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Power Transmission by Laser Beam From Lunar-Synchronous Satellite

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Symbols

A	projected area of transmitter dish, m^2
A_r	area of rover radiator, m^2
b	coefficient of rolling resistance of rover
D	diameter of transmission dish, m
d	diameter of Airy disk (receiver diameter), m
E_1	converter efficiency
E_2	reflectivity of transmitter optics
E_3	laser diode efficiency
F	fraction of total transmitted power received
F_m	rover radiator view factor to lunar surface
F_r	rover driving force
F_s	rover radiator view factor to space
$\mathrm{G_y}$	gray, $1\mathrm{G_y} = 100$ rads
J_0	Bessel function (order zero)
J_1	Bessel function (order one)
K	$= \frac{2\pi}{\lambda}$
M_d	mass of transmitter dish, kg
M_e	approximate mass of satellite, excluding photovoltaics and transmitter dish, kg
M_r	mass of reactor
M_T	total satellite mass, kg
M_{th}	mass of thermal management system
M_v	mass of photovoltaic arrays, kg
n	number of neutrons
P	power of prime source, kW
P_r	power required by rover, kW
Q_{rej}	rover heat load
r	distance of first dark ring from center in far-field pattern
r_r	rover wheel radius
$\mathrm{S_v}$	sievert, $1\mathrm{S_v} = 100$ rems
T_m	lunar surface temperature, K
T_{rej}	rover radiator operating temperature, K
T_s	space (sink) temperature, K
W	rover weight, N
z	distance between far-field pattern and transmission aperture
α_r	absorption coefficient of rover radiator

ϵ_m	lunar surface emissivity
ϵ_r	rover radiator emissivity
λ	laser wavelength
σ	Stefan-Boltzmann constant
Acronyms:	
AlGaAs	aluminum gallium arsenide
APSA	advanced photovoltaic solar array
ARC	Ames Research Center
BOL	beginning of life
CMG	control moment gyroscope
CO_2	carbon dioxide
EOL	end of life
EVA	extra vehicular activity
FFP	far-field pattern
GCR	galactic cosmic rays
He^3	rare helium isotope, valuable in nuclear industry
ILO	initial lunar outpost
LDA	laser diode array
LLOX	lunar liquid oxygen
LSS	life support system
L1, L2, L3, L4, L5	Lagrange point designators
N_2	nitrogen
O_2	oxygen
PV	photovoltaic
RTG	radioisotope thermal generator
SPA	solar panel assembly
SPE	solar proton event
SSF	Space Station *Freedom*
TEC	thermoelectric cooler

Abstract

This study addresses the possibility of beaming laser power from synchronous lunar orbits (the L1 and L2 Lagrange points) to a manned long-range lunar rover. The rover and two versions of a satellite system (one powered by a nuclear reactor, the other by photovoltaics) are described in terms of their masses, geometries, power needs, missions, and technological capabilities. Laser beam power is generated by a laser diode array in the satellite and converted to 30 kW of electrical power at the rover. Present technological capabilities, with some extrapolation to near future capabilities, are used in the descriptions. The advantages of the two satellite/rover systems over other such systems and over rovers with onboard power are discussed along with the possibility of enabling other missions.

Introduction

Manned exploration of our nearest neighbors in the solar system is the primary goal of the Space Exploration Initiative (SEI). An integral part of any manned lunar or planetary outpost will be a system for manned excursions by roving vehicles (rovers) over the surface of the Moon or planet. Technology that optimizes such systems is most needed. A system for transmitting power to a rover is such technology. It relieves the rover of the large mass burden inherent to onboard power generation modules.

NASA has studied the possibility of transmitting power in space by laser beam for some time (refs. 1 and 2). During that time several reports (refs. 3 5) have described systems for transmitting power by laser beam from various locations to the lunar surface. They attempt to describe the systems in terms of their capabilities, the technologies required, and their physical compositions. This report augments these studies by describing two versions of a satellite/rover system for transmitting laser power from lunar-synchronous orbits (the L1 and L2 Lagrange points) to a lunar rover.

The two versions of the satellite differ mainly in their source of primary power, one being powered by a nuclear reactor and the other powered by solar cells. Both sources mainly power a laser diode array that transmits its optical output power through large transmission optics to the rover on the lunar surface.

For maximum usefulness, the rover should have an extended range capability, which typically implies considerable onboard power generation or energy storage capability. The state of the art in power generation and/or energy storage devices is not optimal for an extended lunar mission, and enormous onboard mass would be required for any long mission. This problem is compounded by the long lunar day and night, which severely constrain options such as photovoltaic power production and battery storage. Radioisotope thermoelectric generators (RTG's) have been considered for an unmanned low-power (500 W) rover application on Mars (refs. 6 and 7). It is doubtful that the approximately 30-kW requirement for a manned rover could be met with the low specific power of RTG's. At current specific RTG power production of about 5 W/kg, an RTG-powered lunar rover would have a power system mass in excess of 5000 kg. The use of methanol to power a rover has also been proposed recently; a fuel/oxidizer load of 500 kg was expected to provide a 250-km range on Mars and somewhat more on the Moon (ref. 8). Vehicle mass was not clearly stated in the methanol use study, but an assumption of 1 kWh/km travel over Martian terrain determined fuel use.

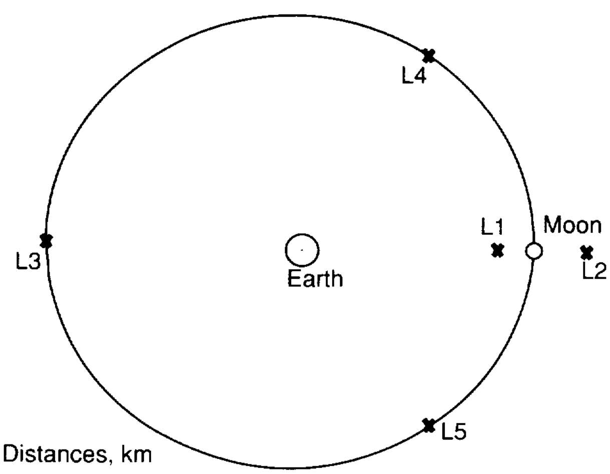
The use of power beamed from lunar satellites to the rover has clear advantages over the conventional methods noted. Not only is there a large mass saving associated with not having to produce power onboard the rover, but additional benefits accrue as well. Abundant power is available to perform a variety of missions, enhancing the capability of the rover beyond that envisioned for an initial lunar base. With the advantage of a lightweight laser-to-electric converter, the rover is highly maneuverable and can operate continuously in either day or night conditions.

The remainder of this study is organized into parts A, B, and C. Part A describes two possible satellite scenarios for power beaming. One scenario uses a nuclear power source, while the other uses a solar photovoltaic power source. Part B describes the laser-powered rover. Part C contains a comprehensive summary of the entire report.

Part A: Satellite Power Station

Locations

In a three-body rotating system composed of the Earth, the Moon, and an infinitesimally small satellite, the motion of the satellite can be described by three equations (ref. 9) in a coordinate system that rotates with the Earth-Moon system. Analysis of these equations reveals that there are five unique points in the coordinate system where a satellite will remain motionless with respect to the Earth and the Moon. (See fig. 1.) These points, called Lagrange points, are places in the Earth-Moon region where the gravitational and centrifugal forces of the rotating Earth-Moon system sum to zero. L1 and L2 are the points nearest the Moon. The distance of L1 above the lunar surface varies between 53 000 and 59 000 km; L2 varies between 59 000 and 66 000 km. (The locations vary, in part, because the lunar orbit is slightly elliptical rather than circular as the referenced analysis assumes.)



Distances, km

Moon orbit radius	384 403
L1 to Moon center	57 835
L2 to Moon center	64 346
Earth radius	6 371
Moon radius	1 738

Figure 1. Lagrange points.

A satellite placed at L1 with the correct initial velocity would tend to remain in a relatively force-free region of space. Only occasional small corrective thrusts would be required to maintain an almost stationary position above the Moon. Figure 2 shows the fraction of lunar surface viewed as a function of the distance above the lunar surface. At the L1 position about 49 percent of the surface is visible at all times. The area of the Moon viewed changes very little. Given thrust, the satellite could transit to L2 on the far side of the Moon to give similar power beaming capabilities from there.

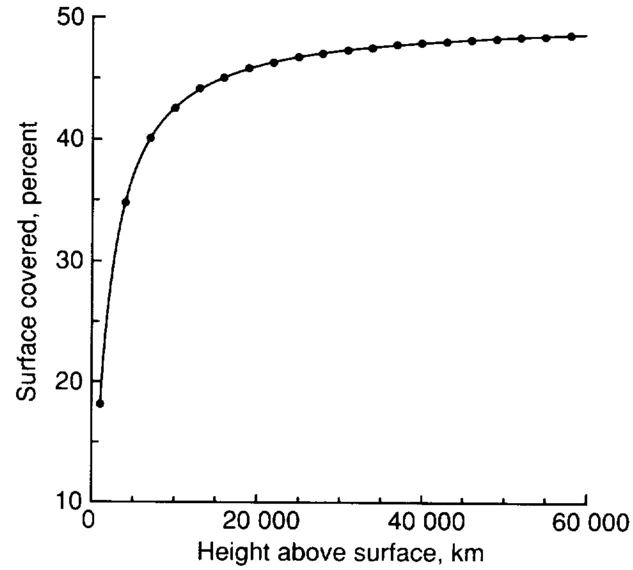


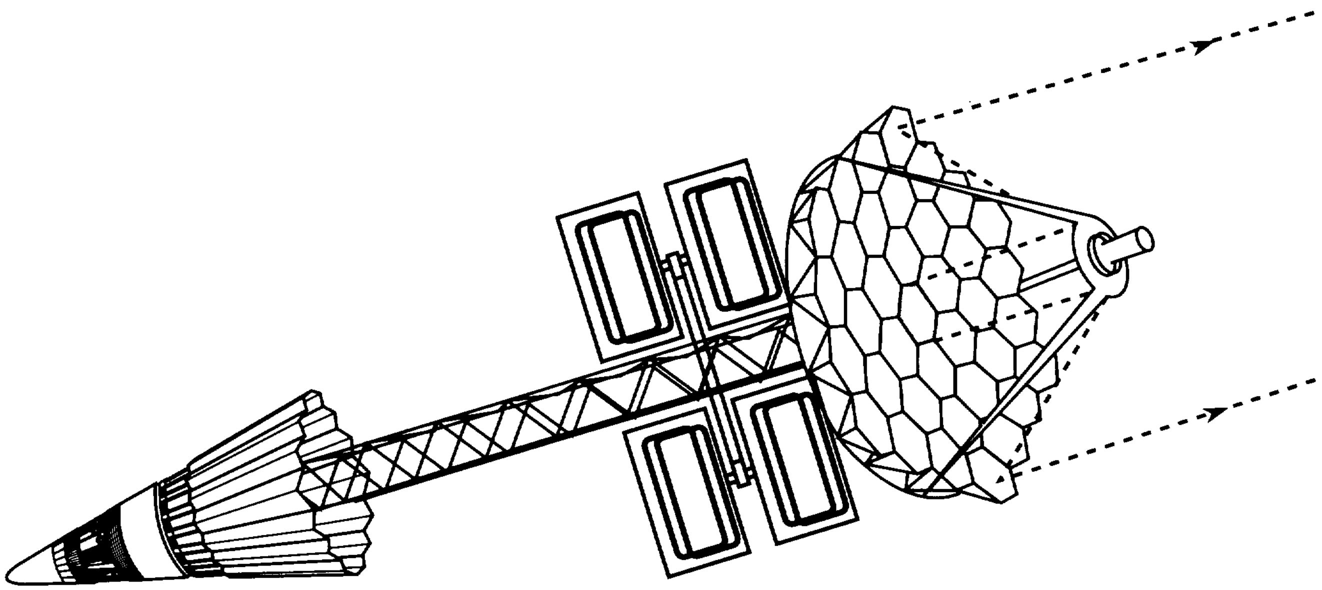
Figure 2. Visible fraction of lunar surface.

Prime Power Sources

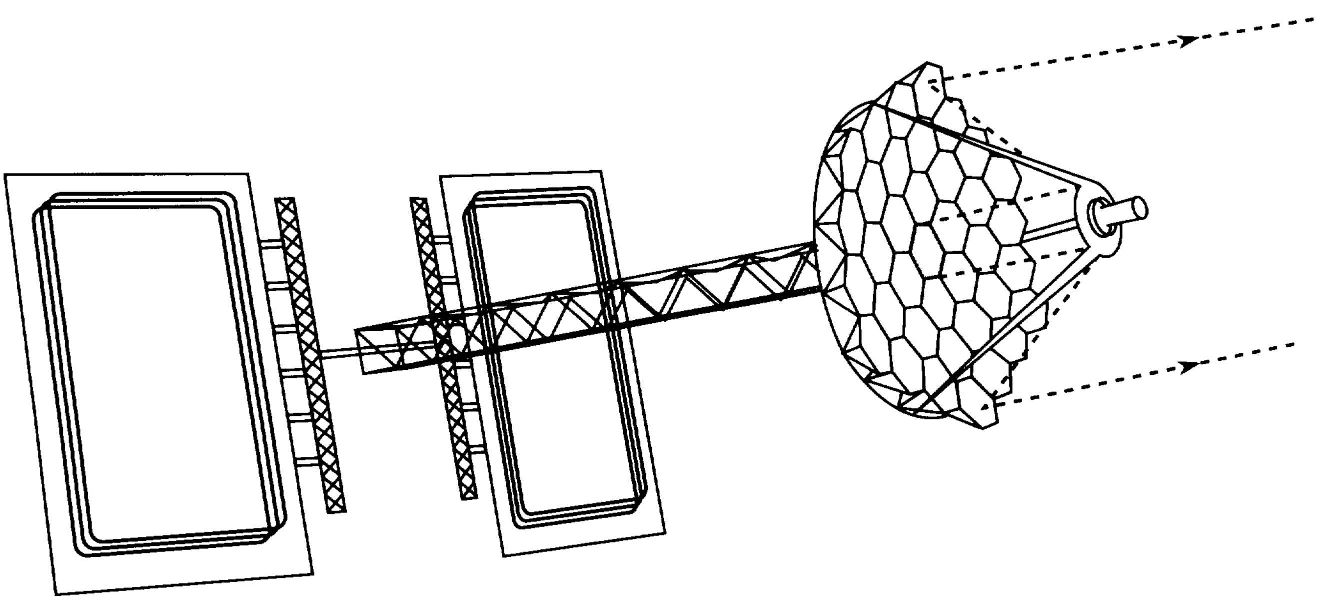
Nuclear

A nuclear reactor is a readily adaptable prime power source for the laser diode array. Such reactors are presently being developed under the SP-100 space reactor program. The SP-100 reactor is designed to produce 100 kW of electric power, but the design can easily be scaled to other power requirements. This reactor uses high-energy neutrons to produce thermal power, which is transported by a liquid metal coolant to the thermoelectric power converters. The converters produce electricity at about 4 percent efficiency. A secondary heat transport loop carries the waste heat to a thermal radiator where it is radiated away. The electrical output of the converter is sent to a power conditioning and regulating system, then on to the laser diode array.

A 20-m boom separates the reactor system from the laser diode array. An illustration of the total system is shown in figure 3(a). The reactor produces a heat loading at the laser diode array of 0.14 W/cm^2, which must be dissipated. Also nuclear radiation produces a dose of 5×10^3 grays (G_y) and a neutron fluence of 10^{13} n/cm^2 over the full-power 7-yr reactor lifetime. These radiation doses are not significant



(a) Reactor satellite.



(b) Photovoltaic satellite.

Figure 3. Satellite systems.

enough to degrade the performance of the laser diode array.

The specific mass of the reactor power system is 40 kg/kW (electric power). Thus, at the required power of 210 kW, the reactor power system mass is 8400 kg.

The reactor would have a minimum 7-yr lifetime, whereupon it would be replaced. The reactor could be permanently parked at L1. Placing the reactor above the Moon eliminates undesirable contamination of the lunar surface.

Photovoltaic

The photovoltaic (PV) power source is based on the present Advanced Photovoltaic Solar Array (APSA) program initiated by the Jet Propulsion Laboratory in 1985 (ref. 10). The APSA program goal is based on a specific power need of 300 W/kg at a power level of 25 kW by the year 2000. A more modest beginning of life (BOL) requirement of 240 W/kg is utilized in the present study, which translates to a specific mass of 4.17 kg/kW.

Briefly, the APSA technology uses a 50-μm-thick carbon-loaded Kapton polyimide film as the blanket substrate material. The blanket consists of multiple three-solar-panel assemblies (SPA's) and blanket leader assemblies. In addition to the blanket substrate material, each SPA consists of a 50-μm-thick cover glass, solar cells with a packing factor of 0.84, 55-μm-thick aluminum back surface reflector, interconnectors, and the necessary adhesives. The number of SPA's in a particular blanket depends upon the amount of power required for a specific mission.

The solar cells chosen for this mission are 10-μm-thick GaAs cells with a BOL operating efficiency of 18.5 percent. The estimated end of life (EOL) efficiency at the end of a 7-yr near-Earth mission is 13 percent. The EOL efficiency estimate assumes passive cooling in a near-Earth space environment, radiation degradation of 12 percent (ref. 11), and a mass contingency factor of 5 percent. This results in an EOL specific mass of 6.4 kg/kW. An EOL power of 233 kW is achieved with twenty-six 2.74- by 26.9-m wings at 9 kW each, resulting in a total array mass of 1500 kg.

An additional mass of 3700 kg must be included in the PV power system mass to account for the associated trusses, beta gimbals, orientation motors, and the power distribution network. This results in a total PV power system mass of 5200 kg. The total photovoltaic satellite is illustrated in figure 3(b).

Stored Energy

Energy storage requirements for the satellite are based on the needs of the following onboard electrical systems:

Communications transceiver	60 W
Coolant pump	1500 W
Laser diodes	132 kW

The first priority is communications with the satellite. The second priority is satellite orientation, since power from photovoltaic cells should almost always be available if the satellite can achieve the right orientation.

The energy storage system chosen is the same as that proposed for Space Station *Freedom* (SSF) (i.e., a nickel-hydrogen battery system consisting of 30 cells in series, each with its own charge/discharge unit). These cells provide approximately 9 kW for 1 h at 112 V. Lower power outputs provide proportionately longer times. Thus, the energy storage system can power the transceiver for about 6 days or the coolant pumps for 6 h. It is not practical to provide the high power required by the laser diodes but for a very short time. Hence the diodes can be sustained at full power for only 4 min, after which time the rover must resort to its own emergency power source. The estimated mass of the energy storage system is 604 kg.

Laser Diode Array

The laser diode array of the satellite is based on two amplifier array modules developed by the McDonnell Douglas Electronics Systems Company. (See ref. 12.) Both modules are thin planar optical waveguide amplifier arrays made from graded-index, separate confinement heterostructure, single quantum well wafers. The output configuration of one module is a linear arrangement of 400-ridge-waveguide apertures, 3 μm wide by a fraction of a micron thick, spaced on 25-μm centers; the other module has 10-ridge-waveguide apertures spaced farther apart. Ten of the 400-ridge-waveguide amplifier modules are stacked plane to plane to form an element of the output array that is almost cubic in shape and has a square emission aperture. (See fig. 4.) The 10-ridge-waveguide modules are used as drivers. Each of its 10 outputs connects to the input of 1 of the 10 stacked amplifier arrays by flexible light pipes.

Beams from the small apertures of the 10 stacked amplifier modules expand to fill cylindrical microlens

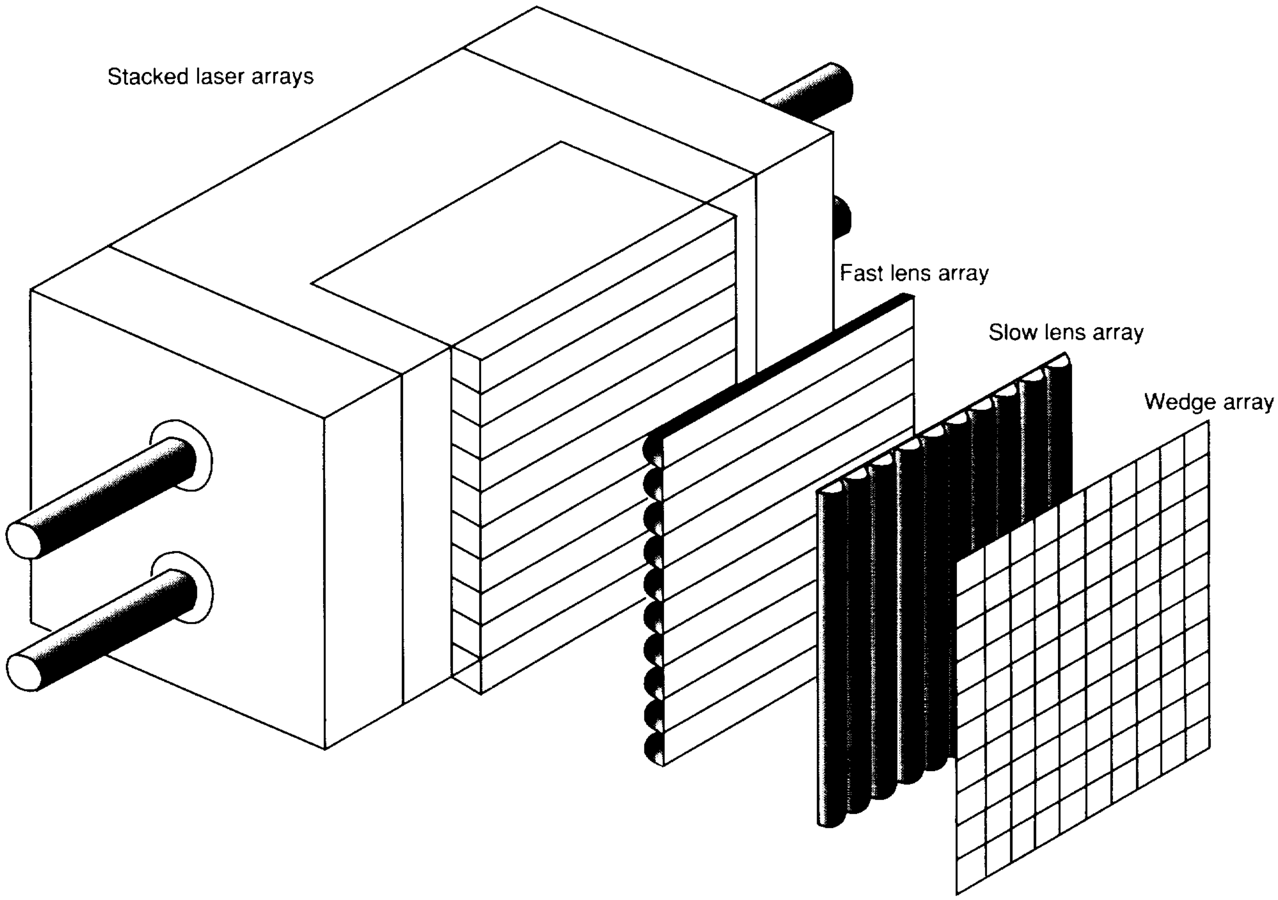


Figure 4. Output element.

arrays 1-cm square and are collimated. With further development, the module is expected to emit a diffraction-limited, collimated beam of 10 W with 50 percent electrical conversion efficiency through a 1-cm^2 output aperture. The laser diode array must provide 90.4 kW of optical power. Therefore, at least 9040 output elements must be provided. Arrangement of the square apertures of the output elements into a simulated circle requires approximately 9850 elements. This increased number of elements allows for optical losses in the output microlens arrays and some laser diode failures.

The output array is preceded by driver stages, the primary purpose of which is to lock all output elements to the same frequency. The last driver stage requires 9850 10-ridge-waveguide modules. These driver amplifiers are physically located near the output module to which they are connected by fiber optics and microlenses. Very little amplification is required of the signal, but amplification is available to solve problems in the output stage. This driver stage requires about 3550 W of electrical power operating at 50 percent efficiency.

Twenty-five 400-ridge-waveguide amplifiers are similarly connected to the last driver stage. This intermediate driver stage requires about 394 W of electrical power.

One 400-ridge-waveguide amplifier in the first driver stage suffices to provide the 25 inputs required by the intermediate driver. However, two are used (one at a time) to minimize the blacking out of large sections of the output stage due to first driver failures. Blackout is minimized by using horizontal and vertical micropositioners to position the two amplifiers with respect to the linear fiber optic input array that couples to the intermediate driver such that 28 totally independent drive positions are available. About 44 W of power are required by the first driver.

Similarly, any one of three master oscillators is selectable (by mirror) to drive the first driver

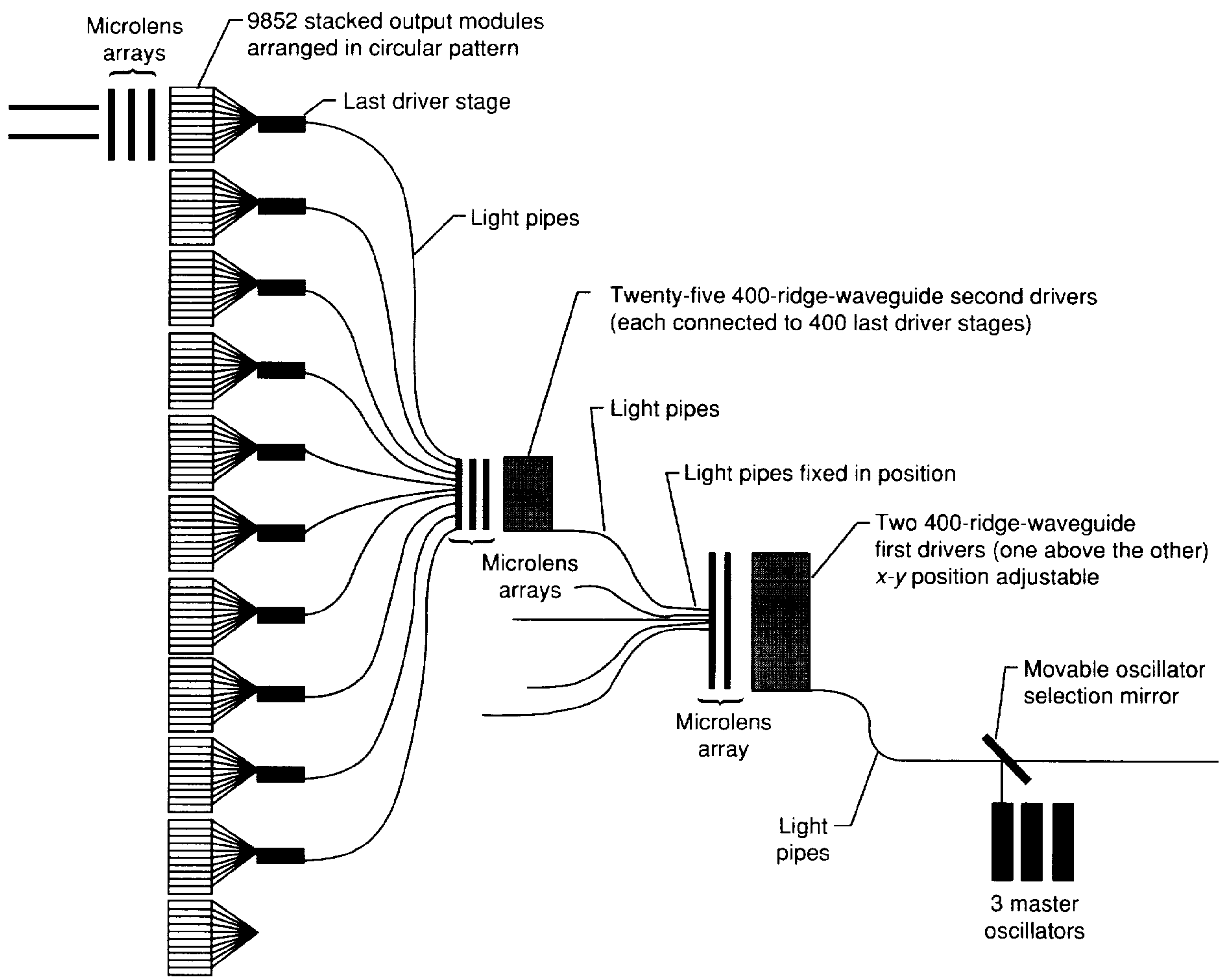


Figure 5. Laser diode array.

stage. The oscillator requires collimating, alignment, isolation optics, and about 36 mW of power. (See fig. 5 for a schematic representation of the laser diode array.)

The mass of a single 400-ridge-waveguide unit (or the mass of a 10-ridge-waveguide unit) with optics and support structure is estimated to be 10 g. The 108 380 units required by the entire laser system would have a total mass of approximately 1080 kg.

Thermal Control

The LDA operating at 50 percent efficiency plus losses in the power distribution system produce 99.6 kW of heat. Since efficiency, phase stability, and heat transfer require operation of the LDA at $20^\circ \pm 0.25^\circ$C, most of the heat must be removed from the LDA.

The proposed system uses water as the coolant. Water from the radiator outlet is fed through each LDA unit. The radiator inlet temperature must be kept below 20°C, and the outlet temperature is kept above 0°C. This requires a minimum mass flow rate of about 1.2 kg/s and a volume flow rate of 0.0012 m^3/s through the radiator. The radiator is fabricated with a metal matrix composite (graphite aluminum). The specific weights of the optimized heat pipe and metal matrix composite radiator (ref. 13) are 0.297 kg/m^2 and 0.495 kg/m^2, respectively. The specific heat rejection of the radiator is 254 W/kg. Using these values, the removal of 99.6 kW of heat requires a fin radiator with an estimated mass of 393 kg and a heat pipe of length 794 m (coiled within the radiator). The estimated mass of the heat pipe is 236 kg. The mass of the water is 1450 kg. Hence, the total mass of the radiator and heat pipe is 2080 kg.

The pump power required to circulate the coolant (ref. 14) is 1150 W. The estimated mass of the pump is 100 kg.

The LDA units are kept within the temperature stability requirement (±0.25°C fluctuations) by thermoelectric coolers (TEC's) and adjustment of the coolant flow rate. TEC's are integrated into LDA units and provide a fast time response to temperature changes. Adjustment of the coolant flow rate provides time-averaged heat removal from the TEC's and requires a probe, controller unit, and flow valve for each LDA block. The masses of the controller units and tubing are 1080 kg and 542 kg, respectively.

The radiators for the LDA system are placed behind the PV panels to be shielded from solar radiation and to increase their radiant cooling capacity. The gross mass of the cooling system (radiator, heat pipe, primary circulation loops, and pump) is 3800 kg.

Laser Beam, Spot Size, Pointing, and Mass

Point Design

Coherent collimated light that is emitted with uniform intensity from a circular aperture forms an Airy disk pattern in the far field (ref. 15). The angular deviation θ of the first dark ring from the major axis joining the centers of the aperture and the far-field pattern (FFP) is given by:

$$\sin\theta \approx \frac{r}{z} = \frac{1.22\lambda}{D} \tag{1}$$

where r is the radius of the Airy disk, z is the transmission distance, λ represents the wavelength of the transmitted radiation, and D is the diameter of the transmission dish. Rearranging equation (1),

$$rD = 1.22z\lambda$$

$$dD = 2.44z\lambda \tag{2}$$

Since the transmission distance z to the lunar limb from L1 is 61 000 km and the laser wavelength is 0.8 μm, the Airy disk diameter will always be less than 2.4 m if the transmission aperture is 50 m in diameter. The Airy disk contains 84 percent of the transmitted power (ref. 15).

With active feedback, pointing error is estimated to be half that of the Hubble Space Telescope, or about 0.025 microradian. At 61 000 km, pointing error radius will be about 1.5 m. Therefore, the Airy disk will move around within a circle 5.4 m in diameter, and the receiver dish must be just as large to assure reception of the entire Airy disk. Actually, the size of the reception dish makes possible the reception of more than the Airy disk. Depending on the exact position of the Airy disk within the receiver, the receiver would also intercept the first bright ring about the Airy disk or most of the bright ring and parts of others, guaranteeing interception of approximately 90 percent of the transmitted radiation.

To supply the 30 kW required by the rover from 50-percent-efficient photovoltaic converters, 60 kW of power must enter the converter. With a receiver that has antireflection coatings, 90 percent beam interception, and 90 percent reflectivity, the total power in the laser beam must be approximately 73.2 kW.

Obtaining the required pointing accuracy will involve the use of control moment gyroscopes (CMG's) or their equivalent in conjunction with small thrusters. The small thrusters will correct for large components of induced angular momentum or will desaturate the CMG's. If the small thrusters dissipate about 90 percent of the induced angular momentum around each axis, freewheeling coolant flow at the periphery of the transmitter dish can be used to control angular momentum around the axis perpendicular to the transmission aperture. The use of coolant flow practically eliminates the mass of a CMG for this axis. The masses of the CMG's required for the other two axes are estimated at 300 kg each. The masses of the six small thruster systems (including propellant) are estimated to be 60 kg each.

Generalized Design

The analysis above is based on the use of a 50-m-diameter transmission aperture and is only one of many point designs that could be used. While the transmitter dish satisfies beam requirements well, it dominates the system mass, suggesting that it is not the optimal system. Total satellite mass (kg) can be *approximated* by summing the masses of its major components (definitions of the terms are given in the symbol list):

$$M_T = M_r + M_{th} + M_d + M_e \qquad \text{(Reactor satellite)} \tag{3}$$

$$M_T = M_v + M_{th} + M_d + M_e \qquad \text{(Photovoltaic satellite)} \tag{4}$$

$$M_r = 40P \tag{5}$$

$$M_{th} = 19.7P \tag{6}$$

$$M_v = 30P \tag{7}$$

$$M_d = 15A \tag{8}$$

$$M_e = 2800 \tag{9}$$

$$A = \frac{\pi D^2}{4} \tag{10}$$

$$dD = 2.44z\lambda \tag{11}$$

$$P = \frac{P_r}{FE_1E_2E_3} + 1.5 \tag{12}$$

$$F = 1 - J_0^2\left(\frac{KdD}{4z}\right) - J_1^2\left(\frac{KdD}{4z}\right) \tag{13}$$

These equations show that for a fixed receiver aperture, the transmitter aperture can be decreased in size, but more of the beam will spill over the aperture of the receiver, thus requiring increased laser power to maintain rover power needs and resulting in larger prime power sources (reactor or photovoltaic). Stated another way, as the mass of the transmitter dish decreases, the mass of the prime power source must increase to provide the required rover power. The above equations have been programmed to show the variation of total satellite mass with transmitter dish diameter for several receiver diameters. (Fig. 6 shows the mass variation for the photovoltaic satellite. Mass variation for the reactor-powered satellite is similar.) For each receiver diameter in the plot, there is a particular transmitter diameter for minimum satellite mass. At smaller transmitter diameters the mass of the prime power source begins to dominate total satellite mass; at larger diameters the transmitter dish dominates. As receiver dish diameter increases, the minimum satellite mass decreases. However, the decrease in the minimum mass gets smaller as the receiver diameter increases, and no preferred receiver diameter is indicated. Rather, a receiver diameter has been arbitrarily chosen that seems compatible with the size of the lunar rover and represents a satisfactory point of diminishing mass savings. At that point the receiver diameter is 4 m and the transmitter dish diameter for the PV satellite is 17 m. The receiver dish intercepts about 73 percent of the incident radiation if the beam is centered on the receiver. To compensate, the LDA must produce 123 percent (90/73) of the power that was produced for the 50-m-diameter dish and be 11 percent more massive (1204 kg).

For the reactor satellite, the receiver dish is 4 m in diameter and the transmitter dish is 17.5 m in diameter. The receiver intercepts 74 percent of the incident radiation of a centered beam. The LDA for the reactor satellite must be 10 percent more massive (1195 kg) than a similar system with a 50-m-diameter dish.

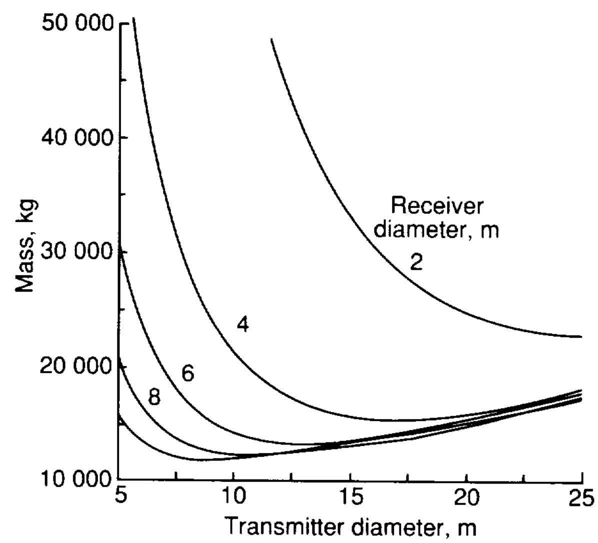


Figure 6. Satellite mass variation.

Since the beam pattern is larger than the receiver, beam movement due to pointing/tracking errors will decrease power reception. For the photovoltaic satellite, the Airy disk diameter will be 7.6 m and will move in a circle 10.9 m in diameter. For the reactor satellite the beam diameter will be 7.4 m, and it will move in a circle 10.7 m in diameter.

Transmission Optics

The large transmission dish for the photovoltaic satellite is a parabolic surface composed of many small hexagonal reflective facets that approximate a 17-m-diameter circular aperture at the periphery of the assembly. The focus of the parabolic surface is 10.3 m from its vertex, where three mutually perpendicular arms join and support a smaller parabolic reflector. From the plane of the aperture to its vertex, the large parabolic dish is 1.8 m deep and has a surface area of 236 m^2. Scaling from the mass of the Large Deployable Reflector (ref. 16), the mass of the parabolic reflector would be approximately 3550 kg. Trusswork for the reflector composed of graphite/epoxy tubes (ref. 17) adds 27 kg to the structure. The mass of the small parabolic reflector and its support is about 19 kg.

The transmission dish for the reactor satellite is 17.5 m in diameter, focal length 10.6 m, and mass 3760 kg.

Satellite Power Station Conclusions

The placement and control of a power-beaming satellite are within present capabilities. Orientation (pointing and tracking) and some technology (laser diode array and transmitter dish) are beyond present capability but should be available within a decade. The photovoltaic version of the satellite would transmit 67 kW of laser power and would be the least massive, but the differences in mass and power of the two satellite versions are not great. A summary of the masses of the major components of the satellites is given in table I.

Table I. Satellite Mass Breakdown

Component	PV system mass, kg	Reactor system mass, kg
SP-100		7120
PV power system	5580	
Energy storage system	600	600
LDA	1200	1200
Thermal control system	3640	3500
CMG's and thruster	960	960
Transmitter dish	3550	3760
Small reflector and support	20	20
Total	15 550	17 160

Part B: Lunar Rover

Design Assumptions and Summary

The design assumptions that are the basis for subsequent lunar rover drive system requirements and design features are provided in table II. Although the table refers to a 1000-km range, which was an initial design parameter, the design of the lunar rover does not preclude additional range if desired. The design does not use primary energy storage for locomotion. Thus, the rover can operate continuously either moving or stationary. The mission duration and crew complement, on the other hand, are critical design parameters because they drive the mass of consumable supplies and life support system requirements. The mass summary of the rover, listed in table III, is derived from subsequent sections of this report.

Table II. Lunar Rover Design Parameters

Feature	Amount
Range, km	1000
Crew number	4
Mission duration, days	30
Mission speed, km/h	10
Maximum slope (firm soil), deg	30
Design life, yr	7

Table III. Rover Mass Breakdown

Component	Mass, kg
Laser receiver system	900
Energy storage system	300
Power distribution equipment	150
Command, control, communications	100
Drive system	370
Cabin thermal management	480
Life support equipment	550
Housekeeping equipment	50
Consumables	1200
Crew	300
Personal gear	50
EVA suits	200
Radiation shielding	1250
Structure	900
Mission hardware	1200
Total	8000

The rover is a cylinder 7.5 m long with an outside diameter of 3.2 m. The cylinder length includes 5 m of laboratory work space and 2.5 m at the aft of the vehicle for extravehicular activity (EVA) suiting operations and manipulator arm control. The driving cab is an additional 2.5 m at the front of the rover. A sketch of the rover is shown in figure 7.

To utilize SSF components, the rover uses SSF international racks 1 m wide laid out as shown in figure 8. The EVA staging area has only bottom racks (no side racks). This yields a total of 28 racks, 12 bottom racks suitable for storage and 16 side racks for mounting hardware. The overhead space is used for liquid storage, which also provides crew radiation shielding. Rack dimensions and volume requirements are further described in the section "Habitation and Life Support System."

Power System Design

The power for the lunar rover is provided by a photovoltaic receiver that converts laser light to electrical power. The vehicle receives laser power while in motion or at rest. This enables the rover to operate continuously in either mode, eliminating limitations on travel distance or constraints on mission operation.

Power Requirements

The system design provides an abundance of electrical power, enabling the rover to perform a variety of functions and missions not possible with other rover concepts. A summary of the power system requirements for the various functions of the rover is provided in table IV.

Power for external illumination is included in the drive system power allocation. Its use would reduce average speed. Only a slight reduction in the average traveling speed would provide significant additional power for illumination during night operations.

The thermal management power consumption is a function of energy dissipated within the pressurized module. It is assumed that all mission, control, lighting, habitation, housekeeping, servicer recharge, and power system losses will ultimately result in waste heat dissipated into the module. This waste heat must be moved from the habitat module at 300 K to the radiator at 400 K.

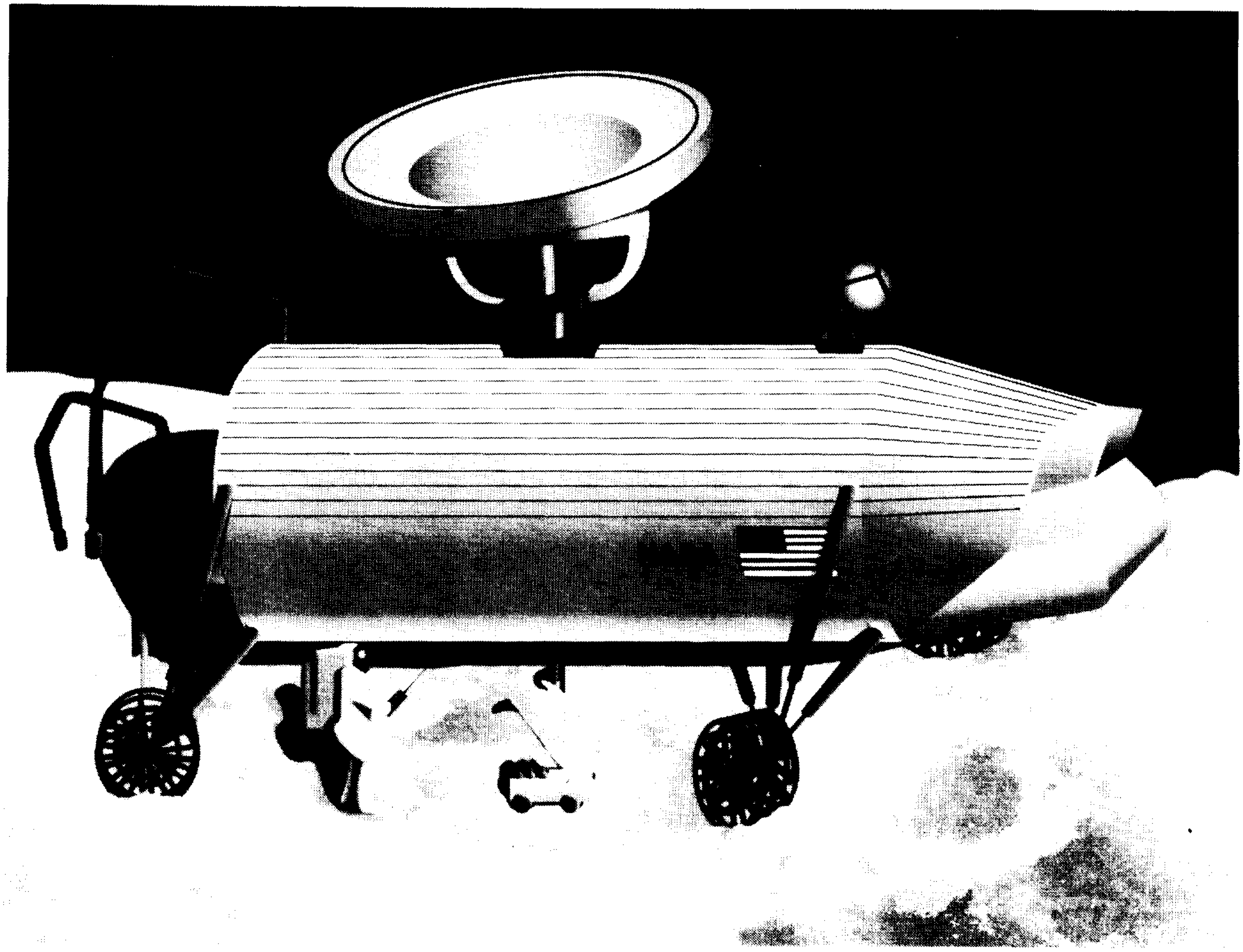


Figure 7. Lunar rover.

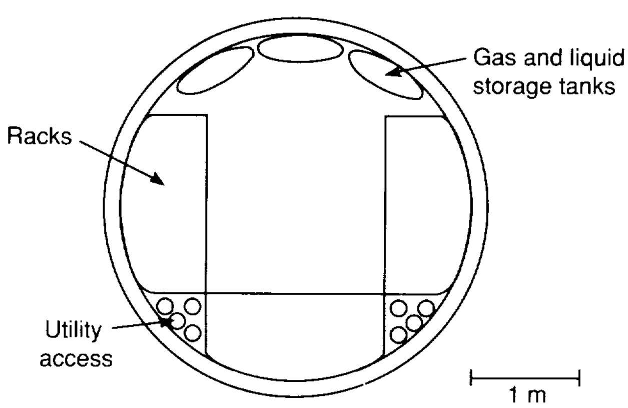


Figure 8. Rover rack arrangement.

However, power consumed outside the pressurized module, which includes the drive motors and the external lighting, would not have to be removed using this system.

Table IV. Power Allocation Summary

Component	Average power, kW	
	Driving	Stopped
Drive system	15.0	0.0
Thermal management	5.0	10.0
Missions	2.0	10.0
Control, interior lighting	2.5	2.5
Habitation, life support	2.0	2.0
Housekeeping, cooking, etc.	0.0	2.0
Battery charger	2.0	2.0
Power system loss	1.5	1.5
Total	30.0	30.0

Mission related hardware, which includes sampling and analysis equipment, will be used primarily when the rover is stationary. However, some power

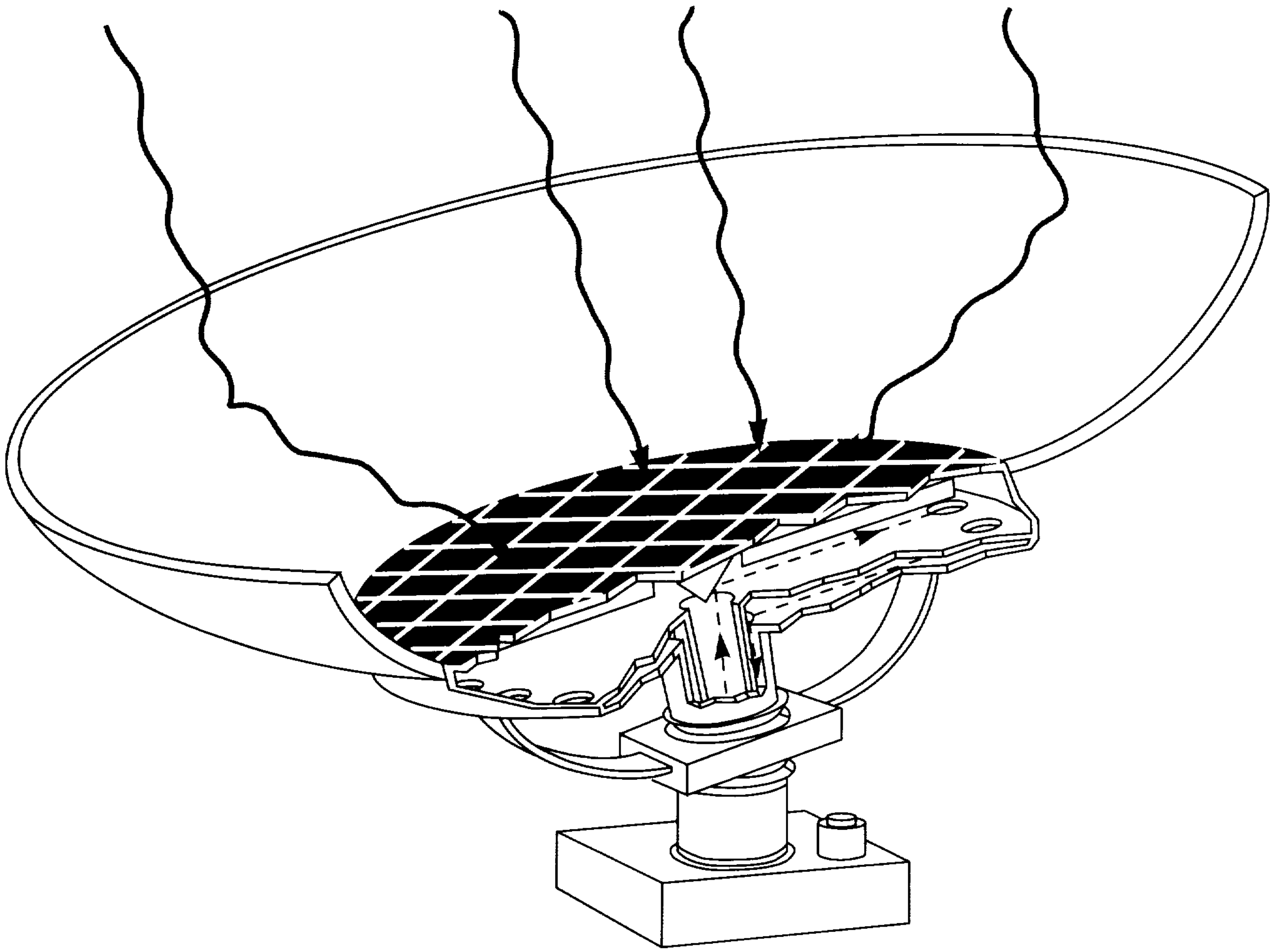


Figure 9. Receiver/converter.

has been allocated for consumption during periods of rover motion to account for possible long-term experiments that cannot be shut down or autonomous functions that may be continuous.

The control functions include communication equipment, computer systems related to rover operation, rover instrumentation and diagnostic equipment, and internal cabin illumination.

Habitation and life support include all critical functions for conditioning the air, such as ventilation and CO_2 scrubbing. Housekeeping includes all noncritical functions for living on the rover that may be switched off during periods of mobility. This includes functions such as water treatment and food preparation.

A battery charging system is included for robotic minirovers, as well as for any mission-related battery-operated equipment. This power is available continuously to eliminate operational constraints.

The power distribution system is assumed to be 95 percent efficient, which is reasonable given the simplicity of the design and the relatively short cable runs required.

Laser Receiver

The laser energy transmitted to the surface is converted to electricity by an AlGaAs photovoltaic converter, as shown in figure 9 (ref. 5). Since the converter is tuned to the laser wavelength, very high conversion efficiencies on the order of 50 percent are achieved. A heat rejection system is included to remove waste heat and to maintain the receiver photovoltaics at 320 K.

The laser receiver mass breakdown is listed in table V. The laser receiver was scaled using

De Young's (1991) design, converting the net electrical output from 15 kW to 30 kW. Each component of the converter mass is doubled, except the tracking device and the coolant piping. The tracking device does not scale with power, since it is primarily structure and control systems associated with receiver orientation. The coolant piping mass increases by a factor of $\sqrt{2}$, which represents doubling the effective piping area. The receiver size is relatively unaffected by increasing power, since the diameter of the reflector dish surrounding the active area is primarily a function of beam jitter and diffusion, which depends on the separation distance, wavelength, and pointing accuracy. The laser receiver is about 4 m in diameter.

Table V. Laser Receiver Mass Breakdown

Component	Mass, kg
AlGaAs cells	1
Active area matrix	2
Laser concentrator	54
Tracking system	100
Thermal management system:	743
Radiator	464
Coolant piping	159
Compressors	120
Total	900

Energy Storage

Although the laser system is sized to provide average power for both stationary and mobile modes of operation, brief periods of above average or below average power will occur regularly as equipment is switched on and off. Onboard energy storage will be used to compensate for these load variations. This energy storage can also be used for emergency conditions when there is an interruption in receiving laser energy or a system failure on the rover that precludes normal operation. This emergency standby power will be adequate to maintain the rover in a "keep alive" mode until rescue or to provide power to perform limited maneuvering.

Two fuel cells in parallel will provide energy storage with an overall capacity of 30 kW and 50 kWh. Near-term fuel cell technology has a specific power of 7 kW/kg and an energy storage density of 225 Wh/kg. The onboard fuel cells have a total mass of 300 kg.

If 30 kWh is dedicated as emergency standby energy, with the remaining stored energy available for routine load variations, the life support system could operate at full capacity for a minimum of 15 h.

Electrical Power Distribution

Figure 10 is a one-line diagram of the rover power system. The system incorporates the concept of a dual bus, which minimizes the effect of a single-point power system failure. Each bus occupies one rack on each side of the vehicle and provides power to racks on the same side of the vehicle. Thus, any redundant critical hardware, such as life support equipment, is installed in racks on different sides of the vehicle. The fuel cells are located in racks adjacent to each bus.

The total mass of the power system components is estimated to be 150 kg (ref. 18). Somewhat contrary to traditional aerospace design, which must sometimes trade between mass and efficiency, mass reduction in power cables reduces efficiency. This power system has a design efficiency of 95 percent.

Mechanical Drive

The lunar rover drivetrain design is lightweight and uses current technology for electric motors and heat rejection. This drive system is well matched to the continuous, relatively large amount of power available from power beaming. As a result, the rover has extended range, increased speed, and enhanced maneuverability.

Lunar Soil and Terrain

The lunar regolith (surface debris layer corresponding to terrestrial soil) consists of particles ranging from fine dust to rocks (refs. 19 and 20). This regolith was formed by meteorite impacts that resulted in shocked fragments of rocks, minerals, and glass spherules (ref. 21). The bulk of the regolith is made up of fine particles that are porous and weakly coherent at the surface but become more compacted with depth. Typically, a thin dusty surface layer about 1/4 cm thick covers a dark gray caked zone, below which there is a 5- to 15-cm-thick, slightly cohesive sandy to silty material (ref. 19). Underlying these zones is a layer that exhibits an increase in firmness and resistance to penetration. The thickness of the regolith is quite variable with location, but is estimated to be between 3 and 20+ m (refs. 21 and 22). The depth of the material appears to be enough to preclude penetration of any conceivable rover wheel through the regolith. Note, however, that the data base for the above values is rather limited (horizontally and vertically) in terms of actual exploration. For example, astronauts were able to obtain core samples only about 60 cm deep (ref. 22).

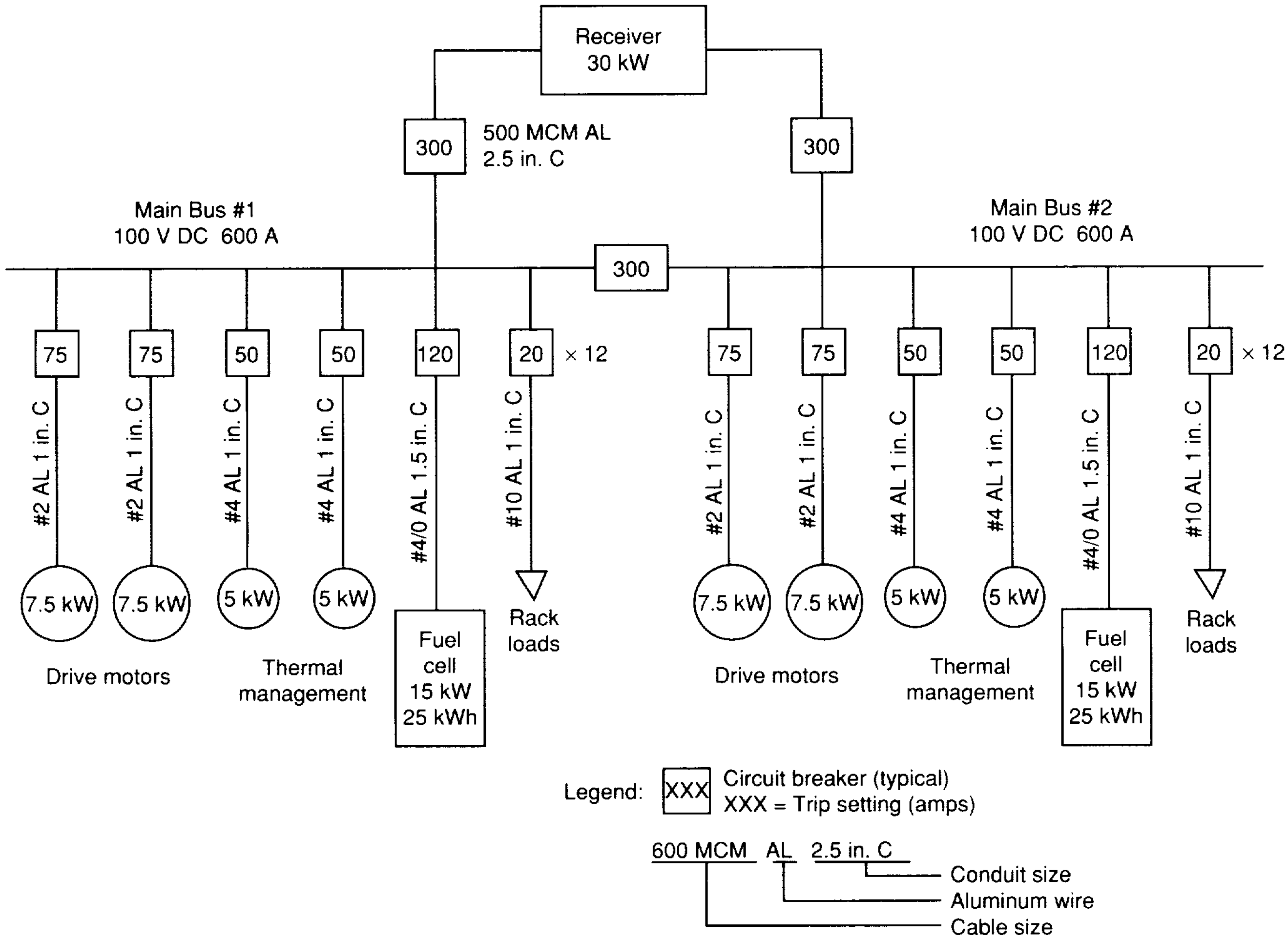


Figure 10. Rover electrical power distribution.

Of main interest here is that, mechanically, lunar soils behave similar to terrestrial soils of comparable particle size, even though composition may be different (ref. 21). In particular, both astronaut footprints and lunar rover tracks penetrated the regolith to depths ranging between 0 and 15 cm (ref. 22). The majority of lunar slopes have angles less than 20° (ref. 23), so the 30° rover climbing limit will restrict travel very little.

Propulsion Requirements

The rover propulsion power requirements include a combination of acceleration, climbing, and horizontal motion in soft soil. Each of these are described separately below, then integrated into the overall requirements.

Acceleration. For an 8000-kg rover to accelerate at a constant rate to 10 km/h in 5 s, 12.3 kW of power is required near 10 km/h.

Climbing. For the lunar gravity, which is one-sixth that on Earth, 18.2 kW is required to maintain the 10-km/h top speed while climbing a 30° slope with firm soil. At lower speeds on the same slope, power would drop proportionately.

Horizontal motion on regolith. The rolling resistance of a wheel is a function of wheel and ground deformation. Given the soft lunar regolith, it appears reasonable to ignore wheel deformation and use the standard equation to calculate the driving force requirements. The required driving force is given by

$$F_r = \frac{Wb}{r_r} \tag{14}$$

where W is the vehicle weight in Newtons, b equals a dimension somewhat less than half the chordal distance across the tire subtended by the regolith when the tire sinks into the regolith (in meters), and r_r equals wheel radius (in meters) (ref. 23). Given a 1.2-m-diameter wheel and $b = 12.5$ cm (typical of soft soil), the power required to maintain forward motion is 7.5 kW, which is the force multiplied by the vehicle speed. This implies that the wheel penetrates into the regolith to a depth of 1.7 cm.

Assuming that the wheel will penetrate the full 15 cm experienced during the Apollo 16 rover missions, then b is nearly tripled and the power requirement rises accordingly. Alternately, given constant power, one could elect to reduce speed from 10 to 3.3 km/h over the softer regolith.

Propulsion power requirements summary. The previously calculated power requirements are summarized in table VI. The third column shows an increased requirement after incorporating an assumed 75 percent overall drivetrain efficiency, which is a conservative assumption.

Table VI. Propulsion Power Requirements

Mode	Power requirements, kW	
	100% efficiency	75% efficiency
Acceleration	12.3	16.4
Climbing	18.2	24.3
Steady speed/horizontal	7.5	10.1
Maximum[a]	30.5	40.7

[a]Implies acceleration on a 30° slope covered with soft soil and speed near maximum.

Thus it appears that if an average power of 15 kW were made available to the rover drive via power beaming, the assumed acceleration and steady horizontal speed criteria could be readily met, or nearly met. Climbing would have to be done at less than 10 km/h, and horizontal travel in soft regolith would also be at slower speeds. Alternately, additional power could be produced from an onboard energy storage system. A specific mission profile would be required to calculate the trade-offs necessary for mission optimization.

Drivetrain Design

Electric Motors

Recent developments in high-efficiency, low-mass electric motors can be used to advantage in a rover vehicle. For example, a journal article (ref. 24) describes a research motor that produces 10.5 kW at 20 000 rpm and weighs 14 kg. Similar, somewhat smaller motors are currently available from Unique Mobility Inc. of Englewood, Colorado. Their DR086x motor produces 7.5 kW at 10 000 rpm and weighs 4.3 kg; a somewhat more powerful and heavier motor, the DR127x, produces 15 kW at 8000 rpm and weighs 6.7 kg. It will be assumed here that four of the smaller motors will be adequate for rover propulsion. These motors, plus associated gear reduction, braking, and heat rejection systems, are mounted in complete units, with one assembly per wheel. Motor controllers are also one per wheel but will reside within the cabin and reject heat into it. Other relevant motor characteristics are summarized in table VII.

Table VII. Electric Motor Specifications (ref. 8)

Characteristic	Amount
Voltage, dc	50 to 100 (with higher voltage preferred)
Dimensions, mm	122 diameter × 133 long
Efficiency (over 70% of speed range)	90
Operating temperature (continuous), K	422
Vacuum operation . . .	Not a problem
Over torque rating (short term)	Up to 300%

Gear Reduction

The 1.22-m-diameter wheels turn at about 43 rpm at a maximum speed of 10 km/h. A double-reduction fixed gear with a ratio of about 250:1 should be readily achievable to match the motor to the desired speed range.

Braking

The Unique Mobility motors can be operated as generators. Typically, this regenerative braking provides 30 percent energy recovery over a complete duty cycle on Earth. A disc brake system similar to one that was proposed for a rover vehicle (ref. 25) will be employed here also.

Heat Rejection

Surface temperatures on the Moon present extremes. For example, at the Apollo 17 site, temperatures ranged from 384 K during the day to 102 K just before sunrise (ref. 21). The Moon reflects about 7 percent of incident light, with the brightest areas being about 3.5 times brighter than the dimmest areas (ref. 21). These conditions create a challenging temperature control problem.

At full power and with an efficiency of 90 percent, 3.33 kW of heat needs to be rejected from the electric drive motors (transmission efficiency should be high enough that little heat will be generated there). At 422 K motor-operating temperature, 1.83 m^2 of radiator surface would be required to reject this amount of heat to space. Rejection of heat to the lunar surface at 384 K would require 5.84 m^2, so this is also a feasible option although a difficult one to implement given the albedo and temperature variabilities of the soil. It will be assumed here that a shrouded heat pipe panel radiator that views space only, with a view factor of about 0.2, will be mounted on the rover body. Thus about 10 m^2 of (projected or actual) radiator area will be required in this design.

The radiator will consist of metal/ceramic fabric heat pipes using water as the working fluid and having a specific mass of 3 kg/m^2 (ref. 26). The flexible heat pipes would expand and contract in length with the heat load to match motor needs. Coolant loops at the motors and heat exchangers at the radiator panels will serve to transfer the heat from the motors to the radiators. The heat-transfer medium in the loops could be pressurized water or an oil-type organic fluid at lower pressure than the water would require. Test results have shown that metal/ceramic fabric heat pipes can be effectively started while frozen (ref. 26).

Control

A full range of variable speed control is available via the CR10-100 Unique Mobility controller. This controller weighs about 2.7 kg and attains efficiency up to 97 percent. It also provides energy recovery by means of regenerative braking.

Drivetrain Mass

The total mass of the drivetrain is 368.0 kg.

Habitat and Life Support System

The vehicle will provide shirtsleeve accommodations for as many as four crew members on 30-day lunar exploration missions. In each of the following sections, the requirements for the designated life support system (LSS) are first established, then followed by a discussion of the required subsystems.

General

The vehicle LSS design is open, using regenerable subsystems based on state-of-the-art physicochemical technologies. This means there will be fewer expendables and the capability to recharge the vehicle subsystems for repeated exploration trips. The level of operation of the various subsystems has been set to meet the technology road map and milestones of the Exploration Technology Program in NASA's Office of Aeronautics, Exploration, and Technology. The LSS technologies were also chosen to be as compatible as possible with those foreseen for the ILO in its emplacement and consolidation phases.

There is not unanimous agreement on the volume and weight requirements for food, air, and materials necessary to support crew members on space missions. One can estimate as high as 32 kg/day of supplies to maintain a person in a totally open LSS, which would require 1 metric ton of supplies per person per month for the rover. Most of the estimated weight is water for personal hygiene, washing clothes, and flushing. The weight requirement can be reduced to about one-third of this figure if less water is used.

Atmosphere

Vehicle atmosphere will be 20 percent O_2 and 80 percent N_2 at 14.7 psi, with minimal CO_2 and trace contaminants. This will require 0.8 kg O_2 and 0.7 kg N per person per day. For a 30-day, four-person crew mission, the weight requirements are 100 kg O_2 and 80 kg N. This does not include air lost in air lock operations or spare capacity.

A two-bed molecular sieve will be used for air scrubbing for CO_2 removal, replacing the four-bed molecular sieve developed for SSF. This will eliminate two desiccant beds upstream of the two CO_2 beds in that model, with concomitant weight and space savings. The CO_2 can be vented or reduced.

CO_2 can be reduced using the well-known Sabatier or Bosch processes. The Sabatier process has significant mass, power, and volume advantages over the Bosch process, but produces methane, which must be vented or otherwise processed. SSF will use a Bosch process, which produces solid, elemental carbon. This vehicle and the lunar base could most profitably use the Sabatier process if the methane could be successfully used as a fuel or otherwise reduced to its carbon and hydrogen components.

Water

Since water constitutes over 90 percent of the consumable loss of an LSS, it is a high priority to reduce this loss. SSF estimates are as high as 22 kg of water per person per day for total use. Estimates of water requirements vary widely across sources, mostly due to uses of water for other than drinking. Most agree that each crew member will need 2 to 5 kg of drinking water daily, depending on their activity level. For a 30-day, four-person crew mission, weight requirements for drinking water are from 240 to 600 kg.

The vehicle will carry two kinds of water: potable water and laboratory water for onboard analytical use. The potable water loop will be partially closed by recovering condensate water (humidity) from the atmosphere and purifying it by multifiltration to remove dissolved organic vapors and metal ions. The laboratory water will probably have to be deionized if it is used in chemical tests. Lab residues will be stored onboard for later disposal or recovery at the lunar base.

Onboard the vehicle, no water will be used for bathing or washing clothes or dishes. Tissue wipes will be used for bathing. Potable water will be used to brush teeth, with residues stored with other wet residues from food preparation.

The crew will wear disposable undergarments and outer clothes that will be changed at regular intervals. Undergarments, and perhaps some other clothes, can be made of degradable materials that can be combined and eventually recycled with other solid wastes after the mission is complete.

Food storage, preparation, and serving materials will be combined in degradable packaging, which will be compacted and returned to the lunar base for recycling with other solid waste. (Food packages create about half of the solid trash weight produced daily on the Space Shuttle.)

Food

The reviewed literature shows strong agreement for food requirements for space crews, ranging from 1.0 to 1.5 kg per person per day. For a 30-day, four-person crew mission, this would be about 150 kg of food. Food volume depends on packaging and food type, but 0.015 m^3 per person per day appears adequate for solid foods. Total food volume for storage would then be about 1.8 m^3.

Food could be all dehydrated, or mixed dehydrated and frozen depending on the thermal management subsystem of the vehicle. Food will be prepared by reconstituting with water and heating in a microwave and/or solar cook unit.

Waste

A crew member produces 0.113 kg of water weight and 0.024 kg of solid weight in feces every day and 0.7 to 0.9 kg of urine. This translates into 16 kg of fecal material and up to 108 kg of urine for a crew of four on a 30-day mission. These wastes should be kept separate and either treated or stored for treatment in a manner compatible with waste recovery for the lunar base.

The minimum waste treatment scheduled for the ILO is recovery of water from urine and from solid wastes, both for nonpotable uses. SSF will recover water from urine for noningestive uses only.

Solid waste recovery is the least well developed of LSS technologies, and development of a long-term system for water recovery from urine for SSF has been shown to be extremely difficult because of the corrosive nature of urine. This vehicle is therefore planned to store urine and fecal materials onboard in separate storage tanks for transfer and subsequent recycling treatment. Fecal materials can be collected and stored in degradable "baggies," which will save use of water for flushing, while urine can be collected directly into a special holding tank as aboard the Space Shuttle.

Trash

Current Space Shuttle experience indicates that a space crew generates 1.0 to 1.1 kg of trash per person per day. The (noncompacted) volume formation rate is approximately 0.015 m^3 per person per day. Food containers account for about half of this by weight (0.47 kg per person per day) with the remainder being mostly paper and biomedical wastes (0.13 kg each), leftover food (0.09 kg), plastic bags (0.06 kg), and tape, aluminum cans, and miscellaneous items (0.10 kg).

Because tape is used mainly for securing people in weightless conditions, biomedical wastes will probably be minimal; and aluminum cans will likely be replaced by different packaging for lunar missions. Thus trash generation aboard this vehicle should be kept below the nominal 1 kg and 0.015 m^3 per person per day formation rate. For a four-crew member, 30-day mission this amounts to 120 kg of trash by weight and 1.8 m^3 by volume that could be compacted and stored aboard the vehicle for later treatment or disposal at the lunar base.

Radiation Environment and Shielding

The two sources of life-threatening radiation are galactic cosmic rays (GCR's) and solar flares (often called solar proton events or SPE's). GCR's are high-energy, heavy and light ions (including hydrogen ions). Solar flares are high-energy proton bursts. GCR exposure on the lunar surface is a constant 0.25 to 0.35 G_y/day. Solar flares are much more dangerous and variable, occurring in an 11-year cycle. Within each cycle there are 1 to 2 major events of 50 G_y (considered lethal), 2 to 5 moderate events of 5 to 10 G_y, and 20 to 30 minor events of 0.5 to 1 G_y each. Current radiation limits for astronauts apply only to low-Earth (not lunar) orbit and are being revised. Regardless, these limits are set at 2 sieverts (S_v) for a cumulative lifetime career, 0.50 S_v/yr, and 0.25 S_v/mo. A sievert is a biological radiation dose that includes a measure of relative biological effectiveness for specific radiation types. To convert grays to sieverts, multiply the dose in grays by a quality factor that depends on biological effect, dose, dose rate, physiological conditions, etc. These multipliers range in value from 1 to 20.

Prediction of solar flare events is notoriously poor, making protection from these events the prime consideration of vehicle shielding. If the vehicle were only as protective as an EVA suit, it would allow an unacceptable exposure. More reasonably, the rover vehicle should be able to limit the exposure of its occupants to less than the set limits for chronic exposure during solar-minimum activity. Some ancillary shielding strategy will be necessary (hiding behind protective terrain, deployable shielding, or an internal flare shelter) to protect the crew from major SPE's. About 1 to 3 m of loosely packed lunar regolith would probably provide sufficient shielding. (No appropriate exposure limits have been set for the lunar environment.) At a density of 60 kg/m^2, 1 to 10 cm of aluminum would be required if the exposure limit were set at 0.50 S_v/yr. The dose equivalent curves for GCR's and SPE's show a rapid drop with increasing thickness of Al up to 10 cm. The curves break noticeably after this point, indicating much diminished benefits with increasing thickness.

Keeping the weight of the vehicle down while providing adequate radiation protection for a 30-day mission will be difficult. Water may be used as shielding (3.6 cm H_2O = 2.3 cm Al), but it is only as effective as methane, which has less than half its density. Methane would be produced as part of CO_2 reduction under the Sabatier process and would make excellent shielding if it could be safely stored. Using water as shielding implies that it should also be recirculated for nonpotable uses, such as cleaning, but this adds to the mass and LSS complexity of the vehicle. Polyethylene has about the same shielding properties as methane and could be used to impregnate aluminum matrices to provide shielding with practically the same structural strength as solid, heavier aluminum components.

The rover design uses a combination of water, methane, and waste water storage in the overhead bulkhead to augment polyethylene-impregnated aluminum shielding.

Interior Volume and Spatial Arrangements

Many studies over the past 30 yr have looked at volume requirements for crews of space vehicles (ref. 27). They agree that the proficiency of the crew is directly related to the free volume available, and that when space is limited, normal housekeeping and hygienic chores consume large portions of the working day.

Background. Davenport et al. (ref. 28) propose an approximate minimum of 3.1 m^3/person for a four-person, 30-day space mission.

The LUNEX simulator used 4.4 m^3 free volume for a two-man, 14-day mission, which included an air lock space that was rarely used. This would be in keeping with the guidelines set by Davenport et al. (ref. 28). But this study also noted severe problems with housekeeping and hygiene as the crew simply tried to stay out of each other's way and get things done. The small volume here was also partially due to a lower overhead height of just 1.7 m.

Wise (ref. 27) found, after an extensive review of all available literature, that "It's not how large you make it, but how you make it large." Interior free volumes can be kept at or below 4.25 m^3/person for up to 30-day missions as long as the available space and interior layouts are designed to meet the crew's functional needs and certain treatments are employed to make the available volume appear as spacious as possible. It is relatively easy, for example, to make a space appear up to 25 percent larger than it actually is simply by using an angled bulkhead.

Basic design. The lunar rover is designed to make functional use of air lock space and incorporates a highly functional interior design. The interior free volume is 30 m^3. The design uses a "three box" concept that includes cockpit (cab), lab/work space, and air lock areas in sequence from fore to aft.

Cockpit (cab). The cockpit is sized to fit all four crew members in two-by-two supportive seats while driving, as the ride will be rough. An EVA-suited crew member will be able to occupy the driver's station. Each fore-aft pair of seats are convertible to a sleep berth so that two crew members can sleep in the cockpit and two more in the lab space on "jungle hammocks" suspended from the overhead. The four seats also allow the cab to be reconfigured as an impromptu wardroom, probably through addition of a lightweight folding round table that would fit between the seats. This could be a smaller version of the one designed at NASA-ARC for the SSF. The cab would be the preferred place to eat and relax because it is the only place that would allow a window view out from the vehicle.

Thinking of the cab as a "four workstation" space suggests that approximately 0.7 to 0.9 m^3 minimum free volume should be allocated per crew member. This would be in keeping with using the cab as a minimum habitable volume for four crew members. Assuming two space suits are carried aboard the rover, the cab should also allow two suited crew members to operate within it in an emergency. Each suited crew member requires approximately 1.5 m^3 to operate controls. Thus minimal cab free volume appears to be resolving around 3.25 m^3. For comparison, a current large American automobile allots about 1.5 m^3 each to front and rear passenger spaces.

The cab may be a separately shaped volume, as in the Boeing rover study, or incorporated into the same cylinder as the lab space. But the latter wastes space because the 3-m interior width is not needed for driving, and a cab does not need to have full standing height throughout. A cab interior free width on the order of 1.5 to 1.8 m is more reasonable, as this would allow two EVA-suited crew members to operate side by side. The design tapers the cylinder of the lab to form the cab. Further shaping of the cab requires consideration of whether operable probes and manipulators will be added to its capabilities.

The depth of the cab free volume aft of the driving console should be at least 2 m to allow the four crew members to gather around a center deployable table approximately 1 m in diameter, and for conversion of fore and aft seats to sleep berths. A curved cab roof proceeds from overhead console height over the driver to the full 2-m interior height toward the rear of the cab. This results in a cab free volume of approximately 4.0 m^3, with an overall length of 2.5 m.

Lab space. The "lab/work space" would be furnished with international racks as used aboard SSF. International SSF racks are 156 cm high (189 cm with mounts), 60 cm deep (76 cm with frame), and can be configured in varying widths (use approximately 1 m wide for planning purposes). For comparison, U.S. SSF racks are 203 cm high, 90 cm deep, and 107 cm wide. Both U.S. and international racks can be subdivided into half-width segments.

This mandates a minimum interior height of approximately 2 m. This height would also accommodate the tallest crew members while providing room for an overhead bar that would assist crew movement through the interior. Studies on movement under simulated one-sixth lunar gravity suggest such an aid is appropriate.

For safety, EVA suit access throughout the vehicle must be allowed. A 95th percentile male in an EVA suit requires 192 cm height, 85 cm width, and 69 cm depth. Note that this is for the current soft suit. A more likely hard suit for surface operations might require more width and depth.

A vehicle can be designed with single-loaded or double-loaded central lab space (i.e., racks on one or both sides of the vehicle). For balance purposes, double-loaded racks are preferred. For a vehicle with a double-loaded central corridor and international racks, the minimum interior volume dimensions would be 200 cm high, 238 cm wide, and long enough to accommodate the number of racks required.

Adding shielding and hull easily puts the vehicle at 3 m wide. The Boeing lunar rover uses a horizontal cylinder for its air lock, which is 3.4 m in width (ref. 29). Accordingly, a 3.0- to 3.4-m-wide cylinder for the central lab space appears feasible.

The length of the lab space cylinder depends on the number and types of rack instruments and equipment. A minimum would seem to be five racks (approximately 5 m) on a side, for an overall lab length of about 5 m.

Air lock. Standard air lock design houses the EVA suits within the air lock and allows enough room to doff/don suits. However, a "suitport" concept explored at NASA-ARC for the SSF has much to recommend it for adaptation to a rover (ref. 30).

The suitport concept places AX-5-type hard suits outside the pressurized interior volume, attached directly to life support charging and checkout systems. Entry is through a backpack door on the suit that locks tight against the pressurized bulkhead. The suit itself can be shielded from the external environment by a form-fitted cover that closes over it. This suitport has the highest volumetric efficiency of any type tested so far, meaning minimal loss of interior atmosphere during pump down for entry and exit. It also prevents contamination of the interior by lunar dust that adheres to the suits. These should be major considerations for repeated lunar surface EVA's from a rover.

The two suitports proposed for this rover could be housed at the aft end of the lab space. The area adjacent to the suitport could house a receiving station for surface materials on one side and an enclosed hygiene facility on the other. Each would occupy one rack width, with another rack width open behind the suitports, adding another 2 m to the lab cylinder length. Total exterior lab cylinder length would now be approximately 7.5 m.

Within the 2.4-m interior diameter of the lab cylinder, two suitports could be positioned side by side, leading to an exterior "porch" on the aft of the rover.

Interior volume summary. The rover could even be designed as an independent "cab plus lab trailer" vehicle and kept to under 14 m, the limit for the Space Shuttle. This would be in keeping with the Rover First concept, which visualizes a rover taking the place of the ILO. There could be detachable "trailer" cylinders of different kinds that would eventually be linked together in the first permanent lunar base.

Cabin Heat Rejection System

Power beaming is a mission-enabling technology for the lunar rover. It will provide high levels of continuous power, thereby expanding the list of potential mission objectives. However, the energy beamed to the rover must ultimately be dissipated as heat.

Daytime Operation

Waste heat rejection in the lunar environment introduces some unique challenges. The heat must be rejected by radiation to space and the lunar surface. Depending on the time of "day," the location on the lunar surface, and the design heat rejection temperature, the lunar surface may be a net heat source or a net heat sink. The sink temperature for radiation can be relatively high because of the high daytime lunar surface temperatures, up to 384 K (ref. 21). Ewert and Clark (ref. 31) calculate an available daytime heat sink temperature of 322 K for a vertical unshaded radiator at the lunar equator.

A simple heat balance allows calculation of the required radiator temperature as a function of heat load and lunar and radiator physical properties:

$$T_{\text{rej}} = \left(\frac{Q_{\text{rej}} + F_s \alpha_r \sigma A_r T_s^4 + F_m \epsilon_m \alpha_r \sigma A_r T_m^4}{A_r \epsilon_r \sigma} \right)^{1/4} \tag{15}$$

Figure 11 shows the relationship between radiator area and temperature for a heat rejection load of 30 kW and for the properties and temperatures specified in the legend (the absorption capability and emissions are assumed to be constant). The practical limit on radiator area will result from the requirement that the system be mobile. The likely design approach would be to design the largest practical radiator for a mobile rover and then select and size the heat pump system to accommodate the required temperature rise. A 30-m^2 radiator was assumed, based on an overall vehicle length of 10 m and a cylinder width of 3 m. The fabric heat pipe radiator is estimated to have a specific mass of 3 kg/m^2, which gives a radiator mass of 90 kg (ref. 26).

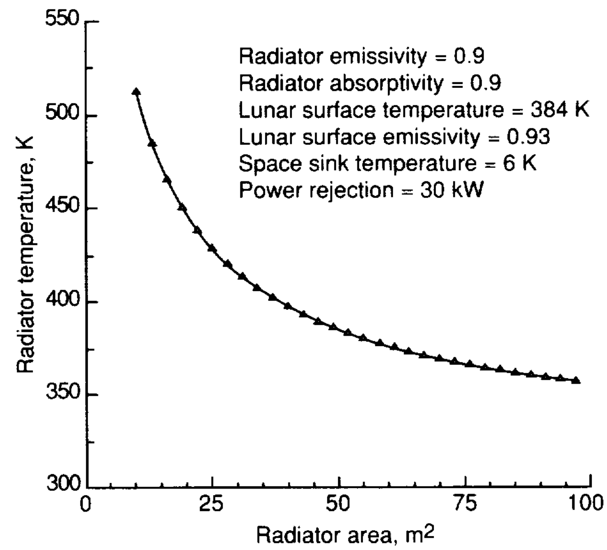


Figure 11. Radiator temperature versus area.

There have been a number of studies to determine the viability of using heat pumps to upgrade waste heat for space applications in order to reduce radiator mass and volume (refs. 32 and 33).

These studies have typically looked at orbital applications where a very low temperature radiative heat sink is available. The general consensus is that for these applications a heat pump system would be marginally beneficial.

The high heat sink temperature that a lunar rover would have available, combined with the need to maintain a livable temperature in the module, necessitate the use of an active temperature control system, since a shirtsleeve environment of 300 K is cooler than the peak heat sink temperature available for an unshaded radiator. From figure 11, a 30-m^2 radiator is required to reject 30 kW at 420 K. Radiator shielding and heat storage capacity may be used to mitigate the heat rejection requirements and to optimize the heat rejection system (refs. 31 and 34).

The in-cabin waste heat rejection system would consist of cold plates coupled to a thermal bus to collect waste heat and a heat pump to upgrade the waste heat. The radiator has a selective coating to minimize insolation.

A promising concept for a heat pump system was proposed by Grossman (ref. 35), in which an absorption heat pump using lithium bromide and water is used to upgrade the waste heat. The primary advantage of the absorption cycle is the lack of moving parts (except for a small pumping device). The cycle employs two working fluids. The temperature lift is obtained when one substance is absorbed into the other. Thus the heat pumping action is provided without compression. Grossman estimates 12.2 kg/kW specific mass for the generator, evaporator, recuperator, and auxiliary parts of the absorption heat pump.

To upgrade the waste heat to 420 K, the ideal Carnot coefficient of performance is 3.5 (thermal watts moved for each watt consumed by the heat pump). Assuming 60 percent of Carnot efficiency is achievable for a real system, the coefficient of performance becomes 2.0. Thus, to reject 20 kW of cabin waste heat at 300 K, 10 kW is consumed by the thermal management system, resulting in 30 kW radiated at 420 K.

Combining the absorption heat pump with a metal/ceramic fabric heat pipe system will provide a very reliable, relatively low mass heat rejection system with a total estimated system mass of 477 kg.

Nighttime Operation

During night operation there is no solar heat radiation and the lunar surface temperature drops significantly, to about 100 K. With this reduced sink temperature, the radiator can operate at a much lower temperature to reject the heat generated in the rover. Figure 12 shows the necessary radiator temperature required to dissipate various heat loads versus the lunar surface temperature. For nighttime operation and a 30-kW load, the resulting radiation temperature is 375 K. Although the heat pump is used to remove the heat, less power is required than during daytime operation.

Emergency Operation

Sometimes it may be necessary to operate the thermal management system under emergency conditions. Assuming a loss of power in which the rover must rely solely on emergency energy storage, all nonessential electrical consumption will be curtailed, thereby significantly reducing the thermal load for the management system. During the day this will not pose a significant problem, since the radiator reaches equilibrium at 300 K with a heat load of 3 to 5 kW. During the night, very low power dissipation

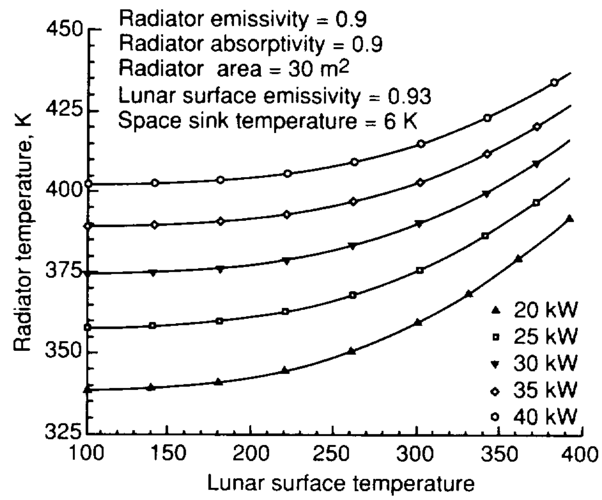


Figure 12. Radiator temperature versus lunar surface temperature.

causes the radiator temperature to drop significantly below room temperature. However, if the radiator is isolated from the cabin (by turning off the thermal management fluid loop), sufficient rover insulation would keep the rover warm enough with minimal heat dissipation within the cabin.

Mission Summary and Hardware Requirements

Lunar rover exploration expeditions are designed to enhance activities conducted at the base. Capability must be provided to permit exploration over a significant distance from the base with equipment for conducting a wide range of research experiments over a period of 1 lunar day (within 30-Earth-day design capability). To determine the types of experiments and hardware necessary for remote exploration, manifests based on the NASA 90-day study and proposed Space Exploration Initiative (SEI) science payloads (ref. 36) were evaluated to determine the mass, volume, and power requirements for exploration activities.

The scenario assumed for this lunar rover mission provides astronauts the freedom to observe and evaluate phenomena without the need to perform significant manual labor. Small rovers and robots developed during other missions are incorporated and adapted to this exploration mission. Teleoperated minirovers will be available to collect samples in the vicinity and transport them back to the main rover for analysis.

Mission Classifications

Many classes of lunar-based science experiments are planned to investigate planetary science, astrophysics, space physics, exobiology, life sciences, and resources and materials use. From these experiment classes, planetary science experiments to determine resources and materials use are significantly enhanced by being conducted as a part of rover expeditions. These investigations are based on evaluation and use of lunar materials. Exploration of large lunar areas, through a series of lunar rover expeditions to characterize the resources and terrain, will significantly improve selection of sites for bases, outposts, and in situ resource development facilities.

Geophysical investigations. These investigations allow astronauts to conduct geologic field work such as mapping and sample collection. Samples will be collected near or on the lunar surface. The amount of supplies necessary for these investigations is based on two fully equipped teams of two astronauts collecting a wide variety of samples in the field, including regolith cores to 10 m long, rock cores to 15 cm long, bulk rocks, rock fragments, pebbles, soil, and regolith. As a result of these investigations, astronauts will analyze samples and construct geologic field maps.

Geophysical ground sample archival. These activities will allow astronauts to return lunar samples to Earth for analysis in terrestrial laboratories or archive unreturned samples on the Moon.

Geophysical mapping. These activities allow astronauts to measure a variety of geophysical parameters in the field to provide information to the Earth about the lunar subsurface structure, local gravity and magnetics, and ranging.

Lunar environmental monitoring. These activities allow continuous monitoring of the lunar atmosphere and geophysical parameters such as heat flow, seismicity, gravity, magnetic field, and space physics conditions.

Lunar geoscience measurements. These activities will allow detailed assessment of collected samples to perform preliminary analyses or advanced scientific analyses of samples that cannot be returned to Earth in pristine condition.

Minirover. The minirover will provide extended mobility to manned exploration. The rover will perform remote sensing and collect and analyze samples. The rover is designed to carry up to 5 kg of scientific instruments and sampling equipment and travel up to 100 km on the surface at speeds up to 0.4 km/h.

Lunar gas extraction experiment. The purpose of this system is to demonstrate the extraction of gases from the lunar regolith and to determine their composition. The system has the capability to collect gases such as hydrogen, helium, nitrogen, and carbon components for potential use at the lunar outpost. If He^3 is collected, it would be exported to Earth for use as a possible fusion fuel.

Hardware Requirements

The components of the payloads for these investigations are summarized in table VIII.

Table VIII. Mission-Related Equipment Manifest (ref. 36)

Experiment	Mass, kg	Volume, m^3	Power, W
Astronaut field package	115	0.5	890
Sample return containers	145	.3	NA
Geophysics package	330	.2	50
Environmental station	70	.1	80
Geoscience laboratory	105	.2	320
10-m drill	305	.2	6120
Minirovers (5)	75	.5	20
Contingency	60	.1	380
Total	1205	2.1	7860

In Situ Resource Demonstration

To prepare for in situ resource utilization, small demonstrations must be conducted. The preferred processes to be considered are summarized in table IX. No power requirements were supplied for these activities; however, increased power to the lunar surface could significantly enhance the timing and the magnitude for demonstrating these capabilities. This increased power availability would enhance developing manufacturing capacity on the Moon to support future space exploration.

Lunar Rover Conclusions

The present lunar rover study, although using a power resource approach different from other recent studies, has otherwise been quite conventional. That is, it has been implicitly assumed throughout that minimization of average power used will result in the optimal (lowest mass) rover. However, the issue of an optimal rover design acquires a new perspective with the use of power beaming. At a small incremental cost in the beam receiver antenna and related electronics, the rover could potentially have available what is now termed maximum power—but continuously, if required, with no time limit on its use. Given

Table IX. In Situ Resource Demonstration Activities

Activity	Preferred process	Application
LLOX generator	Ilmenite	Water, life support, emergency repressurization
Metals coproduct extractor	Carbonyl	Power metallurgy, rods, zeolite
Ceramic by-products unit	Consolidate/sinter	Sintered shades, glass fibers, radiation protection walls
Volatiles unit	Thermal	Helium, hydrogen, carbon, nitrogen

the many unknowns that a lunar rover may encounter during its missions, it would seem prudent to provide it with extended power capabilities and to design for maximum power use to cover unforeseen events and opportunities.

The laser-beam-powered rover design presented in this paper has the following specific characteristics:

- With a total mass of only 8000 kg, the rover can take a crew of four on 30-day lunar exploration missions, traveling at speeds of up to 10 km/h with very few restrictions with respect to maximum incline and soil composition.
- There are no constraints on when, how long, or how far the vehicle can travel within the 30-day mission duration and no restrictions on day or night operations.
- The drive system has inherent reliability with four separate drive motors and employs a lightweight fly-by-wire approach.
- An onboard energy storage system uses fuel cells to account for routine power consumption variations, which can be used for backup emergency power.
- Space Station *Freedom* international racks are used for mounting equipment and instrumentation and for storage. This provides commonality with available flight-qualified hardware.
- Redundant systems are included to ensure crew safety.
- An open life support system is used to minimize mass, complexity, and cost.
- Water, waste water, and methane storage (from atmospheric CO_2 removal) are used to augment aluminum radiation shielding.
- The vehicle is relatively spacious, with 30 m^3 of free volume.
- An advanced thermal management system provides the necessary heat rejection with minimum mass.
- A multitude of mission-related devices and instrumentation, rivaling the capability envisioned for an initial lunar outpost, are included in the manifest.
- Mission activities are enhanced by a fleet of battery-powered minirovers capable of performing remote exploration missions.
- Abundant energy availability allows potential reduced mass designs for systems and equipment onboard the rover.

Several other features of power beaming that enhance rover capabilities should be mentioned. Because power is not generated onboard the rover, use of a heat pump to increase radiator temperature and effectiveness is a credible option. Furthermore, when rover power requirements are lower than available in the power beam, some form of defocusing could be used to match the incoming power with rover requirements. This would serve to minimize thermal radiator mass and area, as only heat from active components and equipment would need to be rejected.

The power-rich environment available allows the mission planners the flexibility to include a wide variety of missions, which may be both heavy and power intensive. The capabilities of the beam-powered rover are enhanced to the degree that it can outperform even the planned conventional lunar base concepts. Consequently, such a rover would be an excellent candidate for a Rover First architecture, in which the early lunar outpost is a manned rover.

Part C: Comprehensive Summary

This study describes two systems for transmitting power by laser beam from the L1 and L2 Lagrange points to the lunar surface. The systems consist of satellites and a rover on the lunar surface. The systems differ in the two satellites used. The primary difference between the two satellites is their prime power source. One satellite is powered by a nuclear reactor, the other by solar photovoltaics (PV). The nuclear reactor is more massive than the photovoltaics and generates thermal and nuclear radiation, but the photovoltaics require continuous solar tracking, which may adversely affect satellite pointing and tracking.

Both satellite versions, except for the prime power sources, are almost identical. The most massive component of the systems is the prime power source (PV 6300 kg, reactor 7900 kg). The second most massive component is the transmitter dish (PV 2800 kg, reactor 3200 kg).

The main advantage of this satellite system over other satellite systems is that 49 percent of the lunar surface is accessible all the time from either L1 or L2. A satellite closer to the Moon views less surface and, since it must orbit to stay aloft, can access a particular location only briefly. At least three such orbiting satellites would be required to provide continuous coverage to a particular location.

Use of a power-beaming satellite considerably lessens the mass of the rover and its power requirements while extending mission time and distance. Use of the L1 satellite system in particular makes it possible to start one or more missions anytime in one hemisphere of the Moon during the 7-yr lifetime of the satellite with the possibility of transitioning to operation in the other hemisphere from L2.

The rover is a large manned vehicle, of approximately 8 metric tons, powered by 30 kW of electrical power, which it receives by laser beam from a satellite. The power is used for locomotion (15 kW), environmental sustenance within the rover (16.5 kW maximum), scientific tasks (10 kW maximum), and energy storage (2 kW) aboard the rover for emergency use. The rover is roughly 7.5 m long by 3.2 m in diameter. It can travel at a rate of 10 km/h over very rough terrain and up inclines as great as 30°. The range of the rover (1000 km out and back) is determined mainly by consumables. Ninety percent of consumables is water. The environmental system recycles some consumables.

The rover mission involves experiments in planetary science, astrophysics, space physics, exobiology, life sciences, and resources and materials utilization. However, its primary mission is exploration, which is extremely important for estimating lunar resources—especially regarding the selection of habitat sites. The proposed power transmission system allows exploration almost anywhere in either hemisphere of the Moon. This concept has been referred to as "Rover First."

The rover may also be used to establish a network of power way stations that could receive and store energy from satellite power transmission during times that rovers do not require such power. (The way stations could eventually enable simpler rovers and different exploration strategies.) Traveling 200 km/day and using remote laser mass spectrometry, such a rover could survey about 30 percent of a hemisphere for minerals within its 7-yr lifetime.

The power satellite can beam its power not only to lunar surface entities, but also to other satellites orbiting the Moon, satellites orbiting the Earth, or space vehicles within a radius of tens of thousands of kilometers. This is a power variable that could engender or be a key factor in missions not yet conceived.

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References

1. Wilson, J. W.; and Lee, J. H.: Modeling of a Solar-Pumped Iodine Laser. *Virginia J. Sci.*, vol. 31, Fall 1980, pp. 34–38.

2. Williams, M. D.; and Conway, E. J., eds.: *Space Laser Power Transmission System Studies.* NASA CP-2214, 1982.

3. De Young, R. J.; Walker, G. H.; Williams, M. D.; Schuster, G. L.; and Conway, E. J.: *Preliminary Design and Cost of a 1-Megawatt Solar-Pumped Iodide Laser Space-to-Space Transmission Station.* NASA TM-4002, 1987.

4. Williams, M. D.; Kwon, J. H.; Walker, G. H.; and Humes, D. H.: *Diode Laser Satellite Systems for Beamed Power Transmission.* NASA TP-2992, 1990.

5. De Young, R. J.; Williams, M. D.; Walker, G. H.; Schuster, G. L.; and Lee, J. H.: A Lunar Rover Powered by an Orbiting Laser Diode Array. *Space Power*, vol. 10, no. 1, 1991, pp. 103–127.

6. Schock, A., Sankarankandath, V.; and Shirbacheh, M.: Requirements and Designs for Mars Rover RTGs. *IECEC-89—Proceedings of the Twenty-Fourth Intersociety Energy Conversion Engineering Conference*, Inst. of Electrical and Electronics Engineers, 1989, pp. 2681–2691.

7. Schock, A.; Or, T.; and Skrabek, E.: Thermal and Electrical Analysis of Mars Rover RTGs. *IECEC-89—Proceedings of the Twenty-Fourth Intersociety Energy Conversion Engineering Conference*, Inst. of Electrical and Electronics Engineers, 1989, pp. 2693–2701.

8. Clark, B. C.; Kalkstein, J. L.; and Meyer, S.: The Case for the Methanol-Powered Planetary Rover. IAF Paper 91-447, Oct. 1991.

9. Moulton, Forest Ray: *An Introduction to Celestial Mechanics*, Second revised ed. Dover Publ., Inc., c.1970.

10. Stella, Paul M.; and Kurland, Richard M.: Latest Developments in the Advanced Photovoltaic Solar Array Program. *IECEC-90—Proceedings of the Twenty-Fifth Intersociety Energy Conversion Engineering Conference, Volume 1*, American Inst. of Chemical Engineers, 1990, pp. 569–574.

11. Karia, Kris: Photovoltaic Power for a Lunar Base. *Proceedings Sixth CASI Conference on Astronautics*, Canadian Aeronautics and Space Inst., 1990, pp. 91–100.

12. Krebs, D.; Herrick, R.; No, K.; Harting, W.; and Struemph, F.: 22 W Coherent GaAlAs Amplifier Array With 400 Emitters. *IEEE Photonics Technology Letters*, vol. 3, Apr. 1991, pp. 292–295.

13. Collicott, H. E.; Coleman, J. E.; and Gottschlich, J. M.: Spacecraft Radiator Design Using Metal Matrix Composites. AIAA-90-1707, June 1990.

14. Hicks, Tyler G.: *Standard Handbook of Engineering Calculations.* McGraw-Hill, Inc., c.1972.

15. Born, Max; and Wolf, Emil: *Principles of Optics*, Fourth ed. Pergamon Press Inc., 1970.

16. Alff, W. H.: *Large Deployable Reflector (LDR).* NASA CR-152402, 1980.

17. Mikulas, Martin M., Jr.; and Hedgepeth, John M.: Structural Concepts for Very Large (400-Meter-Diameter) Solar Concentrators. *Second Beamed Space-Power Workshop*, Russell J. De Young, ed., NASA CP-3037, 1989, pp. 239–257.

18. *Means Electrical Cost Data—1990*, 13th Annual ed. R. S. Means Co., Inc., c.1989.

19. Levinson, A. A.; and Taylor, S. R.: *Moon Rocks and Minerals.* Pergamon Press, Inc., 1971.

20. Mason, Brian; and Melson, William G.: *The Lunar Rocks.* John Wiley & Sons, Inc., c.1970.

21. Smith, Robert E.; and West, George S., compilers: *Space and Planetary Environment Criteria Guidelines for Use in Space Vehicle Development, 1982 Revision (Volume 1).* NASA TM-82478, 1983.

22. Williams, R. J.; and Jadwick, J. J., eds.: *Handbook of Lunar Materials.* NASA RP-1057, 1980.

23. Beer, Ferdinand P.; and Johnston, E. Russell, Jr.: *Vector Mechanics for Engineers—Statics.* McGraw-Hill Book Co., 1962.

24. DeMeis, Richard: Control Muscle for Agile Aircraft. *Aerosp. America*, vol. 26, no. 2, Feb. 1988, p. 32.

25. Price, S.; Chun, W.; Hammond, M.; and Hubert, A.: Wheeled Planetary Rover Testbed. Mobile Robots V, Wendell H. Chun, and William J. Wolfe, eds., *Volume 1388 of Proceedings of the Society of Photo-Optical Instrumentation Engineers*, 1991, pp. 550–559.

26. Antoniak, Z. I.; Webb, B. J.; and Bates, J. M.: Testing of Advanced Ceramic Fabric Heat Pipe for a Stirling Engine. AIAA-91-3526, Sept. 1991.

27. Wise, J. A.: *The Quantitative Modelling of Human Spatial Habitability.* NASA CR-179716, 1985.

28. Davenport, E. W.; Congdon, S. P.; and Pierce, B. F.: The Minimum Volumetric Requirements of Man in Space. AIAA Paper 63-250, June 1963.

29. Griffin, Brand N.: A Mobile Habitat for Early Lunar Exploration. IAF Paper 91-628, Oct. 1991.

30. Cohen, Marc M.: Space Station Architecture. Paper presented at Man/Systems Advanced Development Review (Johnson Space Center, Space Station Program Office), May 6–7, 1985.

31. Ewert, Michael K.; and Clark, Craig S.: Analysis and Conceptual Design of a Lunar Radiator Parabolic Shade. *IECEC 91—Proceedings of the 26th Intersociety Energy Conversion Engineering Conference*, American Nuclear Soc., 1991, pp. 273–278.

32. Edwards, D. K.; and Richards, R. F.: Optimum Heat Rejection Temperatures for Spacecraft Heat Pumps. *J. Spacecr. & Rockets*, vol. 26, no. 5, Sept.–Oct. 1989, pp. 303–307.

33. Grossman, Gershon: Heat Pump Systems for Enhancement of Heat Rejection From Spacecraft. *J. Propuls. & Power*, vol. 6, no. 5, Sept.–Oct. 1990, pp. 635–644.

34. Parrish, Clyde F.; Scaringe, Robert P.; and Pratt, David M.: Development of an Innovative Spacecraft Thermal Storage Device. *IECEC 91—Proceedings of the Twenty-Sixth Intersociety Energy Conversion Engineering Conference*, American Nuclear Soc., 1991, pp. 279–284.

35. Grossman, Gershon: Absorption Heat Pumps for Enhancement of Heat Rejection From Spacecraft. *IECEC-89—Proceedings of the Twenty-Fourth Intersociety Energy Conversion Engineering Conference, Volume 1*, William D. Jackson and Dorothy A. Hull, eds., Inst. of Electrical and Electronics Engineers, 1989, pp. 51–56.

36. Budney, C. J.; Ionascscu, R.; Snyder, G. C.; and Wallace, R. A., eds.: *FY91 Final SEI Science Payloads: Descriptions and Delivery Requirements*. JPL D-7955, Revision A, Jet Propulsion Lab., California Inst. of Technology, May 17, 1991.

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This study addresses the possibility of beaming laser power from synchronous lunar orbits (the L1 and L2 Lagrange points) to a manned long-range lunar rover. The rover and two versions of a satellite system (one powered by a nuclear reactor, the other by photovoltaics) are described in terms of their masses, geometries, power needs, missions, and technological capabilities. Laser beam power is generated by a laser diode array in the satellite and converted to 30 kW of electrical power at the rover. Present technological capabilities, with some extrapolation to near future capabilities, are used in the descriptions. The advantages of the two satellite/rover systems over other such systems and over rovers with onboard power are discussed along with the possibility of enabling other missions.

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