Project Rigel: Mars Sample Return

MarsDrive Mars Sample Return Contest - Winner

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Abstract

This paper covers a design for Mars Sample Return entered in the MarsDrive Mars Sample Return Competition. The name Rigel was chosen because this star is the leading footfall of the constellation Orion, and Orion is the key vehicle in the Vision for Space Exploration. Since this mission has a lot of nested criteria, it begins with a survey of 20 possible configurations condensed into a series of graphs showing the limitations imposed by these criteria. After identifying these limits, the design with the greatest margins is selected for two more design iterations. Schematics, component selections, and mission timelines are then presented for this design. Whenever possible, technology and components from other Mars missions are used. Where new equipment is designed, it is relentlessly simplified to reduce the probability of mechanical failure and development cost issues. Finally, budget and spin-off designs are considered that would build on this design. The spin-offs are considered very early in the design process to allow for follow-up missions using as many components of the original mission as possible, or to allow for changes in primary landing site based on future discoveries.

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Examining the Solution Space

Limitations of Current Entry, Descent, and Landing Technologies

A recent NASA lecture discussed the issue of Entry, Descent, and Landing (EDL) on Mars. In a nutshell, the state of the art has depended on expensive testing from the Viking era for parachute and entry capsule design to the point that with the Mars Science Laboratory (MSL) mission, we will have reached the upper limit of our current engineering for aeroshell design. Even validating existing designs requires launching sounding rockets from high altitude balloons at a cost of $20 million per flight. Due to ballistic coefficients, landing capsules cannot grow larger or heavier than MSL for this design. Further, it was stated that while MSL plans to land an 800 kg payload on the surface, a Mars Sample Return (MSR) mission would need to land 1200 kg [1]. Note that these numbers ignored the MSL landing stage, which would bring the total landed mass for MSL to 1319 kg, plus 219 kg propellant.

Parameter	**Viking [5]**	**MER [7]**	**MSL [9]**	**NASA MSR**
Landing Ellipse	NA	80,000 m	20,000 m	20 m [1]
Capsule Diameter	3.5 m	2.65 m	4.5 m	4.5 m [9]
Entry Mass	900 kg	800 kg	3250 kg	3250 kg [9]
Landed Mass	576 kg	174 kg Rover 348 kg Lander	775 kg Rover 544 kg Descent Stage 219 kg Propellant	1200 kg [1]

Viking development dovetailed the end of the Apollo-era budgets, and was originally to be launched on a Saturn booster under the name Voyager 1973 [16]. There have been statements from NASA scientists that Orion technology could be used for Mars Sample Return [2]. While the Orion capsule is the same shape and is larger than the MSL capsule (5 m as opposed to 4.5 m), the capsule is far too heavy as designed to land on the surface of Mars. A ballistic coefficient of 100 is ideal for Mars entry, whereas a coefficient of 150 is too much. The Orion CM ballistic coefficient is 325 [1]. Theoretically, a lighter version could be created, but the commonality of parts would be sacrificed.

The Solution Space Spreadsheet

For the sake of completeness, this paper initially covers twenty strawman designs in a spreadsheet based on sample return size. It then selects a single design, iterates it with more detailed information, and expands it into three variants optimized for operation at different latitudes.

Most of these designs will exceed the criteria that the total mission cost be held to $2 billion. That said, extending the solution space well beyond this limit for the initial solution space analysis opens several doors.

- While the scaling up of current technology to sample return technology is a fairly short jump, the further scaling towards the human-rated technology scale should be examined to see if any dual-use technologies can be determined, or if there is a suggestion for a post-MSR mission that would help bridge the engineering gap between MSR and human missions.

- Many technologies, in particular methane engines, only currently exist at much larger scales than required by a MSR mission. Given the high cost of engine development, including a larger scale mission within the data set is a logical move for the initial analysis.

- Certain technologies, such as life support, can use of Reverse Water Gas Shift and related ISPP methods. By using these technologies in MSR, we increase the duration of testing under Mars conditions before human missions trust this technology. Therefore including these items, as well as large scale power production and other technologies, becomes a logical move for a forward-looking program.

Showing all 3000 cells in the 20-design first iteration spreadsheet is not physically possible in this paper. Instead, we will first explore the assumptions from the first, second, and third iterations in the table below. In the next section, several graphs will show the data from the 20-design first iteration to show how the core design was selected. Finally, another table will list the selected first iteration design, the second iteration, and the two variants of the third-generation design.

Parameter Spreadsheet Assumptions

Below are the assumptions for the three iterations of design discussed in this paper.

<table>
<tr><th>Parameter</th><th>Assumption</th><th>Data Source</th></tr>
<tr><td>Propellant</td><td>Ethylene and Oxygen</td><td>This combination is ideal for smaller missions due to the reduced need for hydrogen and smaller tanks needed for the return vehicle.</td></tr>
<tr><td>Stage 1 Thrust/Mass Ratio</td><td>0.7662</td><td rowspan="2">These ratios are from Mars Direct [3]. The first stage also follows the general principle of a rocket generating twice the trust required to hover in local gravity. A single engine is used for both ascent and descent, and this method is validated later in the paper.
Deeper work on this indicates the ratio of 0.894 is ideal for a liquid fueled first stage, but that 0.7662 doesn’t induce much gravity loss penalty [20]. Given the high drag of the design, the original figure of 0.7662 is probably closer to ideal for this case, since lower thrust equates to less drag loss.</td></tr>
<tr><td>Stage 2 Thrust/Mass Ratio</td><td>0.2678</td></tr>
<tr><td>Return Entry Capsule Mass</td><td>Ranges: A 6 to 60 kg capsule containing a 0.5 to 5 kg sample mass.</td><td>The minimum is based on Dr. Zubrin’s ISRU demonstration MSR design [4]. The maximum is based on an EADS design [8]. In both cases, the ratios of sample-to-capsule mass are the same, so it is assumed that this would remain true in each intermediate design.</td></tr>
<tr><td>Return Cruise Stage Mass</td><td>Minimum – 24 kg for a 6 kg capsule and 0.5 sample.
Maximum – 50 kg for a 60 kg capsule and 5 kg sample.</td><td>Cruise stage masses vary widely on historic Mars missions. The minimum is based on the Zubrin ISRU Demo paper[4]. The maximum based on the Mars Polar Lander cruise ring, which was a 56 kg cruise stage for a 494 kg entry capsule [5]. In Rigel Iteration 2, the mass was increased to allow for RCS pods.</td></tr>
</table>

Stage 2 (TEI) Dry Mass, Stage 1 (Ascent) Dry Mass	Engine mass: 15.5 percent of thrust. Tank Mass: 12 percent of propellant mass. RCS mass: 7 kg per pod including 2 kg propellant.	Engine mass uses a 15.5:1 thrust to weight ratio. This is based on a study of existing methane engines [17]. Tank mass uses a heavier version of the standard formula because it is divided into two sections per tank. Since the mass of the propellant doesn't change, the mass of the tank doesn't change regardless of shape, so the larger figure of 12 percent rather than 10 percent [4] is used. RCS pods are rooted in current monopropellant designs that were extracted from mass breakdowns of the cruise stages of various Mars landers [7].
Propellant Pumps	Pump mass: 500 grams per pump. Based on J C Whitehead design for MSR applications, Laurence Livermore Labs.	This design fits in the performance gap between light pressure-fed systems and large turbine systems. It uses essentially very small 4-cylinder radial engine design designed specifically for Mars Sample Return that consumes 2 percent of the propellant. The prototypes are 300 grams [18]. It is assumed that a production version would be heavier and higher capacity. To avoid having to do two different designs, four pumps are used for the first stage and two pumps of the same design are used for the second.
Tank Pressurization	Unnecessary in primary design.	The pumps in this design make additional pressurization unnecessary [18].
Propellant Mass	Basic rocket equation with exceptions listed below in Delta V and ISP inefficiencies, plus 1 percent residual propellant.	The equations used allow for 2 percent fuel for pumps (covered under Pumps above and Delta V, below) and 1 percent adhesion loss in the plumbing. The first figure is a Delta V penalty (below) and the second figure is dead weight. Both are included in propellant production requirements.
Delta V	First stage: 4140, but altered to allow for drag to 4157 with first stage circularization or 3900 for second stage. Second stage: 3821 ideal without circularization and 4078 with	The work of J C Whitehead on MAV design allows for the higher relative aerodynamic drag for a small vehicle missing in the textbook figure of 4140. This 4140 figure also allows for orbit circularization with the first stage (full SSTO). [18] By shifting the burden to the TEI stage, relative numbers for both concepts will be compared. Since the spacecraft is very wide, an aerospike is added to the shape so that once it goes supersonic, the shock cone will limit the stress

	circularization. Non-rotating planet used so that design will not be latitude-dependant.	and drag of the remaining vehicle. J C Whitehead's equations assume a non-rotating planet [18], and the primary design of this MAV is for equatorial landing, which would add 433.4 km/h to the rotational velocity (circumference divided by sidereal day). This is another buffer for the equatorial design and allows room in the design for repeat use at higher latitudes if the buffer remains unused in future refinements.
ISP	Ideal for Ethylene Oxygen (369)[4] minus 2 percent for pump fuel [18] and 1 percent for design inefficiencies: 364.7.	In reality, the 2 percent is partially mitigated by forcing the pump exhaust into the thrust vector of the vehicle engine, plus or minus any RCS inputs that happen to be along that same vector. Since the engine operates at low atmospheric pressure over the whole flight envelope, the engine bell design is near-ideal across the ascent. This makes the 1 percent inefficiency allowed over that range realistic for these initial estimates.
Landing Propellant	Stored in first stage tanks and sufficient to give vehicle at landing mass a delta-V of 630 m/s.	The 630 m/s delta-V is from the Human Spaceflight text. [6] Current technology (MSL) can land in a 20 km ellipse, whereas this amount of power allows + or – 4.5 km lateral translation capability. As such, the system would be programmed with several safe landing zones within the landing ellipse, with the closest one selected by software on landing.
Landing Stage Structure/ Landing Gear	Estimates begin at 150 kg and run to 350 kg.	Consists of framework and landing gear. This is roughly on par with the entire MPL (290 kg) and the landing stage of the MSL (544 kg) [5, 12]. Note that this does not include the landing propellant, engines, solar arrays, the LH2 tank, core ISPP system, or avionics. It does include deployment mechanisms for solar arrays and manipulators needed to carry samples from rovers to the return capsule, as well as some of the external components of the ISPP system.
Landing Stage Avionics	Fixed at 20 kg	These are avionics included in the landing stage. They include landing radar, cameras, computers, etc.
ISRU Plant Mass	30 kg (Iteration 1) 60 kg (Iteration 2-3B)	This is roughly double the estimate for a non-redundant ethylene/oxygen production unit in Dr. Zubrin's original paper [4] – this allows for duplex redundancy in hardware.

H2 Tank Dry Mass	Assumes a standard aircraft aluminum alloy 5 mm thick.	This figure is for mass estimates alone, as the real tank would have a metal inner shell surrounded by wound composite for both strength and insulation, and finally insulation layers.
ISRU Electrical Power Demand	The average kg of propellant needed per day (total propellant divided by 432.8 days) multiplied by 575 watts per kg needed.	While Dr. Zubrin's original work starts with 0.5 kg/day and increases by multiples of 10, I derived the power required per 1 kg of propellant per day to simplify conversions [4]. This was compared to the original when the values crossed over and is relatively accurate for power. Because the design requires more oxygen than a simple conversion of all the hydrogen in the tanks would create, one must calculate how many days are required to produce the fuel, how much oxygen is also generated, then add to that the number of days needed with the equipment in oxygen-only mode to finish the job. This must equal 500 days or less. Repeated iterations at different production levels showed consistently that any design must produce ethylene and oxygen for 432.8 days, then oxygen alone for 67.2 days to produce sufficient propellant in 500 days.
Solar Array Surface Area	530 watt-hours per Sol per m2.	The output of the MER rovers averaged between the peak of 750 watt-hours per Sol per square meter and the output at the 1000 day point of 311 watts/sol/m2 [7]. The output will vary based on dust accumulation, dust devil cleaning events, and the 500 day production schedule rather than 1000 day mission. Therefore a middle point was selected for conservative calculations.
Conversion from ISRU demand to Solar Array supply.	Power demand * 12 hours / 530 watt-hours per Sol / m2.	One issue in research was the lack of specificity from various sources on the power requirements being in watts (1 joule), watt-hours (3600 joules), or watt-hours per sol. Extensive digging into these designs implies the demand is based on watt-hours due to the solar power systems designed to meet the needs of these systems. The original ISRU output is based on solar output over a 12 hour day [4], so it has been multiplied by 12 to match the output of the MER-technology solar arrays.
Solar Array	Watt-hours / 12.5 =	This figure appears in one of Dr. Zubrin's

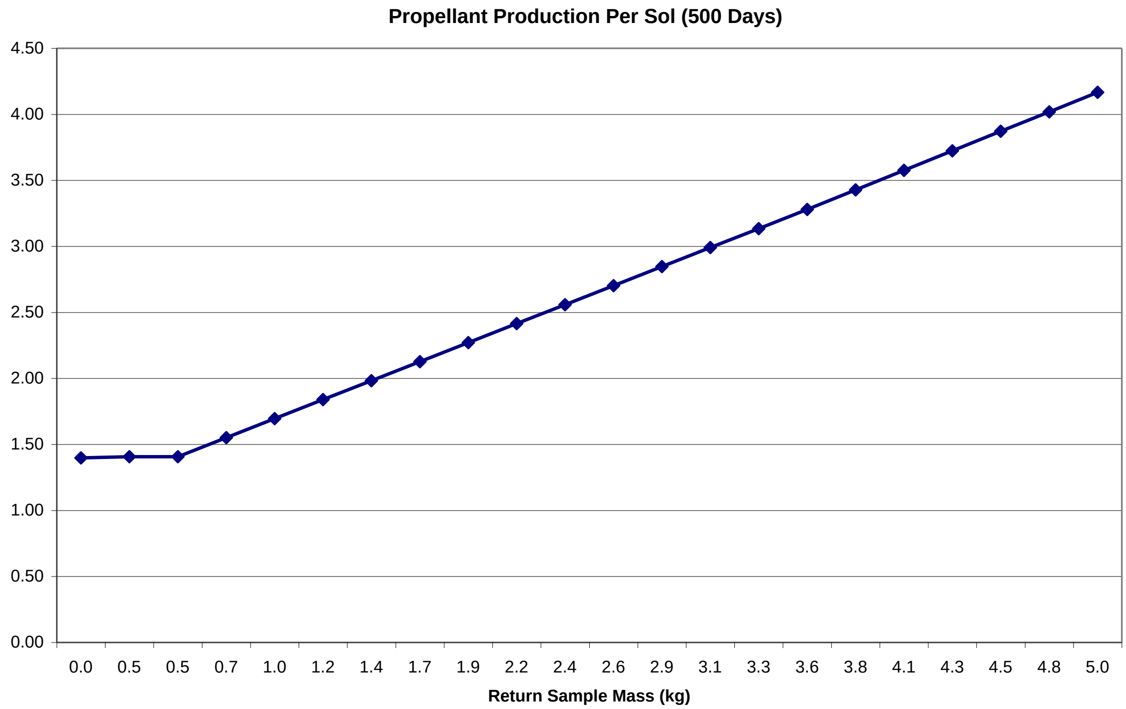
Mass	solar array kg	papers [4] and is verified by comparison with other solar arrays for Mars applications [7].
Landing Aeroshell, Heat Shield, and Cruise Stage Mass, Shape, and Volume	MER – 2.65 m diameter for smallest design. MSL – 4.5 m diameter for all others	The entry mass of even a vehicle that launches only hardware and no sample is 1.445 times that of the MER entry mass. As such, this pushes the ballistic coefficient of the capsule out of the scope of what the supersonic gap-ring parachute can handle [1]. For the sake of demonstration, the 0.5 kg sample size vehicle is listed in both MER and MSL entry capsules.
Rover Mass	3 Solution Sets: Proportional (20-100 kg), fixed at 100 kg, and fixed at 185 kg (MER-sized)	The proportional set scales the rover to match the rest of the vehicle. The smallest rover is over twice the size of Sojourner. Ultimately, because of the ratio of sample return mass to landed mass is so dramatic, the largest (MER-sized) rover is selected to give as many high quality small samples as possible with demonstrated durability in the Mars environment.

Iteration 1 Solution Space Graphs and Conclusions

Iteration 1 assumed a perfect textbook vehicle with zero fuel waste. Issues with RCS mass and using propellant for driving the pumps were not included in this iteration. That said, some issues, such as duplex redundancy (and double mass) for the ISRU systems, is included in this estimate. This was the rough cut to determine the basic shape and size of the vehicle.

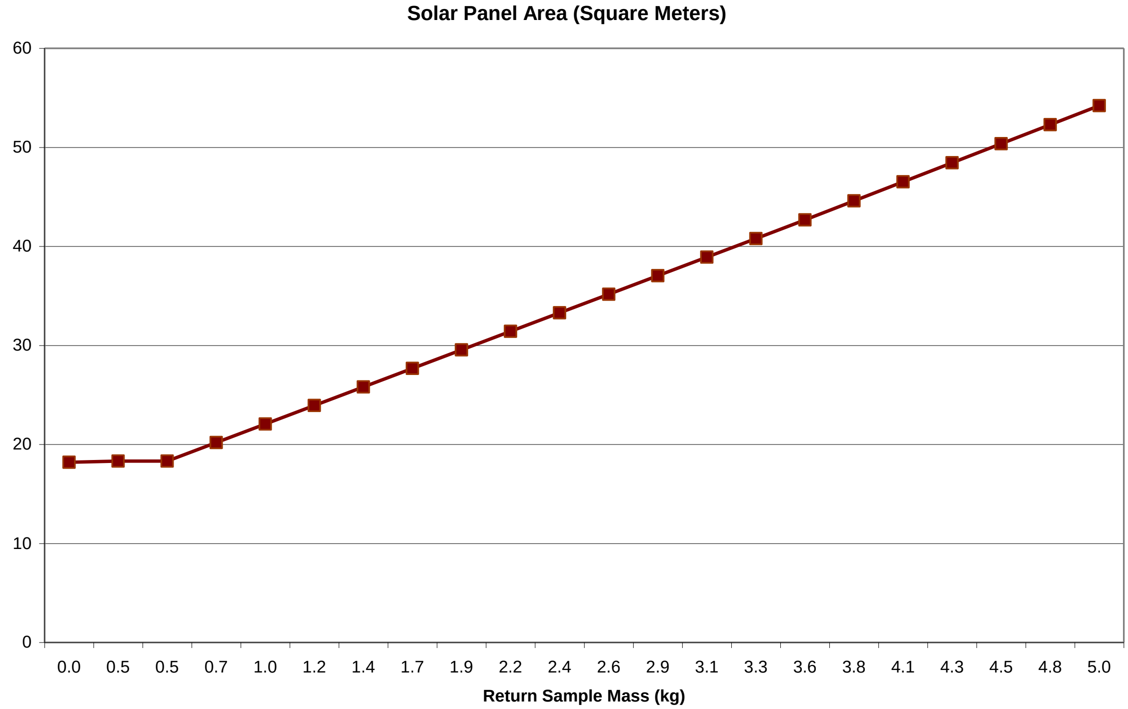
With these charts, there is a flat curve for the first three data points. The first two points, for 0 kg and 0.5 kg return sample mass, respectively, assume that the vehicle is fit into the Mars Exploration Rover (MER) 2.65 meter diameter aeroshell. The numbers after that, starting with a repeat of the 0.5 sample, assume the use of the Mars Science Laboratory (MSL) 4.5 meter diameter aeroshell. One of the first conclusions is that the MER capsules are not practical for a MSR design, but that the MSL capsule has sufficient room for fairly ambitious MSR designs.

Figure 1: Sample Size to Average Fuel Production Rate



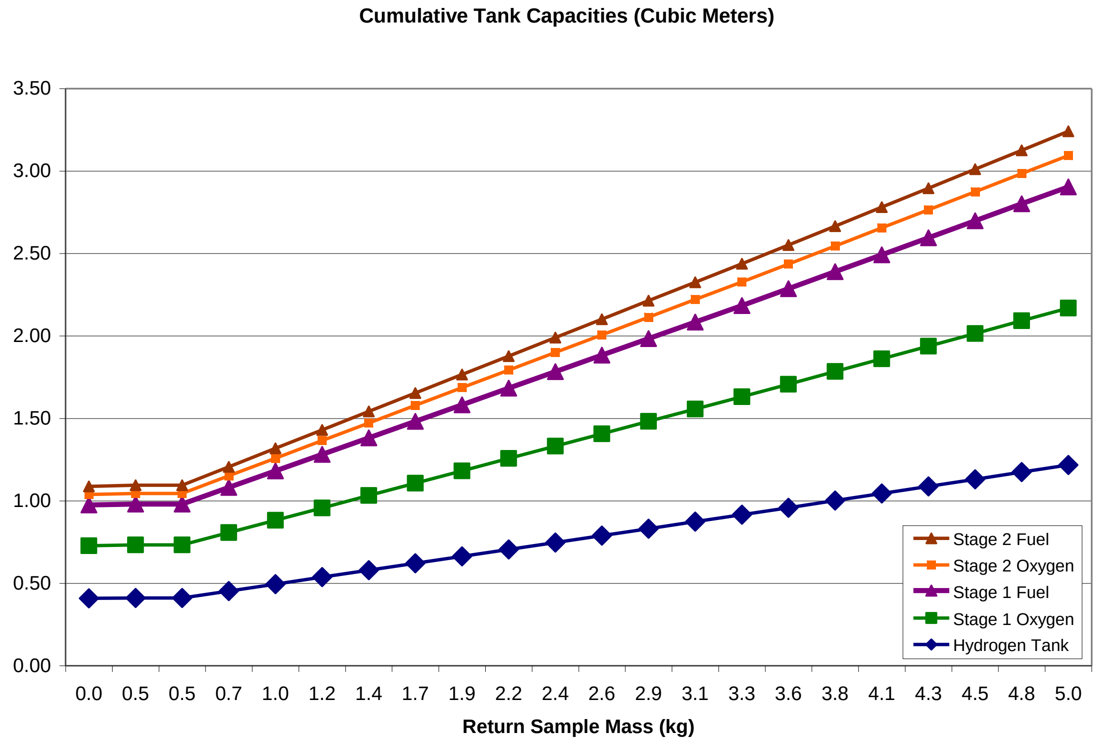
These numbers are based on the assumption of a 500 day stay, with 432.8 days devoted to producing ethylene and oxygen using the hydrogen on board, and the remaining days devoted to producing only oxygen to reach the appropriate fuel to oxygen ratio. The durations of these phases were calculated each time and perfectly consistent regardless of the scale of the vehicle. Note that a system producing 5 kg/day would be big enough for life support on a human mission in terms of recycling carbon dioxide exhaled by crewmembers. A dual-use system with some level of redundancy (say two matched units for Mars and three for a human long-term vehicle) could give field testing of these units and economy of scale.

Figure 2: Sample Size to Solar Array Area



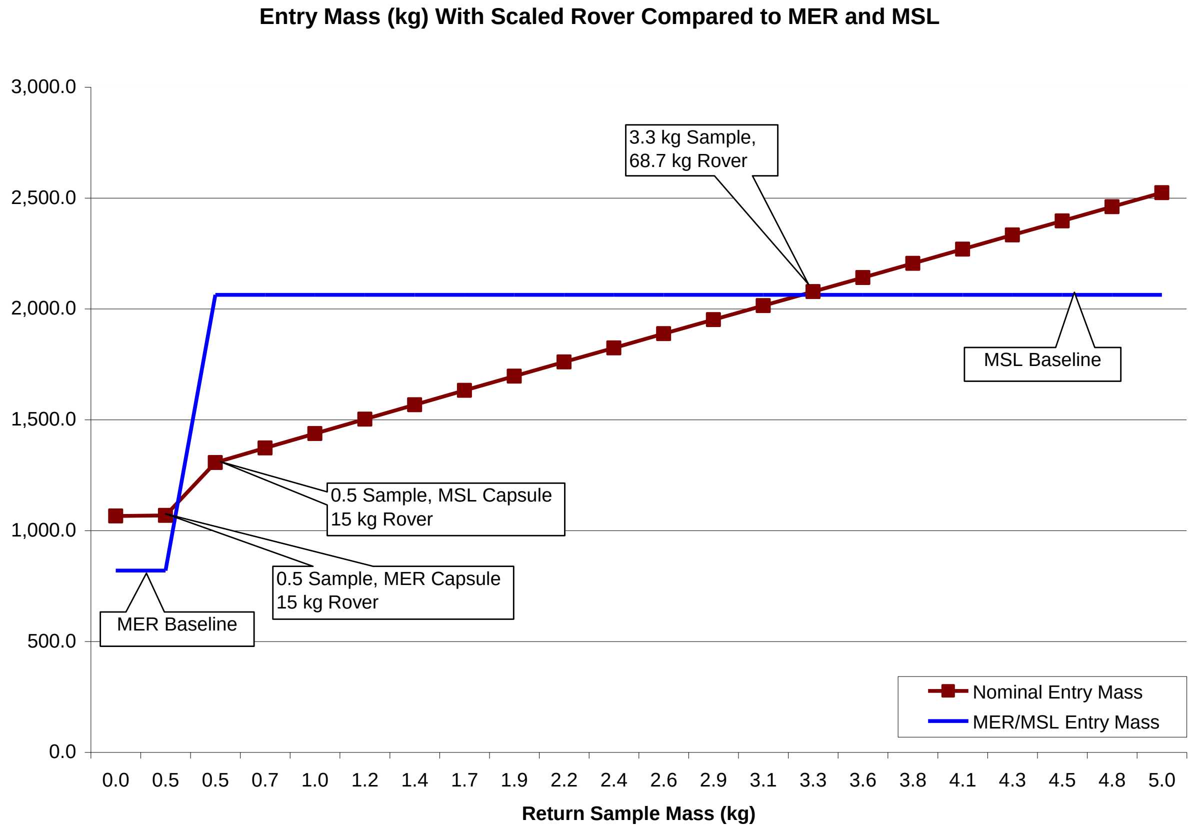
The real problem for scale is shown above. Even an empty sample capsule requires an array of 18 square meters. Cost, ballistics, mass, and other technology issues are less limiting than the issue of how much solar array can deploy autonomously, in rough terrain, with a high likelihood of success, while being packed into the smallest, lightest configuration possible.

Figure 3: Sample Size to Tank Volumes



This graph gives the volume of each major tank in the system, starting with the hydrogen tank and then the first and second stages in turn. Values are cumulative but listed individually – no combined tanks are shown. If tanks were combined, or made into a single pressure vessel with a bulkhead in between, this could be made more efficient for each stage. Also note these are interior capacities, not exterior volumes. This graph proved that fitting a MSR vehicle in the 4.5 meter MSL capsule is realistic, even if the capsule itself is not spherical and the tanks cannot be fused together in this manner.

Figure 4: Sample Size to Vehicle Entry Mass (Scaled Rover)



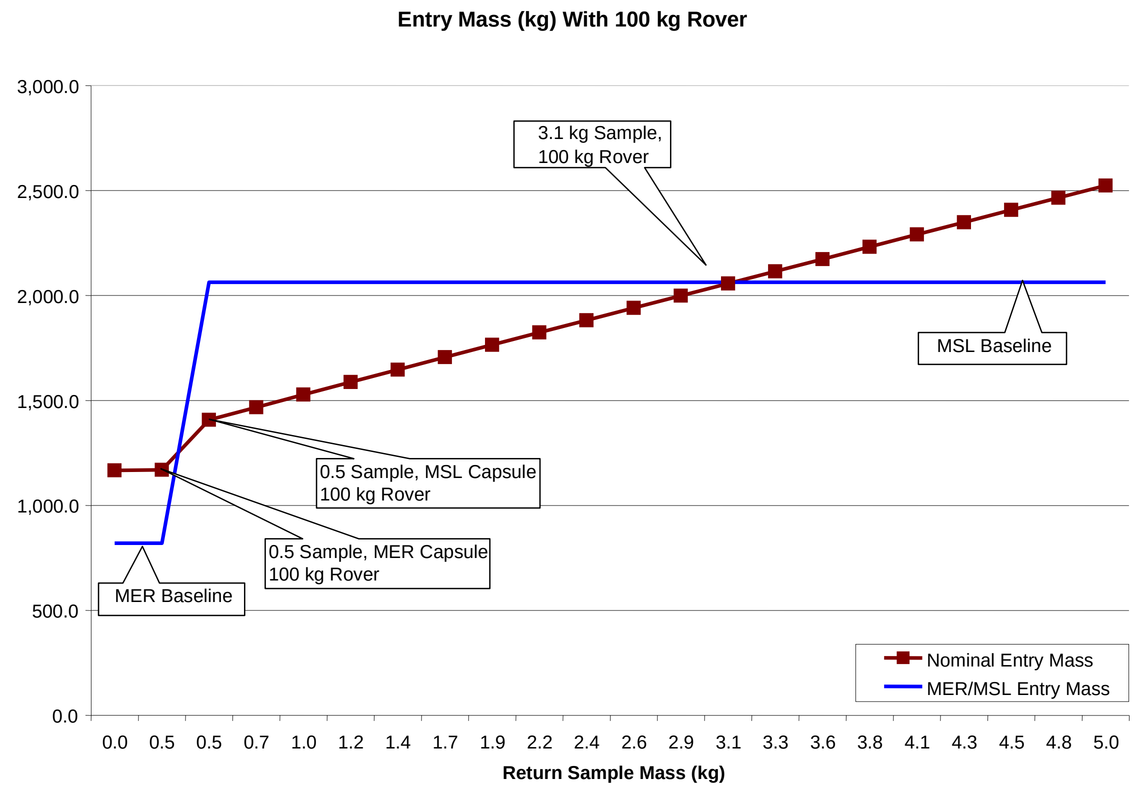
The term "Scaled Rover" means that the rover mass is scaled to the mass of the overall vehicle, with data points evenly spread from 20 kg on the low end to 100 kg on the high end. For reference, Sojourner is 10.6 kg [5] and the MER rovers are 174 kg [7]. It is assumed that even without a rover, local sampling equipment for drilling would still be roughly 20 kg.

This graph tells us several very important things. First, as noted, the first two figures use the MER 2.65 meter capsule, whereas all the rest use the MSL 4.5 meter capsule. Note how much heavier the vehicle is than the MER in terms of entry mass. The actual values for the 0.5 kg sample are 1068.7 kg for the strawman MSR versus 820 kg for MER. Since MER was already nearly double the landed mass of the Pathfinder for the same size entry capsule, it is reasonable to assume the ballistic coefficient of the MER is already close to maximum. To add another 45.7 percent to the mass is unrealistic. It would not slow down enough with the heat shield to deploy the supersonic ring parachute. Further, even if it did work, the scale of the TMI mass is also increased by 23.1 percent, which would result in reengineering the launch vehicle interface to include a larger TMI stage and larger launch vehicle. Also note these are strawman figures for a mathematically perfect design with no margins – the actual numbers will grow with future iterations.

When we jump to the much larger MSL 4.5 meter entry vehicle, the relative entry mass gives a great deal of payload flexibility and design margin. All else being equal, the design can be expanded to return a 3.3 kg sample return before becoming exceeding the MSL mass in the ballistic coefficient. MSL is designed to land at much higher altitudes than earlier missions. Also, MSR has the advantage of being able to handle larger landing masses simply by adding more fuel – at least until the combined mass equals that of MSL. This gives the MSL capsule – MSR mission design combination a very wide range of capability. Given this range, the appropriate course is probably to build a minimalist mission with a lot of safety margin for the first round, then build on the design heritage for one or two follow-up missions.

According to the earlier graph, the solar array needed to fuel this 3.3 kg sample return mission would be 40.78 square meters. Logically, a better use of this margin (at least at this phase) would be to scale up the rover to increase the quality and variety of samples rather than increase the sample mass returned to scale up the quantity.

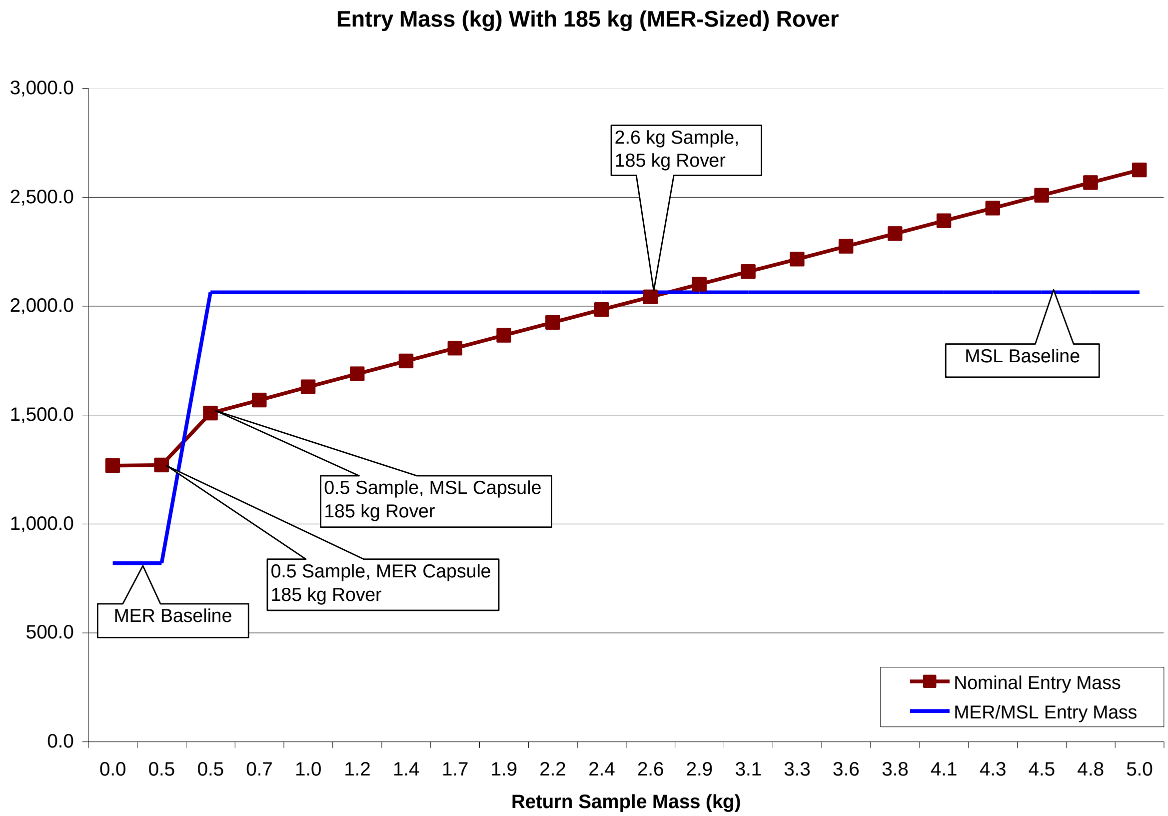
Figure 5: Sample Size to Vehicle Entry Mass (Fixed 100 kg Rover)



For this illustration, all the rovers are simply scaled up to 100 kg rather than making them proportional to the remainder of the vehicle. Note that the maximum sample size now drops from 3.3 kg (with a 68.7 kg rover) to 3.1 kg. In other words, with the addition of 30.3 kg of rover mass, we only cut 0.2 kg from the sample return mass.

Figure 6: Sample Size to Vehicle Entry Mass (MER-sized 185 kg Rover)

While the 100 kg rover above could be designed and may even use some parts from MER, it seems logical to scale the mission as if the entire rover were built from MER components. Creating more copies of the MER rovers with sample return modifications would involve minimal developmental cost. Further, since the vast majority of these systems have been field-tested over two vehicles for twice the required mission of the MSR rover, they can be assumed to be robust enough for this operation. Note that the NASA Fact Sheet states that the rovers are 174 kg, whereas the web site states it weighs 185 kg. Since it will be modified for sample return anyway, the higher figure is used.



Increasing the rover to MER proportions has dropped the maximum sample size to 2.6 kg – which would still require 35.16 square meters of solar array to power the In Situ Propellant Production (ISPP) system. If the sample is scaled down to 0.5 kg, and the rover remains the mass of the MER, we still have a vehicle that is 73.1 percent the entry mass and 77.5 percent the launch mass of MSL. This is the initial, idealized iteration. However, these are very broad margins to work with.

The nominal mission, returning a 0.5 kg sample to Earth, can carry an additional 739.7 kg of equipment including the rover. This vehicle lands with first stage fuel tanks that are 20-30 percent filled to capacity (depending on how much equipment is landed and the sample return mass). As a working model, this uses the best of (by then) past missions while developing only enough new hardware (sample collection equipment, return vehicle, ISPP, large solar arrays) to complete the basic mission.

Design Figures for Iterations 1, 2, 3A, and 3B

Iteration 1 is the selected design from the 20 listed above. Iteration 2 adds RCS to the second stage and some additional margins. This is where the first release of this paper ended. Iteration 3 adds more realistic waste margins to the system, propellant pump fuel demands, and some atmospheric factors. 3A assumes the first stage circularizes the orbit, whereas 3B assigns that task to the second stage.

MSR Project Rigel	**Iter. 1**	**Iter. 2**	**Iter. 3A**	**Iter. 3B**	**Unit**
Stage 1 Rocket Equation					
Propellant Mass	589.63	761.96	933.24	877.12	kg
Mass before burn (Mo)	873.94	1129.36	1357.84	1320.97	kg
Mass after burn (Mf)	284.30	367.40	424.60	443.85	kg
Gravity	9.81	9.81	9.805	9.805	
deltaV (M/Sec)	4140	4140	4157	3900	M/sec
ISP	376.00	376.00	364.7	364.7	sec
Stage 2 Rocket Equation					
Propellant Mass	126.69	152.08	164.52	188.55	kg
Mass before burn (Mo)	196.33	235.68	250.60	277.15	kg
Mass after burn (Mf)	69.64	83.60	86.082	88.6	kg
Gravity	9.81	9.81	9.805	9.805	
deltaV (M/Sec)	3821	3821	3821	4078	M/sec
ISP	376.00	376.00	364.7	364.7	sec
Landing Rocket Equation					
Propellant Mass	163.78	196.40	219.81	217.66	kg
Mass before burn (Mo)	1042.63	1250.33	1360.77	1347.49	kg
Mass after burn (Mf)	878.85	1053.93	1140.96	1129.82	kg
Gravity	9.81	9.81	9.805	9.805	
deltaV (M/Sec)	630	630	630	630	M/sec
ISP	376	376	364.7	364.7	sec
Landing Prop/Take-off Capacity	0.278	0.258	0.236	0.248	
Engine Parameters					
Stage 1 Thrust/Weight Ratio	0.766	0.766	0.766	0.766	
Stage 2 Thrust/Weight Ratio	0.268	0.268	0.268	0.268	
Stage 1 Landing Thrust (kgf)	798.84	957.98	1042.60	1032.42	kgf
Stage 1 Launch Thrust (kgf)	669.59	865.29	1040.35	1012.10	kgf
Stage 2 Launch Thrust (kgf)	52.571	63.108	67.102	74.211	kgf
Engine Thrust/Weight Ratio	15.500	15.500	15.500	15.500	
Stage 1 Engine Mass	43.199	55.825	59.100	57.400	kg
Stage 2 Engine Mass	3.392	4.071	3.410	3.770	kg
Vehicle Components					
Capsule	20.0	20.0	20.0	20.0	kg
Cruise Stage	24.0	24.0	24.0	24.0	kg
Cruise Stage Fuel	4.0	4.0	4.0	4.0	kg
Sample Mass	0.5	0.5	0.5	0.5	kg
Earth Return package (total)	48.5	48.5	48.5	48.5	kg

MSR Project Rigel	**Iter. 1**	**Iter. 2**	**Iter. 3A**	**Iter. 3B**	**Unit**
Stage 2 Dry Stage	21.1	35.1	37.6	40.1	kg
Dry Total Stage 2 Mass	69.64	83.60	86.082	88.6	kg
Stage 2, Propellant + 1% waste	126.69	152.08	166.16	190.43	kg
Stage 2, Mass to Orbit	196.33	235.68	250.60	277.15	kg
Stage 1 Mass, Dry	87.97	131.72	174.00	166.70	kg
1 dry + 2 wet + payload	284.30	367.40	424.60	443.85	kg
Stage 1 Propellant + 1% waste	589.63	761.96	942.58	885.89	kg
Stage 1 Total Liftoff Mass	873.94	1129.36	1357.84	1320.97	kg
Stage 2 Wet + 1 Dry Mass	353.95	451.00	501.35	523.68	kg
Landing Stage - rover, H2	432.28	526.23	567.67	560.02	kg
Landing Gear, etc.	150.00	150.00	150.00	150.00	kg
Landing Stage Avionics	20.00	20.00	20.00	20.00	kg
ISRU Plant Mass (est. kg)	30.00	60.00	60.00	60.00	kg
H2 Tank Thickness (m)	0.01	0.01	0.01	0.01	m
H2 Tank Material Density (kg/m3)	2700.00	2700.00	2700.00	2700.00	kg/m3
H2 Tank Exterior Volume	0.55	0.69	0.83	0.81	m3
H2 Tank Interior Volume	0.48	0.62	0.75	0.73	m3
H2 Dry Tank Mass (kg)	167.56	196.50	222.98	218.69	kg
Solar Array (est.)	64.73	99.73	114.69	111.34	kg
Rover	185.00	185.00	185.00	185.00	kg
Hydrogen Payload	34.31	43.78	53.11	51.56	kg
Landing Mass - landing propellant	878.85	1053.93	1140.96	1129.82	kg
Nominal Landing Propellant	163.78	196.40	219.81	217.66	kg
Nominal Total Landing Mass	1042.63	1250.33	1360.77	1347.49	kg
Max landing Propellant	589.63	761.96	933.24	877.12	kg
Max Landing Stage Allowed	2244.37	2121.92	2057.49	2010.89	kg
Nominal Entry Mass	1567.63	1775.33	1885.77	1872.49	kg
Aeroshell (backshell + Heat shield)	525.00	525.00	525.00	525.00	kg
MSL Entry Mass	*2063.00*	*2063.00*	*2063.00*	*2063.00*	*kg*
Cruise Stage	*400.00*	*400.00*	*400.00*	*400.00*	*kg*
Nominal Mass From Earth	*1967.63*	*2175.33*	*2285.77*	*2272.49*	*kg*
Maximum Mass from Earth	*6083.57*	*6214.58*	*6367.03*	*6260.81*	*kg*
MSL Total Mass	*2463.00*	*2463.00*	*2463.00*	*2463.00*	*kg*
Ratio of Entry Mass (MSR:MSL)	**0.76**	**0.86**	**0.91**	**0.91**	
Ratio of Launch Mass (MSR:MSL)	**0.80**	**0.88**	**0.93**	**0.92**	
Margin from MSL Baseline	495.37	287.67	177.23	190.51	kg
Fuel Selected	Ethylene	Ethylene	Ethylene	Ethylene	
Fuel Ratio	2.6 to 1	2.6 to 1	2.6 to 1	2.6 to 1	
Total of Ratio	3.60	3.60	3.6	3.6	
Fuel Ratio: Oxygen Part	2.60	2.60	2.6	2.6	
Fuel Ratio: Propellant Part	1.00	1.00	1	1	
Oxygen Fraction	0.722	0.722	0.722	0.722	
Fuel Fraction	0.278	0.278	0.278	0.278	
Oxygen Density Factor	1141.00	1141.00	1141.00	1141.00	
Propellant Density Factor	567.92	567.92	567.92	567.92	
Hydrogen Density Factor	70.97	70.97	70.97	70.97	
LH2/Fuel Ratio	0.144	0.144	0.144	0.144	
Ratio in bold above shows the Iteration 3B vehicle maintains an 8 percent margin below the launch mass and a 9 percent margin below the entry mass of the Mars Science Lab.					

MSR Project Rigel	Iter. 1	Iter. 2	Iter. 3A	Iter. 3B	Unit
Stage 1 Propellant					
Mass of Propellant (kg)	589.631	761.960	942.576	885.892	kg
Mass of Oxygen (kg)	425.845	550.304	680.749	639.811	kg
Mass of Fuel (kg)	163.786	211.656	261.827	246.081	kg
Hydrogen Tank (m3)	0.483	0.617	0.748	0.726	m3
Stage 1 Oxygen (m3)	0.373	0.482	0.597	0.561	m3
Stage 1 Fuel (m3)	0.288	0.373	0.461	0.433	m3
Sphere Dia Oxygen	0.893	0.973	1.044	1.023	m
Sphere Dia Fuel	0.820	0.893	0.958	0.939	m
Sphere Dia H2	0.974	1.056	1.126	1.115	m
Sphere Dia if LOX/Fuel in 1 sphere	0.974	1.056	1.126	1.115	m
Sphere Dia of Each Dual Tank	0.976	0.978	0.980	0.980	m
Stage 2 Propellant					
Mass of Propellant (kg)	126.688	152.081	166.160	190.433	kg
Mass of Oxygen (kg)	91.497	109.836	120.005	137.535	kg
Mass of Fuel (kg)	35.191	42.245	46.156	52.898	kg
Stage 2 Oxygen (m3)	0.080	0.096	0.105	0.121	m3
Stage 2 Fuel (m3)	0.062	0.074	0.081	0.093	m3
Combined LOX/Fuel (m3)	0.142	0.171	0.186	0.214	m3
Sphere Dia Oxygen	0.535	0.569	0.586	0.613	m
Sphere Dia Fuel	0.491	0.522	0.537	0.562	m
Sphere Dia if LOX/Fuel in 1 sphere	0.648	0.688	0.709	0.742	m
Sphere Dia of Each Dual Tank	0.607	0.609	0.610	0.612	m
Landing Propellant					
Mass of Propellant (kg)	163.779	196.404	219.810	217.664	kg
Mass of Oxygen (kg)	118.285	141.848	158.752	157.201	kg
Mass of Fuel (kg)	45.494	54.557	61.058	60.462	kg
Volume of Oxygen (m3)	0.104	0.124	0.139	0.138	m3
Volume of Fuel (m3)	0.080	0.096	0.108	0.106	m3
ISPP Demand					
Total Propellant Needed	716.320	914.041	1108.737	1076.325	kg
Total Fuel Needed	198.978	253.900	307.982	298.979	kg
Total LH2 Needed to Produce	28.593	36.486	44.258	42.964	kg
LH2 Boil-off Allowance	0.200	0.200	0.200	0.200	
Total LH2 Mass Needed	34.312	43.783	53.109	51.556	kg
Total LH2 Volume Needed	0.483	0.617	0.748	0.726	m3
LH2 Tank Minor Axis	1.050	1.100	1.200	1.200	m3
LH2 Tank Major Axis	1.000	1.100	1.200	1.200	m3
LH2 Tank Z Axis	0.900	1.000	1.000	1.000	m3
Actual Capacity of Tank	0.495	0.634	0.754	0.754	m3
Surface ISRU Production					
Propellant Needed (kg)	716.32	914.04	1108.74	1076.32	kg
Surface Stay Time Allowed (Days)	500.00	500	500	500	days
Surface Stay Time Allowed (Sols)		486.62	486.62	486.62	sols
Ave Prod Rate Per Day	1.433	1.828	2.217	2.153	kg

MSR Project Rigel	Iter. 1	Iter. 2	Iter. 3A	Iter. 3B	Unit
Refined Prod Rate Per Day	1.407	2.110	2.562	2.487	kg
Refined Prod Rate Per Sol	1.446	2.168	2.493	2.420	kg
Power Demand of ISRU Unit (est)	809.1	1246.6	1433.6	1391.7	watts/hr
Solar Array Mass (Zubrin)	64.727	99.728	114.690	111.336	kg
Solar Array Area (Zubrin) (Fixed)	21.576	33.243	38.230	37.112	m2
Power Demand Per 12 hr Sol	9709.1	14959.2	17203.5	16700.5	watts/sol
Peak MER Power Per Sol/M3	750	750	750	750	watts/m2
Current MER Power Per Sol/M3	310.67	310.67	310.67	310.67	watts/m2
Peak MER Power - Area Needed	12.945	19.946	22.938	22.267	watts/m2
Current MER Power - Area Needed	31.252	48.151	55.376	53.756	watts/m2
MER Equiv Power Average/Sol	530.335	530.335	530.335	530.335	watts/m2
MER Ave Power Area Needed	18.307	28.207	32.439	31.490	m2
Actual Array Size*	19.00	29.48	33.00	33.13	m2
Actual Array Peak Output*	14,250	22,110	24,750	24,849	watts/sol
Actual Array Average Output*	10,076	15,634	17,501	17,571	watts/sol

* This figure does not include the secondary array.

Solar Array Dimensions	Iteration 2	Iteration 3B
Lateral Array Panels (8)	1.1 X 3.2 m	1.1 X 3.64 m
First Inset Array		1.1 X 0.7 m
Second Inset Array		1.1 X 0.3 m
Secondary Array (Approximated from photos of Phoenix lander, from which this array is derived.)	2 m dia.	2 m dia.

Iteration 3A and 3B Clarifications and Changes

Iteration 3A and 3B include more detailed data on Delta V requirements, and a new pump design developed at Laurence Livermore Labs specifically for MSR vehicles. A cursory remediation for aerodynamic issues on ascent is also proposed. The difference between Iteration 3A and 3B is that 3A uses the ascent stage to circularize the orbit of the return vehicle, whereas 3B uses the Trans-Earth Injection stage to do this task. By doing the staging slightly earlier, a small amount of mass was reduced from the vehicle.

Iteration 3A and 3B also include more inefficiency allowances in the design to move it further from the ideal design and closer to something that would be realistically constructed. It also does not allow for the velocity advantages of equatorial launch. This allows future iterations to either land at the poles if the design is found to be accurate or land at progressively lower latitude limits if the vehicle is found to be optimistic.

Delta V — The standard textbook Delta-V for Mars ascent to orbit is 4140 [6]. According to a more detailed study that specifically focuses on a 100 kg MAV that is liquid-fueled and accounts for the atmosphere (with local speed of sound issues, etc.), that figure is actually 3900 for initial takeoff and 257 for circularization, for a total of 4157. This is a fairly

	minor difference of 17 m/s. However, the vehicle can then be redesigned so that the Trans-Earth Injection (TEI) stage actually does the circularization of the initial orbit, thus removing 240 m/s from the Ascent stage and adding 257 to the TEI stage. The design is calculated both ways to see if the increased demands on the first stage to carry the larger second stage are offset by the lower final velocity required for the first stage.
Propellant Pumps	Work has been done at Laurence Livermore Labs in designing and prototyping a 300 gram reciprocating fuel pump for MAV applications [18]. This work assumes a lower specific impulse propellant (310s versus 376 for Rigel), a lighter vehicle (100 kg versus roughly 1000 kg) and far less engine thrust (102 kg versus 865 kg). That said, the design at this point calls for four pumps for the first stage and two for the second, and each scaled up 40 percent to 500 grams. The dual-dual pumps will allow two per side of the ascent stage. Using six identical pumps rather than different designs for each stage should reduce development costs. It is assumed based on the original work that 2 percent of the propellant will be burned to drive the pumps. After consulting with the author of the paper, this was counted as a percentage against the ISP of the engines. This is partly offset by directing the exhaust in the direction of the engine's thrust. It may also be used to augment the RCS system.
Inefficiencies	Iterations 1 and 2 assumed 20 percent hydrogen boil-off, but did not allow for inefficiencies in the ISPP process or the engine design, so this has been allowed at 1 percent for engine design.
Tank Sizes	The tank sizes for the return vehicle varied by less than a centimeter in each case, so the illustrations later in this paper were not modified for this.
Hydrogen Tank	The hydrogen tank did increase in size and the dimensions were redrawn to be accurate in the illustrations below.
Solar Array	The solar array grew by two square meters, therefore the array panels have been lengthened to allow for more surface area.
ISPP	The use of heat exchangers and other elements that work with the cyclical power environment, high thermal gradients, and day-night cycles have been introduced. While the chemical engineering math for these advances has not been calculated yet, they appear to offer substantial efficiencies over the prototypes designed by Dr Zubrin. That said, none of those efficiencies are assumed in the power calculations.

Examining Configuration Issues

General Design Parameters

The goal of this design will be maximum reliability at minimum cost with maximum design margins where possible. The configuration options will consume some of those margins for the sake of reliability and cost.

Component	**Description**
Sample Return Vehicle	This is scaled to the minimum sample size – 0.5 kg sample mass. The size of the solar array is a major limitation. However, keeping the sample size small gives the most margin for design iteration and the least cost to the overall design.
Rover	The sample collection rover is MER-derived and replaces some of the geological analysis hardware with sample collection hardware. The sample collection systems are kept simple but accurate in labeling. They also allow for shallow core samples, soil and small rock samples.
Solar Array Deployment	The array is a series of 1.1 by 3.64 meter sections that are rolled out across the surface. After considering both a rover and a lander deployment of the main array, the lander option was selected for simplicity and a lack of mutual dependency. The lander can use a single array for an equatorial landing, a double array for a mid-latitude landing, or a triple array for a polar landing.
General Cost Reduction Measures	The methodology for cost control will be as follows: 1. Where possible, MER, MSL, and other then-flown components are reused with as little modification as possible. 2. If a technology was developed for an alternate Mars vehicle that was either canceled or not yet approved, that is considered as the next option because some of the R&D cost has already been spent. 3. If something needs to be built from scratch, the physical design will be kept as simple and foolproof as possible. The general rule is that if it can't be prototyped by a hobbyist in the garage, find something that can.

The Vehicle Design

This section covers each design component in detail.

Sample Return Capsule

Overview

The capsule is a passive low density aeroshell which contains a core sphere of crushable material. Within that is a three-chamber canister that contains soil and rock samples, a compressed atmospheric sample, and an uncompressed atmospheric and filtered dust sample. The overall mass is estimated at 20 kg plus the 0.5 kg sample.

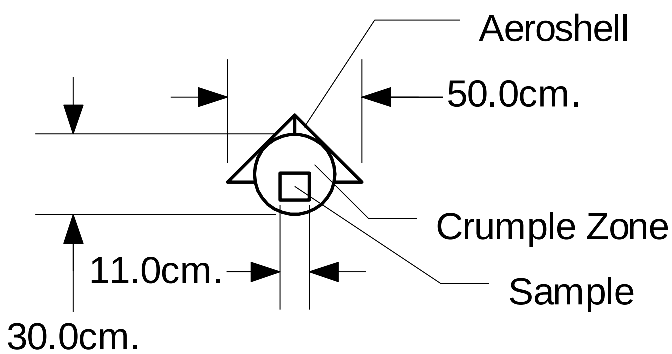
Background

Dr. Zubrin's original MSR discussion described a 0.5 kg sample brought back in a 6 kg passive return capsule [4]. Detailed papers for the ESA from EADS discuss a similar design, but scale it up to a 5 kg sample capsule contained in a 60-80 kg aeroshell [8].

Comparison of EADS and this MSR Design

Component	**EADS Design**	**Rigel Design**
Sample Canister	5 kg with sample	1 kg with sample
Sample Canister Diameter	25 cm exterior	9 cm interior, 10.3 cm exterior
Overall Mass with Sample	60-80 kg	20 kg
Aeroshell Diameter	140 cm	50 cm
Crumple zone diameter	43 cm	30 cm
Cone Angle	45 degrees	45 degrees
Sample Container Shape	Sphere	Cylinder

Diagram



Differences between Rigel and EADS

- The sample capsule is a cylinder rather than a sphere to simplify loading and to allow atmospheric and dust sample sections.

- To give control of weight and balance, and also to amplify the protection of the crumple zone, the vehicle has a disc that fits behind the sample capsule toward the center of the crumple zone. This is also the mount point for fixing the sample capsule in place.

General Tech Level	Passive, low-density capsules arriving at high entry velocities are unprecedented. This is the technology of choice for the Russian Phobos-Grunt mission currently being planned [15]. The vehicle itself (with its lack of moving parts, very simple electronics, and well-understood aerodynamics) requires minimal testing. Early sample returns from the Lunar south pole or other missions that may follow Stardust or Genesis in deep space sample return should standardize on a given passive aerodynamic capsule design where appropriate.
Soil and Rock Sample Container	The soil sample section of this the canister is a 9 cm interior diameter, 6 cm long titanium chamber containing a stack of soil collection tapes from the rover. The rover is equipped with a reel-to-reel “tape” device that allows it to place dime-sized soil or rock samples into individual pockets and seal them in sequence. After each sortie, the section of tape filled during that mission is offloaded to the lander in the event the rover does not return from the next sortie. In the end, the tapes are spliced end-to-end until they form two, 2 by 8.5 cm rolls. The last 2 cm section contains a sectioned bag for small rocks and cores too thick to place in the tape collector. The overall cylinder, when sealed, also contains a sample of Martian atmosphere at ambient pressure. At the center of the three sample rolls is a 6 cm long cylinder that has been used as a soil core sampler by the lander’s manipulator arm. Interior Dimensions: 9 cm diameter by 6 cm long. Exterior Dimensions: 11 cm diameter by 10 cm (this includes the threaded section at the end).
Atmospheric Sample Tank	At the end of this cylinder is a second, 10 by 1.7 cm chamber containing a compressed atmospheric sample. This sample is very slowly pressurized over a period of one sol to 100 times Mars atmospheric pressure, or one Earth atmosphere. The slow pressurization is to minimize possible thermal changes from compression. The compressed sample tank is at the opposite end of the cylinder from the treaded end.
Dust Filter Section	Surrounding both of these containers is a dust filter space. During the ISPP phase, part of the atmosphere pulled into the system is filtered through an air filter material within this space. This zone, 0.7 cm thick, is kept at Martian ambient pressure during the return to avoid compressive transformation of any dust or atmospheric chemistry. The low pressure also adds some vacuum insulation to the core section to minimize any thermal transformation of soil and rock samples. The outer case of this section is relatively thick and is coated with an antimicrobial barrier to avoid any reverse contamination risk.

Since the soil and rock chamber of the canister is empty during most of the surface time, the input valve for the filter is at the very back of the interior cylinder. This reduces the number of openings on the outside of the later-sealed canister and also prevents the filtered dust from pooling on one side and potentially leaving the capsule less balanced.

Samples from this section will be critical to the design of future air filters for crewed vehicles and other robotic ISPP systems.

Sealing Mechanisms

To seal the soil sample cylinder, the cylinder contains two Teflon tape coated threaded elements that are screwed together. Since the threads are filled with tape before launch, they cannot be clogged with dirt or dust prior to the threading operation. The tape itself on both sides would be covered with a thin plastic sheet that could be pulled away with a drawstring just prior to sealing.

Valves for the air samples are closed either mechanically or pyrotechnically in the same manner as a cable cutting guillotine for separating sections of a spacecraft. After closing, a leak test would be conducted and a secondary closing mechanism fired or closed if necessary. Finally, a third pin mechanism would break a glass vial containing an expanding gap filler between the two valves to ensure that the air samples are properly sealed even if an impact or other event jars the valves.

Crumple zone

Surrounding the sample cylinder is the “crumple zone” of low density shock absorbing material. This forms a 30 cm sphere around the capsule. This is surrounded by a 1 cm thick composite shell.

Compression and Balance Disk

The sample cylinder sits on a “plunger” plate 15 cm in diameter and 5 cm above the center point to allow more material between it and the impact point at the bottom of the cone.

One issue with a sample return capsule is weight and balance upon entry, since the mass distribution of the sample cannot be predicted before launch. This impacts not only entry at earth but cruise phase spin stabilization dynamics. After the canister is checked for center of gravity prior to loading, this flat disk may be slid by the loading system up to a centimeter in any direction to offset any sample balance issues.

The disk has a fitting for the sample capsule that can be threaded a specific number of turns to make fine adjustments to the longitudinal axis. The fitting is set within a dual-axis slide that allows it to line up with the sample cylinder regardless as to its offset within the crumple zone material.

Entry TPS Shield

The outer entry cone is a 45 degree, 3 cm thick zone. The overall diameter of the entry capsule is 50 cm. It is made of an ablative material with a high density surface and a low density interior.

Structure

A series of composite spars and ribs bond the entry shield to the crumple zone core. A second skin seals the back of the assembly and contains the electronics in small, thinly-insulated composite boxes. The beacon and strobe are on opposite sides for balance, and two passive transponders are offset from both 90 degrees. Gaps within the structure are filled with very low density foam for added shock insulation on impact.

Mass

The estimated mass, including the sample, was originally calculated at 6.5 kg. To be conservative it will be listed as a total mass of 10 kg including any structural connections.

Electronics

This consists of a passive transponder, an active radio beacon, and a strobe light on a timer. The radio beacons are activated upon departure from the cruise stage. The strobe is activated shortly before impact. The batteries are charged from the cruise stage just before disconnection.

A pressure sensor exists in the atmospheric and the soil sample sections. This alerts the cruise stage if a leak occurs. This may result in rotating the vehicle relative to the solar UV light to allay fears of back contamination. In an extreme case, the leak may result in diverting the capsule away from Earth and avoiding the issue.

Operations: Loading the Capsule

The entire sample capsule cylinder is kept in the sample loading bay prior to launch. The loading sequence is as follows:

1. During the Martian Summer, the air dust sample section is sealed.
2. The air sample section, after pressurization, is also sealed during the Martian Winter. This gives a sample of the atmosphere in opposite seasons for comparison.
3. The capsule is unsealed from its sterilization bag by use of a draw-string. This is pulled away from two directions (either will work) in a manner similar to the arm on the Phoenix lander.
4. The rock sample bag is loaded at the bottom of the sample cylinder, followed by the tape samples and the core sample.
5. The cylinder is sealed with a threaded cap.
6. The cylinder is checked for center of gravity in all three axes.
7. The loading manipulator both adjusts the balance disk in the Crumple Zone to offset the CG within the cylinder, and sets the screw-in fitting to be in the exact center of the loading bay for mating with the sample cylinder.

8. The sample cylinder is moved into the capsule bay by the manipulator and screwed the appropriate number of turns into the move the CG forward to the appropriate location.
9. The cylinder is capped with an additional crumple zone/insulation section and the outer skin door is closed and locked. The insulation is also locked into place within the low density crumple zone with a soft shallow dovetail joint.
10. The Cruise Stage, which has been elevated 15 cm after landing, can be lowered into place if appropriate.

Operations: Capsule Entry

The capsule entry sequence is as follows:

1. The cruise stage charges the capsule battery and runs final tests of the capsule electronics. The capsule may be close enough to Earth that the beacon may be detected from ground tracking.
2. The cruise stage ensures the capsule is correctly oriented and transmits final coordinates to the ground.
3. The cruise stage calculates the appropriate delay and sets a timer on the capsule to sequence the activation of the strobes on the capsule.
4. The electronic coupler between the capsule and cruise stage is severed.
5. The capsule is spun faster to ensure stability using the cruise stage RCS system.
6. A spring-loaded mechanism on the cruise stage gives final separation.
7. The capsule maintains the beacon but not the strobes until atmospheric entry.
8. When the timer completes, several km above the ground, the strobes become active. This simplifies tracking from aircraft and ground crews.
9. The capsule makes landfall.
10. A chase crew in a helicopter switches off the electronics, places the capsule in a secure container and loads it for return to the laboratory.

Planetary Protection (Earth to Mars)

Before leaving Earth, the entire capsule is sterilized by autoclave. The entire capsule section is sealed in plastic that was part of the autoclave process.

Planetary Protection (Mars to Earth)

Since the TPS section has a high probability of damage on impact, the core of the structure is filled with foam that is biologically shielded by the overall skin from and Martian contamination. Since pieces of this foam may be scattered on impact, it is best they have never touched the Martian atmosphere or dust. Filling this space will a low density foam not only gives those contaminants no where to go but adds to the shock absorption on impact.

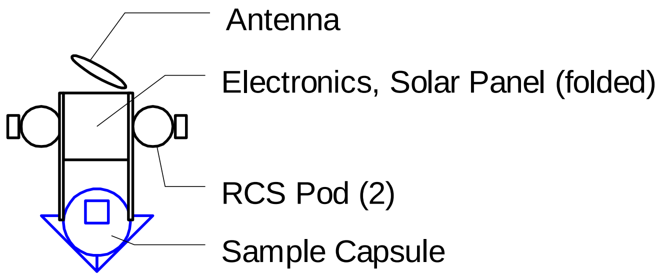
Optional Equipment: Separation and Entry Camera	It may be possible to include a rear-facing rocket camera that would be activated on separation, record video of the cruise stage departure, entry, and landing to solid state memory before recovery. This video would be an added public relations bonus but also provide visual data on the stability, plasma sheath formation, and other dynamics of the entry for use in future designs.

Earth Return Cruise Stage

Overview

The cruise stage is a fairly small RCS, navigation, and communication platform that handles the return capsule from TEI until just before Earth atmospheric entry. The general arrangement has a pair of hydrazine tanks at opposite sides of the stage with a four-engine RCS pod at the end of each tank. At 90 degrees from the tanks are two folding rectangular solar panels. The center of the vehicle contains the electronics, small high-gain antenna, and the mount for the capsule. A low gain antenna is also placed along each solar panel for communications when close to Earth and when the vehicle is facing the opposite direction. A small radiator for the electronics is contained between the capsule and the body to ensure it is in the shade in most orientations.

Layout:
Side View with capsule access exposed



Layout:
Front View (Solar Panels Deployed)

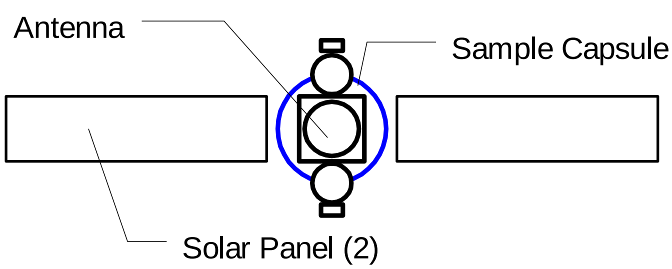
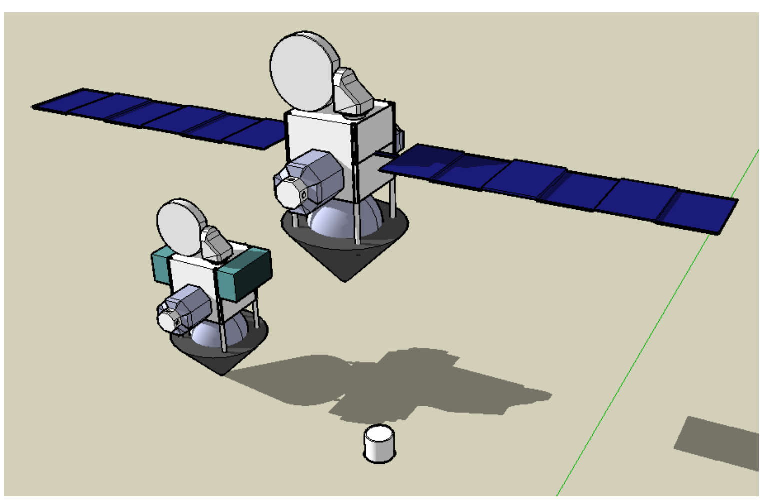


Illustration:
Wings deployed, wings folded, and sample canister



Inverted Configuration

The entire entry vehicle and cruise stage configuration is flipped upside down for the following reasons…

- It enables the antenna and solar panels to deploy at the surface or in Mars orbit.
- It completely protects the entry shield from exposure while on the surface.
- It reduces the overall height of the vehicle by 22 cm.

Dimensions

Mass	24 kg plus 4 kg hydrazine propellant
Dimensions (folded, with capsule)	80 cm (across RCS Pods) 30 cm (across capsule) 102 cm (with capsule raised for sample access) 87 cm (with capsule lowered into flight configuration)
Dimensions (deployed)	300 cm (across panels) 80 cm (across RCS Pods)
Electronics Box	30 cm cube
RCS Pod: Tank Diameter	18 cm sphere (2)
RCS Pod: Engine Cluster	10 cm across, 5 cm thick (2), 4 engines per cluster
High Gain Antenna	28 cm (same as MER [7])
Solar Array Dimensions	30 cm wide, 120 cm long each
Solar Array Output	154 watts at Earth 77 watts at Mars
Battery	10 amp hour Li-Ion (MER)

General Tech Level

The smallest cruise stage launched has been the Mars Polar Lander cruise ring, which at 56 kg (82 kg with the Deep Space 2 capsules and equipment) was able to power and guide a 494 kg capsule [5].

Technology for this return vehicle is fairly similar with two exceptions. First, Mars missions tend to use the payload's computers, whereas the return capsule here will have no such processing power. Secondly, the capsule itself (20 kg) is close to the mass of the cruise stage (24 kg + 4 kg propellant [5]). Therefore the structural mass, fuel, and engine sizes are substantially less massive. Note also that the communications antenna is deliberately sized to match the MER main antenna to allow the system design to be reused.

Structure

The vast majority of this structure is reinforced composite.

A four-column structure connects the sample capsule to the cruise stage. The connection between the capsule and the cruise stage can be raised or lowered 15 cm to allow the sample to be loaded by the lander. It is lowered during transit to Mars, raised to allow the sample to be loaded, and lowered again for the return flight. This mechanism can also be used to make minor changes to the center of gravity before takeoff for proper launch. It consists of four columns set on motorized floating screws, with a flexible sheath around the assembly to keep dust out during the surface stay. If one screw jams, the other three can break a failsafe and force the other shaft to move. If this mechanism is judged too complex, it could simply be moved pneumatically during the loading operation and locked down mechanically before takeoff. The columns are supported laterally by a mounting ring, and this ring also connects the cruise stage to the TEI stage. The cruise stage is the section raised and lowered, with the mounting ring and lander kept in place.

Navigation A star and sun tracker arrangement is used, much as they are for journeys to Mars. Navigation information based on ground tracking can also be uplinked to the computer on board, especially as the vehicle approaches Earth.

Note that the computer onboard the cruise stage is also in charge of guidance of the entire vehicle stack during outbound cruise, Mars descent, Mars ascent, and Trans Earth Injection. The lander computer acts as a backup during the outbound trip.

RCS System The RCS system for the cruise stage allows navigation en route to Earth, placement of the capsule in an Earth entry path, and placement of the cruise stage in either an Earth entry path for destruction or an Earth flyby path for disposal at escape velocity.

The general arrangement is a core of two spherical titanium tanks containing hydrazine and placed in opposite locations on the outside of the platform. This allows the spin of the vehicle (2 RPM) to provide propellant to twin RCS pods – one at each tank, and with four thrusters in each pod. This arrangement is very similar to the MER cruise stage [7] though far smaller and with far less centrifugal force, but with far less fuel demand. Four kg of hydrazine is held between the two tanks. Each tank has an interior diameter of 16 cm and an exterior diameter (including insulation) of 18 cm. The total theoretical delta-V provided by the RCS system is 185 meters per second, minus 1 percent for system loss.

The RCS systems on this stage, TEI stage, and the lander are nearly identical, with the TEI and lander versions providing more thrust over shorter periods.

Solar Arrays While the MER cruise stage provided 300-600 watts of power [7], it also had to maintain diagnostics and thermal management on a 185 kg rover. It also had the highest communication demand on the end of the flight with the lowest solar input with a medium gain antenna, whereas the communications demands for this vehicle when it reaches Earth are over much shorter distances and with far more solar power per square meter.

Perpendicular to the line containing the RCS tanks are the two rectangular solar panels. Both sides of both panels have solar cells, and they will be able to provide minimal power even if folded (provided they are not shaded). These deploy from opposite sides and reach beyond the shadow of the return capsule. They can even be temporarily deployed on the surface if necessary.

It the vehicle goes into safe mode, the panels are canted at a 20 degree angle relative to each other. If both panels are in a single line, and the sun is edge-on, neither panel will receive power and the probe will eventually be lost. By canting them back at an angle, this breaks up this single line and at least one side of one panel will be illuminated regardless of spacecraft orientation. The battery can then be slowly charged until it has enough power to communicate with Earth.

The solar panels are folded into four sections and measure 120 by 30 cm each when deployed. This results in 154 watts near Earth and 77 watts near Mars when the panels are properly aligned. To simplify power handling, the battery is disproportionately large and the system as a whole can run in several power modes depending on current demand. The battery is slowly charged between communications and other demanding bursts to compensate for the small panels.

Telemetry

The main antenna is 28 cm in diameter and is a direct copy of the MER antenna [7]. It is the main antenna for both the return cruise and surface operations.

During the return voyage, since the antenna sits on the axis of rotation, tracking an offset target (Earth) while rotating simply involves rotating the dish on one degree of freedom. The antenna is probably equipped with a radome to minimize transonic stress on ascent and to minimize dust contamination on the surface.

Planetary Protection (Mars to Earth)

After separation from the capsule, and assuming there is insufficient fuel to miss Earth, the cruise stage uses its remaining fuel to place itself in a tumble to ensure the entire structure is directly exposed to and consumed by atmospheric entry plasma. The lightweight structure should be consumed completely, and the titanium tanks outer surface insulation will burn away even if the tanks are empty and do not explode. The cruise stage interior is sealed against dust contamination and therefore will not harbor any potential Martian organisms.

Trans-Earth Injection (TEI) Stage (Second Stage)

Overview

Since the stages are nested, the RCS pods are close to the center of gravity for the two stage configuration and steer from the rear like fins for the TEI-only configuration. Both the TEI and Ascent stages are built with two identical tank enclosures with two hemispheric tanks (one for liquid oxygen and the other for liquid ethylene) in each enclosure. This ensures even mass distribution at the expense of slightly more plumbing. The inner tanks are aluminum (to simplify and cheapen the machining process) reinforced with a jacket of Kevlar or a superior material, and finally insulated with Mylar.

Diagram (skin and structure removed for clarity)

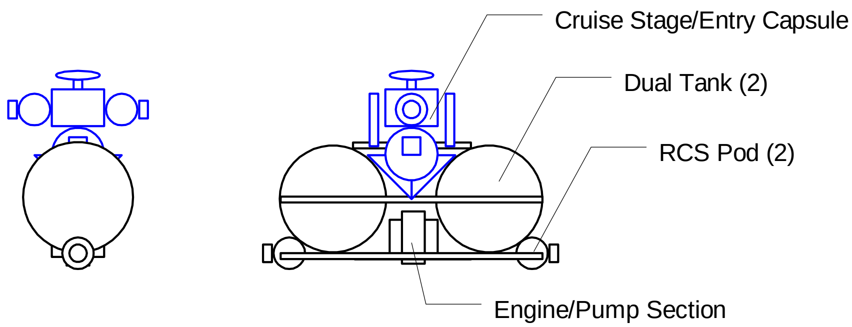
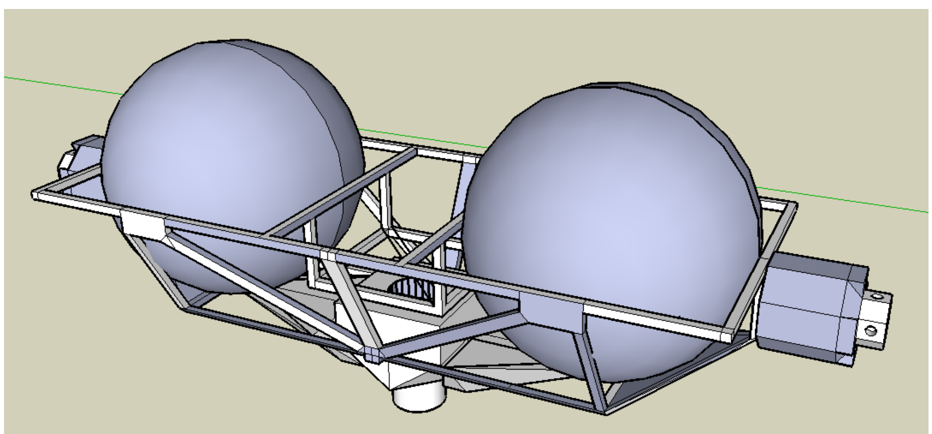


Illustration of TEI Stage



General Tech Level

The dual tank enclosure method (fuel and oxidizer separated by a bulkhead and insulated in a common space) is often used in staging liquid fueled rockets.

New engines must be developed for this vehicle that operate at (comparatively) low thrust and run on ethylene and liquid oxygen. Since the ISP, fuel mixture, and operating temperature of this system is very close to that of a methane engine, of which there are roughly a dozen larger designs, this should not be very difficult. Further, these engines may be reused as RCS for future crewed vehicles.

Structure

A formed composite structure keeps both tanks in place and forms the engine mount and inter-stage mounts for the cruise stage and the ascent stage. The overall vehicle stage is covered with a thin fabric dust and sun shield. Titanium structural tubing and aluminum fittings embedded in the composite may be used as practical for some elements.

This design involves poor aerodynamics and high induced atmospheric stress loads through the transonic phase of ascent. Currently, it is assumed that a future iteration would evaluate whether it would be wise to deploy an aerospike ahead of the vehicle, or design an enclosure for the antenna of the cruise stage that would serve this role. The purpose of this would be to allow the tanks and tubular frame components to ascend in the aerodynamic shadow of this aerospike and therefore not have to deal with these stresses directly. It would also improve the overall aerodynamics and reduce the delta-V penalty of the wide cross section of this design. At this point, it is assumed that either a telescoping or swing-arm mechanism will deploy an aerospike in front of the vehicle before ascent. A future iteration would show the mass of this mechanism as part of the first or second stage, and it would be placed in the most practical way.

Fuel/ Oxygen Tanks

Each tank is split into a lower section containing liquid oxygen and an upper section containing liquid ethylene. (There is 2.6 times as much oxygen as ethylene.) This results in a larger common tank, since the central bulkhead must be figured into the size (1 cm thick across the equator for our design estimate). In practical terms the bulkhead would be curved to allow proper fuel drainage and combined with slosh baffles, but would also be at a higher “latitude” in the tank and therefore of a smaller diameter, so this displacement is a good estimate. Each tank has a fill at the top and a drain at the base. It can be cycled through refrigeration equipment if appropriate.

Engine

The engine fits in a space 13 cm wide by 30 cm long, and the engine hardware (pumps, gimbals, ignition, valves, etc.) fits into a space 20 cm tall by 27.5 cm wide surrounding the engine space. There is room to expand these envelopes if needed. There are two pumps – one for oxygen that draws from both tanks, and one for fuel that also draws from both tanks. Each pump weighs 500 grams.

Fueling Mechanisms

The ISPP system pre-liquefies the oxygen and ethylene for each Sol in holding tanks. Since compression induces heat, the contents of these tanks are allowed to cool overnight before being pumped to the appropriate stages. The ascent stage will probably have leftover fuel from the landing, so it will be fueled first since it already has refrigeration requirements at this point. If the tank is actively refrigerated by recirculation, or stirred in this manner, it can be drawn from the bottom and refilled at the top using this same plumbing.

Some of the plumbing is built structurally into the framework surrounding the tanks and is part of the stage. This allows the lander to connect to the ascent and TEI stage with only two connections, reducing the fueling system's complexity and easing disconnection before flight. Where possible, the fueling pipes on the stage perform a structural role to minimize wasted mass.

RCS

A common RCS system for both the Ascent and TEI stage, but attached to the TEI, includes 2 pods of 7 kg each, and of a very similar design (same 2 kg capacity tanks, etc) as the pods on the cruise stage. These increase the relative mass of the TEI stage, and thus the entire vehicle is scaled up accordingly. This seems a safer option than assuming a gimbaled engine mount and possibly the RCS pods on the Cruise stage can keep the vehicle on course. Secondly, the RCS system will allow separation from the cruise stage and a safe distancing maneuver to prevent the TEI stage from reaching Earth. Third, the RCS system is used to provide ullage trust to force propellant to the bottom of the tanks prior to firing the main engine in microgravity.

Guidance

Guidance for most phases of the mission is provided by the cruise stage. There is one exception which will require a rudimentary mechanism. After cruise stage separation, the TEI stage must back away, orient itself, then fire the remainder of its propellant (in both the main tanks and RCS system) to ensure it will not collide with Earth. A tiny inertial guidance system (roughly as sophisticated as one used on many cars to compliment the GPS) can be used to run a short routine coded by the cruise stage prior to separation.

Power

The stage is powered by a battery charged before ascent and having a lifespan of at least three orbits. It may be charged by the solar collectors on the cruise stage if an issue prevents the vehicle from leaving Mars orbit on schedule.

Operations: Pre-Flight and Flight

1. Pre-launch weather and other monitoring operations begin before the return launch window to allow for possible delays either at the surface or in orbit.
2. The rover, if still active, is moved to a good viewing location. The rover is programmed to visually follow the motion of the ascent vehicle (dark spot) and/or the exhaust plume (bright spot) across the sky in video mode and slowly transmit this video back to Earth later. The manipulator arm, after a final inspection of the return vehicle, is fully extended away from the lander and does the same thing.
3. Batteries for electrical steering and guidance are fully charged from the lander.
4. Memory wires in the solar array flatten it to the surface to minimize damage from the exhaust plume.

5. A final check for winds aloft is made using the lidar. This information is added to the ascent profile and a go/no-go decision is made. This information is relayed to Earth and the rover, along with the exact launch time. In the final two hours of countdown, the go-no go decision is made by the cruise stage computer, not Earth.
6. Fueling valves are closed and pipes are swung away. Dust covers connecting the ascent stage and lander are pulled back. Any vented propellants are allowed to dissipate.
7. The frame of the door shielding the main engine from dust is dropped open on one side but not the other so that the engine plume is directed away from the solar array and across the surface in the opposite direction.
8. Engine gimbals are tested on both stages.
9. Pumps run fuel and oxygen to the ascent engine and it is ignited. On confirmation that the engine is firing, explosive bolts free the ascent stage from the lander and flight begins.
10. After launch, the inertial guidance system used for the separation maneuver supplements the guidance system on the cruise stage. This is used as a pre-test for the mechanism prior to separation and as a temporary backup in case the main one fails.
11. After the ascent stage is expended, explosive bolts separate the two stages. The vehicle is now in a 500 km by 25 km path.
12. The TEI stage RCS system performs a separation maneuver to prevent a subsequent collision. The ascent stage may fire small solid retrorockets to add distance.
13. At the appropriate time, the TEI stage fires to circularize the orbit at 500 km.
14. The ascent stage enters the atmosphere near the launch side, allowing for differences in latitude, rotation of the planet, and atmospheric drag. These should be calculated before launch to allow for possible photography of the reentry from the ground or orbiting spacecraft. The debris field should be estimated.
15. After a brief checkout and orientation using the celestial navigation equipment on the cruise stage, the launch window is determined and the vehicle oriented accordingly.
16. Diagnostics are run on the entire system as time allows.
17. The cruise stage transfers vehicle status information to Earth via a direct link, surface probe, or passing orbiter. The orbiter-to-orbiter link would be tricky since the MSR would be in a low equatorial orbit and any science satellites would be in a low polar orbit.
18. When the launch window is reached, the vehicle enters final countdown. Several seconds before launch, the RCS pods fire a ullage burst to settle the tanks. At the end of this burst the main

engine is fired. Control is provided by the RCS pods and, if absolutely necessary, supplemented by the cruise stage RCS pods.

19. After cut-off, the celestial navigation is rechecked. The TEI main engine may be reignited to make minor corrections with any remaining propellant. Note that the path is currently a near-miss of earth to ensure planetary protection should the system fail at this point.
20. The cruise stage calculates the appropriate maneuvers for the TEI stage and downloads them to the TEI stage control unit.
21. The cruise stage separates. Both the cruise stage and the TEI stage perform a separation maneuver.
22. Following the program left by the cruise stage, the TEI stage orients itself and fires its remaining propellant to ensure it will miss Earth by a greater margin.
23. The cruise stage uses its star tracker or other imaging camera to confirm the TEI stage is following the appropriate course. It then transmits this information to Earth as part of the post-maneuver telemetry.
24. The cruise stage performs a small targeting maneuver to ensure it hits Earth at the appropriate area.

Contingencies

If there is an issue after reaching orbit, the cruise stage can deploy the solar panels and command the complex to communicate with Earth directly, via orbiter-to-orbiter relay, or from orbit to any then-operational surface probes for relay. It can remain in this state indefinitely and may use the remainder of the launch window to communicate with Earth until a solution is found.

If a solution is not found and it is still able to maneuver, other options include leaving the capsule in a higher orbit (or even on Phobos) for eventual retrieval by a less expensive mission. If the mission is a total loss, the tanks should be vented to minimize the hazard of an explosion and subsequent space junk.

Ascent (First) Stage

Overview

The dry mass of this stage was initially increased from Iteration 1 to 2 from 88 kg to 132 kg to deal with the added plumbing and bulkheads for the dual-dual tank solution. It was then increased again for Iteration 3B to 167 kg for added propellant capacity and trust. The overall design is nearly identical to the TMI stage, but larger and without a separate RCS system.

Layout

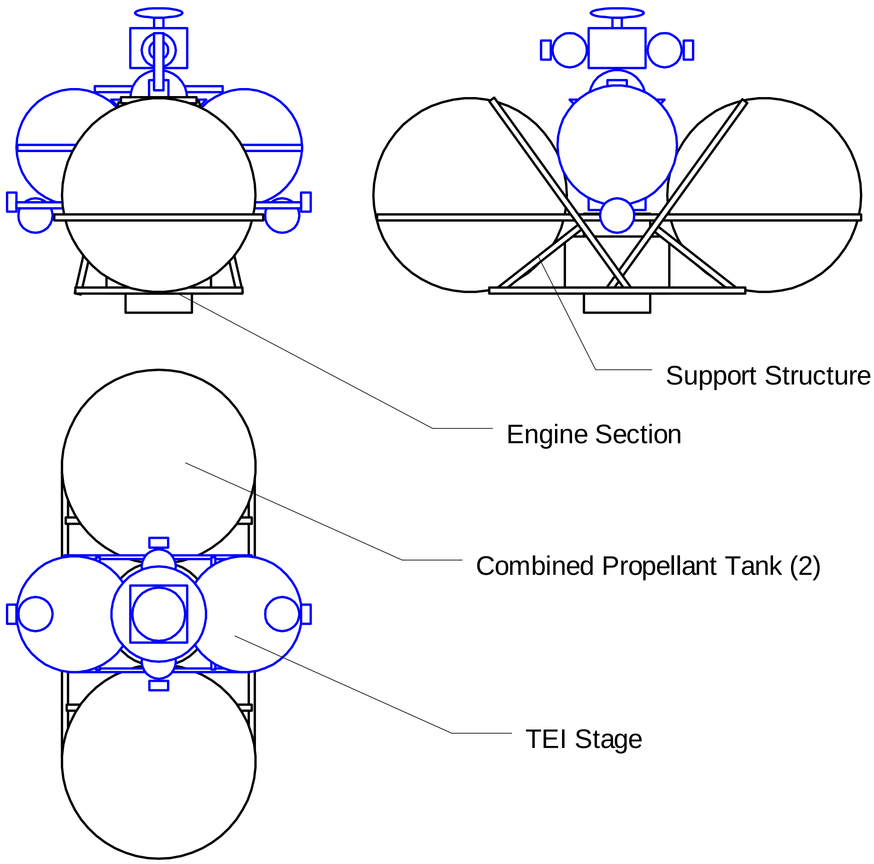


Illustration of Return Vehicle

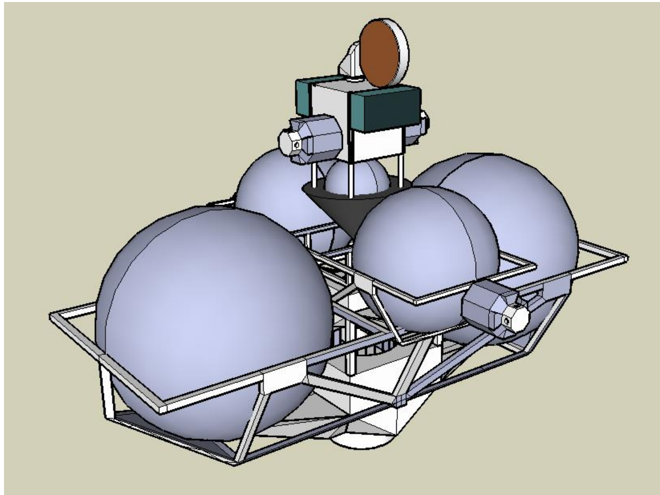
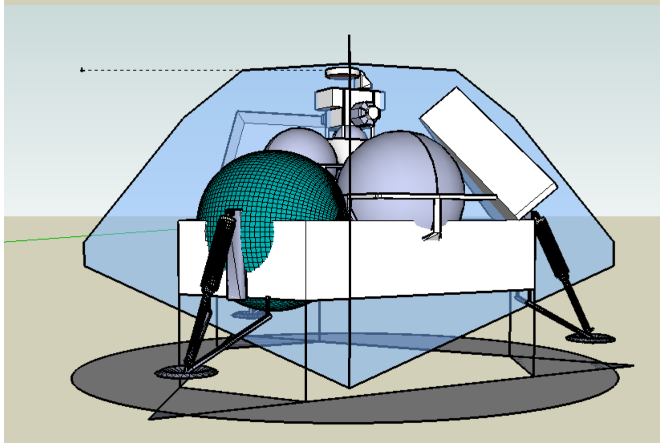
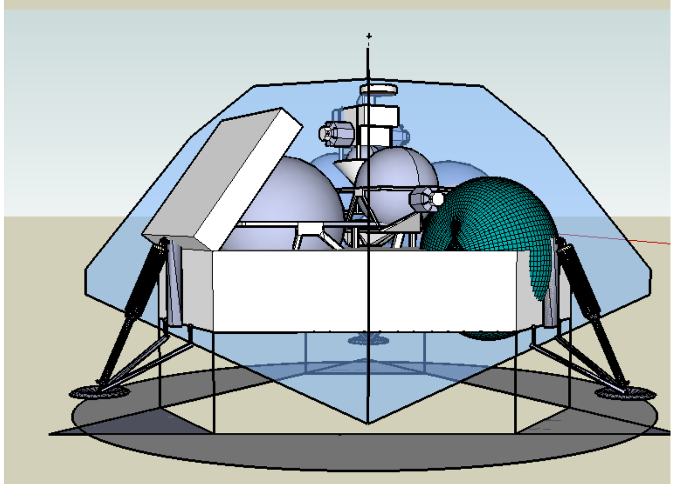
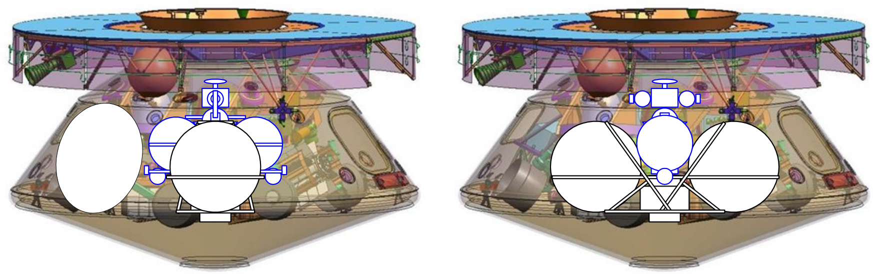


Illustration of vehicle in aeroshell allowance: Alternate configuration (Gear not folded – illustration intended for exhibit display)





Fit to Aeroshell and H2 Tank (Lander Removed) [9 – courtesy: NASA]



The height of this vehicle is perfectly consistent with the rover that the entry vehicle is originally designed to carry. The issue becomes the hydrogen tank. Note that this hydrogen tank is a full 10 cm larger in every dimension than its contents to allow ample insulation and structure. To simplify its position, it is ovoid and angled 15 degrees. Additionally, if necessary, the TEI stage can be made narrower by pulling the main tanks, and therefore the RCS tanks, closer to the center at the expense of making the overall return vehicle slightly taller. Also, the center of gravity could simply be shifted further to the right of this picture and the trust line (and ascent vehicle) moved accordingly.

This will be discussed in more detail in the lander section.

General Tech Level

The TEI Stage and Ascent stage are basically identical in terms of engineering, apart from scale. They will share many common parts, materials, and construction equipment.

The main issue is the development of a variable thrust ethylene/oxygen engine. The output of this engine will be 1032 kgf and may be throttled down to 120 kg force. Given its relatively small size, midrange cryogenic propellants, and midrange operating temperature, these engines should be constructed without an extensive budget. Further, they may be reused in other applications, both robotic and as RCS on crewed vehicles, where mid-range ISP and mid-range storability is a key factor.

See also TEI Stage: General Tech Level.

Structure

See TEI Stage: Structure.

Fuel/Oxygen Tanks

See TEI Stage: Fuel/Oxygen Tanks for more details.

Engine

The engine is required to have 1032 kg of thrust on landing and 1012 kg of thrust on ascent. These are very close and therefore the original idea of using a single engine to do both is validated. This engine is also electrically gimbaled.

This engine, using the ratio of 15:1 thrust to weight that appears standard with larger methane engines, has an allocated mass of 57.4 kg. It fits within a design envelope that is 35 cm wide and 50 cm tall. That engine space fits within the center of an assembly of support hardware that is 55 cm wide and can be up to 40 cm tall.

There is a temptation to think that one could develop a single engine design for both the first and second stages, and simply cluster them for the first stage. However, the thrust demands of the first stage are 14 times those of the second, resulting in either a large (and therefore inefficient) throttle range or a very large (and heavy) cluster of engines for descent and ascent.

Fueling Mechanism	See TEI Stage: Fueling Mechanism.
RCS	The RCS system for the TEI stage is used for both stages. It is located low enough on the TEI stage frame as to be the main RCS system for the ascent. See TEI Stage: RCS. Another RCS system is incorporated into the landing stage and uses the same RCS pods for fine landing control and axial rotation. The main engine is also gimbaled with electric actuators to provide another dimension of control on both descent and ascent.
Guidance	Both ascent and descent is controlled by the cruise stage computer. After ascent burn-out, and lacking any RCS of its own, the ascent stage is inert. It simply fires the pyrotechnics and may also fire small solid rockets to give it distance from the TEI stage. It may also vent the tanks at this point.
Power	The only local power needed by this stage is for ignition and driving the actuators to steer the engine on ascent and descent. This power is driven by the TEI stage on ascent and the landing stage on descent.
Pre-Flight and Flight Operations	See TEI Stage: Pre-Flight and Flight Operations.
Post-Flight Disposal	The ascent stage is placed in a 500 km by 25 km ascent orbit. The TEI stage circularizes this after launch for the rest of the vehicle, leaving the ascent stage to reenter after one orbit. This both eliminates the ascent stage space junk hazard in orbit and reduces the overall mass of the vehicle by staging more efficiently.

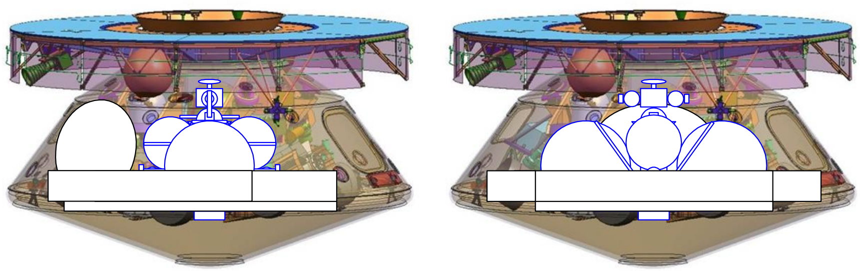
Lander Components

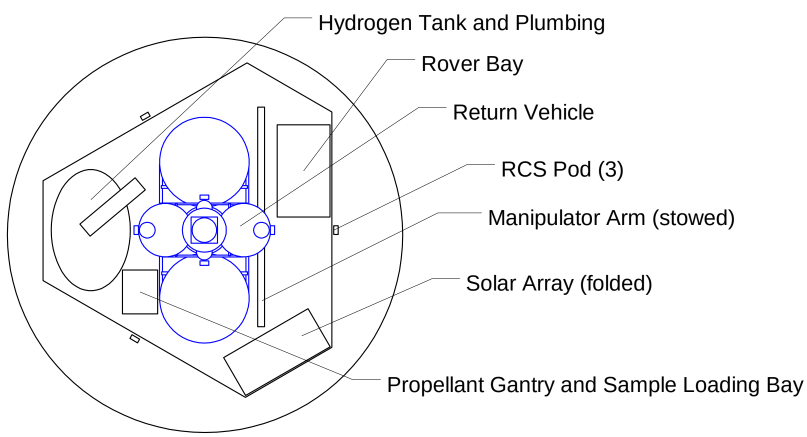
Overview

The landing stage is a Viking-like tripod with a triangular framework. It uses a ground radar and imager to pick out a landing spot during descent. With Viking, the triangle vertexes are blunted and end in landing gear, and the structure is bisected with a line containing the landing fuel tanks and RTGs – with two legs and the majority of the scientific equipment on one side of this line and the communications gear and the other leg on the other side.

The MSR design follows the same symmetry because it is an efficient shape given the form of the return vehicle and the entry capsule. The large section is for rover and solar array deployment and the small section for ISPP production and sample analysis.

Lander Configuration (Landing Gear Not Shown, Deck Equipment Not Shown in Side Views) [9]





The lander deck contains the hydrogen tank and plumbing, propellant and sample handling gantry, a solar array, a manipulator arm, the rover, and the return vehicle. To better show the return vehicle, the deck equipment has been removed from the side views above.

General Tech Level

The terminal guidance is very similar to that of MSL, and uses identical radar. While specific equipment on board is covered in the sections addressing that hardware below, the overall platform has been flown with Viking.

The manipulator arm is folded into a 2.5 meter space and extends 5.5 meters. It contains a camera, a basic claw manipulator, a small laser distance measuring array, and a pneumatic nozzle that allows it, via the atmospheric compression and filtration equipment in the ISPP system, to direct a high-pressure stream of “air”. This allows it to clear dust from not only its own solar panels but those of the rover when it returns from each sortie with samples. Further, it can use equipment on the deck to take shallow core and rock samples as a contingency sample in the event the rover fails, and to anchor the array itself using stakes.

There is no new technology here other than EDL with a variable-thrust single engine system and minimal RCS pods. There will also be software to ensure the manipulator arm doesn’t collide with other equipment. This type of software already is in use on the ISS, but will need to be more intelligent to deal with (say) a dust devil moving a solar array unexpectedly.

EDL Overview

The aeroshell is almost completely identical to that of the MSL, apart from attachment points and related connections. The landing radius for MSL is 20 km, so all landing sites with suitable terrain within a 40 km ellipse (to be on the safe side) are pre-mapped. The vehicle has enough fuel for + or - 4.5 km cross-range capability during the powered descent phase.

Based on Mars Reconnaissance Orbiter photos, mission planners will find a series of locations within the 20 km ellipse, all within 4 km of the nearest alternate site, that have both flat enough terrain for the lander and solar array and enough scientifically interesting sites within range of the rover. These will be assigned values based on their priority. Once the vehicle drops the entry shield and orients itself, the easiest landing site to reach is determined. The system will then decide based on these values and the likelihood of reaching the alternate sites if it should try to go for a less ideal site of greater value instead. Once these things are decided, the lander will turn so that the side deploying the solar array is facing the equator and make a landing in the most level terrain it can find. The small RCS system is necessary for this axial rotation.

Guidance on landing is provided by the radar system from MSL, along with a simple laser rangefinder system that, based on the attitude of the lander and the distances reported by the lasers, determines that the landing site is relatively level before committing to final descent. A landing camera is also used for both wind drift correction (as with MER) and post-landing navigation review.

Upon landing of any one leg, the engine shuts down and is allowed to cool briefly. A door then closes over the engine to prevent dust contamination.

Landing Gear	The major difference with Viking (other than size and capacity) is that in order to level the platform, the legs are actuated to bring the craft level and lock it that way before fuel production begins. Viking (and the Apollo LEM) used crushable honeycomb aluminum in its landing struts to provide cushion. MSR uses one of two systems. Either the struts can be a conventional gas strut that has the pressure levels adjusted upon landing, or it can use a crushable system as before, but then have a wormgear system engage after landing to level it and mechanically lock it in place. The level of the platform is periodically rechecked and adjusted as the mass increases.
Engine Operations	One key issue with a reusable engine for landing and takeoff is the issue of keeping the engine dust-free for over 500 days. To seal the gap, a single panel similar to a roll-top desk slides into place and is magnetically sealed on all four sides. During the slide, electromagnets counter this force and allow it to float, but once they are turned off, the seal is made using permanent magnet strips on all four sides. Shortly before launch, one side of the entire door frame is dropped to the ground without retracting the panel – the whole thing forms a flame deflector to divert the plume between the legs opposite the solar array. This minimizes the chance that the lander will tip or get pulled aside by the plume hitting the array, and also minimizes damage to the lander so that it may survive for an extended mission.
Communications: Direct Earth	The MER-derived high gain antenna at the top of the cruise stage, being the highest point on the lander, covers this function. While the lander can easily support an MSL high gain antenna on the deck itself, that will be considered optional for cost reasons. Since the antenna is exposed not only to surface dust but aerodynamic forces on launch, it is contained in a radome.
Communications: Orbital	Again, a MER-derived low gain antenna allows relay of images and short videos through whatever satellites are operational above Mars at that time. Allowances will need to be made for the fact that the rover and lander, in close proximity to each other, will both be using this relay system. The antenna is placed on the top of the hydrogen tank and angled slightly forward to clear the aeroshell.

Communications: Rover	A space-rated wi-fi LAN, hosted at the lander, allows rover-to-lander communications within short distances to coordinate navigation and manipulator functions. This antenna is also placed on top of the hydrogen tank. While unique at this point to space applications with the possible exception of the ISS, a wi-fi LAN of this type would also be used at a crewed lunar base and may be built using components of the same hardware. It may also communicate with microprobes added by international and/or NGO participants and deployed by the rover. While short range communications can be coordinated by the rover and lander without human intervention, longer range communications beyond the range of this system will be relayed via satellite and require more interface with Earth. It may be worth expanding the machine intelligence at that time to relay from lander to rover via satellite without human intervention, but that is outside the development scope or need for the primary mission.
Propellant Loading	As noted, there is a single gantry for fuel loading for both stages that extends to the level of the top of the ascent stage. Insulated lines run the propellant into the appropriate lines for each set of tanks, and the valves for loading are within the structure of the ascent stage itself. The actual fill points are separated to avoid cross contamination when detached. As with an airlock, the connections are closed on both sides, the gap between is slowly vented with a third valve, and then the connections are unlinked. While this venting takes place earlier in the countdown, the oxygen line is broken fairly late in the event the launch is scrubbed and the tanks need to be topped off.
RCS	Three small RCS units are placed on the landing stage to assist the gimbaled engine in the event a drift event pushes it close to the limits of control. The RCS units mostly are focused on the ability to rotate the craft so that the landing solar array is on the equator-facing side of the craft on touchdown. That said, they can also help with last minute lateral drift and other fine tuning of the landing path at hover, much like the nearly-imperceptible RCS jets at the nose, tail, and wingtips of a Harrier jump-jet.
Guidance	Along with the avionics package on the lander is a computer system that coordinates landing as a back up for the cruise stage. It also coordinates surface operations, for which the cruise stage computer is a backup. See also EDL Overview, above.
Cameras	Cameras on the lander are included at the following locations: • Manipulator arm – Simple color imager similar to a

conventional digital camera. Can be steered independently of the end effector and be raised 6 meters above the surface for panoramas.

- Underdeck for landing – MSL-derived descent imager. Optionally could be mounted to the ascent stage base and used to document the ascent as well – with some possible navigation input to the guidance system (drift, horizon, etc.)
- Sample handling bay – simple stereoscopic black and white imagers for automated guidance of manipulators.
- Hydrogen tank top platform – MSL mast camera used for viewing landscape in high resolution around the landing site from 3 meters above the surface.
- Lander fore-deck – covers blind spot for deployment of rover and solar array prior to manipulator arm deployment – simple color stereoscopic.

The Manipulator System

The 5.5 m manipulator arm has a number of functions.

- Proper deployment of the solar array
- Cleaning of dust from the lander solar array
- Cleaning of dust from the rover solar array
- Contingency core sampling
- Contingency sample loading if the sample bay does not work.
- Very minor (push-pull) repairs or adjustments on equipment.
- Inspection of lander, rover, and ascent vehicle equipment.
- Deployment of tools on the surface
- Visual inspection of rocks and soil on the surface.

The arm can compliment or replace the function of the rover and the sample handling bay. It can use a small tool to grab samples from the rover and load them in a second contingency sample canister, then load the canister into the entry vehicle. While this loading mechanism is cruder and less likely to succeed, it is a back-up system in case the robotic sample handling bay fails.

The arm is deployed after the rover platform is tipped forward. However, until the solar array is also deployed, it can only be swept from its base 30 degrees and angled upward (unless it is deploying the solar array). However, this is the ideal position to inspect the clearance below the rover and solar array deployment platforms.
One key element is the compressed “air” wand. This is a tube alongside the top of the arm and hinged with compression joints to prevent compression of the line from pneumatically forcing the arm into a straighter position than intended. It ends in a small steerable jet in the wrist of the manipulator claw. This jet can be rotated horizontally relative to the mount, but is fixed vertically at 45 degrees downward because the claw itself can be angled as needed. The

source of this compressed filtered atmosphere is the preprocessing unit of the ISPP system. After pulling a quantity of the atmosphere into a holding tank, that tank can either be directed to the ISPP system or to this pneumatic cleaning system.

The cleaning system will also go over the return vehicle components before launch, and is positioned so that it can do so over most of the return vehicle.

Lander Skin and Dust Protection

The lander skin is a thin composite in most locations across an underlying structure of composite and aluminum. Many sections are kept at slight overpressure in order to keep dust out. The overpressure mechanism uses the same pneumatic cleaning system tank and a pressure gauge that releases some pressure into the vehicle when the outside and inside pressure come to within 20 millibars of each other. The skin is sealed to the point that these events should take place no more than once per day.

Launch Pad Components

Overview

This section will explain any aspects of the launch not already explained as part of the lander or return vehicle.

General Tech Level

Returns from the lunar surface have been launched with crews in Apollo and robotically in the Russian Lunar sample return missions. The Russian missions used some very simple yet clever ways to get the payloads back to earth with minimal guidance. They landed in the one place on the moon where a straight vertical take-off would allow the moon to leave it behind as it proceeded in its orbit and let it fall to Earth [10]. We have neither the luxury of a pilot or orbital physics to aid in this return mission.

That said, fueling, sample loading, inspection, and weather clearance are well understood from an Earth prospective and can be comprehended from a Martian one. The major issue seems to be “don’t launch in a dust storm or with high winds”, although high winds from a Martian prospective, especially aloft, would be difficult to determine. It may be necessary to launch at dawn simply to minimize thermals and dust devils.

Fuel Gantry

This is discussed in the Lander section.

Base

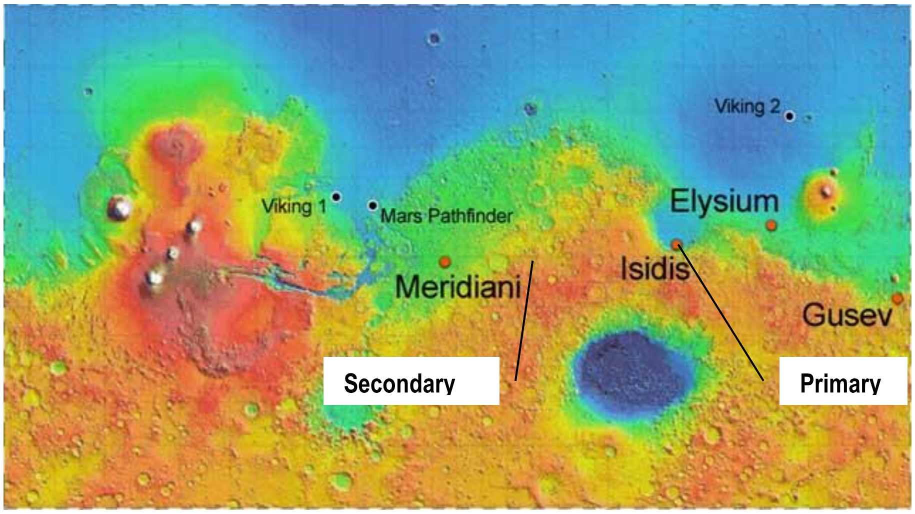
The base contains the structure of the lander that supports the partially fueled vehicle and engine during landing and the fully fueled vehicle just prior to ascent. It has the odd distinction of

	hanging the entire lander from the ascent vehicle and engine structure during landing, then being loaded in the opposite direction from landing to return. This combination leads me to believe that it will use a series of bolts that are removed explosively just before take-off – in the same manner the bolts that hold the shuttle SRBs to the launch pad before take-off are secured. This solution is very robust and repeatedly proven to work under far more stressful conditions.
	The base also has a dust protection jacket that extends from the magnetic doors at the base to the ascent vehicle stage. The interface between the skin of the lander and the skin of the ascent vehicle will have a very long connection edge. Therefore a simple adhesive, when spread over a large surface area like that, may be prohibitive to launch by its collective strength. The shield should be a very thin plastic and be pulled away with a zip thread from several locations just prior to launch but before the launch commit, since failure to remove the seal would need to result in a launch abort until the manipulator arm could fix the issue.
Flame Deflector	See Lander.
Engine Protection	See Lander.
Fueling System	See Lander.
Weather Monitoring – Vertical Lidar	The lander includes a vertical lidar system to monitor winds aloft using the movement of dust and clouds at altitude. A Russian version of this instrument was aboard the failed Mars Polar Lander.
	A radar system, possibly using the communications antenna, seems like a useless idea on Mars, since rain on Earth is what radar is used to detect. However, it seems logical that with so much iron in the wind-blown dust, a system using the proper frequency to match the dust grain sizes could monitor dust movements over some distance. This may be worth future experiments in this area, but should not be considered critical to this mission.
Leveling The Pad	See Lander.
Weight and Balance	As was noted in the sample return capsule section, a small metal disc is used to help balance the payload against the uncertainties of the sample mass and weight distribution. That said, strain gauges within the structure of the ascent stage itself will confer data on the balance of the vehicle after payload and fuel loading.

Pre-Launch Preparations	See TEI Stage.
Launch Operations	See TEI Stage.
Post-Launch Operations	See Ascent Stage for space operations. See Lander and Rover for surface operations.
Optional: Pre-Launch Weather Balloon	As on Earth, launching a weather balloon just prior to launch would be a logical way to assess winds aloft. Any remaining hydrogen could be used to inflate it, and it would be a good additional, practical science payload with a direct impact on launch reliability.

Landing Site and Lander Operations

Primary Landing Site Selection (Map Courtesy NASA)



As we will discuss later, there are three variants to this design based on landing location. The first is designed for an equatorial landing site with arrays that face nearly straight up. The second design uses two of these arrays and can work at middle latitudes. The third is for polar icecap sample return and has three solar arrays.

That said, the first mission should be A) equatorial and B) near an interface between the highlands of the south and the plains of the north. An equatorial launch site, as on Earth, will also give some additional rotational boost to the launch and place it in an equatorial orbit without additional fuel consumption.

During my time at MDRS, I had planned one EVA to find samples in a layer of strata about half-way up the hills in one location. I was pleased to find the valley floor littered with the very samples I came for because they had been eroded down and cleaned by weathering. The Pathfinder and Spirit landing sites were chosen for similar reasons – they would offer a variety of samples form the highlands that had eroded down to the landing site.

One of the alternate sites for MER was the Isidis landing site, shown above. This is sufficiently equatorial to provide power for ISPP, and is situated at a strong elevation contrast point between the highlands and lowlands. With this contrast in elevation, the amount of highland material that has washed in will be more distinguishable and indicative of the highlands than at the Pathfinder site, which is near the delta, or at Gusev, where the sampling variety across locations was not that pronounced until the hills were reached.

Secondary Landing Site Location

Since the MSL capsule can land at higher elevations than any prior mission, and the MSR vehicle is lighter and therefore able to theoretically go higher yet in elevation, we should also consider a location in the highlands between Meridiani and Isidis. This zone also shows the highest concentrations of hydrogen and other underground indications of water than any other equatorial zone. This implies ground water nearer the surface than in other locations along the equator. The secondary landing zone would then be at a level, safe site between 300 degrees longitude (the southwest corner of Syrtis Major Planum) and 330 degrees (north of Terra Sabaea) and along the equator.

Astrobiology and Landing Sites

Oxidation, radiation, and ultraviolet vastly reduce the probability of finding life at the surface, but also offers an opportunity to make a "one way valve" that allows a subsurface specimen to make it to the surface but minimizes the chance that a rover will contaminate the aquifier that was the habitat for that organism. An area of high contrast in terrain (such as Isidis) could result in finding a flow similar to the gullies that have been found recently, but much older and therefore with a sterilized layer between the source of the sample and place where it is found. This also offers the possibility of fossil life or long-term stratification in cliffs along the ancient shoreline. It would seem an ancient shoreline would provide an ideal location to find fossils, since it would concentrate any oceanic or coastal life that may have existed.

With the highland site and a near-surface layer of permafrost the probability of a subsurface biological sample (in either spore or fossil form) that has been blown to the surface by a meteor impact is also magnified. Meteors, like any particle system, contain far more small members of the population than large ones. If the ice layer is closer to the surface, a greater percentage of cratering events, those resulting from shallower meteor impacts, would free permafrost samples and send them to sterile locations on the surface where a rover can safely retrieve them. Alternatively, it is possible a small crash-stage penetration explosive could accompany the lander but not slow down. The lander could then go to the resulting crater from this deliberate impact and retrieve very fresh samples.

Operation: Post-Landing Shutdown

Immediately following touchdown, the following events are scripted:

1. The propellant pumps are shut down, the valves are shut, and the engine gimbals are centered.
2. Engineering data is polled to determine the health of the vehicle.
3. A secondary solar array is deployed from the deck and begins to charge the lander and cruise stage batteries.

4. The deck cameras find the sun and determine where to point the antenna to find Earth. It then proceeds to start a panorama.
5. The high gain antenna on the cruise stage is aimed at Earth and transmits the initial engineering data.
6. Once the engine cools and the dust settles, the landing engine door is closed and sealed magnetically.
7. The panorama is transmitted to Earth.
8. Any remaining fuel in the lander hydrazine RCS tanks is slowly burned to prevent a tank rupture or other incident from causing issues later. This is done at the warmest part of the day and uses only the upward facing engines to minimize surface contamination.
9. After reviewing the panorama, Mission Control sends commands to deploy the rover platform. The rover platform tips away from the lander deck and can deploy up to three ramps.
10. The manipulator arm is deployed and tested. The manipulator arm surveys the area below the rover and solar array deployment platforms.
11. The MER-type rover unfolds itself as was done by the MER, except on this platform and a meter above the ground. It can then roll off whichever of the three ramps is safest, or the entire platform can be tipped down to become a ramp.
12. The rover goes to a safe distance to do an inspection of the lander.
13. The solar power platform is tipped away from the vertical storage position and is hung over the landing strut that faces the equator.
14. The solar array is extended in sections using the manipulator arm to assist if needed. This is discussed in detail in the solar array section. The manipulator may anchor some or all of the solar array end-points to the ground.
15. If propellant remains in the ascent stage fuel tanks, active cooling can be engaged at this point.
16. The ISPP plant is tested and brought online.
17. After initial deployment, the rover grabs a contingency sample and returns it to the sample loading gantry area. If the rover does not function properly, the manipulator arm uses a series of core tubes from the landing deck toolbox. These tubes are designed to fit together in sections within the sample capsule. It then uses a pneumatic device to drive them into the ground within reach of the arm and retrieves them with the claw.
18. Before the rover goes further, the lander’s manipulator arm extends to its full length (now 5 meters above the surface) and does a panorama to give a better sense of what locations are worth scouting for samples.

Operation: Standard Lander Operations Phase	During the 500 days of propellant production, the following events take place. 1. The rover returns every 50-100 days with additional samples which it takes to the sample gantry. 2. During these returns, the manipulator arm uses the air jet to clean the rover's solar arrays. It also cleans the lander solar arrays systematically as needed. 3. The deck cameras and a wind sensor at the end of the solar array look for dust devils each day. If one is seen coming towards the lander, the array can be flattened using memory wire coils to minimize the chance that the wind will get under it and do damage.
Operation: Post Launch	If the lander survives, it is in the odd position of having a huge power source and an antenna that depends on orbiters for communication with Earth. It may continue to analyze samples brought by the rover or by the manipulator arm.

Sample Loading Bay and Gantry

Overview	This is essentially a small bay with a small industrial robot, an “elevator” that carries samples from the bay to the level of the sample return entry vehicle, and a second identical robot at the top of the gantry that places the sample in the entry vehicle. If this system fails, the manipulator arm can substitute.
General Tech Level	This section uses standard industrial robotic techniques, but does it in a hostile environment. So while this is easy to prototype using standard, off-the-shelf technology, doing a field test in a dusty vacuum chamber would be necessary to qualify it. Fortunately, with the MER wheels, arm, and main antenna systems, we have practical experience with robotic actuators on Mars over long periods.
Sample Removal from Rover	Samples at the rover are stored in three locations. The first is a tape that is heat-sealed into small sections – each containing a small dirt, rock, or dime-sized core sample. The second is a series of small interconnected bags that, when the drawstring is pulled, form a disc with the openings of each bag closed and facing the center. The third is a small tray of open boxes that sits in front of the set of bags and can be dumped into an identical tray on the lander for local analysis. When the rover arrives at the lander Sample Loading Bay, the bay uses a camera and a small manipulator arm with a split-cylindrical grappler at the end and retrieves the sample roll from the rover by sliding between the roll and the spindle, expanding to grab the roll, and withdrawing. It can also grab the canister end cap and sample canister itself. When the sample bags are retrieved, they are picked up and stored the same way. The sample tape is transferred to a storage spindle that uses a spring tensioner to keep the tape tight. If the roll is less than 8 cm in diameter when loaded, another may be spliced on in the sample handling bay, and the robot manipulator may spin the two shorter sections together in the manner of a reel-to-reel tape. For samples that are to be examined at the lander, a small tray of boxes is set along the rover sample collection hardware. This can be tipped into a tray handled by the sample bay robot, which will take it to the local sample analysis package for analysis. Alternatively, the whole tray could be taken by the arm and replaced with a new one.
The Sample Canisters	The lander carries several sample return capsules, one of which will be actually returned to Earth. One of these is kept in the sample loading gantry and is used for the dust filter and atmospheric sample

collection described earlier. Another is stored on the lander deck within the manipulator arm's toolbox. This is for a series of core samplers that can be used by the manipulator arm itself even if the rover fails completely. This is described under Other Operations – Contingency Sample below. One core sample, though, will be returned regardless.

Loading Samples into the Canister

The arm loads the bag and tape samples by using the ability to grab these sample collections by the hollow core of each, and placing them in the canister. Once loaded, there is still an empty core section in this canister. The lander can then take a cylindrical core contingency sample from the manipulator arm's collection and slide it into the middle of this section.

Nominally, the canister in the loading bay is used and the same arm that gathered the samples from the rover now places them in this canister. The also grabs the sample canister lid and attaches it to the canister, then grabs the entire canister.

The arm spins the sample canister briefly to settle the contents. It then places it on a scale with two cradle arms and a flat plate. The sample is placed on the flat plate first to be weighed and its balance is determined by a series of four sensors under the plate that measure the difference in pressure. It is then placed sideways across the two arms, first on one side and then on the front, to have the weight and balance measured in all three axes.

Moving the Canister to the Entry Vehicle

At this point, the arm loads the canister onto the "elevator" chain mechanism, which moves the canister to the gantry. At the top, a second, nearly identical arm first adjusts the balance plate and then screws the sample canister into that plate. It then closes the hatch and closes camlock mechanisms (similar to those on a light aircraft engine access door) that latch this lid.

Other Operations-Contingency Sample

Another sample canister is stored on the lander deck within the manipulator arm's toolbox. This contains a central core sampler and a series of six radial core sampler sections that can be wrapped around the cylindrical one to form a contingency sample set. This set can be gathered using the manipulator arm and a pneumatic driver tool to place the core samples. The tool also has a ratcheting system to extract the core tubes from the ground once the sample is collected.

The main manipulator arm can either reach the sample collection bay to hand this sample to the sample collection robot, or it can if necessary load the capsule into the return entry vehicle by itself (albeit without balancing the sample or the mounting plate).

Other Operations-Rover Maintenance	The inspection cameras both on the main manipulator arm and the sample collection area will inspect the rover. The sample arm could theoretically assist in minor repairs (dislodging a rock from a collector, etc.) if needed.
Planetary Protection	All elements in this area are sterilized before launch using an autoclave. They are sealed until landing via a removable plastic bag. This is standard procedure for medical items and pharmaceuticals. Removing a plastic sheet covering from a launch vehicle was done with the nose cone of Mariner 4 just as the engines were fired. It will also be done to the sample collection arm of Mars Phoenix Lander just before it is deployed on the surface.

Lander Science Gear

Overview	In addition to the cameras already sited, the lander performs some analysis of the returned samples. The Sample Analysis package is located in the sample loading bay and will perform experiments that are A) useful in determining if a site is worth revisiting, and B) will have implications for sample handling back on Earth.
General Tech Level	Every scientific instrument listed here is an exact duplicate of an existing item or one that is planned for a mission that will pre-date the MSR.
Atmospheric Chemistry and ISPP	The In Situ Propellant Production (ISPP) system must, by nature, do some chemical analysis to ensure the propellant produced matches standards. This data will give details on the pressure and composition of the atmosphere in Summer versus Winter.
Sample Analysis Package	The rover carries a tray specifically for lander analysis. This tray is transferred to the bay for analysis by the following instruments: The Urey Mars Organic and Oxidant Detector is being developed by NASA for use on the European ExoMars rover [11]. This would be an exact copy. The purpose of this instrument is to analyze concentrations of organics as well as the “handedness” to determine if they are based on life forms or sufficiently random to be non-biologically created. It can also evaluate oxidation of various materials in the Mars environment. The Mössbauer Spectrometer is a duplicate of that on the MER rover [7] – since this instrument was removed from the sample collection rover for this mission to make room on the collection hand, the same experiments are done in the sample bay instead.

LIDAR	This is a duplicate of the LIDAR contributed by Russia to the Mars Polar Lander [5], but with the ability to track winds aloft if possible. This is highly desirable to track high altitude winds before the launch of the return vehicle.

ISPP System

Overview	This system is based on the ISPP system designed by Dr. Robert Zubrin [3] for creation of ethylene and oxygen. The design is targeted for a 500 day run, but is overbuilt so that shorter runs are possible. It should also allow higher per-sol output during periods of higher power input, especially earlier in the mission.
General Tech Level	The system is an aerospace grade version of the prototype with better thermal insulators and lighter sensors. While this is a typical aerospace situation where what you pay for a valve or sensor is as much as one is willing to pay for a valve or sensor, the relative simplicity of the system should result in the criteria for construction being less precise to minimize the cost for this exactitude.
Advanced System Integration	While this system is based on Dr. Zubrin's original designs and workbench prototypes, it has been enhanced to deal with a broader range of end-to-end integration with regard to thermal and electrical efficiency. It is designed to turn the liability of running cryogenics into a hot chemical reactor while maintaining operating temperature, cooling the resulting propellant, dealing with trace gasses in the Martian atmosphere, dealing with limited and cyclical power, and dealing with the thermal environment of the Martian day-night cycle. These liabilities are balanced against each other to make the system more efficient.
Power Supply and Demand	The main solar array is 33.13 square meters and has a peak output of 24,849 watt-hours per sol. Even if never mechanically cleared of dust, it would still have an output of 17,571 watt-hours per sol after 500 days. The array has an active pneumatic cleaning system, so output should be closer to peak most of the time. The demand of the ISPP system generating 2.42 kg of propellant is 16,700 watt-hours per sol. Other equipment, such as electronics and the robotic systems, still have ample power under this system. Excess power is dumped into the ISPP system to increase the production rate and expand the post-ISPP, pre-launch phase as much as possible. This allows some margin for resolution of any issues with either the ISPP or the launch, or to finish the mission if part of the solar array doesn't deploy properly.
Mass	The mass for a non-redundant system is estimated at no more than 30 kg based on the ISPP prototypes developed by Dr. Zubrin [3]. Therefore this system, with duplex redundancy and more plumbing to deal with filtration and pre-cooling, is estimated at 60 kg. This also allows the system capacity to take advantage of periods of higher electricity production by running both systems at once.

Atmospheric Intake	The main dust The dust filter is located on the side of the lander deck near the sample loading bay, and can be cleaned with the manipulator arm cleaning jet if clogged. A secondary dust filter is part of the sample return canister and is located within the bay.
Dust Filtration	After the main and secondary dust filters, an internal filter functions as backup. If the main filter is fouled and cannot be cleared, it may be ejected to allow the internal filter to become the primary.
Atmospheric Chemistry	The atmosphere of Mars is not 100 percent carbon dioxide. However, most other elements (water vapor, carbon monoxide, oxygen) are either created or harnessed by the ISPP process. Argon is inert and therefore will not interfere with operations. The only possible issue is nitrogen. The integrated design described below will distill out as many contaminants as possible both before and after ISPP operations.
Chemical Operations	The system contains a Reverse Water Gas Shift (RWGS) reactor that runs with a rich hydrogen input (3 hydrogen to 1 CO_2), which in turn creates water, carbon monoxide, and hydrogen. The hydrogen and carbon monoxide are then fed into an ethylene reactor, which in the presence of an iron Fischer Tropsch catalyst creates ethylene and water. This reaction generates a lot of heat, and the RWGS reaction requires a high temperature, so the second reaction is used to help drive the first once an electric heater brings the instrument up to proper operating temperature. The water is then converted back into hydrogen and oxygen via an electrolyser [4]. The hydrogen is recycled and the oxygen is sent to the oxygen tank. On a nominal 500 day production run, this reaction is run for 432.8 days, after which the hydrogen supply is exhausted. The remaining 67.2 days, the reverse water gas shift system recycles water with atmospheric carbon dioxide to create oxygen. This is needed to optimize the fuel to oxidizer ratio in the return vehicle.
Methanol Recycling	Experiments show that ethylene production produced 1-2 percent methanol as a secondary product [4]. The chemical reactors only run in daylight when solar power can drive operations, and must be well insulated and re-heated to operating temperature each morning. Methanol does not require refrigeration. In the morning, when the reactor must be heated again to operating temperature, the methanol is burned with oxygen from the quantity made the day before to heat the reactor to operating temperature. The result, besides heat, of this combustion is water vapor and carbon dioxide, both of which can be directed into the next

round of operations. Electrical heaters are also available for this operation, but having a double system that makes use of the heat generated (when enough methanol is available) ensures nothing is wasted. The actual operations will probably use electrical heating on days when there is insufficient methanol, and methanol when the supply is adequate.

Hydrogen Tank

The hydrogen tank is slightly lenticular and angled 15 degrees in order to properly fit in the aeroshell. This configuration gives the most volume possible while keeping the tank as spherical as possible to minimize hydrogen loss, tank mass, and volume within the aeroshell. To minimize leakage, and due to the fact that hydrogen is used very slowly from the tank, it does not have a drain at the bottom – the ISPP draws vapor from the top as it evaporates and permits this evaporative cooling to help keep the overall temperature cryogenic during surface operations.

The hydrogen allotment itself is increased 20 percent beyond that needed for ISPP to allow for boil-off en route, or 51.556 kg. To allow ample room for the tank, insulation, and any other issues, the space allocated for this tank is a full 5 cm thicker than the interior volume.

Day-Night Cycling and Fueling Operations

The production run for each sol is placed in a pair of holding tanks to cool overnight. This permits the heat from the compression operation to dissipate overnight before that same fluid is actively chilled using the refrigeration gear. It will also allow any contaminants to either settle or boil off prior to loading, forming a simple atmospheric distillation system to further refine the fuel before storage.

Daily ISPP Operation Cycles

The table below describes a typical 1 Sol cycle of operations for the integrated ISPP system.

	Time	Action
1	24/7	1. Monitor atmospheric temperature and pressure. 2. Approximate what the temperatures and pressures will be for the next Sol and make engineering settings based on this.
2	Dawn -2 hours, or temperature minimum	1. Check temperature/pressure of fuel temporary storage tank. 2. Heat or cool as necessary and let settle. 3. Vent any gases that are not part of the fuel mix by distillation. 4. Pump the refined propellant to the appropriate return vehicle tank. NOTE: it is to be determined at this stage if the atmospheric staging tank is to be stored in liquid form at 5.2 atmospheres for simplicity and to minimize compression power requirements, or in solid form to aid atmospheric distillation of nitrogen, argon, and other elements. The steps below assume the latter. 5. Check temperature/pressure of atmospheric staging tank. 6. Heat or cool as necessary to ensure all CO2 is in a solid form. 7. Vent any gases (argon, nitrogen, etc) that do not freeze out, but do not vent to the point that the dry ice converts to any other state. 8. Reseal the vents.
3	Dawn	1. Trickle charge batteries. 2. Run early morning diagnostics and any chemical engineering monitoring checks.
4	Dawn + 1 hour	Begin heating the reactors via either electric, methanol, or both.
5	When primary reactors reach op temperature	1. Heat the atmospheric staging tank to flash evaporate the dry ice at approximately the desired pressure. Use regulators to adjust. 2. Run the recently decompressed (and therefore cold) carbon dioxide through the first stage heat exchanger for the oxygen and ethylene coming out of the reactor. At first, when nothing is coming from the reactor, this is a cold-soak process. Later it will aid in condensing out water vapor, then pre-chilling the propellant. 3. Vent hydrogen through the propellant second stage heat exchanger to A) pre-heat the hydrogen to minimize the demands on the system and B) pre-chill the oxygen and ethylene to aid in liquefaction. This two-stage heat exchanger takes two demands (reactor thermal management with cryogenics being loaded and the cooling/compression demand for the resulting propellant) and works them against each other. This is not assumed to be a complete solution, but it reduces the power demands overall.

		4. Run the pre-chilled oxygen and ethylene into the staging tanks. If it is not yet condensed, run the compressors until it does so. Again, the play of compressive heating versus the chill induced by the heat exchangers is actively balanced by monitoring equipment to minimize power demands.
6	When power exceeds the demands of one ISPP unit	1. Run morning photo survey, scientific instrument checkout, and diagnostic. 2. (optional) – send a short burst of engineering and weather data directly to Earth. 3. When the science activities are complete, activate the secondary reactor if there is sufficient power to justify operations.
7	Noon	1. Run noon weather and engineering data collection activity. 2. If there is a sample for local examination in the sample bay, do any scientific work scheduled for the day at this time. 3. Confirm that the batteries can sustain this while still operating at a net positive charge for the day, anticipating the afternoon power supply and demand.
8	When sun-synchronous satellites are overhead	Send the following data: • Any requested data from the previous pass • Engineering data (what has been measured, what is planned, any errors or other diagnostic data) • Weather data (what has been measured, what is forecast) • Routine surface images (black and white survey of cameras looking at the probe itself, dust devil events, dust accumulators, any images requested the previous day, etc.) • Unscripted surface images (if dust devil detected, the film of it. Any other motion detected and recorded)
9	Mid-afternoon	1. Reduce at appropriate time from two reactors sets to one. 2. Run afternoon diagnostic and other scientific monitoring passes.
10	Late afternoon	1. Shut down the ISPP reactor and related hardware. 2. Run the atmospheric compressors to repressurize the atmospheric staging tank. This must contain enough CO_2 to run the next day. The nighttime temperature at the Spirit landing site varies from 183-223 K [21], and CO_2 forms dry ice at these temperatures when compressed to roughly 4-5 bar (depending on temperature), although this is borderline with forming a liquid at 5.1 bar [19]. 3. Use the remaining daylight to top off the batteries for overnight.
11	Sunset	Expose radiators used to aid in the final liquefaction of propellant or formation of dry ice.
12	Overnight	1. Dry ice (and any water ice) forms in the atmospheric staging tank. Other gasses (argon, nitrogen) are separated for later release. Dust is also separated later when the dry ice is allowed to flash to a gas and be vented to the ISPP system. 2. Liquid oxygen and ethylene in the propellant staging tanks is allowed to settle. Any solids precipitate to the bottom and any

		gases remaining (trace hydrogen, etc.) will later be vented back to the reactor pre-stage zone and locked there until the reactor is started. 3. Compressors may kick on at appropriate times to keep these solids, liquids and gasses are in the appropriate states.

Cleaning the Array

1	Cleaning day – the noon before	1. The arm performs an inspection of the array and transmits the photos to Earth during one of the communications passes. Any other engineering data is also uploaded. 2. The computer makes any modifications of the sweep path that will be followed by the arm and uploads that to earth for approval.
2	Cleaning: the previous late afternoon	3. The reactors are shut down early. They are configured for a 1 sol shutdown rather than an overnight shutdown (i.e., they are not kept as warm as practical overnight, but only as warm as would be necessary to minimize thermal stress.). 4. The atmospheric compression tank is fully loaded.
3	Cleaning: Morning	5. The reactors are not activated this sol. It is managed in such a way that the production will be sufficiently increased by the cleaning task that the system can make up for the lost sol of propellant production. 6. Electrical power normally used for the reactor will be used to heat the atmospheric staging tank and run the electronics under a much higher load than normal due to image processing demands.
4	Cleaning: Noon	7. The arm is positioned at the root of the array and at the beginning of the pattern. 8. The atmospheric staging tank is released to the cleaning channel plumbing. CO2 works well as a compression medium for this purpose, be it in solid or liquid form while stored. 9. The atmospheric staging tank is electrically heated far more than normally done for ISPP. The goal is not to slowly trickle refined CO2 to the reactor, but to flash the CO2 into gas at high yet manageable pressure. 10. The gas is routed through the cleaning channel hose and to the end of the arm. 11. Cameras on the end of the arm route the results to image processing software on the vehicle. Multiple cameras will monitor the operation and confirm the “color change” of the array from dusty to clean. 12. If the pressure supply becomes low before the cleaning is complete, the tank is recompressed for almost immediate re-use. 13. If the rover is in proximity and requires cleaning, it may also be cleaned at this time.

5	Post-Cleaning	14. The arm does a once-over inspection and image processing onboard gives an initial grade on the work. If necessary, it is repeated in the appropriate areas immediately, then reinspected. 15. The solar power input is compared to before the cleaning and scaled to the time of day and solar flux (as shown by the cameras) at the time. Note that the cleaning event may impact the cameras either positively or negatively, so this data must not be taken at face value. 16. Any other cleaning (cameras, sensors, etc.) is executed as appropriate. 17. The arm is returned to the resting position. 18. All engineering data, before and after pictures, problem spot images, etc. is transmitted directly to Earth to avoid any delays in return to ISPP operations. 19. The atmospheric staging tank is compressed for the next day and all other ISPP operations are returned to normal.

The Overall ISPP Production Cycle

The production cycle for ISPP, especially on a solar powered lander, is a fairly complex management task. The system is scaled so that if the power system is reduced to average power output immediately upon landing, then remained unaltered during the entire 500 days, the system would meet propellant demands. However, in reality the system will generate much more power early in the mission and less later. It will also be cleaned periodically. Also, there are two ISPP reactor sets configured together within the same thermal zones to allow the primary to help pre-heat the secondary, and to increase the thermal aggregate mass to minimize cooling overnight.

Consequently, the 500 days will be broken down into phases.

Early Mission, high power	• Primary reactor run as many hours as possible. • Secondary reactor run as many noon-time hours as possible each day. Since the compressor does not operate at the same time as the reactor, the electrical supply may maintain both reactors at once, especially since they share the thermal zone.
Mid Mission, between cleaning events	• Primary reactor is run as much as possible. • If there is sufficient power on a given sol to make it practical to activate the secondary reactor, it will be run as well. • The system is periodically shut down for one sol to allow for the manipulator arm to clean the arrays.
Late Mission, Oxygen Only	• Once almost all hydrogen is used, the system

	switches to oxygen-only production. Some hydrogen will be locked into water and recycled repeatedly in this process.
Final Production Phase	• Once sufficient oxygen is made, the reactor may or may not use the last bit of water to go back to the ethylene/oxygen reaction and top off the tanks.
After Production	• The pumps used for propellant handling may be used periodically to stir the tanks. • The hydrogen tank is vented to ambient pressure for safety reasons. • The atmospheric staging tank and manipulator arm may be used to clean off the return vehicle prior to launch. It also inspects the vehicle. • After launch, the remaining working components may be used for other engineering and scientific experiments.

Solar Array

Overview

After deploying a mini-array to power deployment operations, the main array is unfolded. This array is 29.48 square meters and has a peak output of 22,110 watts per sol. Even if never cleared of dust, it would still have an output of 15,634 watts per sol after 500 days. The array uses an active pneumatic cleaning system, so output should be closer to peak most of the time.

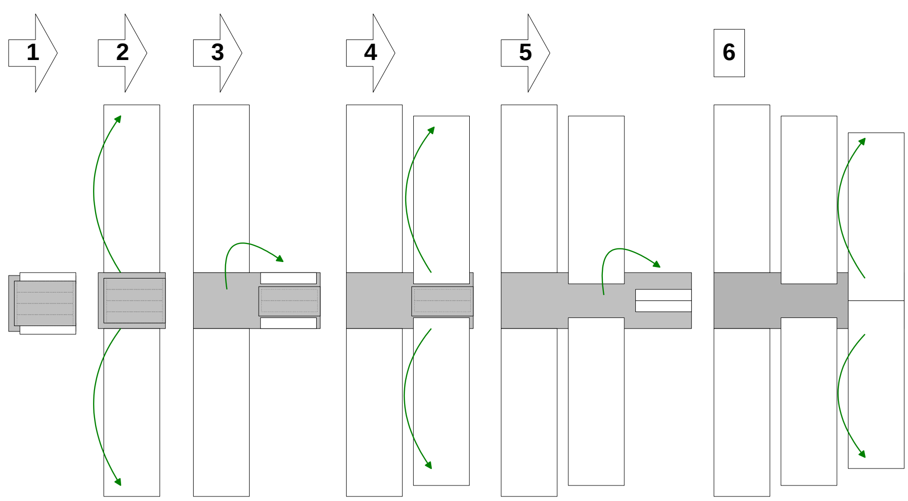
General Tech Level

Solar array output is based on the peak and average output from MER data. The array deployment is as mechanically simple as possible and allows for large rocks to be simply rolled over without interfering with the array. The design is kept so simple that a prototype with dummy solar cells could be built by a hobbyist in a weekend.

Secondary Array Deployment

A secondary fan-fold array is deployed almost immediately after landing to charge the batteries. This is of identical design to the Phoenix Lander (approx. 2 m in diameter) and is deployed from the leg opposite the main solar array near The rover bay to keep it free from other mechanical movement or from shade from other components. It is designed to track the sun and keep power running to the critical systems until the main array is online. It then provides supplemental power to the internal systems.

Primary Array Deployment Sequence



The illustration above is of a simplified six panel array – the actual design uses eight panels, but this version is easier to see in print.

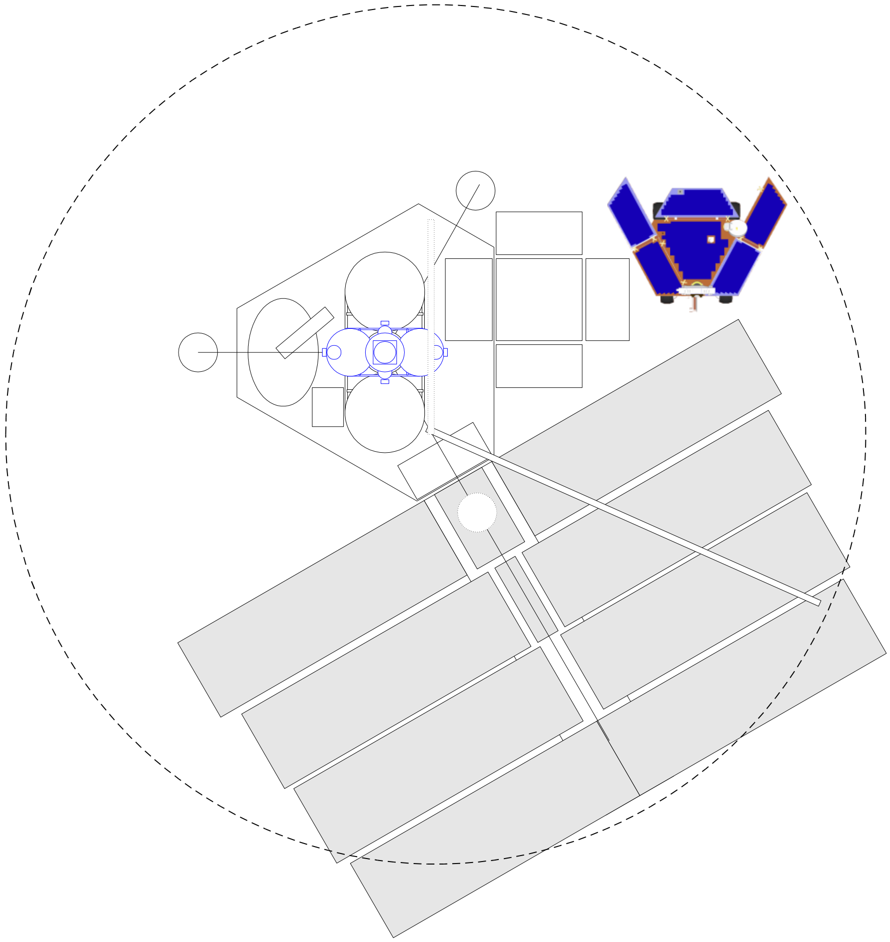
The array is a series of eight rolled-up panels wrapped in a Kaplan fabric quilt. The quilt provides the anchor point for the arrays, but also provides runs within it for electrical lines and pneumatic compression lines. It also supports two smaller inset arrays.

On the far left of this illustration, the array starts as a single square roughly a meter across. This is stacked vertically on the lander's equatorial-facing leg and tipped outward to horizontal shortly after landing. After an inspection confirms the area around the lander is suitable, the array is angled down the leg and the outermost pair of panels are inflated using a compression spine along the right side of the array as shown here (2). The array uses a fabric backing and a spring-loaded set of memory wire guides on the right side of each array to angle it to match the latitude above or below the equator. The backing contains a series of mechanical elbow links that allow the array to bend up to 190 degrees (10 degrees past flat and around rocks if needed, but still forcing the array mechanically to extend as inflated).

The lander air compressor can now continue to inflate the central spine until the next section unfolds and the side arrays for it spread in both directions the same manner (4). This continues until the array is fully inflated and extended. Once extended, no further inflation pressure is necessary. The central fabric contains two inset arrays to provide a total of 29.48 square meters.

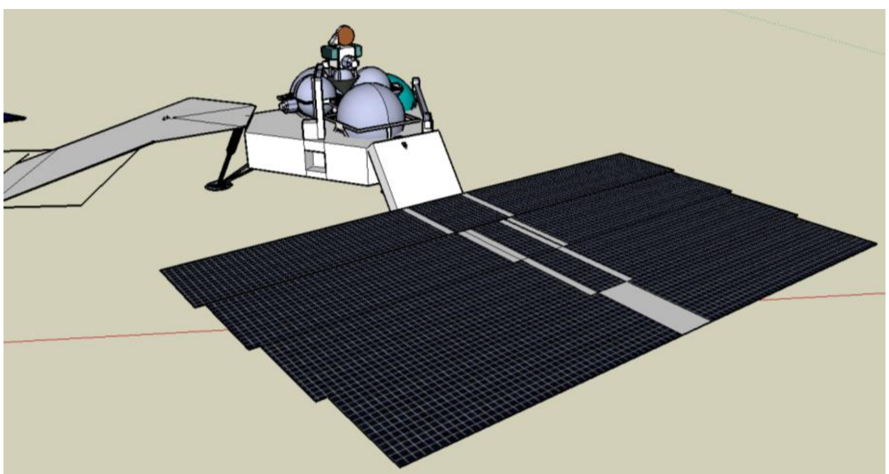
Since the array starts up the leg and the landing site is fairly equatorial, the arrays closest to the lander are still able to operate without shade from the lower arrays due to this "theater seating" arrangement. In the event the compression does not work sufficiently, the manipulator arm can deploy the array mechanically.

Array Cleaning



The manipulator arm has sufficient sweep to clean the vast majority of the array directly (dashed circle). While the rover is parked on either side of the array, it too can be cleaned without dependence on dust devils. See also Lander: Manipulator Arm.

Illustration of deployed solar array (alternate configuration)



Memory Wires to Set Array

The arrays use a memory wire series of guided “spring” supports along the lander side and a hinge at the outward side to angle the

Angle	array properly. While pre-built to a specific latitude (such as an array where all elements raise to 20 degrees), it may be adjusted by varying the voltage in the springs and ratcheting a limiting string accordingly so that the voltage in the springs does not have to be maintained the entire mission. One could even angle the arrays by lowering one side and raising the other for better efficiency with morning and evening light angles. Theoretically, the array could also fire these springs in sequence to undulate a wave through the array – this could aid in fixing a sticky deployment. It could be attempted as an experiment to see if it helps clear dust from the surface as well.
Array Dust Devil Protection	If dust storms are judged to be a risk for this deployment, the manipulator arm has a series of pneumatically drivable "tent stakes" that can be used to anchor loops at the end of each array in the same manner that the core tubes for the contingency sample are driven into the ground. If a dust storm or dust devil is detected, the array can have all the memory wires fire at once and "flatten" the array to the ground to minimize possible damage. Cameras and software on the MER spacecraft already have been reprogrammed to detect dust devils and record their passage. It should be a simple step to have regular monitoring of the area using the various cameras onboard without human interaction.
Anchors and Support	Each array ends in a series of loops that can be anchored with a "tent stake" like system from the manipulator arm. It uses a pneumatic driver system to place the stakes if needed. Multiple loops help ensure that a critical stake does not have to be placed through a rock.
Array Launch Operations	The array will be flattened at the time of launch. A dust skirt similar to that of a hovercraft could be dropped from the bottom of the lander in the event ongoing power is considered critical after launch. Primarily, though, the blast deflector on the bottom of the lander deflects as much blast as possible away from the array.

Rover

Overview	This rover is closely modeled after the Mars Exploration Rovers and will share the vast majority of components to minimize the development costs for this mission.
General Tech Level	The only issue with the MER design is that it was built several years ago, and by the time it would be rebuilt for this mission, the knowledge base and industrial tooling may be less than immediately recalled. One assumes that the staffing for that project has largely translated over to the MSL, and that – given the opportunity to build a combination vehicle; this same group would be able to do so with the added capacity that comes with having done three projects rather than the diminished capacity of having not done one in a while. One addition to the rover arm is a sample collection drill and claw of a simple design. It is kept very simple (2 servos – spin and open) to minimize failure points and development costs. The sample collection tape and bag arrangement is very simple mechanically and could be constructed using standard pharmaceutical materials, assemblies, and sterilization procedures.
Similarities with MER	The solar arrays, communications equipment including antennas, rocker bogie system, batteries, power management, computers, arm, cameras, and so on are identical to MER. These items have been field-proven and slightly over-engineered to cope with long missions under harsh conditions.
Differences with MER	There are several major differences with MER. • The sample arm contains a sample drill/claw in place of the RAT tool and the Mössbauer Spectrometer. • The sample collection assembly wraps around the opposite corner from the high-gain antenna. • The rover can communicate via a wi-fi system with the lander directly and coordinate activities with it.
Sample Collection Arm	The swapping of two instruments for the sampling drill/claw is as much for size and power as for budget. This tool is a hollow drill bit, with the ability to mechanically split and clasp onto small rock samples. The ability to split like this also allows it to drop the very shallow (up to 1 cm) rock cores into the collectors.

Around the base of the sample collection system is a cleaning head that contains a central and outer brush set. To clean the drill bit groove and sample space, the claw is simply pushed into this space, rotated, widened, and then rotated again. Since the bit must be exposed metal, the brushes must be a plastic or fiberglass of some form to avoid vacuum welding (if that is possible at Martian pressures).

In-Situ Sample Analysis

The rover may use the Mini-TES, cameras, x-ray spectrometer, and microscopic imager in the same manner as MER to aid in sample candidate identification.

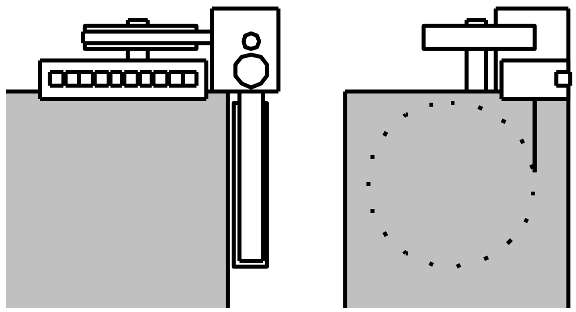
Tape Sample Storage

The sample tape collection system wraps around a corner of the rover base near the sample collection arm. Since the platform is rocker-bogie and hinges at a single point farther back on the chassis, there is room for the source reel in this position. Since the mechanism wraps around the corner of the rover chassis, the amount of room needed for this device is minimized.

1. The source reel feeds a plastic tape to the sample loader. This entire mechanism is sealed against dust until it enters the sample load port. The tape is 2.6 cm across and on a reel that is 15 cm in diameter and 3.2 cm thick including the mount.
   The tape contains circular depressions on both sides. It is slightly creased as it enters the sample load area to form a “V” shape with a spring gasket that keeps the soil from feeding back into the source reel.
   The sample load mechanism has a funnel that faces upward. The core, soil, or rock sample is placed in this funnel and settled onto the tape in one of the two parallel depressions. The funnel is narrower than the tape to avoid spillage and to allow a clean edge along both sides.
   The tape is folded in half, with the “clean” side going over the side holding the sample. This section runs between two capstans that compress both the tape and the sample into a horizontal 1 cm wide shape. It also detects the edges and heats the flat tape around the sample “bubble” to seal it.
2. A second capstan pair rotates the tape from horizontal to vertical.
3. A camera with a soft UV strobe (to minimize sterilization) inspects the tape by looking for the glow of each seal under UV light and checking the sample itself. The underside of the tape is printed with a code that also labels this sample.
4. Eventually, as more samples are taken and the tape feeds forward, it is placed on the take-up reel.

5. When the rover returns to the lander, it feeds several more centimeters of tape to the reel and cuts it at the end.
6. The manipulator arm at the lander removes the tape take-up reel and replaces it with an empty one. The rover then feeds more tape to be fed into the new reel using the manipulator arm and locked using a spring trap device on the reel.

Bag Sample Storage (Diagram zoom/rotate for clarity)



Picture a draw-string bag. Now picture a series of them lined up like one side of an ice cube tray with a single pair of draw strings that can close all of them at once. Since the bags will each contain a rock or core sample, the bag will naturally close to form a disc. This is the sample collection mechanism for larger rocks and core samples. An array of 10 bags, each able to handle a roughly cubic centimeter sample, is arranged in a tray above the spindle for the sample collection tape. It is also arranged below its own spindle (not shown). Having the sample bags and tape in the same location will minimize the demands on the sample collection arm.

The base that holds the bag arrangement contains a pair of hooks that are closed together with a worm gear to draw the string. Once drawn, the string is cut at the edges of the base. The string is drawn around a central spindle to make it easier to pick up by the manipulator arm at the lander. The hooks can be drawn further to a mechanical release so that if one or both strings do not cut properly, the sample can still be retrieved.

Tray Lander Sample Array

Along the outside of the bag sample storage area is a second array of small open-topped boxes set in a tray. This tray can be tipped forward 180 degrees to dump the rock and soil samples into a matching tray presented to it by the lander sample collection arm. It then takes these samples to instruments that analyze them within the lander. If a sample is found to be exceptional, the sample collection arm may add it to the returned package. Otherwise these samples are studied within the lander and discarded. Alternatively, the tray could be removed by the lander and replaced with a new one, or a rotating tray could be used instead to allow the same spindle-reel arrangement to be used for all samples.

Sample Collection Sorties

The mission will nominally be 500 days, minus the initial set-up and concluding activities. The rover will begin with a relatively short sortie to collect the samples determined to be of interest within visual range of the lander. After delivering this reel, it will continue to gather at least two more reels. On the last return, it will also drop off the bag set. It may be possible to replace the bag set on each sortie and assemble a "best of" set at the sample collection bay, but this is outside the scope of this initial design.

Planetary Protection

In addition to lowering the cost of the components, the relatively low tech solution ensures that all components of the tape and bag systems can be autoclaved. As for the apparent contradiction of having a tape that can be both autoclaved for sterilization and heated for sealing when in situ, this type of material is routinely used in sealing pharmaceutical products.

Rover to Lander Communications

The rover may be called upon to coordinate work with the lander by the lander itself. In these cases, the lander is able to establish the position of the rover and command the rover to return to the appropriate location for sample collection and cleaning. It can do this independently, but can be overridden by both Mission Control and the rover's own hazard avoidance systems.

The rover will periodically ping the lander to confirm it is within Wi-Fi range when this is appropriate. If it is within range, it will send basic diagnostic and positional information to the lander to be logged in the event of rover failure. This also gives the lander an idea of where to look for the rover with cameras when the rover returns to visual range.

Since the lander is equipped with a LIDAR that points vertically, it may be worth experimentation to determine if the LIDAR laser can be seen with the appropriate filter on the rover in terms of reflections of the laser on dust in the atmosphere. This may also allow beyond line of site navigation, and possibly even communication, from the lander to the rover.

Mars-Bound Cruise Stage

Overview	The cruise stage for the MSR is, like the entry vehicle, nearly identical to the Mars Science Laboratory design. That said, there are currently two different designs for this cruise stage. One uses the rear panel as a solar collector, and the other as a shade for a radiator that allows the cruise stage to use power from the RTG within the lander itself. In this case, the MSR cruise stage would use the solar power configuration.
General Tech Level	Again, since this is a direct copy of a by-then existing mission, there is no new technology to be created at the point this is built.
Similarities with MSL	The hardware and software of the MSR mission is nearly identical to Mars Science Laboratory with the exceptions listed below.
Delta V and Mass Issues	Since the vehicle is lighter (at least in this design – future iterations may be more massive), it may be possible to either load the cruise stage with less fuel or have increased Delta V from the Cruise stage. Some work has gone into increasing the Trans Mars Injection performance by using the cruise stage as a sort of fourth stage for this operation. Given the lower overall mass and the increased performance of using the cruise stage to give an additional boost, the launch window for orbital transfer and the amount of payload that can be transferred in a less optimal launch window is increased. Therefore the MSR mission becomes more viable overall and could use slightly less powerful third stage or less powerful configuration of the Atlas V booster than MSL.
Location of Software	Assuming MSL also uses the rover computer hardware for guidance, that function would be hosted by the return cruise stage rather than the rover. Also, the EDL phase of the programming would obviously be different owing to the differing landing mechanisms.
Thermal Management	The MSL cruise stage does not require the reactor cooling radiators associated with the MSR version. This reduces mass, cost, complexity, and testing.
Other Cost Reduction Opportunities	Lacking the hardware for thermal management and the tighter mass requirements, the cruise stage for MSR may potentially be both lighter, less expensive, and could use less expensive but heavier options during configuration, provided the newer options did not require testing costs that would exceed the material costs.

Launch Vehicle

Overview	Again, this is identical to or slightly scaled back from the Mars Science Laboratory. That mission uses an Atlas V 541 to launch a 4000 kg payload [12]. This configuration can carry 7600 kg to geosynchronous transfer orbit (GTO) [13]. We could, theoretically with the right launch window, step down to an Atlas V 531 with a GTO capacity of 6900 kg [13]. The 531 configuration indicates it uses three solid rocket boosters versus the 541 configuration with four solid boosters [13].
General Tech Level and Cost	Current USAF estimates place the cost of an Atlas V 500 series launch at $192 million in FY 2004 dollars [13]. It has been launched 8 times thus far with a 100 percent success rate. These numbers are not broken down further. The Delta IV can also launch large (5 meter fairing) payloads. If the mass to GTO could be dropped to 6120 kg, the launch cost could be dropped to $160 million. The Delta IV Medium+ 5.4 configuration has never been launched. Delta IV Heavy has only been launched once and is more expensive than the Atlas V [14].
Advantages of MSR over MSL	Since the vehicle is lighter than MSL, this opens more options for launch vehicle, TMI stage, and launch window. Furthermore, MSR does not involve processing an RTG, further reducing the overall package demand.

Biological Protection

Overview	Biological protection covers two broad categories – Earth to Mars and Mars to Earth. Most of these measures are covered in the sections that address the particular piece of hardware involved.
General Tech Level	The pre-launch planetary protection methods listed here have been used with Viking and to some degree with MER. The strongest level of protection is reserved for the components that touch samples, and those use the same technology as the arm on the Phoenix lander.

Pre-Launch Sterilization

The sterilization procedure relative to this vehicle would be as follows.

1. All electronic components and other related equipment that would be too sensitive to be autoclaved will be packed into sealed boxes. The contents will be sterilized using whatever technology (chemical, UV light, etc.) does not damage the contents, then they will be placed into previously-autoclaved protective boxes.
2. The remaining vehicle components are autoclaved and appropriate components sealed until used at Mars.
3. The autoclaved components (in particular wiring and fuel lines) are tested.
4. Non-autoclaved components are placed within the structure and final structural assembly is completed.
5. The capsule components are sterilized using UV light whenever exposed or sealed if possible.

Realistically, not all this can be done. However, where certain components, especially in sample handling, are involved, those components are sterilized in the same manor as the digging arm for the Phoenix lander and placed in plastic enclosures until landing.

Back-Contamination Protection

The return capsule is sealed in a thin Mylar coating that burns off on entry. The only area exposed to Mars that is then exposed to earth would be the top of the capsule lid, and it too is covered with a thin flammable layer that, even on the back of the capsule, will burn away during entry.

Also, as noted, the cruise phase capsule is lightweight and if it enters, it does so in a tumble to force it to completely burn up or (at the very most) loose two reaction control tanks and engine pods that have been burned by plasma on the outside and flushed with hydrazine on the inside.

Sample Handling at Earth

After the capsule enters the atmosphere and lands in the Utah desert, it is placed in a special box that protects it during the flight back. When it is collected, the crew doing so will be in biohazard suits. Once sealed in the padded collection box, the capsule is transported back without the “need” for this level of protection. It may be worth doing to scrape the ground at the impact site and bag it as well – again mostly for public relations to show that NASA is not taking any chances. The box will be built strongly enough to withstand a crash of either the helicopter or jet en route to the final destination.

Samples will be analyzed in a sterile environment with the atmospheric temperature and pressure reduced to Martian levels. While initial measures will treat the samples with the same care normally reserved for plutonium, after appropriate biological screening is completed, some samples may be distributed to more delicate instrumentation on a case-by-case basis.

Avoiding a Single Point of Failure

Generally, one has two choices with avoiding single point of failure – either create a redundant system that will be dead-weight if the primary system works as designed, or make the primary system split into two systems that each can do the task, but do it more slowly than intended and at greater mass.

With the Rigel design, there are multiple systems with different goals and capabilities but with overlapping capacity to perform tasks. If everything works as designed, nothing is underutilized. If something fails, a secondary system (usually the main manipulator arm) can be pulled into duty to fix the problem. If the arm itself fails, the mission will still function if everything else works. A partial list of these overlaps is shown below.

Primary Plan Method	Backup Method(s)
Deploy secondary array for immediate power.	• Deploy Cruise Stage array temporarily, then retract it when primary or secondary array is set up. • Deploy the primary array early.
Deploy rover platform automatically.	• Deploy manually using arm.
Unfold rover solar arrays automatically	• Deploy manually using arm.
Drive Rover off platform	• Three ramps available. • May make arm strong enough to deploy from elbow hook (reduce mass by not making entire arm able to lift the rover).
Deploy solar array automatically	• Deploy manually using arm.
Clean solar array using arm	• Clean passively from dust devil events. • Design with sufficient power to function with minimal (dust devil) cleaning.
Local communication with rover via “wifi” connection while in line of sight	• Use low-gain omni-directional antennas. • Satellite relay to rover.
Rover gathers full set of samples	• Rover gathers contingency sample first. • Arm gathers set of shallow core samples and rocks from lander vicinity.
Rover returns samples for loading in processing bay	• Rover samples are taken using main arm and put directly in the processing bay or return capsule as appropriate.
Lander computer runs all operations from Earth launch until MAV launch,	• Return cruise stage can handle all operations until ascent.

and all surface functions after that.	

Testing

Overview	Some tests can be run with some innovation to reduce the overall cost of early system development. A few ideas are listed here.
MSR Capsule Flight Test	If desired, the sample return capsule is light enough that it could be launched outside the atmosphere on a sounding rocket. It could even use an additional stage to fire it into the Earth's atmosphere at a higher speed if desired, as was done with some of entry tests in the 1960's. This entry system is so small and light that perhaps the best test would be to include it on the interstage of a commercial satellite launch and let it come back along with the second stage. While the return would require an ocean recovery, saving the cost of a sounding rocket launch may make this appealing.
Rover Testing	After Spirit and Opportunity reach the end of their missions, a sample collection modification could be done to the test rover at JPL and this system could be tested without the building of a completely new test article.
Low Technology Items	The solar array deployment mechanism is so simple it could be built by hobbyists. Lower technology items such as this may be very rapidly prototyped in this manner as part of university or small corporate grants. Similarly, the sample collection arm and the sample handling bay (with two robots and elevator) is exactly the sort of project that a hobbyist or university could do early prototypes of to approximate fit, tension, and programming issues.
Engine Development	Adding criteria to a future X-Prize Cup or Lunar Challenge type of event may use engines that match the performance of the sample return mission could defray several million in development cost for early prototypes.

Ground Operations

Overview	The goal will be to run this operation out of a relatively small office of 12 controllers at peak periods.
General Tech Level	The communications demand of the system is very similar and yet different from MER. Instead of two vehicles on opposite sides of the planet with roughly 24 hour coverage, we have two vehicles in the same location.
Increased Vehicle Automation in Transit	The trend with Deep Space 1 and New Horizons has been toward vehicles that handle their own navigation in transit. While this reduces workload on mission control during cruise phase, it increases development demand for software testing. An appropriate mix of automation and control for this mission will be based on the software maturity for this purpose at it exists then. The goal should be to reduce control communication periods to once per week, with direct involvement in course correction maneuvers.
Increased Rover Automation	MER is already experimenting with more automated software for mobility, sample arm alignment, and navigation. These skills plus coordination with the lander will be key to this mission. The basic job of the rover is to position itself by the lander and sit still while the lander collects the samples and cleans the arrays. It may also be used to circle the lander and take photos of areas the lander arm cannot reach or engineering photos in spectra the lander cannot take with the arm. Since the lander can be examined as if it were any other camera target, the rover should be able to do these operations with minimal software changes.
Lander Automation	The lander has fairly basic demands (make fuel, watch for dust storms) that can be completely automated with diversions from the norm reported immediately and engineering reports done once per week. Power management and fuel production will be recalculated on the fly depending on current circumstances.
Cost for Primary Mission	The logical staffing and control level, then, may be to have a small crew of 5-10 to handle routine operations of rover navigation and sample selection, lander routine operations, and coordination of research with other missions. Future information management systems may allow outside (read: unpaid) investigators to do a great deal of analysis and reporting with data mining and organization tools, then flag specific items for follow up within NASA at a higher level.

Extended Mission Options	After the return vehicle leaves the surface, there are no guarantees that the lander will still have a functioning solar array or that the rover will remain operational. That said, It is possible for the lander to continue using the medium gain antenna to communicate via satellites and do local analysis of returned samples and local science with the remaining instruments. Again, a weekly check-in and command sequence would be adequate.
Eventual Mission Shutdown	One closure activity of potential use would be to have the ISPP system go into oxygen-only production mode and run in this mode until the equipment breaks under Mars conditions as a stress test. If a sample return mission ran close enough to a human mission, it could even mothball itself with the ability to be reactivated as an emergency oxygen oasis. Further, various back up and emergency systems could be tested to failure before the next mission to expand the engineering knowledgebase.

Mission Cost

Overview	Cost calculation in aerospace has always been difficult. It is even more so in this case, because most of the general rules based on history do not apply to this design. Even when attempting to remove this bias from online calculation tools, the project cost soars due to the model treating each item as a completely separate project. That said, numbers were run through the best calculation tool found and these sources of bias are documented, with a final cost estimate given conservatively at the end.

Cost Factors Limiting Standard Models

Lower Cost Factor: EDL Mass	As noted at the beginning, a major emphasis of this design is to use as much hardware at the “flown to Mars successfully” end of the development cost spectra. Other areas are kept as simple as possible and as proven in industry as possible. While the “flown before” or “already built” criteria have been used before to justify mission transitions from Pathfinder to MER or from Mars Polar Lander to Phoenix, they have been with probes that were heavier and more robust than their predecessors. Since MSL is at the very limit of current design, we are in the enviable position of designing a craft that can (indeed, must) back away from the state of the art rather than pushing it farther forward.
Lower Cost Factor: Dry Mass Bias	The cost models currently available online to the public are vague, order of magnitude systems that blur historical data on vehicles with many small components with total mass. Whereas the MSR has this same criteria with roughly half its dry mass, the other half is built as tanks and other heavy structures not normally flown mostly empty on planetary spacecraft. Doing so biases “dry weight” calculations since the MSR is the first vehicle to be launched 70 percent “dry”.
Lower Cost Factor: Common Test Articles	Using equipment from so many flown or then-flown missions allows re-use of many test articles after they have served their purpose with the original missions. For example, after the MER rovers eventually fail, the JPL test version used to validate procedures before sending them to the rovers could be built in the sample collection configuration with minimal effort. The same can be said of other instruments onboard and the aeroshell.

Lower Cost Factor: Common Assembly with MSL	If this MSR configuration were actually approved in some form before MSL were launched, some elements, such as the cruise stage and aeroshell, could be built at the same time.
Potential Higher Cost factor: MSL Success Dependence	With the design of MSL right at the very limit of EDL technology, there is a risk of too much dependency on this design. If the EDL fails and the craft is lost, then we are not building from success. As it stands, the MSL aeroshell is equipped with extensive engineering monitoring capability [1]. We can have confidence in the design once the vehicle lands, and will have the data needed to assess MSL aeroshell risk factors. If MSL fails and MSR is already approved, MSR could be scaled back and/or sent to a lower altitude landing site depending on the type of failure.

Cost Estimation Tools [22]

Cost Estimation Tools	Unfortunately, online cost estimation calculators are very generalized. One online calculator allows one to compute an approximate cost based on dry mass limits the upper bound of the calculations to 1249 kg, whereas this mission has a dry mass of 2055 kg (http://cost.jsc.nasa.gov/SVLCM.html). A second cost calculator (http://cost.jsc.nasa.gov/AMCM.html) results are shown below. As noted, we are in an odd position where it has either flown before or the difficulty in building it is low compared to normal spacecraft development. This seems to average the result between case B and C, below. C is shown because for some elements, such as the return cruise stage, this is somewhat true. Even though six cruise stages will have flown to Mars at the time, none have flown back. Two interplanetary vehicles have returned capsules to Earth, though (Genesis and Stardust) using much more complex systems. For iteration 3A and b, the mass has been increased, and the Block 1/average and Block 2/Hard numbers have been added to this table as E and F, respectively.

<table>
<tr><td rowspan="8">Cost Estimates</td><td>All versions</td><td colspan="2">Quantity: 1
Dry Weight: 2055 kg
Launch: 2011
Mission type: Planetary Lander.
FY: 2004 dollars</td></tr>
<tr><td></td><td>Criteria</td><td>Cost</td></tr>
<tr><td>A</td><td>Block 3, Difficulty Average</td><td>$1059 million</td></tr>
<tr><td>B</td><td>Block 2, Difficulty Low</td><td>$786 million</td></tr>
<tr><td>C</td><td>Block 1, Difficulty Low</td><td>$1006 million</td></tr>
<tr><td>D</td><td>Block 2, Difficulty Average</td><td>$1223 million</td></tr>
<tr><td>E</td><td>Block 2, Difficulty High</td><td>$1901 million</td></tr>
<tr><td>F</td><td>Block 1, Difficulty Average</td><td>$1564 million</td></tr>
<tr><td>Cost Estimate Definitions (directly quoted from page)</td><td colspan="3">Quantity - The quantity is the total number of units to be produced. This includes prototypes, test articles, operational units, and spares.

Dry weight - The dry weight is the total empty weight of the system in pounds, not including fuel, payload, crew, or passengers.

Mission type - The mission type classifies the type of system by the operating environment and the type of mission to be performed. Select one that best describes the system you wish to estimate.

IOC Year - The IOC is the year of Initial Operating Capability. For space systems, this is the year in which the spacecraft or vehicle is first launched.

Block Number - The block number represents the level of design inheritance in the system. If the system is a new design, then the block number is 1. If the estimate represents a modification to an existing design, then a block number of 2 or more may be used. For example, block 5 means that this is the 5th in a series of major modifications to an existing system.

Difficulty - The difficulty factor represents the level of programmatic and technical difficulty anticipated for the new system. This difficulty should be assessed relative to other similar systems that have been developed in the past. For example, if the new system is significantly more complex than previous similar systems, then a difficulty of high or very high should be selected.</td></tr>
</table>

Estimating Difficulty

One element then becomes whether the difficulty level should be considered Low or Average. While this is not documented beyond the definition above, the Reserve Percentage model is instructive.

Reserve Percentage and Development Budgets

The standard Reserve Percentage Table is shown below. As often as possible, the technology for this design is in the 10-20 percent area. Even new technology in the MSR is far from state of the art.

RESERVE PERCENTAGE TABLE

10% -	Off the shelf; hardware exists; no modifications required.
15% -	Modifications required to existing hardware.
20% -	New hardware, but design has been through Critical Design Review (CDR). Also, vendor quotes.
25% -	New hardware, but design has been through Preliminary Design Review (PDR).
35% -	New design but within State of the Art (SOTA); CERs or analogs used to estimate cost. Also vendor ROMs.
50% -	New design; remote (or no) analogs to subsystem; beyond current SOTA (never been done before)

(source: http://cost.jsc.nasa.gov/guidelines.htm)

The reason this table is given is to give more detail into the Difficulty factor for the previous calculation. Given we are in the 10-20 percent range on a project where the range is 10-50 percent, the figure of 2-Low on a scale from 1-5 seems the most accurate.

Launch Vehicle

NASA cost estimation sites leave this cell blank for both the Atlas V and the Delta IV Heavy, probably due to the low flight rate so far (http://cost.jsc.nasa.gov/ELV_US.html).

Current USAF estimates place the cost of an Atlas V 500 series (used for MSL) launch at $192 million in FY 2004 dollars. It has been launched 8 times thus far with a 100 percent success rate. These numbers are not broken down further [13].

The Delta IV can also launch large (5 meter fairing) payloads. If the mass to GTO could be dropped to 6120 kg, the launch cost could be dropped to $160 million. The Delta IV Medium+ 5.4 configuration has never been launched. Delta IV Heavy has only been launched once and is more expensive than the Atlas V [14].

Hybrid Calculation

The best solution at this stage seemed to be to break the vehicle down by mass into several groups based on technology and flight rate and add them together. Trying this revealed that the calculator assumes all these programs are separately managed, as breaking down the figures by group and adding them together tripled the projected cost. Since the hybrid result was more expensive than the worst case single result for the same mass, and this is theoretically impossible, this invalidated the result.

Development and Launch Results

So the design, given the biases noted, costs approximately a $1 billion plus a $192 million launch vehicle.

Operational Costs

The Mission Operational Cost Model calculator gives a rough estimate of operational cost based on the type of mission and the original mission cost (http://cost.jsc.nasa.gov/MOCM.html).

Running this calculation for an investment cost of $1 billion results in an average annual support cost of $35 million. The duration of a mission with a 9 month flight to Mars, 500 days on the surface, and 9 months back is 2.86 years, so if we round that up to 3 years, that gives us an operational cost of $105 million.

Final costs

Working with the following figures:

MSR Spacecraft	$1000 million
Atlas V 500 series	$192 million
Operating costs	$105 million
Total Cost	$1,297 million

As it stands, with the $2 billion cost limit, we still have a vehicle with a fairly large margin. This is very good given the wide range of answers that can be derived from these calculators..

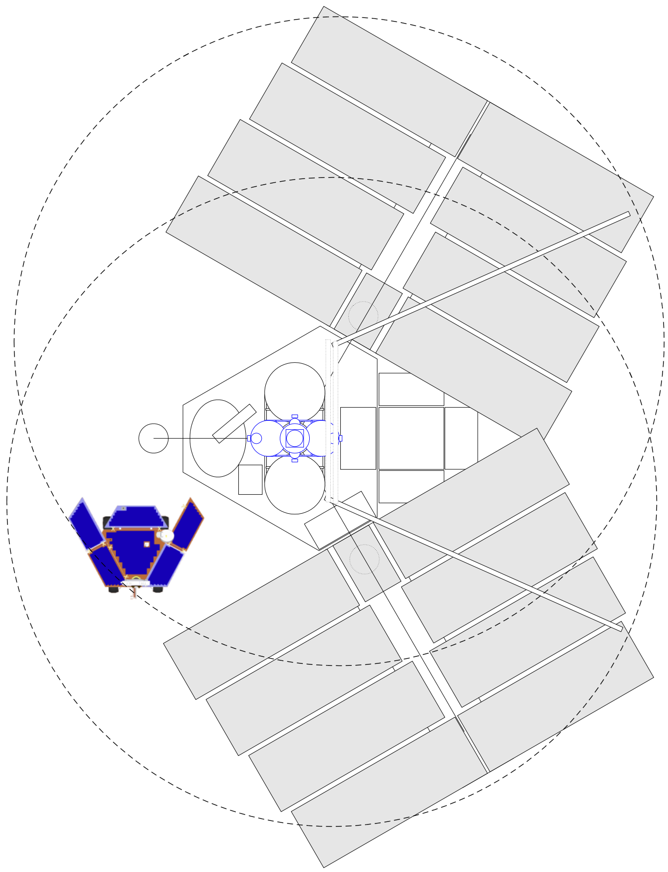
MSR Follow-On Missions

Mid-Latitude Variant

Overview

The design shown so far is optimized for an equatorial landing. One issue with MER is that even the minor latitude difference of 15 degrees from the equator delays the start of operations for the Spirit rover relative to Opportunity by several weeks each Spring. Therefore the first MSR mission is optimized for a landing zone with maximum sunlight.

Diagram



Differences with MSR

As shown, the lander here has a second solar array. One array captures morning and noon sunlight, whereas the other captures noon and evening light in mid-latitudes. The lander must be deployed before the arrays. Since the arrays are angled relative to the ground, they do not actually overlap as shown here. A second manipulator arm allows the secondary array to be cleaned.

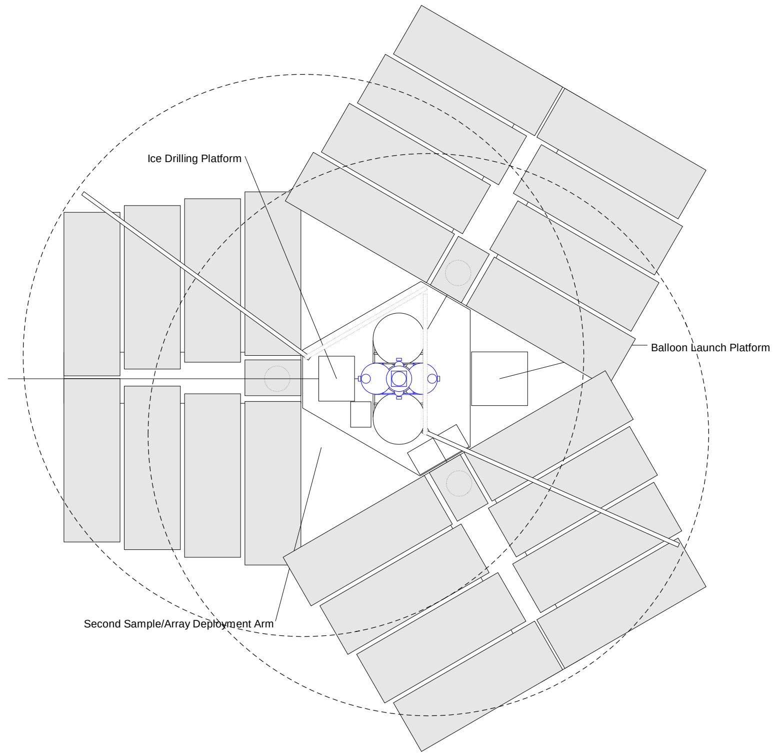
The added mass of the second array and manipulator arm (roughly 100 kg) may be handled by the aeroshell and launch vehicle easily. However, the return vehicle will no longer have the benefit of full equatorial boost relative to orbital velocity. Therefore it will have to be built with larger margins, or the original design would be made larger for greater reuse.

Polar Ice Cap Variant

Overview

Such a mission would, like Mars Phoenix Lander, dig deeper in time rather than further in space to increase our understanding of Mars in that dimension. The sample return hardware would consist of a series of water filters that would both clean water for use in ISPP and gather dust samples over the history of the planet.

Diagram



Differences with MSR

This design only works above the Martian arctic circle, in Summer, while on nearly pure water ice (i.e., the Northern polar icecap). The three arrays capture sunlight at a 90 degree angle as the sun hovers all day around to the horizon. The arrays are also stacked to allow them to avoid shading each other more than necessary.

The hydrogen tank is replaced with an ice boring device with has the combined function of melting water for analysis/return and providing hydrogen for propellant production. The rover is also replaced with a platform for launching hydrogen balloon probes.

Hydrogen Mining

One issue would be whether the bore-hole would yield enough water for ISPP. A quick estimate indicates a bore-hole 11.3 cm in diameter and 27 meters deep could deliver enough water to launch a 0.5 kg sample return using the original spreadsheet numbers (240 kg of water). Also, deep samples like this could theoretically have biological significance, and would give a deep climate history of the more recent past.

Hydrogen Balloon Probes

If the borehole system still works after the sample is collected and the propellant is manufactured, there remains the issue of the rover alternative. If hydrogen can still be mined, and polar surfaces are too rough for initial operations with a rover, it seems logical to launch a series of 1-2 kg science packages using the hydrogen supply to inflate a series of balloons. These packages could use a combination of the lander and any polar relay satellites to relay data to Earth. Fortunately, every satellite orbiting Mars passes over the pole on each orbit, giving a richer array of communications relay options than more equatorial probes.

Balloon packages could relay weather information as well as surface imagery, and upon landing continue to provide data on local conditions. They could even be solar powered mini-stations that could drop ballast and make several hops or ongoing low altitude observations.

Mission Design

A Phoenix lander scientist said words to the effect that they are digging deeper into Mars vertically across time whereas the MER explores it horizontally across space. As a follow-on to this level of science and perhaps an intervening cryobot mission, the Polar Sample Return with the hydrogen balloon sequence would dramatically increase our knowledge of these mysterious regions on both dimensions at once.

It should also be noted that a polar landing would not benefit from the planet’s axial rotation when reaching orbital velocity, so a larger launch vehicle would need to be planned. It may be wise, if the sample return landers become the next decade of flagship missions, to overbuild the first variant to allow it to do the missions of the second and third, then build components for all three at once where it is cost effective and less risky to do so.

Other Possible MSR-Derived Missions

Mission	Description
Robotic Base	The deployment of remote solar arrays, ISRU, rover with home base, and return vehicle capability essentially make the MSR mission a Mars base in miniature. The main issue with a robotic base at Mars is the 26 month gap between possible missions. Depending on hardware in place with such lead-time seems rash unless, like orbiter-lander relay systems now, there are enough elements on site to be relatively certain that at least one will be functional. Surface operations that reflect this redundancy at a particularly attractive scientific location could potentially happen with a sufficiently large initial base and very precisely guided scout class follow-up missions. So far the only advantageous method by which one robotic mission can aid a completely different mission is by the existing system of orbital relay. Ongoing missions, such as weather or seismic networks, would probably be scout-class. That said, there will probably be a window for robotic base operations if either A) a deeply interesting scientific site warrants ongoing missions with in a 20 km radius or B) as an engineering preliminary to crewed missions where a single landing places a very large robotic mission on the surface to human-rate a particular landing vehicle.
Powered Landing Experiments	Since the descent tanks for landing are also the first stage ascent tanks for the return vehicle, they are only filled to a fraction of their actual capacity during Mars EDL. If it becomes necessary to consider crewed missions with greater use of powered descent than is currently practiced, this could be experimented with using this hardware on a future mission without any modification.
ISPP Hopper	Such a mission could use retractable solar arrays to create enough propellant to hop around to various landing sites, and would even be large enough to deploy one or two MER-class rovers at each site and retrieve them. While the science alone makes such a mission compelling, it would also give repeated real-world experience with descent and landing, solar array deployment, and airborne scouting of the surface. At a normal flight rate from Earth of one lander every 26 months, and at a local flight rate of once per month, we could gain a quarter-century of flying experience in a single Martian year, including take-offs, before a crewed mission arrived.
Lunar Applications	Just as with Mars Direct, the same vehicle could be used in lunar environment without ISPP and on a smaller scale [3]. Overlapping hardware with a polar crater sample return or related human-

	precursor or human-extension mission would be ideal. Propulsion systems, landing gear, return vehicles, and control avionics could be used in either environment. Solar power systems would be far more effective in the lunar environment (although the memory wire system would be difficult due to temperature issues), and parts of the ISPP and hydrogen storage systems could be used as a fuel cell system for overnight power storage. A fuel-cell system is proposed for the NASA Lunar base architecture.

Gap Analysis and Iteration 4+

The following areas will be considered for future design iterations of this vehicle or future studies inspired by it. Note is that the tank masses were not calculated in extensive detail in iterations 1-3B because they are to be reshaped in Iteration 4.

Propellant Tank Refinement	Experiment with different, more efficient shapes for the return vehicle. These options include the following: • Converting one or both stages into a flattened spheroid dual tank rather than dual-dual tank complex. • The TEI and/or cruise stage sit laterally, not vertically. • TEI or cruise stage sit on the side and stack after landing.
Trajectory Modeling and RCS	Do more trajectory modeling for launch, landing, and return. Apply this information to properly sizing the RCS pods.
Integrated ISPP Design	Do the chemical engineering CAD model and equations for the integrated ISPP system in terms of thermal, electrical, and pressure loads, along with more accurate production rate and system mass data.
CAD Model	Do a CAD model of the vehicle for fit checks, mass estimates, and stress-load estimates on structural members, and center of gravity calculations. Note that the 3D models in this revised paper are illustrations created based on numbers from the latest iteration in this paper and using an alternative deck layout. This is a simplified preliminary 3D model (created in Google Sketch-up). For further development, we need a solid 3D CAD model (e.g. Solidworks) that allows for volume computations, bolt placement, and so on.
Flight Model	Do a digital flight model in Orbiter and/or X-Plane.
Sample Collection Prototype	Design a CAD model and a full-size prototype of the sample collection tape system and drill/manipulator hand. Potentially partner with a robotic platform for MDRS or other field work.

Solar Array Prototype	Build a solar array blanket prototype with compressive elements and memory wire controls, though probably not active solar cells, to show deployment methods and reliability.

Conclusion – Public Psychology and Mars Exploration

Robots as Storytellers	As Mars exploration activists, we focus on robotic missions in terms of expanding the knowledge and vision of humanity, but we also know that no one threw a ticker tape parade for a sample return capsule. It is important to recognize that the stories our robots tell are often not the ones we expect.
Robots as Actors	Children told about Sojourner or MER often ask when the robot is coming home. They see these machines moving with (as our stories often give them) anthropomorphic qualities and assume that they are lonely or lost when missions end. It is interesting that the Sojourner rover managed to outlive the lander despite the designed lifetimes of each, and that it (based on MRO photos) made it most of the way back to the lander before it, too, died while seeking the lander's signal. On seeing that pixilated orbital picture, I couldn't help but think of climax of Romeo and Juliet. When Spirit and Opportunity eventually fail and are mourned and celebrated, we should think these stories through before launch on future missions. The parallels of Rigel with Viking, Pathfinder, Opportunity and Spirit, not to mention the hopefully-successful Mars Science Laboratory and Phoenix, are fairly clear because of the derivation of technology. Similarly, Ares/Orion is clearly the child of Apollo and STS. One always hopes the children go on to do bigger and better things.

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