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SPACE SOLAR POWER CONCEPTS: DEMONSTRATIONS TO PILOT PLANTS

Connie Carrington, Ph.D.
Flight Projects Directorate
NASA Marshall Space Flight Center
Huntsville. AL USA
Connie.Carrington@msfc.nasa.gov

Harvey Feingold, Ph.D.
Science Applications International Corporation
Schaumburg, IL USA
Harvey.Feingold@saic.com

ABSTRACT

The availability of abundant, affordable power where needed is a key to the future of exploration and development of space, as well as to the future economic growth of communities on Earth. One untapped energy resource uses wireless power transmission (WPT) from space-based facilities that collect the sun's energy and beam it where it is needed. Both microwaves and lasers can be used to transmit power. Our studies have concentrated on the use of microwaves, but laser WPT has recently generated increasing interest, as a promising alternative that could overcome some of the major challenges of microwave WPT. This paper discusses the variety of space-based concepts for collecting the sun's energy and transmitting it to either the Earth or to orbiting spacecraft, organized in categories ranging from near-term demonstrations to a full-size pilot power plant in space.

INTRODUCTION

Abundant, affordable power, at locations where it is needed, is a key ingredient in the future of the exploration and development of space. The same can be said for terrestrial futures that support economic development and prosperity on Earth. Both Earth and space development will also benefit if their power sources are clean and renewable. Affordable space systems need self-sustaining power sources that do not depend on re-supply launches, and countries that are deficient in coal and oil supplies need to be independent of imports from other countries. Space solar power can address the energy needs of both space and terrestrial development.

SSP SYSTEMS AND CONCEPT ORGANIZATION

NASA has been taking a new look at collecting solar energy in space and transmitting it to Earth, to planetary surfaces, and to spacecraft. A variety of innovative concepts have been studied for the space segment component of solar power beaming. These concepts have been organized into five "model system categories" (MSCs), primarily on the basis of increasing power levels and projected dates for technology readiness. Planning schedules include down-selection of concepts and their technologies for flight demonstration at roughly five-to-seven-year increments, contingent on funding.

The first free-flying demonstration is MSC-1, at approximately 100kW, with a technology readiness level of 7 in 2007-2008; it will demonstrate WPT, advanced solar power generation (SPG), and advanced power management and distribution (PMAD). MSC-2 is also a 100kW system, but will operate on a planetary or lunar surface, to demonstrate advanced technologies that cannot be adequately tested in terrestrial or Earth-orbiting environments, and to further advance science and exploration goals. MSC-3 will provide a significant jump in power to approximately 10MW, with technologies ready in 2015-2017, and will demonstrate solar electric propulsion (SEP) in addition to advanced WPT, SPG, and PMAD. MSC-4 represents the first SSP pilot plant at approximately 1-2GW delivered to the ground, in a time period after

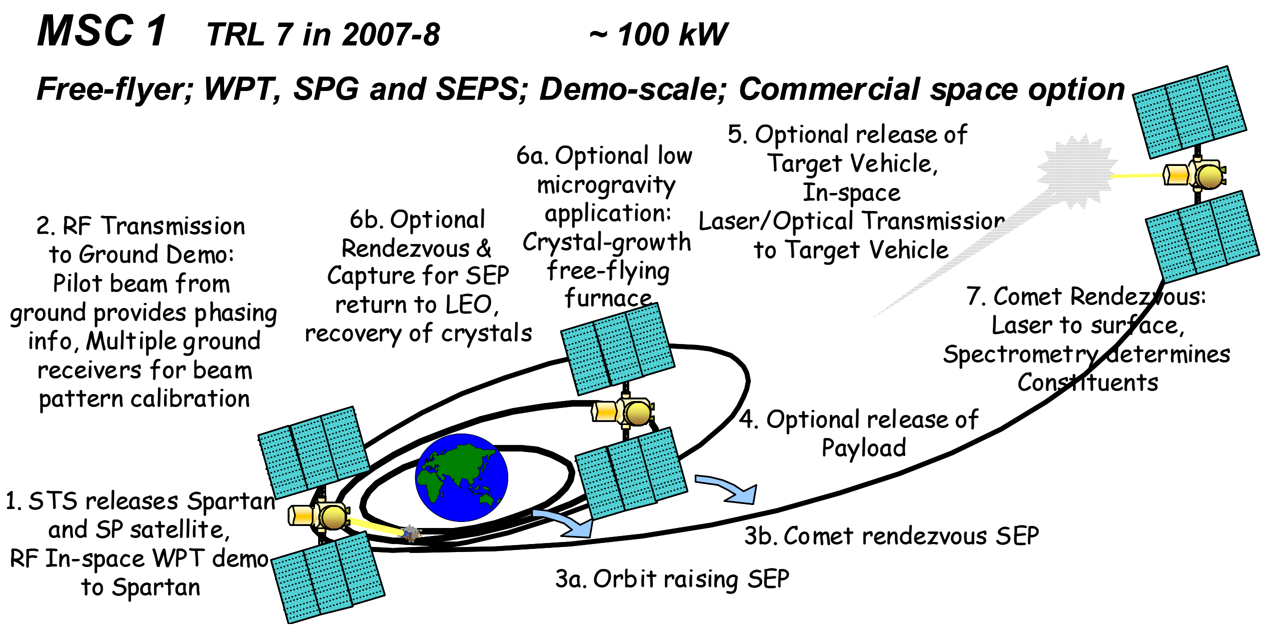


Fig. 1 MSC-1 Candidate Mission Scenarios

2025. A fifth category, at the 10-100GW power level suitable for an interstellar power station, will not be addressed in this paper. Within each MSC several concepts emerge, each having salient benefits and weaknesses. An attempt was made to develop within each concept class an evolutionary series of designs that grow from the MSC-1 demonstration stage to the MSC-4 pilot plant.

NASA organized the SSP work breakdown structure into 14 teams, including research and technology groups defined by technical discipline, applications, market, and environmental groups, and a systems integration working group (SIWG). Faced with defining and analyzing a matrix of concepts, the SIWG first focused on MSC-4 concepts and the delivery of power to the grid on Earth. Hence the pilot plant MSC-4 concepts and their technologies will be discussed first. Concepts for lower-power, near-term MSCs will then be presented.

At several SSP workshops, both the systems and applications working groups developed options for mission scenarios that could be enabled by the power platforms in the MSCs. These options are depicted in Figs. 1 and 2. For the near-term demo in MSC-1, one possible missions scenario is STS-deployment of the small retrievable freeflyer Spartan into the same orbit as the 100kW solar power satellite. Spartan would be equipped with a small microwave rectenna or a PV array for laser power beaming, and the solar power satellite would beam power to Spartan to charge its batteries. The power on Spartan could be used for microgravity experiments, such as crystal growth, and then STS could rendezvous and recover Spartan and its payload for return to Earth. Another potential mission for MSC-1 is RF transmission to a suitable ground site. Due to the relatively modest power levels provided by MSC-1, received power levels on the ground would be low, but would provide an opportunity to perform pattern calibration experiments and to demonstrate ground-based pilot beam functionality. The 100kW power levels of MSC-1 would also be suitable for SEP demonstrations, such as direct drive of Hall thrusters. A science mission scenario could entail SEP transfer of MSC-1 to a comet rendezvous, where the power-rich platform could power remote laser spectrometry to determine comet constituents.

MSC-2's objective is to utilize SSP-technologies for lunar exploration. Boeing has completed a study of lunar polar applications to explore totally unlit polar craters for ancient ice, which may provide clues to origins of the universe, and which could be used for LOX/LH2 propellants. A lunar lander would touch down near a crater of interest on a tall mountain at the pole, which is almost continuously illuminated, where a large solar array would be deployed. The lander then would deploy a rover, whose batteries are charged by remote power beaming from the lander. The rover would explore up to 150km inside the dark interiors of the crater, receiving power as needed from the illuminated lander. In addition to the lunar polar applications study, Boeing and several universities have built and tested several small rovers that have been remotely powered both by microwave and laser beaming.

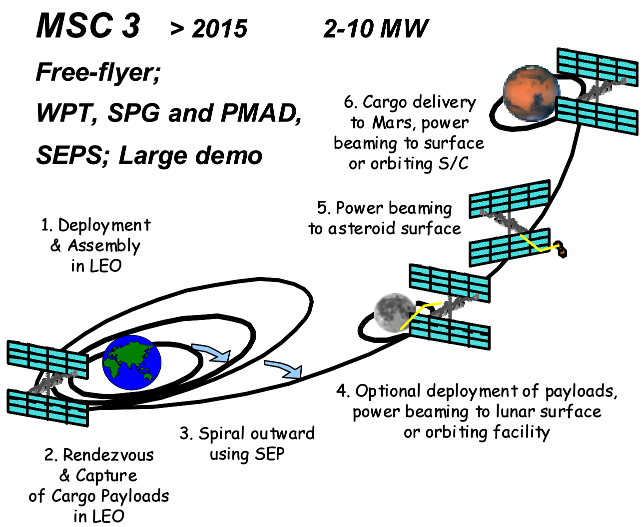


Fig. 2. MSC-3 Potential Mission Scenarios

The 2-10MW of power for MSC-3 would enable exploration scenarios such as power-beaming to lunar or Mars surfaces, or to operations on an asteroid surface, as depicted in Fig. 2. Due to the size of the solar power generation surfaces, MSC-3 would need to be assembled in LEO. With these power levels, it would be capable of providing enough power to transport cargo from LEO out to exploration destinations.

<u>MSC-4 Microwave Concepts</u>

Three primary concept classes have been developed and analyzed for microwave power beaming to Earth: sun-tower configurations, abacus reflector configurations, and the integrated symmetrical concentrator (ISC). All of these concepts are in geosynchronous orbit, so that they dwell for 24 hours over a given receiving site on Earth. Earlier studies performed trades with LEO and MEO satellites, and determined that GEO orbits provided the necessary delivery times to sites, and simplified satellite maneuvers. All of the microwave concepts must accommodate a single, monolithic, massive phased-array transmitter that faces Earth. Spatially distributed microwave transmitters would produce grating lobes at unacceptable levels of radiation on the ground that are located hundreds of kilometers away from the desired receiving site.

Mass and size estimates for all three primary MSC-4 concepts were based on 1.2GW of power delivered to the ground. Configuration comparisons, economic analyses, and sensitivity studies were performed for these concepts, with various technologies that were compatible with their configurations. Size and mass data presented here represent the minimum mass concepts, for technology configurations that presented the best case. Technology performance has been extrapolated to expected performance in the 2020 time frame.

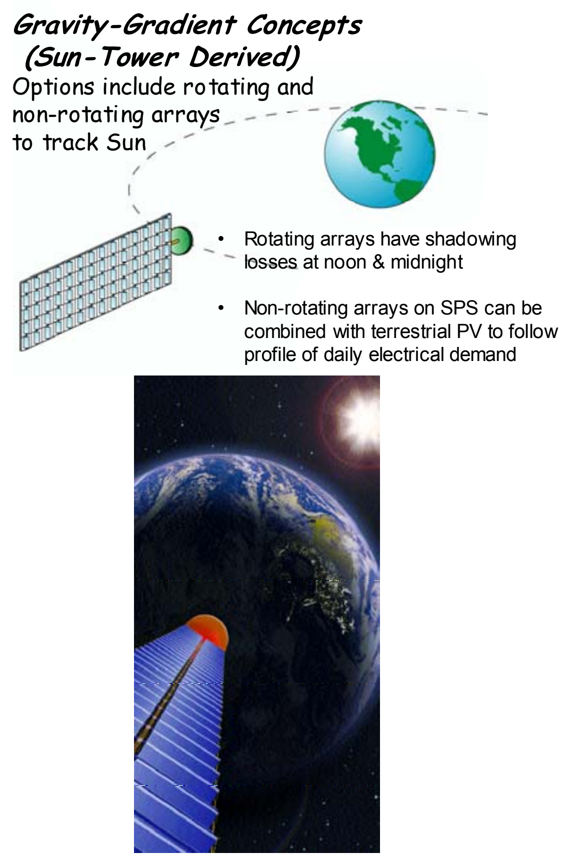


Fig. 3. Sun-Tower and Gravity-Gradient Configurations

Sun-towers and other gravity-gradient configurations as shown in Fig. 3 are characterized by solar array panels aligned along a vertical backbone that collect and conduct power down to the microwave transmitter facing Earth. All power must be conducted by a massive cabling system through the backbone, through a single rotary joint between the solar array assembly and the transmitter, and then into an array of power convertors on the back of the transmitting array. Hence the masses of Sun-towers and their derivative configurations are dominated by the masses of the PMAD and power transmission subsystems.

Another characteristic of sun-towers and gravity-gradient configurations is self-shadowing of the arrays at solar noon and midnight. Trades included rotating and non-rotating arrays. The spacing required to minimize shadowing for rotating arrays increases the solar array structure length and mass, but this needs to be compared to the larger number of closely-packed non-rotating arrays, which have to compensate for longer shadow times. An additional consideration is that power provided by the non-rotating arrays could be augmented on the ground by power from terrestrial PV arrays to match a given location's profile of daily electrical demand.

One of the lowest mass sun-towers is a 20 km tethered-backbone configuration using magnetrons in a 500m diameter transmitter, stretched-lens arrays, and AC power cabling. Approximately 460 pairs of rotating rectangular 40m x 200m arrays are arranged up the tethered backbone, and have been sized to provide the 3.4GW of required power to the transmitter. Array voltage is modeled at 1000V, with a bus voltage of 100kV and transmitter magnetron voltage at 6kV, so converters are required at both the arrays and the transmitter. The initial mass in LEO of this sun-tower is approximately 22,300 MT.

Other MSC-4 concepts were developed to eliminate the self-shadowing problems of the gravity gradient configurations. These new concepts reverted to the perpendicular-to-orbit-plane orientation of the Reference SPS Concept used in the 1970's study of space solar power, but sought to address concerns about the single-point-of-failure of the power-conducting rotary joint between the solar array assembly and the transmitter that was inherent to that earlier concept. One newly proposed configuration replaces the single joint by multiple joints between solar arrays mounted on spars that rotate about a non-rotating backbone mast mounted on the transmitter. Another concept eliminates power-conducting joints entirely by designing the solar arrays and transmitter as one unit and rotating a large but relatively-lightweight microwave reflector in front of the transmitter to direct the power to Earth. This concept is depicted in Fig. 4 and has been named the abacus reflector.

The configuration shown on the left in Fig. 4 is a "kite" arrangement of solar arrays in the abacus frame that minimizes interference between the array edges and the transmitted energy beam. Eleven bays contain sets of the 40m x 200m concentrator arrays, supported by a 3D structure with masts and stays. Alternative abacus configurations are rectangular, as shown on the right, with a prismatic truss structure to provide a reasonable aspect ratio. Fig. 4 also shows a small diameter reflector bearing that could be launched as a unit; earlier configurations had a large diameter rollring around the outside of the transmitter to rotate the reflector. A yoke reflector mount is shown. A single centrally-located mast support for the reflector, with a joint at the reflector center, is an attractive mount option, but has not been pursued due to beam interference and blockage concerns.

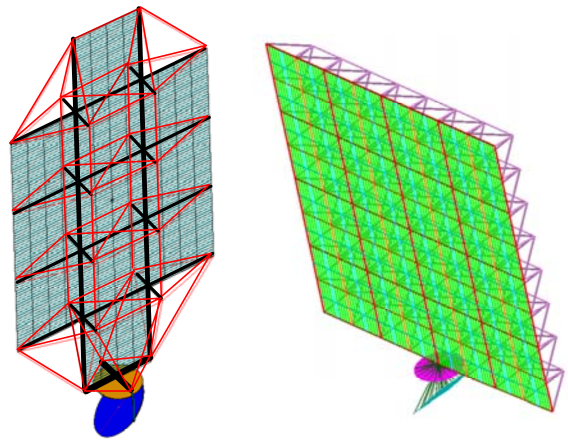


Fig. 4 Abacus Reflector Configurations

Advantages of this concept include continuous anti-sun viewing for radiators behind the array, and an abacus structural frame for the solar wing that accommodates PMAD cabling. The use of the stretched lens array concentrator with a shifting lens to provide seasonal beta-tracking eliminates rotational joints between the cells and the abacus frame, simplifying solar array installation and maintenance. The abacus frame could also be utilized to support other solar collection technologies, such as solar dynamic units. Disadvantages of the reflector approach are primarily challenges to technology, including the assembly or deployment of a large 500mx750m reflector with a surface precision of $\lambda/20$-$\lambda/40$, management of the reflector temperature and thermal distortions, and a stable reflector mount that can meet beam pointing requirements. In addition, the abacus reflector concepts are still plagued by massive power transmission and PMAD systems, and tend to have the most massive structures of all concepts studied.

Lowest mass configurations for abacus reflector concepts include a Brayton solar dynamic system and stretched lens arrays. Both low mass configurations use magnetrons in the transmitter and AC cabling. For the stretched lens array configuration, 962 40m x 200m arrays are needed, and for the Brayton configuration, 1380 units are required. Converters are not needed between the 100kV power distribution system and the 100kV Brayton units, but this configuration still requires converters at the 6kV magnetrons. Since the Brayton concentrators are 100m in diameter, the abacus structure is approximately 4.6km x 3km. The stretched lens configuration, while requiring converters at both the arrays and the magnetrons, requires a much smaller abacus structure, approximately 2.8km x 2.8 km. The initial mass in LEO for the Brayton configuration is approximately 27,000MT, and is 28,000MT for the stretched lens cofiguration. Both are significantly more massive than the sun-tower configurations.

The Integrated Symmetrical Concentrator (ISC) concept in Fig. 5 was proposed to reduce PMAD and cabling masses, as well as structural mass. Incoming sunlight is collected in two large "clamshells" located on the ends of a mast aligned along the orbit normal, reflecting solar energy onto two centrally-located photovoltaic arrays. The energy is converted into electrical power and transmitted by relatively short cables to the transmitting array. The clamshells face the sun, and the mast, PV arrays, and transmitter rotate as a unit to point the transmitter to a receiving site on Earth. The mast is aligned with the orbit normal. A non-power-conducting joint between the clamshells and the mast provide rotational capability for the once-a-day orbital tracking, and the seasonal beta tilt.

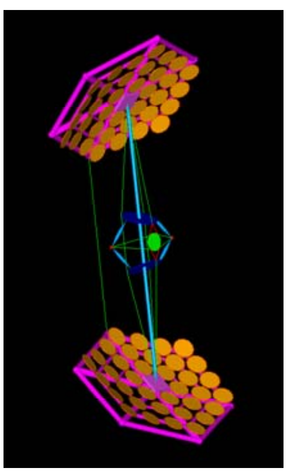


Fig. 5 ISC

Two clamshell configurations were developed: a 24-mirror version, which provides a 2-to-1 concentration ratio on the solar arrays, and a 36-mirror version, which provides a 4-to-1 concentration ratio. Each mirror is planar, approximately 500m in diameter, and is mounted to the back-plane structure at a slightly different angle to form a segmented clamshell primary mirror. Since the ISC is not an optical imaging assembly, the light reflected from each mirror only needs to fall somewhere on the PV array, with a goal of minimizing solar array hot spots. The mast length is sized so that the focal length of the mirrors is greater than 10km, which provides a reasonable spot size for the sun's image on the PV arrays. With this focal length and a local surface-flatness requirement of about 0.5 degrees on the mirrors, hot spots and excessive light spillage around the PV arrays are minimized. Mirrors on the outer edges of the clamshell, which could experience larger deflections than those located interior to the clamshell, will reflect their energy on interior regions of the PV array to reduce spillage. The mirror reflectivity has been modeled at 0.9, and light spillage has been estimated at 10%. Each mirror is 0.5 mill thick Kapton supported by a circumferential inflatable toroidal ring and an inflatable back-plane structure.

An initial ISC concept placed the solar arrays on the back of the transmitter, to minimize power cabling distances. However, the backs of both the solar array and the transmitter need to radiate heat, and thermal radiation estimates of a back-to-back configuration are 90 kW/m^2. Hence the ISC configurations presented here have two separated solar arrays that are each canted 10 degrees.

Since ISC is a concentrating optical system, solar array temperatures have been a concern. The only array technology that holds promise of efficient operation at high temperatures is quantum dot, which was used in all ISC solar conversion models. To roughly size the solar arrays, a top-level array temperature estimate was made, using both 1km and 1.5km array diameters, and a worst-case seasonal sun angle of 23.5^o.

Several mass estimates for the mast have been made, ranging from 170 MT for conventional composite trusses to 50 MT for inflatable trussed masts with stays. System models are using the inflatable mast mass estimates.

The lowest mass ISC estimate is a high concentration configuration using magnetrons, quantum dot solar arrays, and AC power. The 36 mirrors are 470m in diameter, mounted on clamshells that are approximately 4km in diameter. The mast is 7.2km long, and the solar arrays are 1070m in diameter. The initial mass in LEO for this configuration is 18,000 MT.

Changing to a low-concentration ratio configuration, but maintaining the magnetrons, quantum dots, and AC, produces a 24-mirror clamshell, with 573m diameter mirrors. The mast is 7.6km long. Unfortunately the solar array is now 1770m in diameter, and with the additional mass of these arrays and their structure, the mass of the low-concentration ISC is 31,500MT in LEO. Hence 2-to-1 concentration ratios are not competitive with the other MSC-4 configurations.

MSC-4 LASER CONCEPTS

Laser technology advancements in recent years, and frequency allocation and power density issues of microwave WPT systems, prompted consideration of laser WPT options. Their ability to transmit power from multiple apertures distributed over a satellite's structure or flown on separate small satellites is a distinct advantage over the monolithic single transmitters required for microwave WPT. Lasers enable modular design approaches, in which localized SPG, thermal management, and PMAD systems can be sized and built for individual lasers. Systems can be tailored for particular markets, and then expanded by adding modules as the power demand increases. Beam pointing flexibility allows redirection to other receiving sites as weather or power demands change. In addition, the power receivers are PV arrays, which can be used not only for laser power reception, but also for conventional sunlight conversion.

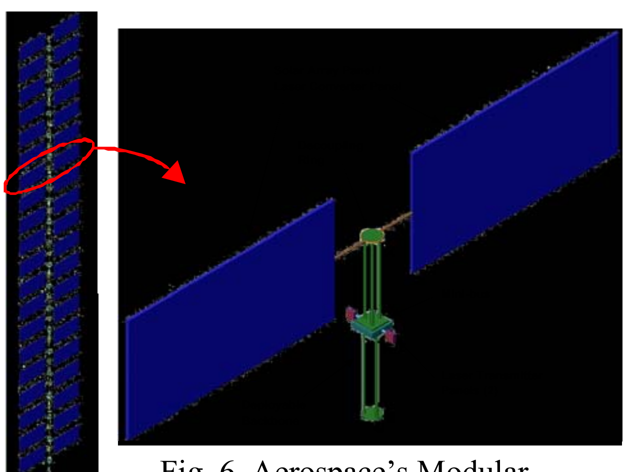


Fig. 6. Aerospace's Modular 5MW Laser WPT Satellite

Both Aerospace Corp. and Boeing have developed MSC-4 concepts that take advantage of the benefits provided by laser systems. Aerospace sized their satellite for a single 40MT launch with 20 laser modules built on a common backbone and providing 5MW. An optional configuration is a 20MT satellite of 10 laser modules, providing 2.5MW. Each module has two laser emitters and two solar arrays. Aerospace arranged their satellites in Halo orbits, which are satellite constellations that appear to fly in concentric rings when observed from the receiving site on Earth. Separation of the satellites can ensure that individual beams from satellites will meet laser skin and eye safety requirements, with intensities of 1-3 suns. The individual lasers use optics in the 20-25 cm range, which reduces thermal management challenges and reduces the mass and cost of the optics. Innovative designs have been proposed, with laser diodes directly mounted on the solar panels, so that radiators can be shared. The modules also house two laser transmitter panels, a portion of a deployable backbone, and a mini-bus containing spacecraft subsystems, including propulsion. The concept is shown in Fig. 6. For 1.2GW delivered to the ground, a constellation of 480 satellites is required, which will fly in a 6000-km Halo orbit.

Boeing has also designed a modular laser power satellite as a gravity-gradient stabilized ribbon-like array assembled from modular units, aligning their array panels edge-to-edge. This non-rotating array configuration will not receive solar energy at noon and midnight, but its PV receiver on the ground will be powered by natural sunlight during noon, providing power to supplement that lost from the power satellite. The ISS-sized panels are 260m x 36m. to provide 1.2GW to the grid; perhaps as many as 1530 panels will be needed, forming a 5.5km long sun-tower configuration, as shown in Fig. 7.

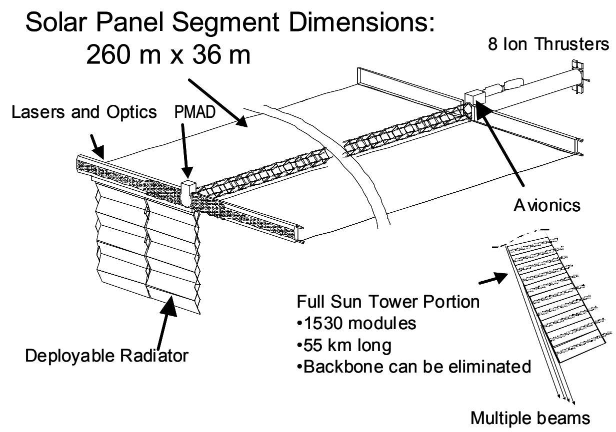


Fig. 7 Boeing's Modular Laser Unit.

MSC-3 CONCEPTS

MSC-3's charter is dual use of SSP technologies for exploration. MSC-3 is a 10MW class vehicle, and will utilize SEP in addition to the advanced SPG and PMAD technologies of power beaming applications.

Boeing has examined applications of SSP transportation systems to missions beyond GEO, including an SEP human Mars mission, an SEP lunar transfer vehicle, and a crewless precursor "power explorer."

A modest human Mars mission was developed that utilizes conjunction class trajectories and cargo pre-emplacement strategies for use in multiple visits to Mars. SEP performance was based on direct-drive Hall thrusters using an Isp of 2000s and a specific power of 3.4 kg/kW. Payload elements were separated into 3 components, each utilizing its own SEP Mars transfer vehicle (MTV). The first element is a cargo MTV with a 58MT cargo lander payload. The second element is another unmanned cargo MTV carrying a 58MT Mars ascent/descent stage. The third element is a piloted transfer vehicle consisting of a 40MT crew-transfer habitat and 6MT Earth-return capsule that can rendezvous with the second MTV in Mars orbit and return to Earth. The SEP Mars mission chosen for evaluation was a 2018 low-thrust Earth-Mars heliocentric trajectory with a 255-day outbound time, a 600-day surface stay time, and a 156-day inbound trip time. Reusable and expendable SEP vehicles were assessed, with propellant for the reusable vehicle totaling about twice that of the expendable, due to the end-of-mission Earth return propellant carried during the entire flight of the reusable vehicle. The total initial mass in LEO for the piloted reusable SEP-MTV is 501MT, and 268MT for the expendable piloted MTV. A sketch of the piloted SEP MTV (with array sizes not to scale) is shown in Fig. 8. The cargo MTVs are 292MT for the reusable vehicle and 185MT for the expendable. Power requirements are 1.70MW for the reusable vehicle and 1.07MW for the expendable.

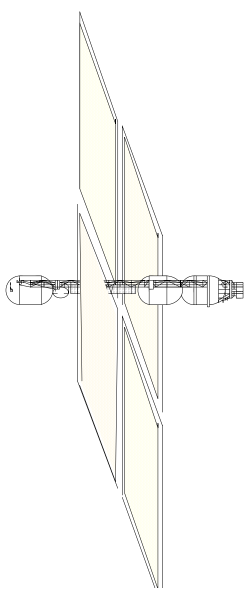


Fig. 8 Boeing Concept for Piloted SEP Mars Transfer Vehicle

SAIC has also examined a variety of SEP exploration scenarios using SSP-derived technologies, for mission scenarios from LEO to GEO, LEO to Earth-Moon L1, LEO to Earth-Sun L2, LEO to Mars, and round trip Mars missions. One example of these missions is a fast round trip to Mars using stretched lens arrays and 100 kW Hall thrusters. The Earth-Moon L1 point was used for vehicle staging, and the outbound cargo payload included most of the support facilities and propellant. A 4.7MW cargo vehicle with a 278MT payload leaves approximately 2 years and 2 months before the crewed vehicle leaves LEO. The crewed vehicle with a 35MT payload takes 185 days from LEO to Mars aereosynchronous orbit, using 40MW from the stretched lens arrays to power 42 Hall thrusters. In Mars aereosynchronous orbit, the crew vehicle will rendezvous with the crew ferry and its propellant tanks and supplies, which has been put in place earlier by the cargo vehicle. After a high thrust transfer to a 500km Mars orbit, the crewed ferry docks with the cargo vehicle. The crew transfers to the ascent/descent vehicle on the cargo vehicle, descends to the surface for a 3-week stay, and then returns to the cargo vehicle. The crew then transfers back to the crew vehicle for return to Earth. The return trip to the Earth-Moon L1 point is about one year, so that the total crew round-trip time is 1.6 years. The cargo vehicle launched mass is 656MT, including 278MT of payload and 254MT of SEP propellant. The crew vehicle launched mass is 255MT, of which 147MT is SEP propellant and 35 MT is crew, their habitat, and their return vehicle. Abacus vehicle configurations are shown in Figs. 9 and 10.

Fig. 9 SAIC Abacus Cargo Vehicle

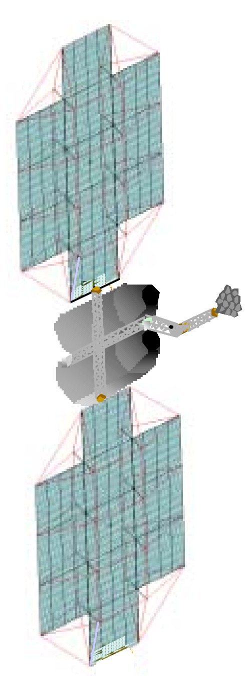


Fig. 10 10MW Abacus MTV

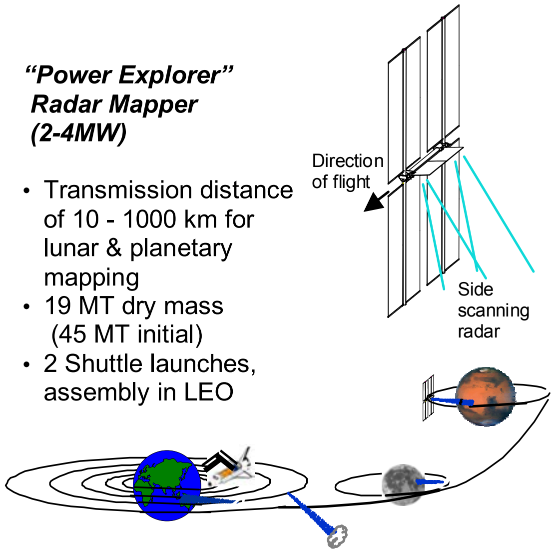


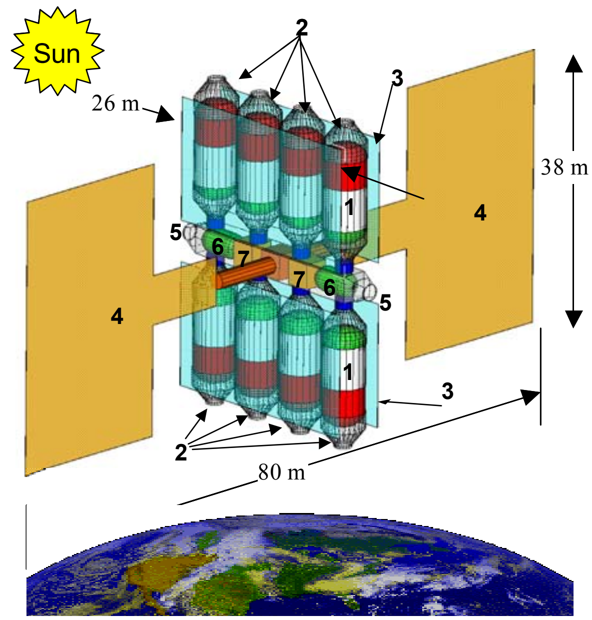
Fig. 11 Boeing MSC-3 "Power Explorer"

Boeing also assessed a crewless SEP precursor vehicle, depicted in Fig. 11. Named the "Power Explorer," it uses four 52m x 31m PV arrays to both drive Hall thrusters for propulsion and to power a radar beam for in-situ planetary prospecting or orbital debris detection. Two identical units, which are docked in orbit, are each deployed from a single launch of a Shuttle or Delta IVH.

MSC-1.5 CONCEPTS

The significant increase in power capability from a 100kW MSC-1 to a 10MW-class MSC-3 prompted the insertion of an intermediate model system category named MSC-1.5, since MSC-2 was already assigned to lunar/planetary applications. MSC-1.5 was categorized as a 200kW to 3MW class demonstration of advanced SSP technologies, in the 2012 time frame. Two concepts in this class are the cryogenic production and storage depot for LEO applications, and Boeing's "Skylight" megawatt-class satellite.

The propellant depot is deployed in a 400 km circular equatorial orbit, where it will receive tanks of water launched from Earth, will convert the water to liquid hydrogen and oxygen, and will store up to 500 MT of cryogenic propellants. The propellant stored in the depot can support transportation from low Earth orbit to geostationary Earth orbit, the Moon, LaGrange points, Mars, etc. The tanks are configured in an in-line gravity-gradient configuration to minimize drag and settle the propellant. Temperatures can be maintained by body-mounted radiators, which will also provide some shielding against orbital debris. 706kW of power is supplied by a pair of stretched lens arrays mounted perpendicular to the orbital plane, which rotate once per orbit to track the Sun. The depot mass is approximately 69 MT without water or LOX/LH2, and is shown in Fig. 12.



1. LOX/LH2 Storage Tank
2. Transfer Vehicle Docking Port
3. Radiators
4. Solar Arrays
5. Water Docking Port
6. Water Storage Tanks
7. Electrolysis System

Fig. 12. MSC-1.5 Propellant Product Depot

Another MSC-1.5 concept developed by Boeing is "Skylight." It is a sun-tower configuration with ten pairs of rotating solar arrays, similarly sized to the ISS arrays but utilizing advanced higher performance SPG technologies. The spacecraft overall length is 170m, and the 2.7MW of power from the arrays could provide approximately 1kW of beamed power using a distributed laser WPT system. Skylight has an estimated mass of 53MT without energy storage. Fig. 13 shows the Skylight concept. Boeing has developed several concepts that utilize advanced ISS-sized arrays, with the goal of eventually replacing the ISS arrays with newer high-performance SPG technologies developed for SSP.

MSC-1 Concepts

This model system category is a near-term 100kW-class space platform whose primary function is to demonstrate SSP-technologies in WPT, SPG, and

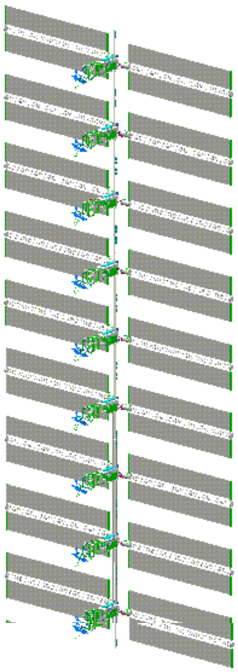


Fig. 13 Boeing's MSC-1/5 "Skylight" Powersat

PMAD, and whose secondary functions are SEP and non-mission-critical advanced technologies. System design mass goals are 5MT for the dry spacecraft, 2MT for technology flight experiments, and 2MT for propellant. Technology performance goals include 100-150kW of on-board power, SPG conversion efficiencies of greater than 30% and at least 250 W/kg, and 300-500V power systems. Demonstration of advanced electric propulsion is also a goal; an example is direct-drive of Hall thrusters. A five-year lifetime, minimal on-orbit assembly, and a single shuttle launch are guidelines.

Boeing has developed several variations of an MSC-1 "power plug in space," for space-to-space power beaming in Earth orbit, and lunar and Mars power spacecraft (LAMP). The power plug was designed to fit into the shuttle cargo bay, and to use the existing ISS array structural design, but outfitted with advanced solar cells. A parabolic WPT reflector is sized at 4.5m in diameter, and the array span is 70.8m x 11.7m. Hall thruster and advanced battery technologies would be demonstrated.

For Earth-based missions, the power plug would be deployed in LEO and then spiral out to a 24-hour period low eccentricity orbit at an inclination of 10^{o} to 14^{o}, where it would remain in sunlight. From this orbit it would be capable of beaming power to GEO satellites in eclipse. For existing GEO satellites, the beamed power would be optical, either at laser frequencies tuned to the receiving satellite's PV arrays, or using high-intensity white incoherent light for multibandgap arrays. Microwave or millimeter power beaming could be an option, but the rectenna on a receiving satellite would most likely be no larger than 10m, which limits beaming to only short distances (<1km for 5.8GHz, or <40km for 245GHz, for ~4% power incident on the rectenna). For laser power transmission, and advanced solar cells on the power plug's ISS-sized arrays, up to 18kW could be transmitted by the beam, over much longer distances.

Boeing has also investigated an orbiting lunar power plug, to provide power or illumination at the Moon. They have developed a charging/thrusting strategy for the transfer orbit that takes about a year, and uses SEP except for trans-lunar injection and lunar capture, where chemical bi-prop is used. The initial mass in LEO is 20MT, of which 2.8MT is SEP propellant and 2.5MT of TLI&LOC propellant, and 260kg is payload. A Mars version of this power plug has also been proposed, and even at the low insolation levels near Mars, up to 17kW may be available through laser power beaming to a Mars infrastructure. The power plug concept for Earth, lunar, and Mars missions is shown in Fig. 14.

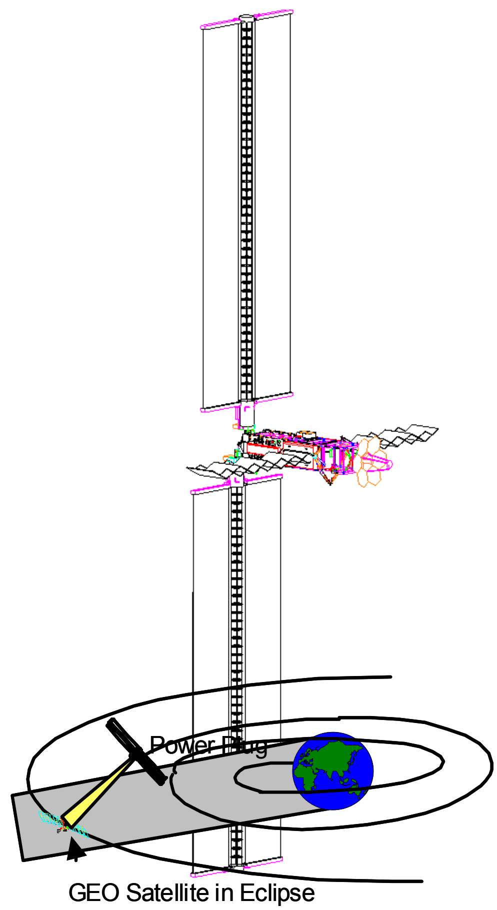


Fig. 14 Boeing's MSC-1 "Power Plug"

Since the MSC-4 ISC is a promising long-term low-mass concept, an ISC-derived 100kW-class concept was investigated. Several small-scale ISC configurations are shown in Fig. 15.

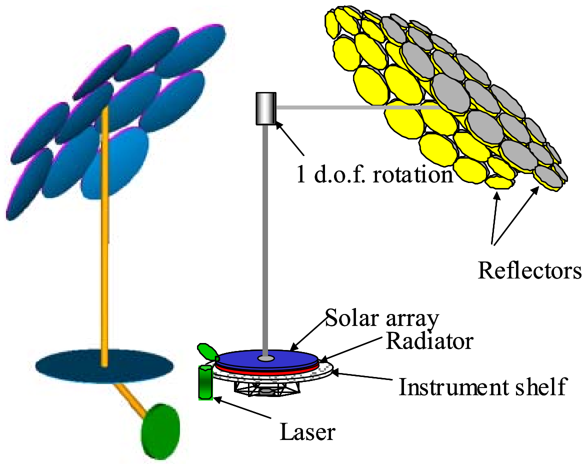


Fig. 15 MSC-1 ISC Configurations

Only one solar array is used, so that the back side of that array will have a better opportunity to radiate heat. An instrument shelf is mounted in a ring around the outside of the solar array and radiator unit, and either a microwave or a laser WPT transmitter can be accommodated

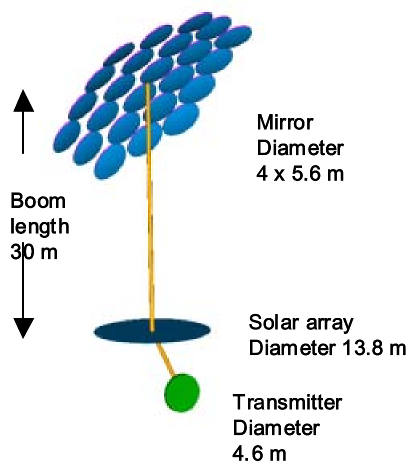


Fig. 16 ISC Configuration of elliptical mirrors with spherical curvature

Uneven and changing illumination on the solar array, from the multiple mirrors of the clamshell, presents difficulties for current and near-term SPG, PMAD, and thermal management technologies. A change from flat to convex mirrors that produce an image illuminating the entire solar array is a solution. Since the clamshells are nominally tilted 45^{o} from the solar array, two options for correcting the elliptical image could be applied: elliptical mirrors with a spherical curvature (Fig. 16), or circular astigmatic mirrors (Fig. 17). Seasonal corrections would also need to be made as the clamshells tilt to track the beta-angle of the sun.

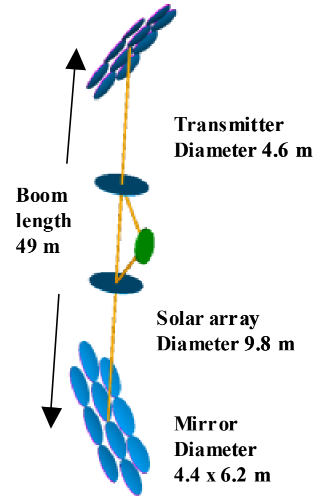


Fig. 17 ISC Configuration with circular astigmatic mirrors

An offset two-clamshell, two-solar array configuration, shown in Fig. 18, has the best chance of rejecting the solar array heat. The transmitter diameter has also been increased to help reduce temperatures, but will now have to be assembled on-orbit.

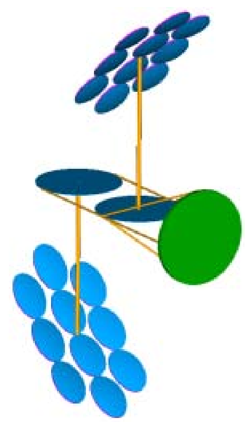


Fig. 18 Offset ISC Configuration

Since these MSC-1 ISC configurations have 2-to-1 or 4-to-1 concentration ratios, they have the same high temperature solar array issues that the MSC-4 ISC must manage. Unfortunately, quantum dots is the only technology that holds the promise of functioning at the temperatures on the MSC-4 ISC solar arrays, and this technology is not at a maturity level to be used for the nearer-term MSC-1 applications.

Other issues for the MSC-1 ISC are the packaging and deployment complexity of a concentrator optic assembly versus conventional solar arrays, close locations of the solar array and transmitter, which increase cooling requirements for both, payload accommodation (which can be done), and propulsion system accommodation. Benefits include solar array views of deep space to aid thermal management, relatively reasonable reflector surface accuracies, the small solar array, short cable distances, no slip joints, and lightweight structures.

A more traditional configuration for MSC-1 is shown in Fig. 19. A standard spacecraft bus, but with a redesigned, higher voltage PMAD system, supports two 5.5m x 30m planar arrays populated with stretched lens arrays. SEP options include two 50kW Hall thrusters, and a $70m^2$ radiator (not shown) to manage up to 90 kW of heat. A preliminary mass estimate of 7MT IMLEO includes 1.4MT of Krypton for transfer to GEO, and 2MT for technology experiments.

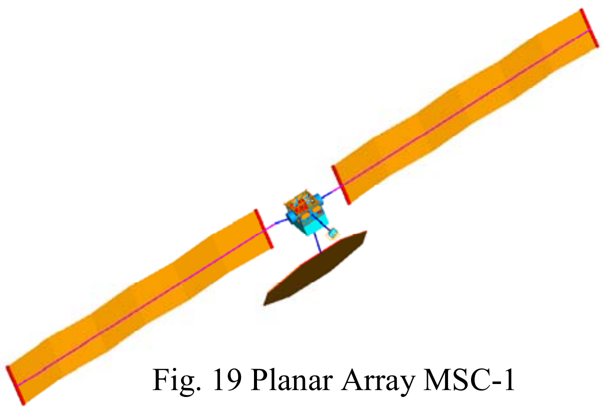


Fig. 19 Planar Array MSC-1

One possible mission scenario using this spacecraft is an ISS laser power beaming experiment, in which a small laser would beam power from ISS to the co-orbiting MSC-1 platform, as shown in Fig. 20. Site 1 of the ISS JEM-EF could provide 5kW of power and thermal management to a laser unit, which would be installed without EVA using the JEM robotic arm. A 12m diameter octagonal PV array on the MSC-1 satellite would receive the beamed power; at a distance of 10-20km in front of ISS, modest laser pointing accuracies of 1 arcmin would be needed. A small amount of power running in a wire around the outer edge of the PV array structure would thermally mark the boundaries of the target for infrared sensors at the JEM-EF laser site, to aid in target centroid detection for beaming acquisition and pointing during power transmission. After the ISS laser beaming experiment is complete, the laser is removed from the EF-1 by the JEM arm and returned through the airlock, where it can later be returned to Earth by the STS. The MSC-1 satellite would then use its on-board arrays to direct drive the Hall thrusters and spiral out and away from the ISS orbit.

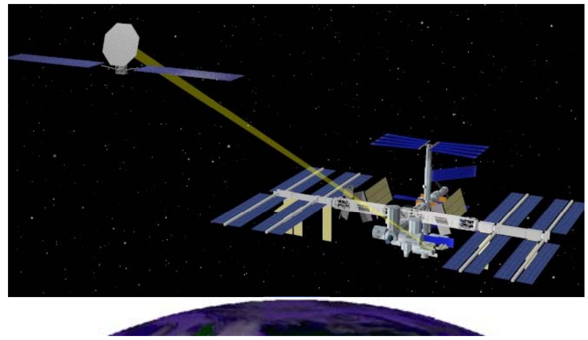


Fig. 20 ISS Laser Power Beaming Experiment

Technologies that could be demonstrated on a power MSC-1 platform include energy storage, innovative thermal management, distributed control of flexible structures, microgravity manufacturing, propellant production and storage, and alternative WPT technologies, such as small-scale ISC concentrator systems. This platform would also provide opportunities to demonstrate advanced materials, deployment of inflatable structures, and space assembly and robotics.

After the WPT demo from ISS, MSC-1 could use its array power to perform experiments in high power communication, space-based radar, or space science. It could also use the array power for WPT experiments with other spacecraft, such as laser annealing of solar arrays, power beaming for energy and control of solar sails, or a lunar/planetary surface demonstration, such as powering a rover. Possible destinations include the Van Allen belts, HEO, GEO, Earth-Moon L1, the moon, Mars, Earth-Sun L2, asteroids, or comets.

Another scenario for a first technology flight demo eliminates the MSC-1 satellite and just demonstrates laser power beaming from ISS to the ground. This experiment could use an existing beaming expander that would fit through the JEM airlock, and could put a reasonably-sized spot on the ground, where a PV array could receive the power. For 5kW at the JEM EF-1 site, received power could be on the order of 0.5-1kW. This demonstration could also perform retro-directive beam experiments.

Other suggestions include collaboration with the DoD, such as using second copies of the Orbital Express satellites to perform space-to-space power beaming and receiving. Collaboration with other governments may also provide suitable satellites.

Using the ISS as a platform for space-to-space power beaming experiments to MSC-1 means that the initial orbit for MSC-1 is 51.6^{o} inclination. Hence the same solar array orientation, attitude, energy storage, propulsion system location, and configuration compromises that ISS has contended with will also effect MSC-1's design. A simpler approach is to deploy MSC-1 in a sun-synchronous orbit, eliminating the need for a second DOF in solar array tracking, greatly simplifying energy storage and thrust vector orientation for propulsion, providing more uniform illumination on the solar arrays, and reducing drag from the orientation of the solar arrays. However, the sun-synchronous orbit option would require a large launch vehicle to place MSC-1 in a high inclination orbit. In a sun-synchronous orbit, WPT demos would need to be from MSC-1 to the ground, or another satellite in this orbit would have to be provided to receive power.

CONCLUSIONS

A great variety of SSP concepts have been developed over the past few years. In each model system category, development of concepts has helped our team identify design issues and the technology advances that might be made to address them. Systems modeling of the concepts has provided insight into which design features and technologies have the greatest impact in reaching the ultimate goal of usable space-based solar power.

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