

# Exploration Space Suit Architecture and Destination Environmental-Based Technology Development

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**This paper continues forward where EVA Space Suit Architecture: Low Earth Orbit Vs. Moon Vs. Mars<sup>1</sup> left off in the development of a space suit architecture that is modular in design and could be reconfigured prior to launch or during any given mission depending on the tasks or destination. This paper addresses the space suit system architecture and technologies required based on human exploration (EVA) destinations, and describes how these systems should evolve to meet the future exploration EVA needs of the US human space flight program. A series of exercises and analyses provided a strong indication that the Constellation Program space suit architecture, with its maximum reuse of technology and functionality across a range of mission profiles and destinations, is postured to provide a viable solution for future space exploration missions. The destination environmental analysis demonstrates that the modular architecture approach could provide the lowest mass and mission cost for the protection of the crew, given any human mission outside of low-Earth orbit. Additionally, some of the high-level trades presented here provide a review of the environmental and non-environmental design drivers that will become increasingly important as humans venture farther from Earth. This paper demonstrates a logical clustering of destination design environments that allows a focused approach to technology prioritization, development, and design that will maximize the return on investment, largely independent of any particular design reference mission.**

## Nomenclature

<i>AU</i>	=	Astronomical Unit
<i>CSSE</i>	=	Constellation Space Suit Element
<i>CxP</i>	=	NASA's Constellation Program
<i>DRM</i>	=	design reference mission
<i>EMU</i>	=	Extravehicular Mobility Unit
<i>EPOXI</i>	=	Extra-solar Planet Observation and Deep Impact Extended Investigation
<i>EVA</i>	=	extravehicular activity
<i>GCR</i>	=	galactic cosmic ray
<i>GEO</i>	=	geostationary Earth orbit
<i>ISS</i>	=	International Space Station
<i>LEO</i>	=	low-Earth orbit

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<i>MLI</i>	=	multilayer insulation
<i>NCRP</i>	=	National Council for Radiation Protection
<i>NEA</i>	=	near-Earth asteroid
<i>NEO</i>	=	near-Earth object
<i>REID</i>	=	risk of exposure-induced death
<i>SPE</i>	=	solar particle event
<i>STEREO</i>	=	Solar TERrestrial RELations Observatory
<i>TMG</i>	=	thermal micrometeoroid protection garment

## I. Introduction: Destinations for Human Exploration

In looking forward to the future of human space exploration, it is important to first consider the possible destinations that humans can realistically travel to, survive in, and possibly live in for extended periods of time with reasonable resources and budget. For example, it can be assumed with some level of confidence that there will be no crewed missions to Mercury due to the required infrastructure, logistics train, and rocket design needed to climb into and out of the inner gravity well of the sun. However, it is reasonable to consider visitation of the Earth-sun libration points. In following this line of thought, and by using current knowledge of the physical environments of destinations in the solar system from which one can return in a decade or less, one can quickly identify the destination design drivers required for exploration-class space suits.

Historically, technology development for human space exploration primarily did not happen until the mission was defined and funded or was done at the component level in efforts to improve existing systems. Low technology readiness level technology development for pursuing advanced concepts has also been limited. The logic in this is understood, given that humans only started venturing beyond the relatively benign environment of Earth in the last 50 years and had little idea of what might be encountered at each destination. Today, however, this approach should be questioned. Humans have either physically stepped on, landed robotic probes, placed orbital vehicles around, or had close fly-bys of every significant body in the solar system – with the exception of Pluto. Now, with the volumes of data growing at a near-geometric rate, the knowledge of the environments in which humans can venture is understood to the point where common design drivers and required design elements can be identified with reasonable confidence. Given this knowledge of the environments and lessons learned from human space flight operations to date, NASA conducted an internal assessment (performed within the Space Suit and Crew Survival Systems Branch at the Johnson Space Center) of the progress that has been made in human exploration space suit technology with respect to the “design space” proposed in this paper.

The following pages address the methodical approach to common and probable destination environments, and how this should affect the prioritization of space suit technology development in the future.

## II. Overview of a Flexible Space Suit Architecture

The space suit architecture developed by NASA’s Constellation Space Suit Element (CSSE) only addressed crew survival, low Earth orbital operations, and lunar surface extravehicular activities (EVAs); however, at the very core, this architecture had many, if not all, key design-driving elements that will be required for human exploration in the solar system. The CSSE team\* addressed this challenge by fully embracing “clean-sheet” design approach and “textbook” systems engineering methodology by first defining the operational concepts, which focused on the development of an architecture with all Constellation Program (CxP) design reference missions (DRMs), and by keeping an eye on life cycle program costs. A comprehensive review of the functional designs, strengths, and limitations of previous US space suits, in addition to what is known of Russian space suits, took place to deduce historical lessons learned based not only on what did not work but, more importantly, on what worked correctly. The goal set forth by the CxP – to accomplish the daunting task of meeting all space suit design requirements in the extreme environments previously detailed with a single system – hinges on an arrangement that not only uses common hardware across multiple mission phases (to reduce developmental and logistics costs), but also features an open architecture that could be reconfigured and can leverage off components used during other mission phases, where possible.<sup>1</sup>

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\* This team was comprised of NASA civil servants and support contractor workforce with the responsibility of defining CxP space suit architecture and associated functional requirements and to later become the NASA oversight and subsystem managers.

The key design figures of merit for the CxP Space Suit were used in evaluating all of the different architectures, some of which later became architecture design drivers: operational performance; work efficiency; launch, entry, and abort overhead; suit attributed mass and volume; field maintenance; commonality (design and hardware); extensibility; technical risk/feasibility; life cycle costs; and development schedule risk. The following were the CxP suit performance criteria that defined the high-level functional requirements for the suit architecture: intravehicular mobility; microgravity mobility; microgravity environmental protection (thermal, radiation, micrometeoroid protection); comfort (un-/pressurized); ease of donning and doffing; crew ability to escape the vehicles while wearing the suit; suit sizing methodology; ability of the suit to have sizing adjustments; high operational reliability; high evolvability and adaptability; extraterrestrial surface mobility; and extraterrestrial surface environmental protection

After 5 years and multiple design iterations, the CSSE suit architecture consisted of the following modular, or swap-able (from one configuration to another), hardware elements: helmet bubble and communications cap; gloves optimized for pressurized usage; boots optimized for 1g vehicle escape; lower arms and legs with mobility joints and umbilical connectors; and restraint mechanisms that are common in design. The fire protection outer cover layer and EVA thermal multilayer insulation (MLI)/thermal micrometeoroid protection garment (TMG)<sup>†</sup> were unique enough to discrete mission phases that it was felt they would not be included functionally in the modular hardware so as to reduce the overhead of carrying around hardware for infrequent use or as bad-day risk mitigation. The outer layer of the TMG is not only fire resistant, it provided low emissivity for reflecting solar radiation – thus the white coloration – as well as cut, puncture, and abrasion resistance. The outer layer of the MLI/TMG might require a different coloration on Mars to meet the emissivity requirements for that environment. In addition, the thin atmosphere of Mars would drive a departure from traditional TMG materials to the use of a high-performance conductive pathway insulate such as aerogel.

With the maturity of human space exploration still in its infancy, and with limited resources to apply to development, a flexible environmental protection suit architecture that would minimize the cost by decreasing the development cost per mission is desirable. However, with a system that operates in varying environments, there is the risk that performance in specific environments would be compromised. Specialized suits and hardware will be warranted as experience in specific environments grows or human habitation becomes more permanent,

Additionally, the CxP portable life support system would be used only on the lunar surface as life support functions are provided by the vehicle when the crew is inside or while performing microgravity EVAs. And, the core torso segment, which was optimized for 8-hour surface EVAs, would be swapped with the all-soft segment used for launch and landing. Joseph Kosmo of NASA recommended a similar design philosophy in 1990,<sup>2</sup> prior to the CxP space suit design effort.

#### **A. How This Works Well for the Different Destinations for Human Exploration**

The fundamental plan was for the CxP to evolve from microgravity to lunar exploration with sortie and long-duration habitation, and to progress eventually to Mars exploration.<sup>3</sup>

The common themes on how the CSSE suit architecture would be used for the CxP resonate with the possible design reference missions being discussed today for future human exploration. At the most fundamental level, every human launch will need to provide protection for the crew against a bad day on the launch pad as well as during a launch abort scenario and protection while reentering the atmosphere on mission completion. However, due to the divergent launch, entry, and abort environmental design consideration from those of EVA, these common themes will not be addressed in this paper.

Each mission will require either a planned microgravity EVA or the capability to perform contingency EVAs in the event the vehicle leaks or other hardware malfunctions require mitigation – particularly during missions with long transit durations. And, if the launch mass of the required space suit hardware is a limiting design constraint, it is highly desirable for these future missions to be able to reconfigure a suit to meet the different needs of the crew (to save mass and volume) and not be required to carry multiple suits per crew member.

Additionally, multi-program life cycle costs and return on investment in technology development can be realized in this approach by designing to the architecture interfaces and only performing multiple designs for the hardware specifically required for the unique environments.

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<sup>†</sup> Consists of Ortho 116 fabric, five layers of aluminized Mylar, and one layer of neoprene-coated nylon. Whereas the neoprene and nylon will burn in air at ambient pressure, the NASA White Sands Test Facility rating of A for the outer layer of Ortho 116 means that the material has a burn length of less than 6 inches for ambient pressure to 10.2 psi with oxygen concentrations of 30%.

### III. Methodology of Destination Environment Design Groupings

#### A. Determination of the Design Drivers

The formulation in a new way of prioritizing technology development efforts for space suits began with the president's new vision for NASA in February 2010 to develop the technology that would be required to make long-duration space exploration more successful when we, as a nation, were ready to step out into space for good.

A different approach was taken for this study. This approach included an inventory of possible destinations in the solar system that humans could reasonably explore, given the likely technology developments in the next 30 years regarding launch vehicle, engine, closed-loop life support systems, and subsequent durations of missions and space suit technology. When the destination list was complete, the subsequent environments and characteristics were assessed and grouped for commonality.

#### B. Destination Design Environments

The environments of the destination locations will be briefly discussed in the sections to follow; however, they will not be discussed in great detail as the individual environments have been documented in the source materials referenced. It is worth mentioning that for this exercise, and to a large extent in space suit design, the exact numbers for environmental design drivers are not critical discriminators in the first-order design of the system. For example, whether the local vacuum of space is  $1 \times 10^{-5}$  torr or  $1 \times 10^{-13}$  torr, for a suit pressurized to 4 lbs/in<sup>2</sup>, is of marginal consequence. The same can be said for designing a suit to tolerate a touch temperature of -125 °C (-193°F) or -148°C (-234.4) in which the design challenge is largely the same and may only impact final material selection or second-order suit heater impacts to the power budget. The specific environmental values that are used for this study, along with rationale and references, are summarized in Table A-1 in Appendix A. The major suit design drivers for suit development will be summarized in each section.

#### C. Low-Earth Orbit Operations (International Space Station, low-Earth orbit satellites)

The low-Earth orbit (LEO) microgravity environment, which is the most familiar in human exploration, is where the largest amount of experience in performing human EVA operations has taken place in the last 50 years. The environment is thus well understood. The local gravitational acceleration, while in the gravity well of Earth, places an object in a state of orbital free fall and, therefore, will be quantified on the order of micro-g's. Additionally, the atmospheric drag at the altitude at which most Space Shuttle missions and International Space Station (ISS) operations take place will be considered negligible with regard to space suit design. The radiation environment, which is greater than the environment to which high-altitude pilots are exposed due to lack of an atmosphere, still resides within the Earth's Van Allen belts.

The amount of radiation that crew members are exposed to in LEO is directly related to orbital inclination. At higher inclinations, spacecrafts are exposed to greater levels of radiation as they actually pass through the Van Allen belts as they connect to the Earth's magnetic poles. Most human spaceflight to date has taken place in inclinations less than 51.5 degrees (orbital inclination of Mir), which is a high enough inclination to be a concern for radiation exposure during parts of the orbit near the magnetic poles. In this paper, an assumption has been made that the orbital inclination of any near-term human spaceflight will be 51.1 degrees (orbital inclination of the ISS) or less. Any missions in LEO with an inclination higher than 51.1 degrees would need to account for increased levels of radiation near the Earth's poles.

The exception is for a region above the Earth known as the South Atlantic Anomaly<sup>‡</sup> where potential EVAs performed<sup>29,30</sup> while the passing through this region are limited to three to five passes for any particular crew member before they are rescheduled; however, the actual limitation is defined by the personal accumulated radiation dosage that is tracked for the mission and for life of the crew member. The current flight rules<sup>16</sup> for EVA radiation exposure state specifically:

- A. For predicted exposure less than the action level (non-restrictive)
  1. Consider delaying the EVA up to 2 days, or delaying or accelerating egress one to two revs if this will reduce the exposure while accomplishing mission objectives consistent with normal crew ground rules and constraints.

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<sup>‡</sup> The South Atlantic Anomaly is a result of the Earth's magnetic field and is not completely symmetric and aligned with the Earth's surface, and thus allows a portion of the solar (particle) flux to extend down through LEO and affects communication with satellites, the Hubble Space Telescope, high-altitude aircraft, and the Space Shuttle.

2. An EVA in progress will continue. Consider not adding unscheduled items to existing timeline if this results in additional EVA crew exposure.
- B. Predicted crew exposure greater than the action level at the end of the EVA (restricted)
1. Delay EVA up to 14 days if still possible to accomplish mission objectives, or delay or accelerate egress one to two revs.
  2. An EVA in progress will continue. Consider expediting tasks not required for primary mission objectives.
- C. Predicted crew exposure greater than the high dose rate limit (high dose rate limits)
1. A planned EVA shall be rescheduled as required to reduce the exposure to below the high dose rate limit.
  2. An EVA in progress shall be expedited by deleting tasks not required for primary mission objectives.
- D. Predicted crew exposure greater than the joint exposure limits
1. A planned EVA shall be rescheduled as required to reduce the crew member's mission exposure to below the joint exposure limit.
  2. An EVA in progress shall be terminated.

Additionally, the National Council for Radiation Protection (NCRP) created a set of career exposure limits, shown in Table 1,<sup>31</sup> for astronauts in LEO. These limits are based on a one-percent risk of exposure-induced death (REID) for the type of radiation experienced in LEO. Longer-duration missions outside of LEO may need to reevaluate these limits based on trade-offs between the resulting shielding mass and accepting a higher REID.

**Table 1. NCRP-132 LEO Exposure Limits<sup>31</sup>**

<b>LEO Career Whole Body Effective Dose Limits (Sv)</b>				
<b>Age</b>	25	35	45	55
<b>Male</b>	0.7	1.0	1.5	3.0
<b>Female</b>	0.4	0.6	0.9	1.7

The solar wind is still a nontrivial influence in this environment. With most of the EVAs performed to date – and likely in the future – being around human-made structures, the effects of the interaction between the solar wind and large metallic structures (or solar panel elements), plasma generation, and conductance is an increasing safety concern in the community.

Therefore, the design drivers for this environment will be: pressurized suit mobility in microgravity; life support consumables in a vacuum; thermal exposure and management in a vacuum around human-made structures; and plasma charging fields.

#### **D. Geostationary Earth Orbit Operations**

Probable future human activity in this region will consist of retrieving, repairing, refueling, or deploying geostationary satellites (35,786 km [22,236 miles] above the Earth's surface) or multi-day experimental missions. This environment is very similar to that of the LEO missions but with the significant difference of being within or outside the Earth's Van Allen belts for some or most of the time.<sup>§</sup> Consequently, the effects of the solar wind – and, to a lesser extent, cosmic radiation – are elevated due to direct exposure from the sun or the concentration of geomagnetically trapped radiation (electron and proton) in Earth's magnetic fields. The inner Van Allen belt extends from an altitude of 1000 to 10,000 km (621.4 to 6,213.7 miles) above the Earth's surface (the South Atlantic Anomaly is a result of the inner proton belts dipping down as low as 220 km [137 miles]), and the large outer radiation belt extends from an altitude of about 19,113 to 44,597 km (11,876.3 to 27,711.3 miles) above the Earth's surface.<sup>17,18</sup>

<sup>§</sup> While the geostationary orbit is above the inner Van Allen belt, it can reside inside or outside of the outer belt due to the compression of the outer belt on the side of the Earth facing the sun and the pressure of the solar wind. Therefore, at times, the satellite might be on the outer edges or outside the belt, depending on the relative position with respect to the sun and current solar activity levels.

The actual radiation levels experienced in geostationary Earth orbit (GEO) can increase significantly based on several factors, including location in the orbit (day or night), solar cycle, and the occurrence of geomagnetic storms.<sup>32</sup> This is the only environment described in this paper; the high levels of radiation experienced in GEO would require significantly more radiation protection than the current Extravehicular Mobility Unit (EMU) for nominal usage.

The duration of such missions would not be expected to exceed a 1- to 2-week duration; therefore, the time element of the design would not be considered a driver. The environmental design drivers for this region would be pressurized suit mobility in microgravity, life support consumables in a vacuum, thermal exposure and management in a vacuum around human-made structures, plasma charging fields, and solar/cosmic/concentrated radiation effects.

#### **E. Lagrangian Points: Earth-Moon**

Interest in human missions to the Lagrangian,<sup>¶</sup> or libration, points in the Earth-moon system has increased in recent years. In a two-body gravitational system in circular orbit about one another (as is the case with the Earth-moon system), there are five regions in which the gravitational balance between the two bodies are in equilibrium and lend themselves well for placement of satellites, observatories, or rendezvous depots for space missions with minimal fuel consumables for positional station-keeping.

The duration of such missions is likely to exceed 3 months, with the potential for more than 1 year depending on the libration point; therefore, the time element of the design would be considered a driver. For the purposes of this paper, Earth-moon Lagrange points L1, L2, L4, and L5 were considered as potential exploration candidates. These locations were chosen based on distance and scientific and exploration interest. Sun-Earth Lagrange points were not considered, as the closest one is approximately five times farther than the moon but of little exploration interest. Most of the design drivers are similar to that of in-transit or GEO missions, even when considering somewhat interesting scenarios, such as constant sunlight/shadow, Halo orbits, or orbit maneuvers between Lagrange points and lower Lunar orbit.

#### **F. Interplanetary Contingency Extravehicular Activity Environment**

This classification, while more mission specific, does define a design environment. This environmental scenario is a catchall for the instances during a mission in which crew members are required to go outside the vehicle to either investigate, repair, or replace hardware associated with their vehicle. NASA's experience during the last 50 years of operations is that Murphy<sup>#</sup> is never far away and having the capability to perform unscheduled, or contingency, EVA is a critical capability for all missions. This environment is largely encompassed by the GEO environment in terms of vacuum, radiation, and plasma charging. However, the thermal environment will probably differ due to varying distances from the sun – for this paper, it is considered to be bounded by the NEO distance limits. Additionally, due to the fact that the probable mission duration (time away from Earth) will be anywhere from 3 months to 10 years, it is imperative that this time away be factored into the suit design, fabrication, and reliability engineering.

Other key factors to keep in mind during transit from one place to another are the amount of radiation crew members receive from low-level but constant galactic cosmic rays (GCRs) and large solar particle events (SPEs) from the sun. Without sufficient radiation shielding against GCRs and SPEs on the vehicle, a crew member may be exposed to a large percentage of his or her career radiation limits prior to reaching the destination.<sup>33</sup> The result would be either decreased EVA time at the destination or increased radiation protection on the suit. Designing transfer vehicles and habitats with sufficient radiation shielding therefore becomes critical to ensuring crew members are capable of performing multiple EVAs at their long-duration destination.

#### **G. Low-mass Near-Earth Object/Near-Earth Asteroid**

The suit design environment of low-mass near-Earth object (NEO)/near-Earth asteroid (NEA) EVAs is an interesting combination of the microgravity environment of LEO EVAs and that of the thermal and dust environment of the lunar EVAs (discussed later). This destination is associated with missions to NEO/NEA that are half the mass of the moon or smaller, and would have a local gravitational acceleration between microgravity and 0.817 m/s<sup>2</sup>, thus rendering normal human ambulation impossible. Based on previous analyses performed of

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<sup>¶</sup> The concept was first conceived by Joseph L. Euler around 1750 when he predicted the collinear points commonly known as L1, L2, and L3. Later, Luis Lagrange, in his work with two-body orbital mechanics, further predicted the existence of points L4 and L5; these points were all later named after Lagrange in his honor.

<sup>#</sup> Societal reference (Murphy's Law) for when something or a situation can go wrong, it will.

candidate NEO targets, all potential targets of opportunity are on the scale of 100m diameter,<sup>36</sup> and thus would result in gravity fields on the low side of that range.

The extremely low gravitational acceleration will require the use of attachment mechanisms (to the object being studied) and mobility aids to transverse the object. In some ways, the lack of meaningful gravity will affect the EVAs and how the crew performs tasks due to the fact that any dust generated/stirred up will likely hover in a cloud around the work site for indeterminable amounts of time and could potentially impact work site visibility, dust coverage of the suit, and dust mitigation strategies. If dust is present on these bodies – which, based on current research, is very likely – the characteristics of the dust (physical and chemical) are expected to fall within the range analyzed both from the moon and from recent studies of comets.\*\* In addition, the sparse selection of targets of opportunity between now and 2040 correlates to a wide range of locations<sup>36</sup> – for the purposes of this paper, the bounding case was considered 0.86AU and 1.3AU – which corresponds to a wide potential thermal range, as well.

## **H. Earth's Moon**

The lunar environment definition for suit design for the CxP encompassed the entire range of lunar extremes as defined by the goal of global access to the lunar surface with a single suit system.<sup>1,3</sup> The dust environment is a known variable, given the experience gained and information gathered as part of the surface EVAs and the dust and rock samples and space suit hardware returned from the Apollo Program.

As part of a “go anywhere, anytime” philosophy, suit engineers now had to consider the design impacts of suit(s) designs that would allow crew members to function in the permanently shadowed crater interiors at the lunar poles with cryogenic touch temperatures as well as the solar furnace-like environments of craters at the equator during lunar noon.

The surface of the moon presents some unique radiation challenges, as well. In addition to the GCRs and SPEs experienced at other locations described in this paper, the lunar regolith itself creates neutron albedo resulting from interactions between GCRs and the lunar regolith. This interaction results in crew members being exposed to an additional 0.123 mSv/day.

The gravity of the moon, while one-sixth that of Earth's, did provide mobility challenges to the Apollo crews since the pressurized suit design hindered natural human ambulation. Advances in space suit mobility elements since that time have significantly minimized the impact of low gravitational acceleration combined with suit pressurization to the design of a space suit. Therefore, design drivers for this environment, in addition to the mission durations of 2 weeks to 1 year and associated reliability design challenges, will be: pressurized suit mobility in reduced-gravity; life support consumables in a vacuum; thermal exposure and management in a vacuum at extreme temperatures; high-abrasion, very fine, and statically charged dust; and potential plasma charging fields.

As with the in-transit condition described above, habitats for any long-term stay on the lunar surface or elsewhere needs to provide sufficient radiation protection to ensure crew members' EVAs are not limited by career radiation limits.

## **I. Martian Moons: Phobos/Deimos Missions**

The environment of the Martian moons is expected to combine lunar dust characteristics with thermal extremes at vacuums that are no greater than those seen on Earth's moon, with the low gravitational acceleration challenges seen with the NEO/NEA EVA environment. As with the in-transit EVA environment, mission duration is expected to play a major component of the design driver challenges. Thermal aspects should be similar to the cold NEO/NEA case.

## **J. Martian Surface Missions**

The Martian surface environment, in many aspects, is the most benign of all those to be considered for human EVAs. The presence of the Martian atmosphere, albeit much less prominent than Earth's, does provide the mechanisms for wind erosion in addition to minimizing thermal extremes, solar wind protection, and some cosmic radiation shielding. Recent discoveries from NASA Martian rovers and orbiters indicate an ever-increasing evidence base for the past existence of liquid flowing on the surface. Between the flowing of liquid on the surface and the atmospheric erosion mechanisms, the Martian dust physical characteristics will be considered low abrasion, albeit more abrasive than what might be found on Earth. However, whereas knowledge of the chemical makeup of the Martian dust is limited, with the spectral information from the orbiting satellites and the spot analyses from the rovers, the generalized list of chemical makeup is growing.

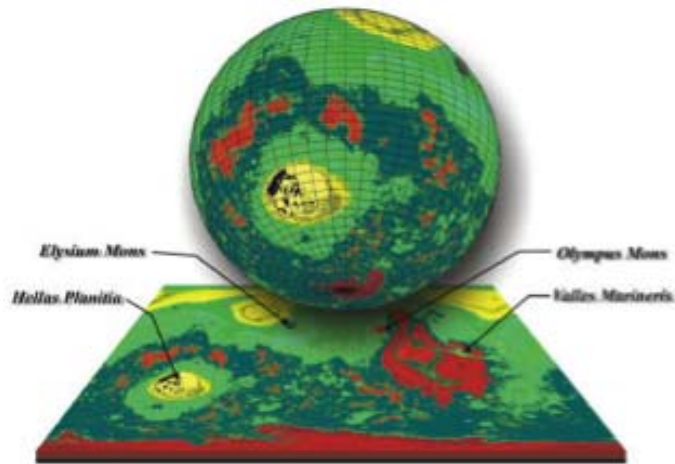
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\*\* Samples of comet Wild 2 returned by NASA's Stardust and data returned from the EPOXI (Extra-solar Planet Observation and Deep Impact Extended Investigation) spacecraft.

Although Mars does not contain a strong protective geomagnetic field like Earth, it does have a thin carbon dioxide atmosphere that would provide some radiation protection to crew members working and living on the surface.<sup>34</sup> The thickness of this atmosphere varies with altitude and location on the planet, as does the amount of radiation protection, as shown in Figure 1.

As stated before, habitats for long-term missions need to provide sufficient radiation protection to ensure crew members' EVAs are not limited by career radiation limits.

The current NASA design reference missions<sup>19,20,21</sup> indicate a probable mission duration upwards of 3.3 years. This poses quite a challenge for space suit engineers to design a suit that is highly operable, does not require frequent maintenance, is durable for significant usage at Mars, and is highly reliable – not requiring repair or replacement – during the mission. Additionally, some surface data from the Spirit Mars Exploration Rover indicate that the surface temperatures can vary from  $-23^{\circ}\text{C}$  ( $-9^{\circ}\text{F}$ ) to  $-90^{\circ}\text{C}$  ( $-130^{\circ}\text{F}$ ) diurnally from late summer to fall, respectively.<sup>22</sup> Some consideration may be required for long mission stays that may span seasons at some of the Martian latitudes.



**Figure 1: Predictions of Skin Dose-Equivalent as a Function of Position on Mars Surface During a Solar Minimum<sup>[34]</sup>**

Therefore, the design drivers for this environment, in addition to the mission durations of as many as 3.3 years and associated reliability design challenges, will be: pressurized suit mobility in reduced gravity; life support consumables in a rarified atmosphere; thermal exposure and management in a rarified atmosphere at cold to moderate temperatures; and low-abrasion, very fine, and potentially chemically reactive dust.

## **IV. Mapping Destination Environments to Mutually Inclusive Design Drivers**

### **A. Destination Mapping Phased Approach**

The study was approached in three phases to provide a systematic review of what is needed in space suit design as a function of the potential destinations for human EVA. The first phase of this study, after defining the list of potential human exploration destinations, was to define a list of space suit design drivers per destination. The second phase took the destination-based design drivers for space suit hardware and focused on the physical characteristics of the local environments, grouping them into common design drivers. These subsequent groupings were: microgravity, reduced gravity, thermal extremes at vacuum, solar, and cosmic radiation; high-abrasion dust; low-abrasion dust; and thermal management in the presence of an atmosphere. The third phase of defining the design drivers focused on unique aspects of missions that would affect the design of space suits; this resulted in: mission length and distance from Earth (hardware reliability, maintainability, and complexity) and long durations of exposure to radiation.

### **B. Phase I: Defining Extravehicular Activity Design Drivers**

#### *1. Microgravity Destinations*

Mobility in microgravity becomes an issue as Newton's third law of motion comes into play: Bodies remain in a state of rest or uniform motion (constant velocity) unless they are acted upon by an external unbalanced force. How this transfers to suit design is in the ability to move and translate from one location to another with minimal resistance from the suit itself, and in smooth motions that will not excite unwanted suit dynamic motion or cause undesired impact forces to interaction with the local environment that would set the crew member in unwanted directions. What is desired is a suit that provides the required pressure and has mobility joints that provide low



torque and no programming.<sup>††</sup> While it is largely independent of the gravity field, it is one of the leading causes of astronaut fatigue during microgravity EVAs.

## 2. *Reduced-gravity Destinations*

The reduced-gravity destination environment group is comprised of potential EVA environments in which the local gravity field is defined as  $1.6 \text{ m/s}^2 < \text{local acceleration} < 6.5 \text{ m/s}^2$ . Of the possible destinations where humans can survive and potentially live long term with a return to Earth within 5 years, the following present themselves as viable destination candidates: Earth's moon and Mars. In a reduced-gravity environment, as defined previously, the two major design drivers are the mobility and mass of the space suit. Similarly, as discussed with the microgravity environment, pressurized mobility of the suit and minimizing suit-induced fatigue on the astronaut are highly desirable given the relatively short EVA time available and the voluminous lists of desired tasks during EVAs. If crew members are exhausted early in the EVA, not all objectives will be met. As has been seen in the past, crew member fatigue is primarily a combination of suit pressurization and tasks required of the crew; the gravity environment will obviously frame what tasks are required.

Secondly, the mass of the suit is important in a few different ways. The gravity environment in which the suit will be used and the length of time the crew member has been out of the Earth's 1g environment should be considered when defining the mass of the space suit. For example, if the crew has only been away from Earth for 1 week and will be operating a 68 kg (150 lbm) suit on the moon, the situation will be manageable (other than the inertial resistance of the suit) as this suit would appear to weigh, on the moon, the equivalent of 11.3 kg (25 lbs). However, if the crew member has been on the moon for 1 year and his or her muscular strength has adapted to the moon (e.g., no muscle resistance training to mitigate muscle atrophy), the suit would appear to weigh 68 kg (150 lbs) on the moon and would adversely affect the fatigue levels of the crew member. Granted, this scenario is unlikely under normal mission operations, but it is used here to exaggerate the point. There is growing thought that EVAs, when done regularly, could prevent atrophy due to the loading of the skeletal system from the suit; however, the ISS paradigm would imply the necessity of exercise protocol throughout a mission to prevent the known long-term effects of weightlessness.

There is also growing thought in the space suit community that a different look should be taken at how the mass of the suit is viewed and managed. The thought is that in reduced gravity environments, such as Earth's moon and that of the smaller moons, natural human ambulation as performed on Earth is not really practical or easy given the presence of reduced gravity. This was seen in the Apollo EVA video footage in which the crew would frequently fall over or would lope across the surface. Loping was easier to do than traditional Earth ambulation and was not as physically taxing. However, some recent simulated reduced-gravity testing performed at Johnson Space Center has indicated that even the suit-less human ambulation changes in the reduced-gravity environment should be investigated further, along with new approaches to suit mobility in these environments.

As with all things relating to space exploration, there is a trade-off between the amount of mass that can be launched from Earth and that required to perform the task optimally in the destination environments. Mass is always king on launch day, so careful mass margin management and impacts on mission objectives at the final destination, should be considered.

## 3. *Ionizing Radiation*

The nonthermal radiation environment for human exploration missions within and outside of Earth's Van Allen belts will increase the risk to human survival in two general situations: high-energy solar events and long-duration exposure to cosmic and solar radiation. Given the propulsion technology of today and the cost of space travel, any destination in our solar system will either require substantial time for the mission or that the time spent at the mission destination be ideally maximized so as to get the return on the financial investment. However, in the early exploration missions, the destination stay duration may be minimized initially to limit the risk with longer durations for subsequent missions.

Future mission architectures will need to take into account the whole systems approach to protecting astronauts from radiation. This approach includes vehicles, habitats, monitoring systems, and the suit. Transit vehicles and habitats will need to have radiation shielding, whereas robust monitoring systems placed locally and around the solar system will be needed to predict and warn of large SPEs. The suit should also be optimized to have increased radiation protection, but not at the expense of sacrificing any significant mobility.

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<sup>††</sup> Some space suit mobility joints are made of sets of circular bearings that are offset at different angles to obtain the desired range of motion for that joint. However, this often results in mobility joints that require proper alignment of the bearings to obtain certain reach zones.

Solar monitoring satellites such as the Solar TERrestrial RELations Observatory (STEREO), as well as other monitoring satellites in the Solar Terrestrial Probes program are essential in monitoring and predicting potentially hazardous SPEs.<sup>35</sup> The two STEREO satellites are nearly identical observatories positioned in heliocentric orbits leading and lagging the Earth in an orbit around the sun, and are used to capture images of the entire surface of the sun. These satellites help provide a three-dimensional view of solar activity and can be used to project the intensity and direction of SPEs that travel throughout our solar system. With the use of monitoring instruments like these, astronauts have some early warning of solar activities that may affect their current EVA. These instruments can also be used to plan EVAs for time periods expected to have less solar activity and thus less radiation from SPEs.

#### *4. Design Impacts due to Usage Duration*

For the same reasons that will be discussed later, most human exploration missions outside of Earth's orbit will necessitate long periods of time away from the safety and resources of Earth. Therefore, it becomes critical that space suit design be robust enough to endure expected usage or be maintainable by a crew with minimal recurring maintenance and required replacement parts during the mission.

This discussion on space suit reliability to a large extent is an uncharted area of study. Historically, space suit hardware is non-commercial custom hardware that is manufactured and operated in non-statistically significant quantities for standard statistical reliability calculation methods. For exploration missions, space suits will be mission-critical items that must be fail safe, but the trades must be done to optimize the acceptable risk posture, mass impacts due to robustness and redundancy (extra mass on suits or spare parts and required tools launched on the vehicle), and cost associated with developing design and testing methods to be able to characterize and predict the mean time between failure and modes of failure.

#### *5. High-abrasion Dust*

High-abrasion dust is characterized generically as in-situ regolith material the size of granules of sand or smaller in which no natural erosion processes are present; i.e., water or atmospheric mechanisms that have eroded or smoothed the edges of the particles once formed. While the extraterrestrial dust world has further segregated philosophically (i.e., arguments based on particle size and whether particles are considered "dust" or "regolith"), for the purpose of this discussion, it is not necessary to further stratify the definition. However, it should be noted that this design driver not only captures the dust particles, but also the in-situ environment – namely, wind or low gravity, both of which can exacerbate the issue.

#### *6. Low-abrasion Dust*

Low-abrasion dust is characterized generically as in-situ regolith material the size of granules of sand or smaller in which natural erosion processes are present; i.e., water or atmospheric mechanisms that have eroded or smoothed the edges of the particles once formed. Note that, similar to high-abrasion dust, this driver also considers the local environment that acts on the dust, including but not limited to wind or gravity fields.

### **C. Extreme Thermal Management at Vacuum**

#### *1. Lunar Pole in Permanent Shadows of Craters – Cold Extreme*

In recent years, there has been evidence of lunar ice at or below the surface of permanently shadowed areas within the craters at the lunar poles. These areas have been part of previous NASA design reference missions. Since water is a primary constituent required for human survival, expensive to launch from Earth to support missions, and the products generated through electrolysis can be used for rocket fuel, any destination that has a form of water available for utilization will be highly desirable.

However, for ice to exist, it must be protected from the solar wind and sublimation process that would require it to be outside the line of sight of the sun, be buried beneath a protective layer of dust, or be at cryogenic temperatures. This will place the astronauts working in an environment of cryogenic touch temperatures and, in turn, will drive the need for development of advanced materials that are highly flexible at these temperatures or of advanced glove or manipulator technology to increase crew productivity. Advancements will also have to be made to provide the thermal management of the crew for long durations at these temperatures.

#### *2. Lunar Equator, Center of Crater at Noon – Hot Extreme*

Earth's moon also offers the other end of the thermal management extreme for possible human exploration in the center of a crater, at the equator at lunar noon. The Apollo Program mitigated the impacts of both the cold polar and

**Table 2. Space Suit System Mass Values for NASA Historical, Currently Operational EVA-capable EMUs and designs.**

Parameter/Weight on Earth: kg (lbs)	Apollo EMU <sup>23</sup>	Space Shuttle EMU <sup>23,24</sup>	ISS EMU <sup>25</sup>	CxP Suit Element (goal requirements) <sup>26</sup>
Space Suit/Pressure Garment	35.4 (78)	43 (94)	65.2 (143.7)	ISS: 42 (92) Lunar: 42 (92)
Portable Life Support backpack	60.8 (134)	65.8 (145)	89.2 (196.7)	Lunar: 48.9 (108)
<b>Total</b>	96.2 (212)	108 (239)	154.4 (340.4)	90.7 (200)
<b>Operational Pressure</b> kN/m <sup>2</sup> (psia)	25.9 (3.75)	Nominal: 29.7 (4.3)	Nominal: 29.7 (4.3) DCS Treatment: 55.2 (8 above ambient)	29.7 (4.3) DCS Treatment: 55.2 (8 above ambient)

the hot equatorial thermal extremes by visiting the mid-latitude areas at lunar twilight.<sup>‡‡</sup> This approach was perfectly acceptable for humankind’s first venture from home; however, the approach will significantly handicap future extensive exploration and permanent habitation away from Earth.

While it is possible that L1 of the sun-Earth Lagrangian/ libration points has higher solar flux from the sun, the solar albedo<sup>§§</sup> resulting from a combination of normal reflection from the lunar surface (the angle of reflection and re-absorption by astronauts is higher) coupled with the solar flux<sup>¶¶</sup> normal to the surface (case for maximum surface coverage) and the “solar cooker” effect of the walls of the crater creates an environment that will be the most radiative thermally challenging of any destination humans may attempt to visit in the foreseeable future.

### 3. Moderate Thermal Extremes

The two environments that fall into this category are LEO (near a structure with significant thermal mass) and the Martian surface. This is an interesting grouping as these two environments represent the milder thermal management design challenges for space suits. These environments are both unique in that they are less extreme as far as how the design must be changed to address the environment.

The radiative thermal environment in LEO takes advantage of local albedo from the structure the astronaut is working around and that is being reflected from the Earth. Given the approximate 90-minute orbit duration<sup>###</sup> (45 minutes in the sun and 45 minutes in the Earth’s shadow), the conductive temperatures are moderated and can be further smoothed depending on the thermal inertia of the structure.

The Martian thermal environment can range from the moderately cold LEO temperatures to what would be a typical winter day in Scandinavia (-11 °C 2 to -8°C [-170°F to 17.6°F]), the hottest depending upon the latitude. The presence of an atmosphere, albeit one that is 1/168th that of Earth, does provide some convective and conductive heat transference that renders the current thermal insulation approach in vacuum inviable.

## D. Phase II: First-order Environmental Impacts to Design

The two most common groupings of destination environments were microgravity (LEO, GEO, sun-Earth-moon Lagrangian points, in-transit mission contingency EVAs, low-mass NEO/NEA, moons of Mars) and reduced-gravity environments (local gravitational fields of one-third Earth or less: Mars and Earth’s moon). The decision to group the environments in terms of the gravity field hinged on the mobility of the human performing the mission tasks and the technology required per the experience of NASA that the technology, tools, and mobility methods are dramatically different for a microgravity environment, a reduced-gravity environment, and that of Earth’s surface gravity. Experiences from Apollo EVAs and the suit mobility designs of the day resulted in the astronauts loping

<sup>‡‡</sup> This is not part of the discussion of this study, but it should be noted that the twilight conditions of the Apollo mission EVAs, in addition to providing thermal mitigation, also provided an optimum balance of lighting conditions. In the absence of an atmosphere to diffract light, the contrast between the directly illuminated surface and that of the shadows is difficult for the human eye to readily adapt. The result is a lack of depth perception and an inability to see into shadows until within the shadow. This would remain an issue for design and operations of future missions.

<sup>§§</sup> The albedo of an object is a measure of how strongly it reflects light from light sources such as the sun. It is therefore a more specific form of the term reflectivity. Albedo is defined as the ratio of total-reflected to incident electromagnetic radiation.

<sup>¶¶</sup> Solar flux, or radiative flux, is the amount of energy moving in the form of photons at a certain distance from the source per angle of incidence per second.

<sup>###</sup> This orbital period is representative of the typical operational orbit for the space shuttle.

across the lunar surface for a long traverse, as they found this to be an easier and more expedient means of traversing than fighting the internal pressurized forces of the suit via a more Earth-like ambulation. Likewise, the current knowledge of the NEOs suggests – and, quite frankly, our hope – that there are none that approach the size of the moon or larger necessitating Earth-like ambulation, despite what Hollywood would have you believe.

The need for human life support (which, in this case, will be defined as requiring, pressure management, thermal control, inspired oxygen, and hydration) within the biological requirements for normal bodily function is necessary for all human excursions outside the bounds of the Earth’s surface. Therefore, this is common to all destination environments and not called out specifically as a design driver – with the exception of when there is an atmosphere or not, as this will influence the technologies and strategies for providing these life support functions. It should be noted that life support systems are one of the more complicated and expensive systems required in space flight and should therefore not be trivialized or forgotten when prioritizing development.

It should also be noted that the impacts due to the internal operational pressure of the suit can significantly affect suit design and system mass. In Table 1, the impact to system mass of the ISS EMU due to an operational pressure of 8 psi, as opposed to the typical 4.3 psi, is significant. Whereas a designed operational pressure of 8 psi is not required for EVAs, it does profoundly decrease the amount of prebreathe time on pure oxygen to denitrogenate the blood to prevent decompression sickness.<sup>27</sup> As a 4.3-psi suit impacts the timeline of a mission due to required prebreathe and uses known technology for design, the 4.3-psi suit is considered operationally desirable for this study; it is not singled out as an environmental design driver.

Lastly, the suit operating pressure inside the vehicle has to be coordinated with the cabin atmosphere and pressure, oxygen concentrations (which affect flammability considerations), EVA prebreathe protocols, and vehicle operational constraints.

### **E. Phase III: Second-order Environmental Impacts to Design**

Once the destination environments were grouped, a mapping to the environmental suit design drivers was performed as can be seen in Figure 2. In environment groups in which all of the included environments contribute to a suit design driver, the line for the group begins at the group boundary and proceeds to the design driver. For design drivers in which all of the environments within a group do not map to the design driver, lines specific to that environment map to the driver. For example, all of the microgravity environments map to the microgravity and to the extreme-thermal-at-vacuum drivers and, thus, the black dotted line maps from the microgravity destinations group border. However, all of the microgravity destinations map to the radiation design driver except for the LEO environment; therefore, all of the microgravity environments – except for LEO – map via lines to the radiation design driver.

As seen in Figure 2, and perhaps more clearly in Figure 3, extreme thermal management in vacuum and radiation protection from a high-level assessment are design drivers in 89% of all possible destinations on which humans are likely to perform EVAs. Coming in a close second at 78% of all destination environments are the design drivers, due to mission duration (time) and microgravity. Following these, we see at surprisingly low percentages: high-abrasion dust at 33%, reduced gravity at 22%, and moderate thermal and low-abrasion dust tied last at 11%.

#### *1. Extreme Thermal at Vacuum*

Not surprisingly, all destination environments listed, with the exception of Mars, have to contend with thermal management in the vacuum of space. In reviewing the data, we see that, of the destinations available for human EVA, the thermal extremes on Earth’s moon encompass all other environments.

## 2. Radiation

Historically, due to mass constraints for launch capability and cost, protection against solar and cosmic radiation has been “best-effort” strategy in which outer garments provide protection against alpha particle radiation but offer limited effectiveness against anything else. To date, NASA has mitigated exposure for LEO operations by monitoring solar activity and limiting EVA time, and by providing vehicle shielding during high-activity or solar events to limit exposure to the crew. A similar approach was used for the Apollo missions, but the information regarding solar activity was limited due to ground-based telescopes and radiation monitors on the lunar lander.

It should be noted that there is no delineation in Figure 3 in the percentages as to what form of radiation each of the environments includes; instead, the percentages are rolled up. Environmental groupings in Figure 2 show that radiation protection is a significant environmental design driver that is common to all destinations outside of LEO and can have a profound impact on human life due to the long mission durations and for long-term exposure. Given the leaps of understanding on the mechanics of radiation and decay, and their effects on humans, it is critical that a concerted effort be applied to suit development for human exploration in this area.

## 3. Design Impacts due to Time

With the exception of LEO and GEO, using the assumption that crewed vehicles will be limited to current chemical combustion technology, all potential destinations for human EVA will require long mission durations at quite a distance from the resources and supplies of Earth. Therefore, it should be recognized that a methodical approach must be taken for developing a highly reliable space suit system. By focusing on the individual design element with regard to the most extreme operational environment and the maximum mission duration, the goal would be to drive the mean time to failure well beyond any mission hardware needs. Such a systematic approach will, over time, drive out the failure modes, increase the design reliability, and build a statistical operational experience base such that failures are well understood and, at times, predictable.

## 4. Microgravity Design Drivers

Similar to the discussion of thermal management at extreme temperatures in a vacuum, the

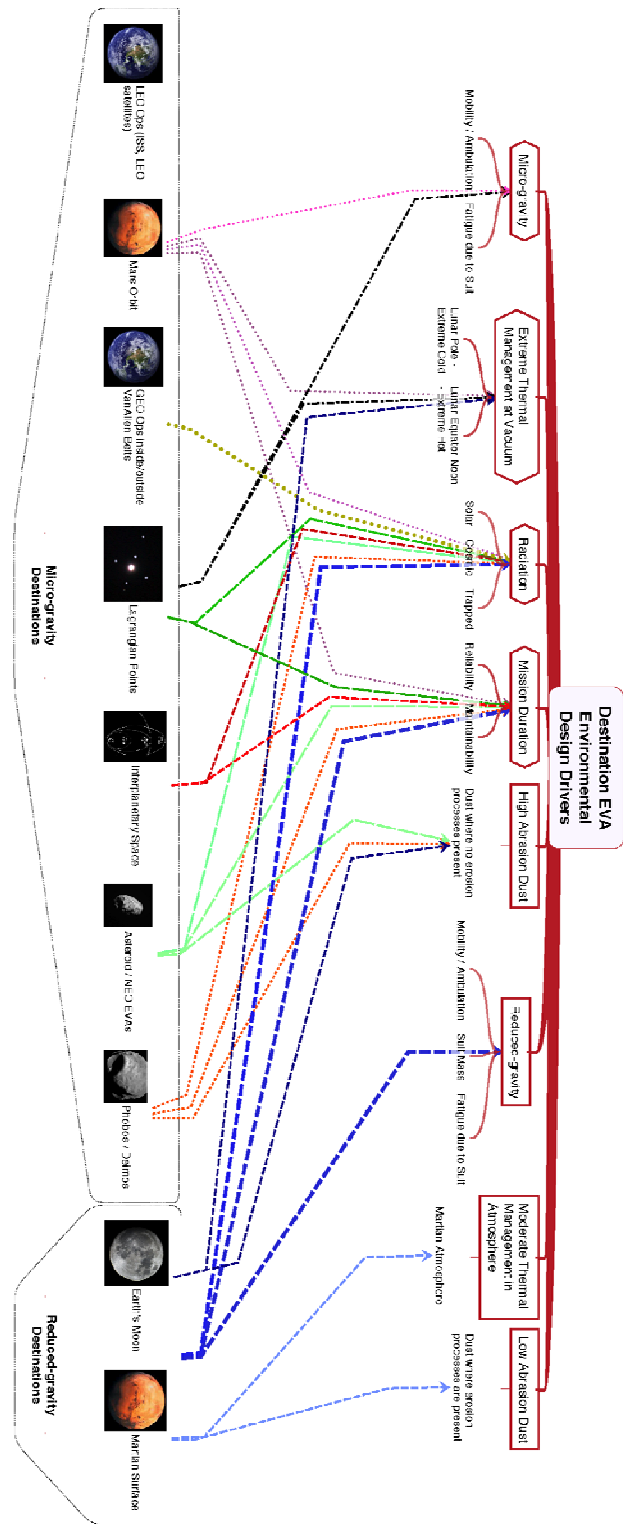
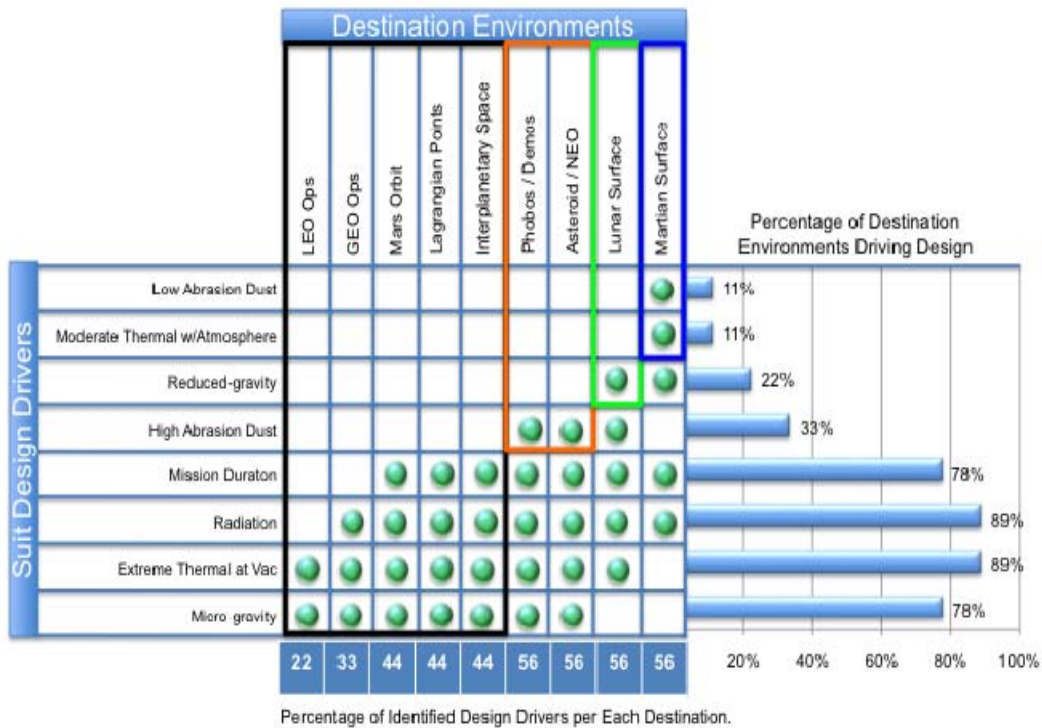


Figure 2 – Mapping of the probable human EVA destination environments to the primary space suit design drivers.



**Figure 3 – Mapping of the EVA destination environments to the design drivers so as to identify commonality percentages. This will aid in identifying the most common design drivers for all the possible human EVA destinations.**

number of destinations with very low to negligible gravitational acceleration by far outweigh the destinations in which a reduced, yet significant, gravity field is present that is relatively hospitable to humans.

### 5. High-abrasion Dust

High-abrasion dust, as a design driver, comes into play in less than half of the environments discussed in this paper, given the number of destinations that pertain to deep-space EVAs or Mars, where the dust has been eroded over time. It should also be noted that our experience with the dust on Earth’s moon is indicative of what is expected for destinations with high-abrasion dust.

### 6. Reduced Gravity, Thermal Management in an Atmosphere, and Low-abrasion Dust

The last three are grouped because the percentages, while not initially expected, make sense when considering all other destinations. All three have to do with Mars and Earth’s moon, which are the only significant bodies within current human exploration. Moreover, Mars is the only other body with an atmosphere that facilitates two of these three design drivers. Further implications of these findings will be discussed later.

## V. Implications to Technology Development Strategies

Given the past success rate of projects to be funded through completion within NASA, it is advised to obtain funding via non-flight program monies, develop the technologies that will give the highest probable return on investment with the greatest likelihood of being needed, and coordinate the effort at the agency level to reduce the likelihood of redundant effort or miss-vectoring.

This study addresses the likelihood of design drivers as a function of the possible destinations that human EVA will potentially encounter, given the likelihood of technological advancements within the next few decades as have been seen in the past 30 years. With this in mind, results could differ from those one would expect, given past efforts in suit design and technology developed to any significant level. In the past, these efforts were defined by a particular mission with a particular destination in mind – usually the first time visiting that destination. In that framework, that paradigm of design and technology development prioritization made sense. However, in a future in which resources to be applied to space suit design and technology development will be scarce and prioritization will

be expected, the need for exploration as well as the destinations to be explored will vary with policy makers in power; therefore, a prioritization based on the likelihood of occurrence should be seriously considered.

If it is clear that Mars is a high-priority destination due to national security, discovery of unobtainium, or survival of the species, the prioritization presented here will be overcome by events. But, lacking such direction, we see here that half of the significant design drivers for space suits encompass 78% to 89% of all destinations for human EVA. What we do see is that only one-third of the suit design drivers are specific to Mars.

From a perspective of return on investment to reach the maximum yield of dollars invested in space suit design and technology development, a new focus should be brought into the forefront for discussion. A modular suit architecture, as discussed in reference 1 and 2, has the potential for a generic set of suit hardware components or elements that would address the majority of destination environments while minimizing the impact to performance. It would provide hardware and design interfaces such that suit components that needed to be changed due to specific and/or unique environmental constraints would be changed. Additionally, the modular nature of the architecture would allow integration of new technologies as needed without a massive redesign effort. Furthermore, by minimizing the costs due to suit redesign, cost savings in terms of launch mass, and only launching the suit components necessary for destinations of that mission, savings in terms of schedule can be realized since the technology can be developed prior to the mission that is being defined; i.e., the sooner you launch, the cheaper it is, given you have saved the money in the out-years due to inflated dollars.

In terms of driving design requirements, Figure 3 illustrates that the largest return on investment to address the most common design drivers to the largest number of destinations can be seen in the grouping defined by the black box. With the addition of the next most common design driver alone, high-abrasion dust, it now opens up the next set of destinations as indicated by the orange box. Once again, adding in the next most common design drivers, reduced gravity, and moderate thermal and low-abrasion dust, then is additional destination is grouped respectively in green and blue boxes. The method of mapping and grouping of the most common design drivers to frequency of destinations will provide an efficient plan for developing technology with the greatest return in the absence of a defined and adequately funded mission destination.

It is not the intent of this paper to assess the current state-of-the-art space suit design with respect to any of the design drivers discussed here. It is the intent to bring to the stage the notion that addressing the design drivers in a systematic and well-managed effort – that will be most frequently encountered in human EVAs in the foreseeable future – will yield the largest return on investment outside of a specific mission and destination.

## **VI. Conclusion**

This study addresses how a generic, modular, environmental protection, space-suit architecture would be beneficial when combined with the study of all potential destinations in the solar system for human EVAs within the next 30 years. This is based on current technological capability and, using linear extrapolation, that which can be achieved based on experiences during the last 50 years of human space flight (no warp drives and force fields available) combined with a systematic prioritization of technology development as defined by likelihood of need for human EVAs. These two, when combined, provide a space suit architecture that is easily modified, depending on the mission destination, and can be upgraded when new technology is available with minimal cost and redesign. One example to illustrate the modular architecture and ability to upgrade as required is the TMG. The TMG can be minimized for use in LEO. When a mission is required to go to the moon, the TMG can be replaced with a version that is specialized for the lunar environment. The TMG can later be replaced with versions that are optimized for the other thermal and micrometeoroid environments defined in this paper. As long as the suit and interfaces are well defined, use of the TMG will minimize the cost of upgrading the suit capability by not requiring a major redesign effort.

The destination list, which is based on these selection criteria, is greatly narrowed and the possible destinations for human (in-person) exploration reduces into a well-defined subset of space suit design drivers that are not likely to change significantly in the near future and can be used now to solve most – if not all – of the major design challenges facing space suit engineers and exploration programs.

The findings and rankings presented in this paper provide a mission-independent, EVA system development approach based on destination environmental space suit design driver likelihood. This approach will help ensure the highest likelihood, and highest return on investment while there is no programmatic destination of record. It will also ensure the opportunity to provide the largest return on taxpayer dollars that will meet multiple future mission destinations. This allows a greater chance of providing better technical solutions to future missions when they are

needed, as opposed to waiting for a mission to be identified and then starting to solve the technical suit design problems, once the programmatic and budgetary clocks have begun to tick.

It is highly recommended that this development approach be considered and managed as a “Flight Program,” meaning that development technical requirements, budgets, and developmental milestones are well defined and managed to agreed-upon completion dates. This will help ensure that these efforts reach the desired engineering solution in a reasonable amount of time and aid in maturing the technology incrementally as the funding is available. Lastly, it should be noted that while the environment is the primary design driver in space suit design, the largest secondary driver is the activity that will be performed in the suit and should not be forgotten when formulating the space suit architecture and considering how to incorporate the needed technologies for the destination environment.



## Appendix A

**Table A-1: The specific environmental values that are used for this study along with rationale and references.**

Destination → ↓ Parameter		ISS / LEO	GEO	Earth-Moon Lagrange Points	Interplanetary Contingency Environment	Asteroid / NEO	Lunar Equator	Lunar Poles	Phobos Surface	Deimos Surface	Mars Orbit	Mars Equator	Mars Poles
Surface Temperature (°F)	Min	N/A <sup>1</sup>	N/A <sup>27</sup>	N/A <sup>53</sup>	N/A <sup>79</sup>	-260 <sup>105</sup>	-289 <sup>131</sup>	-415 <sup>157</sup>	-226 <sup>183</sup>	-243 <sup>209</sup>	N/A <sup>235</sup>	-175 <sup>261</sup>	-202 <sup>287</sup>
	Max	N/A <sup>2</sup>	N/A <sup>28</sup>	N/A <sup>54</sup>	N/A <sup>80</sup>	441 <sup>106</sup>	278 <sup>132</sup>	80 <sup>158</sup>	176 <sup>184</sup>	159 <sup>210</sup>	N/A <sup>236</sup>	68 <sup>262</sup>	5 <sup>288</sup>
Suit Sink Temperature (°F)	Min	-224 <sup>3</sup>	-224 <sup>29</sup>	-125 <sup>55</sup>	-304 <sup>81</sup>	-304 <sup>107</sup>	-300 <sup>133</sup>	-300 <sup>159</sup>	-99 <sup>185</sup>	-113 <sup>211</sup>	-245 <sup>237</sup>	-227 <sup>263</sup>	-249 <sup>289</sup>
	Max	111 <sup>4</sup>	111 <sup>30</sup>	86 <sup>56</sup>	311 <sup>82</sup>	311 <sup>108</sup>	226 <sup>134</sup>	-99 <sup>160</sup>	58 <sup>186</sup>	44 <sup>212</sup>	91 <sup>238</sup>	99 <sup>264</sup>	63 <sup>290</sup>
Touch Temperature (°F)	Min	-180 <sup>5</sup>	-180 <sup>31</sup>	-180 <sup>57</sup>	-180 <sup>83</sup>	-180 <sup>109</sup>	-180 <sup>135</sup>	-180 <sup>161</sup>	-180 <sup>187</sup>	-180 <sup>213</sup>	-180 <sup>239</sup>	-180 <sup>265</sup>	-180 <sup>291</sup>
	Max	384 <sup>5</sup>	384 <sup>32</sup>	384 <sup>58</sup>	444 <sup>84</sup>	444 <sup>110</sup>	384 <sup>136</sup>	384 <sup>162</sup>	271 <sup>188</sup>	271 <sup>214</sup>	271 <sup>240</sup>	271 <sup>266</sup>	271 <sup>292</sup>
Dust/ Particulate environment (Lunar = 1.0)	Min	0 <sup>7</sup>	0 <sup>33</sup>	0 <sup>59</sup>	0 <sup>85</sup>	0 <sup>111</sup>	0.66 <sup>137</sup>	0.66 <sup>163</sup>	0 <sup>189</sup>	0.1 <sup>215</sup>	0 <sup>241</sup>	0.66 <sup>267</sup>	0.66 <sup>293</sup>
	Max	0 <sup>8</sup>	0 <sup>34</sup>	0 <sup>60</sup>	0 <sup>86</sup>	1.5 <sup>112</sup>	1.34 <sup>138</sup>	1.34 <sup>164</sup>	1.5 <sup>190</sup>	1.5 <sup>216</sup>	0 <sup>242</sup>	1.68 <sup>268</sup>	1.68 <sup>294</sup>
High energy debris (incidents/m <sup>2</sup> -day)	Min	0.050 <sup>9</sup>	0.050 <sup>35</sup>	0.033 <sup>61</sup>	0.033 <sup>87</sup>	0.033 <sup>113</sup>	0.050 <sup>139</sup>	0.050 <sup>165</sup>	0.033 <sup>191</sup>	0.033 <sup>217</sup>	0.033 <sup>243</sup>	0.000 <sup>269</sup>	0.000 <sup>295</sup>
	Max	1.2 <sup>10</sup>	1.2 <sup>36</sup>	0.075 <sup>62</sup>	0.075 <sup>88</sup>	0.075 <sup>114</sup>	0.6 <sup>140</sup>	0.6 <sup>166</sup>	0.075 <sup>192</sup>	0.075 <sup>218</sup>	0.075 <sup>244</sup>	0.0005 <sup>270</sup>	0.0005 <sup>296</sup>
Radiation (mSv/hr)	Min	0.02 <sup>11</sup>	417 <sup>37</sup>	0.042 <sup>63</sup>	0.042 <sup>89</sup>	0.042 <sup>115</sup>	0.034 <sup>141</sup>	0.034 <sup>167</sup>	0.042 <sup>193</sup>	0.042 <sup>219</sup>	0.042 <sup>245</sup>	0.023 <sup>271</sup>	0.023 <sup>297</sup>
	Max	0.06 <sup>12</sup>	417,000 <sup>38</sup>	0.083 <sup>64</sup>	0.083 <sup>90</sup>	0.083 <sup>116</sup>	0.098 <sup>142</sup>	0.098 <sup>168</sup>	0.083 <sup>194</sup>	0.083 <sup>220</sup>	0.083 <sup>246</sup>	0.134 <sup>272</sup>	0.134 <sup>298</sup>
Atmospheric Pressure (psi)	Min	0.00 <sup>13</sup>	0.00 <sup>39</sup>	0.00 <sup>65</sup>	0.00 <sup>91</sup>	0.00 <sup>117</sup>	4.0E-14 <sup>143</sup>	4.0E-14 <sup>169</sup>	0.00 <sup>195</sup>	0.00 <sup>221</sup>	0.00 <sup>247</sup>	0.058 <sup>273</sup>	0.058 <sup>299</sup>
	Max	0.00 <sup>14</sup>	0.00 <sup>40</sup>	0.00 <sup>66</sup>	0.00 <sup>92</sup>	0.00 <sup>118</sup>	4.0E-14 <sup>144</sup>	4.0E-14 <sup>170</sup>	0.00 <sup>196</sup>	0.00 <sup>222</sup>	0.00 <sup>248</sup>	0.126 <sup>274</sup>	0.126 <sup>300</sup>
Gravity (g)	Min	0.00 <sup>15</sup>	0.00 <sup>41</sup>	0.00 <sup>67</sup>	0.00 <sup>93</sup>	0.00 <sup>119</sup>	0.166 <sup>145</sup>	0.166 <sup>171</sup>	4.0E-03 <sup>197</sup>	3.0E-03 <sup>223</sup>	0.00 <sup>249</sup>	0.378 <sup>275</sup>	0.378 <sup>301</sup>
	Max	0.00 <sup>16</sup>	0.00 <sup>42</sup>	0.00 <sup>68</sup>	0.00 <sup>94</sup>	0.00 <sup>120</sup>	0.167 <sup>146</sup>	0.167 <sup>172</sup>	6.7E-03 <sup>198</sup>	3.9E-03 <sup>224</sup>	0.00 <sup>250</sup>	0.379 <sup>276</sup>	0.379 <sup>302</sup>
Surface density (g/cm <sup>3</sup> )	Min	N/A <sup>17</sup>	N/A <sup>43</sup>	N/A <sup>69</sup>	N/A <sup>95</sup>	1.0 <sup>121</sup>	2.2 <sup>147</sup>	2.2 <sup>173</sup>	1.3 <sup>199</sup>	0.8 <sup>225</sup>	N/A <sup>251</sup>	1.0 <sup>277</sup>	1.0 <sup>303</sup>
	Max	N/A <sup>18</sup>	N/A <sup>44</sup>	N/A <sup>70</sup>	N/A <sup>96</sup>	6.0 <sup>122</sup>	3.2 <sup>148</sup>	3.2 <sup>174</sup>	1.9 <sup>200</sup>	1.4 <sup>226</sup>	N/A <sup>252</sup>	2.6 <sup>278</sup>	2.6 <sup>304</sup>
Magnetic Field (Gauss)	Min	0.002 <sup>19</sup>	TBD <sup>45</sup>	5.0E-05 <sup>71</sup>	0 <sup>97</sup>	5.0E-05 <sup>123</sup>	1.0E-06 <sup>149</sup>	1.0E-06 <sup>175</sup>	0.57 <sup>201</sup>	0.00 <sup>227</sup>	4.0E-05 <sup>253</sup>	-0.002 <sup>279</sup>	-0.0020 <sup>305</sup>
	Max	0.60 <sup>20</sup>	TBD <sup>46</sup>	1.0E-04 <sup>72</sup>	0 <sup>98</sup>	0.5 <sup>124</sup>	0.005 <sup>150</sup>	0.001 <sup>176</sup>	0.60 <sup>202</sup>	0.60 <sup>228</sup>	5.0E-05 <sup>254</sup>	0.002 <sup>280</sup>	0.0005 <sup>306</sup>
Daylight duration (hours)	Min	0.75 <sup>21</sup>	21.75 <sup>47</sup>	∞ <sup>73</sup>	3840 <sup>99</sup>	0 <sup>125</sup>	354.35 <sup>151</sup>	120 <sup>177</sup>	3.85 <sup>203</sup>	15 <sup>229</sup>	0.5 <sup>255</sup>	10.3 <sup>281</sup>	0 <sup>307</sup>
	Max	0.75 <sup>22</sup>	∞ <sup>48</sup>	∞ <sup>74</sup>	3840 <sup>100</sup>	3840 <sup>126</sup>	354.35 <sup>152</sup>	4383 <sup>178</sup>	3360 <sup>204</sup>	7200 <sup>230</sup>	∞ <sup>256</sup>	14.3 <sup>282</sup>	8244 <sup>308</sup>
Night duration (hours)	Min	0.75 <sup>23</sup>	0 <sup>49</sup>	0 <sup>75</sup>	0 <sup>101</sup>	0 <sup>127</sup>	354.35 <sup>153</sup>	120 <sup>179</sup>	3.85 <sup>205</sup>	15 <sup>231</sup>	0 <sup>257</sup>	10.3 <sup>283</sup>	0 <sup>309</sup>
	Max	0.75 <sup>24</sup>	2.25 <sup>50</sup>	0 <sup>76</sup>	0 <sup>102</sup>	3840 <sup>128</sup>	354.35 <sup>154</sup>	4383 <sup>180</sup>	3360 <sup>206</sup>	7200 <sup>232</sup>	0.5 <sup>258</sup>	14.3 <sup>284</sup>	8244 <sup>310</sup>
Total mission time (days)	Min	1 <sup>25</sup>	1 <sup>51</sup>	7 <sup>77</sup>	90 <sup>103</sup>	90 <sup>129</sup>	14 <sup>155</sup>	14 <sup>181</sup>	365 <sup>207</sup>	365 <sup>233</sup>	365 <sup>259</sup>	365 <sup>285</sup>	365 <sup>311</sup>
	Max	365 <sup>26</sup>	30 <sup>52</sup>	180 <sup>78</sup>	180 <sup>104</sup>	365 <sup>130</sup>	365 <sup>156</sup>	365 <sup>182</sup>	1000 <sup>208</sup>	1000 <sup>234</sup>	1000 <sup>260</sup>	1000 <sup>286</sup>	1000 <sup>312</sup>

### ISS/LEO

- 1: No surface to reference
- 2: No surface to reference
- 3: Calculated by author assuming 0 solar, 33 W/m<sup>2</sup> vehicle IR with 0.5 orthogonal view factor
- 4: Calculated by author assuming full solar, 445 W/m<sup>2</sup> vehicle IR with 0.26 albedo and 0.5 view factors
- 5: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)
- 6: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)
- 7: Captured under high energy debris
- 8: Captured under high energy debris
- 9: LDEF – Long Duration Exposure Facility data – 6 years, 1300 incidents/m<sup>2</sup>: wake side only = 118/m<sup>2</sup>
- 10: LDEF – 6 years, 1300 incidents/m<sup>2</sup> with 100% margin
- 11: Solar min, 400km, 28.5 deg inclination
- 12: Maximum exposure recorded on Skylab
- 13: No atmosphere
- 14: No atmosphere
- 15: Orbital trajectory assumed
- 16: Orbital trajectory assumed
- 17: No surface to reference
- 18: No surface to reference
- 19: “A Comprehensive Model of the Near-Earth Magnetic Field”. NASA/TM-2000-209894. 300-800km range
- 20: “A Comprehensive Model of the Near-Earth Magnetic Field”. NASA/TM-2000-209894. At the polar surface
- 21: Assumed ~90 min orbit
- 22: Assumed ~90 min orbit
- 23: Assumed ~90 min orbit
- 24: Assumed ~90 min orbit
- 25: Experience base of US historical activity.
- 26: Experience base of US historical activity.

### GEO

- 27: No surface to reference
- 28: No surface to reference
- 29: Calculated by author assuming 0 solar, 33 W/m<sup>2</sup> vehicle IR with 0.5 orthogonal view factor
- 30: Calculated by author assuming full solar, 445 W/m<sup>2</sup> vehicle IR with 0.26 albedo and 0.5 view factors
- 31: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)

- 32: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)
- 33: Captured under high energy debris
- 34: Captured under high energy debris
- 35: LDEF – Long Duration Exposure Facility data – 6 years, 1300 incidents/m<sup>2</sup>: wake side only = 118/m<sup>2</sup>
- 36: LDEF – 6 years, 1300 incidents/m<sup>2</sup> with 100% margin
- 37: Assuming best case scenario, @ 0° inclination, closest to the sun in orbit with no geomagnetic storms
- 38: Farthest from the sun in orbit, geomagnetic storm, high solar activity
- 39: No atmosphere
- 40: No atmosphere
- 41: Orbital trajectory assumed
- 42: Orbital trajectory assumed
- 43: No surface to reference
- 44: No surface to reference
- 45: TBD
- 46: TBD
- 47: Corresponds to near-equatorial geostationary orbit with approximately 2 hours of shade per day as it passes behind the Earth
- 48: Corresponds to geostationary orbit of sufficiently high or low latitude to provide full constant sunlight
- 49: Corresponds to geostationary orbit of sufficiently high or low latitude to provide full constant sunlight
- 50: Corresponds to near-equatorial geostationary orbit with approximately 2 hours of shade per day as it passes behind the Earth
- 51: Estimate based upon possible required tasks, variability of radiation exposure due to solar activity and orbital altitude.
- 52: Estimate based upon possible required tasks, variability of radiation exposure due to solar activity and orbital altitude.

### Earth-Moon Lagrange Points

- 53: No surface to reference
- 54: No surface to reference
- 55: Calculated by author assuming full sun, no vehicle/planetary IR contributions – operational controls for 30 second dark periods
- 56: Calculated by author assuming 445 W/m<sup>2</sup> vehicle IR and 0.9 reflectance, 0.5 orthogonal view factor
- 57: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)
- 58: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)
- 59: L1 and L2 are unstable orbits and no dust is assumed.

60: L4 and L5 have stable orbits and dust clouds have been theorized but not positively identified. Most dust would be blown away by solar pressure leaving only larger pieces

61: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only -50%

62: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only +50%

63: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

64: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

65: No atmosphere at this location

66: No atmosphere at this location

67: Negligible net gravity field at this location

68: Negligible net gravity field at this location

69: No surface to reference

70: No surface to reference

71: "Turning Up the Heat in the Corona" Glanz 2007. Science 278 387. Corresponds to interplanetary space near Earth

72: "The Remnant Magnetic Field of the Moon" Coleman 1977. Corresponds to 10 nT, taken from Apollo measurements in Lunar orbit while passing through Earth's geomagnetic tail

73: "Exploration in the Earth's Neighborhood - Architecture Analysis" B. Kent Joosten, NASA 2000. 99.91 continuous sun – only darkness for ~30 sec periods during eclipse

74: "Exploration in the Earth's Neighborhood - Architecture Analysis" B. Kent Joosten, NASA 2000. 99.91 continuous sun – only darkness for ~30 sec periods during eclipse

75: "Exploration in the Earth's Neighborhood - Architecture Analysis" B. Kent Joosten, NASA 2000. 99.91 continuous sun – only darkness for ~30 sec periods during eclipse

76: "Exploration in the Earth's Neighborhood - Architecture Analysis" B. Kent Joosten, NASA 2000. 99.91 continuous sun – only darkness for ~30 sec periods during eclipse

77: Estimate based upon past NASA DRM exercises.

78: Estimate based upon past NASA DRM exercises.

#### Local Interplanetary Space

79: No surface to reference

80: No surface to reference

81: Taken from worst case Asteroid calculation (107)

82: Taken from worst case Asteroid calculation (108)

83: Taken from worst case Asteroid calculation (109)

84: Taken from worst case Asteroid calculation (110)

85: Captured under high energy debris

86: Captured under high energy debris

87: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only -50%

88: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only +50%

89: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

90: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

91: No atmosphere at this location

92: No atmosphere at this location

93: Negligible net gravity field at this location

94: Negligible net gravity field at this location

95: No surface to reference

96: No surface to reference

97: No object to reference

98: No object to reference

99: Calculated by author assuming constant sunlight. Does not consider shade provided by transit vehicle

100: Calculated by author assuming constant sunlight. Does not consider shade provided by transit vehicle

101: Calculated by author assuming constant sunlight. Does not consider shade provided by transit vehicle

102: Calculated by author assuming constant sunlight. Does not consider shade provided by transit vehicle

103: Estimate based upon past NASA DRM exercises.

104: Estimate based upon past NASA DRM exercises.

#### Asteroid/NEO

105: Calculated by the author for 50m asteroid at 1.3AU. Rough estimate based on estimated optical properties with spin state of asteroid unknown

106: Calculated by the author for 130m asteroid at 0.86AU. Rough estimate based on estimated optical properties with spin state of asteroid unknown

107: Calculated by author assuming full shade 10m away from vehicle

108: Calculated by author assuming full solar next to max hot asteroid

109: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)

110: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)

111: Assumes a best case of hard asteroid surface with no dust or particulates except those generated by direct contact with the surface

112: Very little data – could be similar to the Moon but with reduced gravity which would further impede visibility

113: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only -50%  
 114: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only +50%  
 115: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs  
 116: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs  
 117: Calculated by the author, "Opportunities for Near Earth Object Exploration", Johnson - NASA 2010. All candidate asteroids are orders of magnitude too small to hold any atmosphere  
 118: Calculated by the author, "Opportunities for Near Earth Object Exploration", Johnson - NASA 2010. All candidate asteroids are orders of magnitude too small to hold any atmosphere  
 119: Calculated by the author, "Opportunities for Near Earth Object Exploration", Johnson - NASA 2010. All candidate asteroids are on the order of nano-g's  
 120: Calculated by the author, "Opportunities for Near Earth Object Exploration", Johnson - NASA 2010. All candidate asteroids are on the order of nano-g's  
 121: "Asteroid Density, Porosity, and Structure" - Britt et al. Casual min taken from 22 asteroids detailed  
 122: "Asteroid Density, Porosity, and Structure" - Britt et al. Casual max taken from 22 asteroids detailed  
 123: "NEAR Magnetic Field Observations at 433 Eros: First Measurements from the Surface of an Asteroid" - Acuna et. al. Corresponds to ~5 nT from NEAR, only in-situ measurement of asteroid magnetic field to date  
 124: "A Magnetohydrodynamic Model of Solar Wind Interaction with Asteroid Gaspra" Science 263 653. High uncertainty due to fly-by measurement  
 125: Calculated by author assuming constant shade  
 126: Calculated by author assuming constant sunlight. Does not consider shade provided by vehicle  
 127: Calculated by author assuming constant shade  
 128: Calculated by author assuming constant sunlight. Does not consider shade provided by vehicle  
 129: Estimate based upon past NASA DRM exercises.  
 130: Estimate based upon past NASA DRM exercises.

#### Lunar Equator

131: LRO Diviner Lunar Radiometer Experiment – temperature maps corresponding to nighttime temperature  
 132: LRO Diviner Lunar Radiometer Experiment – temperature maps corresponding to small equatorial craters. Flat terrain at noon roughly 240F  
 133: Calculated by author assuming 90 deg zenith angle at crater bottom  
 134: Calculated by author assuming Lunar noon at crater bottom

135: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)  
 136: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)  
 137: Assumed Lunar = 1.0 +/- 1 standard deviation; more data needed to compare Apollo visited sides to entire Lunar surface  
 138: Assumed Lunar = 1.0 +/- 1 standard deviation; more data needed to compare Apollo visited sides to entire Lunar surface  
 139: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only. Realistic value for Lunar bombardment as majority of wake side data is micrometeorite impacts  
 140: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; considered worst case for low junk Lunar orbit. LDEF data was 5:1 junk to micrometeorite ratio  
 141: Simonsen, Lisa, Nealy, J.E., Radiation Protection for Human Missions to the Moon and Mars, NASA TP-3079, Feb 1991. GCR contribution at solar minimum  
 142: Simonsen, Lisa, Nealy, J.E., Radiation Protection for Human Missions to the Moon and Mars, NASA TP-3079, Feb 1991. Assuming normal distribution using minimum and average values  
 143: "The lunar atmosphere: History, status, current problems, and context"; Reviews of Geophysics 37 453. Scientifically non-zero but a vacuum for practical engineering purposes  
 144: "The lunar atmosphere: History, status, current problems, and context"; Reviews of Geophysics 37 453. Scientifically non-zero but a vacuum for practical engineering purposes  
 145: "Farside Gravity Field of the Moon from Four-Way Doppler Measurements of SELENE"; Science 323 900. Taking average and applying 350mGal deviation for gravity anomalies  
 146: "Farside Gravity Field of the Moon from Four-Way Doppler Measurements of SELENE"; Science 323 900. Taking average and applying 350mGal deviation for gravity anomalies  
 147: "Density and Porosity of Apollo Lunar Basalts and Breccias" - converted density from grain to bulk. 22 Apollo samples from different sites - Min corresponds to Imbrium Ejecta, Impact-Melt Breccia  
 148: "Density and Porosity of Apollo Lunar Basalts and Breccias". Converted density from grain to bulk. 22 Apollo samples from different sites. Max corresponds to Ti-included Basalts  
 149: Lunar Prospector electron reflectometer experiment; corresponds to 0.1 nT from magnetic field map  
 150: Lunar Prospector electron reflectometer experiment; corresponds to 500 nT from magnetic field map  
 151: The Lunar Base Handbook, Eckart 2006 pp 118; 1.5 deg Lunar axial tilt and up to +/- 30 deg latitude shift contributes negligible seasonal change to day/night cycle  
 152: The Lunar Base Handbook, Eckart 2006 pp 118; 1.5 deg Lunar axial tilt and up to +/- 30 deg latitude shift contributes negligible seasonal change to day/night cycle

153: The Lunar Base Handbook, Eckart 2006 pp 118; 1.5 deg Lunar axial tilt and up to +/- 30 deg latitude shift contributes negligible seasonal change to day/night cycle  
154: The Lunar Base Handbook, Eckart 2006 pp 118; 1.5 deg Lunar axial tilt and up to +/- 30 deg latitude shift contributes negligible seasonal change to day/night cycle  
155: Estimate based upon past NASA DRM exercises.  
156: Estimate based upon past NASA DRM exercises.

#### Lunar Poles

157: LRO Diviner Lunar Radiometer Experiment – temperature maps corresponding to permanently shadowed part of Hermite Crater. Night common max = -352F  
158: LRO Diviner Lunar Radiometer Experiment – temperature maps corresponding to rims of some crater within 5 deg latitude of poles – very rare; More common polar max: Day -100F; Night -290F.  
159: Calculated by author assuming 90 deg zenith angle at crater bottom  
160: Calculated by author by extrapolating to 1.5 deg axial tilt using 4th order polynomial on lunar plains data; all crater locations colder  
161: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)  
162: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)  
163: Assumed Lunar = 1.0 +/- 1 standard deviation; more data needed to compare Apollo visited sides to entire Lunar surface  
164: Assumed Lunar = 1.0 +/- 1 standard deviation; more data needed to compare Apollo visited sides to entire Lunar surface  
165: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only. Realistic value for Lunar bombardment as majority of wake side data is micrometeorite impacts  
166: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; considered worst case for low junk Lunar orbit. LDEF data was 5:1 junk to micrometeorite ratio  
167: Simonsen, Lisa, Nealy, J.E., Radiation Protection for Human Missions to the Moon and Mars, NASA TP-3079, Feb 1991. GCR contribution at solar minimum  
168: Simonsen, Lisa, Nealy, J.E., Radiation Protection for Human Missions to the Moon and Mars, NASA TP-3079, Feb 1991. Assuming normal distribution using minimum and average values  
169: "The lunar atmosphere: History, status, current problems, and context"; Reviews of Geophysics 37 453. Scientifically non-zero but a complete vacuum for practical engineering purposes  
170: "The lunar atmosphere: History, status, current problems, and context"; Reviews of Geophysics 37 453. Scientifically non-zero but a complete vacuum for practical engineering purposes  
171: "Farside Gravity Field of the Moon from Four-Way Doppler Measurements of SELENE"; Science 323 900. Taking average and applying 350mGal deviation for gravity anomalies

172: "Farside Gravity Field of the Moon from Four-Way Doppler Measurements of SELENE"; Science 323 900. Taking average and applying 350mGal deviation for gravity anomalies  
173: "Density and Porosity of Apollo Lunar Basalts and Breccias" - converted density from grain to bulk. 22 Apollo samples from different sites - Min corresponds to Imbrium Ejecta, Impact-Melt Breccia  
174: "Density and Porosity of Apollo Lunar Basalts and Breccias" - converted density from grain to bulk using included porosity. 22 Apollo samples from different sites - Max corresponds to Titanium-included Basalts  
175: Lunar Prospector electron reflectometer experiment; corresponds to 0.1 nT from magnetic field map  
176: Lunar Prospector electron reflectometer experiment; corresponds to 100 nT from magnetic field map  
177: The Lunar Base Handbook, Eckart 2006 pp 118; Corresponds to 88deg latitude location during Lunar winter, which accounts for 60km range from Lunar Pole  
178: The Lunar Base Handbook, Eckart 2006 pp 118; Corresponds to 6-month day/night cycle; some near-constant (94%) sunlit areas at pole due to local geography  
179: The Lunar Base Handbook, Eckart 2006 pp 118; Corresponds to 88deg latitude location during Lunar summer, which accounts for 60km range from Lunar Pole  
180: The Lunar Base Handbook, Eckart 2006 pp 118; Corresponds to 6-month day/night cycle; some permanent shadowed areas at pole due to local geography  
181: Estimate based upon past NASA DRM exercises.  
182: Estimate based upon past NASA DRM exercises.

#### Phobos Surface

183: "Compositional Interpretation of PFS/MEx and TES/MGS Thermal Infrared Spectra of Phobos" Giuranna 2010; Minimum temperatures seen by MGS and MEx satellites - 130K  
184: "Compositional Interpretation of PFS/MEx and TES/MGS Thermal Infrared Spectra of Phobos" Giuranna 2010; Maximum temperatures seen by MGS and MEx satellites - 353K  
185: Calculated by the author by deriving Phobos IR of 183 W/m<sup>2</sup> from average surface temperatures; 0.95 emissivity. Other assumptions: night side, no vehicle/Mars IR, 0.5 orthogonal view factor  
186: Calculated by the author by deriving Phobos IR of 183 W/m<sup>2</sup> from average surface temperatures; 0.95 emissivity. Other assumptions: subsolar with 0.07 albedo, 445 W/m<sup>2</sup> vehicle IR, max Mars IR of 470 W/m<sup>2</sup>  
187: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)  
188: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)  
189: Assumed similar bounding cases to Asteroid. Could be 1 or more meter of very fine dust due to meteoric bombardment - would explain optical and thermal properties; however, no in-situ measurements

190: Assumed similar bounding cases to Asteroid. Could be 1 or more meter of very fine dust due to meteoric bombardment - would explain optical and thermal properties; however, no in-situ measurements

191: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only -50%; realistic value for micrometeorite bombardment. Could be higher due to theorized Mars dust ring

192: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only +50%; realistic value for micrometeorite bombardment. Could be higher due to theorized Mars dust ring

193: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

194: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

195: "Human Exploration of Phobos and Deimos: Radioprotection Issues." Vasquez 2007 – too small mass, no atmosphere

196: "Human Exploration of Phobos and Deimos: Radioprotection Issues." Vasquez 2007 – too small mass, no atmosphere

197: "Working models for the gravity field and dynamical environment of Phobos." Shi 2011. Accounts for tidal affects from Mars, which can be equivalent to gravitational effects. Also accounts for centrifugal effects.

198: "Working models for the gravity field and dynamical environment of Phobos." Shi 2011. Accounts for tidal affects from Mars, which can be equivalent to gravitational effects. Also accounts for centrifugal effects.

199: "Arecibo radar observations of Phobos and Deimos" Busch 2006. Corresponds to  $1.6 \pm 0.3$  g/cm<sup>3</sup> for near-surface bulk density. Porosity nears 40%

200: "Arecibo radar observations of Phobos and Deimos" Busch 2006. Corresponds to  $1.6 \pm 0.3$  g/cm<sup>3</sup> for near-surface bulk density. Porosity nears 40%

201: "Experimental evidence of the Phobos magnetic field", Mordovskaya 2001

202: "Experimental evidence of the Phobos magnetic field", Mordovskaya 2001

203: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Orbital period of 7.7 hours - daily solar eclipses by Mars ~54min in length

204: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Corresponds to 140 days of constant sunlight at North Pole

205: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Orbital period of 7.7 hours - daily solar eclipses by Mars ~54min in length

206: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Corresponds to 140 days of constant shadow during polar winter

207: Estimate based upon past NASA DRM exercises.

208: Estimate based upon past NASA DRM exercises.

#### Deimos Surface

209: Calculated by the author by applying min/max range from Phobos and applying to Deimos maximum temperature. Similar optical properties.

210: "Infrared Spectra of Deimos and Phobos" Lynch 2007.

211: Calculated by the author by deriving Deimos IR of 156 W/m<sup>2</sup> from average surface temperatures; 0.95 emissivity. Other assumptions: night side, no vehicle/Mars IR, 0.5 orthogonal view factor

212: Calculated by the author by deriving Deimos IR of 156 W/m<sup>2</sup> from average surface temperatures; 0.95 emissivity; other assumptions: subsolar with 0.07 albedo, 445 W/m<sup>2</sup> vehicle IR, max Mars IR of 470 W/m<sup>2</sup>

213: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)

214: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)

215: Assumed similar bounding cases to Asteroid. Could be 1 or more meter of very fine dust due to meteoric bombardment - would explain optical and thermal properties; however, no in-situ measurements

216: Assumed similar bounding cases to Asteroid. Could be 1 or more meter of very fine dust due to meteoric bombardment - would explain optical and thermal properties; however, no in-situ measurements

217: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only -50%; realistic value for micrometeorite bombardment. Could be higher due to theorized Mars dust ring

218: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only +50%; realistic value for micrometeorite bombardment. Could be higher due to theorized Mars dust ring

219: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

220: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs

221: "Human Exploration of Phobos and Deimos: Radioprotection Issues." Vasquez 2007 – too small mass, no atmosphere

222: "Human Exploration of Phobos and Deimos: Radioprotection Issues." Vasquez 2007 – too small mass, no atmosphere

223: Calculated by author; only accounts for gravitational effects. Mars tidal effects and centrifugal effects much smaller than Phobos

224: Calculated by author; only accounts for gravitational effects. Mars tidal effects and centrifugal effects much smaller than Phobos

225: "Arecibo radar observations of Phobos and Deimos" Busch 2006. Corresponds to  $1.1 \pm 0.3$  g/cm<sup>3</sup> for near-surface bulk density. Porosity nears 40%

226: "Arecibo radar observations of Phobos and Deimos" Busch 2006. Corresponds to  $1.1 \pm 0.3$  g/cm<sup>3</sup> for near-surface bulk density. Porosity nears 40%

227: No data available

228: From 1989 Phobos-2 data; assume Deimos magnetic field is less than Phobos

229: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Orbital period of 30 hours - daily solar eclipses by Mars ~84min in length  
230: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Corresponds to 300 days of constant sunlight at North Pole  
231: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Orbital period of 30 hours - daily solar eclipses by Mars ~84min in length  
232: "Comparison of Deimos and Phobos as Destinations for Human Exploration, and Identification of Preferred Landing Sites". AIAA Hopkins 2011. Corresponds to 300 days of constant sunlight at North Pole  
233: Estimate based upon past NASA DRM exercises.  
234: Estimate based upon past NASA DRM exercises.

#### Mars Orbit

235: No surface to reference  
236: No surface to reference  
237: Calculated by the author using data from Thermal Environments JPL D-8160. Assumptions: night side, no vehicle IR, planetary IR of 30 W/m<sup>2</sup> (polar caps) at 500km with 0.5 orthogonal view factor  
238: Calculated by the author using data from Thermal Environments JPL D-8160. Assumptions: day side of Mars with 0.17 albedo, 445 W/m<sup>2</sup> vehicle IR, max Mars IR of 470 W/m<sup>2</sup> at 100km  
239: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)  
240: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)  
241: Captured under high energy debris  
242: Captured under high energy debris  
243: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only -50%; realistic value for micrometeorite bombardment. Could be higher due to theorized Mars dust ring  
244: LDEF – 6 years, 1300 incidents/m<sup>2</sup>; wake side only +50%; realistic value for micrometeorite bombardment. Could be higher due to theorized Mars dust ring  
245: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs  
246: Cucinotta, F.A., M. Durante, Risk of Radiation Carcinogenesis, Human Health and Performance Risks of Space Exploration Missions, 2010. General assumptions for interplanetary space not including SPEs  
247: No atmosphere at this location  
248: No atmosphere at this location  
249: Orbital trajectory assumed  
250: Orbital trajectory assumed

251: No surface to reference  
252: No surface to reference  
253: "Initial Pioneer Venus Magnetic Field Results: Nightside Observations" Russell 1979. Science 205 114. Assumed dark side Venus data and reduced magnetic field to Mars distance  
254: "Magnetic field of Mars: Summary of results from the aerobraking and mapping orbits." Acuna 2001. Corresponds to 5nT, max ambient magnetic field seen by Mars Global Surveyor below ionosphere (170-200km)  
255: Assumed LEO-like orbit (Mars diameter 11% that of Earth – will likely be longer)  
256: High enough orbit could provide near-constant sunlight with only brief eclipse periods  
257: High enough orbit could provide near-constant sunlight with only brief eclipse periods  
258: Assumed LEO-like orbit (Mars diameter 11% that of Earth – will likely be longer)  
259: Estimate based upon past NASA DRM exercises.  
260: Estimate based upon past NASA DRM exercises.

#### Mars Equator

261: Mars Global Surveyor Thermal Emissions Spectrometer data - Estimated from thermal distribution videos, corresponds to night during winter near the equator (estimated -115C)  
262: Mars Global Surveyor Thermal Emissions Spectrometer data - Estimated from thermal distribution videos, corresponds to subsolar location during the day (estimated 20C)  
263: Calculated by author, assuming 0 solar, planetary IR 31.8 W/m<sup>2</sup> calculated from min surface temperature, 0.5 orthogonal view factor. Will likely be closer to -175F night surface temperature due to atmospheric effects  
264: Calculated by author, assuming max equatorial IR, subsolar, vehicle IR of 445 W/m<sup>2</sup> with 0.5 orthogonal view factor and 0.32 albedo. Will likely be lower due to atmospheric effects  
265: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)  
266: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)  
267: Assumed by the author as no better than Lunar. Although weathering/higher gravity, wind poses issue. Higher velocities, can't escape by working slower, dust will get places it likely wouldn't in Lunar  
268: Assumed by the author as marginally worse than Lunar. Although weathering/higher gravity, wind poses issue. Higher velocities, can't escape by working slower, dust will get places it likely wouldn't in Lunar  
269: Assumed by the author to be near-zero due to atmosphere.

270: Assumed by the author from LDEF data – 6 years, 1300 incidents/m<sup>2</sup>. As most debris was 0.5-1.0 mm diameter, most would burn up in atmosphere. LDEF data reduced by 2 orders of magnitude

271: Saganti, et al. Radiation Climate Map for Analyzing Risks to Astronauts on the Mars Surface From Galactic Cosmic Rays, 2004. Assumes solar min at low elevation (Hellas Planatia)

272: Simonsen, Lisa, Nealy, J.E., Radiation Protection for Human Missions to the Moon and Mars, NASA TP-3079, Feb 1991, p18. Assuming annual GCR dose and one large solar flare on the surface of Mars

273: NASA Planetary Fact Sheet – given minimum

274: NASA Planetary Face Sheet – given maximum

275: Mars Global Surveyor Laser Altimeter data – taking average and applying +700/-400 mGal delta for gravity anomalies

276: Mars Global Surveyor Laser Altimeter data – taking average and applying +700/-400 mGal delta for gravity anomalies

277: "In situ observations of the physical properties of the Martian surface." Herkenhoff 2008. Minimum from all in-situ measurements to date

278: "In situ observations of the physical properties of the Martian surface." Herkenhoff 2008. Maximum from all in-situ measurements to date

279: "Magnetic field of Mars: Summary of results from the aerobraking and mapping orbits." Acuna 2001. Estimated from magnetic field map near -30deg latitude. Closer to equator, closer to -0.0005G

280: "Magnetic field of Mars: Summary of results from the aerobraking and mapping orbits." Acuna 2001. Estimated from magnetic field map near equator

281: Calculated by the author. Corresponds to half a Martian day of 24.6 hours ± 2 hours for latitude and seasonal effects

282: Calculated by the author. Corresponds to half a Martian day of 24.6 hours ± 2 hours for latitude and seasonal effects

283: Calculated by the author. Corresponds to half a Martian day of 24.6 hours ± 2 hours for latitude and seasonal effects

284: Calculated by the author. Corresponds to half a Martian day of 24.6 hours ± 2 hours for latitude and seasonal effects

285: Estimate based upon past NASA DRM exercises.

286: Estimate based upon past NASA DRM exercises.

#### Mars Poles

287: Mars Global Surveyor Thermal Emissions Spectrometer data – corresponds to -130C, minimum from this data set near the poles

288: Mars Global Surveyor Thermal Emissions Spectrometer data - Estimated from thermal distribution videos, corresponds to pole at summer during the day (estimated -15C)

289: Calculated by the author, assuming 0 solar, planetary IR 21.3 W/m<sup>2</sup> calculated from min surface temp of -202F and Mars emissivity of 0.9, 0.5 orthogonal view factor. Will likely be closer to -202F night surface temp due to atmospheric effects

290: Calculated by the author, assuming max polar IR of 241 W/m<sup>2</sup> from max 5F temperature and water ice emissivity of 0.96, 30 deg zenith angle, next to 445 W/m<sup>2</sup> vehicle IR, 0.5 orthogonal view factor and 0.32 albedo. Will likely be closer to 5F surface temperature during the day due to atmospheric effects.

291: Calculated by author assuming vehicle optical characteristics (0.15 absorptivity; 0.1 reflectivity)

292: Calculated by author assuming vehicle optical characteristics (0.42 absorptivity; 0.885 reflectivity)

293: Assumed by the author as no better than Lunar. Although weathering/higher gravity, the wind poses a difficult issue. Higher velocities, can't escape by working slower, dust will get places it likely wouldn't in Lunar.

294: Assumed by the author as marginally worse than Lunar. Although weathering/higher gravity, the wind poses issue. Higher velocities, can't escape by working slower, dust will get places it likely wouldn't in Lunar

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296: Assumed by the author from LDEF data – 6 years, 1300 incidents/m<sup>2</sup>. As most debris was 0.5-1.0 mm diameter, most would burn up in atmosphere. LDEF data reduced by 2 orders of magnitude

297: Saganti, et al. Radiation Climate Map for Analyzing Risks to Astronauts on the Mars Surface From Galactic Cosmic Rays, 2004. Assumes solar min at low elevation (Hellas Planatia)

298: Simonsen, Lisa, Nealy, J.E., Radiation Protection for Human Missions to the Moon and Mars, NASA TP-3079, Feb 1991, p18. Assuming annual GCR dose and one large solar flare on the surface of Mars

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303: "In situ observations of the physical properties of the Martian surface." Herkenhoff 2008. Minimum from all in-situ measurements to date

304: "In situ observations of the physical properties of the Martian surface." Herkenhoff 2008. Maximum from all in-situ measurements to date

305: "Magnetic field of Mars: Summary of results from the aerobraking and mapping orbits." Acuna 2001. Estimated from magnetic field map at -80 deg latitude

306: "Magnetic field of Mars: Summary of results from the aerobraking and mapping orbits." Acuna 2001. Estimated from magnetic field map near south pole. North pole is closer to 0.

307: Corresponds to half a Martian year of constant shade at the poles

308: Corresponds to half a Martian year of constant sunlight at the poles

309: Corresponds to half a Martian year of constant sunlight at the poles

310: Corresponds to half a Martian year of constant shade at the poles

311: Estimate based upon past NASA DRM exercises.



312: Estimate based upon past NASA DRM exercises.

## Acknowledgments

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