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Solar Versus Fission Surface Power for Mars

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A multi-discipline team of experts from the National Aeronautics and Space Administration (NASA) developed Mars surface power system point design solutions for two conceptual missions to Mars using In-situ resource utilization (ISRU). The primary goal of this study was to compare the relative merits of solar- versus fission-powered versions of each surface mission. First, the team compared three different solar-power options against a fission power system concept for a sub-scale, uncrewed demonstration mission. This "pathfinder" design utilized a 4.5 meter diameter lander. Its primary mission would be to demonstrate Mars entry, descent, and landing techniques. Once on the Martian surface, the lander's ISRU payload would demonstrate liquid oxygen propellant production from atmospheric resources. For the purpose of this exercise, location was assumed to be at the Martian equator. The three solar concepts considered included a system that only operated during daylight hours (at roughly half the daily propellant production rate of a round-the-clock fission design), a battery-augmented system that operated through the night (matching the fission concept's propellant production rate), and a system that operated only during daylight, but at a higher rate (again, matching the fission concept's propellant production rate). Including 30% mass growth allowance, total payload masses for the three solar concepts ranged from 1,128 to 2,425 kg, versus the 2,751 kg fission power scheme. However, solar power masses increase as landing sites are selected further from the equator, making landing site selection a key driver in the final power system decision. The team also noted that detailed reliability analysis should be performed on daytime-only solar power schemes to assess potential issues with frequent ISRU system on/off cycling.

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Next, the team developed a solar-powered point design solution for a conceptual four-crew, 500-day surface mission consisting of up to four landers per crewed expedition mission. Unlike the demonstration mission, a lengthy power outage due to the global dust storms that are known to occur on Mars could pose a safety hazard to a crewed mission. An earlier power trade study performed by NASA in 2007 concluded that fission power was more reliable—with a much lower mass penalty—than solar power for this application. However, recent advances in solar cell and energy storage technologies and changes in operational assumptions prompted NASA to revisit the analysis. For the purpose of this exercise a landing site at Jezero Crater, located at 18º north latitude, was assumed. A fission power system consisting of four 10-kW fission reactors was compared to a distributed network of Orion-derived solar arrays mounted on every lander, plus a complement of Lithium Sulfur batteries. The team found that a solar power generation and storage system mass of about 11,713 kg on the first crewed expedition would provide the 23 kilowatts (kW) needed to survive a dust storm lasting up to 120-days at average optical depth of 5; note that this may be insufficient to complete return vehicle propellant production. The arrival of 6,102 kg more solar power generation equipment on the second crewed expedition would match fission power's ability to operate at full power, even during a lengthy dust storm, but at nearly twice the cumulative landed mass of the 9,154 kg fission system.

Relative merits—and risks--of the two approaches were also evaluated. To assess latitude sensitivity, the team performed a brief assessment of a second notional landing site, Columbus Crater, located at 30º south. Because global dust storms appear in the Southern hemisphere during the summer, rather than the winter, when the days are shorter, solar array mass did not increase at Columbus Crater—but more battery mass would be required for the longer nights, and additional ISRU strings would be needed for optimal summer/winter cycling. Based on this work, it is clear that the choice between fission versus solar power for Mars surface systems is an important decision, with potential ramifications to crewed landing site selection.

Nomenclature

cg	=	center of gravity
K	=	Kelvin
kg	=	kilograms
kg/hr	=	kilograms per hour
km	=	kilometer
kWe	=	Kilowatt electric
m	=	meters
m^2	=	square meters
mR/hr	=	millirem per hour (radiation exposure rate)
rad	=	0.01 Joules per kilogram (absorbed radiation dose)
sol	=	solar day on Mars (24 hours 39.5 minutes)
τ	=	atmospheric optical depth
V	=	Volt
W-hr/kg	=	Watt-hours per kilogram

I. Introduction and Background

EARLY Mars mission concepts developed by the National Aeronautics and Space Administration (NASA) estimated that 40 kilowatts electric (kWe) power would be needed to support six crew living on Mars for about 500 days.[1] To minimize mass, the crew's return vehicle would land on Mars with empty oxygen propellant tanks and a manufacturing plant would produce Liquid Oxygen (LOX) propellant from the Martian atmosphere. Analysts traded solar versus nuclear power, and settled on a fission system, primarily due to the possibility of global dust storms that threatened solar power's reliability. Initial studies focused on a monolithic, 40-kWe Fission Surface Power System (FSPS) concept[2] that was being considered by the Constellation Program for lunar applications. The FSPS would be deployed about one kilometer (km) from the eventual habitat location and used to power an In Situ Resource Utilization (ISRU) propellant plant during the cargo mission phase; once the return vehicle's propellant

tanks were full, the crew would arrive and switch the FSPS over to habitat and science support during the crewed mission phase. Later work[3] demonstrated advantages—including mass savings—of breaking the 40-kWe FSPS into multiple, smaller power sources by leveraging a joint NASA-Department of Energy (DoE)/Los Alamos National Laboratory (LANL) research effort called the Kilopower fission system concept.[4] The lower mass Kilopower units were assumed to make fission power even more compelling than solar power for this application.

Before initiating the expensive process to develop and flight certify a fission system, NASA's Evolvable Mars Campaign (EMC) managers thought it prudent to refresh the solar versus fission power trade study to include recent technical advances in solar panel efficiency and energy storage density, as well as updates to mission operational concepts. This task was assigned to the Collaborative Modelling for Parametric Assessment of Space Systems (COMPASS) team at NASA's John H. Glenn Research Center.

II. Study Approach

The study was broken into two individual tasks. First, the COMPASS team traded solar power against fission power for an uncrewed, subscale ISRU demonstration mission. The primary objective of this mission would be to test Mars entry, descent, and landing technologies but with a secondary goal of demonstrating that cryogenic propellants can be manufactured from Martian resources within the prescribed mission timeline. The oxygen production rate for this demonstration was specified to be 1/5 of the estimated production need for a human mission. Next, the COMPASS team revisited the power trade study for a crewed mission, using EMC mission assumptions[5] and state of the art technologies.

In the first part of the study, power outages due to dust storms were factored in, but spacecraft power could be reduced without risk; in the second part of the study, the system had to maintain minimum crew keep-alive power regardless of dust storm severity or duration.

III. ISRU Demonstrator Analysis

A. ISRU Demonstrator Task Approach

The first challenge the study team encountered was how to perform an "apples to apples" comparison between the solar and fission powered concepts, given that the solar option would be sensitive to day/night cycling and dust storms whereas the fission option would operate around the clock without weather-related disruption. The objective of the study was to compare fission and solar power systems for the same rate of production so the team decided to evaluate three different solar power cases: Option 1A assumed daytime-only solar-powered production, for approximately 10 hours of production per day, at the 1/5 scale production rate of 0.45 kg/hr; Option 1B assumed the same production rate, but with around-the-clock production; and Option 1C assumed day-time only production, but at twice the production rate of Options 1A or 1B. These three solar options were then compared against Option 2, a 1/5 production rate fission system operating around the clock. The team selected five Figures of Merit for option comparison:

1) Performance, as measured by propellant production rate in kilograms per hour (kg/hr).
2) System mass.
3) System robustness to the Martian environment, particularly dust.
4) System design, development, test, and evaluation cost.
5) System operational life.

B. ISRU Demonstrator Task Assumptions

1. Launch and Landing

The COMPASS team assumed the demonstrator payload would launch on a Delta IV Heavy, sized to fit within a five meter diameter shroud, with a total payload launched mass of 7,500 kilograms. A seven meter diameter Adaptable Deployable Entry and Placement Technology (ADEPT) entry system was assumed[6]. Subtracting estimated entry, descent and landing systems mass based on recent Evolvable Mars Campaign analysis left about 3,500 kg available for the ISRU demonstration experiment, including power systems. The demonstrator spacecraft was assumed to manufacture propellant from the carbon-dioxide atmosphere, rather than soil; therefore the mission could be conducted anywhere on the planet. In lieu of defined mission details, the team assumed the Mars Exploration Rover (MER) Program's Opportunity rover landing site at Meridiani, roughly 2^{o} south, as this had the benefit of Opportunity's 12 years of actual measured solar array performance data, as well as favorable night durations and minimal seasonal variations throughout the year.

2. Martian Environment

To size thermal systems, the study team used MER data from the Opportunity landing site. Surface temperatures were assumed to vary from a night time low of 174 K to a daytime high of 308 K; sky temperatures were assumed to swing from 139 K to 213 K.

Dust storms on Mars include both local dust storms and occasional global storms. While the local storms last at most a few days and have only a modest increase in optical depth, a global storm can last for months, with a significant increase in optical depth[7]. Such global storms seem to occur, on average, about every three Martian years, and are more prevalent during the southern hemisphere spring and summer. For this study, a dust storm of 120 days in duration was assumed, also based on data collected from various Mars observation programs[8]. For the purpose of this study, optical depth (τ, the logarithm of the ratio of incident to transmitted radiant power on a vertical path through the atmosphere) was assumed to vary from 1.0 under clear skies to a worst-case value of 5.0 during a dust storm, based on solar-energy measurements taken by the Spirit and Opportunity rovers. Maximum wind speeds up to 20 meters per second were assumed. Figure 1[9] illustrates the visible difference of various optical depths, as viewed by the Opportunity rover. While it may appear that almost no sunlight was reaching the rover at the peak of the storm, only direct sunlight was blocked. The dust caused light to be scattered, such that at the worst of the storm the diffuse light was approximately 30-40% the amount of direct light on a clear day. It was assumed that, on average, the arrays would see 12 hours of sunlight per sol.

Figure 1. Dust storm time lapse photo taken by Opportunity.

3. ISRU System

Preliminary analysis indicated that an ISRU system sized to produce 0.45 kg/hr could be readily developed, effectively providing a 1/5 scale demonstrator for the 2.2 kg/hr full-scale crewed propellant production needs. ISRU system design and optimization was out of scope for this exercise, so the study team assumed a representative ISRU system[10,11] (Figure 2), consisting of a carbon dioxide collection and freezing subsystem, solid oxide electrolysis processor and recirculation, oxygen liquefaction and storage, power storage and conditioning equipment, and a thermal management subsystem.

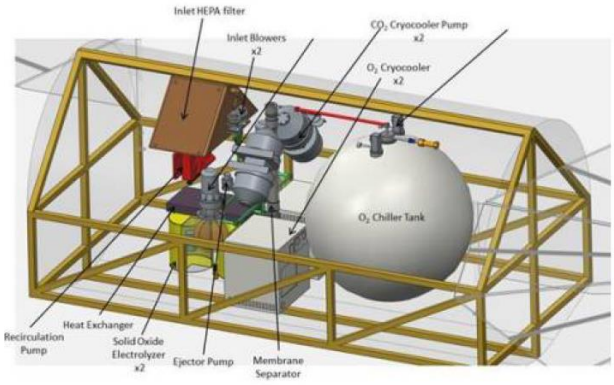

Figure 2. Representative ISRU System.

For the purpose of this exercise, it was assumed the payload must demonstrate liquefaction and storage of oxygen at cryogenic temperatures. A storage system capable of holding 1,500 kg of LOX was assumed for all demonstrator options. Spacecraft systems were assumed to be single-fault tolerant where appropriate.

4. Solar Arrays

For all solar-powered demonstrator concepts, the study team assumed a derivative of the 120 V Orbital ATK UltraFlex™ arrays or equivalent, populated with Inverted Metamorphic Multi-junction (IMM) solar cells of 33% conversion efficiency (measured at Earth distance solar flux, 28°C, beginning of life). Array and gimbal concepts originally developed for the Orion spacecraft's 2.7 standard gravity (g) Trans Lunar Injection structural loads were assumed to be sufficient for Martian gravity and surface wind loads. Arrays were sized based on average solar distance of 1.52 astronomical units, while acknowledging that power could vary as much as +/- 20% depending on actual distance from the Sun during the Martian year. Arrays were sized to meet the daytime ISRU production and housekeeping power requirements plus the power needed to recharge energy storage for nighttime housekeeping power. For worst-case dust storms and resulting incident energy reduction to 35% of the baseline design, solar arrays were sized to provide more than adequate power to meet minimum loads. A permanent dust power loss

factor of 0.95 was assumed, given that the two-axis gimbal articulation allowed arrays to periodically rotate to remove most dust accumulation.

5. Kilopower System

The study team assumed a full-scale, 10-kWe Kilopower system for the fission-powered demonstrator case. Although this turned out to be slightly oversized for the demonstration mission, it had the advantage of providing test data for an eventual crewed mission, without the time or expense of developing a single-use sub-scale unit. A stowed 10-kWe Kilopower system measures approximately 4 m tall by 1.5 m diameter; with the 20 square meter (m^2) radiator panel deployed, the diameter expands to approximately 5 m, as shown in Figure 3. However, for the purpose of this exercise, a simplified unit with a fixed, conical radiator was used. System mass, including 15% growth allowance, was estimated at 1,754 kg. Note that nearly a third of the Kilopower's mass was dedicated to a shield sized to reduce crew radiation exposure to less than 3 millirem per hour (mR/hr) within 500 meters. As noted, the Kilopower concept is a joint NASA/DoE technology development, and was not developed by this study team.

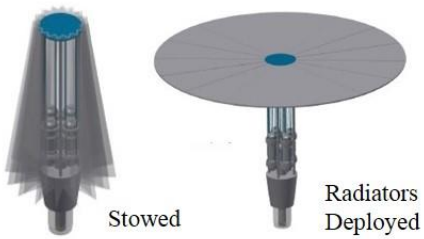

Figure 3. Kilopower Concept.

C. ISRU Demonstrator Concept of Operations

The concept of operations for all four options begins the same: launch on a Delta IV Heavy or equivalent and land at the Opportunity site, 2° south of the Martian equator, in late 2025. The Kilopower option would perform system checkouts for approximately two weeks and then initiate propellant production; the three solar-powered options would first deploy solar arrays before performing system checkouts and beginning propellant production. The solar-powered options would gimbal their solar arrays to follow the sun, and periodically articulate the arrays +/- 45° (Figure 4) to clear accumulated dust. Note that there is a practical limitation on solar array diameter, to prevent ground contact, or contact between multiple arrays, during gimbaling.

In the event of a dust storm, the three solar powered options would pause propellant production and use stored energy to maintain accumulated propellant at cryogenic conditions. For all options, production would continue until 4,400 kg of LOX was produced (1/5 of the estimated crewed mission need). However, only 1,500 kg of LOX would be stored and conditioned in the accumulator; the balance would be vented overboard.

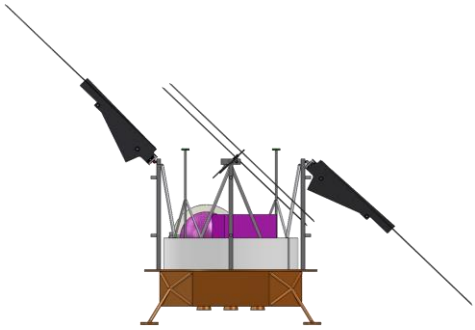

Figure 4. Gimbaled solar arrays.

D. ISRU Demonstrator Conceptual Design

1. ISRU production plant

The basic ISRU plant was the same for each of the options studied. As noted above, the study team leveraged work previously performed by NASA to estimate ISRU production plant masses, with one exception: the cryopump used for atmospheric carbon dioxide collection was replaced with a scroll pump to reduce both radiator mass and volume. This change brought the ISRU subsystem mass down by almost 50 kg.

The Command and Data Handling subsystem was comprised of a Maxwell Technologies SCS750® space-rated computer, an Aitech 710 radiation tolerant communications card, an enclosure and power supply, Ethernet Local Area Network (LAN) card, solar array gimbal controller, memory storage unit, sensors, instrumentation, and wiring.

The Communications and Tracking subsystem consisted of two external Ultra High Frequency (UHF) band omnidirectional antennas and an internal UHF L3 proximity link transceiver for communications to an orbiting relay station. A nominal data rate of 100 kilobits per second (kbps) was assumed, with communications occurring for about an 8 minute window per each sol.

The thermal control system was dominated by a pumped loop cooling system and radiators to remove heat from the oxygen liquefaction cryocooler. To maintain internal electronic components within an operating temperature range of 233 K to 323 K, electronics boxes were protected with Aerogel foam insulation, thermal paint was applied to exposed surfaces, and radiators were used for cooling internal electronics.

Finally, a 7075-T73 Aluminum and composite sandwich primary structure provided the backbone for the conceptual design (Figure 5). The aluminum and carbon fiber-reinforced cyanate ester matrix structure was designed to carry axial launch loads as high as 6.5 g axial and 0.5 g lateral, lateral launch loads up to 2.5 g axial and 2.0 g lateral, and up to 12 g axial loads during landing parachute deployment. The estimated structures mass could be optimized with a higher fidelity stress analysis and the use of more advanced composite materials, orthogrid, or isogid panels.

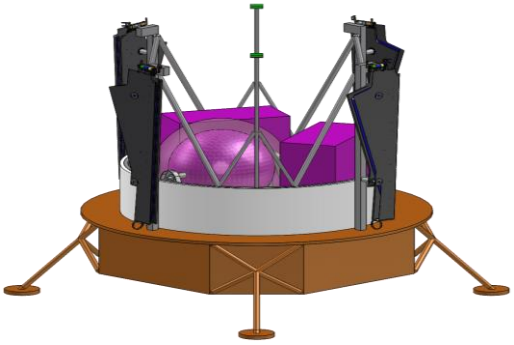

Figure 5. ISRU Demonstrator Option 1A, shown with solar arrays stowed for landing.

2. Solar Option 1A: Daylight-Only, 1/5 Rate Production

The study team estimated that the spacecraft required about 0.22 kW power for post-landing system checkout and array deployment, but the load jumped to 6.45 kW once propellant production began. Night time standby mode was 0.69 kW. The study team calculated that four each 5.6 m diameter UltraFlex™ arrays or equivalent, estimated at 30 kg each, would provide the power needed to produce 4,400 kg of LOX for an ISRU system operating 10 hours per sol (assuming 2 hours per sol of sunlight lost to system warm-up). Lithium-ion batteries would provide standby power for communications and thermal control while the ISRU system cycled off through the night. Including the possibility of pausing production during a dust storm of up to 120 sol duration, propellant production would require about 1,098 sols. Based on a 6 m diameter landed footprint, the maximum center of gravity (cg) height was approximately 2.61 m for this concept. Overall height was 4.61 m.

A breakdown of the 1,128 kg total payload mass, including growth allowances[12], is shown in Table 1.

Table 1. Demonstrator Option 1A manifested equipment list.

Description	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
ISRU Payload	163	18	29	192
Command and Data Handling	45	16	7	52
Communications and Tracking	6	17	1	7
Electrical Power Subsystem	351	30	104	455
Thermal Control (Non-Propellant)	140	18	25	165
Structures and Mechanisms	155	18	28	183
Mars ISRU Demonstrator Sub-Total	**860**	**23**	**194**	**1054**
System-Level Growth		7	74	74
Mars ISRU Demonstrator	**1054**	**25**	**268**	**1128**

In addition to the solar arrays, the power subsystem included 60% depth of discharge, Panasonic cell type Lithium-ion batteries, with an energy storage density of 165 Watt-hours per kilogram (W-hr/kg). A power converter derived from International Space Station heritage hardware would convert the 120 V power provided by the solar arrays to 28 V for other spacecraft loads. A 120 V solar array regulator based on a commercial proprietary design was sized to handle the peak power loads expected at the Mars surface and optimize the power obtained from the solar array. 28 V power distribution cards provide switching to spacecraft loads and 28 V battery charge-discharge cards provides battery regulation and protection. Estimated masses for a 28 V battery controller and power distribution system were based on commercial TERMA hardware. Wire harness mass was sized at 13% of power system mass.

The final spacecraft landed footprint was approximately 6.0 m. With all four arrays deployed the maximum diameter was 15.6 m and 4.81 m tall (with arrays not gimbaled).

3. Solar Option 1B: Around-The-Clock, 1/5 Rate Production

Switching to around-the-clock ISRU production cut the mission time in half, but required larger solar arrays and more energy storage, causing the electrical power subsystem alone to jump more than 1,200 kg in mass as compared to the 1A daylight-only option. Each of the four 7.5 m diameter solar arrays needed for this option was more than twice the mass of the 5.6 m diameter arrays specified for Option 1A, but the bigger impact was battery mass to continue ISRU production through the night: over 1,100 kg more batteries for Option 1B, as compared to Option 1A. This option's ISRU plant, command and data handling, communications and tracking, thermal control, and structures subsystems were all identical to the 1A Option but electrical subsystem changes resulted in a 2,425 kg

total payload mass, more than double the 1A option. Figure 6 shows how much larger the 1B spacecraft footprint would be with the larger solar arrays, as compared to the 1A option.

4. Solar Option 1C: Daylight Only, 2/5 Rate Production

The third solar-powered ISRU Demonstrator option utilized the larger 7.5 m diameter solar arrays to power an ISRU system twice the capacity of the other two options, but only during the approximately 10 hours per sol of daylight. This scheme caused peak power to jump to 12.43 kW, nearly double that of the 1A option. At 1,531 kg, payload total mass fell between the 1A and 1B options. The command and data handling, communications and tracking, and thermal control subsystems were all identical to the other two solar powered payload concepts, but this option carried 143 kg more ISRU mass and 37 kg more structural mass than the other two options.

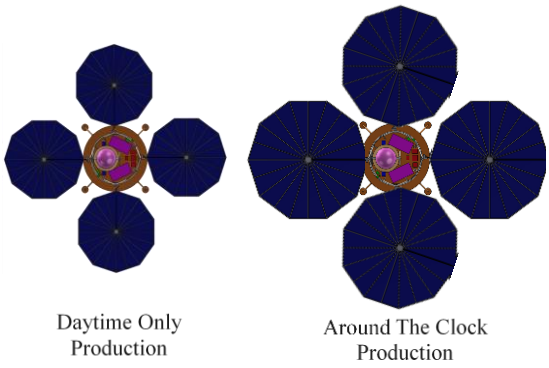

Figure 6. Solar array size comparison between 1A daytime-only (left) and 1B around-the-clock ISRU (right) production.

5. Fission Option 2: Around-The-Clock, 1/5 Rate Production

With a fixed, conical upper radiator, the fission-powered option (Figure 7) required no post-landing mechanism deployment and needed only 106 W keep-live power after landing. Peak power draw of 6.52 kW is achieved during night-time operations, with a peak daytime power need of 6.45 kW. Based on a 6 m diameter landed footprint, the maximum cg height was the same as the solar options at 2.61 m, but overall landed height was slightly higher, at 5.14 m.

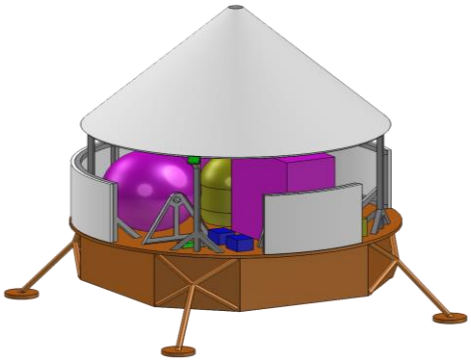

Figure 7. Fission-powered ISRU demonstrator concept.

At 2,751 kg total payload mass (including growth allowance), this option is considerably larger than the 1A or 1C solar power options, but comparable to the 1B option. As with the 1B option, the higher mass is almost entirely due to the electrical power system's 1,804 kg allocation, which is dominated by a 1,754 kg, 10-kWe Kilopower unit. The command and data handling, communications and tracking, and thermal control subsystems were all identical to the solar powered payload concepts; the thermal control system was actually lower mass than the solar options.

E. ISRU Demonstrator Solar versus Fission Comparison

Table 2 compares key parameters of the four demonstrator mission options.

1. Performance Comparison

The closest "apples to apples" comparison with fission is the 1B solar around-the-clock option, as both are producing propellant at the same rate (though it could take the solar option longer to do so given the probability of a dust storm). A mission using the 1A concept would take twice as long as the other missions.

2. Mass Comparison

Options 1B and 2 are roughly comparable in mass *for an equatorial landing*, though the solar concept mass would quickly exceed the fission option mass as the landing site shifts further from the equator. All four options fit comfortably within the 3,500 kg landed payload allocation. The 1A and 1C options could both accommodate two metric tons of additional payload to the surface. All four options would provide flight experience for the ISRU system but Option 2 would also provide flight experience for a full-scale Kilopower unit. Unfortunately, this handicaps the fission option in a mass trade against the solar options because a full-scale Kilopower is about 30% oversized for this particular mission. What's more, a baseline Kilopower unit includes sufficient radiation shield mass to protect crew within one kilometer, which is not necessary for this uncrewed mission.

Table 2. Demonstrator Options Comparison.

Option	Solar 1A: 1/5 rate Daytime Only	Solar 1B: 1/5 rate Around the Clock	Solar 1C: 2/5 Rate Daytime Only	Fission 2: 1/5 Rate Around the Clock Fission Power
Total Payload Mass (including growth)	1,128 kg	2,425 kg	1,531 kg	2,751 kg
Electrical Subsystem Mass	455 kg	1,733 kg	639 kg	1,804 kg
ISRU Subsystem Mass	192 kg	192 kg	335 kg	192 kg
Power	~8 kW Daylight	~8 kW Continuous (with 16 kW of arrays)	~16 kW Daylight	~7 kW Continuous
Solar Arrays	4 each x 5.6 m diameter	4 each x 7.5 m dia.	4 each x 7.5 m diameter	None
Night Production?	No	Yes	No	Yes
LOX Production	4.5 kg/sol	10.8 kg/sol	9.0 kg/sol	10.8 kg/sol
Time to Produce 4,400 kg LOX, including 120-Day Dust Storm Outage	1,098 sols	527 sols	609 sols	407 sols
ISRU On/Off Cycles	1,098	<5	609	<5
Radiation Tolerance	100 kilorad (krad) electronics and ISRU			300 krad electronics, 10 Mega ad (Mrad) ISRU

3. System Robustness

Dust is a major concern for Martian surface operations. Solar array deployment and gimbal mechanisms could be compromised by fine grit, leading to reduced performance for options 1A, 1B or 1C. Similarly, dust accumulation on the conical radiator could lead to reduced performance for option 2. The study team included solar array gimbals to remove dust, based on experience with angled surfaces on NASA's previous Mars rover programs. ISRU system tolerance to repeated cycling for the 1A and 1C options is unknown. In all cases, ISRU system tolerance to fine dust ingestion would have to be understood and potentially mitigated.

4. Cost Comparison

A rough order of magnitude cost comparison was assembled for solar option 1B and fission option 2 (Table 3). Estimates include only Phase B/C/D[13] lander costs, and do *not* include launch services, National Environmental Protection Agency and Nuclear Safety Launch Approval (where applicable), mission-level costs, the ISRU payload and ISRU integration cost, fuel costs, science and science instruments, or technology development up to Technology Readiness Level (TRL) 6. These estimates also ground rule out technology development costs for the fission power system and solar arrays to allow for a better comparison, as these may be carried under a separate program. Estimates assume protoflight development and the effort is contracted to a major aerospace firm. Quantitative risk analysis was performed using a Monte Carlo simulation driven by input parameter uncertainty and the error statistics of the cost estimating relationships. Costs presented are mode values (approximately 35th percentile), in constant United States (U.S.) Fiscal Year 2016 dollars (FY16$).

5. Operational Life Comparison

As noted, a mission using the 1A concept would take twice as long as the other missions, with option 2 having the shortest mission timeline. Extended life analysis was beyond the scope of this study but, in theory, all concepts could remain functional for years after the demonstrator mission is complete. For eventual crewed mission use, the Kilopower unit would be rated for at least 12 Earth years' service life. Solar arrays could be expected to have graceful degradation, allowing reduced performance beyond the planned mission with lost strings or even a lost wing. Battery performance would decrease over time. Long-term performance of electronics and gimbals are unknown.

Table 3. Solar Option 1B versus Fission Option 2 Flight Hardware Cost Comparison (Millions of Dollars).

Work Breakdown Structure/Description		Case 1B (FY16$M)			Case 2 (FY16$M)			Total Delta FY16$M
		DDT &E	Flight Hardware	Total	DDT &E	Flight Hardware	Total	
06.1.3	Command &Data Handling	26.6	9.4	36.0	26.3	9.4	35.7	0.3
06.1.4	Communications and Tracking	3.5	1.6	5.1	3.5	1.6	5.1	0.0
06.1.5	*Electrical Power Subsystem	36.2	80.9	117.1	23.6	175.7	199.3	-82.2
06.1.6	Thermal Control (Non-Propellant)	10.0	6.4	16.4	8.4	5.3	13.7	2.7
06.1.11	Structures and Mechanisms	7.8	10.2	17.9	8.0	11.1	19.1	-1.2
Subsystem Total		**84.1**	**108.5**	**192.6**	**69.8**	**203.1**	**272.9**	**-80.3**
Systems Integration								
Integration, Assembly,Check-out /Systems Test Operations		11.3	6.8	18.1	9.7	9.9	19.6	-1.5
Ground Support Hardware		14.2		14.2	11.8		11.8	2.4
System Engineering/Integration		14.5	22.6	37.1	12.5	33.0	45.6	-8.5
Project Management		10.6	13.7	24.4	9.5	20.2	29.8	-5.4
Launch and Orbital Operations Support		11.8		11.8	9.8		9.8	2.0
Spacecraft Total		**146.5**	**151.7**	**298.1**	**123.1**	**266.3**	**389.3**	**-91.2**
Fee (10%)				29.8			38.9	-9.1
Prime Total				**328.0**			**428.3**	**-100.3**

**for comparison purposes, solar array and fission power system technology development costs have been omitted.*

IV. Crewed Surface Mission Assessment

A. Crewed Mission Task Approach

The study team leveraged demonstrator mission results for the crewed mission task, and applied the EMC concept of operations. Solar- versus fission-powered concepts were compared at two hypothetical landing sites, at different distances from the equator. Although all options were sized to produce crew return propellant, that portion of the mission could more easily tolerate dust storm disruption because it would occur before the crew arrived. The more critical power need was after crew arrival, when at least 14.8 kW of crew and equipment keep-alive power were needed to continue the surface mission and maintain return vehicle propellant conditioning.

B. Crewed Mission Task Assumptions

1. Launch and Landing

The study team assumed that crewed mission elements would launch on NASA's Space Launch System, sized to fit within a 10 m diameter payload shroud. To understand solar power mass sensitivity with respect to latitude the study team selected two hypothetical landing sites (indicated by stars in Figure 8): Jezero Crater, located at 18.9° North, 77.5° East, and Columbus Crater, located 29.5° South, 166.1° West. Jezero Crater was an early candidate during the Mars Science Laboratory landing site selection process and is among the sites being considered for the upcoming Mars 2020 mission[14,15]; Columbus Crater provided a convenient data point at a mid-latitude in the southern hemisphere.

2. Surface Mission

Per EMC working assumptions, surface infrastructure (including power systems) would be re-used by subsequent crews at a single landing site. The first three cargo landers would be pre-positioned before the first expedition crew's arrival on the fourth lander, after their Mars Ascent Vehicle's (MAV) ISRU propellant tanks were confirmed full. Subsequent expeditions would require only two precursor landers—one for cargo and one for their ascent vehicle—plus a crew lander, making a total of ten landers for the first three crewed expeditions during the first decade on Mars. Each expedition crew would spend approximately 500 days on the surface. High voltage

transmission cables laid on the surface would carry power from point of origin to the various landers. Crew would live primarily in a surface habitat, with frequent excursions in a pressurized rover to explore surrounding areas. During the crew's stay, the surface power system would continue to condition cryogenic propellants and maintain system health and status checks on the MAV.

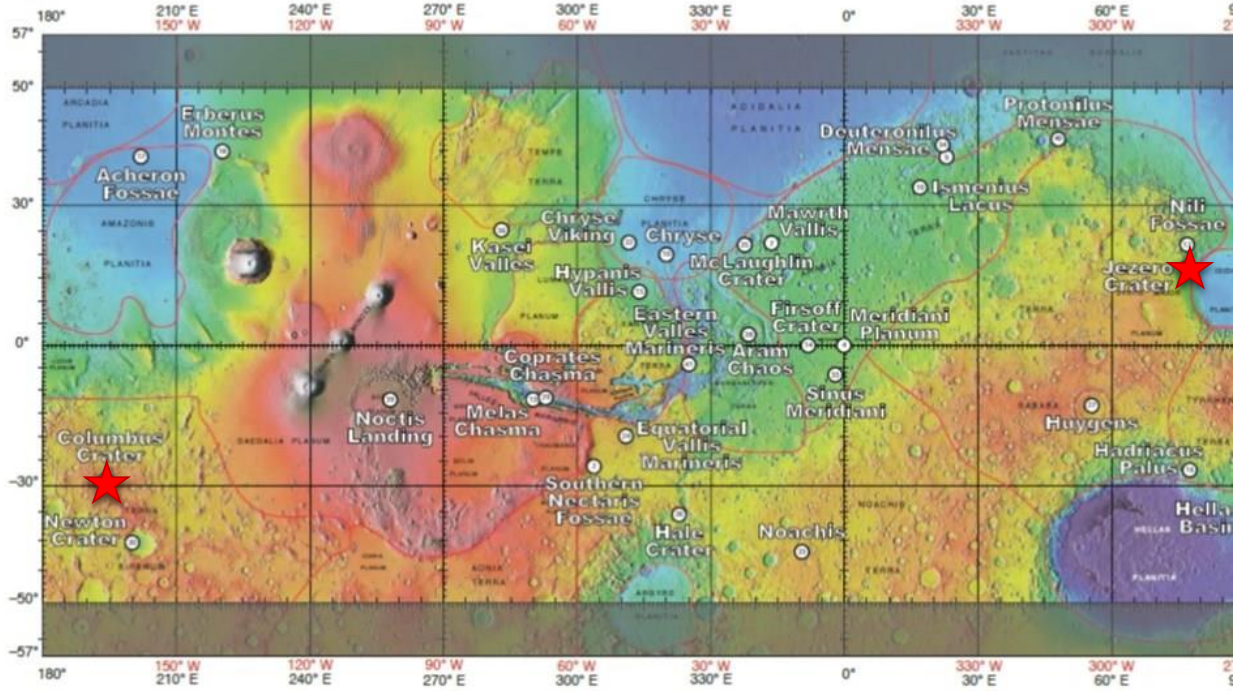

Figure 8. Hypothetical crewed mission landing sites on Mars.

3. Martian Environment

As with the demonstrator mission, at least one dust storm of 120 days in duration was assumed, but seasonal effects were also considered. Sites in the northern hemisphere are closer to the Sun in the winter—when dust storms are more prevalent in the north—so distance from the Sun counterbalances day/night cycle effects, effectively maintaining an energy balance similar to an equatorial site. Dust storms in the southern hemisphere are more prevalent in the summer, when days are longer and the site is closer to the Sun, so distance from the Sun does not counterbalance day/night cycle effects. To take advantage of this, the study team assumed higher ISRU rates during summer for the solar-powered option at the Columbus Crater site, as this eliminated the need to add more solar arrays for the short day/far Sun part of year.

For the Jezero Crater site, the study team assumed night periods up to 13.3 hours, daylight periods as short as 11.4 hours, nominal optical depth of 1.0, and dust storm optical depth of 5.0, or 35% of nominal incident energy. For the Columbus Crater site, the study team assumed night periods as long as 14.5 hours, days as short as 10.2 hours, nominal optical depth of 0.8 during propellant production and 1.0 during the crewed mission, and dust storm optical depth of 5.0.

4. Surface Systems

For the purpose of this study, the team assumed that during the cargo phase the atmospheric ISRU system must produce 22,728 kg of LOX within 14 Earth months (or 420 days), for a minimum production rate of 2.2 kg per hour (not accounting for outages). Note that this may be a conservative estimate for some mission scenarios. For a hypothetical launch in May, 2038 of the first lander, the ISRU system could have as long as 725 to 750 Earth days to manufacture the first crew's return propellant. Production schedule for subsequent expeditions varies, but some expedition timelines could be as short as only 430 Earth days. This analysis only considered atmospheric ISRU; more power would be needed for regolith excavation and processing to produce soil ISRU.

Once crew arrived, LOX would continue to be thermally conditioned in the MAV propellant tanks, along with the methane transported from Earth. Crew would live in a separate surface habitat, from which most other

equipment—such as space suit batteries or robotic rovers—would be recharged. An optional science laboratory would be available for sample processing, analysis, and return preparation

As shown in Table 3, estimated peak surface power need during the ISRU-production cargo phase for this assumed complement of equipment is about 26.3 kW, but could increase to 31 kW during the crewed phase. Note that the peak crewed power load assumes all systems operating, including an optional science facility, but with careful phasing the crewed phase peak power could be lowered to match the cargo phase. During the cargo phase, peak and keep-alive power are the same, since propellant must still be produced. During the crewed phase, life support and MAV propellant conditioning take priority but it was assumed that non-essential functions (laundry or science experiments, for example) could be suspended to minimize keep-alive power. Also note that these values are based on conceptual element designs, and may increase or decrease as mission requirements are defined.

Table 3. Point of Use Crewed Mission Power Needs.

	Peak Power Needed (W)		Keep-Alive Power Needed (W)	
Element	**Cargo Phase**	**Crewed Phase**	**Cargo Phase**	**Crewed Phase**
ISRU	19,700	0	19,700	0
Mars Ascent Vehicle	6,655	6,655	6,655	6,655
Surface Habitat	0	14,900	0	8,000
*Science Laboratory	0	9,544	0	174
Total	**26,355**	**31,099**	**26,355**	**14,829**

**Optional element shown with all systems running. Assume power can be phased to stay below cargo operations peak.*

5. Kilopower System

The study team assumed the same 10-kWe Kilopower system as the demonstrator mission, but assumed the EMC baseline of four units plus one spare, all arriving on the first lander. A power distribution system connected the power farm to subsequent landers. Kilopower mass used in this analysis assumes a one kilometer safe separation distance between the power farm and the crew habitat. Note that Kilopower mass could be reduced by reducing shield mass, but this would increase separation distance (and therefore distribution cable mass and line losses).

6. Solar Arrays

For the solar powered concepts, the study team took a novel approach by assuming solar arrays on *every* lander, rather than concentrated on a single power farm, as was assumed for the fission case. Power distribution cables made each lander a node in a large power network, with batteries providing overnight power. *Note that dust storm power was provided by excess solar array area, not energy storage.* Given enough landers, the accumulated array area could collect enough energy even in a dust storm to meet peak demand, without having to reduce loads to "keep alive" power. Although the first few landers would be at higher risk for dust storm disruption, one appealing advantage was that power generation capability on every lander made most landers self-sufficient (at least in keep-alive mode) immediately after landing, reducing the time pressure to autonomously connect to the power farm. This scheme also made higher latitude sites more accessible for solar power, assuming multiple expeditions contribute to the network.

As with the demonstrator concepts, the study team assumed crewed mission solar panels were a derivative of the 120 V Orbital ATK UltraFlex™ arrays, with two-axis gimbals to remove dust. Solar power generation assumed a daily average flux and worst day of year conditions, rather than an hour-by-hour or day-by-day simulation. A Roll-Out Solar Array (ROSA) option was examined, but the design was not sufficiently mature for the large size, wind loading, and Mars gravity. A ROSA laid flat on the surface would incur mass penalty for additional array area (since it could not track the sun), and would be at significant risk of performance degradation due to dust accumulation. Additional structure could raise the arrays up, but with more mass penalty. It simply made more sense to repurpose the lander structure as a permanent solar array platform, particularly since most landers served no other purpose once cargo was off-loaded.

Because each lander was assumed to carry at least one sol worth of internal batteries for on-board keep-alive power, it was assumed that these batteries could be recharged via the solar arrays during the day to provide night-time keep-alive power at each lander. Given the longer technology development window for a crewed mission versus the demonstrator mission, the study team assumed higher energy density 120 V, 270 W-hr/kg Lithium Sulfur batteries for energy storage. The entire complement of batteries required for full night time power was assumed to be manifested on the first lander.

7. Power Management and Distribution

For the purpose of this study, it was assumed that the Power Management and Distribution (PMAD) systems were more or less the same regardless of how power was generated, with the main difference being the number of

transmission cables accumulated over time and their associated power conversion electronics. Power Management differences were accounted for in the mass estimates. The solar power system requires solar power and battery regulators along with power converters and distribution electronics for the lander housekeeping and ISRU loads. Because each solar-powered lander becomes a node in the larger network, at least one cable is required for every lander, even after all payloads have been unloaded. In the fission power scheme, transmission cables could be repurposed for new landers once all payloads are off-loaded from an older lander and the lander itself is decommissioned. To estimate rough order of magnitude masses, the study team assumed 1,000 VDC transmission, stepped down to 120 VDC at the point of use. It was assumed that transmission cables would be laid on the ground, rather than buried.

C. Crewed Mission Concept of Operations

1. Fission Powered Concept

The concept of operations for all crewed options begins the same: launch the first of three cargo landers on an SLS approximately two years before the first Mars expedition crew arrives on the fourth lander. In the fission-powered scheme, all Kilopower units would arrive on the first uncrewed cargo lander. The Kilopowers could be pre-connected together, and operate from the lander on which they arrived, or offloaded and repositioned (Figure 9). If off-loaded, the study team assumed the lander's one sol, on-board, internal energy storage was sufficient to operate a cargo off-loading system. For the assumed shield mass, Kilopower units must be at least 1 km from continuously occupied crew assets, such as the habitat. Subsequent landers would have to provide internal keep-alive power to cargo for up to one sol while power distribution cables were robotically laid to the Kilopower farm. Some landers, would eventually be decommissioned once all cargo was unloaded and at that point the power distribution cable could be repurposed for the next lander, mitigating the need to bring a new cable for each subsequent lander. The Kilopower units would provide around-the-clock continuous power, regardless of landing site latitude, season, or dust storm prevalence. In the event of a Kilopower failure, the spare unit would be activated. After approximately 12 years (the equivalent of about three crewed expeditions), the Kilopower units would reach end of life and be replaced.

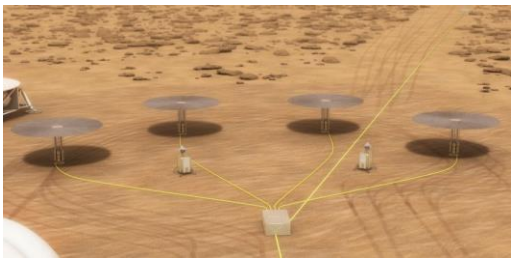

Figure 9. Kilopower deployment.

2. Solar Powered Concept

Upon landing, each lander would deploy up to four solar arrays to provide keep-alive power to on-board cargo. As in the fission concept, power distribution cables would be robotically laid between landers. Every lander must carry a distribution cable and voltage converters, because every lander is a permanent node of the power network, even if there is no powered cargo on that particular lander, or if all powered cargo has been off-loaded. This means that, unlike the fission system, cables and converters from a "retired" lander cannot be re-used for subsequent landers to save mass.

In the event of a dust storm during the first expedition's cargo phase, battery recharging would take priority during the day, with propellant production reduced or suspended. During the crewed phase, non-essential services could be metered to reduce power draw while operating on reduced solar collection. By the end of the second crewed expedition, the larger solar array network could theoretically collect enough energy during a dust storm to continue nominal operations without disruption.

D. Crewed Mission Lander Conceptual Design

1. Power Management and Distribution System

Detailed PMAD system design remains forward work, and final mass will depend on the operating voltage for the landers and their payloads, number of landers, distance between each lander, how robust cable shielding must be, and the transmission voltage selected. Because some equipment, such as a power distribution unit, would be part of a lander's flight design anyway, that function may be available to the surface power system at no additional mass penalty. For this reason, estimated PMAD masses are addressed separately from the surface power generation system mass.

The study team assumed 120 VDC to the main load, with a 28 VDC nominal bus for lander housekeeping loads. A 1000 V – 120 V DC-DC converter would be carried by surface rover from one lander to the subsequent lander to link the two together, and a 120V – 1000V DC-DC converter and power distribution unit would be located on each subsequent lander. Table 4 envelopes the typical PMAD mass that might be expected on each solar powered lander,

given the caveats outlined above. Note that cable mass shown here is quite conservative, and could be lower depending on insulation and abrasion protection selected. Because the Kilopower design includes a Stirling controller/AC-DC converter, a separate 120 VDC to 1000 VDC converter would not be necessary for the fission system's transmission line, but a smaller converter would be required at each lander to step power down to 120 VDC for any powered payloads that remain permanently onboard the lander.

Table 4. Representative Solar Power Management and Distribution (PMAD) equipment list.

Description	Quantity	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
1 km cable + spool + connectors	1	336	336	55	184.8	521
120 VDC to 1000 VDC Converter	1	176	176	30	53	229
1000 VDC to 120 VDC Converter	1	172	172	30	51.6	224
120 V Power Distribution Unit	1	50	50	30	15	65
Total						**1,038**

2. Crewed Mission Fission-Powered Concept

The study team found that four each 10-kWe Kilopower units would provide up to 35 kWe continuous power for all mission phases, at either hypothetical landing site. The first lander would carry the four primary Kilopower units plus one spare, totaling 8,769 kg (including 15% mass growth allowance). Alternating current (AC) output from Kilopower Stirling converters would be transmitted at 1,000 VDC to either the first lander (if the Kilopower units were relocated) or the second lander (if the Kilopower units remained on board the first lander). With 30% mass growth allowance, Stirling cables and controllers added 385 kg to the first lander, for a total fission power generation mass of 9,154 kg through the first crewed expedition (Table 5), not including lander-to-lander PMAD. Thermal management and mechanisms are included in Kilopower total mass.

Table 5. Jezero Crater crew Expedition 1 fission power generation mass (kg).

Description	Lander 1	Lander 2	Lander 3	Lander 4	
Electrical Power Subsystem					
Power Generation					**Expedition 1 Fission Power GenerationTotal**
10 kWe Kilopower	8,769	0	0	0	
Kilopower Power Management					
Stirling AC Cable	62.4	0	0	0	
Stirling Controller	322.4	0	0	0	
FISSION POWER SYSTEM	**9,154**	**0**	**0**	**0**	**9,154 kg**

As noted, as much as an additional 1,038 kg PMAD mass could be needed on the first lander, if the Kilopowers were relocated and the first lander required 1,000 VDC to 120 VDC conversion and distribution for other loads remaining on that first lander. The second, third and fourth landers of Expedition 1 would each require a 1 km spool of high voltage cabling, connectors, and voltage converters. Assuming cables could be retrieved and relocated, at least two of the Expedition 1 cables could be re-used for subsequent missions (after cargo was unloaded from Expedition 1 landers); this means that Expedition 2 and 3 landers could carry lower PMAD mass. In any case, no additional fission power generation mass would be required for Expeditions 2 or 3. By expedition 4, the Kilopower units would be nearing end of service life, so the first lander of Expedition 4 would carry similar mass to the first lander of Expedition 1.

3. Jezero Crater Solar-Powered Mission Concept

The study team estimated that around-the-clock propellant production using the first two solar-powered landers to Jezero Crater would require a maximum power load of 34,255 W during the day and 34,998 W at night, under nominal conditions. During a dust storm, power use would be reduced to 10,985 W during the day and 11,728 W at night. Once crew arrived, combined loads of the first four Expedition 1 landers were 31,915 W during nominal daytime operation and 26,790 W at night; under dust storm conditions, loads dropped to 22,945 W during the day, and 24,060 W at night.

The study team estimated that all four Expedition 1 landers would require four each 12 m diameter UltraFlex™ arrays or equivalent, with each wing estimated at 240 kg, including 30% mass growth allowance, arranged as shown in Figure 10. Deploying arrays on a 9.1 m diameter lander would extend the overall footprint to about 33 m diameter. With arrays in a neutral position on a 2.66 high lander deck, the overall height was about 9.69 m above the surface. As with the demonstrator mission, arrays were deployed high to minimize interactions with the surface or

payloads, and to shed dust by gimbaling (Figure 11). By using the lander deck itself as a stable operating platform, arrays could be brought online quickly without the complication of lifting/relocating them.

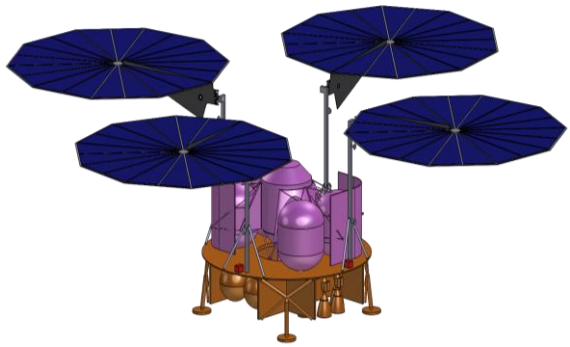

Figure 10. Crewed mission lander, shown with solar arrays deployed and a MAV payload.

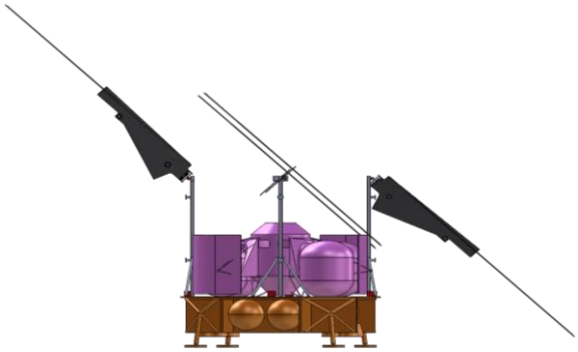

Figure 11. Crewed mission lander, shown with solar arrays gimbaled and a MAV payload.

As shown in Table 6 total landed solar power system generation and storage mass at Jezero Crater was estimated at 11,713 kg over the four-lander Expedition 1, not including lander-to-lander PMAD. As with the fission power case, as much as 1,038 kg of PMAD mass would also be required on each lander; however, unlike the fission case, cable cannot be re-purposed without losing a node of the distributed system, so each solar powered lander would have to carry PMAD mass, regardless of whether there are any powered payloads on board. *Note that the sixteen combined solar arrays on the four Expedition 1 landers only generate enough power to weather a dust storm in reduced-power mode, with propellant production reduced or suspended depending on storm severity*. All three Expedition 2 landers would also have to carry solar arrays to build up enough array area to maintain normal operation during a worst-case dust storm.

Table 6. Jezero Crater crew Expedition 1 solar power generation and storage mass (kg).

Description	Lander 1	Lander 2	Lander 3	Lander 4	
Electrical Power Subsystem	**4,890**	**1,512**	**1,512**	**1,512**	
Power Generation	1,321	1,321	1,321	1,321	
Lander Internal Power Management and Distribution	401	192	192	192	
Energy Storage	3,168	0	0	0	
Structures and Mechanisms	**660**	**476**	**476**	**476**	
Secondary Structure	416	418	418	418	
Mechanisms	244	59	59	59	
Thermal Control (Non-Propellant)	**61**	**45**	**45**	**45**	
Active Thermal Control	2.4	3.4	3.4	3.4	
Passive Thermal Control	41.8	42	42	42	
Semi-Passive Thermal Control	16.8	0	0	0	
SOLAR POWER SYSTEM	**5,611**	**2,034**	**2,034**	**2,034**	**11,713 kg**

Jezero Crater Expedition 1 Solar Power Generation and Storage Total

The study team estimated 5,611 kg of power generation and storage mass for the first Expedition 1 lander and 2,034 kg on each subsequent Expedition 1 lander, with the only difference being the batteries and battery management equipment. Table 7 provides a detailed summary of power generation and distribution mass for the first lander.

4. *Columbus Crater Mission Concept*

The study team found that because there are typically no dust storms in the southern hemisphere during winter, when days are shorter, there was no need to increase solar array area from the Jezero Crater concept. However, the longer nights at Columbus Crater did result in a 9% increase in battery mass. The study team further optimized system performance by assuming additional ISRU strings to operate at higher capacity in summer, with some ISRU strings cycled off all winter. This scheme would add about 40%, or 671 kg, to the ISRU system. Cumulative solar power generation and storage mass for the four-lander Expedition 1 mission totaled 12,679 kg (Table 8), higher than the Jezero Crater system mass. Again, lander-to-lander PMAD system mass is not included in this total.

Table 7. Jezero Crater crewed mission Lander 1 solar power generation and storage mass.

Description	Qty.	Unit Mass (kg)	Basic Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
POWER SYSTEM						**5,611**
Electrical Power Subsystem						**4,890**
Power Generation						
Solar Array Wing	4	200	800	30	240	1,040
Gimbal	4	54	216	30	64.8	280.8
Lander Power Management						
Array Regulator	4	13	52	30	16	68
Battery Regulator	1	86	86	30	26	112
Low Voltage Power Distribution	1	10	10	30	3	13
Lander Power Harness	1	126	126	55	69	195
Low Voltage DCDC Harness	1	10	10	30	3	13
Energy Storage (Battery)	1	2,437	2,437	30	731	3,168
Structures and Mechanisms						**660**
Secondary Structure						
Solar Array Post Assembly	4	80	319	18	57	377
Antenna Boom Assembly	2	5	10	18	2	11
Instrument Box	1	24	24	18	4	28
Mechanisms						
Electrical Power Installation	1	204	204	18	37	241
Thermal Control Installation	1	2	2	18	1	3
Thermal Control (Non-Propellant)						**61**
Active Thermal Control						
Heaters	5	0.2	1	0	0	1
Thermal Control	2	0.2	0.4	0	0	0.4
Thermal Switch	5	0.2	1	0	0	1
Passive Thermal Control						
Multilayer Insulation	1	4.5	4.5	0	0	4.5
Cold Plates	5	0.1	0.5	0	0	0.5
Heat Pipes	5	1.4	6.8	0	0	6.8
Radiator	1	29.2	29.2	0	0	29.2
Thermal Paint	1	0.8	0.8	0	0	0.8
Semi-Passive Thermal Control	1	16.8	16.8	0	0	16.8

Table 8. Columbus Crater crew Expedition 1 power system mass (kg).

Description	Lander 1	Lander 2	Lander 3	Lander 4	
Electrical Power Subsystem	**5,176**	**1,512**	**1,512**	**1,512**	
Power Generation	1,321	1,321	1,321	1,321	
Lander Internal Power Management and Distribution	401	192	192	192	
Energy Storage	3,454	0	0	0	**Columbus Crater Expedition 1 Solar Power Generation and Storage Total**
Structures and Mechanisms	**672**	**476**	**476**	**476**	
Secondary Structure	416	416	416	416	
Mechanisms	256	59	59	59	
Thermal Control (Non-Propellant)	**61**	**45**	**45**	**45**	
Active Thermal Control	2.4	3.4	3.4	3.4	
Passive Thermal Control	42	42	42	42	
Semi-Passive Thermal Control	16.8	0	0	0	
POWER SYSTEM TOTAL	**5,909**	**2,033**	**2,033**	**2,033**	
ADDITIONAL ISRU MASS	**0**	**671**	**0**	**0**	
TOTAL	**5,909**	**2,704**	**2,033**	**2,033**	**12,679 kg**

5. *Solar Power with Fuel Cell Backup*

The study team briefly considered a third solar-powered option: pair the solar arrays with an oxygen/carbon monoxide fuel cell instead of batteries for energy storage. In this scheme, the fuel cell would provide dust storm keep-alive power, with solar array area sized only for nominal operations. This meant that solar arrays could be smaller diameter, or only required on the first few landers. A duplicate of the ISRU system would produce fuel cell reactants, which would be stored in cryogenic storage tanks during clear weather. In a dust storm, the fuel cell would convert oxygen and carbon monoxide to energy and carbon dioxide.

Preliminary estimates assuming a 50% round trip efficiency suggested it would require four each 8.5 m diameter solar arrays about 480 days to manufacture the 16 metric tons of reactants needed to survive just a 70-sol dust storm. In other words, the energy required to make reactants was about the same order of magnitude as the energy required to produce propellant, making the system less mass efficient than the other options considered.

E. Crewed Mission Solar versus Fission Comparison

1. Performance Comparison

The fission power option provides continuous power, regardless of latitude, day/night cycle, seasonal variations, or dust storms. Given enough landers, solar power could also support continuous operation, even during a dust storm, but will always rely on stored energy for night-time operation. Solar-powered propellant production during the first crew expedition could be compromised by a dust storm more severe or longer than recorded observations, resulting in loss of mission.

The fission power system would be isolated on or near the first lander, posing no interference with other surface systems. However, because solar arrays will be mounted on all of the Expedition 1 and 2 landers, interaction with other systems must be considered. For example, to prevent damage to the MAV during departure, arrays must be removed or secured prior to MAV lift-off. Removing four 12 m diameter arrays from a single lander without damaging the MAV will be challenging particularly since safe handling may only be possible at night, when the arrays aren't energized. To meet power needs for subsequent missions, the MAV lander arrays either need to be relocated for continued use or replaced on a later mission, incurring additional mass penalty. If array diameter increases for landing sites further north, it will be even more difficult to package and handle the arrays. Nominally, crew could assist in MAV lander array relocation, mitigating the need for robotic or autonomous array handling. However, after a six month microgravity transit to Mars, the crew will need time to recondition their muscles and adjust to Mars gravity before being medically cleared to exert themselves in this type of Extravehicular Activity (EVA). Mission planners must protect for the worst-case contingency where, for some reason, the surface mission must be abandoned before crew are physically able to perform a strenuous EVA; this could potentially increase robotic array handling mass.

2. Mass Comparison

In the fission power scheme, all the power needed to reliably produce propellant within estimated schedule constraints and sustain subsequent crews arrives on the first lander of the first crewed expedition. Assuming a 12-year service life for the Kilopower units, this power generation mass will support a three-expedition campaign.

Recall that the sixteen combined solar arrays on the four Expedition 1 landers only generate enough power to weather a dust storm in reduced-power mode, with propellant production reduced or suspended depending on storm severity. Therefore, comparing solar vs. fission power generation mass for only Expedition 1 would be misleading. At Jezero Crater, all three Expedition 2 landers would also have to carry solar arrays to build up enough area to match fission power's ability to operate at nominal levels during a worst-case dust storm, plus 10 kW of spare power. Therefore, extrapolating cumulative power generation mass over the first three crewed expedition missions provides a more realistic comparison of the two schemes.

Unlike the equatorial demonstrator mission, solar power does not trade as well for mass at higher latitudes. Assuming the first two crewed expeditions to Jezero Crater accumulate sufficient solar power for nominal operations even in a dust storm, landed solar power generation and storage mass is nearly double that of a comparable fission power system; the discrepancy only increases as the landing site moves further from the equator (Table 9).

A fourth crewed expedition would incur additional mass penalty to replace aging Kilopower units, but the solar power system's three metric tons of rechargeable batteries (the mass equivalent of two Kilopower units) may also need to be replaced at that point.

3. System Robustness

If the campaign is limited to a single landing site, power system mobility will be key to exploring further from the home base. As with the demonstrator mission, dust is a major concern for solar array mechanisms. The compact Kilopower units may have an operational advantage because they can be easily relocated with minimal reconfiguration whereas it may be difficult to re-stow a large, dusty solar array for surface transport and redeployment. Solar arrays are generally not designed to be retractable in a dusty environment. Large solar arrays will also be more sensitive to Martian wind loads than the relatively compact, low center of mass Kilopower units. On the other hand, solar power system electronics will not be exposed to system-generated radiation, and solar arrays can be relocated as soon as they are deactivated; a 10-kWe Kilopower unit won't be safe for crew to approach until about a week after it's turned off, though it could be robotically handled within one sol.

One distinct advantage of solar power is that if every lander were fitted with arrays, each lander could be independent for keep-alive power shortly after landing, minimizing the time pressure to robotically attach surface power system cables to each new lander as it arrives. With up to four arrays per lander, a dust-induced gimbal failure or degradation of a single wing could be easily tolerated with array area accumulated by Expedition 3. Solar power generation closer to the point of use also reduces transmission losses, mitigating the need to generate more power than what is actually needed.

Table 9. Mass comparison for 3-mission crewed campaign power generation and storage options (*not including lander-to-lander power distribution mass*).

Crew Expedition	Power Generation/Storage Mass (kg)		
	Fission Power	Solar Power	
		Jezero Crater	Columbus Crater
Expedition 1	**9,154**	**11,713**	**12,679**
Lander 1	9,154	5,611	5,909
Lander 2	0	2,034	*2,704
Lander 3	0	2,034	2,033
Lander 4	0	2,034	2,033
Expedition 2	**0**	**6,102**	**6,770**
Lander 1	0	2,034	*2,704
Lander 2	0	2,034	2,033
Lander 3	0	2,034	2,033
Expedition 3	**0**	**0**	**0**
Lander 1	0	0	0
Lander 2	0	0	0
Lander 3	0	0	0
Three Mission Total (kg)	**9,154**	**17,815**	**19,449**

**Columbus Crater totals include additional ISRU strings on MAV landers.*

If fission-powered landers are daisy-chained together via a single transmission cable back to a Kilopower farm, there is a risk of disruption due to a severed transmission cable. Risk can be mitigated by adding a second cable, but at a mass penalty. Solar arrays on every lander would make the system less vulnerable to a severed power distribution cable—at least during the day—but the downside is that each lander has to give up cargo space for array mass. Loss of the transmission cable back to the lander that houses the night-time energy storage batteries poses a significant night-time risk, though this could be mitigated by distributing battery mass among landers.

4. Cost Comparison

A rough order of magnitude cost estimate was performed for the crewed mission solar power generation and storage system but not for the fission system. Kilopower fission system development costs have been estimated in previous publications[16]. As shown in Table 10, the solar power generation and storage hardware for the first lander of the first crewed expedition is estimated at about $456 M. Because the first lander carries all of the energy storage mass, subsequent landers were estimated to have lower costs: $196 M for Lander 2, $188 M for Lander 3, and $194 M for crewed Lander 4. Costs for power systems on Expedition 2 landers 1, 2, and 3 would be the same as, or slightly lower than, Expedition 1 landers 2, 3, and 4. Solar power cost estimates include only the power system and directly related structures and thermal control for the first lander. PMAD costs were removed for simplified comparison. Launch services, mission-level costs, other payloads (such as ISRU), payload and system integration, and technology development up to TRL 6 were not included. Estimate assumed prototype development for the solar arrays, gimbals, and batteries and protoflight development of remaining components. Flight spares include solar arrays, gimbals, batteries, power electronics cards, and power system thermal control components. No flight spares or test hardware were included in the estimates for power generation on landers 2-4. Quantitative risk analysis was performed using a Monte Carlo simulation driven by input parameter uncertainty and the error statistics of the cost estimating relationships. Costs presented are mode values (approximately 35th percentile) in constant FY16 U.S. dollars. The omitted PMAD costs (DDT&E and FHW) are estimated to be an additional $124.5M.

Table 10. Lander 1 of Expedition 1: solar power generation and storage estimated costs (FY16$M)

Description	Design & Development Estimate	System Test Hardware	Flight Spares	Design, Development, Test & Evaluation Total	Flight Hardware Total	Total
Electrical Power Subsystem	**105.5**	**42.8**	**45.4**	**193.8**	**224.8**	**418.6**
Solar Array Wing	72.8	33.7	33.7	140.3	135.0	275.3
Gimbal	6.2	6.4	6.4	19.0	25.8	44.8
Lander Power Harness	8.2	0.0	0.0	8.2	*5.6*	13.8
Battery	18.3	2.7	5.3	26.3	*58.4*	84.7
Thermal Control (non-Propellant)	**4.1**	**0.0**	**0.3**	**4.3**	**2.6**	**6.9**
Structures and Mechanisms	**10.4**	**0.0**	**0.0**	**10.4**	**19.8**	**30.3**
System Subtotal	**120.0**	**42.8**	**45.7**	**208.5**	**247.2**	**455.8**

Does not include lander-to-lander power management and distribution hardware costs.

Although Fission power system development and flight certification costs, given the required development, launch safety and security overhead, are higher than solar power costs, there will still be a substantial cost to design, develop, test, and flight certify a crew-rated solar power system compatible with the Mars surface environment.

5. Operational Life Comparison

There is insufficient data for a comprehensive operational life comparison. A 12 year service life is assumed for the Kilopower units, but no life test data is currently available. With few moving parts, mechanical failures are less of a concern than system electronics operational life.

NASA's Opportunity rover has demonstrated solar panel and rechargeable battery performance over a 12 year service life on Mars—but its twin, Spirit, survived only half that long[17]. Lithium sulfur batteries were selected for this study due to their high specific energy and projected future improvements, though they currently have relatively poor service life[18]. To put a Mars mission in perspective: a three-expedition Mars service life equates to about 8,244 battery day/night charge/discharge cycles, but Lithium batteries are currently capable of only about 6,000 charge/discharge cycles at 100% depth of discharge[19]. At a more reasonable 75% depth of discharge, these batteries may last 10,000 cycles[20].

V. Discussion

1. ISRU Demonstrator Mission

By assuming an equatorial landing site, ISRU demonstrator mission analysis was purposely skewed towards solar power as more mass efficient; note that the trade is expected to quickly swing towards fission power at sites further from the equator. This analysis was also skewed *against* fission power, since the study team assumed a baseline 10-kWe Kilopower unit, which was about 20% over-sized for the 1/5 scale demonstrator. What's more, about a third of a nominal Kilopower unit's mass is dedicated to crew radiation shielding, which is not strictly necessary for an uncrewed demonstration. Since both the fission and solar concepts fell comfortably within allowable mass limits for the demonstrator lander, mass alone is unlikely to be a delineator for this type of mission. A final decision will hinge on many factors beyond the scope of this analysis, including technology investment strategies, program budgets, and risk mitigation needs for later crewed missions.

2. Crewed Mission

50 kWe of fission power generation (four primary 10-kWe units plus a fifth spare) is about 20% less landed mass than 35 kW of solar power generation and storage for the first expedition to Jezero Crater, not including a lander-to-lander PMAD system; adding PMAD further increases fission power's mass advantage over solar power, particularly if Kilopower transmission cables can be re-used as previous landers are decommissioned. By the third crewed expedition mission to Jezero Crater, enough solar array area will have been accumulated to match the fission system's performance, even in a dust storm--but at more than twice the cumulative landed mass of the fission system, not including PMAD. Landing closer to the equator would make solar more competitive with fission for landed mass, while moving further from the equator would make solar power less competitive.

The two landing site point solutions that were studied offer some insight into solar power mass sensitivity with latitude, but it remains forward work to evaluate the full range of sites under consideration. From a resource prospecting point of view, sites further from the equator appear more likely to feature water ice[21]—but solar power mass penalty also increases as a function of distance from the equator. Northern versus southern hemisphere seasonal variations must also be considered. The Columbus Crater analysis demonstrates that overall mission mass can be minimized with a careful systems engineering approach that takes advantage of operational schemes to offset environmental constraints.

Because lander-to-lander power transmission mass was assumed to be comparable between the two power generation systems, it remains forward work to develop a detailed PMAD design that optimizes transmission voltage and meets end user needs. At an estimated metric ton per lander, PMAD mass represents a serious potential mass threat.

Note that masses shown are for stand-alone systems, but may change once more detailed lander integration analysis is performed. Thermal and structural support mass for the solar arrays may be stripped out if the lander already carries mass to support these functions (lander internal cold plate function shifts from in-flight systems to power generation after landing, for example). Also note that analysis was based on a single point design solution; many factors could alter the analysis. If propellant production rates increase, either because more propellant is needed or there is less time to make it, power generation and energy storage mass will increase.

As with the demonstrator mission, a final decision regarding Mars surface power generation will depend on many factors beyond the scope of this analysis, including program budgets, technology investment strategies, and crew or mission risk posture.

VI. Conclusions

ISRU demonstrator mission analysis indicates that solar power is more mass-efficient than fission power, *at least for an equatorial landing site* and a somewhat oversized 10-kWe Kilopower unit. A solar-powered concept operating only during daylight hours offers the lowest mass at 1,128 kg, but takes twice as long to meet fission's performance, and potentially has a higher risk of ISRU reliability issues due to daily on/off cycling. An around-the-clock solar-powered system was estimated at 2,425 kg, or about 326 kg less than a fission-powered system of comparable performance. All options evaluated fit within the mass envelope of a commercial launch platform, making mass less of a delineator between them. The fission-powered system's flight hardware costs beyond TRL 6 were estimated at $428M, about $100M more than the comparable solar-powered system. Final selection will depend on long-term technology investment strategies and priorities: is it more important to demonstrate technologies needed for a future crewed mission, or to keep mass as low as possible?

Crewed mission analysis indicates that solar power is more feasible under NASA's new Evolvable Mars Campaign (where surface assets are re-used for subsequent expeditions) than it was under the DRA 5.0 scheme. The mission concept developed by the COMPASS team demonstrates that a solar-powered crew mission is certainly *possible*, though not as mass-efficient as fission power for the two landing sites analyzed. The study team's approach of adding solar arrays to multiple landers helps spread the mass penalty out over time, but requires program managers to accept a higher loss-of-mission risk for at least the first crew expedition (and possibly more, depending on latitude selected and propellant production timelines). As solar array area is accumulated, mission risk is substantially reduced. This approach makes higher latitude sites more accessible for solar power, though it remains forward work to evaluate cumulative array area needed for all landing sites of interest. An additional benefit of lander-mounted solar power is that it makes that lander self-sufficient for keep-alive power shortly after landing, without the time pressure of autonomously connecting cables to a remote power farm before on-board energy storage is depleted.

Cumulative fission power generation mass over the first three crewed expeditions to Mars is estimated at 9,154 kg, compared to 17,815 kg for a comparable solar power generation and storage system at Jezero Crater (not including lander-to-lander distribution and conditioning mass); 19,449 kg would be required for a Columbus Crater solar-powered mission. Clearly, the method of surface power generation will be an important factor in landing site selection (or vice versa). A comprehensive cost comparison remains forward work, as it depends on many factors such as landing site, surface element design and power needs, potential cost-sharing between flight programs, and technology investment strategies. Development of either solar or fission surface power for Mars would require a substantial financial commitment.

Pairing solar arrays with fuel cells seems logical, particularly if in situ resources are available, but the study team concluded that the energy required to make reactants was about the same order of magnitude as the energy required

to produce propellant, making the system less mass efficient than the other options considered. If rechargeable battery service life requires frequent replacement, fuel cells may trade more favorably than solar power over a multi-expedition timeframe.

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