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CAN WE POWER FUTURE MARS MISSIONS?

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ABSTRACT

The Vision for Space Exploration identified the exploration of Mars as one of the key pathways. In response, NASAs Mars Program Office is developing a detailed mission lineup for the next decade that would lead to future explorations. Mission architectures for the next decade include both orbiters and landers. Existing power technologies, which could include solar panels, batteries, radioisotope power systems, and in the future fission power, could support these missions. Second and third decade explorations could target human precursor and human in–situ missions, building on increasingly complex architectures. Some of these could use potential feed forward from earlier Constellation missions to the Moon, discussed in the ESAS study. From a potential Mars Sample Return mission to human missions the complexity of the architectures increases, and with it the delivered mass and power requirements also amplify. The delivered mass at Mars mostly depends on the launch vehicle, while the landed mass might be further limited by EDL technologies, including the aeroshell, parachutes, landing platform, and pinpoint landing. The resulting in–situ mass could be further divided into payload elements and suitable supporting power systems. These power systems can range from tens of watts to multi–kilowatts, influenced by mission type, mission configuration, landing location, mission duration, and season. Regardless, the power system design should match the power needs of these surface assets within a given architecture. Consequently, in this paper we will identify potential needs and bounds of delivered mass and architecture dependant power requirements to surface assets that would enable future in–situ exploration of Mars.

INTRODUCTION

Mars exploration represents one of the main pathways in the Vision for Space Exploration [1]. Mission lineup for the next two decades is identified within NASA's Mars Exploration Program, with additional input from advisory groups, such as the NASA Advisory Council (NAC), and the Mars Exploration Program Analysis Group (MEPAG) [2]. Interdependencies between NASA's Mars Exploration Program and Solar System Exploration Program are discussed in NASA's SSE Roadmap [3].

In general, potential mission architectures for Mars exploration may include orbiters and in–situ robotic platforms, eventually leading to human exploration [1]. In this paper we examine key power system trades and options, that could enable these proposed missions. We also link power system options to programmatics (namely to the various planned Mars mission architectures); and to other technologies, such as launch vehicle options, and entry, descent, and landing (EDL) technologies.

MISSION ARCHITECTURES FOR MARS EXPLORATION

NASA pathway missions are primarily driven by science, although the pathway could also include technology driven missions. The former would address science objectives, outlined in the Solar System Exploration Decadal Survey [4], and in MEPAG documents [2]. The latter would focus on technology demonstrations that could enable future missions. At present, several in–situ and orbiter missions are in their operational phase, namely the Mars Exploration Rovers (MER), a number of orbiters by NASA (i.e., the Mars Global Surveyor, the Mars Odyssey, and the Mars Reconnaissance Orbiter) and the Mars Express orbiter by ESA. Missions under development or in planning include the Scout class Phoenix lander (2007), and the large Mars Science Laboratory (MSL) rover (2009). Final selection for the next Scout mission — targeting the 2011 opportunity — is expected within a few months.

Potential next decade missions (between 2011 and 2020) could include Scout missions in 2011 and 2018; the Mars Science Orbiter (MSO) in 2013; an in–situ mission in 2016 in the form of medium or large rovers; and a Mars Multi–Lander Network mission with multiple (∼4 or more) small landers. One of the options for the 2016 opportunity could be an Astrobiology Field Laboratory (AFL) rover, as a follow on to MSL, addressing astrobiology related science objectives.

Over the past years additional concepts were also considered, including a deep drill for subsurface access to ∼50–100 m, and sub–MER class fetch rovers. Second decade technology demonstration missions (between 2021 and 2030) could include an In–Situ Resource Utilization (ISRU) landers, and other large human precursor missions. To date, these second decade missions are in an early formulation phase, thus the final mission concepts could be significantly influenced by the future direction of the Mars exploration pathway. [5]

Second and third decade explorations could target human precursor missions and eventually lead to manned in–situ missions, building on increasingly complex architectures.

Selected missions for this decade, and potential missions for the next decade and beyond are summarized in Table 1.

These listed missions are categorized as Scout, Moderate, Large and Flagship, where the classes correspond to assigned mission costs of ∼$500M, ∼$750M, ∼$1B and multi–$B, respectively. The mission lineup is strongly driven by budget allocation for the Mars Exploration Program, and could change based on Agency priorities.

Past mission studies, performed under NASA's Mars Exploration Program Division, sized some of these proposed missions, in order to understand the science and technology drivers, the corresponding mission costs, and to ultimately support NASA in the process of identifying future pathway missions. Table 2 identifies landed mass values for these mission concepts. It should be noted that the landed mass is strongly influenced by all elements of a mission architecture, including the chosen launch vehicle, landing configuration, subsystem options, science, operational scenarios (e.g., mission duration), landing site, and assumed mission cost cap. Therefore, these reported landed mass values should be considered notional and used only in the framework of the current discussion.

It is evident from Table 2 that future proposed robotic missions over the next two decades could be achieved with less than 2 metric tons of landed mass. At the same time, a proposed human mission would require a landed mass of 20 to 40 times more [6].

Implications of delivering the required mass to the surface of Mars, and to support these missions with suitable power system configurations will be discussed in the second half of this paper.

MASS TRACKING RESULTS

The payload masses that could be landed on the surface of Mars were generated with MassTracker, a Microsoft Excel–based tool, developed at JPL. MassTracker uses worksheets in a single Excel workbook to collect inputs for a mission architecture, specifically spacecraft dry masses, maneuver ΔV's and "rules–of–thumb". The tool then uses macros to build a worksheet, which calculates the spacecraft wet mass before and after each maneuver in the architecture. [7]

Selected & Potential Missions	Year	Class	Potential Power System Option(s)
Mars Reconnaissance Orbiter (MRO)	2005	Moderate/Large	Solar (typical for Mars orbiters)
Phoenix	2007	Scout	Solar selected
Mars Science Laboratory (MSL)	2009	Large	RPS (MMRTG baselined)
Scout (small mission) TBD	2011	Scout	Solar (cost cap may limit use of RPS)
Mars Science Orbiter (MSO)	2013	Moderate/Large	Solar (typical for Mars orbiters)
Astrobiology Field Lab (AFL) rover OR	2016	Large	RPS (based on MSL heritage)
Mid– or Fetch–rovers (sub–MER class)	2016	Scout–Large	Small–RPS or Solar
Scout (small mission) TBD	2018	Scout	Solar (cost cap may limit use of RPS)
Multi–Lander Network	2020	Moderate/Large	Small–RPS or Solar
Mars Sample Return (MSR) precursor	2022	Flagship	Solar (RPS trades performed)
Mars Sample Return (MSR)	2024	Flagship	Solar (RPS trades performed)
Deep Drill	TBD	Large	RPS (Solar trades performed)
ISRU Testbed, Tech demo	TBD	Large	Solar (RPS trades performed)
Large human precursor	TBD	Flagship	TBD (architecture dependent)
Manned Mars mission	TBD	Flagship	TBD (architecture dependent)

Table 1: **Selected** and potential Mars exploration mission concepts for NASA's Mars Exploration Program — with *downselect options* — for the next decade and beyond, including proposed launch dates, estimated mission classes and baselined power system options.

Missions	Landed Mass (kg)	Power System Option	Comments / Assumptions
Small rovers	160	Stirling w/ 1 GPHS (51 W_e)	Viking type lander
Mid–rovers	700	RPS w/ 4 GPHS or 1.3m^2 solar	Delta IV 4040
	1200	4 GPHS module or MMRTG	Atlas V 511
AFL	1310	4 GPHS modules or MMRTG	Skycrane or Viking type lander
Network Lander	150	solar or 1 GPHS module	4×150=600kg
Drill	886	MMRTG or solar	Atlas V 401, 50 m drill, pinpoint landing
MSR	780–840	solar	design concept with 30% contingency
	730	1 MMRTG + 1 GPHS for heat	design concept with 30% contingency
	720–760	solar + 40 RHU for heat	design concept with 30% contingency
	870–950	solar	design concept with 43% contingency
MSR /w fetch rover	1140–1230	solar	design concept with 43% contingency
Human precursor	550	RPS or solar	Delta IV 4040, pinpoint landing 1–10 m
Human mission	40,000–80,000	TBD (fission?)	Technology development needed

Table 2: Typical landed mass for in–situ Mars mission concepts. The landed mass in this table does not include supporting landed elements, however, the corresponding entry mass − not reported here − is in good agreement with the entry mass calculated by the MassTracker tool. (The "rules–of–thumb" based MassTracker tool assumes that ∼50% of the landed mass could be utilized for payload, and the entry mass is ∼2 times higher that the landed mass, or ∼4 times higher than the payload mass.)

A MER–style Mars mission architecture with a direct, aero–entry at Mars was assumed for this study. A launch C3 of 15 km^2/s^2 was used to determine the maximum allowable mass in trans–Mars orbit for each launch vehicle (see Figure 1). Three mass ratio "rules–of–thumb" were used to split the trans–Mars mass into a cruise stage, an aeroshell, a landing system, and landed payload. The following ratios come from systems experts in the Mars Pre–Projects and Advanced Studies Office at JPL. The assumed ratio of total landed mass to the landed payload is 2 : 1, where the total landed mass is the landing system plus the landed payload. The ratio of entry mass to total landed mass is 1.9 : 1, where the entry mass is the aeroshell plus the total landed mass. The ratio of the cruise stage to the entry mass is 1 : 4. The ratios were plugged into MassTracker and the landed payload mass was increased until the maximum allowable trans–Mars mass for each launch vehicle was reached.

The landed mass requirement for manned missions is predicted at around 40 to 80 metric tons [6], which is more than 2 to 4 times higher than the landed mass (i.e., twice the 9601 kg payload mass) predicted by the MassTracker tool for the planned Ares V launch vehicle. The impact of this landed mass requirement on the EDL configurations is discussed next.

EDL CONSIDERATIONS

Atmospheric entry, descent and landing at Mars present a significant engineering challenge for three reasons: (1) the atmosphere is sufficient to introduce substantial heating, but not dense enough to allow for deceleration achievable at Earth, Venus, Titan or in the atmospheres of Giant Planets; (b) landing is complicated by surface rock abundance, distribution, and size; (c) and the cost of creating a terrestrial EDL testing environment that simulates Mars maybe prohibitive. This section provides a brief discussion on EDL challenges for the ever increasing landed mass requirements, while a detailed assessment is given in [6].

Over the past decades the US successfully landed five robotic systems on Mars. All of these missions had a landed mass below 0.6 metric tons, landed altitudes below 1.4 km MOLA elevation, and landed footprints measured in hundreds of kilometers. These constraints resulted from the performance of the EDL system, and the relevant atmospheric conditions during entry.

Today's performance goals, summarized in Table 3, target a landed mass of 1 t, high MOLA elevation (up to +2 km), and an increased landing accuracy in the scale of tens of kilometers.

Manned missions would require up to two orders of magnitude more mass on the surface compared to current capabilities, that is, about 40 to 80 tons instead of 1 ton. As shown in the previous section, the payload limit on the surface of Mars is predicted below 10 tons. This would necessitate multiple landers, combined with a pinpoint landing capability, measured in tens of meters.

As discussed in [6], robotic exploration technology options that could greatly improve current EDL system delivery limits include:

- Larger diameter parachutes, that would deploy at high Mach numbers ($M \approx 2.7$);
- Inflatable / deployable aerodynamic decelerators (e.g., balloots), that would greatly reduce the ballistic coefficient;
- Supersonic propulsive descent system, which would help reducing entry velocity; and
- Pinpoint landing technologies (tens of meters), focusing on robust, terrain–relative navigation.

In their paper Braun and Manning [6] concluded that *"Mars human exploration aerocapture and EDL systems will have little in common with current and next–decade robotic systems. As such, significant technology and engineering investment will be required to achieve the EDL capabilities required for a human mission to Mars."* These EDL systems, corresponding to the landed mass requirements, may call for a 15 m diameter aeroshell. In comparison, the historic Viking and MER missions employed 3.505

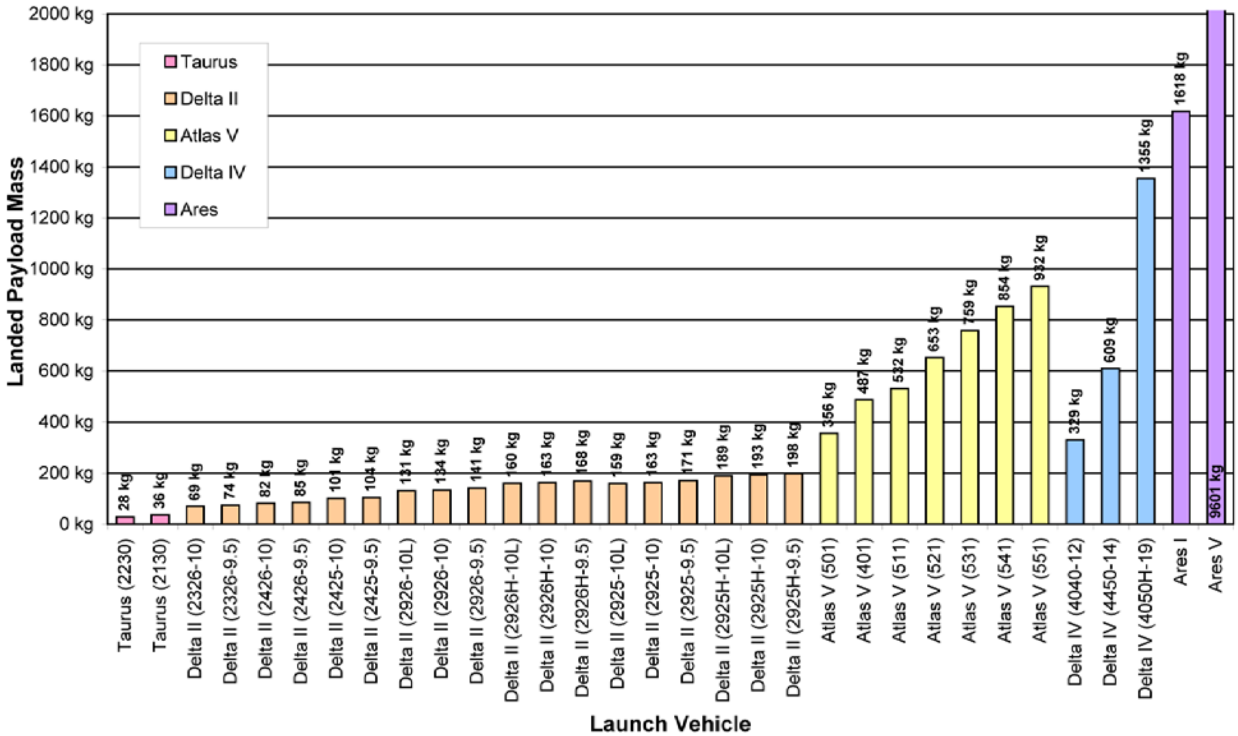

Figure 1: Potential landed payload masses delivered to the surface of Mars with various launch vehicles (for a C3 of 15 km^2/s^2). Note: the result for Ares V is not to scale. [7]

Surface Elevation (MOLA km)	Maximum β (kg/m^2)	Landed mass for 2.65 m diameter aeroshell	Landed mass for 4.5 m diameter aeroshell
-2.0	160	350	1000
0.0	135	300	850
+2.0	115	250	750

Table 3: Approximate aeroshell ballistic coefficients (β), and mass constraints, as a function of landing site elevation, including dispersions. (Adopted from [6].)

m and 2.65 m diameter aeroshells, respectively, while the MSL mission would use a 4.6 m diameter aeroshell. Furthermore, a 15 m diameter aeroshell would introduce technology challenges for the launch vehicles as well, since the current launch vehicle fairing diameters are limited to about a third of the one required for manned missions.

Therefore, human exploration missions will likely require further performance improvements, such as:

- Improved aerocapture / entry Thermal Protection System (TPS) configuration options;
- Improved method for an efficient transition from the entry to landing configuration at supersonic conditions, that would fit within the EDL timeline constraints; (note that this transition would likely to occur at Mach 3 or 4)
- Improved parachute system, by moving to aerodynamic decelerator systems in the hypersonic flow regime; or in the supersonic flow regime while combining it with a propulsive descent system.

POWER SYSTEM OPTIONS

Following the discussions on missions architectures, launch vehicle trades and EDL limits, this section provides a brief overview of potential power system options that could enable proposed future Mars missions. These space power technologies include power sources (both external and internal), and power storage devices. A more comprehensive Mars relevant summary of these power systems can be found in [5], [8], and [9]. Additional details on Radioisotope Power Systems (RPS) are provided in [10], [11], [12], and [13], while energy storage technologies are discussed in [14], and solar power generation in [15].

RPS Options for Mars Missions

NASA and DoE — with their industry partners — are currently developing two types of RPSs, which are considered internal power sources. The first is called a *Multi–Mission Radioisotope Thermoelectric Generator (MMRTG)* and it uses static power conversion. (This type RPS is baselined for the 2009 MSL mission.) The second is called a *Stirling Radioisotope Generator (SRG)*, and it employs dynamic power conversion (i.e., a Stirling converter). Both systems are designed to generate at least 110 W_e at BOL. Concepts of these two systems are shown in Figure 2, while their projected performances are summarized in Table 4.

The first proposed RPS is the MMRTG, which would use 8 General Purpose Heat Source (GPHS) modules. Its specific power is ∼2.9 W/kg, with a corresponding current best estimate (CBE) mass of ∼44 kg. It would generate ∼125 W_e at BOL. This RPS design is multi–mission capable, that is, an MMRTG can operate both in atmospheres and in vacuum. The almost 2 kW_t of excess heat — following the power conversion — could be utilized for spacecraft temperature management. In contrast to these advantages, thermoelectrics have low power conversion efficiencies (∼6.2%), and the overall power system degradation is estimated at ∼1.6%, half of it is due to natural radioisotopic decay and the other half is to the degradation of the thermoelectrics. At the same power output, it uses about 4 times more Plutonium–238 (Pu^{238}) fuel, and it is also heavier than the Stirling Radioisotope Generator. [10] [5]

An upgraded MMRTG design is also under consideration, with a higher power conversion efficiency of ∼8%, using improved or new thermoelectrics. These advanced MMRTGs would target a specific power level of 5 W/kg and above.

The second proposed RPS is the Stirling Radioisotope Generator (SRG). It would use 2 GPHS modules, generating 500 W_t. Two dynamic Stirling generators would convert some of this heat into ∼116 W_e at BOL. The power system degradation is assumed at ∼0.8% per year, due to radioisotopic decay. Degradation of the dynamic power conversion system was not assessed and was considered negligible. SRGs

Figure 2: RPS Concepts Under Development: Multi–Mission Radioisotope Radioisotope Thermoelectric Generator or MMRTG (left), and Stirling Radioisotope Generator or SRG (right – showing only the Stirling converter)

Parameter	MMRTG	Upgraded MMRTG	SRG	Small–RTG
Power per Unit (BOM), W_e	∼125	∼160	∼116	∼12–32
Mass per Unit, kg	∼44	∼40	∼34	∼6–8
# of GPHS Modules per Unit	8	8	2	1
Thermal Power, W_t	2000	2000	500	250
Specific Power, W_e/kg	2.9	5.0	3.4	∼2–4
Conversion type	Static	Static	Dynamic	Static
Converter materials	PbTe/TAGS	Scutterudites	Stirling	PbTe/TAGS
Technical Readiness level	TRL–5	TRL–3	TRL–3	TRL–2
Availability	baselined for MSL–2009	TBD	TBD	TBD

Table 4: Performance summary predictions for 4 RPS designs. Two of them are currently under development by NASA/DoE with industry partners — namely the MMRTG & SRG —; and the other two are suggested for future missions — i.e., the Upgraded MMRTG, and the Small–RPS. The upgraded MMRTG was conceived as a modified version of the standard MMRTG, where the PbTe/TAGS thermoelectrics would be replaced with higher conversion efficiency scutterudite thermoelectrics. Small–RTGs are only in a preliminary development phase, but if developed, they could enable a number of smaller missions. [5]

offer a few distinct advantages over static converter based systems. SRGs have significantly higher conversion efficiencies, in the range of today's ∼22% to the next generation of ∼32%. For the present system the conversion efficiency is about 4 times higher than that of the static conversion. Consequently, for the same power output SRGs require about 75% less Pu^{238} fuel, which can be an important consideration. This lower fuel requirement (∼1 kg vs. ∼4 kg per unit) could also significantly reduce the fuel cost per RPS, by as much as ∼$6M based on an assumed fuel cost of $2000 per gram of Pu^{238} [13]. Because of its lower mass (∼34 kg), the specific power of an SRG is about 3.4 W/kg, compared to 2.9 W/kg for the MMRTG. The SRG design is also multi–mission capable. In short, the SRG design is more efficient; requires less plutonium; and is lighter. The lower Pu^{238} requirement also results in 75% less heat generation, which may simplify cruise–phase thermal management for missions, where a lander is encapsulated in an aeroshell until the completion of atmospheric entry. Beside these advantages, SRGs also have both real and perceived limitations. SRGs are not yet space qualified, and the lifetime for dynamic converters is not yet proven. The SRG g–load tolerance requirement is currently 30g, somewhat lower than that for an MMRTG (40g). This can tolerate the launch environment, but limits landing to soft landing only. In case of failing one of the two Stirling converters, the whole unit could become unbalanced, resulting in the failure of the other converter side. EMI radiation could interfere with sensitive science measurements, and while EMI shielding could somewhat mitigate this effect, it would increase system mass and complexity. Finally, it is required to provide redundancy for these dynamic power systems. This means that each SRG enabled mission must carry a redundant unit, which lessens the power system mass gains against other RPS configurations. NASA and DoE are currently in the process of substituting SRG development from the one reported in Table 4 to an advanced SRG, for which the converter is shown in Figure 2.

NASA's RPS development plans may also consider small–RPSs [11] [8], generating power in the ranges of 10s to 100s of milliwatts or 10s of watts. The former would use multiple Radioisotope Heater Units (RHU: 1 W_t each), while the latter would employ a single GPHS module. Both configurations would utilize thermoelectric (or potentially dynamic) power conversion. Mission concept examples for Mars exploration, enabled by small–RPSs, are given in [16] and [8]. Small–RPS concepts could employ individual units, stacked as needed, or could follow a modular RTG (Mod–RTG) design, where the GPHS modules would be stacked and housed together. Both approaches could provide scalability to the power system, as required by the mission. These systems should be multi–mission capable and high g–load tolerant, in order to enable the largest number of missions.

Batteries

Batteries are energy storage devices, which utilize internal chemical power. They are scalable and highly reliable, but scaled up batteries could have a significant impact on total system mass. A battery's life cycle is influenced by the temperature of the environment; depth and rage of discharge; and degree of overcharge.

Primary batteries are not rechargeable, and typically used for short operations, such as during launch, EDL, and on planetary probes during descent. Examples include Lithium–Thionyl Chloride (Li–SOCl2) and Lithium–Carbon Monofluoride (LiCFx) cells. The specific energy for today's Lithium based primary batteries, at 0°C, is ∼250 Wh/kg, which is expected to increase to ∼400 Wh/kg and ∼600 Wh/kg in 5 and 10 years, respectively. Since battery performance is temperature dependent, it can degrade significantly at low temperatures.

Secondary batteries include Lithium–Ion, Lithium Polymer Electrolyte, Lithium Solid–State Inorganic Electrolyte and advanced Lithium–Sulfur (Li–S) batteries. They are rechargeable, and are only used for energy storage. During peak load operations, these batteries can complement the main power source, and could provide power during overnight operations. Their performance is lower than those for primary batteries. The State of Practice (SoP) specific energy for these batteries at 0°C is ∼100 Wh/kg, which is expected to grow to ∼120 Wh/kg and ∼200 Wh/kg within 5 and 10 years. The battery lifetime is also expected to increase from today's 5 years to 10 and 15 years, respectively. [14]

Solar Power Generation

Solar panels convert solar flux, and external power source, into electricity. (Solar flux decreases with the inverse square of distance from the Sun.) For Mars surface missions, there it further decreases due to the atmosphere and potential dust storms. Landing latitude, seasonal and diurnal changes could also impact solar availability and intensity. For example, the solar constant (S) at 1 AU from the Sun is 1367 W/m^2. At the orbit of Mars — at 1.5 AU — it drops to ∼43%, while on the surface it can be as low as ∼13% to 6.5% to that at Earth [8]. Solar panel size and mass scales linearly with power.

Multi–junction or multi–layer photovoltaic arrays — e.g., Gallium Indium Phosphide / Gallium Arsenide (GaInP/GaAs) cells — use different spectrums of sunlight, resulting in typical conversion efficiencies of ∼22–26.8% [15]. These solar panels degrade at a rate of ∼0.5% per year [17]. Important characteristics of solar cells include: high efficiency; good radiation, UV and atomic oxygen tolerance; long life; robustness for mechanical stress tolerance; high reliability and low cost. Similarly, the arrays can be characterized by their specific power; stowed volume; cost; and reliability. The main solar array categories include body mounted; rigid; and flexible or deployable configurations. Others include concentrator, electrostatically clean and high temperature arrays. The SoP for body mounted array areal power is ∼350 W/m^2. For rigid arrays the specific power is 30–60 W/kg, with a corresponding specific volume of 5–10 kW/m^3. For flexible or deployable arrays these are 40–80 W/kg and 10–15 kW/m^3, respectively, but the arrays may require complex deployment. [15]

Fission Power

The power technologies discussed above are not considered sufficient to support future manned missions, which would require significantly higher power levels. For these, *nuclear fission power* could be considered. Fissions reactors belong to the internal power source category.

Nuclear fission reactors are well established on Earth, with many decades of operational experience (e.g., research & commercial reactors; submarines; carriers). Space reactors are different in many aspects from their terrestrial counterparts. The specific mass is lower, due to transport and EDL limitations, and to high system integration. Reliability and long mission lifetime require autonomous operation, diagnostics and maintenance. The operating environment is harsh both in space and on a planetary surface. Space reactors must endure extreme (cryogenic) environments, lack of gravity, and dynamic loads during launch and EDL. Yet they could provide continuous power, without reliance on an external power source, such as the Sun. An important distinction between fission reactors and RPSs is that uranium fuel for the former is essentially non–radioactive before reactor startup, while RPSs are active throughout the mission, due to radioisotopic decay of the plutonium fuel.

Theoretically, reactors are scalable up to the kW$_e$ to MW$_e$ power levels. For example, the radioactive material (usually Uranium — U^{233} or U^{235}) in a fast spectrum, beryllium–reflected, ex–core controlled design (which is one of many) is contained in fuel pins and arranged inside the reactor core. The power level of a reactor increases with the number of fuel pins. While a 20 kW$_e$ reactor may use ∼150 fuel pins, similar 150 kW$_e$ and 1 MW$_e$ designs employ ∼300 and 600 pins, respectively.

The reactor core would be cooled using one of three methods:

- Liquid metal (lithium–coolant);
- Heatpipe (liquid sodium (Na) coolant);
- Direct gas (He/Xe coolant)

Liquid metal systems provide a flexible power conversion interface and have the lowest mass. This configuration is unproven, it is difficult to test, and system freeze/thaw could introduce a single–point failure to the reactor and consequently to the mission. Heatpipes are flexible, easy to test, and more reliable, and the multiple pipes provide redundancy, although lifetime and reliability data are not readily available and integration with the power converter/heat exchanger may introduce difficulties. Direct gas cooled systems are simple and easy to test, but difficult to integrate with power converters and can present a single point failure.

Increased operating temperatures support higher reactor powers. Reactors in space can employ refractory metals that can tolerate high temperatures, but are highly susceptible to corrosion in a planetary atmosphere. Thus high–temperature in–space reactors may not be suitable for Mars surface applications. The self–sustained fission reaction is controlled with safety/control rods/drums and neutron reflectors in and around the core. Power conversion is achieved through static or dynamic methods, as explained earlier for RPSs. In space, excess heat is rejected through large radiators. On the surface, convection and conduction complement radiation heat transfer; thus these radiators/heat exchangers are designed differently and are more efficient. Shielding configurations are also different between in–space and surface–based reactors. Recent in–space spacecraft designs place the radioactive reactor, the radiator, the payload, and the propulsion system in series. A shield plate is placed between the reactor and the radiators. The conical shadow of the shield provides a radiation–free area, which results in the familiar triangular shaped radiator design. On the surface, shielding may be required all around the reactor to protect not only the instruments but also the surface below and the environment around. Furthermore, a manned mission would require additional shielding to protect humans from the reactor's radiation effects. This will likely result in a significant system mass increase.

A more detailed overview of nuclear systems for Mars exploration is given in [9].

MISSION IMPACT OF POWER SYSTEM OPTIONS

Power system options, especially the use of nuclear systems, could significantly impact all mission phases, mission architectures and designs.

Impact on Mission Phases

RPSs generate heat continuously, which should be mitigated for all RPS enabled missions during *Earth storage*, *launch*, *cruise* and *EDL* phases, in addition to the in–situ operating phase [18]. RPSs are fueled and stored at a US Department of Energy (DoE) facility and integrated with the spacecraft on the launch pad prior to launch. (The storage phase can be as long as 2 years.) RPS integration with the spacecraft is overseen by the DoE. It requires a spacecraft design with easy accessibility, which could introduce an ever–increasing challenge as the number of RPSs may increase for human precursor and manned missions. The ambient temperatures and heat transfer mechanisms also vary throughout the mission phases. On Earth, during the storage and launch phases, the temperatures and pressures are terrestrial. Here the heat transfer mechanisms include convection, conduction and radiation. During the cruise phase in space (vacuum), radiation is the dominant heat transfer mode, while conduction through the RPS housing and along the cooling fins also plays a role. EDL on Mars requires an aeroshell for atmospheric entry. Thus, during the cruise phase the RPS is enclosed inside an aeroshell. The generated excess heat must be removed through a secondary cooling system. A typical configuration would use a fluid loop and external radiators. This adds mass to the spacecraft and complexity to the mission. In the Martian atmosphere heat is again removed through convection, conduction and radiation, although convection is less effective due to the lower atmospheric density.
In comparison, integration of other types of power systems, such as solar panels and batteries, do not represent integration challenges.

Impact on Mission Architectures

At 1.5 AU, Mars is relatively close to the Sun. Therefore, solar availability could point to the use of solar panels. However, solar power may not be suitable for all of the missions listed in Table 1. For Mars in–situ missions, seasonal changes at the polar regions could result in insufficient solar flux that could shut down the mission for up to 6 months [8], and potentially could cause "thermal death". Therefore, the power source selection strategies should be discussed on a destination related basis.

Mars orbiter missions historically used solar power generation, combined with secondary batteries to mitigate eclipses and other non–nominal operating conditions. It is expected that future Mars orbiters will continue to use solar power generation.

On the surface, longer missions to partially or permanently shadowed areas must address spacecraft thermal management issues. The thermal environment could be maintained by resistance heating or through utilization of the excess heat from RPSs or RHUs. In the first case, resistance heating would require secondary batteries, which would be charged from the solar panels during the Martian sol. This method is suitable for near equatorial regions, or for short missions at hight latitudes during polar summers (e.g., Phoenix, MSR). However, it would not be sufficient for long–lived polar missions, in which case the excess heat from a radioisotope source could be utilized to keep components warm. RHUs could augment resistance heating, or keep external components warm, reducing the need for oversized batteries, thus saving mass.

All of the in–situ Mars exploration robotic mission concepts discussed in this paper require power up to the multi hundred watts level. Among these architectures, the RPS enabled missions consider only a single MMRTG. Consequently, for these smaller missions the use of a dynamic conversion technology based SRG is not advantageous, since design principles require a redundant power system for a mission utilizing SRGs. When a single MMRTG based configuration is compared against two SRGs, it is evident that the savings in fuel would be only 50%, but the additional mass and cost impact from the second SRG unit would make this second mission architecture less desirable and more expensive.

On the *Mars Sample Return* mission — recommended in [4] — one of the key technology objectives is to keep the propellant on the Mars Ascent Vehicle (MAV) from freezing. This puts a significant demand on the thermal management system. For a solar powered architecture the propellant could be kept above freezing by resistance heating, RHUs or the combination of the two. For an RPS enabled option the excess heat from the power source or from RHUs could be utilized, while in an alternative configuration a single GHPS module could be employed specifically to provide 250 W_t for MAV thermal management.

The *Multi–Lander Network* mission [4] was studied through three mission architectures. Although it is baselined with solar panels, it has been demonstrated that if a single GPHS module based small–RPS would be made available, it could enable the mission beyond the capabilities of the solar option. This mission, however, was conceived with a hard landing aeroshell — using crushable materials — that would result in high g–loads up to ∼2000g. This would require significant RPS technology development, since RPSs are currently designed to tolerate g–loads up to 40g only. [18]

Mobility concepts could include rovers at various sizes and mission classes. The largest, the *Astrobiology Field Laboratory* rover, would likely utilize MSL heritage, including the baselined MMRTG power source. (RPS enabled missions typically employ a hybrid power system, where during peak power modes power is drawn from the RPS and the batteries, while during low power modes the batteries are recharged [19].) Potentially, the rover could utilize the proposed updated MMRTG. The additional power would increase traversing capability, and high volume data collection & transfer (e.g, high definition streaming video from the surface). Smaller rover

concepts could be based on *MER heritage*, and could use either solar power or small–RPSs [20]. Single GPHS module based small–RPSs could be stacked to provide the required power levels. For example, both MER rovers used 1.3 m^2 solar panels, generating about $\sim$1000 Wh/sol at BOL, which is predicted to drop to $\sim$600 Wh/s at the end of life (EOL). The same amount of power could be generated with two single GPHS module based small–RPSs. The smallest *sub–MER* class rovers are referred to as *fetch rovers*. In NASA studies these were also considered with two small–RPSs, and were designed to support the MSR mission, or to perform independent prospecting for future ISRU missions.
Large scale prospecting on the surface would likely necessitate larger RPS enabled rovers, long mission durations, and significant traversing capabilities. These rovers should cover 10s of square kilometers to map in–situ resources. Traversing capabilities at this scale may need MSL class rover configurations or possibly more.

The *Deep Drill* concept was considered with pinpoint landing, MSL landing heritage, and a subsurface access to $\sim$50 m, although mission trade studies extended the depth to $\sim$100 m. The mission architecture assumed either solar or RPS based power generation. With an optimized design, drilling to 50 m could be supported with 4 small–RPSs, generating about half the power of a standard MMRTG. For the 100 m excess more power would be needed, which could be provided by an MMRTG, while an updated MMRTG could further enhance this mission.

ISRU testbed mission concepts were studied with power requirements around 1 to 2 kW_e, using solar panels. Ultimately, RPSs could be employed maybe up to 5 kW_e, requiring Pu^{238} fuel at the Cassini–Huygens mission level, and combined with dynamic power conversion. Note that the corresponding $\sim$27 kg plutonium fuel would generate 13.5 kW_t thermal power, which must be rejected throughout the cruise phase. The required radiator size would be $\sim$7 times larger than the one designed for the MSL rover's external radiators during cruise phase. This would have a significant mass impact on the mission. In addition, during EDL this excess heat could be removed using phase change materials (PCM), further increasing the overall system mass. Solar panels and a near equatorial landing location could solve this problem. However, if the mission is required to access the Martian poles, then a hybrid system with solar panels, RPSs, and secondary batteries could be used. During polar summers the RPSs could supplement solar power generation. During polar winters the operation could be scaled back and the power and heat generated with the RPSs could keep the system above survival temperatures. Ultimately, a small fission reactor could be also considered, but this technology is not yet available.

Manned Mars missions are expected to power habitats, ISRU plants, surface mobility platforms, a mining apparatus, and science instruments, resulting in a power requirement above 100 kW_e. Obviously this is outside of the capabilities of today's space power systems. Even a proposed multi–kilowatt level HOMER type surface fission reactor would not be sufficient to meet these needs, and technology development would come at a significant cost, as demonstrated through the now canceled JIMO mission.

Power Sizing Limits Based on Payload Mass

An alternative way to assess power availability for in–situ Mars missions is by looking at the deliverable mass to the surface, and from that determine the available power levels, based on a "rules–of–thumb" mass fraction allocation to the power system. Using this methodology, Table 5 summarizes the landed payload mass for selected launch vehicles, generated with the MassTracker tool (see Figure 1). From these mass values a typical 10% was assumed for the power system mass fraction, and the power levels were calculated for RPSs at three specific power levels (i.e., 2.9 W/kg, 5 W/kg and 6 W/kg, representing an MMRTG and future performance goals, respectively). These calculations indicate that the listed launch vehicles would be able to deliver sufficient mass to the surface to support the proposed missions, and to power them. However, manned missions would require power at the 100 kW_e level, which is beyond current capabilities.

Similarly, based on the projected performance of an Ares V launch vehicle, the power system mass allocation would translate to an RPS based configuration, that could generate around 3 to 6 kW_e. This would also correspond to either 21 MMRTGs or 28 SRGs, with RPS fuel requirements of 28 kg to 84 kg, making this architecture not credible, due considerations related to plutonium availability, mission cost, mission risk, and other technology challenges.

A simple scaling calculation was also performed to assess the the potential available power using solar panels. As discussed earlier, manned missions would require landed mass in the 40 to 80 metric tons range. (This would necessitate ∼4 to 8 pinpoint landings for a Ares V based architecture.) Since solar panel mass and power scales linearly, it is possible to scale up the 1.3 m^2 solar panel of the 180 kg MER rovers by 400 fold, resulting in a solar panel size of 520 m^2 and payload mass of 72000 kg (in line with the requirements for a manned mission). A 520 m^2 solar panel translates to a size of 26 m × 20 m. Since the MER panels generated ∼1000 Wh/sol (or ∼40 W_e on average), the scaled up panels could provide ∼400 kWh/sol (or ∼16 kW_e on average). This is significantly lower than the required 100 kW_e. Solar panels generate power from sunrise to sunset, resulting in significantly higher peak powers than the average power stated above. Excess power must be stored and used overnight by a scaled up secondary battery bank, that could store power at the multi-kilowatt level. This would have a significant mass impact on the mission. Deployment, degradation, structural support, dusting, and seasonal variations could further reduce the solar panel performance during its expected long mission duration. Note that this "back of the envelope" assessment for solar power generation was presented for illustration purposes only.

CONCLUSIONS

In this paper we discussed power system options for proposed future Mars missions under NASA's Mars Exploration Program. In a broader framework, these power system options were examined together with other mission architecture elements, namely launch vehicles and EDL limits.

The main focus was placed on Radioisotope Power System options, since these may introduce more complexity to long–lived in–situ robotic missions and related mission architectures, than other power system options.

Power system options for in–situ Mars missions up to the multi–hundred watts level are well understood, and multiple choices are available. For large human precursor, and particularly for manned missions, power systems in the tens to hundreds of kilowatts range represent only one of the many technology challenges that needs to be solved through a dedicated technology development effort.

The Mars Program also identified a potential need for small–RPSs in the 12 to 20 W_e range. If this power system becomes available, a number of additional missions could be enabled by single GPHS module based small–RPSs. Small–RPSs could reduce plutonium requirements and mission cost, when compared to MMRTGs; while extending mission capabilities and mission duration when compared to solar panels.

Historically, for orbiting spacecraft solar power generation was and is found to be the best suited option, due to simplicity; low mass; high reliability; and high solar flux availability.

In summary, future robotic missions over the next two decades are expected to utilize existing power technologies, such as solar panels, batteries and RPSs. Because of the number of RPSs and the mission requirements, these proposed Mars missions are not key drivers for future RPS development. At the same time, manned missions will require about an orders of magnitude more mass and about two orders higher power than that for robotic missions. Thus manned missions would likely need a nuclear fission reactor to provide power at the 100 kW_e level.

Power system options represent only one element of complex mission architectures, there-

Launch Vehicle	Landed Payload kg	Power System Mass kg	Power[†] W_e	Power[‡] W_e	Power[§] W_e
Atlas V (501)	356	36	103	178	213
Atlas V (401)	487	49	141	244	292
Atlas V (511)	532	53	154	266	319
Atlas V (521)	653	65	189	326	392
Atlas V (531)	759	76	220	379	455
Atlas V (541)	854	85	248	427	512
Atlas V (551)	932	93	270	466	559
Delta IV (4040–12)	329	33	96	165	198
Delta IV (4450–14)	609	61	177	305	366
Delta IV (4050H–19)	1355	135	393	677	813
Ares I	1618	162	469	809	971
Ares V	9601	960	2784	4801	5761

Table 5: Notional continuous power generation as a function of payload mass, and specific power levels of [†] 2.9 W/kg, [‡] 5 W/kg, and [§] 6 W/kg. Note that the power system mass is assumed at 10% of the landed payload mass, while the specific power levels are typical for current and future RPS concepts. The actual power system mass, and the resulting power levels would also be influence by the discrete unit mass values or the RPSs. Therefore, these values are provided for illustration purposes only.

fore, in future missions studies these should be addressed together with other technology challenges, including launch vehicle and EDL limits, technologies for in–situ resource utilization, human habitats, and improved instrumentation.

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