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THE MARS HOPPER: AN IMPULSE DRIVEN, LONG RANGE, LONG-LIVED MOBILE PLATFORM UTILIZING IN-SITU MARTIAN RESOURCES

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The requirements for performance by planetary exploration missions are increasing. Landing at a single location to take data is no longer sufficient. Due to the increasing cost, the missions that provide mobile platforms that can acquire data at displaced locations are becoming more attractive. Landers have also had limited range due to power limitations, limited lifetime of subsystems and the inability to negotiate rough terrain. The Center for Space Nuclear Research has designed an instrumented platform that can acquire detailed data at hundreds of locations during its lifetime - a Mars Hopper. The Mars Hopper concept utilizes energy from radioisotopic decay in a manner different from any existing radioisotopic power sources—as a thermal capacitor. By accumulating the heat from radioisotopic decay for long periods, the power of the source can be dramatically increased for short periods. Thus, a radioisotopic thermal rocket (RTR) is possible. The platform will be able to "hop" from one location to the next every 5-7 days with a separation of 5-10 km per hop. Each platform will weigh around 50 kgs unfueled which is the condition at deployment. Consequently, several platforms may be deployed on a single launch from Earth. With a lifetime estimated at 5-7 years, the entire surface of Mars can be mapped in detail by a couple dozen platforms. In addition, Hoppers can collect samples and deliver them to the Mars Science Laboratory for more detailed analysis. The design and performance of the Mars Hopper will be discussed.

I. INTRODUCTION

The requirements for performance by planetary exploration missions are increasing. Landing at a single location to take data is no longer sufficient. Due to the increasing cost, the missions that provide mobile platforms that can acquire data at displaced locations are becoming more attractive. The Mars Exploration Rovers, Spirit and Opportunity, have performed remarkably for 5 years and have covered 4.8 and 10.7 miles respectively. The Phoenix lander has sampled the Martian surface close to the northern polar cap of Mars but only at one location. Spot sampling at a few locations, especially locations that are safe landing sites, will not provide an accurate geophysical map of Mars. In addition, if humans are ever going to land on Mars, we must produce a much more detailed map of the resources, terrain and subsurface in order to know where to land.

Robotic probes for planetary exploration are an excellent way to acquire high quality data to ground-truth orbiting observations. The orbiting systems allow large area coverage and the ground systems provide spot validation. The ability to fly such missions is getting ever more costly and increasing dependent on international collaboration in order to maintain an adequate mission frequency. In addition extended lifetime, long range exploration and sampling to significant depths of several meters requires significantly more power than what can be provided solar cells. Furthermore, the unavailability of Pu-238 means that extended lifetime and long range exploration will become more difficult and in addition alternative power sources for Mars orbiters may not be available after the last Flagship-scale mission to Europa in 2020.

More ambitious technologies must be developed and methods created to increase the science return for each launch, thus increasing the scientific value for the money spent for each mission. Several previous studies have proposed the use of "hoppers" powered by one means or another [1-3]. However, these concepts suffered from both short range and relatively short operational durations. Conceivably, if an instrumented platform could be placed on the surface of a planet that could acquire highly detailed data from the surface and subsurface, travel large distances to multiple sites and perform this task repeatedly, then an entire planetary surface could be accurately mapped and sampled with higher resolution that orbiting platforms and extended range in comparison to current in-situ missions. In addition, if several such platforms could be simultaneously deployed from a single launch vehicle, a surface network of science stations would

be possible that provided long term assessment of meteorological conditions.

The CSNR is designing an instrumented platform that can acquire detailed data at hundreds of locations during its lifetime - a Mars Hopper. The platform will be able to "hop" from one location to the next every 5-7 days with a separation of 5-10 km per hop. Each platform will weigh around 50 kgs unfueled which is the condition at deployment. Consequently, several platforms may be deployed on a single launch from Earth. With a lifetime estimated at 5-7 years, the entire surface of Mars can be mapped in detail by a couple dozen platforms. Furthermore, the basic platform can be deployed to Europa, Titan, and even Venus with alterations- the propulsion system and operations essentially will be the same

II. CONCEPT

The basis for the concept is to utilize the decay heat from radioactive isotopes to heat a block of material to high temperatures. While the heating is taking place, some of the thermal power is diverted to run a cryocooler. The cryocooler takes in the Martian atmosphere and liquefies it at roughly 2.8 MPa. The liquefied CO_2 is transferred to a tank. Once full, the power convertor is turned off and the core is allowed to increase in temperature. After the peak temperature is reached, the liquid CO_2 is injected into the core, heated, expanded through a nozzle, and allowed to produce thrust. One half of the CO_2 propellant is "burned" for ascent. After a ballistic coast, the remaining propellant is used for a soft landing. Once landed, the process repeats.

The Mars Hopper concept utilizes energy from radioisotopic decay in a manner different from any existing radioisotopic power sources—as a thermal capacitor. Radioisotope sources have very high specific energy, j/kg, while having rather low specific power, w/kg. By accumulating the heat from radioisotopic decay for long periods, though, the power of the source can be dramatically increased for short periods. Thus, a radioisotopic thermal rocket (RTR) is possible.

Radioisotope powered propulsion for interplanetary space missions was considered back in the 1960s in the Poodle program. At the time, both Strontium-90 (Sr-90; 29 year half life) and Polonium-210 (Po-210; 138 day half life) were considered. The primary reason for consideration was that the specific energies of Sr-90 and Po-210 were about 1.2×10^5 and 2.4×10^6 MJ/kg, respectively. These values are 12,000 and 240,000, respectively, times greater than the specific energy of chemical propellants, i.e. 10 MJ/kg. While the energy content of radioisotopes was high, the power delivered was low. At the time, the low power capability of the isotopically powered engines could not support the demands of the airframe mass. In addition, compact engines that could withstand the potential of ground impact, in the case of launch abort, were difficult to conceive due to the state of materials technology at the time.

Radioisotopes have been used in the US space program since the early 1960s for the provision of electrical power and thermal management. Pu-238 fueled Radioisotope Thermoelectric Generators (RTG) were used in all of the Apollo landings with the exception of Apollo 11, the Viking 1 and 2 missions, and all planetary exploration missions beyond Mars. Radioisotope Heating Units (RHU) have been used to maintain operational temperatures of spacecraft and robotic platforms such as during lunar night, cold temperatures on mars, and in deep space . Pu-238 has a specific energy of 1.6×10^6 MJ/kg (4.4×10^5 kw-hrs/kg). Factoring in the 6% conversion to electricity for thermocouples [4,5], the RTG had 9.6×10^4 MJ/kg of electrical energy compared to the 0.72 MJ/kg for Li-ion batteries.

Although RTGs have been launched on several vehicles for the past few decades, encapsulation of the Pu-238 in case of launch abort is an ever-present issue. Recent developments in the fabrication of tungsten-rhenium (W-Re) materials at the Center for Space Nuclear Research (CSNR) at the Idaho National Laboratory (INL) indicate that a solid, tough, high-temperature matrix can be formed to encapsulate the radioisotopes [6,7]. The radioisotope material can be fabricated into small particles, perhaps 100 microns in diameter, uniformly dispersed in the W-Re matrix. The current thinking is that the W-Re encapsulation process will prevent the dispersion of the radioisotopes when exposed to the destructive forces associated with spacecraft re-entry and launch abort scenarios or even planetary impacts (in the case of failed in-situ probe deployment). Flow channels built within the matrix material allow for the efficient transfer of heat to the working fluid. Figure 1 shows examples of tungsten elements created at the CSNR. The elements had a 40% by volume load of cerium oxide as a surrogate for plutonium dioxide.

The direct transfer of heat to the atmospheric gases in the inlet (regardless of their chemical composition) will allow sufficient thrust to be produced. There is no need for propellant or oxidizer that could limit the duration and lift capability of the craft. After landing

on Mars, the Hopper stores heat and electricity while performing science measurements. Once the measurement cycle and heat storage phase are complete, the vehicle "hops" to a new location where the cycle is repeated. Theoretically, this process can continue until the radioisotope is depleted or until a critical component on the vehicle fails.
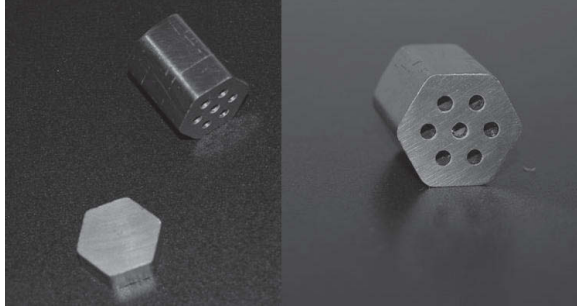

Figure 1. W nuclear thermal rocket fuel elements loaded with CeO_2 (40% vol.), which acts as the PuO_2 simulant [6].

III. SUBSYSTEM ASSESSMENTS

Operation of the Mars Hopper will be significantly different than previous platforms. The basis of the concept requires many, perhaps hundreds, of thermal cycles. In addition, a substantial temperature range will be encountered by the engine materials during each cycle. Finally, the ability to thermally isolate the core from the pressure vessel places a significant requirement on the design. All of these features necessitate innovative solutions. However, preliminary efforts indicate that a Hopper can be designed than requires relatively few parts, utilizes local resources, and can repeatedly perform significant jumps.

The major issues which have been examined are 1) radioisotope options, 2) core material options, 3) power conversion, 4) thermal management, and 5) performance.

Radioisotope Selection

An initial radioisotope selection process found that the primary consideration factors were availability and qualification level for use in power sources. The initial assumption of a $240W_e$ requirement using 25% efficient Stirling engines led to a radioisotope loading of $960W_t$. Table 1 shows the mass of isotope required for $960W_t$ BOM.

While Plutonium-238 has been used expensively in space missions and would require the least change in infrastructure and device qualification, its future availability is uncertain. If Pu-238 was available for the Hopper mission, this would likely be the best material choice from a time perspective. Conceivably, old Pu-238 sources that are below the specifications required by NASA for current RTGs could be used for the Hopper with modest impact on the design.

Isotope:	Specific Power (W/g):	Mass Compound (kg):
Pu-238	0.392	2.45
Sr-90	0.254	3.78
Cm-244	2.269	0.42
Am-241	0.094	10.21

Table 1: Isotope Mass Requirements for $960W_t$

Other isotopes such as Curium-244 and Americium-241 are attractive options, but would require qualification testing and are difficult to acquire in sufficient quantities. Strontium-90, however, has been used in terrestrial power sources and is readily available at Hanford [8]. However, the Bremstrahlung radiation from Sr-90 decay requires substantial amounts of heavy shielding. Most recently, the advantages of using Americium, i.e. the reduced radiation levels, have also proven feasible if the availability question can be addressed.

Preliminary calculations indicate that the radiation levels at the surface of the Hopper will be well below the 2 mSv/hr currently experienced in MMRTGs. The low level is the result of having the Pu-238 encapsulated in tungsten, reducing the gamma ray dose, and having the fuel elements surrounded by beryllium, reducing the neutron dose. The dose level is well within acceptable levels for handling and integration.

Core material selection

While the RTR core matrix material is used primarily for heat storage, it is also necessary for the matrix to have high thermal conductivity and high melting temperature.

Several elements have been assessed on the basis of specific heat, thermal conductivity, and melting point and assigned a figure of merit. Table 2 shows a comparison of candidate materials. This analysis determined that Beryllium is the optimal matrix material. Beryllium has a density of 1.85 g/cm^3, a melting point of 1551 K, thermal conductivity of 200

W/m K (at 298 K), and a specific heat of 1825 J/kg K (at 298 K). The very large specific heat is the primary characteristic that makes Be attractive.

As seen in Table 2, the heat capacity of Be is the dominant characteristic. In fact, the capacity increases dramatically with temperature as shown in Figure 2. This dependence on temperature makes the core temperature calculation as a function of propellant mass flow nonlinear. The remarkable ability of Be to hold heat enables the Mars hopper concept.

Material	T_{melt} (K)	Thermal Conductivity (w/m-K)	Heat Capacity (j/kg-K)
Carbon	3823	165	710
Tungsten	3695	173	130
Beryllium	1551	200	1820

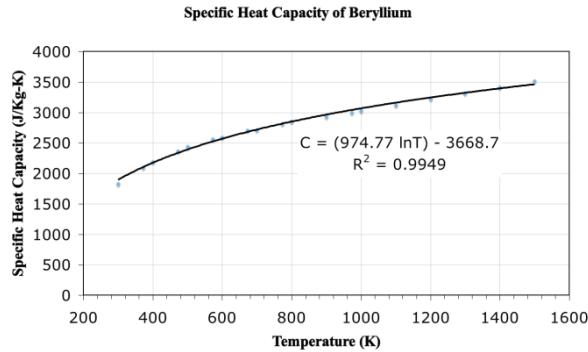Table 2. Comparison of fundamental properties of candidate matrix materials.

Figure 2. Specific heat capacity of beryllium as a function of temperature. Plot of data taken from [9].

The heat requirement was assessed to provide enough electrical power to run the compressors and also heat the core to the peak operating temperature. A significant design issue is the ability to thermally isolate the core from the engine walls. Conceivably, any conductive paths may be minimized and thermal insulation used to keep the core isolated. In order to determine the heat lost, a core size as well as insulation properties are assumed. The assumptions made for the preliminary analysis are seen in Table 3.

Energy Storage:	100 MJ
Flow Channel Diameter:	2.5 mm
Insulator Thickness:	5 cm (AETB-8)
Insulator Thermal Conductivity:	0.13 W/m-K
Radiating Temperature:	500 K
Max Core Temperature:	1300 K

Table 3: Assumptions for Heat Loss

Power Conversion

The CSNR is currently evaluating the least massive method to use the heat from the core to drive a power conversion system generating the electricity to drive the cryocooler. Three options have been identified that can provide power:

1. Closed loop Stirling engine.
2. Closed loop He Brayton cycle
3. Thermoelectric conversion

The Stirling engine is attractive in that space-qualified units have been built and tested by the Sunpower Corporation [10]. Figure 3 shows a 115 W_t Stirling engine that generates 35 W_e. The hot head temperature is 650°C and the rejection temperature is 120°C. The engine efficiency is quoted as 34% with the overall conversion efficiency at 29%. Potentially and subject to further evaluation, with the lower rejection temperature available at Mars, the engine ouput could be raised to 44 W_e. The total mass of the system is 309 g with a specific power of 109 W/kg.

The efficiency of a radioisotope thermoelectric generator (RTG) system is a key mission design driver due to its impact on the cost, availability, size and weight of the power source. To date, the conversion efficiency of thermoelectrics (< 7%) has been relatively poor compared to those of e.g. Stirling cycle converters (~30%) [4], and emphasis needs to be on maximizing the electrical power conditioning. Each percent of power conditioning loss will increase the size of the RTG system by approximately 14%, which reflects directly on the mass of the vehicle.
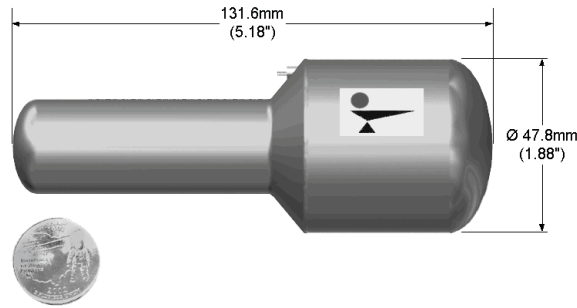
Figure 3. Space qualified Stirling engine with a 35 W_e output [10].

Research in thermoelectric materials suggests that significant improvements in efficiency are possible through the development of high performance materials including e.g. oxide thermoelectrics, rare-earth intermetallics, engineered crystal lattices and skutterudite-based materials which have thermal and electronic transport properties that can be manipulated independently [5]. The efficiency of current RTGs can also be improved by using thermoelectric materials suitable for higher temperature operation (up to 1000ºC) [5]. These developments have either led to new commercially available thermoelectric generators, or new thermoelectric materials which are close to market. The challenge for space applications is the qualification of these materials and the inclusion of new higher efficiency systems in the next generation of RTGs. Despite these advances, based on the current state of the art, it is unlikely that thermoelectric sytems could exceed the efficiencies of Stirling conversion systems.

There is a place for thermoelectrics in the concept proposed in this study either as supplementary to the primary Stirling system or in a scenario where the repetition rate of the hopper is extended, and a lower power requirement becomes acceptable if the overall mass of the conversion system is reduced. An additional advantage of thermoelectrics is the absence of moving parts.

Current efforts are underway to identify possible methods to reduce the mass of the Stirling system and to compare the estimated mass of the direct drive system.

IV. THERMAL MANAGEMENT

Three major issues exist in thermal management: 1) Thermal isolation, 2) Heat transfer to the propellant, and 3) Thermal cycling.

The thermal isolation of the low power thermal source is critical in order for the core to reach the required temperature in a practicable time period. The heat transfer requirement impacts on the length of the core and its mass. The thermal cycling qualification element of the design will impose lifetime limits for the entire system.

Preliminary calculations of the thermal isolation requirement indicate that the core can be sufficiently isolated so that only radiative cooling dominates over the system. This is accomplished by placing the core on an insulating zirconium carbide disk that overlaps the core sufficiently for structural support during launch with minimal overlap for heat conduction. The results of the STAR-CCM steady-state calculations are shown in Figure 4. The core has a 2 cm thick layer of zirconium carbide surrounding the outer surface only (orange layer in Figure 4). The entire assembly is assumed to be in a hard vacuum. The calculations show that the core can achieve a maximum temperature of 1206 K uniformly distributed across the radius. The outer surface of the zirconium carbide layer reaches a temperature of just under 1000 K. The radiative loss from all surfaces is equivalent to the 1000 W_t being generated by the radioisotope source. The calculation shows that roughly 50 W of thermal power are lost via conduction through the zirconium carbide disk.
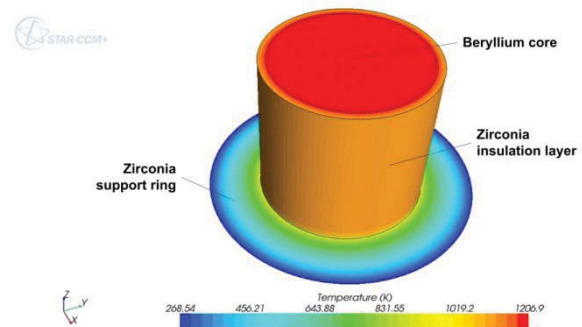

Figure 4. STAR-CCM model of the steady state temperature reached by a radioisotope encased in a Be core surrounded by a zirconium dioxide layer. The entire assembly is in a vacuum.

The second design challenge will be the ability to transfer the heat rapidly from the Be matrix to the CO_2 gas. One element of the preliminary design of the core was the separation of the heating produced by the radioisotope and the cooling by the CO_2. In the model the radioisotope was encapsulated in W rods (See Section 4), which are embedded in the Be and not exposed to any coolant flow. This implies that the core temperature can be allowed to increase over a period of hours. The cooling system is composed of a number of independent channels passing through the Be core.

The heat transfer rate will impact the core's length-to-diameter ratio and, thus, the core mass. Based on results from the NERVA nuclear rocket studies in the 1960s and on preliminary analytical results, the core matrix material should be able to transfer the heat at the required rate. However, the transfer rate is dependent on the temperature difference between the Be and the propellant. The core-material temperature continually decreases resulting in changes in both the heat capacity and the heat transfer rate. Computational modeling using time-dependent, finite-differencing techniques is underway.

The third challenge is to minimize any degradation of performance as a result of thermal cycling, which has not yet been examined. Over a period of several days the core will cycle from 250 K to 1206 K. Similarly, the nozzle and the plenum feeding the CO_2 to the core will also experience thermal cycles. Preliminary designs for the plenum have incorporated a bellows to allow for expansion and contraction of the core. In order to allow for differences in expansion of the Be and the zirconium carbide outer layer a small gap or the segmentation of the insulator layer may be necessary.

V. ESTIMATED PERFORMANCE

Preliminary results show that a platform carrying a 200 kg payload can hop repeatedly over the Martian surface until some part fails and the craft falters. From a power supply standpoint, the Hopper can operate for decades with the interval between hops growing with time. Initial results indicate that the Hopper could cover over 10 km each time. In this scenario, roughly one half of the liquefied propellant is expended during the ascent. The craft then coasts on a ballistic trajectory reaching a peak altitude >1 km. At the appropriate altitude after reaching the peak in the ballistic trajectory the engine restarts, i.e. a propellant tank valve opens and the craft descends. As a consequence of the reduced mass during the descent phase, roughly 5% of the propellant is left as

a contingency at touchdown. Assuming a thermal power provided by the radioisotope of roughly 1000 W, the ship should be able to repeat the cycle every 2-3 days.

Preliminary results for two instrument payloads are shown in Table 4. The results assumed that Pu-238 was encapsulated in a W matrix with a 50:50 volume ratio. The W was then distributed into the Be core which is further encapsulated in a Ti can. The entire core is within a Ti pressure vessel which comprises the engine. Ranges for the two scenarios were calculated using locally written codes that included a rough approximation for the Martian atmospheric density change with altitude and a simple estimation of drag. A tankage fraction as well as a structure fraction of 0.09 for the CO_2 was assumed. In all cases, the maximum core temperature of 1000 K was assumed corresponding to an initial specific impulse of approximately 124 s.

The results show that a range of over 10 km can be achieved for either the 10 kg or 200 kg payload using between 1-2 kg of Pu-238. The three scenarios shown in Table 4 were chosen to cover the range of potential missions. The 200 kg payload is approximately equal to the Phoenix lander. Case 2 assumes the same platform as Case 1 except only a 10 kg instrument package is used. These two results show that a common platform can be used for various sized payloads. Case 3 demonstrates how small the system could be. The core containing 2.16 kg of Be would have a radius of just over 6 cm. The repetition intervals for each case are roughly 4, 4, and 5 days respectively at the assumed power levels.

VI. SUMMARY AND CONCLUSIONS

Preliminary results indicate that a small, compact system with relatively few parts can be designed that will utilize the in-situ Martian resources to "hop" around the surface. Such a probe will enable high-fidelity data to be acquired at hundreds of sites over a period of a few years. Potentially, this system can travel from the equator to a pole in two years. Thus, a comprehensive set of data using the same instruments can be acquired over much of the surface of Mars. In addition, depositing several of these platforms on Mars will allow the entire surface to be mapped within a decade using only a few launches from Earth. Finally, the proof of concept for this system can be performed using electrically heated cores with a modest investment.

Parameter	Case 1	Case 2	Case 3
Mass payload (kg)	200	10	10
Total Energy stored (J)	1.0e8	1.0e8	6.0e6
Isotope power (W)	1000	1000	500
Mass Pu-238 (kg)	2.0	2.0	1.0
Mass Beryllium (kg)	55.	55.	2.16
Mass flow(kg/s)	16.6	13.36	0.68
Mass initial (kg)	1061	852	50.
Mass propellant (kg)	580	580	22.
Mass final/mass initial	0.45	0.32	.56
Ascent burn time (s)	16.45	22.6	18
Initial Thrust (N)	20115	16157	947
Thrust at Burnout (N)	11854	9094	511
Burn Out Velocity (ms^{-1})	230	327	213
Range (km)	15.9	30.9	10.0

Table 4. Predicted operational characteristics of the Hopper for two payload masses

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