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Mars Direct: A Simple, Robust, and Cost Effective Architecture for the Space Exploration Initiative

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ABSTRACT

The concept of a coherent Space Exploration Initiative (SEI) architecture is defined and is shown to be largely unsatisfied by the conventional Earth-orbital assembly/Mars orbital rendezvous mission plan that has dominated most recent analysis. Coherency's primary requirements of simplicity, robustness, and cost effectiveness are then used to derive a secondary set of mission features that converge on an alternative mission architecture known as "Mars Direct."

In the Mars Direct plan two launches of a heavy lift booster optimized for Earth escape are required to support each 4 person mission. The first booster launch delivers an unfueled and unmanned Earth Return Vehicle (ERV) to the martian surface, where it fills itself with methane/oxygen bipropellant manufactured primarily out of indigenous resources. After propellant production is completed, a second launch delivers the crew to the prepared site, where they conduct extensive regional exploration for 1.5 years and then return directly to Earth in the ERV. No on-orbit assembly or orbital rendezvous is required in any phase of the mission, and the same set of booster, crew hab, and ERV used to support Mars missions can also be used to support a lunar base.

This paper discusses both the martian and lunar forms of implementation of the Mars Direct architecture. Candidate vehicle designs are presented, and the means of performing the required in-situ propellant production is explained. The in-situ propellant process is also shown to present very high leverage for a Mars Rover Sample Return mission flown as a scaled down precursor version of the manned Mars Direct. Methods of coping with the radiation and zero gravity problems presented by a manned Mars mission are discussed. Prime objectives for Mars surface exploration are outlined, and the need for substantial surface mobility is made clear. Combustion powered vehicles utilizing the in-situ produced methane/oxygen are proposed as a means for meeting the surface mobility requirement. While the Mars Direct plan can be implemented utilizing only chemical propulsion, it is found that substantial improvement in mission capability can be achieved through the introduction of nuclear thermal rocket propulsion, and that the architecture is highly amenable to the introduction of such technology whenever it becomes available. It is concluded that the Mars Direct architecture offers an attractive means of rapidly realizing a coherent SEI, thereby opening the doors of the solar system to humanity.

INTRODUCTION:
Defining a Coherent Space Exploration Initiative

The need currently exists for a coherent architecture for the Space Exploration Initiative (SEI). By a coherent architecture what is meant is a clear and intelligent set of objectives and a simple, robust, and cost-effective plan for accomplishing them. The objectives chosen should offer the maximum payoff, and their accomplishment should enhance our ability to achieve still more ambitious objectives in the future. The plan, in order to be simple, robust, and low cost, should not make inter-dependant missions (i.e. lunar, Mars, and Earth orbital) that have no real need to be dependant on each other. The plan should, however, employ technology that is versatile enough to play a useful role across a wide range of objectives, so as to reduce costs through commonality of hardware. Finally, and most importantly, technologies must be chosen that maximize the effectiveness of the mission at the planetary destination. It is not enough to go to Mars; it is necessary to be able to do something useful when you get there. Zero capability missions have no value.

While the above principles may appear to be common sense, they were violated in every particular by many recent SEI studies[1], and as a result, a picture has been presented of SEI that is so costly and unattractive that congressional funding of the program is very much in doubt. Such architectures have driven costs through the roof by employing totally different launch vehicles for the

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Moon and Mars, totally different space transfer vehicles and propulsion technologies for the Moon and Mars, totally different excursion vehicles for the Moon and Mars, a completely artificial dependance of the Mars missions on the lunar missions, and a requirement to base the lunar missions on a massive orbital assembly, refueling, and refitting infrastructure at Space Station Freedom. Furthermore, both the Lunar and Mars missions studied have been close to zero capability, with no serious attempt made to provide surface mobility, and with Mars explorers spending less than 5% of the total Earth-Mars round trip mission transit time on the surface of the Red Planet.

Meeting the demands of coherence drives the design of the SEI architecture in certain very definite directions. To wit:

1. Simplicity and Robustness require that the Lunar and Mars missions not depend upon any LEO infrastructure. In addition to being tremendously costly to develop, build, and maintain, such infrastructure is intrinsically unreliable, and difficult to repair, and its use adds risk to all planetary missions based on it due to the difficulty in verifying quality control of any space-based construction. The demand for the elimination of LEO infrastructure argues in favor of using both advanced propulsion and/or indigenous propellants, both of which can contribute to reducing mission mass to the point where no on-orbit assembly is required.

2. Low Cost requires that the same launch vehicles, space transfer vehicles and propulsion technology, and to the extent possible, excursion vehicles be used for both the Moon and Mars, as well as other destinations. Low cost also demands the elimination of LEO infrastructure, as the potential cost savings made possible by re-use of space transfer vehicles at such infrastructure are insufficient to balance the cost of the infrastructure. This can be seen by noting that the cost of such infrastructure is currently estimated to be about 3 orders of magnitude larger than the value of the vehicle hardware elements (engines, avionics) that it would be able to save with each space-based refit. Thus, about 1000 refitted missions would be required before such a facility broke even - a somewhat distant prospect. Low Cost also demands that the most cost-effective trajectories be taken at all times (i.e. conjunction class trajectories for Mars), and that an initial group of opposition class Mars missions using completely different hardware from the main sequence of conjunction class missions not be undertaken.

3. High Effectiveness requires that the astronauts be endowed with three essential elements once they reach their destination. These three essentials are:

a) Time
b) Mobility
c) Power

Time is obviously required if the astronauts are to do any useful exploration, construction, or resource utilization experimentation on the surface of the destination planet. This clearly means that opposition class Mars missions (which involve 1.5 year flight times and 20 day surface stays) are out of the question. It also means that architectures involving Lunar or Mars orbital rendezvous (LOR, MOR) are very undesirable, for the simple reason that if the surface stay time is long, so is the orbit time. The LOR or MOR architectures are therefore left in a predicament of whether to leave someone in the mothership during the extended surface stay, exposed to cosmic rays and the rigors of zero-gravity conditions and accomplishing nothing; or leave the mothership unmanned for an extended period and have the returning crew trust to fate that it will be ship-shape when they return. If it isn't, their predicament may be hopeless. The alternative to LOR and MOR architectures are those that employ direct return to Earth from the planets' surface. This is possible to do on a Lunar mission with all terrestrial produced propellents, however the mission capability is greatly enhanced if Lunar produced LOX can be used for the return. Direct return from the surface of Mars absolutely demands that indigenous propellants be used.

Mobility is absolutely required if any useful exploratory work is to be accomplished on a body the size of Mars or even the Moon. Mobility is also needed to transport natural resources from distant locations to the base where they can be processed, and is also required to enable crews to visit distant assets, such as optical and radiotelescopic arrays on the Moon. The key to mobility on both the Moon and Mars is the generation of indigenous propellants for use in both high powered ground rovers and rocket propelled flight vehicles. On the Moon the resource of choice is Lunar LOX, which can be burnt with terrestrial fuels such as hydrogen or methane. On Mars, chemical fuel and oxidizer combinations such as methane/oxygen or carbon monoxide/oxygen can be produced for both surface and flight vehicles, and in addition rocket thrust for flight propulsion can also be produced by using raw carbon dioxide propellant heated in a nuclear thermal rocket engine.

Power can be generated in the large amounts required for indigenous propellant production on both the Moon and

Mars only by using nuclear reactors. Once indigenous propellants have been produced, they form a very convenient mechanism for storing the nuclear energy, thus providing explorers with mobile power, for example by running a 100 kWe generator off the internal combustion engine on a ground rover. The presence of a power rich environment, both at the base and at remote sites, is essential to allow the astronauts to pursue a wide variety of scientific and resource utilization activities.

We thus see that the requirements for simplicity, robustness, low cost, and high effectiveness drive SEI toward an architecture utilizing direct launch to the moon or Mars with a common launch and space transfer system, and direct return to Earth from the planet's surface utilizing indigenous propellants, which are also used to provide surface mobility and mobile power. One such coherent architecture that has been devised is known as "Mars Direct."[2,3]

THE MARS DIRECT ARCHITECTURE

The Mars side of the "Mars Direct" architecture works as follows (fig.1). An an early mission opportunity, for example, Dec. 1996, a single shuttle derived "Ares" heavy lift launch vehicle with a substantial cryogenic upper stage lifts off the Cape and hurls onto direct trans-Mars injection an unmanned 40 tonne payload. This payload consists of an unfueled methane/oxygen driven two-stage ascent and Earth return vehicle (ERV), 6 tonnes of liquid hydrogen cargo, a 100 kWe nuclear reactor mounted in the back of a methane/oxygen driven light truck, a small set of compressors and automated chemical processing unit, and a few small scientific rovers. This payload aerobrakes

Fig. 1 The Mars Direct mission sequence. Every two years two boosters are launched. One sends an unmanned ERV to fuel itself with methane and oxygen manufactured on Mars at a new site, the other sends a manned hab to rendezvous with an ERV at a previously prepared site.

into orbit around Mars and then lands with the help of a parachute. As soon as it is landed, the truck is telerobotically driven a few hundred meters away from the lander, and the reactor is deployed to provide power to the compressors and chemical processing unit. The hydrogen brought from Earth is quickly catalytically reacted with Martian CO_2 to produce methane and water, thus there is no need for long term storage of cryogenic hydrogen on the martian surface. The methane is liquefied and stored, and the water electrolysed to produce oxygen, which is stored, and hydrogen, which is recycled through the methanator. Ultimately these two reaction (methanation and water electrolysis) produce 24 tonnes of methane and 48 tonnes of oxygen. An additional 36 tonnes of oxygen is produced via reduction of additional Martian CO_2. The total bipropellant produced is 107 tonnes, or a leverage of 18:1 compared to the hydrogen brought from Earth needed to produce it. Ninety-six tonnes of the bipropellant will be used to fuel the ERV, while 11 tonnes are available to support the use of high powered chemically fueled long range ground vehicles.

The propellant production having been successfully completed, in 1999 two more Ares HLLVs lift off the Cape and throw their 40 tonne payloads onto trans-Mars injection. One of the payloads is an unmanned fuel-factory/ERV just like the one launched in 1996, the other is a habitation module containing a crew of 4, provisions for 3 years, a pressurized methane/oxygen driven ground rover, and an aerobrake/landing engine assembly. Artificial gravity is provided to the crew on the way out to Mars by tethering off the burnt out Ares upper stage and spinning up at 1 rpm. The manned craft lands at the 1997 landing site where a fully fueled ERV and fully characterized and beaconed landing site await it.

Fig. 2. Mars Direct surface base. Shown is the two deck disc-shaped hab module, the two-stage conical methane/oxygen driven ERV, an inflatable greenhouse, and a pressurized ground rover. A 100 kWe nuclear reactor positioned in the crater in the background has long since completed its job of driving the production of the ERV's propellant supply, and now provides a copious source of power to the base, with backup power available from solar panels. (Painting by Robert Murray)

This surface rendezvous plan has several levels of backup available to assure mission success. First of all we have the advance characterization of the site by 2 years of local robotic exploration out of the ERV, with placement of a transponder on the best possible landing site identified in the vicinity. The ERV also mounts a radio beacon much like an ILS transmitter at an airport giving the crew exact position and velocity data during approach and terminal landing. It may be noted that both Viking landers touched down within 30 km of their targeted sites without active guidance. With the aid of a feedback targeting control system, a guiding radio beacon, and a pilot to fly terminal descent, the landing should be within a few meters of the targeted point. However if the landing should prove inaccurate by tens or even hundreds of kilometers, surface rendezvous can still be achieved through the used of the pressurized methane/oxygen internal combustion driven ground rover carried in the hab, which has a one-way surface range of up to 1000 km. If the landing rendezvous fails by distances of planetary dimensions, the second ERV following the manned hab out to Mars can be redirected to the manned landing site to provide a third level of backup. However, assuming the surface rendezvous is accomplished as planned and the ERV checks out, the second ERV will be landed several hundred miles away to start making propellant for the 2001 mission. Thus every other year 2 HLLVs are launched, for an average launch rate of 1 HLLV per year to pursue a continuing program of Mars exploration.

An artists conception of what the first Mars Direct surface base might look like is shown in fig. 2. An extensive descriptions of the vehicle elements shown is given in reference 3.

The crew stays on the surface for 1.5 years, taking advantage of the mobility afforded by the high powered chemically driven ground vehicles to accomplish a great deal of surface exploration. With 11 tonnes of methane/oxygen bipropellant allocated for surface operations, about 22,000 ground kilometers can be traversed, ranging up to 500 km out from the base. Thus each mission can explore an area of approximately 800,000 square kilometers, which is roughly the size of the state of Texas. At the conclusion of their stay, the crew returns to Earth in a direct flight from the martian surface in an ERV. All personnel sent to Mars thus spend all of the stay time at Mars on the surface where they can be shielded from cosmic radiation and have natural gravity. No one is ever left in orbit. As the series of missions progresses, a string of small bases is left behind on the Martian surface, opening up broad stretches of territory to human cognizance.

LUNAR MISSION PLAN

The Mars Direct vehicle systems can also be used to accomplish Lunar missions in the following way. First, an Ares booster launch is used to throw a 59 metric ton payload consisting of a standard hab module plus a cryogenic Lunar orbital capture and lunar descent (LOC/LD) stage onto trans-lunar injection. The LOC/LD stage is then used to land the hab on the Moon. After one or more such habs have been thus enplaced at a given site, the crew is flown out to the Moon within a Mars Direct ERV. The ERV in this case has its (Mars Ascent) first stage deleted, but its second stage is fueled with methane/oxygen bipropellant, and this provides sufficient thrust and delta-V for an Earth return direct from the lunar surface. Landing on the Moon at the prepared site is accomplished with the aid of the same cryogenic LOC/LD module used to land habs and cargo. After landing, the crew exits the ERV and enters the pre-landed hab(s) and proceeds to operate on the Lunar surface for an extended period, after which they re-enter the ERV and execute a direct return to low Earth orbit (LEO).

Prior to Lunar liquid oxygen (LOX) becoming available, the mass of the fully fueled Mars Direct ERV exceeds the Lunar delivery capability of a single Ares launch by 5 tonnes. This problem could be resolved by scaling down the lunar ERV by 20% compared to the Mars version. However if hardware commonality with the Mars ERV is desired, a simple solution would be to pre-land a cargo flight of liquid oxygen at the chosen site. Since such a cargo flight could land about 21 tonnes of LOX, one such flight could support 4 manned missions to that destination (the ERVs would fly out fully fueled with methane and 2/3 fueled with LOX), plus any number of cargo flights. At the conclusion of these missions, Lunar LOX production could be in place and Lunar LOX available at the site, eliminating the need for any further LOX delivery flights.

An artist's conception of a Lunar base supported by Mars Direct transportation hardware is depicted in fig. 3. While the upper stage of the Mars Direct ERV appears to be oversized for a Lunar mission ERV, it is well suited to support the large crew rotations that will be required by a Lunar base.

This Lunar mission architecture has many advantages. First of all, no Lunar orbit rendezvous (LOR) is required. This is almost essential for a Lunar base mission (as opposed to Apollo short-stay missions), as it would be unthinkable to have astronauts exposed to zero-gravity and cosmic radiation in a Lunar orbiting craft while awaiting the return of a 6 to 12 month duration lunar excursion crew, and on the other hand landing the entire crew would l

Fig. 3. Mars Direct Vehicle elements being used to construct a Lunar base. The hab modules can also serve as cargo carriers and are flown one way to the Lunar surface. Only the upper stage of the methane/oxygen propelled ERV is delivered to the Moon. The astronauts will fly out in the ERV and use it for a direct return to LEO. No Lunar Orbit Rendezvous is required. (Painting by Robert Murray)

leave the vital Earth return mothership in an unstable Lunar orbit for an extended period with no-one minding the store. If its condition were to deteriorate, only limited corrective action could be taken the crew might be stranded; if a failure on board were to remain undetected until LOR, the returning crew would find itself in a very difficult situation. Secondly, the use of methane/oxygen for direct return from the Moon has advantages compared to either the hydrogen/oxygen or conventional storable bipropellant (NTO/MMH) alternatives. Compared to hydrogen, methane is almost indefinitely storable on the Lunar surface, thus facilitating extended stays. Methane/oxygen has a higher specific impulse than NTO/MMH (373 s compared to 343 s), however, once Lunar oxygen is introduced, this advantage becomes greatly multiplied and the effective Isp of the methane/oxygen system exceeds 1700 s. What this means is that using NTO/MMH, 25 tons must be landed on the Moon to return a 10 ton ERV to LEO, while using methane and lunar oxygen, only 13 tons need be landed. Thus 12 additional tons of cargo can be flown to the Lunar surface with every manned flight, which means that if one crew and 1 cargo flight are flown each year, the total cargo delivered has been increased 50%. Finally, the "Lunar Direct" architecture has the key advantage of being totally coherent with the Mars Direct architecture, using the same vehicles. Thus experience with Mars systems can be obtained in near-Earth space on lunar missions, and overall Space Exploration Initiative program costs can be greatly reduced through reduction in the number of elements of the total space transportation architecture.

The complete set of vehicles required for this combined Lunar/Mars Direct architecture is shown in Fig. 4. While no detailed costing of this architecture has been done, it is difficult to imagine how a Space Exploration Initiative based on such a limited number of hardware elements could cost anything resembling the $400 billion price tag frequently cited for more conventional architectures..

Fig. 4 The complete space transportation system required to accomplish both the lunar and Mars objectives of SEI using the basic Mars Direct architecture. The high degree of vehicle commonality can help keep SEI costs to a minimum.

THE ARES LAUNCH VEHICLE

The Mars Direct launch vehicle, known as an "Ares," is shown in fig. 5. The launch vehicle's design is optimized for Earth escape missions. From past Shuttle-Z studies[4] four SSME's are highly desirable for the core stage. This increases thrust and reduces mass after the solid rocket boosters (SRB's) separate - critical when using a heavy upper stage. The in-line upper stage performs the same role as the S-IVB did on the Saturn-V.

It completes the burn into LEO and then is restarted several orbits later and injects the payload onto a heliocentric trans-Mars orbit. The external tank is significantly modified from a standard shuttle tank. It has an elliptical top dome on the oxygen tanks instead of ogive and has a thicker barrel section to support the upper stage, payload, and shroud. The loads from these three elements all flow into the top dome/barrel intersection. The hydrogen tank is stiffened around the SSME engine pod and a new bipod is placed on the side of the barrel section to support pitching loads of the engine pod. Both oxygen and hydrogen tanks retain the same capacity as a standard external tank.

Optimization of the Ares for a versatile range of missions leads to the choice of a 250 klb thrust engine for the upper stage. Increasing thrust decreases trans-Mars injection performance because of the increase in inert engine mass. Decreasing thrust only slightly decreases TMI performance because of higher gravity losses--both suborbital and during the escape burn. 250 klbs is selected to enable good thrust-to-mass ratios for heavier Lunar and LEO missions; however, the same trans-Mars performance could be achieved with 7 RL-10 engines on the upper stage. Note that that the Ares can achieve about 25% more payload capability on Lunar missions (59 t) than Mars missions (47 t). This extra performance is needed to carry out a similar mission since Lunar missions cannot utilize aerobraking at the Moon and cannot achieve an 18:1 return propellant ratio using only Lunar derived oxygen.

This same launch vehicle has significant capability going only to LEO. The baseline vehicle can lift 121 t to a 300 km circular orbit. If less performance is needed, the upper stage can be omitted, giving a capability of about 75 t. Since a vehicle launching to LEO is heavier, it suffers

Ares Launch Vehicle Definitio

Payload Capabilities (All Weights in tonnes)	
Trans-Mars (C_3 = 15 km^2/sec^2)	**47.2**
Trans-Lunar (5 day transfer)	**59.1**
LEO (160 by 160 Nmi, 28.5 degrees)	**121.2**
LEO/NUS (160 by 160 Nmi, 28.5 deg)	**75.0**
Height (m)	**82.3**
Gross Mass (Without Payload)	**2,194.6**
Stage-0	
2 Advanced Solid Rocket Boosters	**1,214.5**
Stage-1	
External Tank (Including Residuals)	**35.6**
SSME Engine Pod (4 SSME's)	**28.6**
Usable Propellant in ET	**723.5**
Total SSME Thrust (kN, 104%)	**8,706**
Specific Impulse (sec)	**453**
Staging Relative Velocity (km/s) (LEO to Mars Range)	**4.2 to 5.5**
Stage-2 (Ignited Sub-Orbital)	
Usable Propellant	**158.8**
Inert Mass	**13.2**
Single Engine Thrust (kN)	**1,113**
Specific Impulse (sec)	**465**
Payload Fairing (ALS Design)	**20.4**

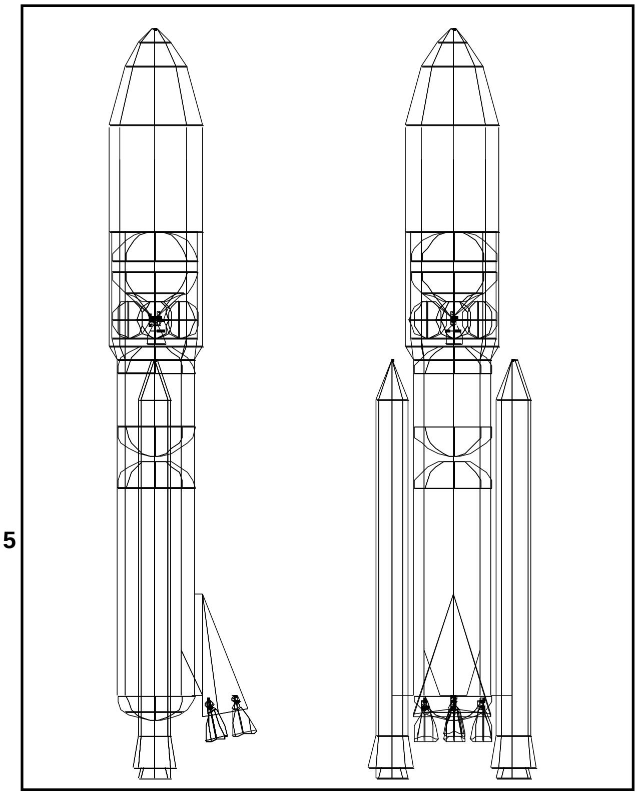

Fig 5. The Ares launch vehicle can send 47 tonnes on direct trans-Mars injection, 59 tonnes to trans-Lunar injection, or 121 tonnes to LEO.

more gravity loss. By adding another 250 klb thrust engine on the upper stage, the maximum LEO performance increases to about 130 t.

The Ares upper stage Isp was assumed to be 465 seconds and Advanced Solid Rocket Boosters were used. These should be available by the mid 1990s. If instead of these parameters a more conservative vehicle based upon SRBs and a 450 s upper stage is assumed, then the Ares can still deliver 40 t to TMI, 50 t to TLI, and 106 t to LEO.

The key feature of the Ares is its upper stage. Stage and a half vehicles based upon shuttle technology can also be designed to lift 121 t to LEO, provided they are equipped with about 6 SSMEs and 4 SRBs. Compared to such vehicles, then, the Ares trades off its upper stage with 250 klb thrust against 2 extra SRBs and 2 extra 500 klb SSMEs for equal LEO performance. This alone may be considered a fairly good argument for the Ares. Such Stage and a half vehicles, however, have no performance beyond LEO, and even experience a major degradation of performance if delivery to a high LEO (such as a nuclear safe 700 km altitude) orbit is required. The Ares, however, because it is staged, experiences comparatively little degradation in performance in sending payloads to high LEO orbits, and in fact can deliver substantial payloads direct to the Moon, Mars, GEO, or Jupiter, for that matter. The Ares' ability to use its upper stage to deliver payloads direct frees the entire SEI architecture from dependance upon a space-based space transfer vehicle, with its attendant show-stopping infrastructure of orbiting repair shops, cryogenic propellant depots, and so forth.

The Ares' fairing is 10 m in diameter. This allows it to accommodate either deployable flex-fabric aerobrake modules such as those shown in fig.4, or alternative rigid conical high L/D aeroshells. In neither case is there any need for on-orbit assembly.

TETHERED ARTIFICIAL GRAVITY SYSTEM

In the Mars Direct plan a tether is used to create artificial gravity on the Earth-to-Mars leg only. Shortly after the habitat is injected onto its trans-Mars trajectory the upper stage separates from the bottom of the crew habitat and maneuvers around to the other side of the habitat and begins moving away, pulling the tether off the habitat's roof as it goes. Once the stage has drawn out the full length of the tether it fires its reaction control thrusters and accelerates tangentially. This gradually pulls the tether taught and begins to create artificial gravity for both the habitat and the stage. When the stage reaches a relative velocity of 400 m/s its engines stop and all remaining cryogenic propellants are dumped. (The cryogenic propellants powered the main engines that put the vehicle on the trans-Mars trajectory, whereas the reaction control engines, using hydrazine, performed the tether extension and spin-up maneuver.) The tether is 1500 meters long and rotates at one RPM, giving the crew an acceleration of 0.38 earth-G (one Mars-G). The habitat is connected to the tether with a pyrotechnically releasable end fitting. This allows the tether to be rapidly dropped in the event of the onset of any unanticipated irremediable tether dynamic modes. Since the tether links the hab to only the burnt out Ares upper stage, and not a mission critical item, the tether can be dropped and the mission continued in a zero-gravity mode. As a matter of routine, the tether is dropped shortly before the hab begins its approach towards Mars aerocapture.

Tether design

The tether would consist of six flat-woven braids with interconnecting cords every meter. Separate braids allow up to three braid failures from meteor damage before the remaining braids would be overstressed and fail. Even if three braids were severed, the spin rate of the habitat-upper stage could be reduced to prevent overstressing the remaining braids. Weaving the braids into rectangular cross-sections also reduces the weakening affect of a meteor strike. Prior to launch the tether is wrapped around stanchions on the habitat roof. This minimizes the chance of tangling as the upper stage draws away, pulling the tether off the roof as it goes.

Tether Mission Operations

Having a rotating spacecraft travelling through interplanetary space presents many design challenges: How are maneuvers performed? How are communications maintained between Earth and the spacecraft? How is power gathered from the sun using solar arrays? How will navigation sensors view stars, Mars and its moons?

Maneuvers have been performed on spinning spacecraft before. Pioneer Venus Orbiter and the Pioneer Venus Probe Carrier were spinning, interplanetary spacecraft with precise targeting requirements at Venus. They used repeated, time-phased thruster firings to create a net delta-V perpendicular to the angular momentum vector. We would do much the same. In order not to increase or decrease our spin rate (and gravity), the thruster firing must be along the line of the tether or parallel to the angular momentum vector of the spinning system. Since the tether is taught, thruster firings that push the habitat toward the upper stage have the effect of reducing tether tension. As long as the thruster acceleration is less than the centripetal acceleration then the tether will stay taught. Trajectory correction maneuvers in deep space do not require large changes in velocity (typically about 20 m/s) and there is plenty of time to accumulate the change in velocity over several days. Since the tether-spacecraft system is rotating in a fixed plane then maneuvers in the rotation plane are done by timing when the thrusters fire, conversely, maneuvers out of the plane are done with continuous low-thrust burning, parallel to the angular momentum vector.

Communications would be achieved with a steerable high-gain antenna mounted on the roof of the habitat. To prevent radiating through the upper-stage counterbalance or the habitat, the spin orientation must be aligned so the Earth-spacecraft line is not perpendicular to the angular momentum. Figure 7 shows that the trajectory turns only about 130 degrees about the sun and that the Earth moves 30 degrees ahead of the line between the spacecraft and Sun. Hence, the Earth moves through an angle of about 160 degrees. This means a spin orientation which is about 10 degrees from edge-on at the beginning of the mission and will not have either Earth or Sun pass through the spin plane during the trip to Mars. Electrical power is produced with solar arrays that hang down, over the edge of the aerobrake on deployable booms. Each of the two panels has a gimbal to slowly track the sun as the angle between the spin plane and spacecraft-sun line changes. Navigation will be performed by observing the sun, stars, Phobos and Deimos, and Mars Itself. Either scanning sensors can be fixed to the habitat or trackers can be located on the same despun platform that the high gain antenna is located on.

MINI MARS DIRECT

The Mars Direct architecture is meant to enable a direct assault manned Mars mission by the turn of the century. In such an accelerated scenario, Mars precursors beyond the currently planned Mars Observer mission (1992), the

Soviet Mars 94 mission (1994), and the actual flight of the first unmanned ERV payload (as early as 1997) would be unlikely. However, if for funding or other reasons, a leisurely schedule was decided upon, then the implementation of a unmanned Mars Rover Sample Return (MRSR) mission would use the extra time to maximum advantage by providing mission planners with useful scientific and technological data.

Current plans for the MRSR mission[5] include a spectrum of scenarios, differing from each other in cost, complexity, technological risk, and scientific return. One of these, which has been termed the "Low Risk Scenario," (LRS) may be considered a baseline for alternative mission plan comparison. In the LRS plan, a Titan IV/Centaur launch vehicle is used to hurl the MRSR payload onto trans-Mars injection. Arriving at Mars, the payload aerocaptures into an elliptical Mars orbit, after which it enters the atmosphere and lands with the aid of a parachute. The surface payload includes a fully fueled 2.5 stage (drop tanks are used) ERV employing conventional storable bipropellants, a small science payload, and a micro-rover with strictly local (< 50 m range) capability. In the course of 1.5 years of surface operations, 5 kg of samples are collected by either the rover or a set of drag lines (analogous to fly-casting fishing gear, these are attached to the ERV as a backup to the rover) and loaded onto the ERV. About 1.5 years after the landing, Earth and Mars are correctly aligned for a minimum energy return, and at this time the ERV blast's off onto direct trans-Earth injection. Arriving at Earth, the sample return capsule performs a direct Apollo-type entry, and is snatched by an aircraft after its parachute has deployed.

An alternative to this plan is to use the same propellant production processes employed on the manned Mars Direct mission to provide leverage the MRSR mission. The advantages of using in-situ propellant production on the MRSR mission has been pointed out many times in the past.[6,7] We term the MRSR mission using a scaled down version of the Mars Direct propellant processes "Mini-Mars Direct" (MMD).

In the Mini-Mars Direct scenario, a Titan IV/Centaur launch vehicle is also used, however upon arrival at Mars the payload executes a Viking style propulsive capture into an elliptical Mars parking orbit, after which it enters the atmosphere and lands. The payload in this case consists of an unfueled ERV, and propellant production plant with a 800 We power source, 100 kg of liquid hydrogen, a large science payload, and a 400 kg long distance (up to 100 km range) ground roving vehicle. In the course of the 1.5 year surface stay, the return propellant is produced, and about 20 kg of samples (4 times the LRS mission return payload) is loaded into a 2 stage ERV (drop tanks are not required). At the conclusion of the surface stay, the ERV takes off onto direct trans-Earth injection, after which its sample capsule does an Apollo entry followed by parachute air-snatch just as in the LRS mission.

A comparison of the conventional LRS and the Mini-Mars Direct MRSR mission plans is given in Table 1.

Table 1. MRSR Mission Plan Comparison (masses in kg)

Payload	Low Risk Scenario	Mini-Mars Direct
ERV	2048 (wet, 2.5 stage)	300 (dry, 2 stage)
Lander	940	500
Extra Power	0	120
ISPP Plant	0	180
Hydrogen	0	100 (36% margin)
Drag Lines	20	20
Instruments	110	160 *Payoff*
Rover	10	400 *Payoff*
Capsule	6	20 *Payoff*

It can thus be seen that in addition to returning 4 times the sample, the MMD scenario also lands about 4 times the useful surface science payload as the LRS mission. Furthermore, aerocapture is not required. This is a very important point, as a Mars aerocapture has never been done, and therefore a MRSR mission plan incorporating this technology might only be considered truly "low risk" if a precursor Mars aerocapture test flight mission were flown in advance. However holding the MRSR mission in abeyance until such a mission could be flown and its data incorporated into the design of the MRSR mission could delay the MRSR mission by 5 to 10 years. Such a delay in turn would either prevent the MRSR mission from being flown in time to function as a useful precursor to the manned mission, or perhaps worse, cause the entire manned Mars exploration program to be stalled. Since the human (technical and managerial) infrastructure of SEI may reasonably be expected to cost about $5 billion a year (regardless of level of real activity), a programmatic delay of 5 years might end up costing the American taxpayers something like $25 billion. If aerocapture is removed from the MRSR mission, then an aerocapture precursor and the MRSR mission can be flown on parallel schedules and such an excessive programmatic delay avoided.

If, on the other hand, it is deemed acceptable to fly the MRSR mission with aerocapture without an aerocapture precursor, then the MMD plan can also utilize this technology, at which point its leverage over the LRS miss grows from its all-propulsive factor of 4 to about a factor of

10 in greater useful landed science. Alternatively, the MMD mission can sit tight in its all-propulsive incarnation, and be a lower-risk option than the LRS plan, while still preserving its 4 to 1 advantage.

It should also be noted that the availability of in-situ produced methane/oxygen bipropellant allows the use of internal combustion engines on the rover. This makes the 400 kg rover on the MMD mission a much more capable machine than a RTG powered of equal mass rover flown on a conventional mission. For example a 400 kg conventional rover might carry a 0.8 kWe RTG massing 180 kg. An internal combustion engine massing 40 kg could easily generate 40 kWe, for 50 times the power at less than 1/4 the weight. This means that the MMD rover could be a big-wheeled vehicle with capability for overcoming massive obstacles and thus accomplishing a degree of regional exploration that a RTG rover could not even approach. The MMD rover could thus serve as something like a 1/5 scale test vehicle for a manned pressurized rover, a task for which the feeble RTG powered model would be much less useful. The high power available would also enable a greatly increased rate of data transmission from the MMD rover, significantly enriching the science return from the mission and helping to reduce risk by giving ground controllers an order of magnitude improvement in detail of knowledge of mission events.

While the MMD mission requires about 0.8 kWe more power than the LRS mission in order to perform the propellant processing, the propellant once produced provides a ready medium for storing a massive amount of energy. This means that power for the MMD mission can be provided entirely by a solar array, which could be an advantage if political considerations make it desirable to fly the mission without the use of an RTG.

Finally, the MMD mission is an excellent technology precursor for the manned Mars Direct mission, demonstrating in the field the enabling exploration technology of in-situ propellant production. Since the MMD MRSR mission uses this technology itself to great advantage, it is not a representative of the unfortunate category of mission plans in which a technology demonstration is imposed as a requirement for a mission which could do just as well without that technology. The MMD plan thus represents a true symbiosis of ends and means, characteristics which are generally the case of a prosperous mission, and which mark it as a high potential candidate to be included within a coherent space exploration program.

METHANE AND OXYGEN PRODUCTION

The Mars Direct architecture utilizes the in-situ production of CH_4/O_2 bipropellant for Earth return and surface mobility from the very first mission. It can do this, because unlike the case of the Moon, the processes required to produce propellant on the surface of Mars are simple and very well understood. In fact, all of the chemical processes used in the Mars Direct plan have been in large scale use on Earth for over a century.

First there is the acquisition of the required native raw material. Since the hydrogen component of the bipropellant mixture represents only about 5% of the total propellant weight it can be imported from Earth. Heavy insulation of tanks with multi-layer insulation (MLI) can reduce in-space boiloff of liquid hydrogen to less than 1% per month during the 6 to 8 month interplanetary transit without any requirement for active refrigeration. Since the hydrogen raw material is not going to be directly fed into an engine, it can be gelled with a small amount of methane to prevent leaks. Gelling of the hydrogen cargo will also reduce boiloff further (as much as 40%) due to suppression of convection within the tank[8].

The only raw materials thus required from Mars are carbon and oxygen. The atmosphere of Mars, as measured by the 2 Viking landers, is composed of 95.3% CO_2, 2.7% nitrogen, 1.6% argon, and trace quantities of water, oxygen, and carbon-monoxide[9]. Carbon and oxygen are thus the two most plentiful elements in the Martian atmosphere and can be acquired "free as air" anywhere on the planet. The atmospheric pressure measured at the 2 Viking sites varied over a Martian year between 7 and 10 mbar, with a year round average of about 8 mbar (6 torr) observed at the higher altitude Viking 1 landing site on Chryse Planitia. Pumps which can acquire gas at this pressure and compress it to a workable pressure of 1 bar or more were first demonstrated by the English physicist Francis Hawksbee in 1709. Even better pumps are available today[10].

In order to insure quality control in the propellant production process, it is desired that no substances of unknown composition, to wit, Martian dust, be allowed to enter the chemical reactors. This can be accomplished by first placing a dust filter on the pump intake to remove the vast majority of the dust, and then compressing the CO_2 to about 7 bar pressure. When CO_2 gas is brought to this pressure and then allowed to equilibriate to ambient Martian temperature conditions, it will condense into the liquid state. Any dust which managed to evade the pump filters will then go into solution, while nitrogen and argon will remain gaseous and thus can be removed. If CO_2 is then vaporized off the holding tank it will be distilled 100%

pure, as all dust will be left behind in solution. Distillation purification processes working on this principle have been widely used on Earth since the mid-1700s, when Benjamin Franklin introduced a desalination device for use by the British Navy.

Once pure CO_2 is obtained, the entire process becomes completely controllable and predictable, as no unknown variables can be introduced by Mars. Thus with the design of adequate quality control on the CO_2 acquisition process, the entire rest of the chemical production process can be duplicated on Earth under precisely the same conditions that will be present on Mars, and reliability guaranteed by an intensive program of ground testing. Very few of the other key elements of a manned Mars mission (engines, aerobrakes, parachutes, life support systems, on-orbit assembly techniques etc.) can in fact be made subject to an equivalent degree of advance testing. This means that, far from being one of the weak links in the chain of a Mars mission, the in-situ propellant process can be made one of the strongest.

Once the CO_2 is acquired it can be rapidly reacted with the hydrogen brought from Earth in the methanation reaction, which is also called the Sabatier reaction after the chemist of that name who studied it extensively during the latter part of the 19th century.

The Sabatier reaction is:

$$CO_2 + 4H_2 = CH_4 + 2H_20 \qquad (1)$$

This reaction is exothermic and will occur spontaneously in the presence of a nickel catalyst (among others). The equilibrium constant is extremely strong in driving the reaction to the right, and production yields of greater than 99% utilization with just one pass through a reactor are routinely achieved. In addition to having been in widescale industrial use for about 100 years, the Sabatier reaction has been researched by NASA, the USAF, and their contractors for possible use in Space Station and Manned Orbiting Laboratory life support systems. The Hamilton Standard company, for example, has developed a Sabatier unit for use on Space Station Freedom, and has subjected it to about 4200 hours of qualification testing. It is interesting to note that the Hamilton Standard SSF Sabatier units, which use a proprietary Ruthenium catalyst with a demonstrated shelf life of greater than 12 years, are sized to react about 3 kg of CO_2 per day, which is the full capacity required to perform the Mini-Mars Direct sample return mission.

A Hamilton Standard Sabatier unit designed for Space Station Freedom is shown in Fig.6.

Fig. 6. A Hamilton Standard Sabatier unit designed for use on Space Station Freedom. The reactor is about 0.5 m long, and has sufficient capacity to support the Mini-Mars Direct MRSR mission.

The fact that the Sabatier reaction is exothermic means that no energy is required to drive it, and this in turn implies that the limiting rate at which it can be made to proceed on Mars is the rate at which the CO_2 feedstock can be acquired. CO_2 can be compressed and liquefied out of the Martian atmosphere at an energy cost of about 0.08 kWe-hr/kg. With 100 kWe available to drive the pumps, the manned Mars Direct mission can acquire all of the 33 tonnes of CO_2 needed to completely react its initial supply of 6 tonnes of liquid hydrogen into methane and water in just 26 hours. Similarly, with 0.8 kWe, the Mini-Mars Direct mission can acquire the 550 kg it needs to react away its hydrogen supply in about 55 hours. This would not actually be done in either mission, as it would lead to a needless oversizing of the Sabatier reactors and the pumps. The point, however, is that the hydrogen can be reacted away at a rate much higher than it will boiloff, and thus there is no problem with the long term storage of the cryogenic liquid hydrogen on the Martian surface.

As the reaction (1) is run, the methane so produced is liquefied either by thermal contact with the hydrogen stream or (later on after the liquid hydrogen is exhausted) the use of a mechanical refrigerator. (Methane is slightly less cryogenic than liquid oxygen.) The water produced is condensed and then transferred to a holding tank, after which it is pumped into an electrolysis cell and subjected to the familiar electrolysis reaction:

$$2H_2O = 2H_2 + O_2 \qquad (2)$$

The oxygen so produced is refrigerated and stored, while the hydrogen can be recycled back to the Sabatier reaction (1).

Electrolysis is familiar to many people from high school chemistry, where it is a favorite demonstration experiment. However, this universal experience with the electrolysis reaction has created a somewhat misleading mental image of an electrolysis cell as something composed of Pyrex beakers and glassware strung out across a desk top. In reality modern electrolysis units are extremely compact and robust objects, composed of sandwiched layers of electrolyte impregnated plastic separated by metal meshes, with the assembly compressed at each end by substantial metal end caps bolted down to metal rods running the length of the stack. Such solid polymer electrolyte (SPE) electrolyzers have been brought to an extremely advanced state of development for use in nuclear submarines, with over 7 million cell-hours of experience to date. Testing has included subjecting cells to depth charging and loads of up to 200 g's. Both the Hamilton Standard and the Life Sciences companies have also developed light weight electrolysis units for use on the Space Station. Once again, these units are of adequate capacity to perform the propellant production operation for the Mini-Mars Direct mission. The SPE units that Hamilton Standard has supplied for use by Britain's Royal Navy have the correct output level to support the propellant production requirements of the manned Mars Direct mission. These units have operated for periods of up to 28,000 hours without maintenance, about 4 times the utilization required on the manned Mars Direct mission. The submarine SPE electrolysis units are very heavy, as they are designed to be so for ballasting purposes. SPE electrolysis units designed for space missions would be much lighter (see below).

If all the hydrogen is expended cycling the propellant production process through reactions (1) and (2), then each kilogram of hydrogen brought to Mars will have been transformed into 12 kg of methane/oxygen bipropellant on the martian surface, with an oxygen to methane mixture ratio of 2:1. Burning the bipropellant at such a ratio would provide a specific impulse of about 340 s, assuming an nozzle expansion ratio of 100. This amount of propellant mass leveraging would be satisfactory for the Mini-Mars Direct mission, providing the sample returned was kept to 10 kg, and the rover was reduced from 400 to 350 kg, with an extra 50 kg of hydrogen taken compared to the estimate given in Table 1. However the optimum oxygen to methane combustion mixture ratio is about 3.5:1, as this provides for a specific impulse of 373 s and the hydrogen to bipropellant mass leveraging of 18:1. It is this level of performance is the basis of the optimal design of the manned Mars Direct mission, as well as the mass estimates for the MMD mission given in Table 1.

If this optimal level of performance is to be obtained, an additional source of oxygen must be obtained beyond that made available by the combination of reactions (1) and (2). One possible answer is the direct reduction of CO_2.

$$2CO_2 = 2CO + O_2 \qquad (3)$$

This reaction can be accomplished by heating CO_2 to about 1100 C, which will cause the gas to partially dissociate, after which the free oxygen so produced can be electrochemically pumped across a zirconia ceramic membrane by applying a voltage. The use of this reaction to produce oxygen on Mars was first proposed by Dr. Robert Ash at JPL in the 1970s, and since then has been the subject of ongoing research by both Ash (now at Old Dominion University), Kumar Ramohalli and K. R. Sridhar (at the Univ. of Arizona), and Jerry Suitor (at JPL). The advantage of this process is that it is completely decoupled from any other chemical process, and an infinite amount of oxygen can be so produced without any additional feedstock. The disadvantages are that the zirconia tubes are brittle, and have small rates of output so that very large numbers would be required for the manned Mars Direct application. (The numbers would not be excessive for the MMD mission.) Improved yields have recently been reported at the Univ. of Arizona, so the process may be regarded as promising, but still experimental.

An alternative that would keep the set of processes employed firmly within the world of 19th century industrial chemistry, would be to run the well known water-gas shift reaction in reverse. That is recycle some of the hydrogen produced in the electrolysis unit into a third chamber where it is reacted with CO_2 in the presence of an iron-chrome catalyst as follows:

$$CO_2 + H_2 = CO + H_2O \qquad (4)$$

This reaction is mildly endothermic but will occur at 400 K, which is well within the temperature range of the Sabatier reaction. It has been shown by Meyer[11] that if reaction (4) is cycled with reactions (1) and (2), the desired mixture ratio of methane and oxygen can be produced with all the energy required to drive reaction (4) provided by thermal heat output from the Sabatier reactor. Reaction (4) can be carried out in a simple steel pipe, making the construction of such a reactor quite robust. The disadvantage of reaction (4) is that in the temperature range of interest it has an equilibrium constant of only about 0.1, which means that in order to drive it to the right it is necessary to both overload the left hand side of the equation with extra CO_2 while condensing out water to remove it from the right hand side. This is certainly feasible, and actually constitutes a fairly modest chemical engineering design problem. However a number of alternatives that are at least equally promising have been advanced. One of the most elegant of these would be to simply combine reactions (1) and (4) in a single reactor as follows:

$$3CO_2 + 6H_2 = CH_4 + 2CO + 4H_2O \qquad (5)$$

This reaction is mildly exothermic, and if cycled together with reaction (2) would produce oxygen and methane in a mixture ratio of 4:1, which would give the optimum propellant mass leveraging of 18:1 with a large extra quantity of oxygen also produced that could function as a massive backup to the life support system. In addition, salvageable CO would also be produced that could conceivably used in various combustion devices or fuel cells. If all the CO and O_2 produced is included, the total propellant mass leveraging obtained could thus be as high as 34:1. Researchers at Hamilton Standard intend to put to the test a number of methods of driving this reaction early in 1991[12].

Probably the easiest method of obtaining the required extra oxygen is just to take some of the methane produced in reaction (1) and pyrolyze it into carbon and hydrogen.

$$CH_4 = C + 2H_2 \qquad (6)$$

The hydrogen so produced would then be cycled back to attack more Martian CO_2 via reaction (1). After a while a graphite deposit would build up in the chamber in which reaction (6) was being carried on. (This reaction is actually the most common method used in industry to produce pyrolytic graphite.) At such a time, the methane input to the reactor would be shut off and instead the chamber would be flushed with hot CO_2 gas. The hot CO_2 would then react with the graphite to form CO, which would then be vented, cleaning out the chamber.

$$CO_2 + C = 2CO \qquad (7)$$

Such a plan, incorporating two chambers, with one carrying out pyrolysis while the other is being cleaned, has been suggested to the authors as the simplest solution to the extra oxygen problem by a group of researchers at Hamilton Standard[13].

The Hamilton Standard group also provided mass estimates for propellant production systems for both the Mars Direct and MMD missions based on a system combining reactions (1), (2), (6), and (7). The estimates are given in Table 2.

Table 2. Hamilton Standard Mass Estimates for CH_4/O_2 Plant

Reactor	Mini-Mars Direct	Manned Mars Direct
Sabatier	36 kg	164 kg
Electrolysis	90 kg	477 kg
Pyrolysis	105 kg	450 kg
Total	231 kg	1091 kg
Requirement	3.6 kg/day	360 kg/day
Capability	7.2 kg/day	540 kg/day

The mass estimates in Table 2. assume 2 complete units each with 100% mission capacity for the system used in the MMD mission, and 3 units each with 50% capacity for the system employed on the manned Mars Direct mission.

The reason for the different approach to redundancy on the two missions is that the small scale units used on the MMD mission have essentially the same mass whether they are full or half capacity, while the much larger units used on the manned Mars Direct scale in a roughly linear manner with capacity.

In summary the methods required to produce methane and oxygen bipropellants on Mars for the Mars Direct are well understood and already in an advanced state of technology development. It has suggested in some quarters that while these propellant production processes are promising, they should be relegated to inclusion in downstream missions, with the initial set conducted using only terrestrial propellants. This hardly seems appropriate, as these propellant production methods are in a more mature state of development than nearly everything else associated with the manned Mars mission, and except for the Titan IV/Centaur launch vehicle, everything in the MRSR mission as well. Moreover, the carrying out of an initial set of manned Mars missions without the leverage afforded by in-situ propellant production would be the ultimate in architecture incoherence, as the initial set of non-in-situ propellant missions would require a different set of vehicle hardware, and a massive launch and orbital infrastructure that would be very costly and later prove unnecessary. Furthermore, if much in the way of useful surface exploration is to be accomplished, the in-situ process will be needed anyway - and on the very first mission. Since we can have the propellant production process right from the start of the manned Mars exploration program, and since we must have it if useful surface exploration is to be done, we might as well take full advantage of it and use it to provide the Mars ascent and Earth return propellant as well.

RADIATION HAZARDS AND MISSION RISK

The Mars Direct mission plan uses conjunction class (close to minimum energy) orbits for interplanetary transfer. The use of such orbits require Mars mission round trip times of roughly 2.6 years. It has been sometimes remarked that such long missions create excessive mission risk due to the deleterious effects of radiation in space and zero gravity. For this reason it has been argued that opposition class (high energy) missions are necessary, as these can reduce the round trip mission time to about 1.6 years.

The opposition class mission has numerous disadvantages. In the first place, its higher energy trajectories increase the total mission delta-V to the point where the total initial mass of the mission is driven up by about a factor of 2 compared to conjunction class missions. The larger amounts of propellant utilized necessitate a correspondingly greater total number of engine-burn-minutes, with the probability of an engine failure increasing in direct proportion. The higher energy orbits also entail higher energy aerocaptures maneuvers, with increased probability of skip out of the planetary atmosphere, and deceleration g loads increased to about 8 to 10 g's compared to the 3 to 4 g's common on conjunction class mission aerocapture maneuvers. The high g loads of opposition class mission aerocapture could pose an unacceptable hazard to crews weakened by 1.5 years in a low gravity environment, navigation and control margins are narrowed by the higher entry velocity, and the violence of the maneuver could create thermal and mechanical problems for the spacecraft as well.

The opposition mission reduces total round trip mission time primarily by reducing the stay time at Mars down to a minimal 0.1 years, with interplanetary round trip transit time remaining at about 1.5 years. This tends to give the mission a very inefficient, if not slightly absurd, character, as the time available to accomplish Mars exploration is reduced to nil. The situation is somewhat analogous to a family which decides to fly to Hawaii for Christmas vacation, taking a 6 hour flight out to the island, 20 minutes to taxi up to the airport terminal, exiting for a 20 minute sortie around the airport, and then returning to the aircraft for a 20 minute wait on the runway followed by another 6 hour flight home.

Finally the opposition class mission must spend part of its flight in a swing into the inner solar system to a distance of about 0.65 astronomical units from the Sun. At this distance, the radiation dose experienced from a solar flare would be 2.4 times that felt at Earth's distance, and 5.5 times that felt by a spacecraft in orbit about Mars. This is very important, because the effect of high sudden doses of radiation are non-linear, and a single 200 rem dose experienced by an opposition mission crew as they flew within the orbit of Venus would be far more dangerous (severe radiation sickness would result) than 5 doses of 40 rems delivered over a 1.5 year period to a conjunction class mission crew hanging in orbit about Mars (no observable symptoms would be expected). In addition, the doubling of heat loads during the opposition class mission transit through the inner solar system would create a significant thermal design problem, and result in the catastrophic failure of the mission should the required cooling system fail.

The opposition mission is a dead end. It drives mission mass up to the point where the repeated missions required for a sustained human presence on Mars would be prohibitively expensive, and it is incapable of supporting any significant program of surface exploration. The question is, is the taking on of the burden of all the additional risks, cost, and inefficiency entailed by the opposition class mission really necessary? We believe that it is not. In the paragraphs below we shall support this assertion by showing that the proper design of a conjunction class mission can eliminate the zero gravity and radiation exposure rationales that have repeatedly been cited as the rationale for choosing the opposition class option.

While it is the case that the canonical minimum energy (Hohmann transfer) flight time to Mars is 0.707 years (258 days) each way, assuming circular planetary orbits, the fact of the matter is that for the real orbits of Earth and Mars trajectories can be found that are marginally more than minimum energy which reduce one way transit times to as little as 0.345 years (126 days). A set of such trajectories chosen for the 1999, 2001, and 2003 mission opportunities are shown in figures 7,8, and 9. All missions assume a trans-Mars injection C_3 of 15 km/s^2, (which is the design mission for the Ares' 47 tonne TMI throw capability), aerocapture C_3 limits at both Earth and Mars of 25 km/s^2, and a trans-Earth injection C_3 limit of 10 km/s^2 (which is the ERV's design performance). It can be seen that transit times for these missions vary between 0.345 and 0.614 years (126 and 224 days), with a canonical typical average being 0.5 years (182 days) each way. The total round trip interplanetary transit time of the conjunction mission can thus be kept to about 1.0 years, 33% <u>less</u> than the 1.5 years required[14] by the opposition class mission.

These missions could be made shorter, especially the return trip times, if slightly more energy is added to the Earth Return Vehicle. In figures 7 through 9 the delta-V was minimized for leaving Mars since the ERV is the mass driving element of the Mars Direct mission scenario. By adding another 500 m/s capability to the ascent vehicle the return trip times can be reduced by about a month.

Figs 7, 8, and 9 depict the trajectories taken by Mars Direct missions during the years 1999, 2001, and 2003, respectively. The average one-way flight time on the slightly accelerated conjunction class trajectories chosen is only 180 days. This is about 33% less than typical high energy opposition class trajectories

If the interplanetary transit is to be done in zero gravity, than the reduced transit time of the conjunction class mission reduces zero g exposure to a corresponding degree. On the other hand, if artificial gravity is desired, then the reduced delta-V requirement of the conjunction class mission makes the mass penalty associated with artificial gravity systems that much easier to tolerate. If rigid artificial gravity systems are desired, then aerocapture of the extended spacecraft becomes awkward, if not impossible. The abandonment of aerocapture technology is tolerable for the conjunction class mission[15], but drives the mass of the higher energy opposition class mission off the scale. If tethered artificial gravity systems are anticipated, then the use of conjunction class trajectories together with the leverage offered by the use of in-situ propellant allows the reduction of trans-Mars payload mass to the point where useful levels of artificial gravity can be generated by tethering off the burnt-out TMI stage. Thus the counterweight at the end of the tether is not mission critical, and if tether dynamics were to go awry, the tether could be dropped without loss of the mission. Such is the artificial gravity adopted by Mars Direct. This could not be the case with an opposition mission tether system, which due to increased TMI payload mass would require mission critical payload elements to be placed at both ends of the rotating tether.

The adoption of the "type 1" fast-transit conjunction class orbits chosen for Mars Direct does not reduce total round trip mission time significantly. Instead, stay time at Mars is increased compared to a nominal minimum energy conjunction class mission. A comparison of the mission plans is given in Table 3.

Table 3. Flight Times and Stay Times of Mars Missions

	Min. Energy	Mars Direct	Opposition
Total Transit	1.4 years	1.0 years	1.5 years
Mars Staytime	1.2 years	1.5 years	0.1 years

The question at hand is, how do these mission profiles effect the radiation dose the crew may be expected to receive in the course of a manned Mars mission, and what is the magnitude of the health hazard that such a dose represents?

In Table 4. we present some basic data drawn from references 16, 17, and 18 that deal with the radiation hazards on a Manned Mars mission.

Table 4. Radiation Dose Experienced on Space Missions

	Unsheltered	Sheltered Dose
GCR in Space (Solar Min)	50 rem/year	33 rem/year
GCR in Space (Solar Max)	20 rem/year	15 rem/year
GCR on Mars (Solar Min)	13 rem/year	8 rem/year
GCR on Mars (Solar Max)	6 rem/year	4 rem/year
Solar Flare, 1 AU, 2/56	31 rem	16 rem
Solar Flare, 1 AU, 11/60	37 rem	7 rem
Solar Flare, 1 AU, 8/72	46 rem	1 rem
Solar Flare, Mars, 2/56	11 rem	6 rem
Solar Flare, Mars, 11/60	10 rem	2 rem
Solar Flare, Mars, 8/72	9 rem	0.2 rem

In Table 4. the "Mars" doses are the doses on the Martian surface. "Sheltered" doses are based upon the assumption of 35 gm/cm^2 of shielding. On a spacecraft this could only reasonably be provided within a small "storm shelter" for use during solar flares, while on Mars it could be provided to the entire habitat through the use of regolith. "Unsheltered" doses are based on the assumption of 5 gm/cm^2 of spacecraft structure. All doses given in Table 4 are taken from the references cited, except for the "Mars, sheltered" doses, which are extrapolated from the reference data.

It can be seen that the Mars surface doses are much less than those experienced in 1 AU interplanetary space. This is because the Martian atmosphere, while only 20 gm/cm^2 thick in the vertical direction, actual provides the equivalent of 65 gm/cm^2 of hemisphere averaged shielding when rays impacting the astronauts after an oblique transversal through the atmosphere are averaged in[16]. In addition, the surface of Mars itself blocks out 50% of all cosmic rays. Finally, with a distance from the Sun averaging 1.52 AU, the average dose from a solar flare experienced in near Mars space would only 43% of that at 1 AU.

Now let us try to estimate some "worst case" doses to the expedition crew. We assume that a solar flare equal to the average of the three worst recorded cases (2/56, 11/60, and 8/72) occurs at a rate of 1 per year during solar max, and at a rate of 0.2 per year during solar min. We estimate the average distance from the Sun during a conjunction class orbit at 1.3 AU, and during an opposition class orbit as 1.2 AU. We assume that 25% of the time of each solar flare is spent unsheltered, and 75% within shelter. We assume that the crew of the spacecraft sleep in the storm shelter and thus spend 30% of their GCR exposure time within shelter, and 70% out of shelter. Since on the surface of Mars, the entire habitat could be sheltered we assign 70% of the Mars surface GCR exposure to the sheltered category, and 30% to the unsheltered. The calculated doses for the three mission plans (opposition, Minimum energy, and Mars Direct) are given in Table 5.

Table 5. Radiation Dose Experienced on Mars Missions

	Min. Energy	Mars Direct	Opposition
Transit Doses			
GCR (Solar Min)	63.0 rem	45.0 rem	67.5 rem
GCR (Solar Max)	25.9	18.5	27.8
Sol. Flare (S. Min)	2.5	1.8	3.2
Sol. Flare (S. Max)	12.7	9.1	16.0
Mars Doses			
GCR (Solar Min)	11.4 rem	14.3 rem	1.0 rem
GCR (Solar Max)	5.5	6.9	0.5
Sol. Flare (S. Min)	1.1	1.3	0.1
Sol. Flare (S. Max)	5.4	6.8	0.5
Total Dose			
Solar Min	78.0 rem	62.4 rem	71.8 rem
Solar Max	49.5	41.3	44.8

To place these doses in perspective, it may be noted that every 60 rem of radiation (received over an extended period, such as the doses given above) adds 1% of extra risk of a fatal cancer at some point later in life to a 35 year old woman, while 80 rem adds 1% of extra risk of fatal cancer to a 35 year old man.

It can be seen in Table 5. that the Mars Direct mission, with its slightly accelerated conjunction class orbits combined with a long surface stay, actually offers an average mission radiation dose somewhat less than the opposition class mission. As stated above, however, the freak chance of a single large catastrophic dose is much higher on the opposition class mission due to its close-in pass to the Sun. There thus appears to be no radiation-dosage rationale for choosing the opposition mission plan over Mars Direct.

In summary, we find that neither zero-gravity nor radiation exposure concerns offer any countervailing advantages to offset the inefficiency, risk, complexity, and high cost of the opposition class mission. We therefore recommend that the opposition mission plan be dropped from the NASA baseline.

ACTIVITIES ON THE SURFACE OF MARS

Science on the surface

The two primary scientific themes for human exploration of Mars are: Planetary evolution, climate change, and life; Human habitability of Mars. The first theme encompasses the search for life (past and present), climate history of Mars, and geological history. The second theme deals with resources on Mars to support humans, medical issues and human factors.

The steps to achieving science objectives can be broken down into three distinct phases. These phases are: 1) Precursor, to obtain environmental knowledge necessary for human exploration; This phase has been underway since the Mariner mission of 1965. 2) Emplacement, human landings to explore Mars on a regional basis; 3) Consolidation, to build a permanent base and conduct global exploration.

Possibly the most intriguing science objective is the search for fossil-life on Mars. This type of work will be closely linked to the study of the geologic history of Mars. Geologic history will identify the paleolake site where most of the fossils are likely to be found. A good example of a possible site for fossil-life studies is at latitude 22°, longitude 12° in the Margaritifer Sinus region.

One of the keys to human exploration of Mars will be the atmospheric science and climate history. Studies of these areas will provide us with clues to the questions of similarities between early Earth and Mars. In addition we will learn what aspects of the current climate of Mars can be directly useful to human exploration.

The primary science that will be conducted on Mars will be geoscience and geologic history. The goals of this work will be:

- to develop and test a geological model of the current surface and interior of Mars.
- to use the geological model to identify resources which are necessary or useful to support a human presence on Mars.
- to identify and understand the internal and surface processes which have been active throughout the planet's history.
- to establish a time frame for the occurrence of events in the planet's history.

Base/landing location

It is unlikely that early human missions will have truly global access. Therefore, selecting an appropriate site for the initial base on Mars is critical. In selecting a site for the early base there are two primary drivers: first, getting as near as possible to interesting scientific sites but staying within safe landing zones, second, to be in a location with as low an elevation as possible for radiation protection and aerocapture reasons. There is one other driver that will alter all design parameters and that is the discovery of a large reservoir of water on Mars.

Assuming that no large source of water is found on Mars before humans start landing, what is a good location to put the base?

A good answer to this question is the southern plains of Lunae Planum just north of Ophir Chasma (on the equator at 65° lat.). This places the base at an average elevation (between 6 and 7 km) and within a couple of thousand km's of most of the scientifically interesting sites on Mars.

Mars consists of seven primary geological features: northern plains, southern cratered terrain, volcanos, lava flows, canyons, channel terrain and craters. A 0° longitude, 65° latitude landing site would be in the largest area of channel terrain. In addition it is within 300 km of canyon terrain. With a range of 1000 km the northern plains to the west and north and the southern cratered terrain to the east are within reach. 2000 km to the west is the edge of the lava flow areas. Another 600 km beyond that is are the volcanos of the Tharsis Montes. Therefore, within an area of 2500 km in radius, one can reach six of the seven major geological features of the planet Mars.

Table 6. shows the key features near the 0° long., 65° lat. landing site.Mars Direct provides for surface sorties up to 500 km from a single base site. Provided ground vehicles of such range are available, then a string of 4 landing sites placed 800 km or less apart would allow one to access all 14 of the interesting sites on the list in Table 1. On the other hand, if the range of the ground vehicles were limited to 100 km or less, then about 12 landings would be required. Furthermore, even with 12 landings, the mission scenario equipped with 100 km rovers would only be able to access about 1/8th the territory available to 4 missions equipped with 500 km range rovers. Thus we see the importance of achieving at least such medium (500 km) range surface mobility if our Mars exploration program is going to be reasonably cost-effective.

Surface Vehicles

The Mars Direct method of exploration provides a number of features specifically for surface operations. To begin with there is a powerful energy source in the nuclear SP-100. This power source can produce large amounts of fuel for both rockets and surface vehicles. Next there is the deliverable payload. A single Mars Direct launch can place 30 metric tons on the surface of Mars. Finally, all of the crew members of the mission are on the surface. This makes the most hands available for surface operations. Since the mission mass scales primarily in proportion to total crew size, elimination of redundant crew left in orbit reduces total mission ETO mass, cost, and number of crew at risk, while preserving exploratory capability.

The primary advantage of using the Mars Direct method of exploration for surface operations is that it allows for a chemical combustion powered surface vehicle. The environment on Mars is very harsh and the need to cover long distances will be great.

The fact that the vehicle is chemically powered as opposed to electrical power is very beneficial as seen in Table 7..

Table 6 .Surface Features of Interest in the Exploration of Mars

Feature	Distance (km)	Direction
Ophir Chasma	<300	southwest
Juventae Chasma	<300	southeast
Slope and bedrock material	<300	south
Cratered plateau material	<300	east
Chaotic material	<300	east
Degraded crater material	<300	south
Hebes Chasma	600	west
Center of Lunae Planum	650	north
Northern plains	1200	northwest
Kasei Vallis	1300	north
Viking 1 landing site	1400	northeast
Paleolake site	1500	northeast
Volcanic flows	2000	west
Pavonis Mons	2500	west

Table 7 Power to Mass Ratios for Candidate Mars Rover Propulsion

Power Source	Power/Mass (W/kg)
RTG	5
DIPS	8
Photovoltaic (during daylight)	16
H2/O2 Fuel Cell	55
Internal Combustion Engine	1000

We see that a combustion engine can have a power/mass ratio about a factor of 20 higher than that of an H2/O2 fuel cell. Now for a given life support system mass, the vehicle's range will be directly proportional to its speed, which is in turn proportional to the power. Furthermore, if one of the other options try to match the combustion engine's power level, its weight will rapidly become excessive. For example, if the rover is equipped with a 50 kW (about 65 hp) of power, the mass of the required internal combustion engine would only be about 50 kg, while that of a set of fuel cells would be 900 kg. The combustion powered car could thus take along 850 kg of additional science equipment and consumables compared to a fuel cell powered vehicle of equal power, and again have much greater endurance, capability, and range. Furthermore, the fact that the combustion powered vehicle is virtually power unlimited allows sortie crews to undertake energy-intensive science at a distance from the base that would otherwise be impossible. For example, a combustion vehicle sortie crew could drive to a remote site and generate 100 kW to run a drilling rig. Rover data transmission rates can also be much higher, which in turn increases both crew safety and sortie science return. Combustion engines can also be used to provide high power for either main base or remote site construction activity (bulldozers, etc.) Thus we see that the greater power density of combustion powered engines will provide for greater mobility with a much smaller, lighter, and far more capable vehicles, and a more potent and cost-effective Mars exploration program all-around.

The use of combustion powered vehicles is fuel intensive, however. For example, it is estimated that a 1 tonne pressurized ground rover would require about 0.5 kg of methane/oxygen bipropellant to travel 1 km. Thus a 800 km round trip excursion would consume about 400 kg of propellant. Traveling at an average rate of 100 km a day, this would only represent a 8 day sortie. In the course of a 600 day surface stay, many such excursions would be desired to make effective use of the available time. Importing from Earth the large amounts of propellant required to support an adequate level of activity would pose a very heavy burden upon the space transportation system. In the Mars Direct plan, however, only 610 kg of hydrogen needs to be imported to produce 11 tonnes of bipropellant for surface use, enabling 27 excursions such as that described above, with a total available surface mileage of 22,000 km. Thus we see that the use of combustion powered vehicles is closely tied to the in-situ manufacture of propellant.

There might be very different designs for each of these types of surface vehicles but there are some fundamental questions that should be answered before detailed designs are pursued. Primarily, what is the best engine to use on Mars? The environmental factors on Mars make it impossible for a completely conventional Earth internal combustion engine to work on Mars. Therefore what is the best alternative? In order to address this question properly we must look at what differences there are between Earth and Mars, and between various engines, before selecting the best engine to power the surface vehicle.

The key differences between Earth and Mars with respect to the design of a vehicle engine are noted in the following list:

- Mars has no oxidizer in the atmosphere
- Mars is much colder on average (Mars avg. 215°K, Earth avg. 288°K)
- Mars has a much thinner atmosphere (1/100 of Earth)
- There will be a very limited supply of fuel on Mars
- There will be a limited ability to maintain and fix equipment on Mars
- Liquid water does not exist on Mars
- There is a large daily temperature variation on Mars (avg. 60°K)
- Great limitation on size of equipment
- Great limitation on equipment mass

The lack of an oxidizer in the atmosphere can be easily dealt with by having the vehicle carry an oxygen tank. The real problem in designing an internal combustion engine for Martian use is not the lack of oxygen in Mars' atmosphere, but the lack of nitrogen, which will cause a pure methane/oxygen engine to run very hot. The most critical problem then, is how to cool the engine.

Cooling an engine on Earth is done primary though three modes. First, the air that is sucked in as a working fluid also takes a great deal of heat away as exhaust (typically 40%), with the nitrogen in the air playing a key role in reducing the combustion temperature. Second, where an active cooling system such as water or oil cooling exists the heat of the engine is taken away to be cooled by a heat exchanger. Finally, passive cooling of the exposed surfaces of the engine is typically a very small fraction of the total heat removed from the system (<5%).

On Mars all of these three methods could be used. The combustion temperature can be reduced by utilizing Martian CO_2 as an inert gas to take the place of nitrogen as a heat sink[19]. This gas could be pumped into the engine as it is driven, or accumulated in liquid form and placed in a holding tank during the night when the vehicle is not being driven. Alternatively, the vehicle could be supplied with a large reservoir of liquid CO_2 at the base for use during a given sortie. This would not be desirable, however, as a large supply of CO_2 would add excessively to the propellant weight and thus limit the vehicle's range and speed. A more imaginative alternative would be to recycle the exhaust through a high temperature radiator, allowing only as much CO_2 to be vented to the atmosphere as is being created by combustion. Some of the water in the exhaust could be condensed out, either in the radiator or with the aid of supplemental regenerative cooling available from the liquid methane or oxygen. This water could then be stored in a holding tank and be brought back to the base for synthesis into more methane/oxygen bipropellant at the end of the sortie. The remaining exhaust, its temperature drastically dropped, would then be recycled back into the combustion chamber to sandbag the combustion temperature down to tolerable levels. By combining such an approach with the use of higher temperature materials (ceramics, titanium, etc.) than are commonly used in automobile engines, and possibly using the soft cryogenic propellants to directly regeneratively cool the hottest parts of the engine itself, it should be possible to run such an engine with only a small reservoir of liquid CO_2 required for start-up purposes.

NTR Augmentation of Mars Direct

While the Mars Direct architecture can be initiated solely with the use of chemical propulsion, it is particularly amenable to the introduction of nuclear thermal rockets (NTR) as soon as that technology becomes available. The reason for this is that the Ares launch vehicle can deliver NTR stages directly to a nuclear safe (700 km) orbit, and the architecture involves no LEO infrastructure positioned below this orbit. In addition, the modest size, mass, and per-unit cost of NTR engines makes NTR the advanced propulsion technology of choice for use in the expendable Mars Direct mode. Such expendable utilization would be unthinkable for multi-MW nuclear electric propulsion (NEP) units, for example, the mass of which are typically an order of magnitude larger than NTRs.

In keeping with the overall approach of the Mars Direct architecture, then, namely the elimination of the need for orbital infrastructure, NTR technology is incorporated into the plan simply as a third stage to the Ares launch vehicle. Such a configuration is shown in Fig. 10. The NTR stage has a specific impulse of 900 s, a power of 900 MWth, and a thrust of 45,000 lb. With the addition of this stage the Ares can throw 70 metric tons on to a minimum energy trans-Mars injection trajectory (C3=15), an increase of about 50% over the 47 metric ton TMI all-chemical Ares baseline.

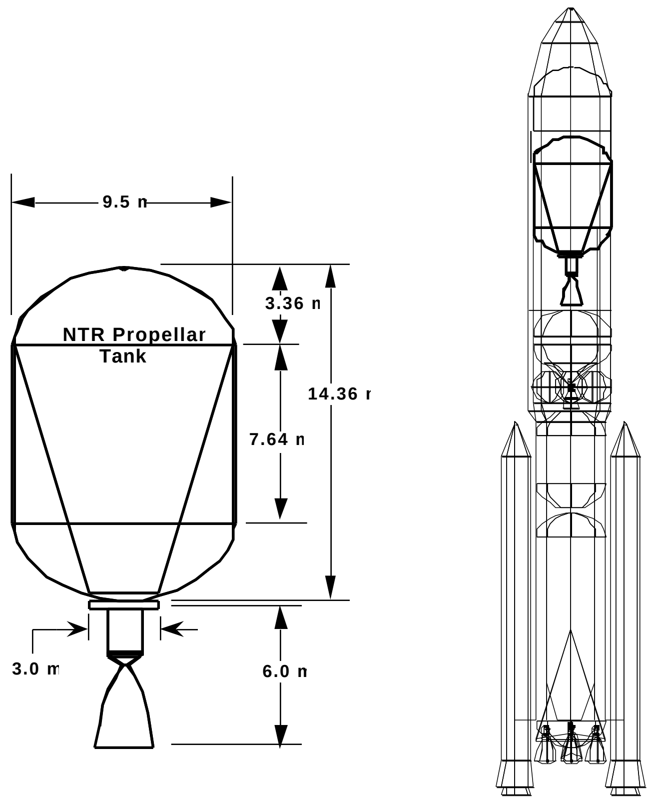

Fig. 10. Ares with NTR third stage

The performance of the NTR augmented Ares is limited by the fact that the NTR must be fired from LEO, and the baseline 2 stage Ares vehicle is not optimized for LEO delivery, as with 250,000 lb thrust its second stage is under powered. (The 250 klb thrust is optimized for direct trans-Mars injection.) If the thrust of the second stage is increased, the LEO delivery capability can be increased, and the TMI throw also increased accordingly. Such results are shown in Table 8

Table 8. NTR Augmented Ares TMI throw Capability

Stage 2 Thrust	LEO Payload	TMI Payload
250 klb	121 tonnes	70 tonnes
500	130	76
1000	144	83

In the above calculation, NTR T/W of 5 and tank fractions of 0.1 are assumed. Thrust to weight ratios in this range are expected for a 1990s version of updated NERVA class NTR technology.

Thus, if we augment the Ares upper stage to maximize its capability as a LEO delivery system, about 83 tonnes can be thrown to Mars with each launch (excluding the mass of the NTR stage, which is expended). This is about 77% higher than the Ares baseline, a TMI mass increase of 35 tonnes, which translates into an extra 20 tonnes of cargo delivered to the martian surface with each launch.

If we stick with our early plan of two launches per mission, this will allow us to increase our crew complement of each flight to 12. Alternatively, if the size of the missions are kept the same, using NTR will allow us to launch each mission with a single booster, instead of split between two.

As a third alternative, crew sizes can be kept at 4 for each two launch mission, but large amounts of cargo landed with them, allowing for the rapid buildup of a sizable base. Finally, if fast interplanetary transits are deemed important, the NTR stage can be used to throw a 4 man Mars Direct hab onto a fast (80 to 120 day transit, C3=50) trajectory to Mars, while using its larger minimum energy TMI throw to deliver an ERV augmented with a third stage to the martian surface. With the help of this third stage, the methane/oxygen driven ERV is able to execute an equally fast Earth return trip.

Three different alternatives for NTR stages for use in lunar missions are shown in fig. 11. All three are designed to be flown off of a single launch of a basic 121 tonne to LEO Ares launch vehicle (or any other 121 tonne to LEO HLLV.)

Alternative A is a reusable NTR transfer stage that delivers a 66 tonne payload from LEO to Low Lunar Orbit (LLO), and then releases the large (47.5 tonne capacity) propellant tank to return to LEO using the propellant contained in the small (2.5 tonne capacity) run tank. The payload then is separated from the large tank and landed on the Moon with a set of cryogenic engines, while the large tank remains in LLO where it is expended. After returning to LEO, the NTR waits until another 66 tonne payload mated to a full large tank is lifted to LEO where they mate and dock to prepare the next flight.

Alternative B is an expendable NTR stage that delivers a 68.5 tonne payload from LEO to LLO. The payload is then landed on the Moon with a set of cryogenic engines, while the NTR stage is expended in LLO.

Alternative C is an NTR stage that propels a 53 tonne payload from LEO to LLO, and then executes a 1.7 km/s burn to bring the payload to an approximate halt about 2 km above the Lunar surface. The NTR engine is then turned off, and the NTR stage and the payload separate, and are each landed on the Moon at separate locations using small sets of storable bipropellant engines. Since the ideal delta-V to land on the Moon from a dead halt at an altitude of 2 km is only 80 m/s (300 m/s was used in the calculation to allow for hover and margin), this alternative effectively uses the NTR's high specific impulse to effect the nearly 6 km/s delta-V between LEO and the lunar surface.

The results of all three alternatives are shown in fig. 12 and compared to the payload delivery capability of the all chemical Mars Direct lunar architecture. It can be seen that by using NERVA derivative NTR, payload per ARES flight can be increased from 68% to 104% over the chemical baseline. It should also be noted that in addition to delivering 48 tonnes of payload to the lunar surface, alternative C, the true "Lunar Direct" option, also soft lands the NTR and its tank. If the NTR is a dual mode reactor, or at least of a design that is convertible to one, it can be used to produce power on the surface, while the tank can be used to provide a large pressurizable volume for the base. If the mass of these two components are added in, then alternative C may be considered to deliver a total of 56.5 tonnes of payload to the Moon, an increase of 140% over the cryogenic chemical propulsion baseline.

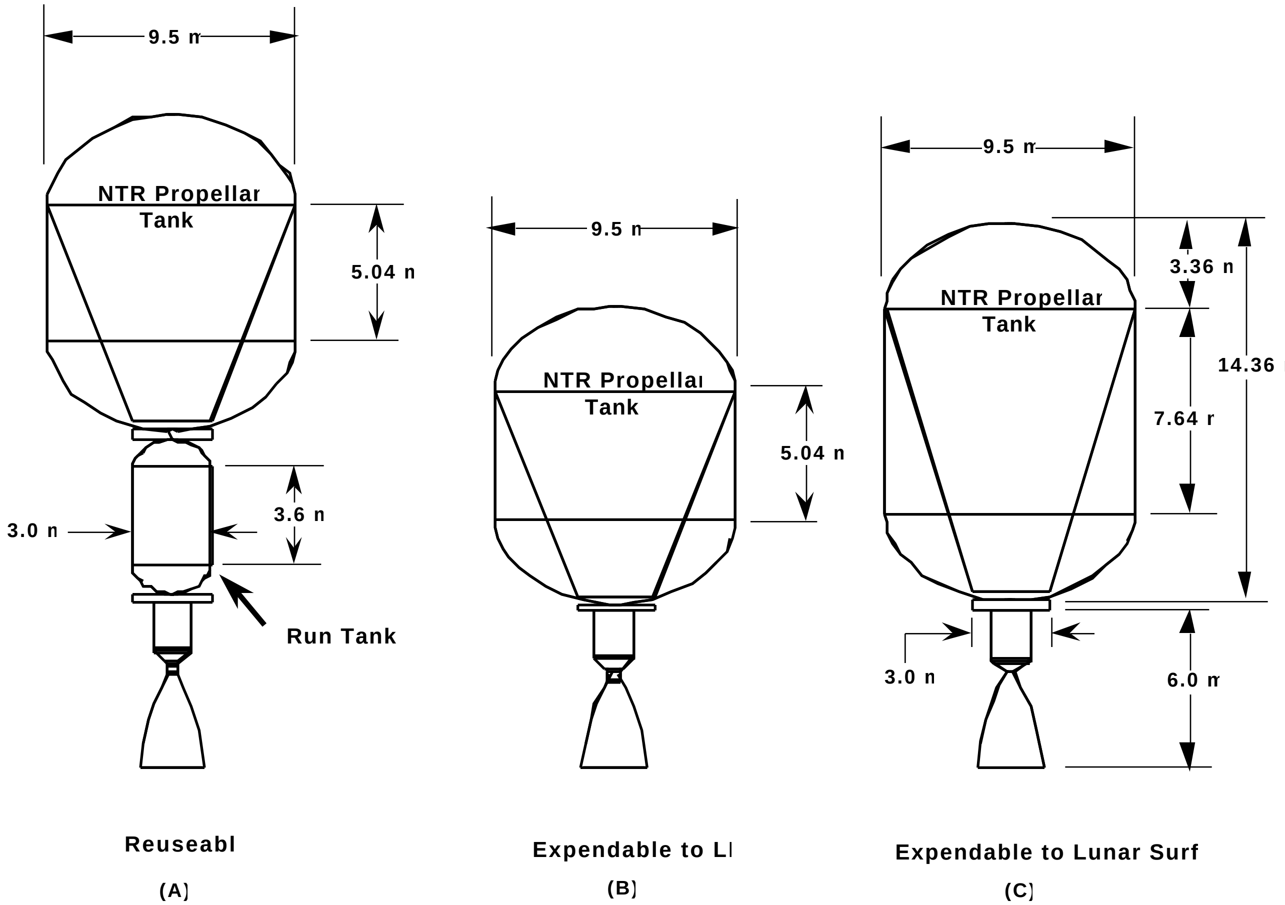

Fig. 11. Lunar NTR stage options.

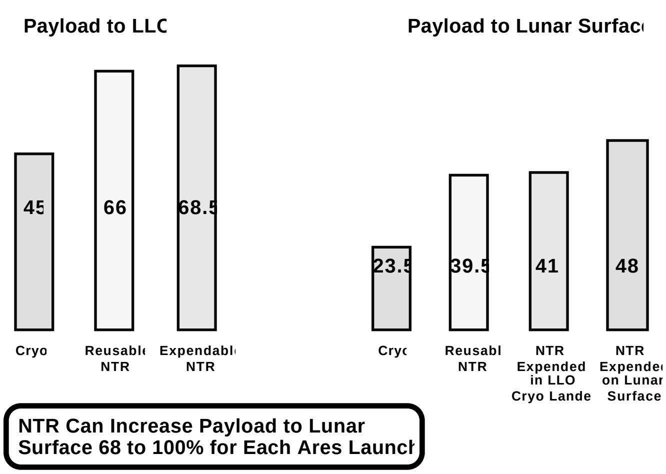

Fig. 12 Benefits of NTR stages for Lunar missions. The benefits of using NTR in a Lunar Direct architecture are substantially higher than using NTR in a LOR plan.

At this point the utility of lunar LOX needs to be factored into the scenario. Assume that a 6 tonne (dry) lunar hopping vehicle using hydrogen/oxygen propellant is employed for surface to surface long distance sorties from the base. The maximum round trip delta-V between any two points on the surface of the Moon is about 6.4 km/s (for two near-orbital ascents and descents), so that with an Isp of 465 s, the fully fueled mass ratio of such a cryogenic hopper would be about 4. It would thus require 18 tonnes of propellant, of which 2.6 would be hydrogen transported from Earth. (If the hopper used methane/oxygen propulsion, the mass ratio would be 5.75, and about 6 tonnes of methane would have to be transported to the Moon to support each maximum distance sortie.) The methane/oxygen ERV has a dry mass of 12 tonnes, and so requires 3.4 tonnes of terrestrial methane plus 12.1 tonnes of Lunar LOX to return a mission crew to Earth. Using these numbers we can calculate the number of sites on the surface of the Moon that can be visited per Ares launch. The results are shown in Table 9.

Table 9 Lunar Sites Visitable per Ares Launch

Method	Payload to Surface	Sites Visitable
Cryo/No LLOX	23.5 tonnes	1*
Cryo/LLOX	23.5	3
NTR/No LLOX	48	2
NTR/LLOX	48	12

It can be seen that without LLOX, the use of NTR doubles the number of individual sites that can be visited per Ares launch. However with LLOX available, the leverage of the NTR delivery system *increases* to a be factor of 4 greater than the cryogenic chemical/LLOX option. Thus it can be seen that in the Mars Direct lunar architecture, far from being competing technologies, LLOX and NTR are highly synergistic, and the benefits of both technologies taken together are greater than the product of the benefits of each technology taken by itself.

*(The cryo/No LLOX mission cannot quite deliver a complete fully fueled ERV. Either the ERV must be scaled down to 80% its Mars mission size, or an oxygen delivery mission must be pre-landed at the chosen site. Since about 21 tonnes of LOX could be pre-landed in one such mission, and the Cryo/No LLOX mission is only short by 5 tonnes of landed payload, a single such oxygen delivery mission could support 4 manned landings at the site, after which presumably lunar LOX production would be operational.)

NIMF Vehicles for Global Mobility

NTRs can also be designed to use martian CO_2 as their propellant. Since this can be acquired at low energy cost through direct compression out of the atmosphere, rocket vehicles so equipped will give Mars explorers complete global mobility, allowing them to hop around the planet in a craft that can refuel itself each time it lands. Such a vehicle concept, known as a NIMF[20,21] (for Nuclear rocket using Indigenous Martian Fuel) is illustrated in Fig. 13.

The high molecular weight of CO2, while very detrimental

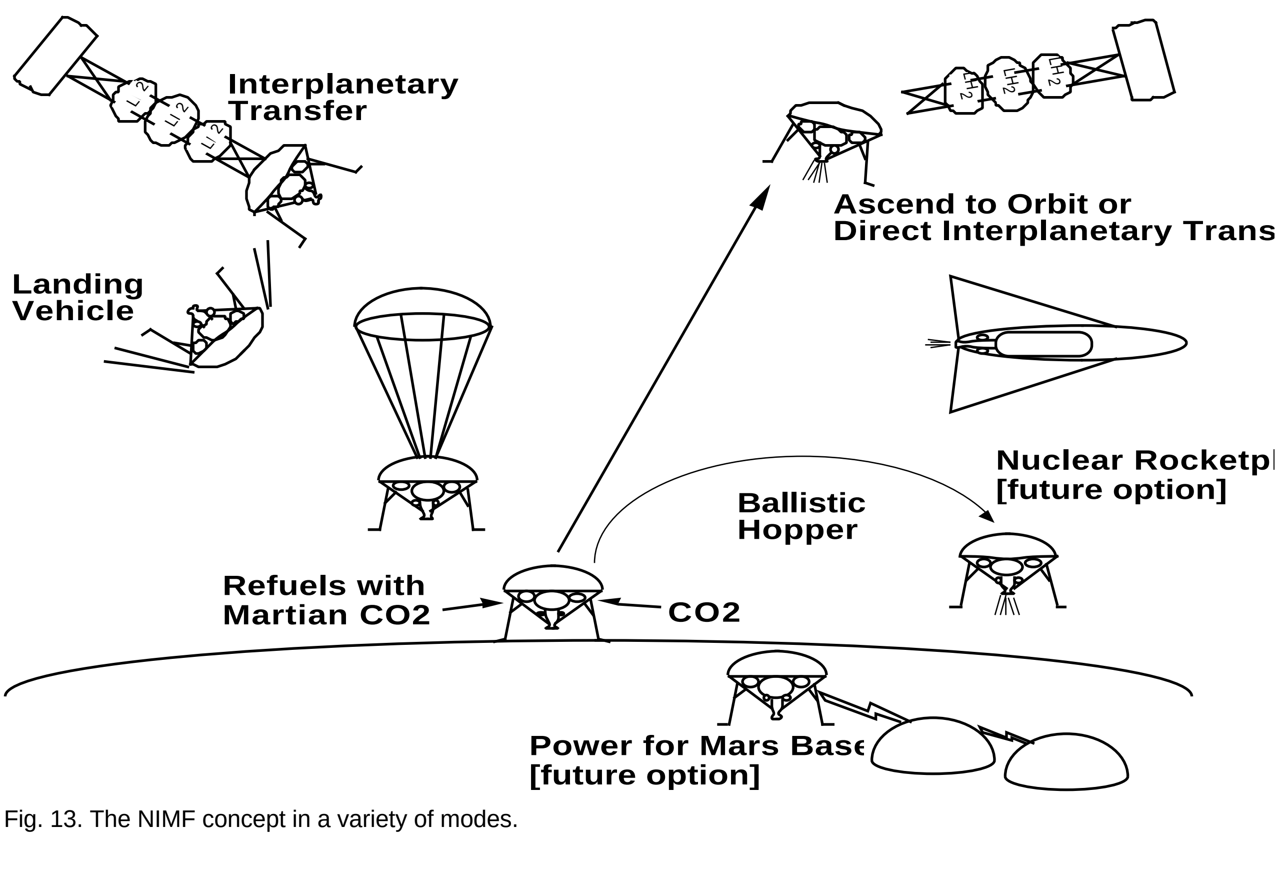

Fig. 13. The NIMF concept in a variety of modes.

to specific impulse, allows for a much higher thrust to be generated by a NIMF engine operating at a given power level than a conventional hydrogen fed NTR of the same power. Assuming a propellant temperature of 2800 K, a specific impulse of 264 s can be obtained with a nozzle expansion ratio of 100. Such a performance would give the NIMF the capability of attaining a 250 km by 33000 km (250 by 1 sol) elliptical orbit about Mars. However, even a modest propellant temperature of 2000 K would still give it the actually or more important ability to hop from one point on the surface of Mars to any other point in a single hop. Because CO_2 becomes an oxidizing medium when heated to elevated temperatures, conventional NERVA type carbide fuel elements cannot be used in a NIMF engine. Instead either oxide or oxide coated fuel pellets would have to be used. Uranium-thorium oxide has a melting point of about 3300 K, and such pellets coated with a layer of either zirconium or thorium oxide to retain fission products, could well enable operation at 2800 K. Alternatively, preliminary data[22] indicates that "traditional" NERVA uranium carbide fuel elements coated with graphite can have their graphite coated with a further layer of thorium oxide, and that such thorium oxide outer coatings are resistant to both CO_2 and solid-solid reactions with the graphite at temperatures up to 3000 K. Because the NIMF requires high T/W ratios to take off from the martian surface, a particle bed geometry for its core is probably the most appropriate choice.

CO_2 can be liquefied out of the martian atmosphere by simple pump compression at an energy cost of about 84 kWe-hrs/tonne. What this means is that the NIMF, using a 25 kWe power source (either DIPS, deployable solar, or dual mode NTR) can completely fuel itself in less than 50 days, without any dependence on surface infrastructure.

Since a typical conjunction class stay is about 550 days, the use of the NIMF offers an increase in the effectiveness of a Mars expedition by about a factor of 18, since with an average refueling time of 30 days, the astronauts would be able to use the NIMF to visit and explore 18 martian sites instead of the usual 1. Once such global reach is available to sortie parties, there will no longer be a need to land new Mars missions at widely scattered sites. Instead, the exploration imperative can be met by NIMF sorties, while successive Mars mission landings are concentrated at a single site. There a large base can be developed with a sufficient crew size for a significant division of labor and thus the beginnings of real industrial and agricultural capabilities. The pioneering and mastery of the utilization of local resources achieved at such a base will make it the beach-head for the eventual settlement of the Red Planet.

In summary, we see that the Mars Direct architecture is highly amenable to the introduction of NTR technology, which in turn can lead to NIMF technology. This defines an evolutionary growth path for the capability of the Mars Direct architecture which leads naturally to massive increases in humanity's ability to explore, and ultimately colonize, our neighboring worlds.

CONCLUSIONS

In conclusion we find that the Mars Direct architecture provides a simple , robust, cost-effective, and coherent plan for the Space Exploration Initiative. It enables an early commencement of useful SEI operations, and right from the start it conducts missions in such a way as to minimize cost and maximize exploratory return. The Ares booster used by Mars Direct has been shown to be a highly attractive and versatile option for the nation's next heavy lift launch vehicle. The stepped up conjunction class trajectories used in Mars direct are found to minimize crew radiation exposure more effectively than the inefficient, costly, and risky high energy opposition class trajectories that have been the focus of much recent attention. The in-situ propellant production processes used by Mars Direct are found to have an extensive historical and industrial basis, and to already exist in a high state of maturity within certain areas of the space program itself. Such propellant processes have also been shown to be required if substantial surface mobility on Mars is to be achieved. The Mars Direct in-situ processes have also been shown to provide a basis for a very attractive Mars Rover Sample Return precursor mission. Power requirements for the manned Mars Direct propellant production can be met by near term surface nuclear electric systems. We have identified NTR as the advanced propulsion technology that is most compatible with the Mars Direct architecture, and have shown that it provides Mars Direct with a growth path leading in an evolutionary way to an order of magnitude increase in exploratory capability on both the Moon and Mars. We therefore recommend that the Mars Direct architecture in all its phases be made the subject of intense study by the nation's space planning bodies, as a leading option for getting the Space Exploration Initiative off the ground.

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