AIAA'87

AIAA-87-1901

The Ballistic Mars Hopper: An Alternative Mars Mobility Concept

J. C. Sercel and J. J. Blandino
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

K. L. Wood
California Institute of Technology
Pasadena, California

AIAA/SAE/ASME/ASEE 23rd Joint Propulsion Conference

June 29-July 2, 1987/San Diego, California

The Ballistic Mars Hopper: An Alternative Mars Mobility Concept

Joel C. Sercel*
John J. Blandino*
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

Kristin L. Wood**
California Institute of Technology
Pasadena, California

Abstract

The ballistic Mars hopper is proposed as an alternative mobility concept for unmanned exploration of the martian surface. In the ballistic Mars hopper concept, oxygen and carbon monoxide produced from the martian atmosphere are used as propellants in a rocket propulsion system for an unmanned vehicle on suborbital trajectories between landing sights separated by distances of up to 1000 km. This mobility concept is seen as uniquely capable of allowing both intensive and extensive exploration of the planet using only a single landed vehicle massing approximately 2000 kg. The technical challenges associated with In-Situ Propellant Production (ISPP) on the surface of Mars are reviewed. A rocket propulsion subsystem capable of using oxygen and carbon monoxide as propellants is described. Finally, results of mission analysis and a hopper landing hazard simulation are reported. It is concluded that an attractive Mars hopper can be developed based on relatively near-term technology.

I. Introduction

This paper describes the results of a preliminary investigation into the feasibility of a ballistic hopper for use in unmanned exploration of the planet Mars. The ballistic hopper is a mobility concept that uses rocket propulsion to transport a vehicle and scientific payload through multiple suborbital ballistic hops covering large distances in relatively short time periods. As shown in this paper, the gravity and atmosphere of the planet Mars make ballistic mobility an ideal form of surface transportation on the red planet. The moderate martian gravity allows for relatively long-range hops (up to 1000 km per hop) with reasonable rocket propellant mass requirements. The atmosphere of Mars is ideal for ballistic mobility because it can be used as feedstocks for in-situ propellant manufacturing and is dense enough to be used in aeromaneuvering or aerobraking to reduce propellant requirements, but is not so dense as to introduce excessive drag losses during launch.

*Member Jet Propulsion Laboratory Technical Staff
**Graduate Research Assistant

Scientific Objectives

Though study of mission scientific objectives was not a primary focus of this investigation, a brief discussion of scientific objectives is appropriate to outline the motivation for investigating the ballistic Mars hopper. From a scientific return perspective, the Mars hopper can be viewed as a lander capable of investigating multiple widely-spaced sites. While traversing the distance between sites, the hopper can accomplish imaging and atmospheric soundings from a variety of altitudes and locations. At each site, the hopper can deploy independent meteorologic and seismic instruments making possible the acquisition of simultaneous data from widely remote locations, thus enhancing the scientific value of such investigations. As such, the ballistic Mars hopper will allow long-distance martian exploration involving both extensive and intensive scientific capabilities. In contrast, a wheeled or legged roving vehicle could not provide atmospheric or imaging data from high altitude and could not investigate such diverse and widely spaced sites.

To assess the desirability of ballistic mobility from the perspective of planetary science return, the value of long-range mobility capabilities must be known. While a careful study of the scientific benefits of long-range mobility was considered beyond the scope of this investigation, a few observations were made which suggest that long-range mobility will be a desirable attribute for an unmanned Mars mobility system. For example, with a land area of 72,000,000 km^2, the planet Mars has roughly the same land area available for exploration as the planet Earth (because Earth's surface is largely covered by water). The geology of Mars is as rich and varied as any planet in the solar system with points of scientific interest scattered widely over its surface. Like Earth, the martian surface has been shaped by endogenic processes such as volcanism, atmosphere-driven erosion, and apparently by the presence of flowing water. Unlike Earth, the effects of meteoroid impact cratering on Mars are plainly visible on a global scale.[1] Orbital observations suggest a possible history of mantle motions on Mars, but unlike Earth, Mars lacks continents.[1] Today's martian atmosphere is too diffuse to allow the existence of liquid that volatile materials frozen into the surface may be a key to understanding the martian climatological history.[1] To understand all these phenomena, it is clear that martian exploration will require a mobility system that allows detailed site investigation at multiple, widely separated locations.

System and Mission Concepts

Three system concept options were investigated: a hopper only, a hopper with a sample return capability, and a hopper with a sample return capability and a deployable rover vehicle. In all three concepts, the hopper and its payload with a minimal initial propellant loading are assumed to be delivered to the martian surface by a conventional spacecraft system. A detachable aeroshell will probably be required to protect the system from the intense thermal environment encountered while entering the atmosphere at orbital velocity. Once the hopper has been safely landed on the martian surface at the first site, it will deploy its radiators and begin local scientific investigations. Depending on which system concept is used, samples from the local environment can be taken and/or a rover vehicle deployed to accomplish a localized rover mission and collect samples. At the same time, the hopper's in-situ propellant manufacturing facilities are activated and propellant is produced from the martian atmosphere. After a period of several months, the hopper's propellant tanks are full enough to accomplish a ballistic hop to a second landing site that has been previously selected on Earth.

Deployed scientific instruments and radiators are retrieved before launch. At lift-off, the main propulsion system provides a maximum acceleration of approximately twice the gravitational acceleration on the martian surface. Immediately after lift-off, the deployable landing struts are retracted to minimize atmospheric drag. At an altitude of about 50 km, the propulsion system is shut down, burnout velocity is achieved, and the effects of atmospheric drag become negligible. A suborbital ballistic trajectory is followed for 5 to 15 min before reentry begins at an altitude of approximately 50 km. It is reasonable to assume that imaging can be conducted throughout the trajectory and that atmospheric sensing can be accomplished during both ascent and descent.

A two-stage parachute system is proposed to slow the spacecraft following the initial reentry process. The first-stage parachute is a relatively small (less than 5 m diameter) high altitude supersonic drogue to aid in deceleration from supersonic velocity. After the vehicle has transitioned to subsonic velocities, a larger (approximately 40 m diameter) second-stage parachute is deployed. Second-stage parachute deployment occurs at an altitude of about 5 km to reduce terminal velocity. This reduction in terminal velocity is desirable to decrease required landing delta-V and to increase loft time for maximum atmospheric sensing and surface imaging scientific return.

Terminal velocity under the second stage parachute is approximately 50 m/s. While the second stage parachute is deployed, landing struts are extended. A radar altimeter or similar sensing instrument is used to measure altitude above ground level. At a few hundred meters altitude the parachute system is separated and the propulsion system activated for terminal decent. It is assumed that a soft landing condition with a touchdown velocity of approximately 1 m/s can be accomplished. This will allow inertial sensors on the vehicle to be used to sense a destabilized condition such as would be caused by landing on an excessively large boulder or slope. If such a destabilized condition exists, thrust level can be increased and the vehicle's attitude control system used to recover the attitude of the spacecraft before attempting a second touchdown. A total landing delta-V allotment of 200 m/s has been assumed in this study to account for such a descent strategy.

This overall procedure can be repeated several times over the course of a 2 to 3 yr mission allowing several landing sites to be studied. At the end of the nominal mission, if a sample return capability is included in the system design, a few kilograms of collected samples and their sample maintenance equipment can be delivered to a low Mars orbit. This is accomplished with a minimal reduction in overall system mobility by using the main hopper propulsion system to deliver the sample to orbit after all the suborbital hops have been completed. In concept, the sample, its container, the propulsion system, and a simple avionics system are separated from the hopper vehicle and become an autonomous launch module. Without the added mass of the aerostructure, the propellant production facility, and the scientific instruments, the launch module is capable of delivering the sample and its container to low Mars orbit. Once in orbit, the launch module acts as the passive partner in a rendezvous maneuver with a sample return vehicle. The sample is transferred to the sample return vehicle and is returned to Earth orbit for study.

The spacecraft configuration concept for the hopper vehicle is shown in Fig. 1. This configuration concept reflects attention to several unique requirements placed on a ballistic Mars hopper. For example, the aerodynamic structure must provide minimum drag in launch mode and maximum drag in reentry mode. A typical reentry velocity for this system is 1 km/s, corresponding to a calculated Mach number of approximately 4.5. This transient peak velocity suggests that a reentry thermal control system can be designed based on a heat shield composed of standard high-temperature metal alloys. Not shown in the figure is the cradle possibly used to carry a separate rover vehicle with the Mars hopper. Table 1 gives an estimated mass

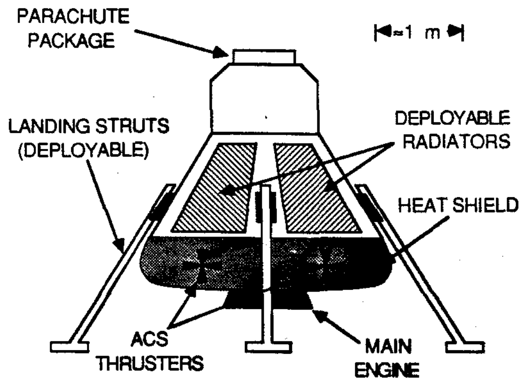

Fig. 1. Hopper configuration concept

Table 1 Hopper system estimated mass summary

ITEM		MASS, kg
PAYLOAD		775
Rover Vehicle	500	
Rover Cradle	100	
Science	50	
Avionics	25	
Communications	50	
Samples and Container	50	
PROPULSION SYSTEM (DRY)		1100
Propellant Tanks & Feeds	70	
Main Engine	210	
Attitude Control Thrusters	20	
ISPP Unit	650	
RTG Power Supply	150	
STRUCTURE		350
Landing System	50	
Parachute System	50	
Aerostructure	250	
TOTAL		2225 Kg

summary for one of the Mars hopper system concepts studied here. As can be seen in Table 1, total system dry mass for this vehicle is expected to be less than 2500 kg, including rover and cradle. As such, total mass delivered to the surface of Mars is not appreciably different for this system than for the Mars Sample Return mission which has been studied at the Jet Propulsion Laboratory (JPL) and the Johnson Space (JSC) Center.[2]

II. Propellant Production

The key to a practical ballistic mobility system for unmanned surface exploration of Mars is the production of usable rocket propellants from indigenous feedstocks. A number of material sources exist on the planet which in principle, could be used to provide the feedstocks for propellant manufacturing. The system concept adopted for this study assumes the production of carbon monoxide and oxygen from carbon dioxide present in the martian atmosphere. The carbon monoxide and oxygen are then stored as cryogenic liquids and burned together in a rocket engine to produce thrust. The following is a brief description of how such a propellant production facility might function. The details of this system are somewhat speculative because the technology of In-Situ Propellant Production (ISPP) is still being developed. The purpose of this discussion is not to advocate the suggested scheme as the only option for propellant production on Mars, but rather to suggest that if feasible, this approach to production of rocket propellant from the martian atmosphere makes the hopper concept attractive.

A simplified block diagram of the adopted ISPP system is shown in Fig. 2.[3] Before describing this system, a brief discussion of its key components is in order. At the heart of this system is the zirconia electrolyte oxygen cell. This device allows oxygen gas to be separated from a gaseous mixture of carbon dioxide, carbon monoxide, and oxygen. The cell functions by ionic conduction of oxygen through a zirconia (ZrO_2) membrane. Specifically, when a voltage is applied to the membrane in the presence of a gas mixture containing oxygen, oxygen in the mixture is reduced to $O^=$ ions at the cathode. These $O^=$ ions then migrate to the anode where they surrender their excess electrons and reform to oxygen gas. The electrochemical theory behind the operation of the zirconia membrane in this device was developed by Nernst and is well understood. Only recently, however, has it been demonstrated that thermal dissociation of CO_2 can provide sufficient oxygen in a gas mixture at the cathode to allow a zirconia cell to produce pure O_2.[4]

Another important component of the ISPP system assumed in this study is the molecular adsorption gas compressor. This device operates on the principle that at low temperature, certain gases can be adsorbed onto

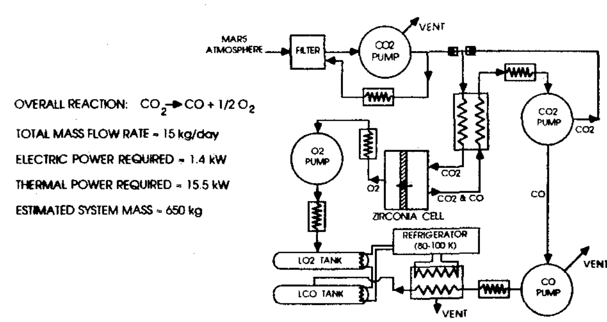

Fig. 2. ISPP system block diagram

appropriate high surface area materials. When the temperature of the material is subsequently elevated, the gas is desorbed. If the desorption is accomplished in a closed container, extremely high gas pressures can be obtained. By containing the high surface area material in an appropriately valved container, an adsorb-desorb cycle can be used very effectively to pump the gas. Such a pumping cycle is shown in Fig. 3.

Because these systems operate effectively with simple check valves as the only moving part, their potential reliability is high.[5] Also, the pumping cycle can be accomplished using low quality waste heat such as might be produced in an Radio-Isotope Thermal-Electric Generator (RTG) electric power supply at a maximum temperature of about 600 K. In the design of a sorption pump it is important to carefully select the solid, high surface area material according to its suitability with the gas to be pumped. For example, both zeolite and activated carbon have been proposed for use with CO_2 gas, while nitrogen and hydrogen have been effectively pumped using metal hydride materials.

Now that the critical components of the ISPP technology assumed here have been introduced, the overall system operation depicted in Fig. 2 can be outlined. The martian atmosphere, which is about 96% CO_2 and 4% trace gases such as argon and nitrogen, is introduced into an adsorption compressor at the ambient temperature of approximately 200 K and a pressure of about 7 mBar. The CO_2 adsorption compressor outputs the gas at a temperature of 600 K and a pressure of about 1 Bar. A back-flush capability is provided to clean the filter periodically. Because the trace gases are not affected by the adsorption process appropriate to CO_2, they can be vented at this first adsorption compressor.

The adsorption compressor delivers the CO_2 to the zirconia cell where it is heated to approximately 1000 K using high quality heat from the hot side of an RTG power supply. At this high temperature some of the CO_2 dissociates allowing oxygen to be pumped through the zirconia membrane which is maintained at a temperature of 1250 K. After being expelled from the zirconia cell the oxygen is cooled in a multistage process involving oxygen adsorption compressors and Joule-Thompson valves. Finally, the oxygen is deposited in a storage tank as liquid at 90 K.

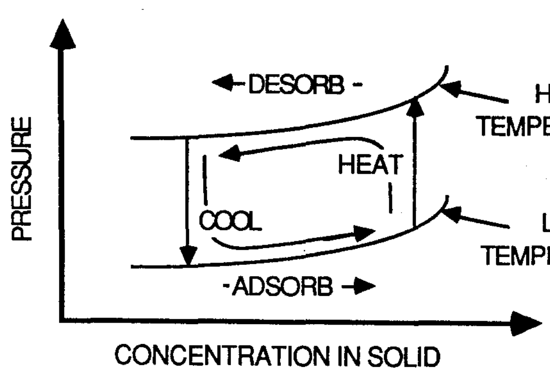

Fig. 3. Adsorb-desorb pumping cycle

Meanwhile the CO_2-CO mixture which is expelled from the cathode side of the zirconia cell at 1000 K is passed through a fin and tube counterflow heat exchanger where it is used to preheat the incoming CO_2 flow from the first CO_2 compressor. After being cooled additionally in a radiator heat-exchanger, the CO_2-CO mixture is introduced to a second CO_2 adsorption compressor. This compressor separates the CO_2-CO mixture. The CO_2 is heated and returned to the cathode side of the zirconia cell at 1 Bar pressure while the CO is cooled and introduced into a CO compressor.

The CO compressor is used to operate a Joule-Thompson refrigerator which serves two functions. First, a fraction of the CO is liquefied and stored at 80 K in propellant tanks for fuel. Second, the remainder of the CO is used as an expanding coolant for propellant tank thermal control as well as cooling various other components of the ISPP unit before being expelled to the ambient Mars atmosphere. The fraction of the CO which is expelled in this manner depends upon the oxidizer-to-fuel (O/F) ratio required by the propulsion system. This system can be modified to produce oxygen and carbon monoxide at oxidizer to fuel mass ratios varying from near stoichiometric (O/F = 0.571) to any oxidizer-rich mixture required by the propulsion system.

The mass and power estimates given in Fig. 2 for this system are approximate and are based on analysis conducted at JPL.[8] To obtain more precise estimates of these values, future study and experimental research will be required. In addition, before this concept can be used, many technology development issues must be addressed. It is stressed, once again, that this system is suggested only as an example of how an autonomous ISPP unit could be designed for oxidizer and fuel production on Mars. It is quite possible that future studies will discover concepts with significant advantages relative to the one described here.

The key technological development issues associated with the eventual use of this type of system have been thoroughly described in a recent publication.[3] To paraphrase, key issues include autonomous system control, low pressure-drop filters, zirconia cell life and performance, and effective adsorption pumps for each of the gases required. When these system technology issues are addressed, the critical feasibility issue for this system will be reliability. It is expected that by using redundant components and parallel operation of small components where possible, the reliability issue can be effectively addressed.

IV. Rocket Propulsion Subsystem Design

A prime area of investigation in this study was the performance of a carbon monoxide/oxygen burning propulsion system. It was found that by comparison with other bipropellant liquid fuel rocket propulsion systems, this propellant mixture is not a very high performance option. However, the potential of providing all of the propellant from indigenous feedstocks using feasible technology makes this system quite attractive. Also, in comparison with the performance of solid propellant or monopropellant rocket engines, the performance of a carbon monoxide based propulsion system it quite good.

A brief description of the engine performance calculations conducted as part of this study is given here. As a first iteration, various oxidizer to fuel ratios were used with the assumption of complete combustion. This allowed an approximate calculation of adiabatic combustion temperature as a function of oxidizer to fuel ratio. Given this information, it was determined that stoichiometric operation with this propellant is not feasible because of excessively high combustion chamber temperature. Operation at an O/F ratio of 1.17, however, resulted in an acceptable combustion temperature of 3934 K.

This design point was used in conjunction with the One Dimensional Equilibrium (ODE) rocket performance analysis code developed at the Lewis Research Center. The program was run under the assumption of gaseous fuel and oxidizer input at 300 K and 298 K, respectively, and a combustion chamber pressure of 500 psia. With the more accurate assumption of chemical equilibrium including the effects of dissociation, the computer calculation projected a combustion chamber temperature of 3300 K. The dominant reaction products determined in the computer calculation included CO at a mole concentration of 0.122, CO_2 at a mole concentration of 0.495, and O_2 at a mole concentration of 0.384.

A thrust chamber geometry based on the Space Shuttle Orbital Maneuvering System (OMS) with an exit to throat area ratio of 200 was assumed. At the exit plane of the thruster, a gas pressure of 11.2 mBar was calculated, suggesting that this engine is slightly under-expanded when operating at Mars sea level pressure. System ideal specific impulse at these conditions was determined to be 278 lbf-s/lbm. Total thrust level was then calculated assuming a conical half-angle of 15 deg to approximate nonaxial exhaust flow.

Additional calculations were performed to enable the first order design of fuel and oxidizer turbopumps. A compressor efficiency of 0.75 and a turbine efficiency of 0.80 were assumed in these calculations. Pump inlet pressure was assumed equal to propellant storage pressure or 40 psia. Regenerative cooling was assumed to be accomplished by the fuel which sustained a 17% pressure drop in the cooling jacket before introduction to the combustion chamber. Propellant flow to the gas generator was found to account for a reduction in system specific impulse to 267 lbf-s/lbm. A conservative specific impulse of 260 lbf-s/lbm was used in the system performance analysis portion of this study.

Engine ignition is provided by electrical spark igniters in both the gas generator and the main combustion chamber. Feed lines provide start-up propellant to the gas generator at 40 psia before the turbopump is up to operational pressure. After several seconds of operation, combustion in both the gas generator and the combustion chamber is self-sustaining.

The results of a point design calculation are repeated here to illustrate typical operating conditions for this propulsion system on the Mars hopper. At a thrust level of 37,000 N (8300 lbf) total mass flow rate is determined to be approximately 14 kg/s. Total system turbopump power required is 3317 HP. Gas generator propellant mass flow rate is about 0.5 kg/s. Total propellant mass was determined to be approximately 2500 kg, of which 1152 kg is CO and 1348 kg is O_2. Two cryogenic propellant storage tanks at a storage pressure of 40 psia are required. Propellant storage thermal control is accomplished in part via an active thermodynamic vent system using vapor-cooled shields. Propellant storage tank masses for this design point are determined to be 54 kg for the fuel and 45 kg for the oxidizer not including structure. Main engine mass is determined to be approximately 210 kg. A detailed Attitude Control System (ACS) design was not accomplished, but a mass allotment of 20 kg was made to account for four 200 lbf ACS thrusters distributed evenly on the spacecraft periphery.

The propulsion system described above is unique in two ways: (1) it is based on the use of a unique propellant combination, and (2) it assumes an autonomous restart capability after landing. The development of a carbon monoxide-oxygen propulsion is seen as a straightforward task. The most significant feasibility issue associated with this system will be ensuring a reliable restart capability. This will be especially critical for the main engine in the landing maneuver in which a delayed engine start could result in system failure.

V. Trajectory and Mission Performance Analysis

The performance of the Mars hopper can be modeled in a manner quite similar to that used to model the performance of Intercontinental Ballistic Missiles (ICBMs). Specifically, the hop maneuver can be thought of as encompassing three distinct phases of operation: (1) a launch phase, (2) a ballistic coast phase, and (3) a reentry phase. The geometry of a single hop is depicted in Fig. 4. In this illustration, the thickness of the martian atmosphere and the distance traveled have been exaggerated for clarity. In simplified form, the total

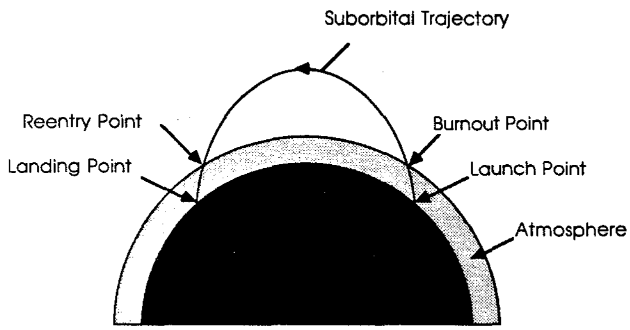

Fig. 4. (Mars hopper trajectory geometry)

trajectory analysis problem for the Mars ballistic hopper can be broken into a set of distinct sub-problems: (a) determine the location of the desired landing site, (b) determine the reentry position and velocity required to achieve that landing site, (c) determine the conditions at burnout required to achieve the reentry conditions, and (d) determine the launch profile required to achieve the burnout conditions. The issues associated with these sub-problems are discussed briefly below as phases of the trajectory analysis problem followed by a discussion of calculated system performance.

Detailed solution of the site selection problem will eventually include consideration of several factors. Foremost among the issues to be considered in site selection will be scientific return potential and lander touchdown hazard considerations. Past studies have shown, for example, that many of the most interesting potential landing sites from a scientific return perspective involve rough terrain on which touchdown for a lander is unsafe. It is expected, therefore, that tradeoffs will be made between safety and scientific interest in selecting a landing site. A study of these tradeoffs can't be conducted without detailed knowledge of the targeting accuracy of the hopper in landing at the desired site. Because of time and resource limitations, hopper landing accuracy was not quantitatively investigated as part of this study. Instead, a simulation of landing hazards for two types of terrain was accomplished as discussed below.

Key problems associated with the reentry portion of the hopper trajectory include thermal control, deceleration loading, accuracy and total braking delta-V. In future, detailed designs of Mars hopper systems, each of these problems must be addressed. At least three design options exist for this phase of the trajectory. First, the rocket propulsion system can be used exclusively to provide braking and accurate maneuvering, thereby solving most the problems associated with reentry. Unfortunately, this option results in a doubling of hop delta-V and would probably render the concept noncompetitive. Second, an aeromaneuvering system could be used to not only dissipate the reentry velocity, but also extend the reentry phase of the trajectory to considerably increase hop range. This is a very attractive option but was not baselined in this study because it was considered to rely on excessively sophisticated guidance and control systems. It is recommended that future studies seriously consider this option.

As a final option, a simple high-drag aerostructure can be used to decelerate the vehicle upon reentry. This option was adopted for this study because it is easy to analyze and requires no technology development. The dominant forces acting on the baselined hopper concept during reentry are atmospheric drag and gravity. For significant deceleration to occur during reentry, the drag forces must be larger than the gravitational forces. To ensure that this occurs, the ballistic coefficient of the vehicle in reentry mode must be made large. The first stage of deceleration occurs when the vehicle is at high altitude (greater than 10 km) and at peak velocity. The vehicle's aerostructure heat shield is used to provide drag during this phase of the trajectory in which the vehicle transitions from approximately Mach 5 to approximately Mach 2. To accomplish controlled deceleration below Mach 2, it was assumed that a two-stage parachute system is used as described in the system and mission section of this paper.

Prior to reentry, the Mars hopper flies in a suborbital trajectory on an elliptic path that intersects the martian atmosphere at the launch burnout point and the reentry point. During this phase of flight the vehicle's motion can be easily described using standard techniques of orbital mechanics. In the trajectory analysis conducted for this study, maximum range trajectories were used for the purpose of calculating burn-out velocity requirements based on the ICBM targeting method of Bate, Mueller, and White.[6] Critical assumptions made in the calculation included neglecting planetary oblateness effects and planetary rotation. Planetary oblateness effects are known to have a minimal effect on trajectories of this duration. Planetary rotation was neglected because no launch and landing site selection was made as part of this study. Actual performance for eastward launches will therefore be higher than calculated here while westward launches will suffer a relative performance penalty in terms of total hop distance. Because hop distance in all cases studied was a small fraction of the planetary diameter, error resulting from neglecting planetary rotation is small.

Because energy is continually added to the hopper trajectory during the launch phase, two-body mechanics can't be used to model the performance of the hopper in the time from launch to burnout. Detailed modeling of this portion of the trajectory requires continuous integration of the equations of motion for the vehicle, including the effects of gravity, atmospheric lift, drag, and variable thrust. Fortunately, this calculation has been done many times in past studies of launch vehicles and rule-of-thumb approximation techniques have been generated to allow simple analytical calculation of launch performance. For this study, it was assumed appropriate to allow for a delta-V loss of 20% to account for gravitational and atmospheric losses in the Mars hopper launch.

The final result of trajectory calculations conducted here is shown in Fig. 5. Residence time required for propellant production is presented as a function of hop distance for a hopper with a rover and for a hopper without a rover. The assumptions made in this calculation were a propellant production capability of 15 kg/day, a specific impulse of 260 lbf-s/lbm, a landing delta-V allotment of 200 m/s, and a launch delta-V calculated as described above. For the system with a rover, total dry mass was calculated to be 2200 kg; the system without a rover required only 1500 kg of dry mass. Both systems were sized for a maximum propellant loading of approximately 2500 kg. Maximum range for these systems is thought to be limited to 750 to 1000 km per hop. Typical residence time required for moderate hop distances of several hundred kilometers is a few months. A simple calculation, therefore, shows that these devices will be able to cover up to a few thousand kilometers while sampling several sites over the course of a two to three year mission. By using hoppers with aeromaneuvering capabilities it is expected that this performance can be significantly improved upon.

VI. Hopper Landing Hazard Simulation

Potential landing hazards represent a critical issue relating to the feasibility and desirability of the ballistic hopper for Mars mobility. Specifically, if the hopper is relegated to landing only in areas of relatively benign geography or if the chances of catastrophic failure on landing are excessive its value to the scientific community may be limited. To address this issue, a simulation computer code was developed in which boulder obstacles in the martian terrain were simulated in a discrete grid model.

Both a nominal terrain model and a second-order terrain model were treated in this simulation. The nominal terrain model was based on a conservative terrain somewhat like that encountered by the Viking landers.[7] The second-order terrain model corresponded to an estimate of one level more complex terrain, such as that found in the foothills of the Rocky mountains. One thousand hopper landings were simulated for both terrain models. In each simulated landing, the location of each of the feet in the hopper's tripod base was compared to the location of boulders in the simulated boulder field. Encounters with boulders larger than 0.1 m were recorded.

The results of these simulations are presented in Fig. 6. The fraction of boulder encounters for boulders ranging in size from 0.1 to 0.8 m is given in bar graph form. To ensure a 0.99 probability of successfully avoiding a hazardous boulder encounter in first-order terrain, a Mars lander must have the capability of successfully touching down on boulders as large as 30 cm in diameter. To achieve approximately the same level of safety in second-order terrain, a Mars lander must be designed to successfully touch down on boulders as large as 60 cm in diameter. Such capabilities can be achieved in a Mars hopper by careful landing system design. Landing hazards, therefore, do not represent a major issue in determining the feasibility of ballistic mobility for Mars surface exploration.

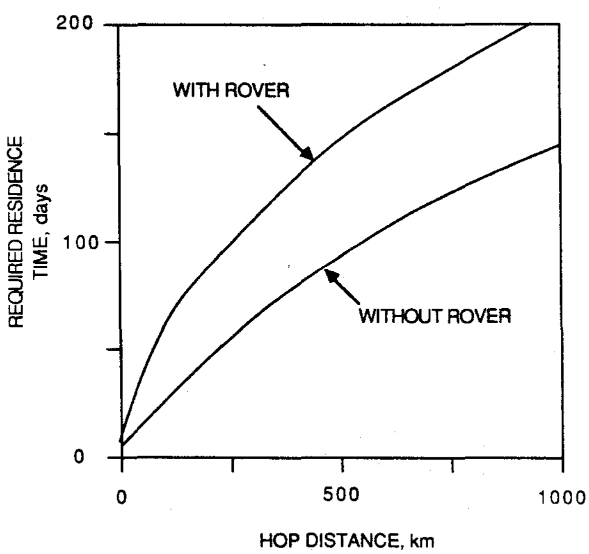

Fig. 5. Residence time required for propellant production as a function of hop distance

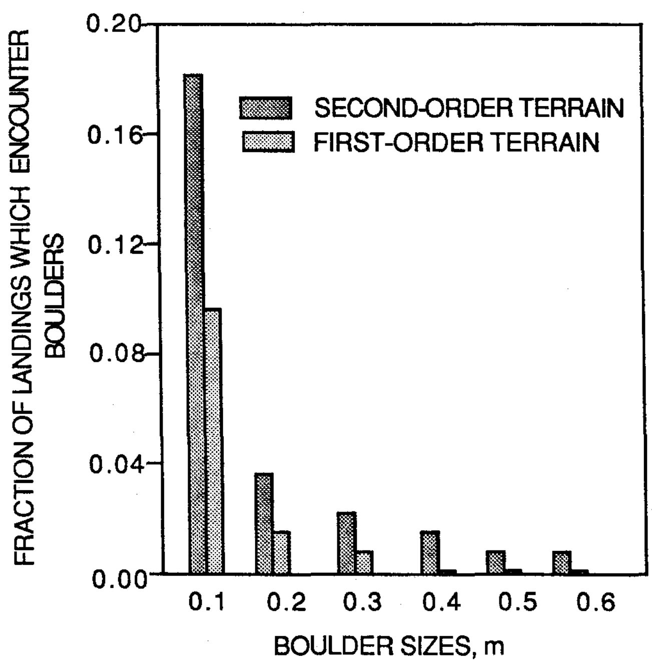

Fig. 6 Fractional rate of encountering various sized boulders on martian surface.

VII. Conclusions

This paper described a first order-study of a unique Mars mobility concept. To the level of detail which this study was able to investigate, the ballistic Mars hopper appears to be an attractive candidate mobility concept for unmanned exploration of the planet Mars. It is expected that an attractive Mars hopper can be developed based on near-term technology. Key technology development areas exist in the propulsion system and the propellant production system. From a scientific return perspective, ballistic mobility will allow a unique capability to accomplish simultaneous extensive and intensive martian exploration. The mass required to be delivered to the martian surface for this system will not differ significantly from the requirements of a Mars sample return mission concept which has been studied by the National Aeronautics and Space Administration.

Acknowledgements

This study was conducted in the spring 1986 academic quarter for Mechanical Engineering 130b, a design course offered at the California Institute of Technology and supported by the National Aeronautics and Space Administration as part of the Advanced Space Design Program, University Space Research Association. Part of the work described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under contract with the National Aeronautic and Space Administration. The authors would like to thank Professor Culick for allowing them to take part in this class and study a non-conventional mobility concept. The authors acknowledge R. Frisbee, M. Appel, and R. Klemetson of the JPL Propulsion Systems Section for valuable assistance in this project.

References

1. J. A. Wood, The Solar System, published by Prentice-Hall, New Jersey, 1979.

2. DeVries, J.P., Mars Sample Return Mission, 1984 Study Report, JPL D-1845, Jet Propulsion Laboratory, Pasadena, California, September 1984. (Internal Document)

3. Lawton, E. A., and Frisbee, R. H., A New Look at Oxygen Production on Mars ISPP, JPL D-2661, Jet Propulsion Laboratory, Pasadena, California, September 1985. (Internal Document)

4. Richter, R., Basic Investigation into the Electrical Performance of Solid Electrolyte Membranes, JPL 82-75, Jet Propulsion Laboratory, Pasadena, California, August 1985.

5. Klein, G. A., and Jones, R. A., "Molecular Absorption Cryogenic Cooler for Liquid Hydrogen Propulsion Systems," AIAA-82-0830, June 1982.

6. Bate, R. R., Mueller, D., D., and White, J., E., Fundamentals of Astrodynamics, published by Dover Publications Inc., 1971.

7. Feasibility of a Mars Multi-Rover Mission, JPL Internal Report JPL 760-160, February 1977.

8. Frisbee, R. H., Mass and Power Estimates for a Martian In-Situ Propellant Production System, JPL D-3648, Jet Propulsion Laboratory, Pasadena, California, October 1986. (Internal Document)