Nuclear Systems for Mars Exploration[1,2]

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Abstract—This paper identifies breakpoints for various power and propulsion technologies, with a special focus on fission-based sources in support of NASA's Mars exploration program. Transportation, orbital, and surface missions are addressed through an assessment architecture developed for this study. This architecture is based on three key considerations: decomposition of generic Mars missions into phases, a lumped parameter approach, and a bounding case analysis. With these simplifications breakpoints are identified beyond which new technologies, such as nuclear fission power, are required to achieve mission objectives. It is found that in-space propulsion and power generation are sized by launch vehicle delivery limits and trajectory options. Similarly, power levels for surface-based reactors are affected by transportation system and EDL limits imposed by current technologies. After summarizing the breakpoints for today's state of the art, development targets are identified to enable space-based nuclear power and propulsion systems to perform at their full potential.

TABLE OF CONTENTS

1. INTRODUCTION 1
2. ASSESSMENT ARCHITECTURE 2
3. TRANSPORTATION ISSUES 2
4. EDL ISSUES 7
5. POWER GENERATION ISSUES 9
6. CONCLUSIONS 16
ACKNOWLEDGMENTS 17
REFERENCES 18
BIOGRAPHY 20

1. INTRODUCTION

As the latest phase of our Mars exploration program draws to completion, NASA continues working on future mission concepts. Contributing to this effort, JPL's Advanced Mission Studies Office identified three possible exploration pathways, forecasting the next four decades. Assuming the success of our current missions, the first path in the roadmap covers a mainstream approach termed "current pathway". It follows a conventional path with a number of planned missions for the next decade, followed by robotic and subsequent human explorations in the third and fourth decades. The second path, termed "reduced scope or go competed", reflects an uncertainty of future exploration efforts, reacting to unfavorable technical, economical, or political influences. This rather pessimistic path sees the completion of the upcoming Mars Science Laboratory (MSL) program, but without predicting any follow-on missions. The third path responds to "momentous discoveries", such as finding signs of past (or present) life on Mars, and consequently igniting an accelerated golden age of exploration in the form of advancing robotic exploration as early as the next decade, with subsequent human exploration and colonization of Mars during the following two decades (Figure 1). However, finding existing life on Mars could result in a delay to colonize Mars, until bioethical issues are resolved.

The first two Mars exploration pathways predict a gradual increase in mission complexity, offset only by program timeline. To achieve the objectives of these missions, enabling technologies (e.g., propulsion and power systems) must evolve and in effect change our current technology paradigm. As shown in Figure 2, today's space exploration can be characterized by mass dependence and consequent power limitation. Launch vehicle technologies limit the

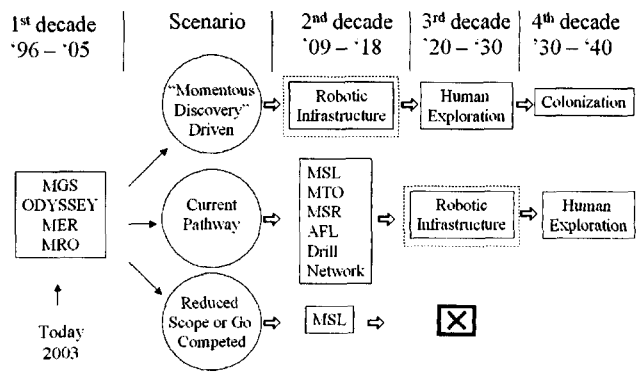

Figure 1- Mars exploration program evolution [1]

[1] 0-7803-8155-6/04/$17.00 © 2004 IEEE
[2] IEEEAC paper #1118, Version 3, Updated November 19, 2003

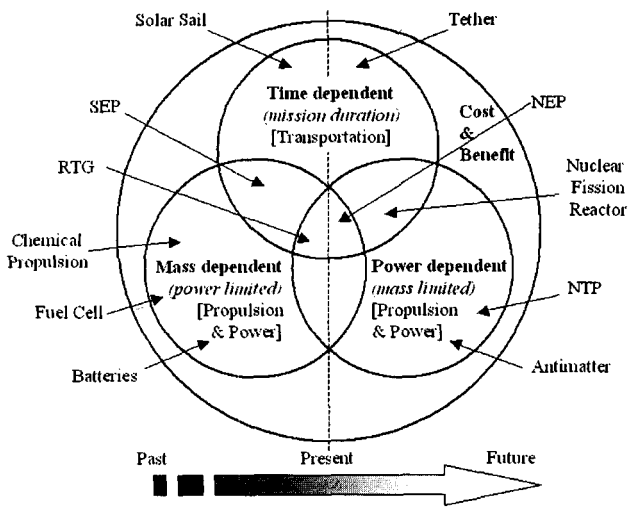

Figure 2- Technology paradigms

maximal deliverable mass to Earth orbit and beyond. Space missions are designed around these bounds, thus hampering power availability for transportation, science, and housekeeping. Chemical propulsion, fuel cells, and batteries belong here, characterized by restrictions to both power and duty cycle. In the future, a new paradigm can be envisioned, where advanced propulsion and power sources would provide power far beyond our current limits. While for the distant future we may consider exotic power sources based on antimatter or nuclear fusion, for the near term nuclear fission power is the most likely candidate. Solar sails and tethers do not generate power by themselves, and may operate for an extended period of time; hence these solutions belong to a time-dependent category, not explored further in this study.

Thus, this paper focuses on two of the three paradigms, one dependent on mass and the other on power, with an emphasis on the role of nuclear fission power. First, the assessment architecture of this study is introduced, followed by descriptions and performance characteristics of

- Transportation
- Entry, descent, and landing (EDL)
- Fission and decay-based power generation
- Other conventional technology options

After identifying the limitations of today's technologies, key areas are stated where advancements could facilitate a transition from one paradigm to the other, demonstrated within the framework of the Mars Exploration Roadmap.

2. Assessment Architecture

Power and propulsion system technologies cover a broad range of options developed to various technology readiness levels (TRL). Space missions to date have utilized these technologies, which have been selected based on mission objectives. Therefore, to assess the breakpoints beyond which nuclear power sources represent the only viable alternative, their performance must be compared against more conventional technologies.

The assessment architecture used for this study consists of three components. The first reduces the number of parameters to only a few; the second limits the sensitivity analysis to the upper bounds of these parameters; and the third decomposes a generic Mars mission into distinct stages.

Lumped Parameter Approach

Space mission complexities pose hard challenges reflected through a multitude of dependent parameters. To account for all is beyond the scope of this work. Instead, a lumped parameter approach is adopted, reducing these parameters to only mass, power, and time, from which all other parameters can be derived. Based on these key parameters main technology breakpoints are identified.

Bounding Case Approach

Technologies are sometimes scaleable and cover a wide application spectrum. The bounding case approach, adopted here, helps to minimize assessment effort by identifying these upper limits or technology breakpoints beyond which new technologies are needed to achieve mission objectives.

Mission Stages

A typical Mars mission consists of a partial or a full set of the following three stages:

- Transportation stage
- In-orbit stage
- On-surface stage

Each of these can be characterized by mass, power, and time.

With this methodology, generic mission concepts can be tested inexpensively on a conceptual level. Once one or more favorable answers are reached, further in-depth studies are needed to address utility and to identify the best candidate configuration for a given set of mission objectives. (Note that other considerations, such as safety and cost — though critical — are not discussed.)

3. Transportation Issues

The orbit of Mars is more eccentric than that of Earth. It is at ~1.5 AU (~1.4 to 1.6 AU) from the Sun, with an orbital inclination of 1.85°, relative to Earth. Due to orbital phasing, low-energy launch opportunities to Mars occur about every two years. It is typical to launch the spacecraft to parking Low Earth Orbit (LEO), orient it to an appropriate inclination, and then launch it to a transfer orbit between Earth and Mars with the last stage of the launch vehicle or using an onboard propulsion system. From that point on the spacecraft follows a trajectory, which is based

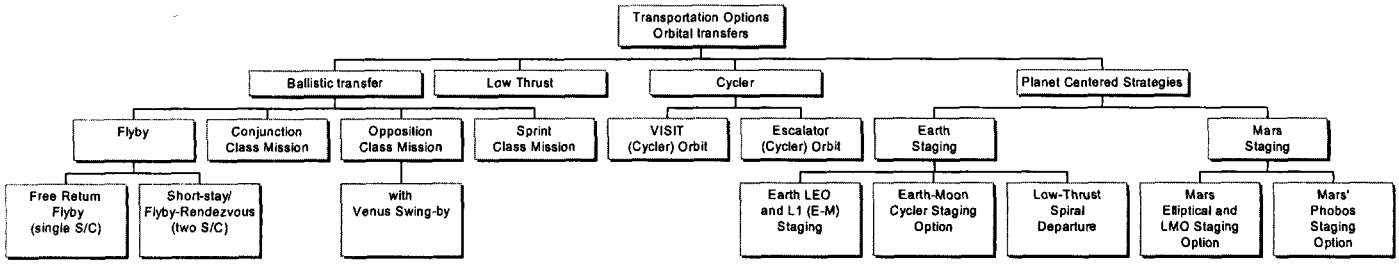

Figure 3 - Return trajectory options between Earth and Mars

on its initial impulse or its onboard propulsion system or both. Before placing any propulsion and power systems into the transportation framework, it is important to note the competing propulsion technologies and trajectory options. Combinations of these options define a Mars mission architecture trade space. Trajectories are influenced by launch date and propulsion system options. Similarly, trip time and payload mass requirements call for a suitable propulsion system. Hence, both trajectory and propulsion system options are discussed below.

Trajectories

The three main orbital transfer pathways are:

- Ballistic (using high-energy impulse)
- Low thrust (but high specific impulse, I_{sp}) or
- Cyclers

Each of these can be subdivided based on trip time and energy [2]. Due to the phasing between Earth and Mars and the departure time, return trip missions can be optimized for a number of variables, for example shortest mission time, fastest transfer time, longest surface stay, or largest deliverable mass. Figure 3 summarizes these return trajectories, applicable to both manned and sample return missions. Detailed description of these trajectories is given in [3], and [4]. For most one-way scientific and cargo missions, the transfer time corresponds to the outbound leg of a given return trajectory. One-way manned missions do not have a mainstream acceptance; however, they can result in a 25 to 35% cost saving while building up a colony and resources on Mars [5][6]. It should be noted that trip times are dependent on assumptions for the propulsion system, final mass to be delivered to Mars, and the launch time. Therefore, the numerical values provided in this paper should be viewed only as rough estimates.

High-thrust trajectories refer to ballistic transfers. While a *Hohmann transfer* does not take planet phasing into account and in this pure form the trajectory cannot be used, it is considered an ideal transfer, minimizing the total energy. Such a trip to Mars requires a round trip ΔV of 11.2 km/s. This can only be achieved by dividing the round trip into an outbound leg, a stay period, and a return leg. The resulting round trip time is 2.66 years (~971 days), including a stay time of 1.24 years (~453 days) [7]. Type 1 and 2 transfers can be faster than Hohmann transfers and can be envisioned as Hohmann transfers to a dummy orbit beyond Mars, terminating and/or initiating at one of the two Mars orbit crossings. *Type 1* (T1) round trips transfer to the first Mars orbit crossing. For this case the stay time increases to 1.65 years, but the round trip time is only reduced by 0.2%. *Type 2* (T2) round trips pass the target orbit and transfer at the second opportunity on the way back. It reduces roundtrip time by about 8%, but significantly reduces stay time (to 0.43 years) [7]. *Free-return flyby* is the simplest and least energy intensive trajectory, based on the ballistic Hohmann transfer and a single spacecraft, which minimizes propellant requirements. During the outbound trip the spacecraft passes Mars, achieving about 2 hours of optimal viewing. There is insufficient time for a piloted landing. The return requires 1.5 heliocentric revolutions due to planet phasing. The total flight time is about 3 years. While it fulfills technological requirements, the long flight time and short stay time makes this option undesirable [3][4]. *Flyby-rendezvous* or *short-stay* are similar to the ballistic transfer type free-return flyby, but they utilize two spacecraft. The first spacecraft arrives and lands 30 days before the second spacecraft's flyby. It takes off in time to rendezvous with the second spacecraft and does the same return as the free-return flyby mission. It still results in a proportionally too long transfer time compared to a short 30-day stay. For this case and also for the free-return flyby, the Earth-Mars flight time is ~230 days, while the Mars-Earth return flight-time is ~840 days due to planet phasing. [3][4] *Conjunction class* represents a long-stay mission architecture, with a total round trip time of ~950 days, which includes a stay time of up to 560 days [8]. Such a mission may require a Saturn class rocket and In-Situ Resource Utilization (ISRU). Fast 150-day one-way trip times would need Nuclear Thermal Propulsion (NTP) (sometimes referred to as Nuclear Thermal Rocket (NTR)). *Opposition class* represents a short-stay architecture, with short outbound and long return transit times (or reverse). The advantage is a short 1.6 to 1.9 year total mission time without or with Venus swing-by, respectively. The Venus swing-by is more favorable from an energy point of view, but it subjects the spacecraft to greater thermal and radiation loads. Minimum energy opportunity for this class occurs every 26 months. The stay time on the surface is, however, only 30 days; hence 95% of the total mission time is spent in transit. Niehoff et al. [3] provides a summary table of available launch dates between 2002 and 2015 for conjunction, opposition, and sprint class trajectories. The above "brute force" methods are expensive, requiring the

largest available launch vehicles (e.g., Delta IV Heavy or Titan IV).

Low-thrust trajectories require longer trip times than those for high-thrust trajectories, and the spacecraft spends a significant time crossing the Van Allen radiation belt. Due to a higher specific impulse propulsion system, the delivered mass may be higher. Adding ΔV by a Venus swing-by can further reduce propellant requirement and consequently further increase payload mass, but it adds to the trip time. (Venus is about 30% closer to the Sun than Earth). Low thrust transfer options are used by electric propulsion systems. A typical mission includes a 50-day outward spiral from LEO. After reaching escape velocity the trip to Mars takes about 510 days. Another 40-day spiral-in follows from Mars orbit capture to reach Low Mars Orbit (LMO). The stay time is between 100 and 200 days. The return trip consists of a 25-day spiral-out from LMO, a 230-day transfer from Mars to Earth, and a 16-day spiral-in from Earth capture [3]. For manned missions it is suggested that, to avoid the Van Allen radiation belt, embark the crew on a high-thrust rocket and join the low-thrust spacecraft after it spiraled beyond Geosynchronous Earth Orbit (GEO). Manned missions on a low-thrust trajectory would, however, results in a set of problems, including prolonged exposure to galactic radiation, additional cost of countermeasures, and physical and psychological support for the crew.

Cycler orbits represent perpetual travel between the orbits of Mars and Earth. Nakagawa et al. describes a cycler-like trajectory in [9]. "NEPTranS" (Nuclear Electric Propulsion (NEP) Transportation System) is envisioned to make full use of the high ΔV capability and added payload benefits of a NEP system. The low-thrust cycler trajectory between Earth and Mars is designed for over 10 years of operation, using a 100kWe NEP system (I_{sp}=5000 sec; η=70%). The repeating phases of the trajectory consist of

- Spiraling out from LEO
- Transfer to Mars
- Spiraling in to LMO

The architecture includes a launch to LEO by a Delta IV-H, an Atlas V, or the Space Shuttle. Two traditional cycler orbit examples are VISIT and UP/DOWN Escalator cyclers. Versatile International Station for Interplanetary Transport (*VISIT*) cyclers orbit the Sun four times, while Earth orbits the Sun five times; hence VISIT reencounters Earth every five years. As for Mars, VISIT completes three orbits around the Sun, while Mars does it twice. This orbit repeats itself every 15 years and should be potentially retuned after 20 years. Frequent transfers between destination points can be achieved by employing a network of three or more VISIT cyclers [10]. An *UP/DOWN-Escalator* cycler uses Earth gravity-assist passes to precess and keep up with the progressive Earth-Mars phasing orientation — rotating its semi-major axis by about 50° counterclockwise between successive phases. The UP phase includes a short Earth-Mars transfer with a Mars flyby, where the payload is transferred. After reaching aphelion (farthest point from the Sun) beyond the orbit of Mars, the DOWN cycle begins until two Earth gravity-assist encounters and another payload transfer occur. The cycle repeats; however, such escalator cyclers require periodic course corrections at aphelion [3].

Planet-centered strategies allow for higher delivered mass to Mars [3][4]. They may involve in-orbit assembly and the utilization of Lagrange points around the Sun or destination planets. For *Earth Staging*, a LEO departure assumes that all of the LOX/LH propellant for a chemically fueled Mars spacecraft originates on Earth. The spacecraft is assembled in LEO, and then at the appropriate time at perigee it fires its onboard propulsion system to depart for Mars. This approach can also be applied to NEP, Solar Electric Propulsion (SEP), and NTP-enabled spacecraft. Assembly at LEO allows for more mass to be delivered to Mars; however, it adds more complexity and higher risk to the mission. Lagrange point departure from the Earth-Moon L1 point assumes that the hydrogen and oxygen (oxidizer) propellants are supplied from Earth and the Moon, respectively. Fueling occurs at the gravitationally quasi-stable L1 point. An initial burn takes the spacecraft out of L1 on an elliptical Earth orbit, benefiting from Earth's gravity assist. At perigee a second burn is required to reach escape velocity from the Earth-Moon system. For an Earth-Moon cycler orbit departure the spacecraft leaves LEO, flies by the Moon for supplies (e.g., lunar oxygen), and then returns to perigee for supplies (e.g., hydrogen fuel). A burn sends the spacecraft into an elliptical "phasing" Earth orbit, taking it two-thirds of the way to the Moon. Two phasing orbits allow the Moon to reach the proper position, when another burn at perigee raises the orbit for a second lunar flyby. This cycle repeats until the spacecraft is ready for Mars departure, at which time it fires its rockets at perigee to leave the Earth-Moon system. For *Mars Staging*, LMO is the simplest Mars arrival option but the least fuel efficient. The spacecraft enters a close circular orbit around Mars using a long rocket burn at periapsis. In another option the spacecraft enters a "loose" elliptical orbit around Mars by firing its rockets for a short time at periapsis to remove ΔV. At apoapsis it fires its rockets again to further reduce ΔV and consequently to fall toward Mars for entry and landing. The inner moon of Mars, Phobos, may have a composition of a carbonaceous chondrite meteorite, which could be converted into hydrogen and oxygen through ISRU. (Martin Marietta [now Lockheed Martin Corporation] built full-scale ISRU units, also called In-Situ Propellant Production [ISPP] units. The units demonstrated both methane and oxygen production [11].) A moderate periapsis burn can change the spacecraft's velocity to match orbits with Phobos, then land there. Using it as a "natural space station", launch mass requirements for piloted Mars missions can be reduced and additional propellant manufactured and stored there. In the case of a manned mission, astronauts could tele-operate robots on Mars from Phobos for several months.

Aeroassist, such as aerobraking and aerocapture, may reduce the fuel requirement, in which case the drag of the Martian atmosphere slows down the spacecraft by converting its kinetic energy into heat. Aerobraking can, however, increase mission time, and aerocapture may increase mass by about 10 to 20% for a larger heat shield or by 5 to 10% for a ballute. Aeroassist options with a nuclear power source on board require careful consideration. For orbiter missions, taking an in-space reactor inside the Martian atmosphere may raise planetary protection concerns. When aeroassist maneuvers are used to circularize the orbit of a landing spacecraft, subsequent de-orbiting would necessitate a sizable propulsion system to remove ΔV. (Aerobraking has been demonstrated on the Mars Global Surveyor and Mars Odyssey missions [12].)

Propulsion System Options

Today's launch vehicles employ chemical propulsion systems. These chemical rockets can propel the payload to Mars directly or indirectly. As discussed above, a launch vehicle can send payload to a positive escape velocity (C3) directly or to a parking LEO. From LEO the spacecraft launches to Mars by the use of another chemical rocket (kick motor) or by other onboard propulsion systems, such as NTR, SEP, or NEP. Propulsion systems are customizable enabling technologies, influencing both trip times and deliverable mass limits. Complex missions may use a wide variety of options. For example, a manned mission can have separate manned and cargo parts. The manned part may use chemical or nuclear thermal propulsion, while the cargo part may use all available options and technologies. The assessed systems of this study, which were considered for their high or moderate TRL, include chemical, nuclear electric, and solar electric propulsion.

Table 1. Rocket propulsion performance summary [13]

Type	Parameter	Min.	Max.
Solid rocket	Thrust (kN)	0.050	5000
	$I_{sp\text{-}vacuum}$ (sec)	270	305
Liquid bi-propellant	Thrust (kN)	0.0001	12000
	$I_{sp\text{-}vacuum}$ (sec)	150-315	450
	Run time (sec; min)	Few sec	100's min
Nuclear Thermal	Thrust (kN)	Up to 12000	
	$I_{sp\text{-}vacuum}$ (sec)	800-1100	6000
	Run time (min; hour)	Few min	Hours

Rocket propulsion describes high-thrust propulsion systems that include chemical and nuclear thermal rockets, with the performance characteristics shown in Table 1. *Chemical propulsion* employs solid and liquid propellants, or the combination through hybrid systems. The I_{sp} of solid, monopropellant, bi-propellant, and hybrid systems range between ~270 to 305 sec, ~140 to 235 sec, ~320 to 450 sec, and ~290 to 350 sec, respectively. Chemical systems hence have the advantages of high thrust levels and long heritage. Typical chemical systems may achieve ΔVs up to 8 to 11 km/s. Disadvantages include moderate performance, combustion complications, and safety concerns.

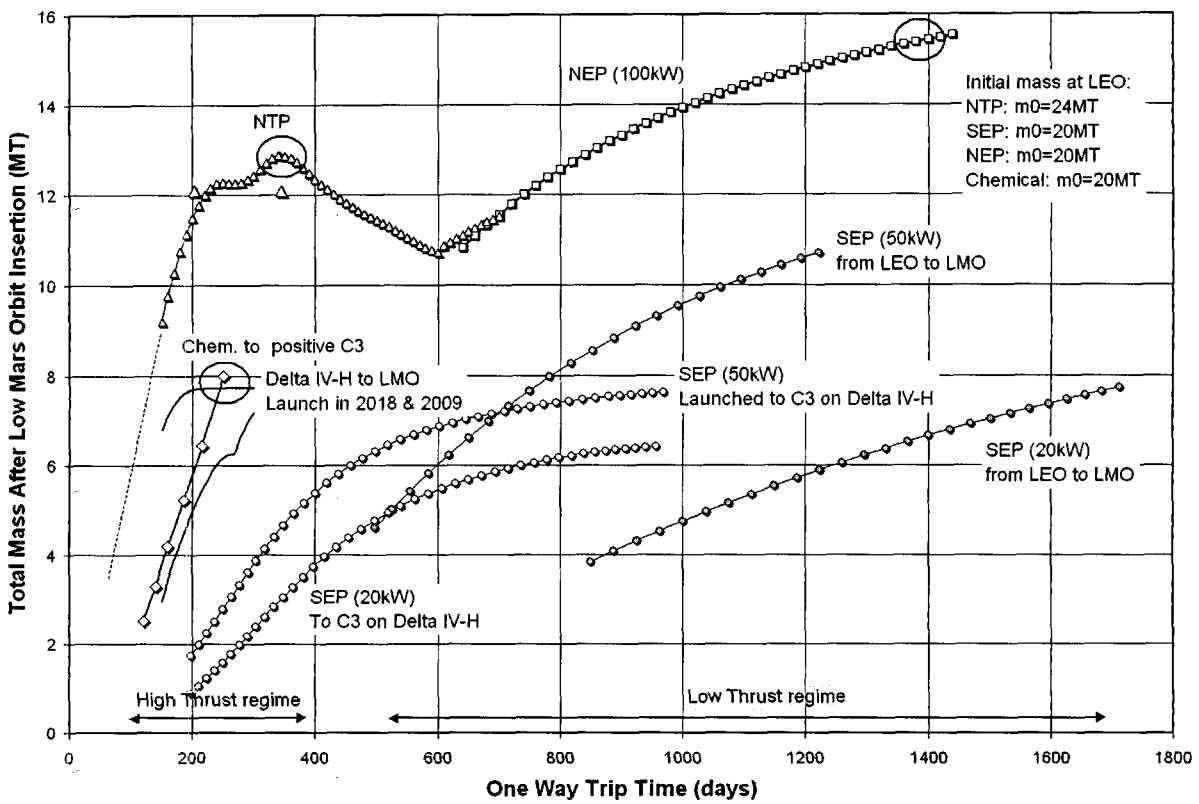

Figure 4 - Transportation system breakpoints

The deliverable mass performance of today's launch vehicles for direct or indirect delivery options can be obtained from [14]. Delta IV Heavy launch vehicles have the highest delivery capacity available today. For a Hohmann-type ballistic transfer trajectory, the maximum total mass that could be delivered to Mars is ~8 MT (metric tons). With reduction of interplanetary transfer time, the deliverable mass also decreases. If the payload is inserted into a typical 1000 km Earth orbit, the mass corresponds to ~20 to 22 MT. From this orbit an onboard propulsion system can be used to send the spacecraft to Mars. Additional analysis was performed to calculate the mass delivery capacity of a typical Delta IV Heavy (4050H) launch vehicle (I_{sp}=315 sec) to a 1000 km altitude circular orbit around Mars [15]. The trajectories have been optimized such that a deep space maneuver is included if it improved performance. Five launch opportunities were examined between 2009 and 2018. The calculations are based on an optimum launch date. Consequently, the actual performance is expected to be lower, when a specific launch window is included, due to a less optimal opportunity. Figure 4 shows the total deliverable mass to Mars. These calculation results are in good agreement with the data obtained from the NASA KSC database [14].

Nuclear thermal propulsion, with a thrust level up to 12000 kN (see Table 1), can be divided into three reactor categories:

- Solid core, with I_{sp}=800 to 1100 sec
- Liquid core (conceptual), with I_{sp}=3000 sec
- Gas core (conceptual), with I_{sp}=6000 sec

The main advantage of nuclear thermal propulsion is the high specific impulse combined with high thrust. For the same initial-to-final mass ratio, nuclear systems achieve twice the ΔV compared to chemical systems. Consequently, nuclear thermal rockets could potentially cut the one-way trip time to Mars from ~6 months to ~2 to 3 months. Alternatively, for the same trip time it could double the delivered mass. NTP and NEP systems may achieve ΔVs in the 22 to 33 km/s range. In addition, while conventional liquid chemical propulsion systems use both fuel and oxidizer from separate storage tanks, NTP eliminates the weight of one tank, one propellant fluid, and one pump. Instead of combustion, the reactor heats up the gas passing through it. This technology is yet to be proven in space; it is expensive and the thrust-to-weight ratio is low. To quantify the performance of a small engine NTR, an assessment was performed for near-term small cargo missions [16]. In this study two separate solid core NTRs were considered on Type 1 and 2 trajectory opportunities, between 2007 and 2026. The initial masses on a 407 km circular LEO were assumed at 12 MT, 24 MT, and 80 MT based on the delivery capacity of Delta IV-M, Delta IV-H, and Shuttle derived Magnum class heavy lifter launch vehicles, respectively. The small engine is based on MITEE (MIniature ReacTor EnginE) studies for the Air Force, while the higher thrust configuration is based on Confederation of Independent States (CIS) (formerly Russian) technology [17]. It was calculated that, for ballistic trajectories, a Delta IV Heavy launch vehicle with a small NTR could deliver a 12.1 MT payload to Mars in 204 or 343 days, depending on the trajectory option and the launch date. Further analysis was also performed [16] to assess the trip time dependant performance of a small NTR system. The optimal trajectory, which corresponded to the minimum energy case, resulted in a trip time of 342.9 days and a total mass of 12.9 MT (or an 8.7 MT payload mass), as shown in Figure 4. Given the optimal values, several trip time fixed trajectories were calculated. Changing the trip time below or above the optimal reduced the payload mass, because more propellant was needed for the required velocity change. These results were in good agreement with the MITEE and CIS point design values.

Electric propulsion systems require large power systems onboard the spacecraft. The two common power source options are solar energy for SEP and nuclear fission for NEP. When selecting a solar electric propulsion system for a Mars mission, it should be considered that solar radiation decreases at a rate of $1/r^2$, where r represents the distance from the Sun. On an orbit around Mars the Sun's radiation is ~43% of that near Earth, resulting in a 2.25 times larger solar panel requirement. This requirement has a significant effect on spacecraft mass and will be discussed with power source sizing in a later section. Further discussion on solar power generation is given in Section 5.

Electric propulsion options are summarized in Table 2. Electrostatic propulsion systems apply electrostatic forces to accelerate charged particles directly. If the particles are ionized atoms or molecules, the system is called an ion engine. Colloid thrusters employ fine sprays of liquid droplets instead of ionized particles. Electromagnetic (or plasma) propulsion systems accelerate matter using magnetic forces on a charge-natural, ionized gas (plasma). Electrothermal propulsion uses electrical energy to heat the working fluid (i.e., the propellant), which is then expands through a nozzle to achieve high exhaust speeds, the same way as that for rocket engines. Of these three electric propulsion options, only electrostatic and electromagnetic

Table 2. Electric propulsion summary [18]

Type	Parameter	Min.	Max.
Electro-thermal	Power (kW)	0.4	2
	Thrust (N)	1	10
	I_{sp} (sec)	300	800
	Run time (sec; hour)	Few sec	Hours
Electro-static	Power (kW)	5	50
	Thrust (N)	0.03-1	0.03-1
	I_{sp} (sec)	2k-5k	6k-10k
	Run time (years)	Years	Years
Electro-magnetic	Power (kW)	50	1000
	Thrust (N)	0.3	20
	I_{sp} (sec)	2000	5k-10k
	Run time (years)	Years	Years

systems are suitable for main propulsion systems. (Electrothermal systems, such as arcjet, have undesirable performance characteristics, namely low thrust and low I_{sp}.) For the first two systems, the specific impulse varies between 2000 and 10000 sec, with a thrust level up to tens of Newtons. To achieve 20 N of thrust the power system should provide 1 MW of power. For the upcoming Jupiter Icy Moons Orbiter (JIMO) mission a 100 kWe power source is under consideration to power its NEP system, but it is expected for future reactors to scale-up to higher power levels. Electric propulsion can offer very high specific impulse, but the thrust level is low. Electrostatic systems (e.g., ion propulsion and Hall thrusters) can be used for robotic surface missions, and also in in-space SEP and NEP systems. Electromagnetic (e.g., MPD) systems are suited for piloted and cargo missions and NEP.

The performance of *solar electric propulsion* systems has been assessed [15] based on the following assumptions:

- Two power levels were considered: 20 kWe and 50 kWe, representing current and next-generation solar power sources.
- In one scenario the spacecraft was launched to a 1000 km circular LEO; from there a SEP system was used to carry the payload to a 1000 km LMO.
- In a second scenario the spacecraft was launched to a positive C3 using a Delta IV Heavy launch vehicle, then captured to a 1000 km LMO.
- The duty cycle, assumed at 90%, was approximated by reducing the power by 10% from 20 kWe and 50 kWe to 18 kWe and 45 kWe.
- The calculations were optimized for I_{sp}.

As shown in Figure 4, for the first scenario the total deliverable mass and transfer time to Mars from LEO (including the spacecraft, propulsion system, payload, propellant, etc.) are ~11 MT and ~1200 days for a 50 kW system and ~8 MT and 1700 days for a 20 kW system. The same correlations between mass and trip times for the second scenario yields lower mass at Mars. This is due to the mass delivery characteristics difference between positive C3 and LEO for Delta IV Heavy launch vehicles. Mars is a low-energy destination; therefore, the I_{sp} optimizes at a rather low value for shorter flight times. Hence, the shorter trip time sections of the curves are rather academic.

Performance characteristics of a 100 kWe (η=70%) *nuclear electric propulsion* system were calculated [15]. After launching the spacecraft to a 1000 km LEO on a Delta IV Heavy launch vehicle, with an assumed mass at LEO of 20 MT, a NEP system is used to deliver the spacecraft from LEO to Mars. The calculations were optimized for I_{sp}. The results are shown in Figure 4 and Table 3. To achieve higher masses at LEO, in-orbit assembly is required.

In assessing potential future performance capabilities of NEP and SEP systems, Frisbee & Hoffman [19] evaluated megawatt-class SEP and NEP vehicles for Mars cargo missions. These assumed power and mass levels corresponded, however, to technologies well beyond our current capabilities; hence this reference is included here only for completeness.

Table 3. 100 kWe NEP system performance summary

Parameter	Min.	Max.
Flight time (days)	640	1440
Power (MW)	0.1	
I_{sp} (sec)	2500	6100
Heliocentric flight time (days)	361	625
Escape spiral (days)	222	608
Capture spiral (days)	56	206
Initial mass (MT)	20	
Final mass (MT)	10.8	15.5
Propulsion time (days)	450	1314
ΔV (km/s)	15	
Heliocentric transfer angle (revs)	0.68	1.22

In summary, Figure 4 provides comparison of the approximate bounding values for various transportation technology options covering the Mars exploration roadmap for the next few decades. It should be note that the NTP system-related calculations are based on an initial mass of 24 MT at LEO, while all other systems were assessed at 20 MT. The optimization routine for electric propulsion systems gave unrealistically low I_{sp} values at short transport times; therefore, the lower ends of the electric propulsion curves should not be used. Point values for NTP are in good agreement with time varying calculations. Similarly, the performance values for Delta IV Heavy launch vehicles, obtained from a NASA KSC database [14], correspond well with independent calculations. While Figure 4 does not provide exact performance values for mass and trip times, it can provide rough estimates on technology breakpoints and how these transportation system options relate to one another.

4. EDL Issues

Before discussing reactors, specifically surface-based nuclear fission reactors, *entry, descent, and landing (EDL)* limits are addressed. The highest power generated with a surface-based power source is influenced by the system mass, which is dependent on EDL limits. Thus, placing high performance and heavy nuclear reactors on the Martian surface requires improved soft landing capabilities.

Planetary EDL can be achieved through active or passive means. Active EDL is conducted by propulsive deceleration, which is similar to the method used during Moon landing. This option provides good control and results in accurate landing, but it requires a significant amount of propellant, hence lowering the landed mass.

Table 4. EDL limits — aeroshell scaling analysis [20]

Aeroshell Type	Parachute Envelope	Aeroshell Diameter (m)	Launch Mass (kg)	Entry Mass (kg)	Landed Mass (kg)	Launch Vehicle Required ($C3=15km^2/s^2$)
Viking	**Viking**	**3.75**	**2251**	**1971**	**1372**	Delta IV M+(4,2)
Viking	Viking	4.05	2579	2299	1603	Delta IV M+(4,2)
Viking	Viking	4.572	3210	2930	2048	Delta IV M+(5,2)
Ellipsled	**Viking**	**3.75**	**3408**	**3000**	**1633**	Delta IV H
Ellispled	Viking	4.05	3951	3500	1909	Delta IV H
Ellispled	Viking	4.572	4992	4460	2438	Delta IV H
Ellispled	**Viking**	**3.75**	**3200**	**2792**	**1453**	Delta IV H
Viking	Mach 3	3.75	3481	3201	2427	Delta IV M+(5,2)
Viking	Mach 3	4.05	4014	3734	2834	Delta IV H
Viking	Mach 3	4.572	5038	4758	3616	Delta IV H
Ellispled	**Mach 3**	**3.75**	**4953**	**4500**	**2895**	Delta IV H
Ellispled	Mach 3	4.05	5752	5249	3379	Delta IV H
Ellispled	Mach 3	4.572	7287	6689	4308	Delta IV H

Note: Bold indicates calculated data points based on analysis data
Viking parachute envelope: Mach 1.4 to 2.2 and dynamic pressure 400 to 800 N/m^2
Mach 3 envelope: Mach 2.0 to 3.0 and dynamic pressure to 1300 N/m^2

Passive entry uses an aeroshell and a parachute. The aeroshell decelerates the landing craft from hypersonic to supersonic velocity by converting its kinetic energy to heat through friction between the atmosphere and an ablative heatshield. From that point on parachutes (supersonic and subsonic) take over, reducing the velocity to near zero. Soft landing is achieved with airbags or retrorockets.

At present, Viking-type aeroshell and parachute designs provide the highest landable mass. Aeroshell size is dependent on the diameter of the launch vehicle fairing. A Delta IV-H launch vehicle with a 5 m diameter fairing can accommodate a typical 4.572 m diameter aeroshell. When combined with a Viking-type parachute, the landed mass is ~2 MT, as shown in Table 4 [20]. This limit is dependent, however, on weather conditions and landing location. At a high-elevation landing site under adverse weather conditions, such as for moderate (20 to 40 m/s) variable winds and low seasonal air density, the landed mass might be limited to as little of ~1 MT. The Mars exploration roadmap predicts an ever increasing mass requirement on the surface, necessitating improvements to this limit.

One way to achieve higher landing mass is by improving the aeroshell and parachute designs, thus increasing the mass for each single landing. Aeroshells under development include ellispled (mid L/D) and spherical dome designs. A qualification program for Mach 3 parachutes is also under consideration. Development cost for these technologies is minor when weighted against the multibillion-dollar development effort for nuclear reactors, especially when technology returns are accounted for. Benefits include higher descent and landing mass, allowing for larger nuclear fission reactors on the Martian surface with higher power levels. A combination of a Mach 3 parachute and a Viking type or an ellipsled aeroshell could land ~3.6 MT and ~4.3 MT, respectively. For the latter configuration the launch mass is ~7.3 MT, which is consistent with the payload mass for Delta IV-H launch vehicles (~8 MT), as discussed in the transportation section. It can also be seen that Mach 3 parachutes may indirectly contribute to reactor development by allowing higher power levels on the surface.

A second possibility is to accumulate ground assets at a given location through pinpoint landing. Landing accuracy can be improved with guidance. For example, the Pathfinder mission had a landing accuracy of ~150 km, as shown in Figure 5. With optical navigation but without entry

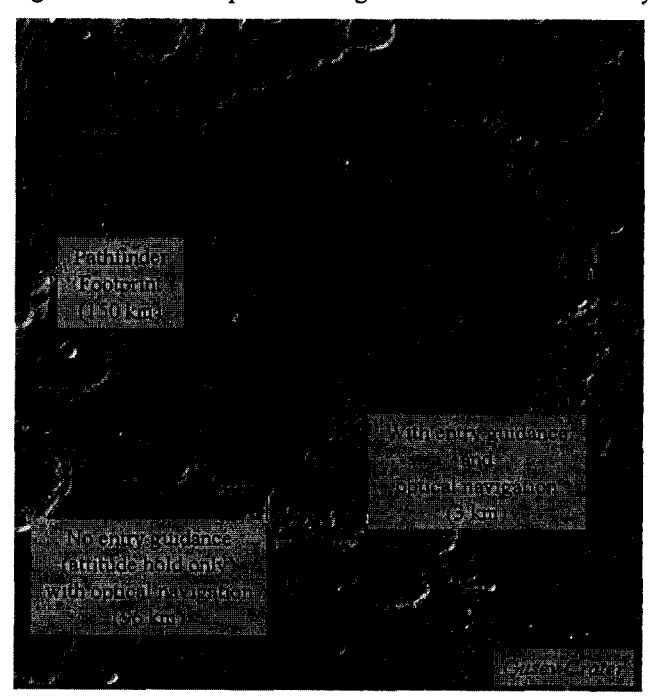

Figure 5 - Landing accuracy [20]

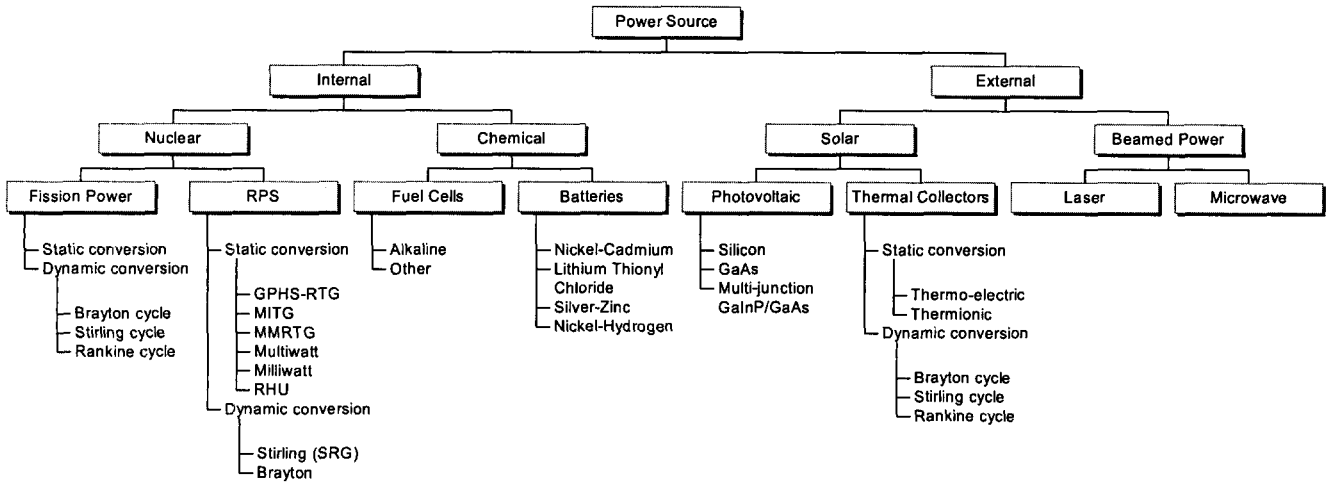

Figure 6 - Power source categories

guidance, this can be improved to ~96 km, while with both entry guidance and optical navigation accuracy is predicted at ~3 km, still far from pinpoint landing (reproduced after [21]). Recent studies for pinpoint landing accuracy on Mars set a target of ~100 m distance between surface assets, with a landing mass of ~1 MT. This mass could accommodate a dedicated 6 to 7 kWe surface reactor (but without additional payload). Considering a landing mass of ~3.6 MT with a Viking-type aeroshell and a Mach 3 parachute, a dedicated surface reactor can be sized up to as much as 50 kWe, as will be shown in Section 5.

5. Power Generation Issues

In this section breakpoints are identified for nuclear fission and radioactive power systems (RPS), that can be used for in-space and on-surface power generation. Nuclear systems use internal sources to generate power. Radioisotope thermoelectric generators utilize nuclear decay heat, while reactors operate on nuclear fission. Batteries and fuel cells are chemical systems using internal energy sources, while solar and beamed power systems are based on external power sources. (Power source types are broken down in Figure 6.) For completeness the conventional technologies are judged against fission-based systems. Thermal-to-electric power conversion is common to all of these systems; thus the various methods and technologies are introduced first.

Table 5. Thermal-to-electric conversion efficiencies [22][23] [24]

Converter	Type	Efficiency (%)	Comments
Static	Conventional TEC	~7-10	In space
	TIC	~10-20	In space
	AMTEC	~20-30	Development
	TPV	~20-30	Development
Dynamic	Brayton	~20-35	Development
	Stirling	~25-30	Development
	Rankine	~15-20	Development

Power Conversion Options

Nuclear fission and decay-based power systems and solar radiation generate thermal energy which, in turn, is converted into electric power. Conversion is achieved through static or dynamic methods.

Static converters operate without moving parts and employ thermo-electric or thermionic systems. In thermo-electric conversion (TEC) a potential is produced the same way as in thermocouples. Thermionic conversion (TIC) depends on the production of a current due to the flow of electrons generated by thermionic emission from a hot electrode. Under development, Alkaline Metal Thermal to Electric Conversion (AMTEC) [22][23] converts infrared radiation into electricity using liquid metal ions, which are charged atoms. Thermo-photovoltaic (TPV) converters change infrared radiation emitted by a hot surface into electricity. Design goals for AMTEC and TPV technologies (currently at TRL-3) are set to increase RTG performance through a three-fold efficiency increase to ~20 to 30%.

Dynamic systems have moving parts that transform heat into mechanical energy, which is then used to generate electricity. Conversion is achieved through a Stirling, Brayton, or Rankine thermodynamic cycle. Stirling-cycle converters employ a single-phase working fluid. The cycle consists of two isothermal processes (compression and expansion) and two constant volume processes (heating and cooling), with a conversion efficiency of ~25 to 30%. Brayton-cycle converters represent an option for low mass and highly scalable (kilowatts to megawatts) power generation and use a single, compressible working fluid. The cycle consists of a constant-pressure expansion/heating, an adiabatic expansion/cooling, a constant-pressure compression/cooling, and an adiabatic compression/heating with a power conversion efficiency of ~20 to 35%. Rankine-cycle converters use a two-phase fluid system, employing a boiler, turbine, alternator, condenser, and

pump, essentially in the same manner as terrestrial power stations, and offering conversion efficiencies of ~15 to 20%.

As shown in Table 5 [22][23][24], dynamic converters are more efficient, while static converters provide higher reliability and simplicity. Better power converters benefit system performance in two ways. They increase electric power output or reduce radioactive fuel and system mass requirements; in effect, higher efficiencies increase specific power. Thus, significant development efforts are dedicated to improving power conversion efficiency.

Radioisotope Thermoelectric Generators

Radioisotope thermoelectric generators (RTGs) are powered by nuclear decay of radioactive fuel and can be used for both in-space and on-surface applications. RTGs, which convert heat to electricity, consist of two main parts, a heat source and a power conversion system. The heat source includes the radioactive fuel, which is encapsulated in clad and shell layers. These protective layers prevent the release of radioactive material to the environment in case of an accident during launch or atmospheric reentry. RTGs have many advantages compared to conventional technologies. They are self-contained; can operate continuously for an extended period of time; are compact, strong, and highly reliable; are unaffected by radiation environment; and are independent from solar energy. Modular design enables scalable power levels and thermal outputs, which is then converted into electricity through static or dynamic conversion methods. RTGs are designed for an operational lifetime of ~5 to 15 years, although this is routinely extended based on the predictable decay characteristics of its plutonium fuel (Pu^{238}), with a half-life of 88 years.

The first US spacecraft to use an RTG was launched in 1961. The SNAP-3B7 (Space Nuclear Auxiliary Power) RTG generated 2.7 W of power and operated for 15 years. Pioneer 10 (SNAP-19) and Voyager 1 and 2 (MHW-RTG for Multihundred Watt Radioisotope Themoelectric Generator) have been operating for up to three decades. The MHW-RTG provided 157 We (2400 Wt) (BOL), with dimensions and weight of 40 x 61 cm, and 38.5 kg. Today's soon to be phased out General Purpose Heat Source RTGs (GPHS-RTG) generate about 285 to 290 We of electric power (4400 Wt) (BOL), with dimensions of 114 x 43 cm and a system mass of 75 kg (see Figure 7). The system includes 8 kg of Pu^{238} divided into 18 GPHS modules. The Galileo and Cassini spacecraft utilized two and three of these GPHS-RTGs, respectively. To date, the Department of Energy (DoE) supplied 45 RTGs for 25 space missions [25].

In addition to flight-tested RTGs, a number of power system are under development or are being considered for future development. Modular Isotopic Thermoelectric Generators (MITGs) are not yet flight tested, but they are designed to generate 282 We (3000 Wt) of power. The dimensions are 75 x 16.5 cm, and the weight is 27.2 kg [26]. Two new advanced radioisotope power systems (ARPS) are under development by the DoE and NASA. One is a small RTG-

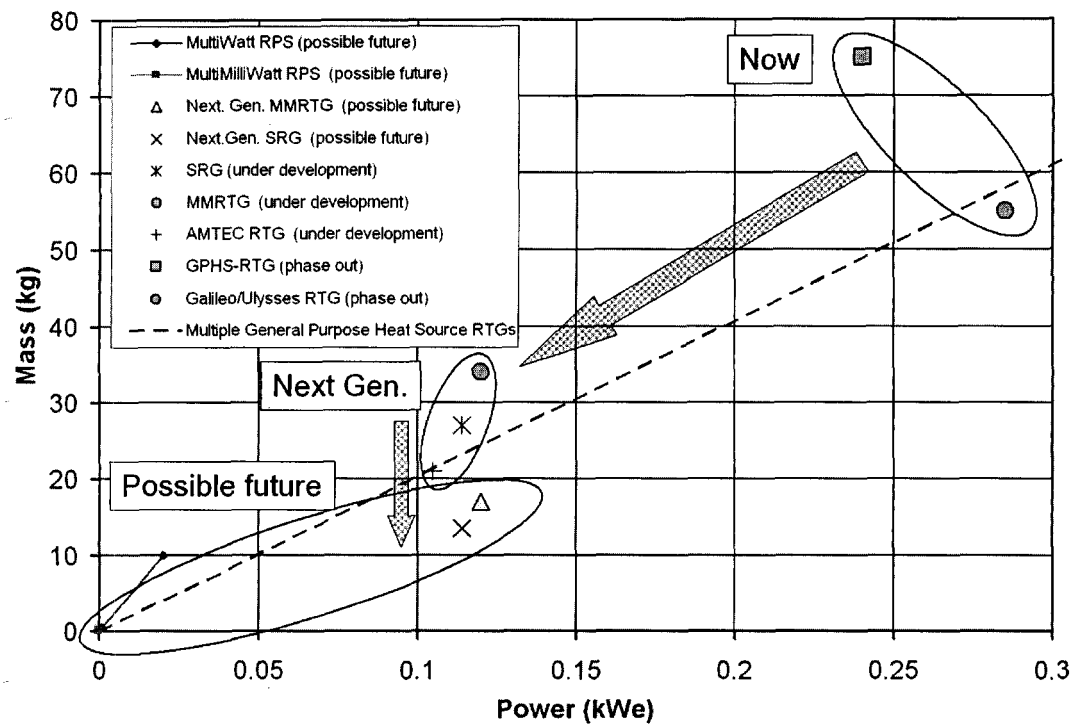

Figure 7 - RTG development trends

based electric power system built on design heritage; it is called a Multi-Mission Radioisotope Thermoelectric Generator (MMRTG). It is half the size of a GPHS-RTG, with a system mass of 34 kg, which includes ~4 kg of Pu^{238}, delivering ~120 We, and designed for a minimum lifetime of ~14 years. MMRTGs can be used for both in-space and on-surface missions (e.g., this type is under consideration for the upcoming MSL mission). A second system uses a higher efficiency dynamic Stirling conversion that is four times more efficient that that of a thermoelectric conversion system. This Stirling Radioisotope Generator (SRG) requires less fuel (~1 kg of Pu^{238}) for the same power production (~114 We). The system mass for an SRG is 27kg [27] [28] [29][30]. Performance and mass data on ARPSs for three power conversion technologies, namely TPV, AMTEC and Stirling, are given in [31].

Thermal-to-electric conversion efficiencies for space proven RTGs are ~6 to 7%. Recent testing of a 2 kW solar dynamic power system demonstrated Brayton conversion efficiencies of over 29% using 1970's component technology [32]. The same conversion technology can also be used with radioisotope-based systems.

Future second-generation SRG and MMRTG power sources are envisioned with higher power conversion efficiencies, resulting in the same power level but half the system mass. Multiwatt (1 to 10 We) and milliwatt (10 to 100 mWe) systems are also in the planning phase [33]. Further information on RTGs can be found in [27].

Radioisotopes can be also used for heat generation. Radioisotope Heater Units (RHUs) provide 1 Wt of thermal power with only 2.7 grams of fuel. A typical 3.2 x 2.6 cm RHU weights 40 g; with 2% efficiency it could generate ~20 mW of electric power (the Galileo spacecraft used 120 RHUs). These units are stackable; several packed into a protective shell may provide as much as 0.1We [25] [34].

RTG development efforts focus on a number of factors. Economic considerations point to reusability, reflected in the MMRTG design. Regulatory concerns call for a reduction of radioactive fuel. This can be achieved by either scaling down power generation (multiwatt or milliwatt systems) or increasing power conversion efficiency (SRG or Brayton systems). These trends are captured in Figure 7.

Nuclear Fission Reactors

Nuclear fission reactors are well established on Earth, with many decades of operational experience. Examples include research and commercial reactors, and naval applications (submarines, carriers). Space reactors are different in many aspects from their terrestrial counterparts. The specific mass is lower due to transport/EDL limitations and to high system integration (e.g., no containment, emergency cooling). Reliability and long lifetime require autonomous operation, diagnostics and maintenance. The operating environment is harsh both in space and on the planetary surface. Space reactors must endure cryogenic environments, lack of gravity, and dynamic loads during launch and EDL. Yet they provide continuous power without reliance on an external power source such as the Sun. An important distinction between fission reactors and RTGs is that uranium fuel for the former is essentially non-radioactive before reactor startup, while RTGs are radioactive throughout the mission.

Space reactor power systems consist of:

- Reactor core where heat is generated through self-sustained nuclear fission reaction
- A primary heat transport loop that removes heat from the core
- Various control mechanisms and shielding that sustain the reaction inside and shield radiation outside of the core
- Power converter to convert thermal energy into electric power
- Excess heat is rejected through radiators

Due to the complexity of nuclear systems and the large number of development programs, the summary provided in this paper is incomplete. Detailed descriptions of space nuclear fission reactors and programs are readily available in open literature. Therefore, this section addresses only some of the basic concepts and provides examples for past development efforts (see Tables 6 and 7) and high-level design trends for the present and near future (see Figures 8 and 9).

Table 6. Historical fission reactor performance

Reactor and power conversion types		Mass (kg)	Power (kWt)	Power (kWe)	Lifetime
SNAP 10A (SNAPSHOT)	TEC	435	46	0.5	Demonstrated for 43 days
SP-100 10 kWe	TEC	-	-	10	Study
SP-100 baseline	TEC	4518	2400	100	Development for ~7 years
SP-100 advanced	TEC	3500	2400	100	Study for ~7 years
SP-100 dynamic	Brayton	9506	383	100	Study
SP-100 dynamic	Brayton	14337	2657	550	Study
SP-100 dynamic	Stirling	20004	2500	825	Study
Topaz (Cosmos-1818, -1867)	TIC	1200	150	5	Demonstrated for 342 days
Topaz II	TIC	1000	135	5.5	Estimated ~1.5 to 3 years

Table 7. Historical NTR peak performance

Parameter	Value	Reactor
Power	4100 MW	Phoebus-2A
Thrust	930 kN	Phoebus-2A
Hydrogen flow rate	120 kg/s	Phoebus-2A
Equivalent I_{sp}	845 s	PEWEE
Minimum reactor specific mass	2.3 kg/MW	Phoebus-2A
Average coolant exit temperature	2550K	PEWEE
Peak fuel temperature	2750 K	PEWEE
Core average power density	2340 MW/m^3	PEWEE
Peak fuel power density	5200 MW/m^3	PEWEE
Accumulated time at full power	109 min	NF-1
Greatest number of restarts	28	XE

Theoretically, reactors are scalable up to the kWe to MWe 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 kWe reactor may use ~150 fuel pins, similar 150 kWe and 1 MWe designs employ ~300 and 600 pins, respectively. (Other fuel types include isotopes [Pu^{239}, AM242m] and compound fuel [e.g., U-metal, UZrH, UO2, UCZrN], arranged in pin, particle, foil, or prismatic geometries.)

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

- Liquid metal (lithium-coolant)
- Heatpipe (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 can introduce a single-point failure. Heatpipes are flexible, easy to test, and safe, 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 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. 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. 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. This can increase the system mass significantly.

Historically, the US has had limited experience with nuclear reactors in space. During the past 50 years both the US and the USSR developed and launched space reactors and RTGs, gaining experience primarily with one or the other. Sending 45 RTGs and one reactor into space, the US accumulated mostly ground test data on fission reactors. In contrast, the USSR launched only 6 RTGs, but 37 space reactors.

An incomplete list of past programs in the US demonstrates the diversity of research and development efforts. These included SNAP/SPUR (fission and RTG, TEC and Rankine, and under it the Medium Power Reactor Experiment — MPRE and the Gas Cooled Reactor programs); Rover/Nerva (NTP); SP-100 (fission); DoE 40 kWe thermionic reactor program (fission); Air Force bi-modal study; and the multimegawatt program (fission). Russian programs include Rorsat, Topaz 1 and 2, and the Confederation of Independent States (CIS) Nuclear Thermal Rocket Program.

The SNAP program developed fission reactors (even numbers) and RTGs (odd numbers) in the 1960s and early 1970s at a cost of $840 million in then-year dollars. The reactors used U-ZrH fuel and liquid metal (NaK) cooling with TEC or Rankine power conversion. Several ground tests had been performed, and one had flown in Earth orbit (SNAP-10A). [35] Russian fission reactors were typically U-Mo alloy or UO2 fueled and liquid metal (NaK) cooled, with TEC (>30%) or TIC (2%) power conversion. During a short operational lifetime (<= 1 year) only low power levels were achieved (2 to 5 kWe). Rorsat designed 2 kWe TEC conversion systems, operating ~4 to 5 months. Two Topaz 1 reactors were launched in 1987. These multicell TIC units generated 5 kWe (BOL) of power. Topaz 2 was developed but not launched. As a single-cell TIC system, it was designed to generate 6 kWe (135 kWt) of power during its 1 to 3 year lifetime, with its 27 kg of 96% enriched U^{235} fuel. The reactor mass was 1061 kg, with a radiator area of 7.2 m^2 and reactor diameter and height of 1.4 and 3.9 m, respectively [36]. To date 37 Russian reactors have been

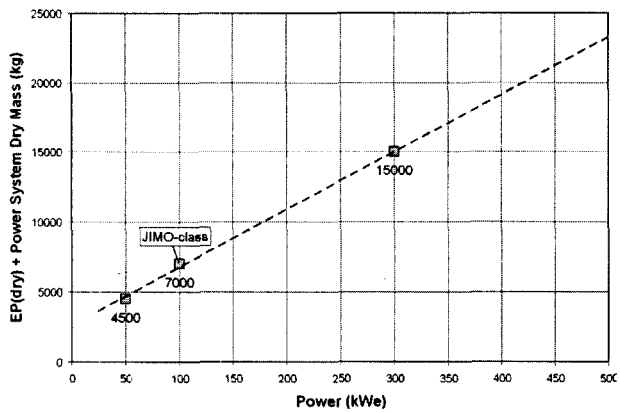

Figure 8 - In-space propulsion sizing [26]

flown in space. The power and mass data for these fission reactors is summarized in Table 6.

The NERVA program ran between 1955 and 1973, testing nuclear thermal rockets named KIWI, NRX, and Phoebus, and at a cost of ~$1.4 billion [37][38]. Top performance data for nuclear thermal propulsion from the NERVA program is shown in Table 7.

Ongoing development efforts today address in-space NEP, NTP, and bi-modal propulsion, small surface reactors, and power conversion technologies. Without trying to provide a full literature review, a few examples are given below. It is expected that the upcoming Project Prometheus Program and its flagship mission the Jupiter Icy Moons Orbiter (JIMO) will answer key questions about nuclear fission-based technologies. While the exact configuration of JIMO is still under consideration, a possible option for a 100 kWe NEP system is described by El-Genk and Tournier [39]. A correlation between mass and power for in-space nuclear fission reactors is shown in Figure 8, with the data extrapolated up to 0.5 MWe [26].

The Heatpipe-Operated Mars Exploration Reactor (HOMER) is a derivative of the Heatpipe Power System (HPS) designed at Los Alamos National Laboratory (LANL) as a lightweight reactor for electricity production on Mars. The designed power range is 1 to 20 kWe, which may scale up to 50 to 250 kWe for manned flight. The 20 kWe (125 kWt) point design reactor weights 385 kg, with a total mass of 1385 kg (including reactor, instruments, and control; power conversion; radiator; shielding; and 20% contingency) [40]. Power conversion technologies apply differently for in-space and surface reactors. For low-power RTGs and surface reactors, Stirling conversion is the obvious choice from a mass perspective, with a crossover point to a Brayton system around 15 to 20 kWe. For high-power in-space reactors above the 30 to 40 kWe power range, Brayton conversion is preferred [41].

Future testing of nuclear rockets is outlined in the Subsurface Active Filtering of Exhaust (SAFE) concept. It would allow variable sized engines to be tested for long times [42]. Another SAFE acronym (for Safe Affordable Fission Engine) refers to an electrically heated system that allows testing and development of fission systems in non-nuclear facilities, saving time and money. Resistance heated tests on the 30 kWe SAFE-30 testbed was performed at NASA-MSFC [43]. NASA-GRC is testing small surface reactors with Brayton power conversion. Brayton conversion represents low mass and highly scalable (kilowatts to megawatts) power generation. To date, a 2 to 15 kWe class power source with a conversion efficiency of ~30% has accumulated over 40,000 hours of operation. Other small power sources, such as a 2 kWe testbed and a 25 kWe engine, are designed as a modular building block for 100 kWe-class electric propulsion and Mars surface power applications [44]. A small 3 kWe surface reactor with Stirling power conversion is described in [45]. The total mass for this system is 775 kg. To calculate the mass-power relationship for a small surface reactor, this point design value was scaled up, as shown in Figure 9 [45].

This brief summary provides only a snapshot of the ongoing research activities for nuclear systems. The power-mass relationships for in-space and surface reactors, plotted in Figures 8 and 9, can, however, help determine rough performance breakpoints when matched against transportation and EDL limits. For example, from transportation limits the ~15 MT mass can support a ~200 kWe reactor in space at LMO, while today's EDL limit of ~1 to 2 MT will bound the surface reactor's power at around 5 to 7 kWe.

Other Technologies

In this subsection alternative technologies to nuclear fission power are discussed briefly, such as solar power, fuel cells, and batteries. These serve as a comparison basis to the fission-based technologies discussed above.

Solar power is based on solar radiation and is considered an external power source. Solar flux decreases with the square of distance from the Sun, the presence of a Martian

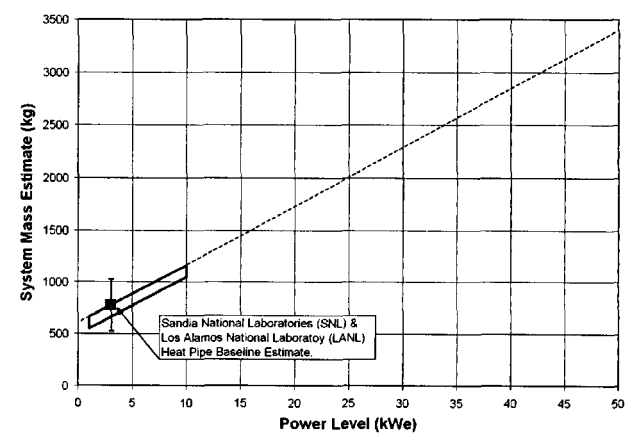

Figure 9 - Small surface reactor sizing [45]

atmosphere, and potential dust storms there. The solar constant (S) at the orbital distance of Earth from the Sun is 1367 W/m^2. Compared to that (100%), solar irradiance values are significantly lower at Mars, as measured in orbit (43%), on the surface under clear conditions (22%), and under cloudy conditions for local storms (13%) and global storms (6.5%).

Solar radiation can be converted into electric power using solar thermal collectors or photovoltaic (PV) arrays. Solar thermal collectors employ static or dynamic conversion methods, as discussed previously. They may provide an alternative to PV arrays. Large, lightweight solar concentrators may use thin-film inflatable technology, offering a factor of five improvement in areal density (kg/m^2) compared to conventional rigid panel concentrators. PV arrays employ solar cells for power conversion. In this assessment three types of PV solar cells are considered and compared against the performance of nuclear power sources. Single junction cells, such as silicon (Si: η_{conv}=14.8%) and gallium arsenide (GaAs: η_{conv}=18.5%), convert photons of near-infrared energy to usable energy. Multijunction or multilayered solar cells, such as multijunction gallium indium phosphide / gallium arsenide (GaInP/GaAs: η_{conv}=22%), use different spectrums of sunlight, hence increasing conversion efficiency [24].

Solar power may not be suitable for certain missions. Seasonal polar environments result in insufficient illumination, shutting down the mission for up to 6 months. End of life (EOL) power production capability depends on cell degradation, which is 3.75, 2.75, and 0.5%/yr for Si, GaAs, and multijunction GaInP/GaAs, respectively [24]. Specific performance ranges from 14 to 47 W/kg, with a high-end performance of 66 W/kg at the beginning of life (BOL). Body-mounted arrays weight less than planar (flat panel) arrays, while aluminum honeycomb panel technology weights approximately 3.19 kg/m^2.

Solar panel mass is estimated as a function of power, generated on the Martian surface and sized for cloudy global storm conditions after 3 years of operation. For a 100 W configuration the solar panel size corresponds to 8.52, 6.61, and 5.19 m^2 for the three cell types. The matching mass values are 27.2, 21.1, and 16.57 kg, respectively. Solar panel size and mass scales linearly with power such that a ten-fold increase in power results in the same magnitude increase in panel size and mass, although this type of scaling is significantly bound by EDL limits. For example, assuming a 2 MT landing limit with 50% of the mass assigned to one of the three solar panels types, the achievable mission power is limited to about 4 to 6 kWe. These results are also assessed against system mass values for a small surface-based fission reactor and for multiple GPHS RTGs. As shown in Figure 10, for power levels above ~3 to 5.5 kWe small fission reactors provide a mass advantage against solar panels. Similarly, above ~4 kWe surface-based reactors become a better choice than multiple GPHS RTGs. The considered reactor mass data includes shielding and power conversion, while the solar panels are calculated without any of the supporting components and structures. Therefore, the crossover values given above are conservative, where nuclear reactors become a better choice at even lower power levels. Is should also be noted,

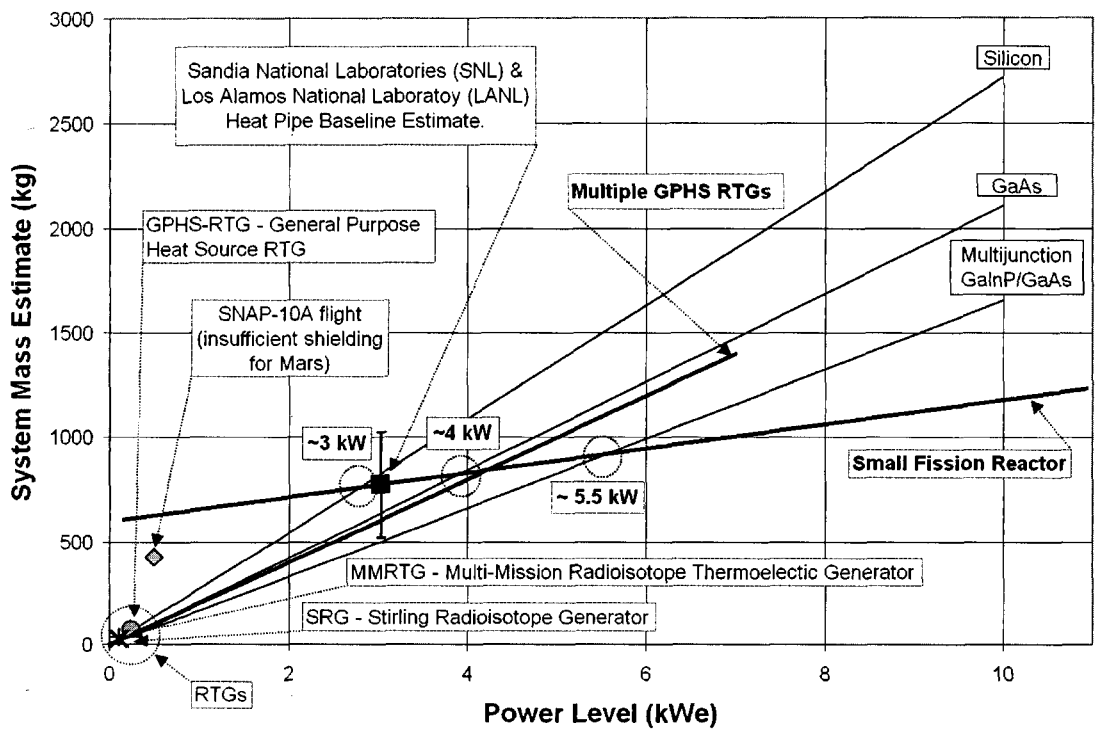

Figure 10 - Comparison between solar and fission power parameters

however, that the comparison in Figure 10 is limited to system mass against power only. Additional parameters, such as duty cycle, cost, maintenance, physical dimensions, safety, and planetary protection issues, among others, will significantly influence the trade space and hence the final power source selection for any given mission.

Fuel cells are electrochemical devices that convert internal chemical energy to electricity through an oxidation reaction. For space applications alkaline fuel cells are common, based on hydrogen-oxygen fuel. When combined, electricity and heat are produced without combustion or pollution, but with water as an operational by-product, that can be further utilized during the mission. Because the fuel is converted to electricity directly, the fuel cell efficiency is higher (~70%) than that of other power sources. Such a system does not have moving parts; hence its operation is quiet and highly reliable. These self-contained generators provide continuous operation independent of sunlight. Because the reactant is included within the system, tank size and system mass increase with mission duration. Individual fuel cells are stackable. The number of stacked fuel cells and the surface area determine the voltage and current and consequently the total electric power generated [46].

In the 1960's fuel cells provided power to NASA's Apollo, Apollo-Soyuz, and Skylab programs, based on alkaline electrolyte (molten KOH) technology. Fuel cells were used on 18 Apollo missions, operating for over 10,000 hours combined without an in-flight incident. Each of the three 28 V fuel cells on-board provided 1.5 kWe of power (2.2 kWe burst), operated in parallel, and weighted 113 kg. Second-generation fuel cell technology [46] is used on the Space Shuttle Orbiter, producing ~10 times the power of the Apollo-type models. Each of the three generators on the Shuttle employs 96 individual fuel cells with alkaline (KOH) electrolyte technology, providing electricity for the 28 Vdc bus and generating 12 kW of power (16 kW burst). The dimensions and system mass for each generator are 35 x 38 x 115 cm and 118 kg, respectively. The efficiency is 70%, corresponding to high specific power (275 W/kg) and low reactant mass. The system starts up in 15 minutes and can operate up to 2600 hours (~13 missions) before refurbishment, although the typical continuous operation is around 7 to 14 days due to fuel storage and mission time limitations. As a by-product, the 3 fuel cells can produce about 104 kg of water per day at a rate of 0.36 kg/kWh. During 113 Space Shuttle missions, fuel cells operated for 90,264 hours. Further information regarding the history of fuel cells can be found in [47]. Fuel cell technologies can greatly benefit short-term space missions requiring high power. Long duration missions beyond 7 to 14 days may, however, necessitate other systems when high and continuous power is needed and time is measured in months or years.

Batteries provide energy storage using internal chemical power as part of the electric-power subsystem. Space missions typically employ nickel-cadmium (Ariane-4), lithium thionyl chloride (Titan-IV), silver-zinc (rovers), or nickel-hydrogen (Space Station) batteries. These scalable energy storage systems are highly reliable but heavy, often significantly affecting the total system mass. To increase voltage or current, the units are connected in series or parallel, respectively. Battery life cycle is influenced by temperature, depth-of-discharge, rate of charge and discharge, and degree of overcharge. Reconditioning of the battery through periodic full discharge can prevent capacity loss. Primary batteries generate power within the spacecraft and usually last for less than an hour. They are used on launch and reentry vehicles and rovers, providing not well-regulated power, typically 28±5 V. Secondary batteries on a spacecraft are used only for energy storage. They are important during two operating scenarios: during peak load (2 to 3 times the average load) and eclipses (resulting in thousands of discharge cycles) [24]. Mission duration may necessitate battery change.

Two examples for today's high-end batteries and potential future replacement developments are Space Station batteries and flywheels. The Space Station's Battery Orbital Replacement Units (BORUs) have a five-year design life. Each of the 48 BORUs weighs 166 kg with a dimension of 3.66 x 3.66 x 3.66 m, and each Battery Charge/Discharge Unit (BCDU) can regulate 2 BORUs, providing up to 6.6 kWe of electric power [48]. Flywheels are under development to provide an alternative to batteries on the Space Station. They offer greater capacity and cleaner and more economical power than chemical batteries, without the need for regular replacements. Energy is stored by spinning up the wheel to ~60,000 rpm, using an electric motor powered by the station's solar panels. When the motor is switched to generator mode, de-spinning the wheel discharges energy [49].

Batteries are appropriate for short missions (<1 week) or as backup power options for longer missions (>1 week). They can supplement main power sources and can deliver high power (many kW) over a short discharge (e.g., up to 1 day). Hence, batteries are not applicable for missions that last for weeks, months, or more, and that require continuous high power.

Power beaming by microwave from space to Earth was first suggested in 1968 [50]. Examples for this method are given in [51]. In the 1980's NASA extended the technology to laser-based power beaming between space assets, then later from ground-to-space [50][52]. Landis provides a numerical example for power beaming from Earth to the Moon. With a GaAs laser diode array, a lens diameter of 2 m, a distance of 4×10^8 m and assuming diffraction limited beam spread (accounting for atmospheric turbulence), he calculates the total spot radius at the Moon as 250 m with a corresponding illuminated area of 0.2 km^2. Using a 12 MWe power source at the sending end, the received power at the Moon is ~50 kWe after all conversion and beaming losses are accounted

for. This corresponds to an end-to-end beaming efficiency of ~0.4% [53].

When power beaming is considered for Mars missions, the possible cases may include (a) orbit-to-orbit, (b) orbit-to-surface, (c) surface-to-orbit, and (d) surface-to-surface configurations. Although the distance between an orbiting spacecraft and the Martian surface is less than the aforementioned Earth-Moon distance (even if aerostationary orbit is assumed), the power beaming efficiency is still expected to be in the lower single-digit range. Other issues to consider include pointing accuracy and visibility between the two spacecraft for cases (a) to (c), atmospheric absorption and unfeasibly large receiving antennas on the surface for cases (b) and (c), and EDL mass limits with consequent power limitations for cases (c) and (d). Of the four cases, case (d) seems to be the only promising configuration, if combined with pinpoint landing of numerous surface assets. Landed crafts in close proximity provide good intervisibility and low beaming loss due to distance and atmosphere. (Beamed power decreases inversely with the distance squared.) In case of a landed mass of ~3.6 MT (see EDL limits before), the maximum power generated by a dedicated surface reactor is ~50 kWe. Assuming 1 to 2% beaming efficiency, the received power is ~0.5 to 1 kWe. Considering the trades from added mass by the power conversion systems on the receiving end and the small amount of transferred power, this option should be weighted against providing independent power to each surface asset. From aerostationary orbit a 200 kWe reactor could beam ~1 to 2 kWe of power to the surface (end-to-end). While this seems reasonably high, the corresponding 0.1 to 0.2 km^2 collector antenna area on the Martian surface (i.e., 20 to 40 football [soccer] fields) would present severe logistical problems with EDL, deployment, and maintenance. Microwave and laser beaming technologies differ in many ways, including antenna configurations; however, beaming efficiencies and antenna size are similar between the two, and hence the same conclusions apply.

Based on these assumptions it is concluded that power-beaming technologies require significant improvements (i.e., 2 orders of magnitude in conversion efficiency from ~0.4% to ~40%) before they can be seriously considered for future Mars missions.

6. CONCLUSIONS

Transportation and power generation options are assessed to identify breakpoints for near and future Mars exploration missions, with an emphasis on nuclear fission power. The study focused on key parameters, such as mass, power and time, as specified in the assessment architecture. It is evident from the information presented thus far that enabling and supporting technologies impose severe limitations on nuclear systems and power availability, both in orbit around Mars and on the Martian surface. While fission reactors can be scaled up, as demonstrated through terrestrial power stations and naval applications, the actual bottlenecks are due to transportation, EDL, and power conversion. Hence, future developments must advance these technologies to allow for a paradigm shift from power limited to power enabled.

Reactor power level is a function of the system mass and the technology options. Thus, in-space power generation is predominantly affected by the limits of today's launch vehicle technologies. At present, the Delta IV Heavy is our most capable launch vehicle. It can deliver payloads to LEO or directly to a C3 escape velocity (with its third stage kick motor). On a direct approach to LMO the total mass is limited to ~8 MT. The mass limit to LEO is ~20 to 24 MT, dependent on orbital parameters. Nuclear propulsion (NEP, NTP) is conceived to initiate from a "nuclear safe orbit" (~1000 to 2500 km) due to obvious planetary protection issues. (Non-nuclear LEO missions launch to as low as a few hundred kilometers.) Missions initiating from LEO use a second propulsion system (SEP, NEP, NTP, or chemical) to transfer the spacecraft to LMO. A 100 kWe NEP system could deliver ~15 MT to Mars in ~1400 days, but most of the time would be spent spiraling out of and into the gravity wells of Earth and Mars. A ~15 MT spacecraft at LMO could support a ~200 kWe in-space nuclear reactor, assuming that the reactor mass is ~10 to 11 MT and the payload is ~4 to 5 MT. This is impractical, however, because it assumes two separate nuclear reactors in orbit: a 100 kWe system for transportation and a 200 kWe system for power generation. Instead, a single reactor should be sized iteratively, supporting both propulsion and power generation. This would allow for faster interplanetary transfer and higher mass and in-space power generation. An NTP system could deliver ~13 MT to Mars in ~200 to 350 days, depending on the trajectory option. This fast transfer scenario could marginally support a 100 kWe JIMO class in-space reactor (~7 MT) but leaving only ~1 MT for payload. The specific impulse of an NTP system is twice that of a chemical propulsion system. Hence, it can deliver 50% more mass for the same trip time or the same mass twice as fast. Optionally, a bi-modal nuclear system could be used for both propulsion and in-space power generation. Because Mars is a low-energy destination, SEP options do not necessarily provide an advantage in deliverable mass. With the increase of power (e.g., from 20 to 50 kW) the delivered total mass also increases, although it can result in a significantly larger (not quantified) mass for the solar panels at the expense of the payload mass. When SEP is launched directly to positive C3, the initial mass is limited to about ~8 MT, providing little advantage compared to a direct launch by a chemical rocket. (These reported mass values represent total mass, which includes the propulsion system, remaining propellant, payload, and so on.)

Further improvements to deliverable mass limits call for heavy lifters beyond Delta IV-H. In-orbit assembly in LEO or at a Lagrange Point can multiply initial mass limits,

compared to single launches. Gravity assist (Venus swing-by) and aeroassist (aerobraking, aerocapture) can reduce propellant requirements and increase payload mass for the same delivery capacity but could introduce planetary protection related issues. Similarly, propellant generated in-situ (e.g., on the Moon or Mars and its moons) could be beneficial in the same way. Interplanetary transport time can be reduced or mass limits increased by improvements to specific impulse and thruster efficiency. These options should be further examined when design or in-situ data becomes available.

On-surface power generation is bounded by EDL limits. The maximum landing mass on Mars with a Viking-type aeroshell and parachute is ~2 MT at low elevation and good weather conditions. At high elevation and in adverse weather this limit reduces to ~1 MT. The higher mass could support up to ~15 to 20 kWe, and the lower mass ~5 to 7 kWe of power generation. With the same aeroshell and a Mach 3 parachute the limit increases to ~3.6 MT. An assumed mass breakdown of ~2 MT / 1.6 MT between the small surface-based nuclear reactor and the rest of the landing craft would allow for ~25 kWe. Future larger launch vehicles with increased fairing size would accommodate larger advanced aeroshell designs (ellipsled – mid L/D– and spherical dome), resulting in higher landing mass. Moderate investment to a Mach 3 parachute could also increase the landing mass by about 80%. Alternative means, such as ballutes for aeroassist maneuvers, should also be investigated. Landing accuracy with entry guidance and optical navigation is ~3 km. This precludes multiple landing missions with power sharing and Mars base architectures. Pinpoint accuracy landing studies target ~100 m distance between landed assets with a landing mass of ~1 MT. Manned Mars bases will require hundreds of kWe or even multi-MWe of power and several 100 MT of landed mass. To achieve this, significant improvements are needed to all technologies throughout the mission, including larger launch vehicles to increase deliverable mass, fast transfer for the crew, higher landing mass, pinpoint landing, and higher power provided by a high conversion efficiency nuclear reactor.

Obviously, advances in technology are needed to improve the nuclear reactors as well. At present, both in-space reactors (e.g., Topaz) and surface testbed reactors (e.g., HOMER, SAFE-30) are small but could be developed further and scaled up to higher power levels. Improving power conversion efficiency could also raise reactor power or reduce system mass. RTG trends target increased power conversion efficiency and reduced system mass with consequently lower radioactive fuel requirements.

Beside the upper bounds for nuclear fission power sources the lower bounds are defined by conventional power sources, such as solar and fuel cells and batteries. Small surface reactors become more mass efficient than solar panels around 3 to 5.5 kWe. Other advantages include continuous operation without significant degradation and without maintenance. While batteries and fuel cells can perform in the multi-kWe power level range, these technologies are not suitable for continuous long operation measured in months and years. Thus for these types of missions only nuclear reactors and RTGs are appropriate. Beamed power is still far from practicality and from finding its way into the space program.

In summary, based on the assumptions and today's technology limits, in-space nuclear fission reactors can provide ~100 to 200 kWe power for orbital missions at Low Mars Orbit and ~10 to 25 kWe power for surface-based missions on Mars. These limits are based on numerous assumptions. Consequently, the results and comparisons should be used only for initial mission feasibility assessments. At later mission design stages specific detailed analysis is required to finalize mass, power, and time requirements and to assess trade options.

The various breakpoints compiled here paint a picture of upper bounds for our technology capabilities for nuclear systems. One could certainly argue that there is a chance to break these limits by projecting performance potentials and possibilities for the future. It is the opinion of this author, however, that without significant financial commitment, political will, and a proven track record of rapid technology improvements or the introduction of new ground breaking technologies in any given field, the technology limits outlined in this paper remain, at least for the next decades of our Mars exploration program.

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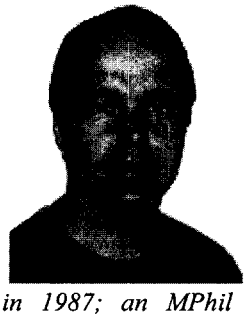

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