Extraterrestrial Regolith Derived Atmospheric Entry Heat Shields

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ABSTRACT

High-mass planetary surface access is one of NASA's technical challenges involving entry, descent and landing (EDL). During the entry and descent phase, frictional interaction with the planetary atmosphere causes a heat build-up to occur on the spacecraft, which will rapidly destroy it if a heat shield is not used. However, the heat shield incurs a mass penalty because it must be launched from Earth with the spacecraft, thus consuming a lot of precious propellant. This NASA Innovative Advanced Concept (NIAC) phase I project investigated an approach to provide heat shield protection to spacecraft after launch and prior to each EDL thus potentially realizing significant launch mass savings. Heat shields fabricated *in situ* can provide a thermal-protection system for spacecraft that routinely enter a planetary atmosphere. By fabricating the heat shield with space resources from materials available on moons and asteroids, it is possible to avoid launching the heat-shield mass from Earth. Regolith has extremely good insulating properties and the silicates it contains can be used in the fabrication and molding of thermal-protection materials. In this paper, we will report on the findings of the NIAC phase I study, (Hogue et.al. 2012 ASCE Earth & Space 2012) (Hogue et. al. 2012, NIAC final report).

Introduction

Such in situ developed heat shields have been suggested before (Lewis 1996). Prior research efforts (Hintze et.al. 2009), (Roberson et.al. 2009), (Balla et.al. 2011) have shown that regolith properties can be compatible with very-high temperature resistance.

Routine access to space and return from any planetary surface requires dealing with heat loads experienced by the spacecraft during reentry. We address some of the key issues with the EDL of human-scale missions through a highly innovative investigation of heat shields that can be fabricated in space by using local resources on asteroids and moons. Most space missions are one-way trips, dedicated to placing an asset in space for economical or scientific gain. However, for human missions, a very-reliable heat-shield system is necessary to protect the crew and sensitive payloads from the intense heat experienced at very high entry velocities of approximately 11 km/s ~ Mach 33 (Apollo). For a human mission to Mars, the return problem is even more difficult, with predicted velocities of up to 14 km/s, ~ Mach 42 at the Earth-atmosphere entry. In addition to human return, it is very likely that future space-travel architecture will include returning cargo to the Earth, either for scientific purposes or for commercial reasons. Platinum, titanium, helium 3, and other metals, elements and minerals are all high-value commodities in limited supply on Earth, and it may be profitable to mine these substances throughout the Solar System and return them to Earth, if an economical method can be found.

Mission Architectures

In this study, we examined three mission Design Reference Architectures (DRA) to determine, at least to a rough order of magnitude (ROM) the mass savings that can be realized using in situ fabricated heat shields. The use of silicate-rich regolith in the manufacturing of heat shields in space is only applicable to the entry into oxidizing atmospheres such as that of Earth and Mars. The hydrogen-rich atmospheres typical of the giant planets of the Solar System would quickly reduce silicates and all metal oxides in the regolith to its metallic elements and mostly vaporize the entire heat shield. One DRA is based on building the regolith heat shield using materials from the moons Phobos and Deimos for either descent onto Mars or return to Earth. Another is to build the heat shields on or in the vicinity of the Moon for an Earth return or a Mars mission. A third architecture envisions building heat shields on or near an asteroid for Earth return.

Architecture I - In Situ Heat Shield Fabrication at Phobos/Deimos for Earth-bound Mars Return Spacecraft and Mars EDL of Surface Exploration Craft

This scenario involves the manufacturing of a Mars aerocapture heat shield on a Martian moon and its mating and assembly onto the incoming spacecraft prior to Mars aerocapture where it is needed.

It is proposed that there will be two heat shields: one for Mars aerocapture, and one for Mars EDL. The Mars aerocapture heat shield will be fabricated on Phobos and then transported via a solar electric propulsion (SEP) tug to intercept the Mars-bound spacecraft. The SEP tug would pre-deploy via a Hohmann transfer into a staging orbit around the sun, where it would stay until the Mars-bound spacecraft intercepts it and executes a rendezvous. Once the heat shield is attached to the Mars spacecraft, then it is ready for aerocapture. Further work will be done on these orbital dynamics in a future work effort.

In the second stage of the journey (from Low Mars Orbit to the Mars surface), a new EDL heat shield will be used, and the Mars aerocapture heat shield will be jettisoned and replaced with a new heat shield to provide the highest quality and most reliable EDL capability. This will also reduce the amount of heat shield mass that has to be transported by the SEP tug for a deep space rendezvous. The Δv required to travel from Phobos to a Low Mars staging orbit is approximately 538 m/s with an aerobrake and 845 m/s without an aerobrake. Figure 1 shows an artist's concept of a regolith-derived heat shield protecting a payload as it enters the Martian atmosphere.

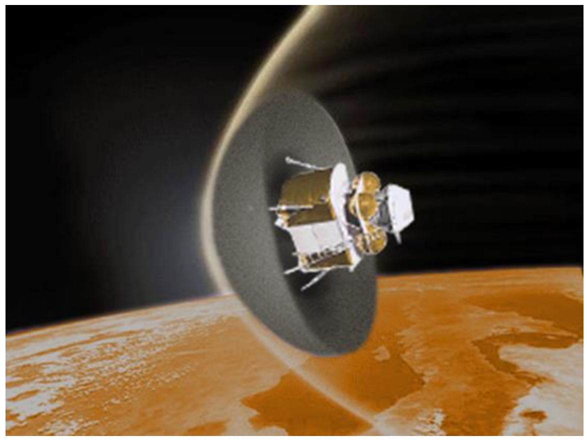

Figure 1. Artist's concept of a regolith derived heat shield entering the Martian atmosphere.

In this study it is assumed that the propellant is transported from Earth (with zero boil-off technologies) to provide the EDL heat shield transfers between Phobos and the Mars staging orbit, but it is possible that there will be in-situ water found either on Phobos itself or a nearby asteroid or comet, so that in-situ produced propellant in the form of hydrogen and oxygen could also be used eventually. We assumed that the EDL heat shield itself to have a diameter of 20-25 m, (Sostaric 2010) for a human scale mission with a lander mass of 50-60 t. With a gear ratio of approximately 6:1 then the total mass launched to LEO if propellant is brought from Earth would be 84 t. This results in an approximate mass savings of 246 t – 84 t = 162 t inserted mass in low Earth orbit (IMLEO). Taking into account the dry mass of a Phobos to Trans-Mars Solar Electric Propulsion (SEP) tug (using Phobos In-situ derived H_2) the resultant mass savings IMLEO are now reduced to 246 t – 126 t = 120 t. Since the in situ fabrication equipment (5,000 kg) also has to be transported to Phobos at a IMLEO cost of 16 t, this results in final savings on the first mission of 104 t or $520 M at $5,000/kg launch costs to LEO. Increasing savings on subsequent missions can be expected since the in situ fabrication equipment and SEP would already be in place on Phobos.

Architecture II – Lunar Heat Shield Fabrication facility for Moon-Earth returns and Moon-Mars missions

A lunar fabrication facility for heat shields on the surface of or orbiting the Moon may serve outbound transport spacecraft that need an aerocapture shield upon arrival at Mars. The servicing of spacecraft returning from Mars inbound to Earth in this scenario is unlikely as mission managers may choose not to take the risk of coming to Earth vicinity at interplanetary speeds without a shield.

The presence of robotic assets on the lunar surface is part of other architectures currently proposed and we propose to leverage and extend these capabilities to obtain the regolith needed for heat shield fabrication. The alternative options of on-surface and in-orbit fabrication are being studied and traded to best service crewed and cargo spacecraft leaving the Moon bound for Earth. The availability of a heat shield fabrication asset at the Moon would change any mission designed to shuttle between the lunar surface and Earth: in fact, such spacecraft could be launched from Earth without any heat shield since it only needs it during the final minutes of its mission during Earth EDL. We are studying the implications of such scenario on mass budgets at launch, lunar orbit insertion, lunar landings and operations and lunar launch.
Two mission architecture option hypotheses for the use of Lunar-made heat shields were considered:

1) Fabricate a heat shield on the Moon, outfit it to an orbiting spacecraft and use it to aerobrake the spacecraft returning to Earth orbit or Earth Direct Entry (for sample return, mining ore, science payloads, and crew).
2) Fabricate a heat shield on the moon, deliver it to Low Earth Orbit (LEO) and assemble it to an outbound Mars Spacecraft for aerobraking or aerocapture at Mars.

Calculations using Tsiolkovsky' s rocket equation using a Δv of 4.04 km/s and an I_{sp} of 450 s yield an associated required propulsion capability of propellant and spacecraft weighing 2,834 kg in order to deliver this heat shield to LLO. An additional 474 kg of wet propulsion stage mass is needed to return the heat shield to LEO using aerobraking. This amounts to a total mass of 3,308 kg initial mass launched to LEO required to transport the heat shield round trip to the Moon from LEO. The total terrestrially manufactured heat shield system mass in LEO is then 1,247kg + 3,308 kg = 4,555 kg.

The IMLEO mass savings per lunar orbital mission from using an ISRU regolith derived heat shield is then 4,555 kg – 2,133 kg = 2,422 kg. At a LEO launch cost of $5,000/kg this will result in a potential cost savings of $12.1 M per mission, so over a 20 mission campaign, cost savings would be $242.2 M.

These kinds of cost savings could make a lunar commercial venture profitable or reduce the costs to the point where economies of scale help the business case. Further work is needed to do sensitivity studies and account for additional surface systems support equipment but this example illustrates the potential benefits.

In the second examined case we propose to fabricate a heat shield on the Moon, deliver it to LEO and assemble it to an outbound Mars spacecraft for aerobraking or aerocapture at Mars. Using NASA Mars DRA 5.0 as a reference, the TPS and associated heat shield structure is assumed to be 41 t at a Mars staging orbit. With a gear ratio of 6:1 for LEO to Mars (as part of a 57 t lander) (Rapp 2005), then the

IMLEO is 246 t; however a fair comparison must consider that the IMLEO of the heat shield itself is 41 t, so the lunar derived heat shield IMLEO must be less than 41 t to be viable.

The same heat shield system weighing 41 t could be fabricated on the Moon and then transported to LEO, therefore saving the propellant required to launch it from Earth to LEO. If 10,000 kg of ISRU equipment is assumed to be required on the Moon's surface to fabricate the Mars heat shield robotically and produce propellant to launch it from the Moon, and then this equipment is amortized over 10 Mars missions, and that amounts to the equivalent mass of 1,000 kg delivered to the Moon per mission. This would require 3,326 kg of wet mass for the transfer stage to the lunar surface per mission and the 1,000 kg ISRU payload resulting in 4,326 kg IMLEO per mission for delivering the ISRU equipment.

The ascent stage and the lunar orbit to LEO transfer stage must then be transported to the lunar surface requiring a wet transfer stage (LEO to Lunar Surface) of mass 58,828 kg. The ascent stage and transfer stage have a combined dry mass of 17,685 kg and are then filled with lunar derived ISRU propellants. The lunar regolith heat shield for Mars entry weighing 98 t would have to be transported from the lunar surface to LEO. It is assumed here that the heat shield does its own Earth aerobraking, so that a separate LEO aerobraking heat shield can be avoided.

By adding up the various components, then ISRU 4,326 kg + Dry LEO Ascent Stage & Dry Moon to LEO: Wet Transfer Stage from LEO 58,828 kg + Lunar Ascent & Moon to LEO Transfer Stage Dry 17,685 kg = 80,839 kg total IMLEO per mission for the regolith heat shield for Mars aerocapture. Comparing this to the IMLEO 41,000 kg baseline heat shield and transportation system mass that would be needed if it were not fabricated on the Moon, this results in a IMLEO deficit of 41,000 kg – 80,839 kg = -39,839 kg.

This analysis has shown that the additional Δv burden imposed by the gravity well of the Moon and the added mass of using the heavier regolith as TPS means that the lunar-made regolith heat shield for Mars is not a viable solution. Launching a heat shield from Earth and transporting it directly to Mars without a lunar fabrication intermediate step can minimize the IMLEO.

Architecture III: In Situ Heat Shield Fabrication at an Asteroid for Earth-bound Spacecraft

The mining of asteroids for resources of value is now being evaluated seriously thanks to the rapid advances of robotics and new launch capabilities. Several corporations funded privately have emerged to invest in the search of target asteroids and technologies needed to explore and mine them. The economic viability of such endeavors is still very much in question because so many known and unknown risks exist; our current knowledge of these objects remains fragmented and superficial and does not yet enable us to develop adequate technologies for landing on, mining and processing asteroids.

If a typical baseline near-Earth asteroid Δv of 5.5 km/s is assumed, then the mass savings associated with an in-situ produced heat shield can be calculated. It is assumed that the payloads that can be brought back from an asteroid must be reasonable so that in the event of a loss of control, the mining ore spacecraft would burn up in the Earth's atmosphere and not pose a hazard to the population. Therefore a payload mass of 7.7 t was chosen (similar to Orion), which could be returned with a 5 m diameter heat shield made of asteroidal regolith attached to the transfer stage. Assuming that the 5 m heat shield has a mass of 1,247 kg as in the Orion lunar example above, then the return propellant for the ore with the heat shield payload would also have to be factored in (with the use of aerobraking at 50% Delta V savings, Cooper & Arnold, 1990) and amounts to a mass cost of a wet transfer stage of 8,487 kg and in order to transfer this return stage with an Earth fabricated heat shield from LEO to the Asteroid, it would require a LEO – Asteroid transfer stage of 26,558 kg IMLEO. The total baseline comparison mass for a mission with Orion-type heat shield returning 8.8 t of ore would then be LEO to Asteroid stage 26,558 kg + Asteroid to LEO aerocapture 8,487 kg + Heat Shield 1,247 kg = 36,292 kg IMLEO per mission. At $5,000 per kg launched to LEO this amounts to a cost of $181.5 M, which means the market value of the ore must exceed $23,566 / kg to produce a profit. Since the mining equipment delivery cost is not included here, the actual asteroid mining cost would be even higher. Since the current market price of platinum (one of the most valuable metals on the market) is approximately $50,000 / kg, this means that a rough profit of $26,434 / kg could be produced. With a LEO payload of 7,700 kg, this translates into potential profits of $203.5 M per mining mission, although the cost of purification and return to the Earth's surface would still have to be subtracted from this margin. However this rough estimate shows that mining an asteroid for platinum is potentially lucrative.

The heat shield can be fabricated at the asteroid instead. In this case the ISRU heat shield fabrication equipment would have to be transported to the asteroid and it is assumed that the propellant is brought from Earth, since a metallic-type asteroid may not have water ice present for ISRU propellant production. Alternatively, oxygen can still be produced from such asteroid by using ISRU processes that reduce the metal oxides in the ore such as Molten Regolith Electrolysis (Sibille et.al. 2009)(Sibille & Dominguez 2012), hydrogen- (Clark et.al. 2009) and carbon- reduction (Gustafson et.al. 2010) of the regolith. These processing approaches would certainly be part of the metal extraction schemes to obtain the targeted metals. It thus makes sense that the oxygen released from the metals be harvested and used as a monopropellant in ion engines for example. The wet mass of the ISRU fabrication equipment transfer stage is 2,729 kg and it carries 5,000 kg of ISRU fabrication equipment so that the total IMLEO for this asteroidal heat shield fabrication capability would be transfer stage mass 13,642 kg + ISRU 5,000 kg = 18,642 kg IMLEO. If this was amortized over 50 missions it would be reduced to 372 kg per mission.

In addition, the mined ore of 7,700 kg + 2,300 kg regolith heat shield would have to be transported back from the asteroid to LEO for processing at an orbital outpost prior to returning the high grade ore to the Earth as explained above. This would require a wet mass transfer stage of 9,521 kg and in order to transfer this return stage to the asteroid it would require a LEO – Asteroid transfer stage of 25,980 kg IMLEO. The total mass for an asteroidal regolith-derived heat shield mission would then be 372 kg + 9,521 kg + 25,980 kg = 35,873 kg IMLEO per mission.

By comparing these two cases it can be seen that the net savings per mission amount to 36,292 kg - 35,873 kg = 419 kg IMLEO. The bulk of the mass being transported is propellant, and since the regolith derived heat shield is heavier, then the benefits of ISRU fabrication are somewhat more profitable (~$2M) versus sending an Earth fabricated heat shield directly, but the risk is higher. If an asteroid could be found with good ore and water ice for propellants then the benefits of ISRU would be far greater, since in-situ propellants could be used which would avoid sending the return journey propellant at substantial mass savings

If the asteroid did contain water for propellants or a nearby asteroid did, then the total IMLEO would be ISRU 745 kg + Dry Return Stage 865 kg + LEO to Asteroid Wet Transfer Stage 2,363 kg = 3,973 kg. By using ISRU propellants at the asteroid, the regolith-derived heat shield and ISRU become viable with a mass savings of 36,292 kg - 3,973 = 32,319 kg IMLEO that translates into $161.6 M per mission cost avoidance at $5,000/kg launched to LEO.

Now the economic case becomes much more attractive, since 3,973 kg IMLEO costs $19.9 M to launch. This corresponds to a mining cost of $2,580 / kg. With an assumed market price of $50,000 /kg of platinum ore, for example, then the profit margin is $47,420/kg. With a LEO payload of 7,700 kg this yields rough earnings of $365 M in LEO per mission or $162 M higher than without ISRU propellants and in-situ heat shield fabrication.

Regolith-based Heat Shield In-Space Fabrication Concepts

Three types of fabrication methods for several heat shield formulations were investigated. One method is to sinter the native granular regolith to form a solid mass. Sintering is accomplished at temperatures high enough for the particles that make up the regolith to begin to fuse or stick together without fully melting. The resulting solid body has a lighter density compared to the density of a full melt. The second method to fabricate the heat shield is to use the post-processing hot regolith from an ISRU reactor. Several reactor designs involve processes to remove oxygen from regolith that result in a by-product stream of very hot if not melted material similar in composition to the original regolith. The ISRU by-product material stream can then be used to create sintered materials from the regolith also so this method was combined with the sintering method as one of the potential ways to effect sintering. We discarded the use of fully melted stream materials because their higher density, higher thermal conductivity and generally weaker mechanical strength once solidified make them less desirable candidates for heat shield applications. The third method is to bind the regolith particles together using a high-temperature RTV. In this case, the goal is achieve a ratio of binder to regolith low enough to limit the mass of binder needed while creating a mixture that can be cast in desired shapes and perform as heat shield components.

The density of the resulting heat shield material is of significant consequence to its viability. If the density is too light, the heat shield will not withstand the stresses of atmospheric entry. However, if the material is too dense, its thermal conductivity will be too high to adequately protect the vehicle or payload.

A. Sintering Method

Two regolith simulants were used in the sintering experiments, Hogue et.al.. They are: JSC-1 Mars and JSC-1AC Lunar. These simulant materials were selected for their availability and on the basis of their representation of relevant characteristics of lunar and Martian regolith for this work. Although JSC-1 Mars is not considered a good simulant of any particular soil material found on Mars, it approximates the silicate and iron oxide content of such materials. JSC-1A is standard simulant produced to reflect the chemical and mineral composition of lunar basalts and JSC-1AC includes the wider particle size distribution that approximate these natural materials before they undergo any processing. The sintering process was refined for temperature and time to produce the samples used in flame and arc jet testing. To ensure a good fidelity test, a sample shape was designed based on similar 4" diameter by 2" thick samples (iso-q) designed for tests in the arc jet. For the sintered samples, Fondu Fyre™ molds were made.

B. RTV Binding

The RTV used is a two part silicone RTV formulated to retain its elastomeric properties at high operating temperatures. It also retains flexibility to low temperatures. For the RTV bound samples, plastic molds were made from a polymer via 3D machining methods. Examples of the iso-q molds are shown in Fig. 2 for the sintered and RTV bound test samples.

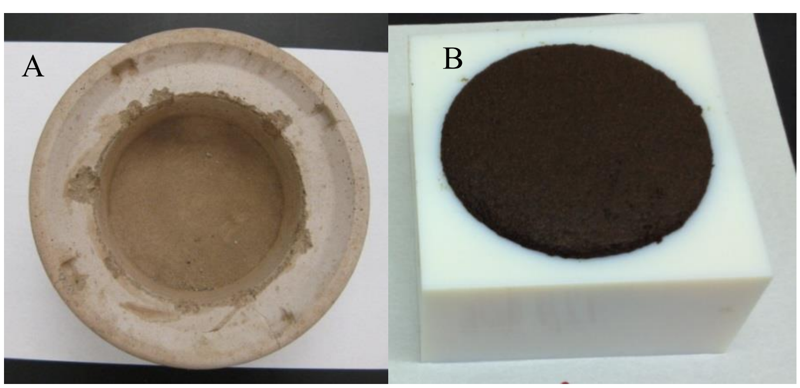

Figure 2. A. Fondu Fyre mold for sintered samples. B. Polymer mold showing a JSC-1 Mars simulant/RTV sample as it cures.

High Temperature Flame Impingement Tests

Two sintered and six RTV bound samples were fabricated for internal KSC flame testing and are listed in Table 1.

Table 1 Samples for KSC Internal Testing

Simulant	Composition
JSC-1AC Lunar	Sintered
JSC-1AC Lunar	Sintered
JSC-1A Lunar	RTV Bound
JSC-1A Lunar	RTV Bound
JSC-1A Lunar	RTV Bound
JSC-1 Mars	RTV Bound
JSC-1 Mars	RTV Bound
JSC-1 Mars	RTV Bound

To evaluate the viability of the initial test sample formulations, it was decided to expose the samples to the heat of an acetylene cutting torch for five minutes. This exposure represents the longest planned exposure in the Arc Jet Complex tests and approximates the heat load and some ablation that will be placed upon the samples in the Arc jet facility. The torch was positioned approximately six inches from the front surface of the sample. The flame temperature estimated at that distance from the torch nozzle is approximately 2200º C, (Perez 2012).

Each sample was placed horizontally on a large welding table supported by Fondu Fyre™ bricks. To measure rear temperature, a type K thermocouple was attached to be back side of each sample. The output of the thermocouple was read via an Omega Omnical. To measure the front temperature, a Fluke model Ti 32 IR camera was used. While rear temperatures could be measured throughout the test, the Fluke IR camera was limited to readings up to 620ºC so these readings were taken during the cooling period immediately following the five-minute torch impingement was completed. Each test run was recorded via a video camera. Front and rear temperatures were recorded via the above-mentioned devices for eight minutes after torch termination. The test set up is shown in Fig. 3.

In all of eight test runs, the maximum rear surface temperatures did not exceed unacceptable levels. The maximum temperature was measured a few minutes after flame test termination. This indicates that no significant heat was transmitted to the rear of the samples during the test. Post-test condition of the front surfaces of the samples is shown in Fig. 4. An example of the temperature data is shown in Fig. 5.

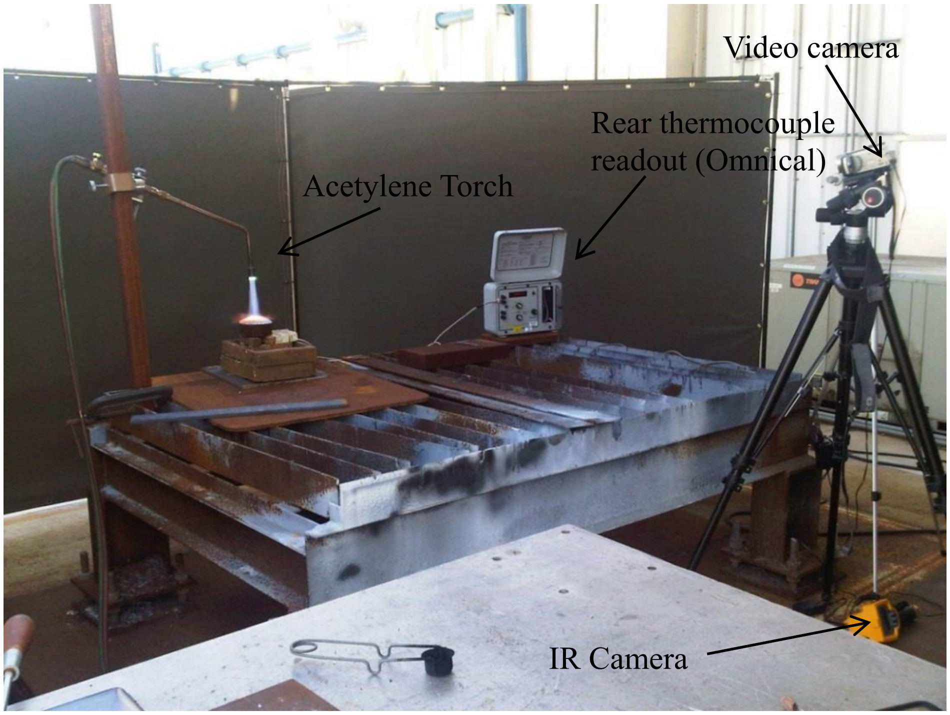

Figure 3. High temperature flame test set-up at the ESC Weld Shop (LETF) at KSC showing a heat shield material sample under test.

Figure 4. Post-test photo of the front surfaces of the eight flame test samples. The two JSC-1AC Lunar sintered samples are on the left-most column. The remaining samples on the top row are: RTV/JSC-1 Mars, RTV/JSC-1A Lunar, RTV/JSC-1 Mars. On the bottom row: RTV/JSC-1A Lunar, RTV/JSC-1 Mars, RTV/JSC-1A Lunar

Arc Jet Testing at Ames Research Center

The IHF Arc Jet test facility at ARC can expose test samples to a low pressure but very hot Argon and air plasma stream that models the thermal and dynamic conditions of atmospheric entry.

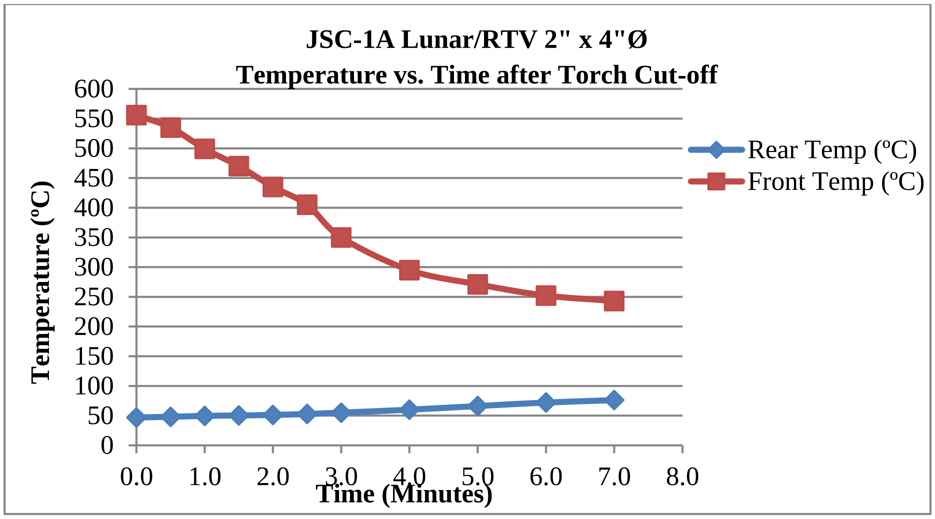

Figure 5. High Temperature Flame test temperature data acquired for the RTV/JSC-1A lunar sample. Measurements start at the beginning of the material's cooling period after termination of the exposure to the flame.

Ten of the 4 inch-diameter by 2 inch-thickness samples were fabricated for this testing. Sintering of the JSC-1 Mars simulant resulted in cracking and excessive shrinkage so these samples were not used. The three sintered samples sent for arc jet testing were made from JSC-1AC lunar simulant. The remaining seven samples were RTV bound formulations of both JSC-1A Lunar and JSC-1 Mars regolith simulants. Table 2 summarizes the selection of the ten Arc jet samples tested at the IHF Arc Jet Facility.

Two test conditions were created in the IHF arc jet tests. Condition 1 was set to a heat flux of 48 W/cm^2 and condition 2 was set to 92 W/cm^2. The test protocol for the samples is given in Table 3, (Hogue & Sibille 2012). There are three arms called stings in the IHF: East, west, and overhead. Each arm/sample assembly can be sequentially rotated to position the center line of the sample in the center of the plasma jet.

Table 2 Arc Jet Test Samples*

Simulant	Formulation
JSC-1AC Lunar	Sintered "A"
JSC-1AC Lunar	Sintered "B"
JSC-1AC Lunar	Sintered "C"
JSC-1A Lunar	RTV
JSC-1A Lunar	RTV
JSC-1A Lunar	RTV
JSC-1A Lunar	RTV
JSC-1 Mars	RTV
JSC-1 Mars	RTV
JSC-1 Mars	RTV

Four test runs were performed over a period of two days. The calorimeter mentioned in Table 5 was used to set the heat flux (W/cm^2) and other system parameters for each test condition. The calorimeter consists of a copper iso-q model instrumented with thermocouples (TC) and was placed on one of the stings. Readings from the calorimeter allowed IHF personnel to tune the Arc Jet energy to the correct range for each condition. Test condition 2 is equivalent to the heating experienced by a space shuttle during re-entry of the Earth's atmosphere.

Front surface temperature of the samples was measured by three pyrometers. The pyrometers were requested to cover a surface temperature measurement range of 600-2500°C. The pyrometers were directed at the center of the each sample during and after the sample's exposure to the Arc Jet flow.

Table 3 Arc Jet Test Protocol (Terrazas-Salinas et.al. 2009, Hogue & Sibille 2012)

Run #	Condition ID	Sting	Test Model	Exposure (sec)	Model Instruments
1	1	OH	4" Slug Calorimeter	*	1 K TC
	1	E	JSC-1A Lunar/RTV	60	2 K TC
	1	W	JSC-1 Mars/RTV	90	2 K TC
2	1	OH	JSC-1 Mars/RTV	150	2 K TC
	1	E	JSC-1A Lunar/RTV	240	2 K TC
	1	W	JSC-1AC Lunar Sintered "C"	300	2 K TC
3	2	OH	4" Slug Calorimeter	*	1 K TC
	2	E	JSC-1A Lunar/RTV	60	2 K TC
	2	W	JSC-1 Mars/RTV	90	2 K TC
4	2	OH	JSC-1AC Lunar Sintered "A"	150	2 K TC
	2	E	JSC-1A Lunar/RTV	240	2 K TC
	2	W	JSC-1AC Lunar Sintered "B"	300	2 K TC

* Calibration exposure durations were per the discretion of the IHF test engineer.

An infrared (IR) camera was focused to view the entire exposed face of the test article. All pyrometers and the IR camera were calibrated with a blackbody source. Also, a video camera was used to record each test. Pre and post-test still photos of the samples were taken. To determine mass loss, each sample assembly (sample, back plate, thermocouples) was weighed both before and after the test. Photos of several samples under test in the arc jet are shown in Figs. 6-8.

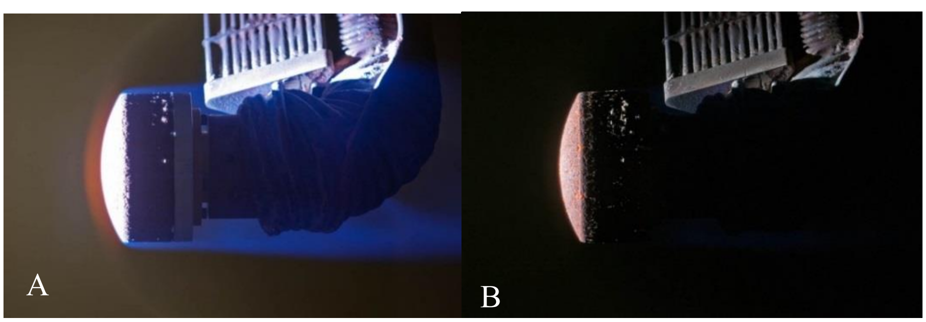

Figure 6. A. JSC-1 Mars/RTV at the start of run. B. The same sample at the end of a 150 second exposure at 48 W/cm^2 showing little or no melting or ablation.

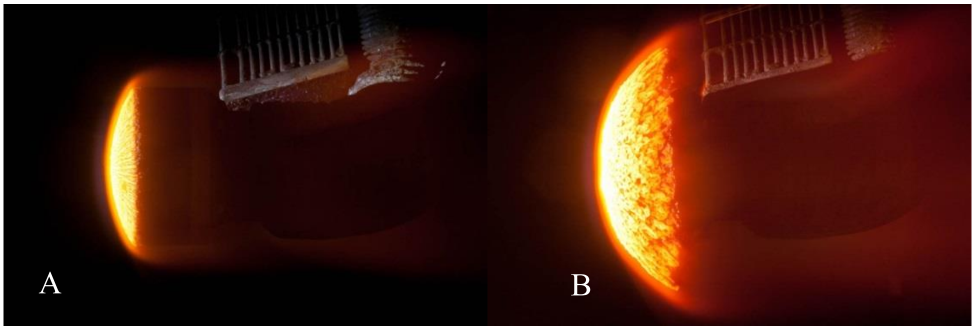

Figure 7. A. JSC-1A Lunar/RTV at the start of run. B. The same sample at the end of a 240 second exposure at 92 W/cm^2 showing melting and flow of its surface.

The highest recorded rear temperature (~ 260° C) was for sample JSC-1AC Lunar sintered "B" which was tested at condition 2 for 300 seconds (Fig. 8). While melting deformed and ablated most of the sample, it displayed sufficient thermal protection on its rear surface.

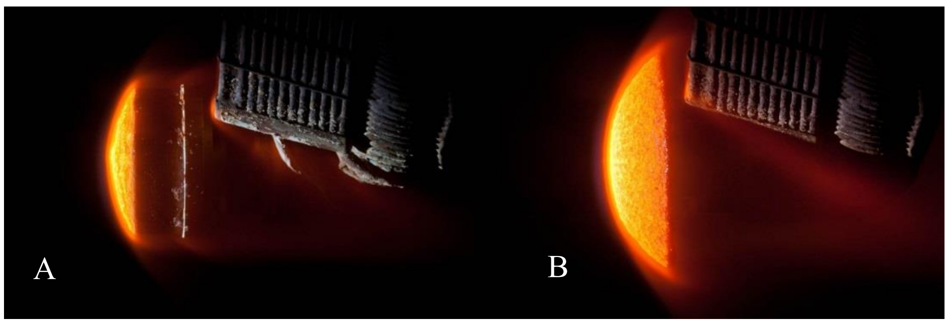

Figure 8. A. JSC-1AC Lunar Sintered "B" at the start of run. B. The same sample at the end of a 300 second exposure at 92 W/cm^2 showing melting and flow of its surface.

Mass loss for each of the sample assemblies (regolith sample, Aluminum back plate, thermocouples, bonding RTV) was acceptable. Except for the longer exposure times at condition 2 (92 W/cm^2) we measured little appreciable mass loss. Sample JSC-1AC sintered "B" displayed the most observable ablation and change. The melted regolith flowed over the surface of the sample and formed an approximately one inch wide and 3/8 inch thick glassy crust along the edges which was very brittle and broke off after the test. Pre-test and post-test photos of this sample are shown in Fig. 9. Also shown in Fig. 22 is a side photo of sample JSC-1AC sintered "A" showing a similar glassy ring. This ring, the result of surface melting and flow of molten regolith to the edges of the sample also changed the overall shape to a larger radius and enlarged the forward diameter by about two inches. Sample JSC-1AC lunar sintered "B" also had the highest peak rear temperature recorded at 1237.7 seconds (987.7 seconds after test end).

Summary and Conclusions

Within the scope of the testing to date, the feasibility of using extraterrestrial regoliths as the construction material for atmospheric entry heat shields has been confirmed from the results of the acetylene flame and arc jet testing.

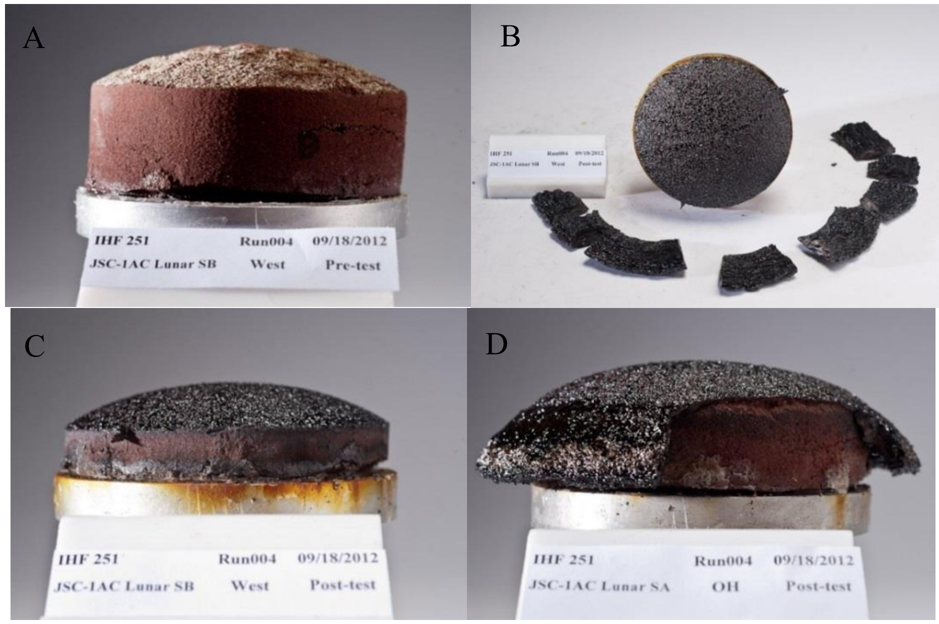

Figure 9. A. Pre-test side view of sample JSC-1AC sintered "B". B. Post-test photo showing glassy ring. C. Post-test side view showing loss of thickness in Sintered "B". D. Side photo of sample JSC-1AC sintered "A" showing an example of the glassy ring and its cross-section.

While some of the arc jet-tested samples were heavily ablated, they provided adequate low temperatures on their rear surfaces. These rear surface peak temperatures were recorded several minutes after arc jet test termination.

While the highest energy input (92 W/cm^2) at a five minute duration was comparable to a space shuttle re-entry from low Earth orbit, interplanetary atmospheric entry energies can be on the order of about 300 W/cm^2 or higher, (Laub & Venkatapathy 2003). For these type of atmospheric entries, a much thicker regolith derived heat shield would be required than the two inch thick samples evaluated. If the heat shield is fabricated on Phobos or an asteroid, where there is little gravity, then fairly large heat shields can be used to protect returning payloads to Earth.

A number of sintered and RTV bound formulations were evaluated. However, samples of JSC-1 Mars with the larger RTV concentrations performed well. In future work, JSC-1 Mars will be replaced by a more representative simulant material using materials that represent characteristics of Phobos and Deimos materials instead. Other lunar material simulants that reflect anorthosite-rich highlands materials are also planned.

Much mission architecture work still needs to be performed to determine the full cost/benefit of using regoliths as atmospheric heat shield material. Cost savings for a 20-mission Mars campaign (10 unmanned and 10 manned missions) are estimated to be about $35 billion dollars if the massive heat shields for each mission did not have to be transported from the surface of the Earth to Mars.

Acknowledgements

The authors would like to acknowledge the support of NASA's NIAC program, the Office of the Chief Technologist, and the NASA KSC Space Technology Office. We are grateful for the assistance of NASA KSC NE project management office. Special thanks go to Mr. David Early and Mr. Ivan Townsend from Team QNA-ESC, Kennedy Space Center, FL for their work on the acetylene torch tests. Also, we give special thanks to Mr. Enrique J. Carballo, IHF Lead Test Engineer, Ms. Imelda Terrazas-Salinas/ARC-TSF, and the ARC IHF personnel who successfully performed the arc jet tests.

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