Methods for Achieving Long Range Mobility on Mars

Robert Zubrin†
Martin Marietta Astronautics
PO Box 179
Denver, CO 80201

Abstract

This paper discusses alternative methods for enabling human explorers to achieve long range mobility on the surface of Mars. Vehicle types considered include ground rovers, winged and lighter-than air atmospheric vehicles, and suborbital ballistic vehicles making use of rocket propulsion. Energy sources considered for such vehicles include batteries, fuel cells, radioisotope generators, solar photovoltaics, chemical combustion engines, and nuclear reactors. It is found that the production of fuels and propellants out of indigenous Martian resources greatly enhances the potential for long range mobility, so much so that it may be considered the essential central element of any manned Mars exploration strategy.

Introduction

Mars is a very big place. With a surface area of 144 million square kilometers, the Red Planet has as much terrain to explore as all the continents and islands of Earth put together. Moreover, the Martian terrain is incredibly varied, including canyons, chasms, mountains, dried river and lake beds, flood runoff plains, craters, volcanoes, icefields, dry-ice fields, and "chaotic terrain," among others. The U.S. Geological survey currently records no less than 31 types of Martian terrain on its "Simplified Geologic Map[1]," and this before high resolution imaging of Mars has even been done. Some of the Martian terrain features, such as the 3000 km long Valles Marineris, are of continental extent, and thus the exploration of even a single such feature will require continental scale mobility[2]. Mars is believed to have had a warm, wet climate, suitable for the origin of life, for a period of time longer than it took life to evolve on Earth. Thus the search for life, either extant or fossilized, will be of the highest priority for Mars explorers, as around its result turns the question of whether life is a universal or unique phenomenon. The results of the Viking missions have shown that, if life does exist on Mars, it is rare, and its finding will take more than a bit of searching. The experience of professional paleontologists on Earth has shown that the hunt for fossils will also require much footwork and a very wide net. Furthermore, while the hope of producing a positive result from such investigations depends critically upon the surface mobility of the exploration teams, the ability to demonstrate a convincing negative result will require searching virtually the entire surface of the planet.

The battery powered Lunar rover used during the Apollo program had a one-way range of about 20 km, giving it a sortie range of 10 km from the landing site. A manned Mars expedition equipped with equivalent transportation would be able to explore only about 300 square kilometers, regardless of the length of its surface stay, and nearly *half a million* such missions would be required to examine the entire surface of Mars *one time*. Even if it were considered sufficient to simply examine a variety of points of interest, the limited mobility afforded by such a vehicle would be a severe impediment and vastly increase the cost of mounting a manned Mars exploration program. For example, Table 1 shows a list of points of interest in the Coprates triangle area, surrounding a landing site at 0 degrees latitude and 65 degrees west longitude.

Table 1 .Surface Features of Interest in the Exploration of Mars

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

† Member AIAA

It can be seen that if surface mobility were limited to 100 km range (10 times the Apollo Lunar rover) at least 12 landings would be needed to visit all the sites. If the mission had a surface mobility of 500 km, then only 4 missions would be required to visit all 14 sites, and these 4 missions could access 8 times the surface area as that available to the 12 missions conducted by crews with 100 km range.

Manned Mars missions are likely to cost billions of dollars each. Currently, much thought is being expended on ways to reduce this cost by introducing technologies such as nuclear thermal propulsion, aerocapture, or cheaper launch vehicles. While such efforts are to be heartily encouraged, it must be pointed out that introduction of any of these technologies will cost billions, and only reduce Mars mission costs by (at most) a factor of 2. The expansion of surface mobility, on the other hand, is likely to be cheaper, and can potentially increase mission exploratory effectiveness by several orders of magnitude.

It is thus clear that there is nothing more important in determining the cost-effectiveness of a program of manned Mars exploration than the degree of mobility provided on the surface of the planet.

Surface Vehicles

Wheels, treads, half-tracks, and even motorized legs are all viable options for the propulsive mechanism of a surface roving vehicle on Mars. The essential issue is what provides the power to move the vehicle's mechanism. The advantage of chemical power as opposed to electrical power is very large, as seen in Table 2.

Table 2 Power to Mass Ratios for Candidate Mars Rover Propulsion

Power Source	Power/Mass (W/kg)
RTG	5
DIPS	8
Photovoltaic (during daylight)	16
H_2/O_2 Fuel Cell	55
Internal Combustion Engine	1000

It can be seen that a combustion engine can have a power/mass ratio about a factor of 20 higher than that of an H_2/O_2 fuel cell. Now for a given life support system mass, the vehicle's range will be directly proportional to its speed, which is in turn proportional to the power. Furthermore, if one of the other options try to match the combustion engine's power level, its weight will rapidly become excessive. For example, if the rover is equipped with a 50 kW (about 65 hp) of power, the mass of the required internal combustion engine would only be about 50 kg, while that of a set of fuel cells would be 900 kg. The combustion powered car could thus take along 850 kg of additional science equipment and consumables compared to a fuel cell powered vehicle of equal power, and again have much greater endurance, capability, and range. Furthermore, the fact that the combustion powered vehicle is virtually power unlimited allows sortie crews to undertake energy-intensive science at a distance from the base that would otherwise be impossible. For example, a combustion vehicle sortie crew could drive to a remote site and generate 100 kW to run a drilling rig. Rover data transmission rates can also be much higher, which in turn increases both crew safety and sortie science return. Combustion engines can also be used to provide high power for either main base or remote site construction activity (bulldozers, etc.) Thus we see that the greater power density of combustion powered engines will provide for greater mobility with a much smaller, lighter, and far more capable vehicles, and a more potent and cost-effective Mars exploration program all-around.

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

Solutions to the specific design challenges posed in the design of combustion engine driven ground vehicles are discussed in reference 3.

The range of a ground rover powered by chemical combustion will depend critically upon the energy/mass ratio of the propellant utilized. While in principle any bipropellant combination could be used, transportation logistics dictate that at least most of the propellant used be manufactured on Mars out of indigenous materials. A list of potential combinations is given in Table 3.

Table 3. Potential Bipropellants for Use in Mars Mobility Vehicles

Bipropellant	Energy Density	
	kJ/kg	kJ/liter
H_2 / CO_2	21000	1500
N_2H_2 / CO_2	4200	4000
H_2 / O_2	13500	4725
CO / O_2	6540	7720
CH_3OH / O_2	7665	7535
CH_4 / O_2	10080	8570

The Martian atmosphere is 95% CO_2, and thus the H_2/CO_2 and N_2H_2/CO_2 combinations given in Table 3 can function as air-breathing engines, much in the manner that internal combustion and jet engines do on Earth. In these cases, therefore, the Energy/Mass ratio given is that of the energy release per unit mass of the non-CO_2 fuel, since the CO_2 does not have to be carried by the vehicle. It can be seen that from the point of view of energy/mass ratio, the H_2/CO_2 engine is superior to all other options considered. However storage of hydrogen is likely to present formidable problems that may make the use of such a system on a ground rover impractical. In that case, the high energy density of CH_4/O_2 would appear to make it the preferred option.

Ballistic Vehicles

While ground vehicles driven by chemical combustion may endow Mars explorers with sortie ranges on the order of 500 km, the rough nature of the Martian landscape dictates that true long range mobility can only be achieved by vehicles capable of flight. Candidate flight systems include ballistic vehicles employing rocket propulsion. and various types of atmospheric craft. Since the former are at least conceptually simpler, they will be considered here first.

The standard equations of orbital motion[4] can be used to calculate the range of a ballistic vehicle traveling from one point on the surface of a planet to another. Such a vehicle travels in an ellipse with one focus at the center of the planet. After an extended series of manipulations which involve differentiating θ, the angle between periapsis and the point where the orbit crosses the planet's surface, with respect to the orbits eccentricity, e, while keeping the orbit's energy constant, one can obtain:

$$\cos\theta = -2(1 - W)^{0.5}/(2 - W) \qquad (1)$$

where W equals the square of the ascent velocity divided by the low orbital, circular velocity of the planet ($W= (V_i/V_{circ})^2$.)

Equation (1) gives the maximum range of travel of a ballistic vehicle under the idealized case of a spherical planet with no atmosphere and an initial velocity generated instantaneously at the planet's surface. The results it yields for a vehicle hopping on Mars (R= 3380 km, Vcirc = 3.57 km/s) are given in Table 4.

<u>Table 4. Ideal Maximum Range for a Ballistic Vehicle on Mars</u>

V_i (km/s)	W	cosθ	φ (radians)	Range=φR (km)
0.5	0.0196	-0.99995	0.020	68
1.0	0.0785	-0.99913	0.083	281
1.5	0.1766	-0.9950	0.200	676
2.0	0.314	-0.9825	0.375	1266
2.5	0.491	-0.9456	0.663	2240
3.0	0.707	-0.837	1.157	3911
3.5	0.962	-0.377	2.367	8000

In Table 4, $\varphi = 2\pi - 2\theta$, is the number of radians across the planets surface transversed by the hop. The artillery equation for the maximum range of a projectile fired on a flat planet is;

$$\text{Range} = V_i^2/g \qquad (2)$$

where g is the gravitational acceleration at the surface of the planet ($g = 3.77\ m/s^2$ on Mars). It can be seen that for small V_i, equation (2) predicts ranges that are only slightly less (66.3 km for V_i = 0.5 km/s, 265 km for V_i = 1.0 km/s) than those predicted by equation (1). However as V_i and range are increased, the curvature of the planet becomes more important and the results of (1) and (2) diverge. When V_i is set equal to V_{circ} (3.57 km/s), the vehicle is orbital and the range given by equation (1) goes to 11097 km, which takes the craft to the opposite point on the surface of the planet, while that predicted (very inaccurately) by equation (2) is only 3380 km.

As stated above, equation (1) is idealized in the sense that it assumes instantaneous acceleration. However a rocket vehicle accelerating in a gravity field will experience "gravity losses" due to the necessity of accelerating the craft in a direction opposite the gravity vector for at least part of the trajectory. Calculation of gravity losses with precision requires the use of a simulation code. However, useful approximate results can be obtained for vehicles with a lift-off thrust-to-weight ratio greater than 1.4 by assuming that gravity losses increase the total ΔV required to attain a trajectory that corresponds to a given V_i by a factor of 1.15.

On Mars, a ballistic vehicle can take advantage of the atmospheric drag to reduce the ΔV required to land from a ballistic trajectory to about 500 m/s, regardless of the ascent ΔV.

Candidate bipropellants for a Mars ballistic hopper include CO/O_2, CH_4/O_2, and H_2/O_2. The CO/O_2 rocket has the advantage that all of the components required to manufacture its propellants can be extracted out of the 95% CO_2 Martian atmosphere. However with a specific impulse of 270 s, the performance of such a vehicle is rather low. The H_2/O_2 vehicle, with a specific impulse of about 460 s, enjoys high engine performance, but suffers as a practical system from low vehicle propellant mass fractions, serious cryogenic propellant storage problems, and the danger of hydrogen leaks. However the greatest difficulty faced by the H_2/O_2 hooper concept is the fact 1/7th of the propellant it expends is hydrogen. While hydrogen exists on Mars in trace quantities in the atmosphere and in large amounts in subsurface permafrost, the difficulty of its extraction will make it a scarce resource, more wisely hoarded for life support augmentation than blown away in vast quantities as rocket hopper propellant. Hopping around Mars using H_2/O_2 propellant may thus prove to be an excessively expensive way to travel. Lower in performance than H_2/O_2 but avoiding its cryogenic storage, leakage, and mass fraction problems and using much less hydrogen, is the CH_4/O_2 option. A CH_4/O_2 rocket would produce a specific impulse of 370 s out of a bipropellant combination 1/18th of which is hydrogen. All three chemical bipropellant combinations require about 5 MWe-hr/tonne to manufacture, an energy requirement that is much too large to allow such vehicles to manufacture their own propellant. Such vehicles must therefore operate in a series of sorties from a fixed base where propellant manufacturing facilities are located.

An alternative type of hopping vehicle is also possible, using a nuclear thermal rocket (NTR) engine with Martian CO_2 as its propellant. Since this can be acquired at low energy cost (85 kWe-hrs/tonne) through direct compression out of the atmosphere, rocket vehicles so equipped would give Mars explorers complete global mobility, allowing them to hop around the planet in a craft that can refuel itself each time it lands. Such a vehicle concept, known as a NIMF[5,6] (for Nuclear rocket using Indigenous Martian Fuel) is illustrated in Fig. 1.

The high molecular weight of CO2, while very detrimental to specific impulse, allows for a much higher thrust to be generated by a NIMF engine operating at a given power level than a conventional hydrogen fed NTR of the same power. Assuming a propellant temperature of 2800 K, a specific impulse of 264 s can be obtained with a nozzle expansion ratio of 100. Such a performance would give the NIMF the capability of attaining a 250 km by 33000 km (250 by 1 sol) elliptical orbit about Mars. However, even a modest propellant temperature of 2000 K would still give it the more important ability to hop from one point on the surface of Mars to any other point in a single hop. Because CO_2 becomes an oxidizing medium when heated to elevated temperatures, conventional NERVA type carbide fuel elements cannot be used in a NIMF engine. Instead either oxide or oxide coated fuel pellets would have to be used. Uranium-thorium oxide has a melting point of about 3300 K, and such pellets coated with a layer of either zirconium or thorium oxide to retain fission products, could well enable operation at 2800 K. Alternatively, preliminary data[7] indicates that "traditional" NERVA uranium carbide fuel elements coated with graphite can have their graphite coated with a further layer of thorium oxide, and that such thorium oxide outer coatings are resistant to both CO_2 and solid-solid reactions with the graphite at temperatures up to 3000 K. Because the NIMF requires high T/W ratios to take off from the Martian surface, a particle bed geometry for its core is probably the most appropriate choice.

CO_2 can be liquefied out of the Martian atmosphere by simple pump compression at an energy cost of about 85 kWe-hrs/tonne. What this means is that a 45 tonne NIMF with a propellant capacity of 350 tonnes[6], using a 25 kWe power source (either DIPS, deployable solar, or dual mode NTR) can completely fuel itself in less than 50 days, without any dependence on surface infrastructure. Thus in contrast to a chemical bipropellant vehicle, a NIMF engaged in a sortie from a base need only carry enough propellant for a 1-way hop. This reduces the ΔV required for a given sortie in half, and effectively doubles the effective specific impulse of the NIMF relative to a chemical alternative. Since a typical conjunction class stay is about 600 days, the use of the NIMF offers a large increase in the effectiveness of a Mars expedition, since with an average refueling time of 30 days, the astronauts would be able to use the NIMF to visit and explore 20 Martian sites instead of the 1 that an expedition without a hopper would be limited too, or the 2 to 4 that could be afforded (from a base energy and hydrogen resource point of view) by an expedition equipped with a chemical hopper.

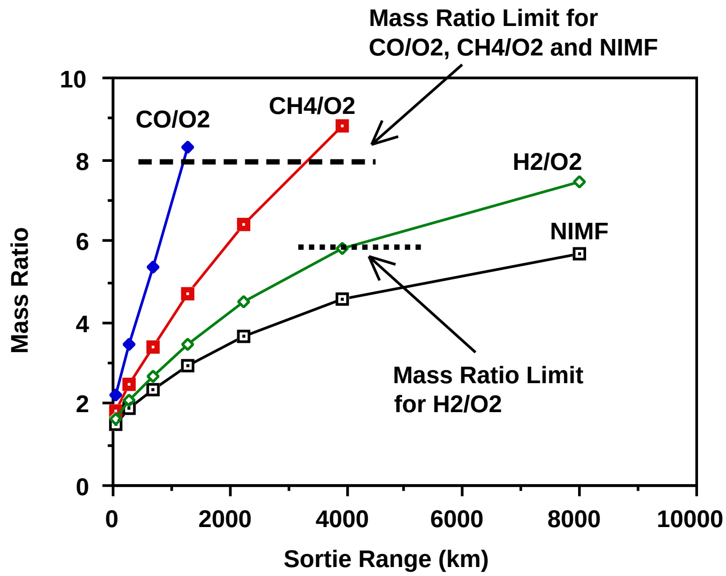

Fig. 2. A comparison of the performance of Mars ballistic hoppers. Mass ratio limits restrict CO/O_2 hoppers to a maximum sortie range of 1200 km, and CH_4/O_2 and H_2/O_2 to 3200 km. The NIMF's sortie range is unrestricted.

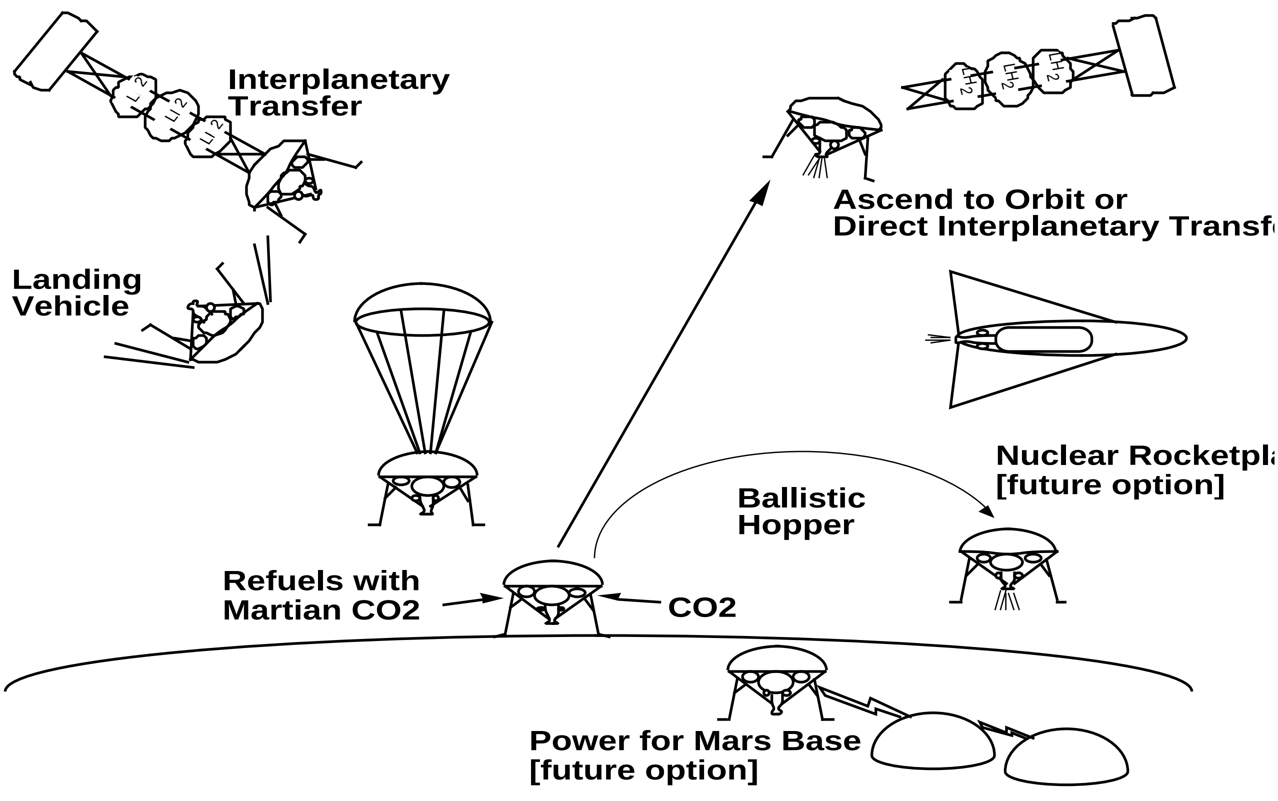

Fig. 1. The NIMF concept in a variety of modes.

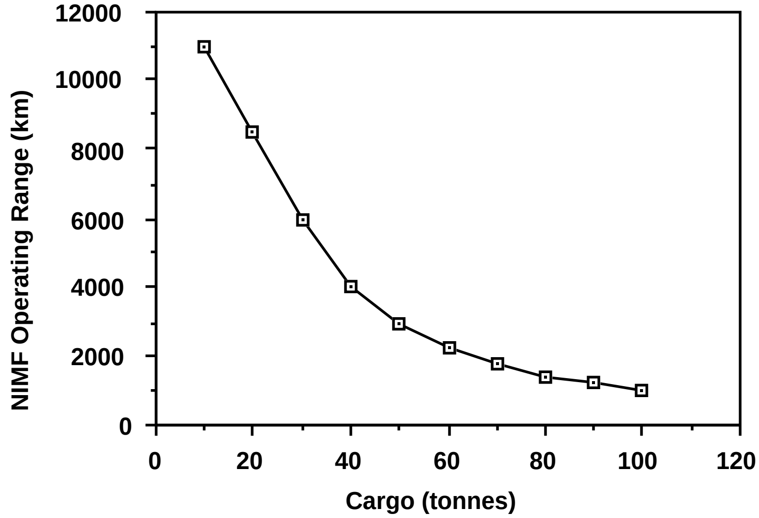

Fig.3.Cargo Delivery Capability of a 45 tonne NIMF with 350 tonne propellant capacity.. No supporting infrastructure is required at either end of the trip.

A comparison of the operating range of all types of Mars ballistic hoppers considered is presented in Fig.2. It can be seen that the NIMF's ability to refuel itself after landing gives it a strong advantage over all chemical alternatives.

NIMF vehicles would also have the capability to transport large amounts of cargo long distances point to point across the Martian surface. The cargo capability of a 45 tonne drymass NIMF with 350 tonne propellant capacity is shown in Fig. 3.

It can be seen that the surface to surface cargo delivery capability of the NIMF is quite large, allowing it to deliver a cargo equal to its own mass approximately 4000 km across the Martian surface, which is roughly the distance from the Martian equator to the poles. A Mars base supported by such a NIMF would thus be able to access

Fig. 4. A Ballistic NIMF on the Martian Surface. From top to bottom is the control deck, habitation deck, CO_2 compressors, main propellant tank, and reactor surrounded by a coaxial propellant tank. (Painting by Robert Murray, Martin Marietta)

resources from all over the planet, which would allow concentration of resource utilization equipment and personel at a single site. This in turn would facilitate development of a large base with sufficient crew size for a significant division of labor and thus the beginnings of real industrial and agricultural capabilities. The pioneering and mastery of the utilization of local resources achieved at such a base will make it the beach-head for the eventual settlement of Mars. This concentration of base infrastructure could be achieved with no sacrifice of science return, as the exploration imperative would be met by NIMF sorties, instead of scattered landings by successive missions from Earth.

A ballistic NIMF on the Martian surface is depicted in Fig. 4.

Ballistic hoppers can also have surface to orbit capability, and thus hoppers using indigenous propellants can be used to reduce the Earth to Orbit mass of manned Mars missions, and thus their cost. Indeed, using anything but indigenous propellant in a Mars Ascent Vehicle (MAV) appears to be rather absurd once transportation costs are accounted. Consider the following: The current cost of shipping material from Earth to Orbit is about $10,000 a kilogram; with chemical propulsion 1/4 of this could be sent to the Martian surface as useful payload. If we factor in the cost of the additional systems required for trans-Mars injection, aerocapture, and landing, an estimate of $50,000 per kilogram emerges as a reasonable minimum cost for Earth-to-Mars transportation. A small MAV typically requires about 40,000 kg of propellant, thus the decision to employ terrestrial propellant for such a vehicle instead of indigenous material *requires spending $2 billion per mission to ship to Mars a massive amount of stuff that is already there.* Bringing coals to Newcastle was never so foolish.

In the long term, ballistic hoppers using indigenous propellant offer the possibility that a Mars settlement may be able to export material products, such as metals extracted from rare minerals that may exist on Mars in concentrated form[8], to Earth for sale at a profit. Such vehicles could deliver the cargo to orbit, after which it could be transferred to an interplanetary vehicle for delivery to Earth. If the interplanetary vehicle was a cycling upper stage driven by a Martian propellant or a solar or magnetic sail[9] spacecraft, no terrestrial propellant would be required. Alternatively the hopper could deliver the cargo to Phobos, where either a gas gun, ram accelerator[10], or electromagnetic[11] or tether catapult[12] device could be emplaced, and used to fire the cargo off

Fig. 5. NIMF rocketplane. Mach 4 flight allows the use modest wings in the thin Martian atmosphere. Four ventral nozzles are allow vertical takeoff and landing using hot CO_2 gas supplied by the reactor. (Painting by Robert Murray, Martin Marietta)

the Earth for capsule aero-entry and marine recovery. In either case, the ascent vehicle most suitable for supporting such a commerce would be the one whose use imposed the smallest tax on the settlement's resources.

Atmospheric Vehicles

Mars' atmosphere, with a density about 1% that of Earth at sea level, is sufficient to enable the use of various types of aircraft.

In the past, the low levels of power available from electric systems served to center the attention of prospective Martian aviators on either lighter-than-air craft[13] or ultra light subsonic aircraft[14] with very high L/D ratios. Subsonic aircraft, with their long delicate albatross-like wings, could never survive re-entry and would have to function as dedicated airplanes, as of course would balloons. Furthermore, the lifting (and thus transport) capability is of both of these types of systems is negligible, and they would be of questionable safety in the high-wind conditions that sometimes occur on Mars, being subject to damage from such storms even when on the ground. Thus while of some interest as methods of deploying drone observation and other reconnaissance instruments, such systems are of doubtful utility as a method of manned transportation on Mars. (An interesting possible form of propulsion for drone reconnaissance craft for use on either Mars or Venus would be piston or jet engines employing H_2/CO_2 combustion. The high propellant energy/mass ratio made possible by such CO_2 atmosphere breathing engines would enable a specific impulse of 600 seconds. The cycle is too wasteful of hydrogen to consider for large manned craft, however.)

Rocket propulsion, on the other hand, provides power levels sufficient to maintain supersonic flight, which is far more suitable to the thin atmosphere of Mars, where flying conditions resemble those found at 75,000 feet on Earth. The required wing area decreases in inverse proportion to the square of the aircraft's velocity, so that in the middle supersonic range (M = 3 to 5) the Martian aircraft becomes transformed into a sturdy delta winged vehicle, perhaps looking somewhat like a Space Shuttle Orbiter (Fig. 5). Such a vehicle could serve double duty, acting as either a Mars airplane or as a Mars/orbit descent and ascent vehicle. The high L/D ratio available to such a vehicle would also reduce both heat loads and g-loads of orbital re-entry compared with that encountered by a purely ballistic vehicle, as well as providing astronauts with much greater control in choosing a landing spot.

A rocketplane used for Mars local transportation would be inherently far more versatile and maneuverable than a purely ballistic hopper. A winged aircraft can take off and then turn back and land at its home base; once a ballistic hopper takes off it is committed and has much less freedom to choose the time and place of its landing. For most flight missions of intermediate range (400-2000 km) the winged vehicle uses less fuel than a ballistic counterpart of the same mass. Of course, a ballistic vehicle of a given capability is likely to have less mass than a winged vehicle, as its structure is much simpler, and this gives a ballistic vehicle the edge over a rocketplane for very long distance flights or ascent to high orbits. However, as the lift to drag ratio (L/D) of a winged rocketplane rises, the rate of rocket propellant consumption required to maintain level flight drops in inverse proportion.

Among the rocketplanes, one propelled in the manner of a NIMF would enjoy a qualitative advantage, as at a certain point, a winged NIMF utilizing CO_2 propellant would become capable of in-flight refueling, either through direct jet gas intake or through in-flight gas collection and liquefaction. The first option would give the vehicle infinite aerodynamic cruising range, while the second would allow for a drastic reduction in ground lift-off weight for an ascent to orbit. The performance of all rocketplane options, however, can be improved through a simple jet intake augmentation of rocket thrust.

The equations governing a rocketplane in level flight are:

$$Mg = L \tag{3}$$

and

$$MdU/dt = T - D = -(dM/dt)c - D \tag{4}$$

Here U is the forward flight velocity, M is the aircraft mass, c is the rocket exhaust velocity, T is the thrust, L is the lift, and D is the drag. Equation (3) simply states that the weight equals the lift, while (4) states that acceleration equals thrust minus drag. Combining equations (3) and (4) we obtain:

$$MdU/dt = -(dM/dt)c - D(Mg/L) \tag{5}$$

In the special case of a glide, where dM/dt=0, equation (5) reduces to:

$$dU/dt = -Dg/L \tag{6}$$

In the other special case of level flight at constant velocity, (5) reduces to:

$$dM/M = (g/c)(D/L)dt \tag{7}$$

Equation (7) can be integrated and the solution is:

$$M_i/Mf = \exp[(g/c)(D/L)t] \tag{8}$$

where M_i is the mass of the aircraft (including propellant) at the beginning of powered flight, and M_f is the mass after t seconds of powered flight.

Consider a winged NIMF with a dry mass of 40 metric tons driven by CO_2 with an exhaust velocity of 2600 meters per second. The vehicle flies at Mach 4.0 (about 1 km/s on Mars) with a lift to drag ratio of 4. Assuming a reasonable supersonic lift coefficient of 0.25, a wing area of 300 square meters (on the order of that of the Space Shuttle) is required to maintain level flight at an altitude of 4 km at the beginning of the cruise phase when the craft is its heaviest. Mach 4.0 (about 1 km/s on Mars) The L/D ratio is a conservative estimate, allowing for a drag coefficient of up to 0.38 for a cross sectional area of 49 meters squared. Using these values, equation (8) can be simplified to:

$$M_i/Mf= \exp[t/2760] \tag{9}$$

Let us asumme the craft has on board 104 metric tons of fuel at take off, then the rocket equation shows that 50 metric tons will be expended in accelerating the vehicle to 1 km/s, and 6 metric tons will be required to stop and land the vehicle using either vertical thrust ("Harrier style") or tail end nozzles ("X-13 style") after a flare up maneuver has reduced the terminal velocity to about 250 m/s. The remaining 48 metric tons are used for cruising for in level flight. Since M_i/Mf in this case equals 94/46 = 2.04, equation (9) shows that the total time in level powered flight at 1 km/s will be 1972 seconds.

Assuming that the initial acceleration to 1 km/s is at 10 m/s^2, 50 km will be covered during the initial acceleration phase. Equation (6) shows that the average deceleration during the glide phase will be (1/4)(3.77) = 0.94 m/s^2. Thus about 740 seconds will be required to glide-decelerate to a velocity of 300 m/s, after which the terminal landing maneuver will take place. During the glide, the average velocity of the aircraft is about 600 m/s, so up to 480 km can be travelled during the glide phase. The total distance traveled will be thus be 50 + 1972 + 480 = 2502 km. A ballistic hopper with the same dry weight and fuel load would be able to travel about 2100 km, provided it could accomplish most of its deceleration by aerodynamic braking. Such a craft would however be totally committed to its destination shortly after takeoff. If it wanted to have comparable flight plan flexibility to the rocketplane it would need to decelerate using rocket thrust, and its range would only be about 750 km.

However, at Mach 4 at 4 km altitude, the mass flow required for this aircraft to maintain cruise (at its maximum cruise thrust) is only 120 kg/s, and this could be provided by in-flight intake of atmospheric CO_2 through an inlet of 12 square meters. Such an inlet area could easily be obtained by placing a long, narrow slot along the lower surface of the aircraft, and would give the craft infinite cruising range. Larger slots or faster speeds could make possible in-flight acquisition of CO_2 for the storage tanks, which would serve to reduce ground lift-off fuel requirements for an ascent into orbit.

While such jet augmented configurations involve significant increase in engineering difficulty and complexity when compared with a simple ballistic or winged rockets, their potential for improved performance is so great that they are certainly worthy of study.

Propellant Manufacturing Processes

Ground vehicles employing combustion engines might consume 10 to 30 tonnes of propellant in the course of a 600 day conjunction class mission surface stay, while ballistic or winged rocket flight vehicles might consume 30 to 300 tonnes of propellant per flight. The importation of propellant from Earth to Mars to support such activity would be prohibitively expensive. The manufacture on Mars of propellants out of indigenous materials is thus essential if surface systems capable of long range mobility are to be employed.

As discussed above, by far the easiest propellant for local manufacture on Mars is raw CO_2 for use in a NIMF vehicle, which can be produced simply by running a roughing pump to compress Mars' 8 mbar atmosphere to the point of liquefaction at 6 bar (88 psi.)

Of the chemical bipropellant combinations, the manufacture of CH_4/O_2 is easiest to perform, especially if the amount required is small enough that it is practical to import the hydrogen component (5% of the bipropellant by weight) from Earth.

Methane is produced when hydrogen is combined with Martian CO_2 in the Sabatier reaction, so named after the chemist who studied it extensively during the latter part of the 19th century.

The Sabatier reaction is:

$$CO_2 + 4H_2 = CH_4 + 2H_20 \qquad (10)$$

This reaction is exothermic and will occur spontaneously in the presence of a nickel catalyst (among others)[15]. The equilibrium constant is extremely strong in driving the reaction to the right, and production yields of greater than 99% utilization with just one pass through a reactor are routinely achieved. In addition to having been in wide scale industrial use for about 100 years, the Sabatier reaction has been researched by NASA, the USAF, and their contractors for possible use in Space Station and Manned Orbiting Laboratory life support systems. The Hamilton Standard company, for example, has developed a Sabatier unit for use on Space Station Freedom, and has subjected it to about 4200 hours of qualification testing. It is interesting to note that the Hamilton Standard SSF Sabatier units, which use a proprietary Ruthenium catalyst with a demonstrated shelf life of greater than 12 years, are sized to react about 3 kg of CO_2 per day. Each unit is about the size of a can used to contain 3 tennis balls. A battery consisting of 10 such units would be sufficient to produce enough methane to support operations by a ground rover traveling 20,000 km per year.

The fact that the Sabatier reaction is exothermic means that no energy is required to drive it, and this in turn implies that the limiting rate at which it can be made to proceed on Mars is the rate at which the CO_2 feedstock can be provided. This means that any hydrogen brought from Earth to feed this process can be reacted away at a rate much higher than it will boil off, and thus there would be no problem with the long term storage of the cryogenic liquid hydrogen on the Martian surface.

As the reaction (10) is run, the methane so produced is liquefied either by thermal contact with the hydrogen stream or (later on after the liquid hydrogen is exhausted) the use of a mechanical refrigerator. (Methane is slightly less cryogenic than liquid oxygen.) The water produced is condensed and then transferred to a holding tank, after which it is pumped into an electrolysis cell and subjected to the familiar electrolysis reaction:

$$2H_2O = 2H_2 + O_2 \qquad (11)$$

The oxygen so produced is refrigerated and stored, while the hydrogen can be recycled back to the Sabatier reaction (10).

Electrolysis is familiar to many people from high school chemistry, where it is a favorite demonstration experiment. However, this universal experience with the electrolysis reaction has created a somewhat misleading mental image of an electrolysis cell as something composed of Pyrex beakers and glassware strung out across a desk top. In reality modern electrolysis units are extremely compact and robust objects, composed of sandwiched layers of electrolyte impregnated plastic separated by metal meshes, with the assembly compressed at each end by substantial metal end caps bolted down to metal rods running the length of the stack. Such solid polymer electrolyte (SPE) electrolyzers have been brought to an extremely advanced state of development for use in nuclear submarines, with over 7 million cell-hours of experience to date. Testing has included subjecting cells to depth charging and loads of up to 200 g's. Both the Hamilton Standard and the Life Sciences companies have also developed light weight electrolysis units for use on the Space Station. Once again, these units are of adequate capacity to perform the propellant production operation for ground rover application. The SPE units that Hamilton Standard has supplied for use by Britain's Royal Navy have the correct output level to support the propellant production requirements of manned ballistic hoppers or rocketplanes. These units have operated for periods of up to 28,000 hours without maintenance, about 2 times the 600 day surface stay of a conjunction class manned Mars mission. The submarine SPE electrolysis units are very heavy, as they are designed to be so for ballasting purposes. SPE electrolysis units designed for space missions would be much lighter (see below).

If all the hydrogen is expended cycling the propellant production process through reactions (10) and (11), then each kilogram of hydrogen brought to Mars will have been transformed into 12 kg of methane/oxygen bipropellant on the Martian surface, with an oxygen to methane mixture ratio of 2:1. Burning the bipropellant at such a ratio would provide a specific impulse of about 340 s, assuming an nozzle expansion ratio of 100. This amount of propellant mass leveraging would be satisfactory for rover and limited ascent vehicle or hopper use. However the optimum oxygen to methane combustion mixture ratio is about 3.5:1, as this provides for a specific impulse of 373 s and the hydrogen to bipropellant mass leveraging of 18:1.

If this optimal level of performance is to be obtained, an additional source of oxygen must be obtained beyond that made available by the combination of reactions (10) and (11). One possible answer is the direct reduction of CO_2.

$$2CO_2 = 2CO + O_2 \tag{12}$$

This reaction, which can also be used to produce CO fuel for CO/O_2 engine use, can be accomplished by heating CO_2 to about 1100 C, which will cause the gas to partially dissociate, after which the free oxygen so produced can be electrochemically pumped across a zirconia ceramic membrane by applying a voltage. The use of this reaction to produce oxygen on Mars was first proposed by Dr. Robert Ash at JPL in the 1970s, and since then has been the subject of ongoing research by both Ash (now at Old Dominion University), Kumar Ramohalli and K. R. Sridhar (at the Univ. of Arizona), and Jerry Suitor (at JPL)[16]. The advantage of this process is that it is completely decoupled from any other chemical process, and an infinite amount of oxygen can be so produced without any additional feedstock. The disadvantages are that the zirconia tubes are brittle, and have small rates of output so that very large numbers would be required for manned flight vehicle application. (The numbers would not be excessive for rover application only.) Improved yields have recently been reported at the Univ. of Arizona, so the process may be regarded as promising, but still experimental.

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

$$CO_2 + H_2 = CO + H_2O \tag{13}$$

This reaction, which also produces CO, is mildly endothermic but will occur at 400 K, which is well within the temperature range of the Sabatier reaction. It has been shown by Meyer[17] that if reaction (13) is cycled with reactions (10) and (11), the desired mixture ratio of methane and oxygen can be produced with all the energy required to drive reaction (13) provided by thermal heat output from the Sabatier reactor. Reaction (13) can be carried out in a simple steel pipe, making the construction of such a reactor quite robust. The disadvantage of reaction (13) is that in the temperature range of interest it has an equilibrium constant of only about 0.1, which means that in order to drive it to the right it is necessary to both overload the left hand side of the equation with extra CO_2 while condensing out water to remove it from the right hand side. This is certainly feasible, and actually constitutes a fairly modest chemical engineering design problem. However a number of alternatives that are at least equally promising have been advanced. One of the most elegant of these would be to simply combine reactions (10) and (13) in a single reactor as follows:

$$3CO_2 + 6H_2 = CH_4 + 2CO + 4H_2O \tag{14}$$

This reaction is mildly exothermic, and if cycled together with reaction (11) would produce oxygen and methane in a mixture ratio of 4:1, which would give the optimum propellant mass leveraging of 18:1 with a large extra quantity of oxygen also produced that could function as a massive backup to the life support system. In addition, salvageable CO would also be produced that could conceivably used in various combustion devices or fuel cells. If all the CO and O_2 produced is included, the total propellant mass leveraging obtained could thus be as high as 34:1.

Probably the easiest method of obtaining the required extra oxygen is just to take some of the methane produced in reaction (10) and pyrolyze it into carbon and hydrogen.

$$CH_4 = C + 2H_2 \qquad (15)$$

The hydrogen so produced would then be cycled back to attack more Martian CO_2 via reaction (10). After a while a graphite deposit would build up in the chamber in which reaction (15) was being carried on. (This reaction is actually the most common method used in industry to produce pyrolytic graphite.) At such a time, the methane input to the reactor would be shut off and instead the chamber would be flushed with hot CO_2 gas. The hot CO_2 would then react with the graphite to produce CO, cleaning out the chamber in the process.

$$CO_2 + C = 2CO \qquad (16)$$

Such a plan, incorporating two chambers, with one carrying out pyrolysis while the other is being cleaned and producing CO, has been suggested to the author as the simplest solution to the extra oxygen problem byf J. McElroy and his research group at Hamilton Standard.

The Hamilton Standard group also provided mass estimates for propellant production systems for both large scale and small scale utilization based on a system combining reactions (10), (11), (15), and (16). The estimates are given in Table 5.

Table 5. Hamilton Standard Mass Estimates for CH_4/O_2 Plant

Reactor	Small Scale	Large Scale
Sabatier	36 kg	164 kg
Electrolysis	90 kg	477 kg
Pyrolysis	105 kg	450 kg
Total	231 kg	1091 kg
Requirement	3.6 kg/day	360 kg/day
Capability	7.2 kg/day	540 kg/day

The mass estimates in Table 5. assume 2 complete units each with 100% mission capacity for the system used in the small scale application, and 3 units each with 50% capacity for the system employed for the large scale application. The reason for the different approach to redundancy on the two systems is that the small scale units have essentially the same mass whether they are full or half capacity, while the large units scale in a roughly linear manner with capacity.

In summary the methods required to produce CO_2, methane, oxygen and carbon monoxide propellants on Mars are well understood and already in an advanced state of technology development. It has suggested in some quarters that while these propellant production processes are promising, they should be relegated to inclusion in downstream missions, with the initial set conducted using only terrestrial propellants for surface mobility, ascent, and Earth return. This hardly seems appropriate, as these propellant production methods are in a more mature state of development than nearly everything else associated with manned Mars missions. Moreover, the carrying out of an initial set of manned Mars missions without the leverage afforded by in-situ propellant production would require a different set of vehicle hardware, and a massive launch and orbital infrastructure that would be very costly and later prove unnecessary. Furthermore, if much in the way of useful surface exploration is to be accomplished, the in-situ process will be needed anyway - and on the very first mission. Since we can have the propellant production process right from the start of the manned Mars exploration program, and since we must have it if useful surface exploration is to be done, we might as well take full advantage of it and use it to provide the Mars ascent and Earth return propellant as well.

Conclusions

This paper has discussed alternative methods for enabling human explorers to achieve long range mobility on the surface of Mars. It is found that it is highly advantageous to use combustion engines to power Martian ground vehicles, and that that their utilization creates a strong incentive for the manufacture of propellants on Mars out of indigenous materials. The use of ballistic or winged flight vehicles allows for a further increase in surface mobility by more than an order of magnitude. The economics of space transportation dictate that such vehicles can only be used if their propellants can be produced locally. The most practical surface and flight vehicles are those whose propellants are easiest to manufacture. For surface rovers this criterion indicates that CH_4/O_2 may be the optimum propellant. In the case of flight vehicles, those employing raw CO_2 propellant heated by a nuclear reactor enjoy a strong advantage over competing systems. In addition to enabling complete global mobility for Mars exploration, such systems allow for the ready transportation of cargo point to point on Mars, and from the surface of Mars to orbit. The latter capability can make possible the development of industry on Mars with global access to Mars' resources, as well as export of useful product from a Mars settlement to Earth. In doing so, such vehicles provide the enabling technology for the creation of human civilization on the Red Planet.

Acknowledgments

The list of sites for Mars exploration given in Table 1 was provided by Owen Gwynne, of NASA Ames Research Center. The paintings were done by Robert Murray, of Martin Marietta Astronautics.

References

1. D. H. Scott and M. H. Carr, "Geologic Map of Mars," U.S. Geological Survey, 1978.

2. C.P. McKay and T. R. Meyer, "Architectures for the Scientific Exploration of Mars," AIAA 91-0327, 29th Aerospace Sciences Conference, Reno, NV, Jan. 1991.

3. R. M. Zubrin, D. A. Baker, and O. Gwynne, "Mars Direct, A Simple, Robust, and Cost-Effective Architecture for the Space Exploration Initiative," AIAA 91-0326, 29th Aerospace Sciences Meeting, Reno, NV, Jan 7-10, 1991.

4. R. Bate, D. Mueller, and J. White, "Fundamentals of Astrodynamics," Dover Publications, New York, 1971

5. R. Zubrin, "Nuclear Rockets Using Indigenous Martian Propellants," AIAA 89-2768, 25th AIAA/ASME Joint Propulsion Conference, Monterey, CA, July 1989.

6. R. Zubrin, "The Engineering of a Nuclear Thermal Landing and Ascent Vehicle Utilizing Indigenous Martian Propellant," AIAA 90-1995, 26th AIAA Joint Prop. Conf., Orlando, FL, July, 1990.

7. J. Powell, Private communication, July 1990.

8. B.M. Cordell, "A Preliminary Assessment of Martian Natural Resource Potential," AAS 84-185, presented to the Case for Mars II Conference, Boulder Co, July, 1984.

9. R. M. Zubrin and D. G. Andrews, "Magnetic Sails and Interplanetary Travel," AIAA-89-2441, AIAA/ASME 25th Joint Propulsion Conference, Monterey, CA, July 1989. Published in Journal of Spacecraft and Rockets, April 1991.

10. A. P. Bruckner and A Hertzberg, " Ram Accelerator Direct Launch system for Space Cargo," IAF-87-211, 38th Congress of the IAF, Brighton, United Kingdom, Oct. 1987.

11. H. Kolm and P. Mongeau, "Basic Principles of Coaxial Launch Technology," IEEE Trans.on Magnetics, Vol. Mag-20, No. 2, March 1984.

12. D. Baker and R. Zubrin, "Lunar and Mars Mission Architecture Utilizing Tether-Launched LLOX," AIAA 90-2109, 26th AIAA/ASME Joint Propulsion Conf. Orlando, Florida, July, 1990.

13. M. B. Clapp, "Dirigible Airships for Martian Surface Exploration," pp.489-496,The Case for Mars II, 1984.

14. A. Colozza, "Preliminary Design of a Long-Endurance Mars Aircraft," AIAA 90-2000, 26th AIAA/ASME Joint Propulsion Conference, Orlando, Florida, July, 1990.

15. R.L. Ash, W. L. Dowler, and G. Varsi, "Feasibility of Rocket Propellant Production on Mars," Acta Astronautica, Vol 5. pp.705-724, Pergamon Press, 1978.

16. K. Ramohalli , E. Lawton, and R. Ash, "Recent Concepts in Missions to Mars: Extraterrestrial Processes," Journal of Propulsion and Power, Vol.5, Num. 2, pp 181-187, Mar-April 1989.

17. T. Meyer, Private Communication, Sept. 1990.