameter of 2060 mm and external diameter 2900 mm. The maximum total mass available is 200 kg placed at 4 to 6 positions. Maximum dimension of the equipment's placed on the ring is 450x450x450 mm (can be 700 mm high by waiver). It is electrically connected to the Vehicle Equipment Bay (OBC).

The mission costs

An example of the first cost break-down is as follows :

Launch	10 M$
Viking platform	20 M$
ASAP rectenna	5 M$
Experiment	10 M$
Ground segment	3 M$
Operations 1 year	5 M$
Total	≈ 50-60 M$

Acknowledgements

During the work of this student working group two persons contributed especially.
Thank you Dieter Kassing of ESA and Sven Grahn of Swedish Space Corporation.

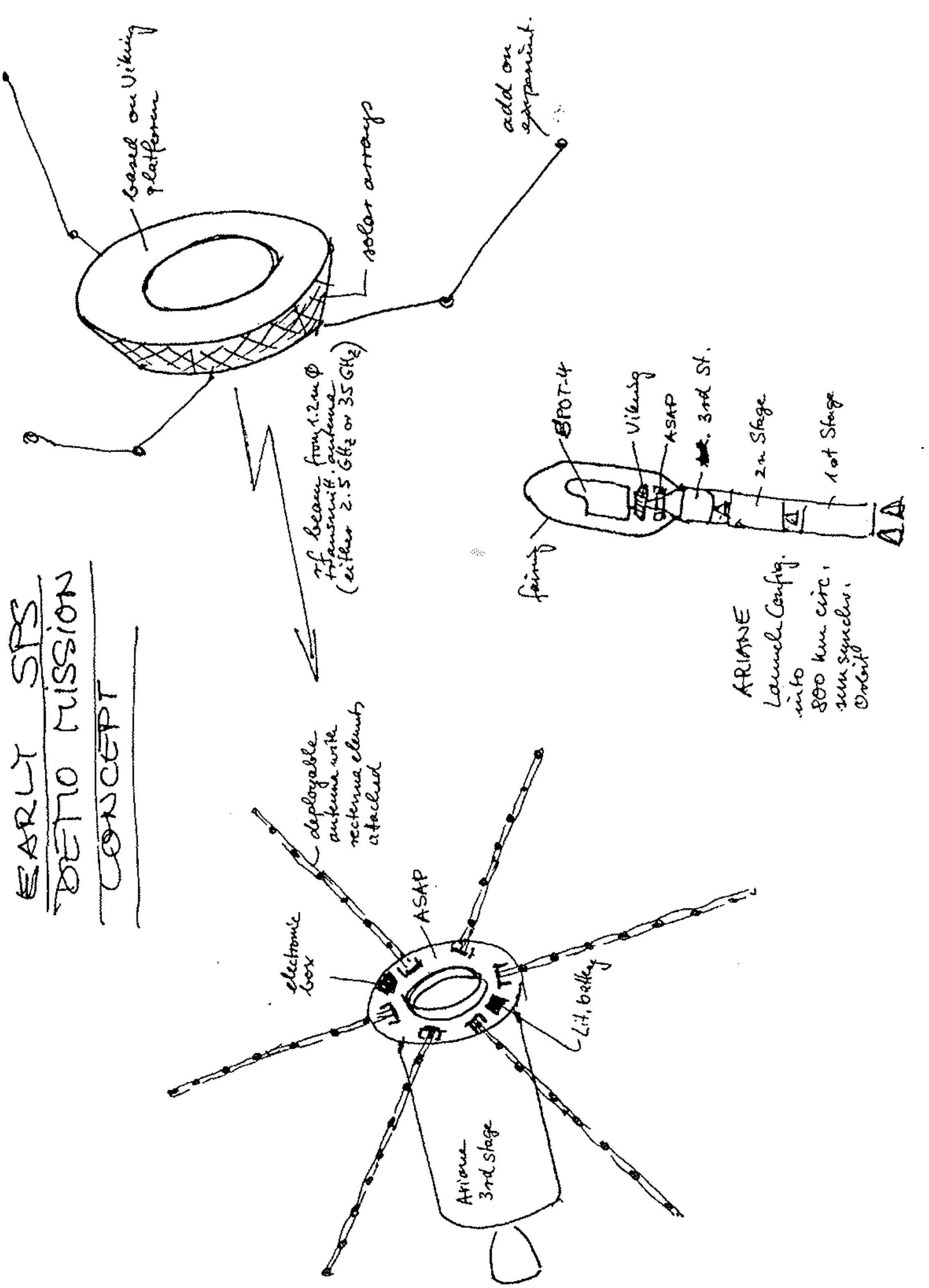


Figure F.1 Viking ASAP demo mission

Appendix G

Scheduling: Macproject II

For the creation of the schedule of SSPP MacProject II software has been used. The outline of the project is broken down into phases which are broken down further into tasks, milestones and dependency lines.

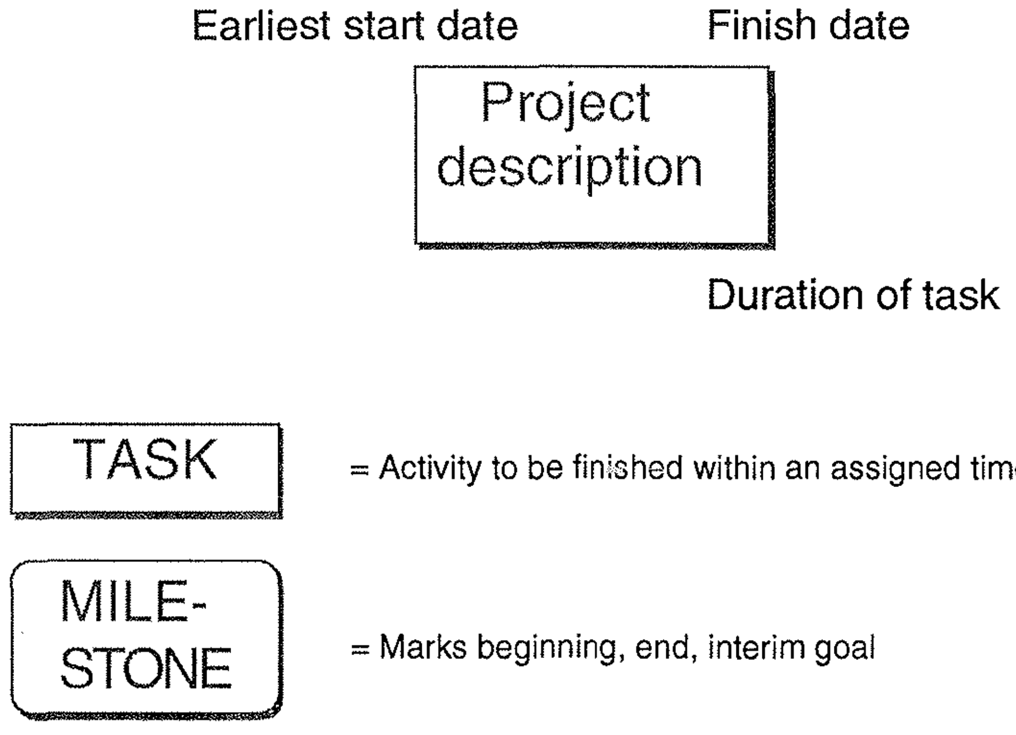


Figure 1

The tasks are presented in square boxes (figure x.1) **REPORT GROUP: CHANGE THE NUMBER OF THIS FIGURE.** A task is a distinct, identifiable activity within the project that can be accomplished within a reasonable amount of time. The basic information which describe a task are the name of the task and the amount of time it will take. In each upper left corner the time is mentioned where a task starts, in the upper right corner where it ends. The duration is mentioned in the lower right corner in months.

The milestones are presented in boxes with rounded corners. A milestone marks the begin or the end of a project or sub-project, but it can also be used to mark an interim goal for a phase of the project or a set of tasks. Milestones serve as visual cues and most often do not have a duration.

The boxes are connected by dependency timelines. These depict how the tasks and milestones interact. Usually these do not take any time and appear as diamonds in the task timelines. Sometimes durations appear on the dependency lines between boxes. The number without a star is a finish to end dependency, which means that it cannot start after a certain amount of time after the previour task. The one with a star is a start to start dependency, which means that the task cannot start after the start of the previous task.

The schedule for the SSPP is broken down into several levels. level 1 is the overall development program. It is broken down into the several subprojects of level 2. These subprojects are again broken down into the sub-projects and tasks of level 3. Every level is shown as a single chart. In this report two kinds of charts are displayed: schedule charts and task timelines.

A schedule chart is called a PERT chart in the project management terminology. It graphically depicts all the tasks and milestones, the complete project schedule, and the dependency relationships among tasks and milestones. In the schedule charts task are represented as boxes, milestones as rounded rectangles and supertasks as rectangles with a white oval inside them.

The task timelines show tasks as horizontal bars that span elapsed time for a task or a milestone. The bars are lined out in chronological order. Super tasks have a half-oval at each end of the bar for planned duration. A milestone that has no duration appears as a diamond. Tasks appear as simple bars. The white portions of the bars symbolize the duration of a task. The shaded portions of the bars symbolize the slack.

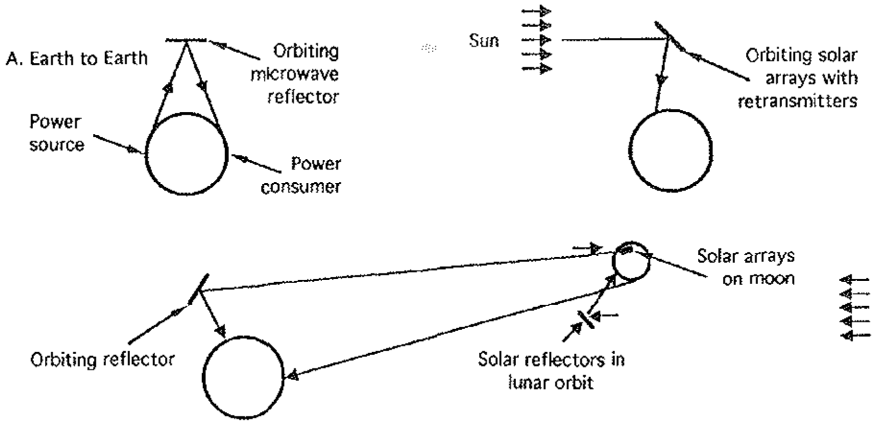
Appendix H

Past and Current Space Solar Power Projects

The purpose of this appendix is to provide a limited amount of information on all significant space power projects, proposals and demonstrations. There are references provided where available for those interested to locate more information on the individual designs. We outline the main properties - power levels, orbit selection and technology choice - and innovations of each project. In addition, some are illustrated for clarity. References for all projects are given at the end of the entry for that project.

The projects have been divided into two main groups. Firstly large scale projects, those of 500kW and over, and secondly those below this figure. This second category will include purely experimental projects with direct space power application. Within these categories the projects are arranged roughly chronologically.

The three basic types of space power are outlined in the diagram below. Most of the concepts mentioned below correspond to one of these models, with the exception of those concerned with powering satellites and the demonstrations.



The three basic Space Solar Power configurations for Earth Power
A. Earth to Earth. B. Space to Earth. C. Moon to Earth

1. Big Projects (over 500kW)

Glaser's concept (1968)

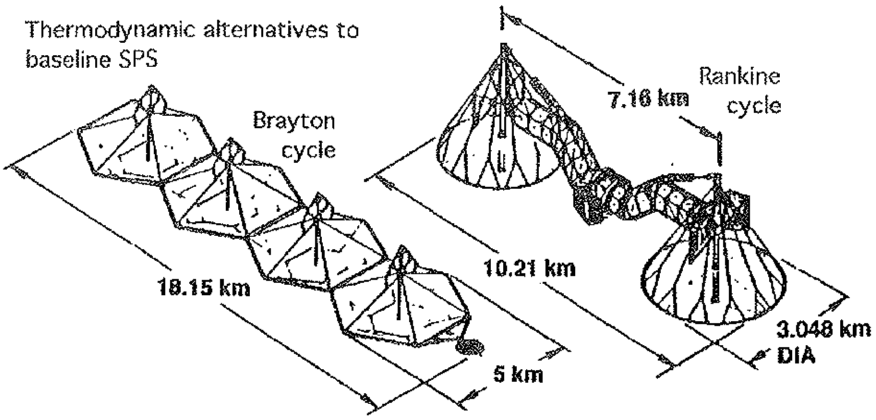
The initial serious proposal for space power utilisation, Glaser suggested the use of large platforms in geostationary orbit. Photovoltaics were used for conversion of solar energy to electricity and the power beamed to earth using microwaves at 2.45GHz. Also suggested were nuclear reactors in orbit, as they are safer in orbit than on the Earth's surface, beaming power down to earth.
Glaser, P.E., *Power from the Sun: Its Future*, Science, vol 162 no 3856, pp 857-861, 1968

NASA/DOE Reference System (1980)

In 1980 NASA and the U.S. Department of Energy (DOE) completed their Space Solar Power study, started in 1977, which considered all relevant technologies and many design concepts before creating one detailed point design, called the reference design. This consisted of a system of 10km by 5km platforms of silicon photovoltaic cells in geosynchronous orbit, with a 1km diameter transmitting antenna for 2.45GHz microwaves. These satellites would be used to provide 5GW of continuous baseload power to earth, collected at 5km diameter rectenna sites on Earth. The study concluded that such a system would be economically viable, but a subsequent overview by the National Academy of Sciences rejected this saying that certain assumptions, notably the low launch costs assumed in the initial study, were unrealistic.
_____, *Solar Power Satellite System Definition Study*, NASA/DOE, 1980

DOE/NASA Solar Thermal Concept (1980)

In the NASA/DOE study all technologies were considered. As well as photovoltaics several thermodynamic conversion techniques were considered, but finally rejected by comparison with the final reference concept. Illustrated here are conceptual designs for Brayton and Rankine cycle satellites designed by Rockwell International.
_____, *Use of Space for Earth Power*, Rockwell International, 1992



General Dynamics/NASA study of lunar resources for satellite construction (1980)

This report outlines ways in which direct substitution of lunar materials for those in the NASA/DOE reference concept could reduce the cost of this satellite. Thus while the design is the same the economics and development plan for such a satellite are significantly different.
Bock, E. et al, *Lunar Resource Utilisation for Space Construction*, vols 1-3, General Dynamics Convair, 1979

Rockwell Post-Contract IR&D

A reevaluation of the NASA/DOE study conducted subsequently, this concept uses newer technologies (multiple bandgap solar cells, magnetrons) and significant quantities of lunar materials to produce 15.4GW on earth.
_____, *Use of Space for Earth Power*, pp.22,Rockwell International, 1992

Pioneering the Space Frontier: Report of the President's Commission on Space (1986)

A set of recommendations for future American activities in space. The report spoke favorably of the possibilities of future power from space.
Pioneering the Space Frontier: Report of the President's Commission on Space, Bantam Books, 1986

Energy Storable Orbital Power Station (ESOPS) (1987)

An ISAS/Toshiba concept for a ten MW space to earth space power satellite, ESOPS would fly in low earth orbit. Transmission would be for 35 minutes per revolution, with an orbital period of 144 minutes. Between transmission times the energy would be stored as heated fluid, with LiF used as the thermal transfer fluid for a Brayton cycle engine .
Akiba, R. et al, *A Concept of the Energy Storable Orbital Power Station (ESOPS)*, Acta Astronautica, vol 15 no 11, pp 893-900, 1987

The Synthesis Group Report

This study considered four possible future paths for space activities, including the possible future industrialisation of space to provide power for earth.
America at the Threshold, Synthesis Group

NASA Lunar Energy Enterprise Case Study (1989)

This task force examined the major options for the utilisation of extra-terrestrial power. It outlined the major developments in key space power-related technologies and evaluated the significance of these developments in relation to space power as an alternative energy option for use on Earth and as a potential stimulus for space infrastructure developments. It also examined possible usage of extra-terrestrial resources. They also studied Solar power bases constructed on the moon to supply the majority of the power needed on Earth.
_____ , *Report of NASA Lunar Energy Enterprise Case Study Task Force*, NASA Technical Memorandum 101652, 1989

Earth to Space Transmission Concepts (1989)

Several concepts for large scale Earth to Space power beaming have been proposed, including large industrial parks powered by a system of equitorial earth-based transmitters. A variation on this theme is to use this power to drive electrical propulsion on a large platform to be used for payload transfer between low earth and geosynchronous orbit.
Brown, W., *History and Status of Beamed Power Techn ology and Applications at 2.45 GHz*, Second Beamed Power Space Workshop, p.181, NASA, 1989

Solar power satellites built of lunar materials (1985/1989).

Commissioned by the Space Studies Institute from Space Research Associates. The purpose of the report was to consider the extent of possible uses of lunar materials in space power satellites. All relevant technologies were considered as in the NASA study, but with the emphasis on optimising the design for use of lunar materials.

The 1985 study considered transportation of materials from the moon to geostationary orbit to be a factor of 50 less expensive than from Earth and all reasonably abundant lunar materials were considered available for use. The conclusion was that 99% of a space power satellite could be constructed from lunar materials.

In 1989 a follow-up study was commissioned. This time there were restrictions placed on the materials available for use. Only those which could simply be extracted were considered.

When Space Shuttle external tanks, glasses and glass-glass composites were allowed, the study consluded that 64.9% of a space power satellite could be lunar in origin, with a 55.1% cost reduction when compared to the NASA/DOE baseline study.

Considering availability of iron and oxygen in addition, 91.7% lunar construction was reported, with a 69.8% cost reduction. In this case the price of using GaAs solar photovoltaics with concentrators became comparable to planar silicon structures.
_____ , *Solar power satellite built of lunar resources*, Space Research Associates, 1985
_____ , *Near-term non-terrestrial materials usage in solar power satellites*, Space Research Associates, 1989

SPS 2000

This design model has been proposed by the ISAS Space Power Satellite Working Group to generate a clear image of realistic space power systems. They selected an equational circular low earth orbit for such systems because of projected launch system availability before 2000. The satellite consists of a prism shaped structure stabilised by a gravity gradient. It collects energy using photovoltaics and aims to transmit 10MW at 2.45GHz to Earth. The study is intended to consider not only the technical aspects of space power but also the environmental aspects of its implementation.
Nagatomo, N. & Kiyohiko, I., *An Evolutionary satellite power system for international demonstration in developing nations*, SPS '91. Power from Space, Société des Ingénieiurs at Scientifiques de France, p. 356, 1991

Project SELENE (Space Laser Electric Energy)

Systems studies and technology development work towards creation of a 10m segmented telescope to send 10MW from FEL (free electron laser) in the micrometer range to spacecraft. The power would be collected with conventional solar cell technology. The study also examines methods for optimising solar cell response for the beam and improvements in FEL technology.

Project Selene, NASA, 1992

2. Smaller Projects / Demonstrations

Microwave Ionosphere Non-Linear Interaction Experiment (MINIX) (1983)

A mother-daugher power beaming experiment launched on a sounding rocket in 1983. A single magnetron was used for transmission.
Japan External Trade Organization, *Solar Power Satellite R&D in Japan,* New Technology Japan, 1991

Russian SPS (TsNIIMash)

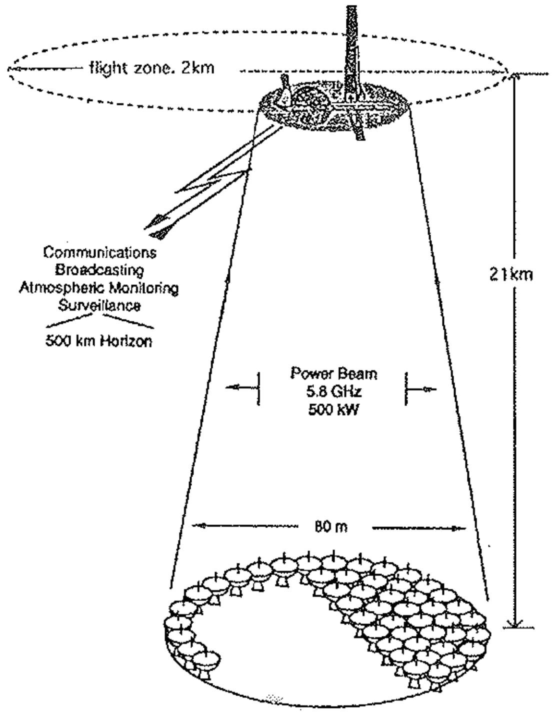
Illustrated below, this is a seventy tonne structure in circular low earth orbit. It produces 250kW for beaming to earth. The power station is manned by cosmonauts who carry out adjustment and repair work, and is launchable using existing vehicles.
Mozjorine, Y.A. et al, *Small-scale space power stations: Feasability and usage prospects,* SPS '91. Power from Space, Société des Ingénieiurs at Scientifiques de France, p. 381, 1991

IGRE's 100kW demonstration project

A general discussion of the issues involved in providing space power, with an emphasis on third world participation. IGRE is a proposed not-for-profit organisation (Institute for Global Rural Electrification) to bypass the structures of government and build a hardware infrastructure. Initial demonstrations would transmit 100kW at 2.45GHz to Earth. Low earth to geostationary orbit transfer would be accomplished using ion rockets powered by cells on the spacecraft.
Leonard, R,S., *The IGRE's 100 kilowatt demonstration project,* SPS '91. Power from Space, Société des Ingénieiurs at Scientifiques de France, p. 393, 1991

SHARP

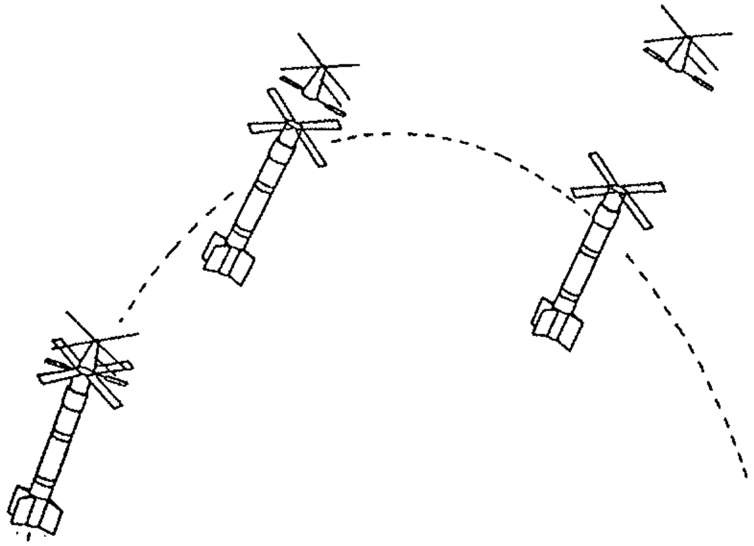
This is a project aimed at the creation of a high-altitude communications platform for Canada. In 1987 a small airplane flew powered 500kW of energy beamed at 5.8GHz. A larger airplane has been built and tests are imminent. The planned configuration is illustrated below.
Schlesak, J.J. et al, *SHARP Rectenna and Low Altitude Tests,* IEEE Global Telecommunications Conference, CRC, 1985



Microwave Energy Transmission in Space (METS)

The first space-borne experiment in the METS program, ISY METS, will be launched in 1993. One objective of this experiment is to demonstrate the feasibility of wireless energy transmission in space. The other objective is to study non-linear plasma effects due to the high power microwave energy beam in the space environment. A microwave beam of 936W at 2.45GHz will be transmitted from the mother section of the rocket to a daughter section using a newly developed phased array antenna. Previous work in this program included a ground-based energy transmission test using a small airplane as a target, the Microwave Lifted Aircraft Experiment, MINAX, in 1992.

Kaya, N. et al, *Rocket experiment METS - Microwave Energy Transmission in Space*,SPS '91. Power from Space, Société des Ingénieiurs at Scientifiques de France, p. 336, 1991S



The ISY METS experimental configuration

Space Flyer Unit Energy Mission

The Space Flyer Unit (SFU) is a free flying platform retrievable by Space Shattle. The ISAS SPS Working Group is proposing the SFU Energy Mission as a follow-on mission to conduct SPS related research. The following experiments are proposed on the mission: Termo-dynamic Power Generation Experiment (TDPGE), Electric Propulsion Experiment (EPEX), Microwave Energy Transmission in

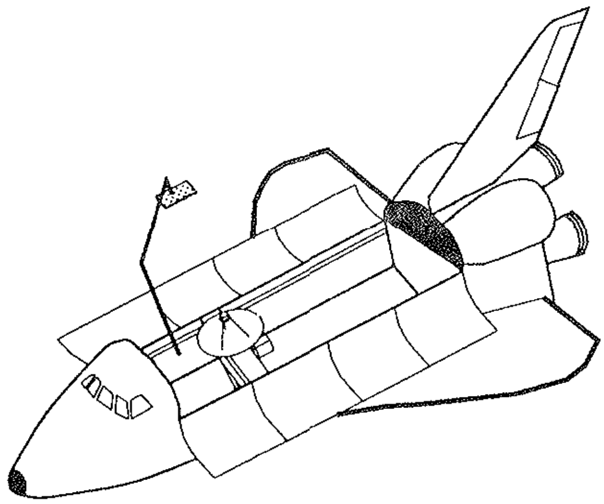
Space (METS), Space Tether Experiments (STEX), Autonomous Satellite Retrieval Experiment (ASREX) and Laser Propulsion Experiment (LPE).
Japan External Trade Organization, *Solar Power Satellite R&D in Japan,* New Technology Japan, 1991

Japan Power Satellite (JPSAT)

JPSAT, formerly known as Power Supplying Satellite (PSS), is a feasability study to build on the results of METS in order to design a larger scale solar power satellite, providing from tens to hundreds of kW. This power would be transmitted to customer satellites at 24GHz. The design is modular, with each small module integrating solar cells, FET amplifier and a micro-strip transmitter. Together, these transmitters will form a full phased array.
Matsumoto, H. et al, *A feasability study of power supplying satellite (PSS)* SPS '91. Power from Space, Société des Ingénieiurs at Scientifiques de France, p. 375, 1991

Demonstration of microwave power transmission in space (1991)

This is a three phase plan. The first phase is for an internal shuttle-based experiment, with a rectenna at the end of the Shuttle remote manipulator receiving power at 2.45GHz from an antenna located in the cargo bay. The second phase is for a similar experiment carried out with a free-flyer and the third using a free-flyer but utilising higher frequency transmission (35GHz) to transfer more power with smaller antennae.
Chang K. et al, *Demonstration of microwave transmission in space*, SPS '91. Power from Space, Société des Ingénieiurs at Scientifiques de France, p. 343, 1991



Chang's first phase experiment

Eurospace Powersat Study (1992)

A study commissioned by ESA examining possible development of near-term, low cost ($10M for the first one) demonstrations of microwave or laser beaming from a satellite in low earth orbit. The end goal is to create a system to supply any increase in Space Station Freedom's power requirements.
______, *Powersat Study*, Eurospace, 1992

Appendix I

Questions to be Addressed

The predictions arrived at by the preceding assumptions are not desirable. By breaking at least one of the assumptions, alternative futures can be imagined. The most acceptable assumption to change is likely to relate to new sources of energy. A proposal to provide additional energy to Earth via Solar Power Satellites (SPS's) was presented originally by Peter Glaser [Glaser, 1968] and has been explored by many researchers since then.

When such a program is undertaken, however, a number of questions arise which must be answered before large scale solar power can be provided. During the first two weeks of the ISU summer session a number of brainstorming sessions were held to determine what these questions might be. These questions are listed below.

1 Economic/Businss Issues

Cost and Economic Viability

Does current analysis of the economics of the SSPP program allow for commercial funding?

In order to decide on funding sources and other financial issues it is important to know to what degree the SSPP program as a whole is viable as a commercial venture. If the Return On Investment (ROI) is negligible, it is clear that commercial funding will be impossible. If there is some degree of commercial return from the project, combined governmentaVcommercial funding may be possible. It is essential to establish this balance from an analysis of the program at the outset.

Subsidiary questions relating to this issue are: Are there space infrastructure projects worth undertaking which would in turn reduce overall program costs and improve ROI (eg. developing own launch service?) What is the impact of an early drop in launch costs arising from other programs?

The question then follows: to what extent is government/public finance necessary, and what form should it take? On the basis of the analysis described above, the requirement for government-based funding can be ascertained. The form of government funding undertaken must be determined. For example, it may be possible that governments would be required to underwrite the risk only, or to offer fiscal incentives for commercial participation in the program. Alternatively, governments may be required to provide large scale funds directly.

Finance

What are the financial sources?

Potential contribution from funding sources such as governments, multinational bodies (EEC, UN), private industrial enterprises, and financial institutions is to be determined. The degree to which developing countries are able to fund the program will impact the amount of funding available (both governmental and commercial), and must therefore be determined.

What is the method of financing?

This includes addressing the balance of debt and equity and the use of financial instruments such as leasing and barter.

Should financial advantages be offered to funding sources for an early financial commitment to the program?

Incentives that may be offered for an early financial commitment to the program should be considered. The form that these incentives might take relates to management structure issues (see below).

What is the cost impact of a 'juste retour' (fair return) policy on industrial contracts?

The cost implications of establishing a 'juste retour' policy for governmental participation (such as that currently used in ESA) must be addressed, along with the question of how to minimize the potential disadvantages, should such a policy be adopted for political reasons.

What is the reliability of government/public support?

It is important to determine the likelihood of a sustained government commitment to the program until completion or commercial return. Possible competition for funds would come from both other space programs and the successful development of other large scale energy sources (e.g. nuclear fusion). This analysis may impact both the program's degree of reliance on public funds, and the management structure (see below).

What is the time scale for funding requirements, and how may this be optimized? How do funding sources change with different program phases?

The time scale funding questions address the evolving needs of the project and the involvement of the funding parties. Within program schedule constraints, a degree of financial flexibility exists which may be optimized. For example, not all debt needs be incurred at the beginning of the program.

What is the program schedule and how does it impact cost?

The complexity and potential for large overruns on a project of this nature makes it imperative to exercise the greatest care in establishing its initiation date, phasing, and duration. This issue is important in all cases, but particularly so for demonstrations and early commercial opportunities, where the project's funding might depend heavily on fast initiation and development.

Management and Organization

What management structure should be implemented?

This question aims at establishing the convenience of managing the project(s) through intergovernmental agreements (IGAs), industrial consortia, or through the creation of semi-autonomous organizations (e.g. Intelsat, Eumetsat), and to what extent the management structure should follow past experiences. In this context, additional questions on the allocation of decision-making, power and economic incentives for parties involved, and a possible balance between these two benefits must be adressed. For example, commercial enterprises might be left wi th less weight on the hierarchy relative to their participation, but with more share of profit. Innovative structures, reflecting the international character of the project and other peculiarities may be considered.

How do the incentives for government investment impact management issues?

Government may choose to support the program on the basis of future financial return (national prosperity), environmental benefits, and other concerns. Understanding government motivation will have an impact on program management structure and the organization to best capitalize on the nature of the support.

2 Demonstration-Specific Issues

Cost of the demonstration program

What is the cost of the desired demonstration program?

In order to decide what will be achieved, it is essential to assess the amount of money available for the demonstration program. This will be imponant to define the specific goals that can be achieved with the demonstration model and the dates of completion.

How can the demonstration programs be optimized for the acquisition of funding sources?

Specific strategies used to finance the demonstration program may be different than the ones used for the commercial or full scale program. At this point, the test may not be financially viable, but may serve as a step in the development of further uses. It may therefore require a particular financial structure and sources of financing.

Goals of the demonstration program

Should the demonstration program concentrate initially on early commercial uses before demonstrating large-scale space to Eanh power technologies?

Depending on the goals and money available for the demonstration program, its definition could greatly change. It could be used to develop technologies for the large scale space program or as an intermediate step for the early commercial use programs.

Should the development of new technologies be a goal, or should we concentrate on the demonstration of existing space solar power technologies ?

The definition of the goals of the demonstrator is also required to specify where the emphasis of the program will be. Demonstration of the technology, the benefit to human life, or the financial aspects of future projects shall be performed.

Early Commercial Use Issues

For an initial investigation on the early commercial possibilities of a SSPP, three fundamental questions must be addressed. First, what are the market opportunities of a SSPP ? Second, what is the commercial viability of such a market ? Finally, how can the required demonstration projects be used for early commercial use?

The Investigation of market opportunities

The investigation of market opportunities includes, but is not limited to the character of the market and the market size. Analysis of market opportunities is an essential precursor to the determination of an application's economic viability.

What is the character of the market?

Initially, the character of the market should be assessed. This can be examined from at least three different perspectives: potential applications, potential customers, and the competition faced during early market development.

Potential applications include Space-to-Space, (e.g. remote power supply to electric propulsion systems or to satellites) and Space-to-Earth. This portion of the question would address, for example, the identification of locations in, developing nations that have no other viable power access, as well as developed nations needing to develop alternative energy sources. A third application would even be Earth-to-Space. We feel it would be prudent to investigate the possibilities of using current power systems in supplying energy to space via power beaming.

The second portion of the market analysis would be the identification of potential customers .Customers such as governments or private enterprise will dictate the reliability and consistency of the market demands. Cyclical or unreliable energy demands will strongly impact commercial market opportunities.

The third question to be asked is the type and the extent of competition an SSPP would face. If new commercial markets are identified, other organizations (commercial or governmental) may choose to enter the market and compete, thus potentially reducing the market share of an SSPP.

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What is the size of the market?

lhe market size will determine the initial scope of an SSPP. This can be examined in three parts. First, will the provision of new energy supplies stimulate growth in the effected areas? If it does, then how will the growth of these areas impact the availability of what the new systems can be supply. Fmally, at what time scale will this growth occur?

The commercial viability of these markets

A cost breakdown for each one of the proposed applications should be calculated. Given these data, one should study the likely return on investment for each application, and determine if a staged implementation would offer any advantages.

Commercial viability might be reached if some capital costs are written off by government institutions. What parts of the project(s) could be sponsored by govemments? For what amount and under what uncertainties?

Clearly, the cost of space systems is such that if space hardware constructed for the demonstration phase could be re-used, the savings over all new construction would be substantial, and should be investigated fully. This consideration may impact the selection of suitable demonstration.

Large Scale Commercial Application

If the demonstration of solar beamed power proves to be commercially viable in the near terrn, then development of large scale commercial applications (SSPP) should be considered. Additional questions that either raise new issues or re-investigate previously stated issues need to be reexamined:

1) What is the cost of the project? 2) What is the market? 3) What will be the availability of investment capital? S) What will be the expected completion date? 6) When will the project break even? 7) What interfaces should be established with the existing structures of management? 8) What will be the global economic impact of the project?

What is the cost of the project?

Taking into account previously demonstrated costs, it is important to show accurate cost analysis to convince potential investors of project credibility. Due to past failures and inconsistency in cost analysis (e.g., Space Station Freedom cutbacks), it becomes critical to have a thorough understanding of the question, "how much will a project of this scale cost?" The main costs to be considered should include the building, operation, and maintenance costs. In addition, the cost of lost investment opportunities should also be investigated.

What is the market?

An analysis of the market must be performed in order to demonstrate project feasibility and scope. For example who will be the major consumers of solar generated beamed power? Will the main consumers be on the level of countries, power utilities, or individuals? The cost of the energy produced by solar beamed power will need to be compared to the cost of electricity produced by other energy sources. What will the cost of electricity be to the end user? Will that cost be competitive considering the electricity market at project completion?

What will be the availability of investment capital?

In light of other potentiai programs for government to invest in, and other more lucrative investments for private enterprise, will there be any available capital to put into a space solar power program?

What will be the expected completion date?

It will be necessary to establish a schedule and time frame that is realistic and one that is not intimidating to potential investors, i.e. return on investment has to occur within a reasonable length of time. What should the time frame be for such a large and elaborate project?

When will the project break-even?

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Large Scale Commercial Application

If the demonstration of solar beamed power proves to be commercially viable in the near terrn, then development of large scale commercial applications (SSPP) should be considered. Additional questions that either raise new issues or re-investigate previously stated issues need to be reexamined:

1) What is the cost of the project? 2) What is the market? 3) What will be the availability of investment capital? S) What will be the expected completion date? 6) When will the project break even? 7) What interfaces should be established with the existing structures of management? 8) What will be the global economic impact of the project?

What is the cost of the project?

Taking into account previously demonstrated costs, it is important to show accurate cost analysis to convince potential investors of project credibility. Due to past failures and inconsistency in cost analysis (e.g., Space Station Freedom cutbacks), it becomes critical to have a thorough understanding of the question, "how much will a project of this scale cost?" The main costs to be considered should include the building, operation, and maintenance costs. In addition, the cost of lost investment opportunities should also be investigated.

What is the market?

An analysis of the market must be performed in order to demonstrate project feasibility and scope. For example who will be the major consumers of solar generated beamed power? Will the main consumers be on the level of countries, power utilities, or individuals? The cost of the energy produced by solar beamed power will need to be compared to the cost of electricity produced by other energy sources. What will the cost of electricity be to the end user? Will that cost be competitive considering the electricity market at project completion?

What will be the availability of investment capital?

In light of other potentiai programs for government to invest in, and other more lucrative investments for private enterprise, will there be any available capital to put into a space solar power program?

What will be the expected completion date?

It will be necessary to establish a schedule and time frame that is realistic and one that is not intimidating to potential investors, i.e. return on investment has to occur within a reasonable length of time. What should the time frame be for such a large and elaborate project?

When will the project break-even?

The break-even point is reached when revenue completely covers cost, and any extra revenue is profit. Investors are interested in the moment that the project will become profitable. Shorter termreturn (3 to 5 years) on investment, compared to longer term return (more than 5 years),--will determine different types and combinations of investors. For example, a short term break-even point will interest private investors. A long term period may only be supportable by governments. When is it no longer possible to support a project solely through private enterprise? --- -

What interfaces should be established with existing management structures?

Due to the large scale nature and international aspects of the project, it will become necessary to manage it in terms of cost effectiveness, the continuation of the different space programs, the-standardization of space flight equipment, and international cooperation. What interfaces will most

effectively operate under these constraints?

What will be the global economic impact of the project?

A large scale project will affect the global economy and will have to be justified politically, socially, and economically. Questions concerning economic impact include: Will new jobs be aeated or lost? Will there be new technologies created and distributed? Finally, w-hat effects will there be on other world-wide power industries?

4 Political, Social, and Legal Issues

Public Concern / Perception:

Does the public perceive a need for beamed power?

There could be a major gap between the short and long term energy source availability and the public perception of the problem. The individual might not even be aware of the inaeasing energy demand of the world. This could be due to the fact that people do not see the connection between personal energy consumption and the energy demand for society as a whole. As long as there is no immediate lack of energy, the public might not realize that there is a need for developing new and alternative energy sources. Because of high total costs for a SSPP, the public perception issue is important both for the near and long term applications.

What kind of resistance will there be to beamed power?

The use of beamed power is likely to provoke resistance from environmentalists, political groups, nations, etc. In order to carry out a SSPP, and especially, a large scale SSPP, it is important to be aware of the possible objection to the project. The present design concepts for a SSPP involve space power stations visible from earth and large rectennas on the ground. The obvious aesthetic impact of these structures is one of several issues that will affect the public's view of a SSPP. In terms of nonterrestrial aesthetic impact, will the creation of "new heavenly objects" be accepted by the public? Massive opposition to a SSPP could ultimately prevent the development and implementation of a SSPP.

How can the public be informed about a SSPP and its possible side effects?

There is a need for information when implementing new technology. The use of beamed power from space will ultimately depend on the acceptance from the public. Therefore, choices on what type of information, how it is distributed, and to whom, is of vital importance to the degree of public support. These issues are in turn connected to the assessment of perceived personal and societal benefits compared to the problems associated with beamed power.

Control of Operations

If we consider a SPS as an instrument that transmits energy from space to a spot on the ground, then a SPS could be used as a weapon. One condition for public acceptance of a SPS may be that we assure the public that the beam density doesn't exceed some critical value and that the beam cannot be turned outside a rectenna intentionally.

When a SPS is considered as a power plant, the power capability, supply, and price are important factors for the public. In this program, the evaluation of two points will be examined. The above considerations will make it difficult to choose the manager of this model.

In addition to the questions raised above for the short term, there are some long term questions that have to be considered for large scale applications of Solar Space Power. One of the most important questions in this context is the question of how the whole project should be organized. International cooperation, and how this cooperation will be assured, must address the different interests and the various political standpoints of the different countries that would like to participate.

Apportionment of Costs and Benefits

One of the main questions regarding the potential future use of a SSPP, not only from an economic, but also political point of view, will concern the apportionment of the costs and benefits of a SSPP. In this respect, both in the early development phase (demonstration projects) and later on when large scale solar power plants are going to be considered, the problem of how to finance the project will have to be solved.

Specifically, the following questions will have to be addressed:

a. Who will invest in technology demonstration projects and early commercial SPS systems? The SPS program is a very innovative and expensive project which will require not only funding from the usual sources, such as government, but also commercial sources. Consequently, market studies will have to be conducted at a very early stage for funding from electric and power utilities, communications spacecraft companies, resource manufacturing companies, and others.

Who will benefit financially from and receive power from early SPS systems? In this initial phase, questions will have to be answered as to whether only the investors will benefit financially from the technological advances or will the knowledge gained be available for everyone to use. Agreements will have to be made to decide who will hold the rights to the inventions and discoveries made in this phase. A similar approach will have to be made on the financial benefits of small scale power applications, such as communication satellite users.

b. Who will use the energy? Will there be different benefits for different countries?

How will the energy be assigned and charged for?

Will the bill be according to the energy demand?

Will the bill be according to the financial strength of the user?Will there be CO2 credits?

To determine the users of the power generated on a large scale, studies will have to be conducted for the breakdown of the users among the different countries and/or within a single country. How the energy will be sold and who will pay for it. It will have to be decided how energy will be divided among the participating organizations.

Social Side-effects

During the near term phase, the possible social side effects are thought to be small. But we need to understand what the impact of future large energy systems may be on communities in developing countries to plan ahead.

In the developed world, the use of space power could possibly replace other power sources. But assuming that the amount of energy available worldwide will increase dramatically, especially in the developing world, there may be a sizable impact on these communities. The availability of useful energy, where previously there may have been none, could change people's lives by providing new services. The provision of energy could also impact the high birth rate in developing countries, and thus help to reduce the global population growth.

The provision of a rectenna will provide both employment and power to a local community. This will help acceptance of the siting of a rectenna in a community. The construction of a rectenna will impact the local landscape and environment and will probably change the infrastructure of the community. These factors also need to be examined.

Legal Framework

The goal of jurists is to determine the legal framework in which other people have to develop a Space Solar Power Program. The legal status of outer space needs to address the major principle of freedom of use. This includes both the principle of peaceful purpose (e.g. how can we prevent the use of SPS for military purposes) and the principle of non-appropriation. Specifically, the problem of appropriation of extraterrestrial materials located on the Moon and asteroids to build SPS's needs to be addressed.

The allocations of frequencies and slots in GEO and LEO is needed to avoid collision and electromagnetic interference. Will the International Telecommunications Union (ITU) be competent to do it? Can the allocation of slots be separated from the allocation of frequencies?

Do we need to create a multinational consortium to manage SSPP? What model should be used? Possibilities include deep sea, INMARSAT, INTELSAT, ESA, ITU, and the U.N. How does one establish and manage it? What role is there for private corporations? Who will finance it? Who will be the owner? Who will control individual SPSs? What rights do technology developers and financiers have? How can the developing countries participate in it? How do we manage it to provide direct environmental and economic benefits to the earth, in particular, the developing countries? Recall that the lack of energy is one of the most important causes of desertification in Africa. How will the energy be assigned and charged? Who will decide on the locations of rectennae?

Responsibility and liability questions exist, including the input ability of states (article 6 OST.) What if a SPS were to damage the Earth upon uncontrolled re-entry? What about collisions with other spacecraft? What if a SPS forms additional orbital debris? What liability is there concerning microwave or laser beams in the space environment? In Earth's environment? If global warming is associated with SPS systems, who will be liable? Is this the problem of both non-governmental entities and governments involved in the program?

Settlement of disputes poses other questions. Will there be mandatory jurisdiction? What kind of jurisdiction will it be; arbitral, judicial, or some other type? What procedures will be followed?

5 Technical Issues

Short Term Technical Issues

This section is divided in the same way as previous sections; (1) demonstration, testing and early commercial development and (2) large scale commercial applications. The focus of the questions related to near-term demonstration, testing and early commercial development will be on those issues which demonstrate technical feasibility of beamed power and which need to be resolved to enable follow-on large scale power production. Results from near-term demonstrations will be used in the process of scale-up for large scale production. The definition of the near-term demonstration task will evolve and should emphasize communication with the long-term task in an iterative process. As problems that need early demonstration or proof of concept are encountered in the development of large commercial applications, they can be incorporated into the demonstration task.-Three major areas of technical issues for near-term demonstration, testing and early commercial developments are discussed; power system, spacecraft, and technology development.

Beam Characteristics

Energy transmitting characteristics have to be selected during the first stages of the program so that the demonstration of the Space Solar Power Program is demonstrating fully all possibilities of beamed power. There are several possible methods to transmit energy from space to earth. Methods vary from passive mirrors reflecting the surlight to earth to active beam generators.

The frequency as well as the power density for an active beam generator should be selected so that there is minimal attenuation in the atmosphere. The power density has to be large enough to increase the efficiency of the power transmission link, but low enough to prevent any effects caused by energy absorption of the atmosphere. Safety issues are discussed in section "Environmental & Safety Aspects"

Although microwave beaming seems to be the best choice at the moment, the alternatives should at least be analyzed. The development of techniques and materials may change the effectiveness of different methods.

Power Conversion

What is the best and most efficient conversion method from solar energy to a beam transmitted down to earth? The conversion should be as efficient as possible. The best solution might be to convert the entire spectra that the sun radiates. The present technology is however not so efficient. For a

realizable power station, there are only few narrow bands on the radiation that can be converted to other energy forms.

Integration of SPS Power to Existing Electricity Power Networks

This is a most fundamental question which must be addressed. The outcome of this demonstration will determine how much funding power companies and other financial institutions will be willing to invest in any form of commercial SPS project. The demonstration will be very limited in the amount of power it can supply to the electricity grid, but it should show the scalability of any solution and the possibility for future development.

Power Beaming Pointing Efficiency

To gain public confidence in the use of microwave beamed power, any demonstration will have to show that the energy source is directable. It should also show the safety aspects of the system i.e. shutdown if pointing becomes a problem. If SSPP is adopted as a large producer of energy then the pointing efficiency could become a legal requirement of this kind of system. This capability will also impact on the ability of the system to produce a constant supply of power.

Electromagnetic Interference

SSPP Task A raised a number of questions concerning electromagnetic,interference with existing users of the spectrum: for example, telecommunications and radio astronomy. Obviously electromagnetic interference problems will be a function of the SPS transmission frequency, and today, most SPS interference questions have focused on 2.45 GHz in the Industrial, Science and Medicine (ISM) frequency band.

Some SSPP interference issues that must be addressed include:

• How to limit out of bandwidth power transmission levels (side-lobe energy)?

• How to minimize the interference effects of transmission frequency harmonics?

Spacecraft Control-Structure Interaction

The basic issue facing the structural and control engineers is the control-structure interaction problems observed in large space structures. These are due to the overlapping between the attitude control and structural frequency spectra, resulting in controllability and observability spillover effects. Presently, ground based experiments are under way in order to counteract the high mode excitation, however, further experiments in orbit will have to be carried out in order to confidently deploy large space structures.

Space Construction

Space construction will be imperative in order to deploy large space structures in orbit with the present state-of-art in launch vehicle technology. A feasibility study will have to be carried out in order to demonstrate deployment and robotic techniques. Would manned EVA construction be necessary? How would reliability in construction techniques be insured?

Spacecraft Orbit Selection

The issue of orbit selection is a fundamental one. For demonstration purposes, should a LEO be selected; and if so, which is the best orbit for the proposed design constraints, with considerations to launch windows.

Space Transportation System

It will be necessary to reduce weight of payload as much as possible. This could affect the design and deployment of a SSP satellit. For example, it may be necessary to increase the use of inflatables.

Solar Cells Development

It is necessary to make more efficient and compact individual solar cells. This relates to a SPS's dimensions and weight. If more efficient and compact solar cells, total cost will be reduced for a SPS. Also, if we can make fewer solar cells for SPS, then reliability will increase. Therefore, we need to determine what energy conversion technology should be used: For example, solar array technologies like thin films vs. crystalline, Si vs. GaAs, thermodynamic system or perhaps a combination of photovoltaic systems should be addressed.

Technical Aspects of Long Term Commercial Opportunities

The long-term technical problems of SSP are divided into four categories: single SSP satellite design, entire SSP network design, resources and manufacturing, and maintenance and operatibns. We areassuming that at the end of Phase B, the viability of small SSP systems will have already beendemonstrated, but at that time that there will not be any large-scale use of space solar power. Thus, the challenges that must be answered in the next phase involve implementing space solar power on a large scale to the point where it will provide a significant fraction of the Earth's total energy needs-- as well as the energy needs of systems in space.

The most important problems associated with the single SSP satellite are those involving scaling. Atthe end of Phase B, we should have small SSP satellites in operation providing power on a limited basis. Presumably, the final SSP satellites will be expanded and enlarged versions of these initial demonstration satellites if they are to provide power for much of the Earth. There will be several problems with such scaling. These include: maintaining control of the beam, maintaining both attitude and orbit control, ensuring stability of the entire structure, and reducing vibrations, particularly in the vicinity.of the transmitter. Also, there are other types of scaling problems, such as adapting the engineering failsafes needed to prevent misdirection and overamplification in Phase B to Phase C satellites without overly disturbing transmitting flexibility.

In addition to scaling, orbit selection of single SSP satellites will be a major problem. In Phase B, orbits will have been selected for demonstration purposes. But in Phase C', orbits for single satellites will have to be selected within the context of the entire SSP network, regarding not only physical constraints such as solar wind, space debris, and the Earth's radiation belts and magnetic field, but also user demands, be they on Earth or in space. Therefore, technical countermeasures to these physical constraints will have to be installed on each satellite.

Another question involves the SSP receiving stations. Once the best type of receiver is demonstrated in Phase B (e.g. microwave rectenna or laser receiver), the problems for Phase C include: Where should ground receivers be located? What will the architecture of the ground or receiving system be? And what is the optimum size of the receiver? This last question is especially important to the SSP satellite, because if a microwave system is used then the size of the rectennae will also determine the size of the transmitter. With regard to the transmitter, it is also important to decide whether the satellite should be required to change its beaming angle, so that it will be able to transmit power to various locations on Earth and in space.

Apart from the technical issues for the solar power satellite itself, there are also items that adress a complete network design as could be envisaged for large scale use of space solar power. We have mainly considered the problem of delivering solar power to the electrical power grid on earth.

Relating to the overall architecture and concept of large scale solar power satellites we have identified the following points.

What should the structure of SSP look like? Should it be a monolithic system (one or a few large satellites) or a more distributed system (constellations of satellites)? Should a single SSP system be capable of delivering energy to both Earth and to space-based receivers or are multiple SSP networks necessary? Also, should substantially different technologies and frequencies be used in these cases (microwave beaming for transmission to Earth and laser beaming for space-to-space)?

How do we decide between the requirements of continuous power or energy storage? If continuous power is needed, does this require SSP satellites in GEO or in LEO with relay satellites? For the LEO case, what orbit should be selected. Also, if relay satellites are used, how will efficiency of energy transmission be affected? If storage is required, how will SSP be more competitive than terrestrial solar energy?The issue of compatibility with existing network systems should be adressed (frequency, voltage, stability). Related to this, how do we select and enforce an industrial standard for SSP satellites and rectennae? Will these standards reduce production costs?

Finally, who and where are the end users for SSP (earth, satellites, moon)? This could impact the network concept. For instance, should a SSP satellite be positioned around the moon, both for demonstration and to provide energy for a lunar base that can then provide resources for the largescale SSP system?

Resources & Manufacturing is another group of aspects which has been addressed.How does the use of Earth resources compare to mostly using Lunar resources? Using extra-terrestrial materials will require new materials transformation processes. Materials test on earth is limited by the amount of lunar material we have. How are we going to determine the efficiency of those machines and processes? What about the problems associated with lunar resources extraction? Is the use of lunar materials for construction really more cost effective ? In order to reduce the Delta-V, and consequently reducing the price of putting objects in space, it has been proposed to use extraterrestrial resources and to build a SSPP in space rather than on earth. This possibility imposes technical and technological challenges. In fact, the use of lunar materials, for example, implies the settlement of a lunar base, and requires vehicles that land on and depart from the moon. Definition of studies, crew selection, life sciences research, etc. have to be done. Are lunar mass drivers really more efficient than simple lunar launchers? Is L2 really the best Lagrangian point for resource collection?

What space transportation systems and methodology would be required for use of Earth resources only and what systems would be required for use of Lunar and other non-terrestrial resources?

Such large structures as SSP with current launch volume restrictions give rise to the question of what the construction and maintenance method should be, and what level of automation is most effective for this (robots versus manned systems)?

Related to the station crew, will a permanent crew per station be required or can "roving" crews be used? What will be the effects of long missions on the crew (maintenance, construction, control, etc of SSPP).

Can resources from near-Earth asteroids economically be used for SSP? How about Shuttle and Energia main tanks? If these tanks are going to be used by SSP, then perhaps some essential construction materials can be added to them prior to launch?

Large scale use of SSP will increase the importance of avoiding space debris. How can we minimize the contribution SSP will make to this problem? How should this affect the question of whether SSP be constructed in orbit or deployed? Will space debris have to be cleaned in order for SSP to function?

The last group of aspects relates to maintenance and operations. Research, development, manufacturing, operations and maintenance must be considered, in essence the entire life cycle cost. Significant costs could be incurred due to environmental damage and management and should be included. Life cycle costs and cost per year should both be estimated and compared for all energy sources.

A quality expected of any public energy company is it's reliability. Unpredictable variations of availability can be very annoying for consumers. Whether it is the sole source of power or only one out of many, reliability will be expected of any SSP system. The question of how we can insure the reliability of a SSP system is then very important. A network of satellites and relays can be designed that will insure good consistancy in an ideal situation, but this will not insure reliability. For instance, atmospheric absorption might unpredictably lower the availability of power. Also, a breakdown of any component of the network will have potentially disastrous consequences and might take a long time to repair. The question of how much redundancy should be built in the system and what other measures might be taken in order to provide the expected power in all conditions and therefore insure that SSP becomes a useful energy source must be addressed.

Modularity of the system should be considered for concerns of both operations and maintenance. Modularity may increase development costs but could reduce manufacturing costs and could allow for quicker and easier subsystem replacement when damage is incurred. In addition, modularity may give opportunities for increases in operational capabilities by allowing for continuous improvements.

Solar cells of SSP satellites decrease in efficiency with time. Moreover, they can get hit by space debris and create some debris themselves. Other components of a satellite can also break down. It is therefore important to organize a program of maintenance which will insure long term consistancy in the supply of power. The question of how to decide when to repair or to replace a satellite as well as the question of what should be done with a satellite which has reached the end of it's useful life must be addressed (e.g. regeneration of solar cells in space).

Integration of SSP with other types of energy production could be advantageous. Can SSP and other energy sources be used in conjunction with each other to improve the output of all sources? How could the SSP be integrated into a large scale energy program?

How much flexibility do we want SSP to have in terms of the amount of power it can deliver, both in terms of extra power and increased speed? How will this extra flexibility affect safety considerations?

6 Environmental and Safety Aspects

Due to overlapping issues concerning both short and long term effects on the environment, these two aspects are combined. They are organized into the following five areas: Living Organisms, Atmosphere, Rectenna, Alternative Energy Sources and Launch Systems.

Living Organisms

What biological effects do microwaves produce on living organisms?

• Human

The main impediments to the implementation of SSP will be based on political and social issues, rather than on technical issues. Since no means of generating power is without risk, many public concerns about power generation have to be taken into account. What will be the effect, if any, on the health of the general public? The experiments performed so far have not convinced the public that energy beaming, especially using microwaves, has no effect on health. There are still questions about the effects on many areas of health, including the nervous system, bone and tissues, internal organs, reproduction and circulatory system. Long term effects, such as those on cancer and DNA, must also be investigated. The problem of possible dangerous effects of microwaves on living organisms, in particular on the human body, represents the main reason for public skepticism.

The recognized problem arising from over-exposure to microwaves is due to heating (thermal effects). It still is uncertain whether non-thermal effects exist. If this is the case, it would mean that even low power levels could be dangerous. Hence, the area of international standards for microwave dosage must be addressed. What are these standards? Are present regulations sufficient? These standards must be continually monitored so that, as more information on the effect of microwaves becomes available, the regulations can be modified accordingly.

It is the duty of the SSP group to achieve public acceptance by educating the people to these issues. The main emphasis should be on informing the people how solar power can be a future energy source, by showing and proving its convenience and safety. This has to be done on a scientific basis and not only for commercial convenience.

Others

What are the effects of microwave power beaming on non-human organisms (plants and animals)? Can we protect them from harmful exposure?

Protection of plants and animals on Earth should be a consideration during the design of the a SSP. When using SSP, it may be impossible (or impractical) to avoid radiating microwave energy to plants and animals entirely. If this is the case, the possibility of reversing the effects on plants and animals which have been damaged should be investigated.

In the first experiments with space solar power, a ground-based demonstration program or a small satellite may be used to improve the technology and further prove the viability of power beaming. These experiments may be on a small scale, i.e., tens to hundreds of kilowatts. As part of these experiments, some initial experiments of the effect of microwaves on biological systems can also be conducted. Several microwave frequencies based on the Industrial, Scientific and Medical (ISM) organization requirements may be chosen for these tests. During the power transmission tests, subjects, such as plant life as well as fish, birds and other small creatures - with the approval of the Society for the Prevention of Cruelty to Animals [SPCA], could be placed in the beam. These tests would potentially be of short duration and therefore not provide any long-term exposure data. This data could allow at least preliminary estimates of the effects of microwaves on the local plant and animal life.

Atmosphere

What are the interactions between microwaves and the atmosphere?

Continuous beaming of microwaves through the atmosphere may be a part of a small scale demonstration. For a small orbiting satellite, a beam may be on continuously to simplify the spacecraft design. Potential interactions of the microwave beam with the atmosphere should therefore be investigated. These small-scale experiments could address physical reactions (e.g. heating) of and chemical reactions in the atmosphere. Both LEO and GEO place special requirements on the tests. A LEO satellite will pass over many countries. With the satellite at GEO, the same area will be continuously exposed to microwave energy. Effects of the beam sweeping through the air versus having the beam "staring" at one point may be addressed.

For future large scale programs it is necessary to address the same questions as mentioned in the above paragraph. However due to the increased power of the beam, additional effects on the atmosphere could be found which were not discovered in the small scale demonstration. Due to the increased size of the rectennas and the higher power beam the physical and chemical effects could be different. With satellites at GEO, special emphasis needs to be put on local atmospheric heating around the rectennas sites. The impact on global heating should also be considered.

Rectenna

What effects will the rectenna have on the local environment?

It is particularly important to pay attention to the environmental effects of the rectenna placement because this will be the more evident, tangible manifestation, for the general public of the SSP demonstration program. The way the environmental concerns will be taken into account will largely determine public acceptance of Space Solar Energy. It is important to involve and to inform the public with the environmental studies. Neighborhoods will be deeply concerned about the impact of such a large project on their health and on their quality of life.

In space solar beamed power systems, the rectenna receives microwaves from the satellite and converts them into electricity. Rectennas could cover a large ground area, and their environmental impact would not be negligible.

In addition, the receiving plant will need additdonal facilities like buildings for administration, maintenance or public information / relation, protective fences, power lines, access roads, whose impact will also have to be assessed.

Many environmental impacts have to be studied:

• Effects of the placement of the rectenna on the local flora and fauna;

• Impact on the marine enviromnent if the rectenna are installed on an ardficial island;

• Polludon: chemical degradadon of the rectenna;

• Hydrology: effects on the underground water reserves;

• Geology: modificadon of soils (preparadon of the area for the construcdon and access roads), erosion due to the depletion of the flora under the rectenna;

• Visual impact / aesthetic of the landscape (power lines and rectenna facilides);

• Sociological effects on the local population: expropriadon of dwellers, induced econornical changes on the local life; aand

• Recycling of the area at the end of the SSP exploitadon;

Also, control of the test beam and devising ways to shut it off, if it gets out of control, are needed to enhance safety.

SSP versus other Energy Sources

What are the differences with respect to environmental effects between SSP and other energy technologies?

The difference relating to the environment between SSP and present energy sources should be studied. The choice of SSP over other energy sources should be based on convincing arguments. We must gather information on the environmental effects of both current energy sources and those under development, including SSP. In the case of SSP, we must begin the analysis with simulations and preliminary experiments. This experiment will allow the SSP system to be more accurately assessed and compared with other systems. The environmental effects of each of the technologies can then be analyzed and an informed decision can be made regarding space solar power.

Launch Systems

What are the environmental effects of the feasibility demonstrations and/or small commercial test launches ?

In the development of a space power satellite, the impact of the launch vehicle on the environment must be considered. The number and size of transportation systems, the impact point(s) of their stage(s), debris created in orbit, and their engine's exhaust products will all have to be assessed. These environmental factors are important in regard to two key areas, mainly public preception of the environmental impacts of an energy source that supposedly is clean and the evaluation of the actual environrnental impacts of a future large scale commercial project.

Even though the absolute magnitude of environmental impacts due to a feasibility~demonstration might be quite small, it is still very important from a public relations point of view to minimize the detrimental environmental impact in order to allay public concerns.

The effects of engine exhausts on the ozone layer must also be taken into account. This will require - analyzing the mechanism of ozone depletion and how the exhaust from SSP launch systems rnight contribute to this phenomenon.

Appendix J

Electric Propulsion Demo With Power Beaming for Orbital Transfer or Lunar Transfer Vehicle

Preliminary Idea

The use of power beaming for space transportation can be considered for electric propulsion systems.

The reason power beaming is of interest to these class of vehicles is that the power-generation system of electric propulsion vehicles is very massive: from 5 to 40 kg/kW. For a 1-megawatt (MW) transfer vehicle, this mass will be 5,000 to 40,000 kg. A rectenna for microwave power reception will have a mass of 1 to 2 kg/kW (1000 to 2000 kg for the 1 MW system). The power level of 1 MW was chosen based on the analysis of References 1 and 2. This power level is only preliminary and is subject to a new optimization based on power beaming technology. Smaller vehicles with only a 10- to 100-kW power level are also potential candidates for this demonstration.

Because the power source is a major part of the total vehicle mass, using beamed energy may be a way of improving the performance of a high-power electric-propulsion system. The improvement lies in removing a large fraction of the mass of the power system from the transfer vehicle and thereby reducing its mass and the total propellant required to perform the mission. Reducing the propellant mass improves the acceleration of the vehicle and reduces the trip time.

Launcher

The launcher may be a Atlas, Titan or Ariane for the $800M demonstration. This test would use a power level from 10- to 100-kW. A Space Shuttle-class vehicle for the Task C demo is needed; a 100-kW to 1-MW power level will fit into the cost constraint.

Power Technology

The planned power technology is solar photovoltaic arrays.

Customer / User

The customer for the system would be SSPP. The other customers would be satellite transportation companies or other government agencies (NASA, ESA, NASDA, Russia) that need space vehicle transportation.

Spacecraft Concept

References 1 and 2 provide the description of the vehicle concepts. The electric transfer vehicle has ion propulsion for very high specific impulse and is the most-attractive option of electric propulsion for this system. It would use an inert gas propellant, such as xenon, krypton or argon. The vehicle concept is a low-thrust design with a flexible structure. The primary components are the power rectenna, the electric thrusters, the power conditioning and distribution, the propellant feed system and the other subsystems (attitude control, navigation, command and data handling).

Organization

Probably a government program for this demo.

This is purely a space-based system so the environmental impact on Earth is negligible.

Orbit

The orbit is either low Earth orbit (LEO) to lunar orbit or LEO to some intermediate orbit. The purpose is to prove power beaming technology rather than having a specific orbit in mind. Thus, any orbit in between LEO and the Moon is acceptable.

Time Scale and Deadline

The time scale for this demonstration is not fixed but for the $800M demo, the 10-year limit must be met. Based on past studies and the available technology, ion propulsion and all other technologies can be available for this demo within the $800M limit.

Cost Target for Early Demos

Costing for task C will require a definition of the trajectory and the test duration. For a lunar mission with a 100-W power level, the cost may be up to $500M.

References

1) Palaszewski, B. "Lunar Transfer Vehicle Design Issues Using Electric Propulsion Systems", AIAA Paper, Monterey, California, July 1989

2) Palaszewski, B., Electric Propulsion for Lunar Exploration and Lunar base Development", LBS-005, Houston, Texas, April 1988.

Appendix K

Low-Cost Launch Technology Demo For Earth to Orbit Propulsion

Preliminary Idea

There is a critical need for lower cost space transportation. The Earth to Orbit segment of SSPP (or any other space endeavor) can be up to 45 percent of the total system cost.

Launcher

The launcher may be a derivative of a pressure-fed booster/launch vehicle design from the Bob Truax/Air Force studies of the early 1960's (with the Space Technology Laboratories [STL]) , the planned Strategic Defense Command (SDI)/U.S. Air Force Delta Clipper and the current U.S. Navy SEA-LAunched Rocket (SEALAR) for the $800M demonstration. This demo could also be included in Task C.

Customer / User

The customer for the system would be SSPP. The other customers would be satellite transportation companies or other government agencies (NASA, ESA, NASDA, Russia) that need space vehicle transportation.

Spacecraft concept

The ideas for low-cost transportation have been documented in the references. A prime candidate for the low-cost system is a pressure-fed booster. This system uses a low-pressure rocket engine that has minimal maintenance requirements and can theoretically launch large payloads to orbit with minimal refurbishment (repair). This type of propulsion system has fewer parts and is fabricated from simpler materials than the typical aerospace vehicle. A proposed booster would have heavier components that do not require the stringent technical tolerances that are needed on existing vehicles. The lower-technology materials would require larger rocket engines to get into orbit, but the overall effect would be a reduction of cost.

Organization

Probably a government program for this demo.

References

1) Bob Truax/Air Force studies of the early 1960's (with the Space Technology Laboratories [STL])

2) Strategic Defense Command (SDI)/U.S. Air Force Delta Clipper

3) Bob Truax, U.S. Navy SEA-LAunched Rocket (SEALAR)

4) Palaszewski, B. "Launch Vehicle Performance Using Metallized Propellants", AIAA Paper, Sacramento, California, July 1991.

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