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LUNAR AND MARTIAN HARDWARE COMMONALITY

Hubert P. Davis, Robert E. Johnson, Paul G. Phillips,
Donald S. Spear, William R. Stump, Franklin U. Williams
Eagle Engineering
Houston, TX

ABSTRACT

A number of different hardware elements were examined for possible Moon/Mars program commonality. These include manned landers, cargo landers, a trans-Mars injection (TMI) stage, traverse vehicles, unmanned surface rovers, habitation modules, and power supplies. Preliminary analysis indicates that it is possible to build a common two-stage manned lander. A single-stage, reusable lander may be practical for the lunar case, but much less so for the Martian case, and commonality may therefore exist only at the subsystem level. A modified orbit transfer vehicle was examined as a potential cargo lander. Potential cargos to various destinations were calculated for a Shuttle external tank sized TMI stage. A nuclear powered, long range traverse vehicle was conceptually designed and commonality is considered feasible. Short range, unmanned rovers can be made common without great effort. A surface habitation module may be difficult to make common due to difficulties in landing certain shapes on the Martian surface with aerobraking landers. Common nuclear power sources appear feasible. High temperature radiators appear easy to make common. Low temperature radiators may be difficult to make common. In most of these cases, Martian requirements determine the design.

INTRODUCTION

NASA's post Space Station options may include a return to the Moon and/or a manned Mars program. It may be easier to do all or part of both of these programs at the same time if some hardware can be made common. Cost savings through commonality require both lunar and Martian programs underway within five years or so of each other. Programs separated in time by more than this are much less likely to benefit from commonality because of advance of the technological state of the art.

USE OF A TRANS-MARS INJECTION STAGE FOR OTHER MISSIONS

Recent studies of a manned Mars mission identify the need for a very large chemical propulsion stage which provides the first maneuver of the trans-planetary space vehicle (Ref. 1). A "conjunction class" mission carrying about 340 metric tons of mission module and Mars landing vehicles (two) required a "Trans-Mars Injection" stage propellant load near the capacity of the Shuttle external tank (ET)--about 640 metric tons of hudrogen/oxygen propellant--and needed the engine thrust provided by a single high expansion ratio variant of the Shuttle main engine.

This led to the conceptual synthesis of a stage which was assembled and checked out on Earth, launched into the Space Station orbit as the Shuttle ET (i.e., using its propellant to power the STS), placed into LEO by a direct insertion ascent profile, then reconfigured and refueled at the Space Station. This concept has the advantage of eliminating the effort otherwise needed in LEO to assemble and test a modular tankage vehicle of the same class.

The possible utility of this large space propulsion vehicle for missions to lunar orbit, the lunar surface, and the several candidate future missions between the (500 km) Space Station orbit and GEO-stationary orbit is explored here in a tentative way. Figure 1 shows the original manned Mars concept and the modified TMI stage in a lunar lander configuration.

TMI Stage Mass Properties and Engine Performance

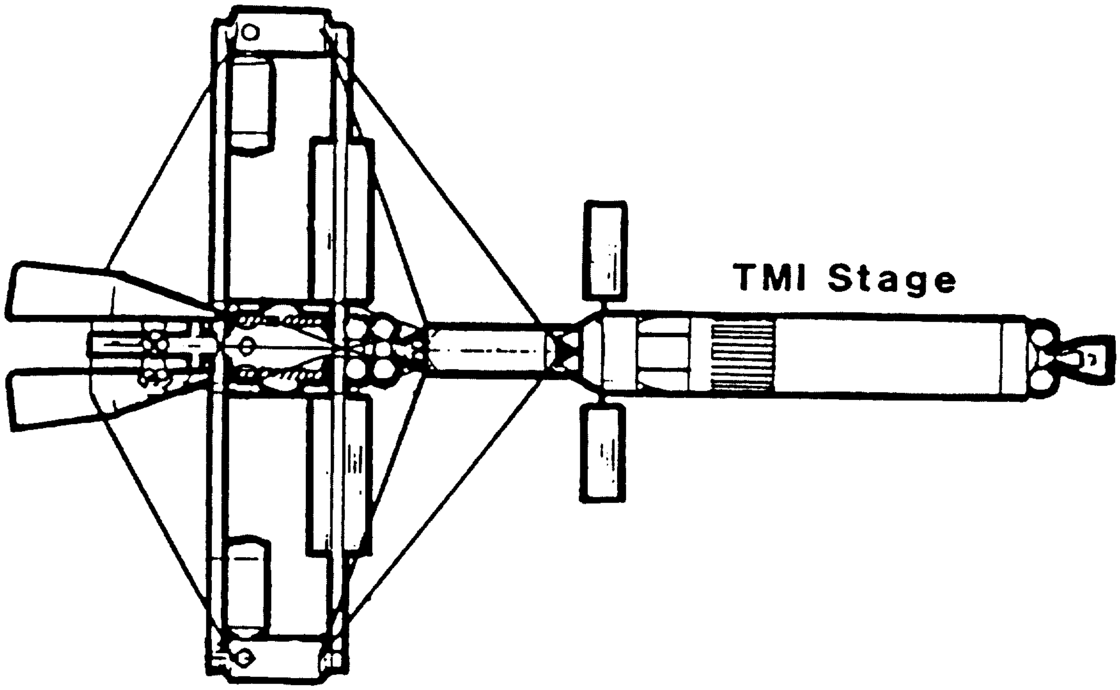
The elements and their estimated masses (Ref. 2) and the main propulsion system (MPS) engine performance are shown in Table 1.

TMI Stage Mission Performance

Velocity increments and mission performance for several all-propulsive missions are shown in Tables 2 and 3, respectively. The performance for each of these missions was calculated only for the case where the initial payload was the same as the payload for all subsequent mission phases. Obviously, other mission/payload combinations and partial propellant loading are possible. Additionally, full or partial re-use of the stage can be accomplished, rather than expending the stage. It may prove preferable to recover only the propulsion/avionics components and replace the propellant container for each mission with a once-used ET.

The use of this means of transporting mass from the Space Station would require a large-scale space program to provide enough mission demand to justify the initial investment and to develop the logistics capability to modify the stage in orbit and reload it with propellant. A lunar surface base is one type of program that might require its large

Figure 1 - TMI Stage
Original Manned Mars Spacecraft Concept



Modified TMI Stage Landing 175 MT on Lunar Surface

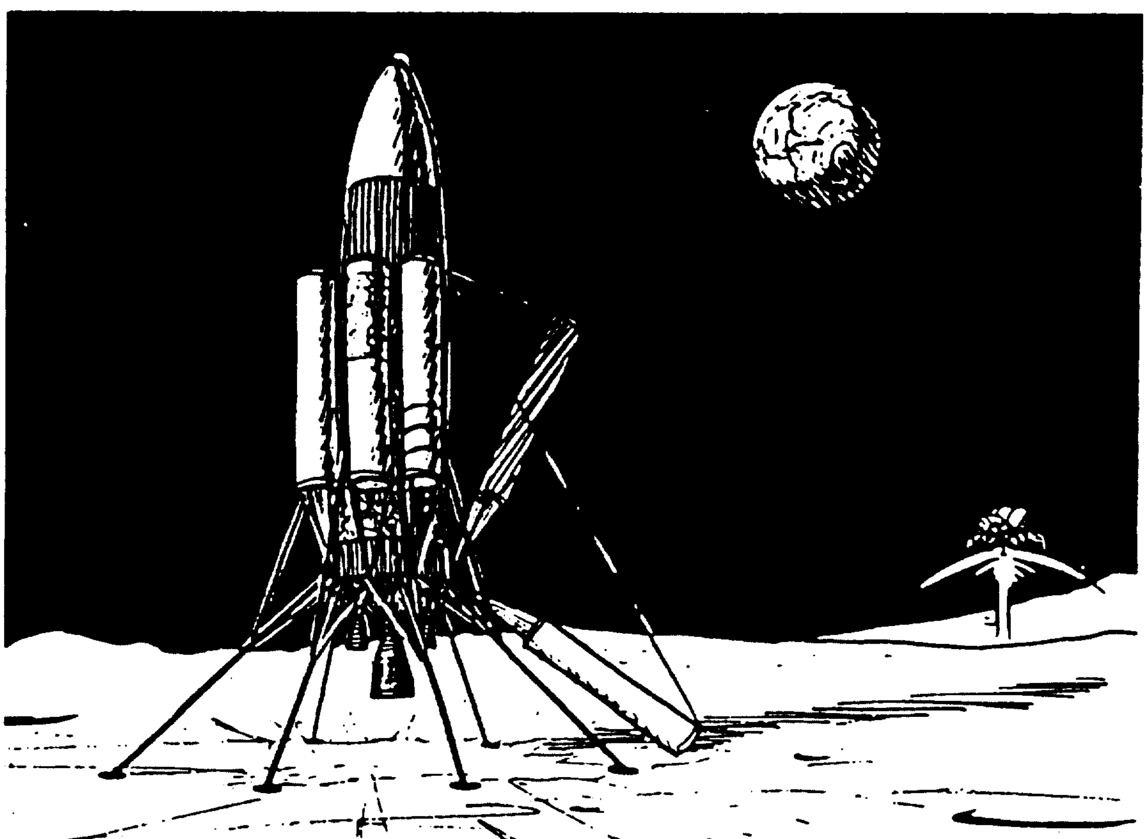


TABLE 1 - TMI STAGE OTV MASS PROPERTIES

Subsystem	Basic Mass launched	OTV Added in space	Total
"Light weight" ET	30,840		30,840
OTV MPS--1 SSME, high e	3,400		3,400
2 RL10's for roll control	340		340
Basic avionics suite	540		540
Propellant lines & valves	680		680
Helium pressurant system	910		910
Thrust truss	1,000		1,000
Attitude control sys. (dry)	410		410
Pyrotechnics & separation	230	50	280
Passive thermal control	540	1,810	2,350
25% Reserve--new items	1,080	460	1,540
Subtotal--dry mass	39,980	2,320	42,300
Unusable fluids			
Helium pressurant		680	680
Attitude control residuals		110	110
MPS residuals @ 1/4 %	1,770		1,770
Flight performance reserves	1,770		1,770
Subtotal	3,540	790	4,330
STAGE END--BURN MASS			46,630
USABLE FLUIDS			
MPS propellant		707,600	707,600
Attitude control propellants		2,270	2,270
Fuel cell reactants		540	540
Purge helium		1,130	1,130
Subtotal--consumables		711,540	711,540
START BURN MASS			758,170
"Mass Fraction"			0.933

Main propulsion system steady-state I_{sp} 470 seconds
Mission effective specific impulse 467.4 seconds

TABLE 2 - TMI STAGE OTV PROPULSIVE MANEUVERS, M/SEC

MISSION	#1	#2	#3	#4	#5	Total
Translunar Insert.	3,200					3,200
LEO - Lunar Orbit	3,200	30	1,070			4,300
LEO - L. Surface	3,200	30	1,070	90	2,040	6,430
LEO - LO - LEO	3,200	30	1,070	1,070	3,230	8,600
LEO - GTO	2,530					2,530
LEO - GEO	2,530	30	1,710			4,270
LEO - GEO - LEO	2,530	30	1,710	1,710	2,560	8,540
Planetary	3,200	3,200				6,400

LEO--Low Earth Orbit, GEO--Geosynchronous orbit, GTO--Geo-transfer orbit

TABLE 3 - TMI STAGE OTV MISSION PERFORMANCE

MISSION	Total dv	Mass Ratio	MR-1	Final Mass	Payload	Notes
TL Insert	3,200	2.011	1.011	700,090	653,460	
LEO - LO	4,300	2.555	1.555	455,100	408,460	
LEO - LS	6,430	4.070	3.070	230,480	183,850	Less lndg. gear
LEO - LORT	8,600	6.528	5.528	114,410	81,380	Round trip
LEO - GTO	2,530	1.737	0.737	960,150	913,520	Requires AKM
LEO - GEO	4,270	2.538	1.538	460,110	413,480	Less boil-off
LEO-GEORT	8,540	6.442	5.442	130,050	83,410	Round trip/boil-off
Planetary	7,770	5.455	4.455	158,860	112,230	High C3

payload capability. This stage could deliver a payload of about 175 metric tons from low Earth orbit directly to the Lunar Surface Base--perhaps necessary for economic placement of a large-scale, self-contained, highly automated lunar surface oxygen facility.

Another example would be the placement in GEO of a large, highly-shielded space station for GEO service crew habitation. The payload of 408 metric tons would permit several space station "common modules," radiation barriers, a large nuclear-electric power supply, and significant operational capability to be emplaced in one flight.

If large, relatively near-term space projects are contemplated, this system is a candidate. The Centaur 'G', now under development for the Galileo and other missions, may evolve into the workhorse OTV of the 22,700 to 68,200 Kg propellant class by use of drop tanks mounted at the Space Station. The first new OTV could possible be one of very large payload class such as this TMI stage, if a demand for such large payload delivery capability develops.

MANNED LANDERS--DESIGN CONSIDERATIONS

The use of a common or nearly-common vehicle for performing manned landings on Mars and the Moon would require that the vehicle meet several disparate performance and environmental requirement sets. The difference in gravitational attraction will dictate that different engine thrust levels be available in order to perform hovering flight and near-surface translational maneuvering. For Mars, an engine that can throttle over a wide range or some engines that are not used on the Moon may be required.

The Mars landing vehicle must accommodate entry heating and will almost certainly employ aerobraking to reduce the descent propulsion system size and mass; the lunar vehicle descends through a near vacuum and is untroubled by descent heating, but cannot make use of aerobraking. The presence of an atmosphere on Mars will cause the dust cloud raised by the terminal descent to persist and envelop the vehicle, whereas on the Moon, disturbed surface particles follow a ballistic path and do not dwell about the vehicle.

The different gravity fields of Mars and the Moon must be considered for every aspect of the lander crew interface--from physical support to egress and ingress. The Mars surface suit may have to be umbilical-supplied for makeup gas, coolant, and power, as otherwise, the pressure suit and backpack may be too heavy on Mars for a human to stand unsupported or gain the necessary mobility on foot. A long duration suit may weigh several hundred Earth pounds.

The texture of the surfaces may be sufficiently different that different landing gear design become necessary. Heat rejection on Mars cannot use the Apollo-era ice sublimator. A compression cycle "heat pump" coupled with external convectors or space radiators are needed for Mars; however, the dust problem of Mars must be carefully considered in assessing heat refection devices.

General arrangement may take several forms for either a Mars or a Moon landing vehicle. The requirements for ascent from the surface of Mars, however, are much more severe than from the Moon, such that a dedicated ascent stage with drop tanks for Mars ascent appears necessary. A lunar vehicle which uses hydrogen/oxygen propellants can descend and ascend with the stage. Thus, a common design for Mars and lunar landers does not look reasonable, if optimum performance for each is desired.

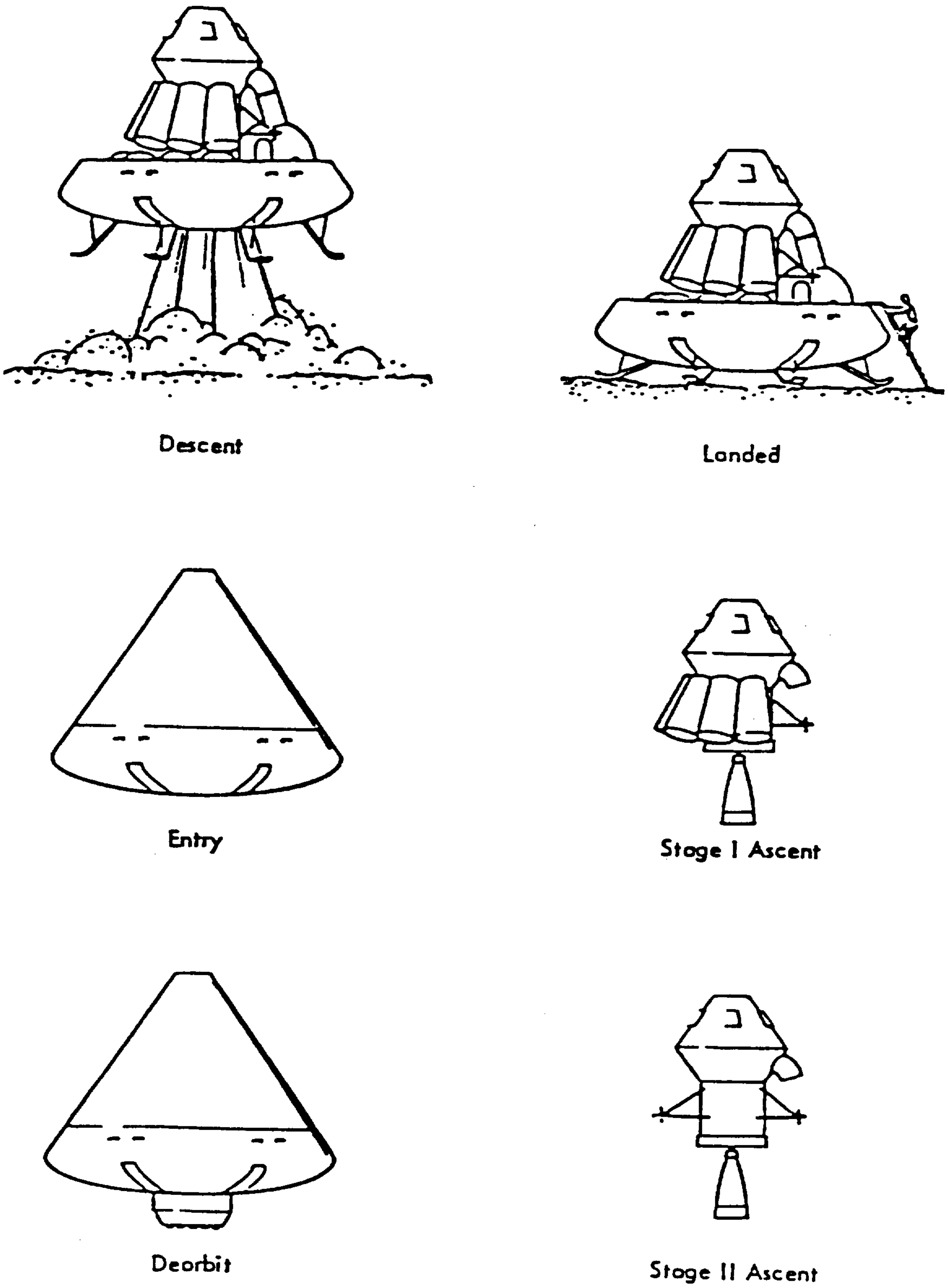
In spite of these problems, a common lander might be designed that would be primarily a Mars lander, capable of lunar landings. Table 4 shows the basic characteristics of a two and one-half stage lander, as it might be configured for Mars only and the Moon only, and the common configuration configured for both Mars and lunar missions. The Mars lander or Mars Excursion Module (MEM) design drives the lunar lander. The common lander in the lunar configuration does not carry the ascent drop tanks and the afterbody shroud, but is still assumed to carry the heat shield and the large tanks required for Mars first-stage ascent. Surprisingly, the descent propellant required for both cases is about the same, so there is no penalty associated with a common descent stage. All four of these landers carry a crew of 4 with 60 days life support, have an aerodynamic L/d of .5, a length of 8 meters, and a diameter of 9 meters. They will all look something like Figure 2, but the lunar version can be stripped of outer shell. All versions carry a 3.3 metric ton storm shelter. All versions use liquid oxygen/monomethyl hydrazine propulsion for both ascent and descent. This fundamental design is described in more detail in references 3 and 4.

TABLE 4 - LUNAR/MARS MEM CHARACTERISTICS

	MARTIAN ONLY MEM	LUNAR/MARS MEM (MARS CONFIG.)	LUNAR/MARS MEM (LUNAR CONFIG.)*	LUNAR ONLY MEM
DEORBIT MASS, MT	70.90	71.24	51.38	46.21
ASCENT 2ND STAGE PROPELLANT, MT	4.20	4.20	0.00	0.00
ASCENT 1ST STAGE PROPELLANT, MT	18.70	18.70	3.54	2.47
TOTAL DESCENT PROPELLANT, MT	22.10	22.20	24.95	22.44
ASCENT STG. TANKS & SYSTEM, MT	1.31	1.31	1.31	0.17
ASCENT CAPSULE (LESS PROPUL.),MT	2.42	2.42	2.42	2.42
DESCENT STAGE (LESS PROPULSION)	18.96	18.97	16.05	15.90
LANDED CARGO, MT	1.91	1.91	1.91	1.91
ASCENT CARGO, MT	0.14	0.14	0.14	0.14
CONSTANT ASCENT THRUST, KLBF	40	40	40	40
ASCENT THRUST/WEIGHT, KLBF/KLBM (MARS OR LUN. WEIGHT, STRT BURN)	1.75 (MARS)	1.75 (MARS)	14.35 (MOON)	20.56 (MOON)
CONSTANT DESCENT THRUST, KLBF	120	120	120	120
DESCENT THRUST/WEIGHT, KLBF/KLBM (MARS OR LUN. WEIGHT, END BURN)	2.87 (MARS)	2.85 (MARS)	11.87 (MOON)	13.20 (MOON)
ASCEND TO, KM (AND DESCEND FROM)	500x32,963 (24 hour)	500x32,963 (24 hour)	200x200 (circular)	200x200 (circular)
PROPULSION ISP, SEC	360.5	360.5	360.5	360.5
ASCENT STAGE 2 DELTA V, KM/SEC	2.66	2.66	0.00	0.00
ASCENT 1ST STAGE DELTA V, KM/SEC	3.43	3.43	1.92	1.92
DESCENT DELTA V, KM/SEC	1.23	1.23	2.17	2.17

* DROP TANKS AND AFTERBODY REMOVED

Figure 2 - Rockwell Lander with MSFC Updates (taken from REF. 2)



MEM - MISSION PHASE CONFIGURATIONS

MANNED LANDERS--DEVELOPMENT AND TESTING

Reference 4 (the last major MEM study, done in 1967) considered testing of a Mars-only lander on the lunar surface as an option in a development and test program that would also include an unmanned Earth entry test and several manned tests in Earth orbit. The MEM heat shield was to be built to withstand Earth entry, such that an unmanned Mars test could be avoided. A lunar landing of a Mars-only MEM is not a requirement for a test program and would, in reality, be a small additional lunar program with some testing benefit. Given the advance in technology since 1967, it may now be possible to test a Mars lander with perhaps one- or two-manned Earth orbital and entry flights.

OTV DERIVED CARGO LANDER

The Johnson Space Center has performed a conceptual design of an aerobraking orbit transfer vehicle (OTV) which uses the heat shield structure to support the propellant tanks and other vehicle systems. This 1990's vehicle concept is described in NASA TM 58264, March 1985. An attempt was made to adapt this space-based LEO to GEO and return vehicle to the task of landing on the surface of either Mars or the Moon while carrying an unmanned, or "cargo," payload. Figure 3 illustrates the original vehicle and its modified version landing on Mars.

Although no analysis has yet been performed, early indications are that a common heat shield may be designed for the LEO-GEO-LEO and Mars aerobraked landing missions. Placement of cargo and landing gear on the JSC OTV concept is complicated by the fact that the flight path during engine thrusting is approximately normal to the aerobraking path. This may be manageable by careful placement and attachment of both cargo and landing gear. Gimbal travel and engine throttling may be needed to an unprecedented extent to maintain stability. The space-based OTV may be reduced in mass for any mission not encountering an atmosphere, including lunar landing, but the basic structural arrangement must remain for the Mars and lunar vehicles to be considered "derivatives."

OTV DERIVED LANDER--MASS PROPERTIES AND ENGINE PERFORMANCE

The JSC OTV has an inert mass cited as 5,032 kilograms to house 38,000 kg of O_2/H_2 propellant which can transport a 7,120 kg manned crew module round trip from LEO to GEO and return. As both the lunar landing and Mars landing missions are much more sensitive to inert mass than are

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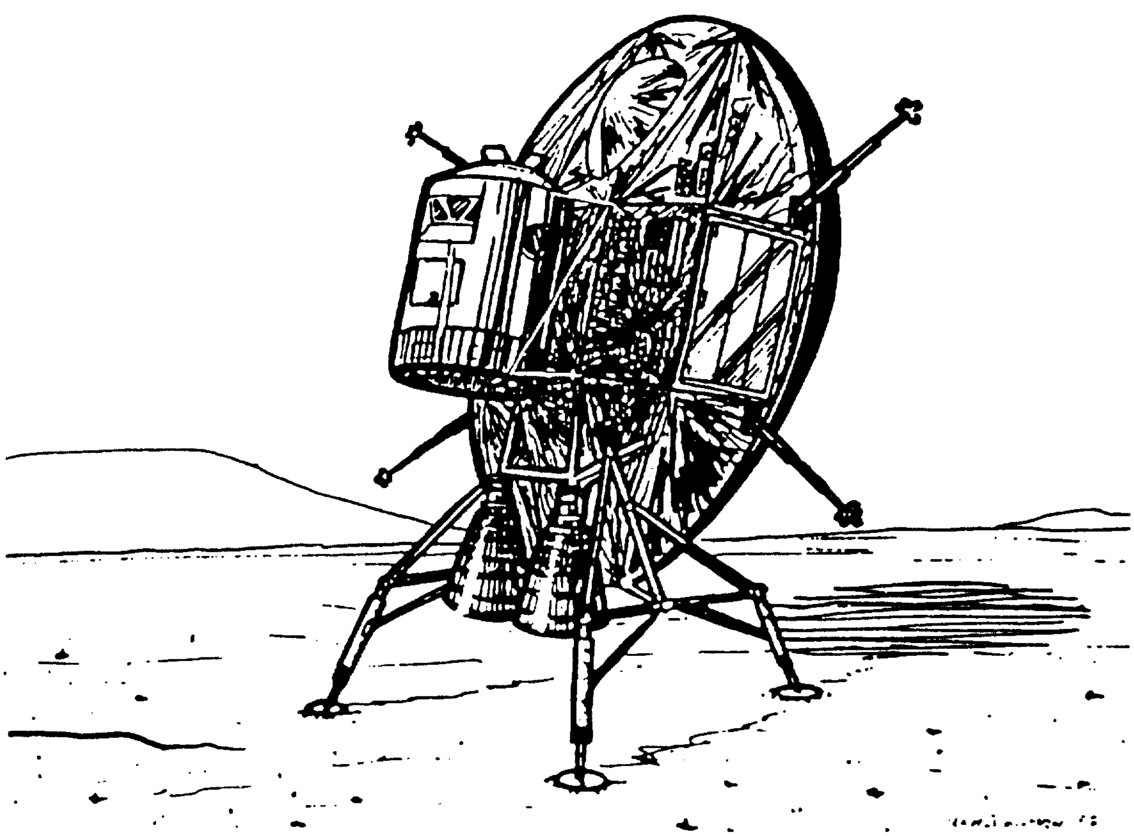
Figure 3 - JSC OTV
Aerobraking into Low Earth Orbit

Modified Cargo Lander on the Martian Surface



GEO transfers, the "first pass" mass property estimates given in Table 5 for the lunar and Mars derivatives presume a composite structure OTV with a refined 25% lighter thermal protection system (TPS). In addition, a second generation main propulsion system (MPS) engine rated at 48,900 N thrust is assumed to be available, permitting the use of three engines, rather than two, and producing a higher delivered specific impulse than the RL 10 11B. Both of these assumptions for the "derivative" vehicles are consistent with their somewhat later need date than the space-based LEO-GEO vehicle. For the lunar case, a 973 m/sec delta V was used for lunar orbit insertion and 2,100 m/sec for descent. For the Mars case, 1,230 m/sec was used for de-orbit and descent (aerobraked).

OTV DERIVED CARGO LANDER PERFORMANCE--LUNAR MISSION

With the relatively low energy requirements of this mission, which originates in low lunar orbit with topped-off propellant tanks, the OTV-derived lander appears to have the performance capability to land from low lunar orbit with an unmanned payload of over 55 metric tons. There is considerable doubt, however, that a side-mounted payload of this magnitude can be balanced by engine gimbaling. Also, about 40% more engine thrust would be necessary in order to hover. A more appropriate mission mode may be to have the cargo lander perform its own lunar orbit burn, leaving the cis-lunar OTV only the task of establishing the trans-Moon trajectory and returning to the Space Station. When the vehicle is required to accomplish the insertion into lunar orbit, the payload is reduced to 31.9 metric tons, which is probably more than adequate for base buildup and resupply.

OTV DERIVED CARGO LANDER PERFORMANCE--MARS MISSION

The same comments apply here, only with a great deal more force than for the Moon due to the higher gravitational field and indicated payload for the Mars landing mission. Over 84 metric tons is neither required nor feasible, so again we must look for additional propulsion tasks for the Mars landing vehicle. If cryogenic propellant insulation can truly accommodate nine- to ten-month missions, an interesting possibility may be for the Mars cargo lander to proceed independently of the manned planetary vehicle and pre-position its cargo at the desired landing site, conducting its entire mission independent of the principal Mars space vehicle.

TABLE 5

ESTIMATED MASS PROPERTIES--OTV DERIVED CARGO LANDER
Estimated Mass in kilograms

SUBSYSTEM MASS	BASIC OTV	LUNAR DELTA	LUNAR LANDER	MARS DELTA	MARS LANDER	NOTE #
TPS	351	-263	88	11,592	11,943	#1
Structure--TPS	281	-140	140	9,216	9,496	
Structure--Other	510	127	637	170	680	
Payload Accom.	265	40	305	133	398	
Internal insulat.	239	60	298	79	318	
Power & Distrib.	218	54	272	73	290	#2
Reaction Control	204	153	357	204	407	
Avionics	210	63	274	63	274	
Tankage	1,301	0	1,301	0	1,301	
Main Engines	841	-136	705	-136	705	#3
Landing Gear	0	1,270	1,270	5,511	5,511	#4
Other	0	0	0	0	0	
Subtotal--Dry Mass	4,420	1,228	5,647	26,904	31,324	
Residual & Reserve Propellants	610	0	610	0	610	
Contingency Inerts	880	250	1,130	5,380	6,260	
EOM Inert Mass	5,910	1,478	7,387	32,284	38,194	
Payload (manned)	6,800		31,900	unmanned	84,400	#5
Total EOM Mass	12,710		39,280		122,600	
Re-entry Mass	13,020		39,280		122,600	#6
Usable Propellants	38,100		38,100		38,100	
Boil-off	190	-130	50	-100	80	#7
RCS Propellants	340	1,020	1,360	1,360	1,700	
Fuel Cell Reactant	120	120	240	150	280	#2
Start-burn Mass	51,770	1,000	79,040	1,400	162,750	
Mass Ratio	4.072				1.328	#5
"Mass Fraction"	0.847		0.808		0.486	
I_{sp} (Mission Effec.)	470.1		458.3		453.7	#8

NOTES:

1. 75% TPS removed for lunar, 15% of extra payload added for Mars
2. Assumes "power on" from internal power for OTV transfer/coast
3. Assumes new technology engines reduce dry mass/unit thrust
4. Assumes 3.5% of landed mass for lunar, 5% for Mars
5. Lunar & Mars landed cargo is found by iteration
6. No apogee raise maneuver for lunar on Mars landings
7. Assumes passive thermal control, top-off before deployment
8. Estimated by multiplying the steady state I_{sp} (483 sec) by the ratio of useful propellants to total fluids consumed, then deducting 1% for stop-start losses

SURFACE HABITATS

Significant design criteria such as crew size, stay time, pressurized volume, maximum weight, or dimensional limits for a Mars/Lunar Surface abitat Module (SHM) have not been defined as yet; therefore, the discussion that follows is based on assumptions which may or may not be applicable to later, more refined studies. As an example, the selected SHM shape (i.e., cylindrical vs hemispherical) obviously affects module weights considerably, but the shape will be dictated by other considerations such as SHM function, launch vehicle characteristics, orbit-to-surface delivery mode, etc.

Also, meeting program goals of minimizing development and testing costs can dramatically influence the design and manufacturing approach of the SHM. Interest in the evaluation of Space Station Common Module (CM's) for the SHM role stems primarily from this consideration. The Space Station CM used to consider Mars vs. lunar Surface SHM's is taken from reference 4.

MARS SURFACE HABITATION MODULE

The requirements for a Mars SHM are generally more severe than those for a lunar SHM in that the natural environment imposes more design complications. To illustrate, a comparison of the environmental factors affecting structural design is given in Table 6.

TABLE 6

ENVIRONMENTAL FACTORS AFFECTING SHM (TYPICAL)

FACTOR	EARTH	MARS	MOON
Atmospheric Pressure (mb)	1000	7-9	-0-
Temperature (^{o}K)	300	215-280	220
Soil Density (g/cc)	-	3.9	1.0-1.6
Gravity	1.0 g	0.38 g	0.165 g

In addition, the composition of the Mars surface material, combined with occasional storms with winds up to 100 mph, can create a significant erosion hazard. Although radiation protection from the SHM structure will be significant, for long times, the statistical probability of solar flares will likely require additional shielding or safe areas for crew protection. It is believed that the micrometeoroid hazard is somewhat less on Mars than on the Moon, but again, it is likely that additional protection would be needed if the SHM structure is designed to handle pressure, landing, and intertial loads only and deployed on the surface of Mars.

These concerns have led other studies to examine ways of burying or covering the SHM on the Mars surface. Such an approach appears feasible with considerable benefits. However, this will require considerations of budkling and local instability if the current family of Space Station CM's is considered. Certainly, there are ways of encapsulating the SHM without loading the skin structure with the overload from the Mars surface material. Such approaches as boring a self-supporting tunnel or trenching and erecting a roof structure could mitigate the penalty associated with CM re-design.

The long Earth-to-Mars transit time and the requirement for entry thermal protection would appear to cause significant differences in the Mars and the Moon SHM's. If needs for cxommonality in design are significant, these problems can be solved by deployable, single-use shields or panels. Trades of weight, cost and complexity of this approach versus separate SHM designs will be required to determine which approach is better.

Structurally, except for the possible load from burying the module, the Space Station Common Module would require modifications for attach points for Earth-to-Mars transfer and for deployment loads. The structure should be slightly over-designed for pressure loads for the Mars SHM application.

<u>LUNAR SURFACE HABITATION MODULE</u>

The requirements for a lunar SHM appear to be less rigorous than for a Mars SHM; notably, lower "g" and no "entry" heating will result in more design flixibility and, very likely, a design similar to the current Space Station CM design could be used, provided transportation and functional needs are satisfied.

Thermal control, radiation hazards, and micrometeoroid protection consideration may lead to a desire to place the SHM below the lunar surface as was discussed for the Mars SHM. If so, it is recommended that

that free-standing covers support the surface material rather than penalize the SHM design for this additional long-term load.

As an example, an early NASA-JSC design for a CM called for an 0.06 inch (.15 cm) thick wall of 2219.T87 aluminum alloy. Wall thickness was set by micrometeoroid criteria rather than pressure loads or flight loads. If it is determined that the micrometeroid hazard for the lunar SHM is less than for the Space Station, some structdural weight reduction is possible; otherwise, the additional weight of the lunar soi8l if the SHM is placed below the lunar surface could likely be accommodated by most of the current CM designs. Checks for local buckling and stability around cut-outs would be needed as well as Earth-to-Moon transportation load conditions. Design modifications to accomodate hard attachment points and support pads for surface deployment will be needed.

While there are many details yet undefined, it is feasible to consider a pressure module such as the space Station CM for both Mars and a lunar SHM. Perhaps the most significant impact will be the method of delivery of the Mars SHM to the surface; the cylindrical shape may not be feasible for the aero-entry system design. As far as a manned, pressure module, the development and certification of the Space Station CM will go a considerable way to the development of a manned surface habitat.

LONG-RANGE MANNED TRAVERSE VEHICLES

Before examining the feasibility of a long-range traverse vehicle capable of operating on both the lunar and Martian surfaces, it is necessary to establish the feasibility of such a vehicle for operation on either surface. A proposed requirement for a traverse vehicle is the capability to travel from the equator to the pole and back. This allows the manned exploration of half of the planetary surface from one base, which in some opinions, must be possible befroe committing to a surface base. On Mars, this trip is 11,300 km (7,000 miles), while on the Moon it is only 5,600 km (3,500 miles). Assuming travel at an average speed of 24 kilometers per hour (15 mph) for 12 hours per day and one day of scientific activity for each travel day, the trip on Mars will last 80 days. Weight of a Mars vehicle will be twice that of a lunar vehicle. The Mars requirements drive the design.

Figure 4 illustrates a Common/Mars Traverse Vehicle design concept. The nuclear power plant is in the trailing segment and cabin systems are located in the leading segment. This configuration is an adaptation of a vehicle currently under study (Ref. 6) Table 7 is an approximate mass summary. Subsystem weights are based on reference 6 information.

The first order feasibility of this vehicle depends on the power supply. A brief examination of fuel cells indicated that this trip is too long for their use as the primary power supply. Fuel cells should be adequate for short range missions, however, and they are included in the design as a secondary power system. A nuclear power supply is used as the primary power supply. The reactor segment provided about 15 square meters of radiator surface operating at 1300^{0} K on the upper portions of the body. The reactor is sized at 100 KW electric, assuming 50 Kg per KW typical of SP-100 type reactors.

The locomotion system consists of independently suspended 72-inch by 30-inch wheels. Power is provided to each wheel by electric motors similar to the Lunar Rover Vehicles. Track and elastic loop wheel systems were considered for greater obstacle clearing capability. Track systems, however, are notoriously unreliable and heavy, and are usually better suited to solid with high cohesion (Ref. 7). Fatigue in the elastic loop material for loop wheels appears to limit the reliability to an unacceptable level (Ref. 8).

The pressurized cabin system layout is similar to the 4 x 4 MOLAB configuration described in reference 9. The airlock is contained in the aft portion of the cabin and is provided with shielding for solar flare protection. The environmental control and life support is a closed cycle system. The Martian vehicle has the option of obtaining some consumables from the atmosphere. Secondary power is provided using fuel cells. This secondary system allows limited use of the vehicle in the event of reactor failure or prior to the reactor delivery.

The structural design of the vehicle is determined by the higher Martian weight. Examination of dynamic interactdion with the respective environment is necessary. Surface separation on the Moon during obstacle clearing may be higher than on the Martian surface. Braking capabilities will also vary with the lunar and Martian weights. Since the vehicle

Figure 4 - Long Range Traverse Vehicle

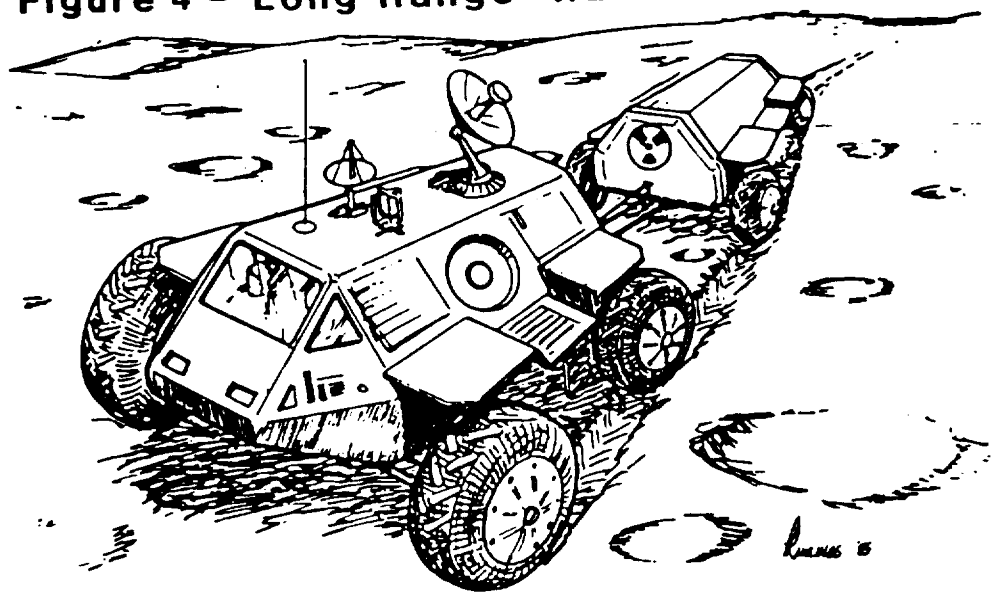


Figure 5 - Unmanned Rover

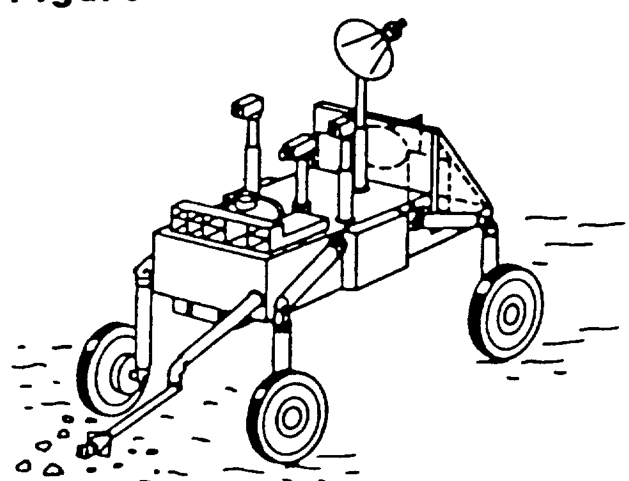
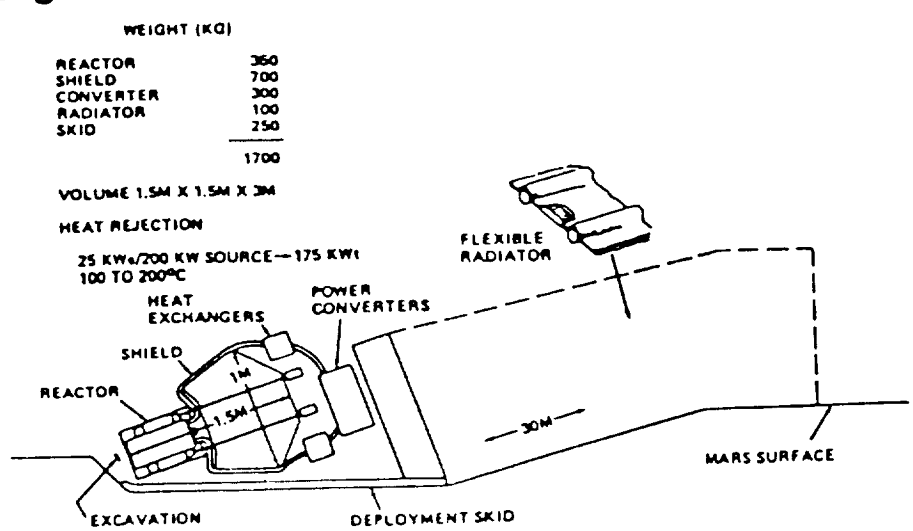


Figure 6 - Power Supply (from REF. 13)



weight is lower on the lunar surface and momentum is the same, lunar braking will be more difficult.

Differences in the soil characteristics may result in different operational capabilities. The range of angles of internal friction for each soil are approximately the same, but the Martian soil may be generally less cohesive than lunar soil. As a result, the vehicle will have greater slope climbing capability on the lunar surface.

The reactor radiators do not presesnt a commonality problem since they operate at over 600^{o} K. Study shows that radiators operating at temperatures higher than 600^{o} K do not have problems operating in either environment. Heat dissipation from the cabin segment is likely to occur at significantly less than 400^{o} K, thoughm, which may presesnt a commonality problem.

While space reactors are currently in development, the reactor used in this application will be unique. This reactor must be controllable while current applications do not appear to have wide variation in load characteristics. The high amplitude and random motion effects on the reactor that may be anticipated during Martian and lunar traverse must be considered also.

A vehicle designed for lunar and Martian long-range traverse appears feasible, assuming a nuclear reactor can be developed for this application. The vehicle will be over-designed for lunar applications, but a significant weight penalty is not anticipated.

UNMANNED ROVER VEHICLES

Unlike manned traverse vehicles, unmanned rover vehicles have received considerable recent attention. Reference 11 describes a 1984 Mars Rover initially planned for a mid-1980's follow-up to the Viking program. In addition, a vehicle very similar to this design has been baselined for the Mars Sample Return mission described in Reference 12. For the purposes of this comparison, this vehicle configuraiton will be used.

This configuration has a roughly rectangular body mounted on four articulated supports. One design uses elastic loop wheels for locomotion (Ref. 8) while the other uses wheels (Ref. 9). Figure 5 shows the wheeled vehicle.

The interaction between the terrain and the vehicle will not present capability conflicts for the same reasons described for traverse vehicles. Braking problems should be of little consequence since the vehicle will travel at slow speed.

Thermal considerations may cause problems in the use of a common vehicle since it appears that the thermal control system will operate at considerably less than 400^{o} K.

The computing system requirements may be different for the lunar application since ther period of radia contact with the vehicle will be considerably longer than for the Martian application. The Mars Rover must be capable of automated travel for periods of one day, while it is likely that the Lunar Rover can be controlled directly from Earth. A Martian vehicle baseline, however, should have exceptional operational capabilitiets on the lunar surface.

It appear that the major development requirement for this vehicle will be the image processing capabilities of the computing system. Some testing for the Mars application could be accomplished on the Lunar surface, but in reality, this would be an additional program.

Overall, the use of a common rover vehicle for the Moon and Mars appears feasible and desirable. Slightly increased design costs should far outweigh a complete repetition of the total design.

BASE POWER SUPPLY

Discussion of common electrical power systems for a mars base considers a power requirement of 25 Kwe. A Photo-PV system, including a regenerative fuel cell (RFC) and reactor (Rx) were selected for more detailed analysis. A major factor affecting the design of a common PV system is the relative solar intensity on each body. The lunar surface receives approximately 1,353 W/sq m during the day. On Mars, daytime solar intensity varies from 708.8 to 487 W/sq m, depending on the season. A nominal value of 582.8 W/ sq m, 0.43 relative to the lunar surface is specified in Table 8. This difference in solar intensity would relquire the Mars solar array to be 2.13 times larger than a solar array positioned on a lunar base.

In addition to overcoming the difference in solar flux between the Martian and lunar surfaces, the longer lunar night (18 Earth days), poses problems for a common design. Compared to a Mars RFC system, 35 times as

TABLE 7 - MANNED TRAVERSE VEHICLE ESTIMATED MASS, KGMS

TOTAL MASS	22,650
CABIN SEGMENT	12,150
Environmental Control & Life Support	1,140
Other Crew Systems	390
Food	130
Experimental Equipment	170
Communication	50
Navigation	40
Data	270
Displays	90
Shielding	2,000
Secondary Power	1,900
Pressure Hull	1,920
Structure and Drive System	4,050
REACTOR SEGMENT	10,500
100 KW Reactor	5,000
Shielding	2,000
Structure and Drive System	3,500

TABLE 8 - POWER SYSTEM COMPARISON

System type	Output (KWe)	System (Kg/Kwe)	Solar array or radiator area (sq m)	Solar array mass (Kg)	RFC mass (Kg)	Total mass (Kg)
LUNAR						
Photovoltaic	25 day only	48	396	1,188	-	1,188
Regen Fuel cell	25 constant	1070	1,117	3,351	23,402	26,753
Nuclear Reactor SF-100 program	25 constant	68 to 136	3 to 26	-	-	1,700 to 3,400
MARS						
Photovoltaic	25 day only	102	846	2,538	-	2,538
Regen Fuel cell	25 constant	337	2,383	7,149	1,268	8,417
Nuclear Reactor SP-100 program	25 constant	68 to 136	3 to 26	-	-	1,700 to 3,400

many storage tanks would be required on the Moon to deliver continuous power through the lunar night. From these two factors, a common PV-RFC design is not likely.

A nuclear reactor will deliver continuous power throughout the local night. At temperatures above 600^0 K, waste heat radiator performance on the Moon and Mars is the same, making a common radiator design possible. Based on the SP-100 program, a 25 Kwe nuclear reactor was choses for consideration (Figure 6, taken from reference 13). The weight is 1700 Kg or 15 W/Kg. In this design, the reactor is towed into an existing hole or a hole made from an explosive charge. Once in place, the generation system is electrically connected to the manned base by a tether. The recessed position of the reactor and the added distance produced by the tether allow the system to use a lighter reactor shield.

The power plant consists of a nuclear reactor as a heat source, a radiation attenuation shield to protect the payload, the electric power conversion equipment, and a heat rejection system to eleminate waste heat. Power conversion is by the direct thermoelectric conversion of heat to electricity.

This design uses refractory metals in the power system's construction. Due to the reactive nature of refractory metals with the carbon dioxide atmosphere of Mars, different materials may have to be used. The metal would not be a problem on the Moon since there is no atmosphere.

Development and testing of this type of nuclear reactor is on-going in the tri-agency (NASA, DOE, and DARPA) SP-100 program. This type of system is expected to provide poer for mid-1990's missions.

REFERENCES

1. Young, A. C. (1986), Mars Mission Concepts and Opportunities, presented at the Marshall Space Flight Center Mars Workshop, June 10-14, 1985, Huntsville, Alabama.

2. Davis, Hubert P., A Manned Mars Mission Concept with Artificial Gravity, Eagle Engineering Inc., Houston, Texas, NASA JSC Contract # NAS9-17317, presented at the Marshall Space Flight Center Mars Workshop, June 10-14, 1985, Huntsville, Alabama.

3. Stump, William, R., Babb, Gus R., and Davis, Hubert P., Mars Lander Survey, Eagle Engineering Inc., Houston, Texas, NASA JFC Contract #NAS-17317, presented at the Marshall Space Flight Center Mars Workshop, June 10-14, 1985, Huntsville, Alabama.

4. Canetti, G. S., Definition of Experimental Test for a Manned Mars Excursion Module, Three volumes, Contract # NAS-6464, North Americal Rockwell Corp., Columbus, Ohio.

   Volume 1--Summary Final report, Oct. 1966-Aug. 1967, 1968, 78 pages, 68X12107*, Report # NASA-CR-65911 SD-67-755-1.

   Volume 2--Design Final Report, 1968, 571 pages, 68X11978*, Report # NASA-CR-65912 SD-67-755-2.

   Volume 3--Test Program Final Report, 1968, 256 pages, 68X12010*, Report #NASA-CR-65913 SD-67-755-3.

5. Space Station Common Module Reference Configuration Review, presentation and NASA MSFC, March 4, 1984.

6. Personal communication for Kirt Bilby, Summer Intern, NASA JSC Advanced Programs Office, July 1985.

7. Bekker, M. G., Finelli, J. P.; and Pavlics, F.; Lunar Surface Locomotion Concept in Advances, in Space and Science and Technology, <u>Vol, 9</u>, pp. 297-326, 1967.

8. Trautwein, W., A Mobile Planetary Lander Utilizing Elastic Loop Suspension, JPL Telchnical Memorandum 33-777.

9. Weatherred, C.; Wong, R.; and Haesslein, H.; The Molab 4 x 4 Configuration, in Advances in Space Science and Technology, <u>Vol. 9</u>, pp. 327-361, 1967.

10. Mitcham, G. L.; McRae, F.; and Romano, S.; The Molab 6 x 6 Configuration, in Advances in Space Science and Technology, <u>Vol. 9</u>, pp. 327-361, 1967.

11. Dobrotin, B.; French, J.; Paine, G.; and Purdy, W.; 1984 Mars Rover, AIAA 78 - 81, January 1978.

12. De Vries, J. P.,; Norton, H. N.; and Blanchard, D. N.; Mars Sample Return Mission, 1984 Study Report, September 28, 1984, JPL-D-1845.

13. Giudici, Robert J.; Electrical Power Systems for Mars, Preliminary Design Office, Marshall Space Flight Center, presented at the Marshall Space Flight Center Mars Workshop, June 10-14, 1985, Huntsville, Alabama.

14. Angelo, Joseph A. and Buden, David, Space Nuclear Power and Man's Extraterrestrial Civilization, Los Alamos National Laboratory.