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Mars Rocket Vehicle Using In Situ Propellants

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The Mars In-Situ Propellants Rocket (MIPR) is a small vehicle proposed to fly autonomously on Mars, using in-situ propellant production to manufacture rocket propellant directly out of the Martian atmosphere, thus demonstrating the feasibility of using local resources to "live off the land." It is proposed as a payload on a Mars Surveyor class lander as a reusable "hopper" vehicle. The vehicle explores the Martian surface under rocket power and can repeatedly take off and land, carrying a suite of science instruments over a range of hundreds of meters to several kilometers per hop. The flight demonstration will accomplish a range of technology objectives important to both unmanned probes and to future human missions, including demonstration of a suborbital Mars launch vehicle, demonstrating storage of cryogenic propellants on the Mars surface, demonstration of a pressure-fed cryogenic propulsion system for Mars ascent vehicles, demonstration of a lightweight space engine, and use for the first time of propellants manufactured in situ on another planetary body. In addition to these technology objectives, the MIPR vehicle can carry a science payload that will advance our understanding of the surface and atmosphere of Mars.

Project Description: Mars In-Situ Propellants Rocket

Introduction and Background

A CRITICAL—and enabling—technology for the human exploration of the planet Mars is in situ propellant production (ISPP).[1] ISPP involves the manufacturing of propellants on Mars using indigenous resources as feedstock in the chemical processes. The primary resource on Mars available for ISPP is the atmosphere, which is 95% carbon dioxide (CO_2). This CO_2 can be converted directly into oxygen (O_2) and carbon monoxide (CO), or, with some hydrogen brought from Earth, into O_2 and methane. These propellants fuel the crew's Mars ascent vehicle: propellant mass typically will be 60–80% of the vehicle mass. Analysis of a candidate human Mars mission shows a potential reduction of up to 30% in the Earth-launched mass if ISPP technology is utilized at Mars.[2]

The importance of ISPP is recognized in the NASA Strategic Plan,[3] which states "The Space Science Enterprise missions will also demonstrate the feasibility of utilizing local resources to 'live off the land.'"

A demonstration of ISPP was scheduled for flight on the Mars Surveyor 2001 Lander, where one of the technology payloads was a demonstration of a small-scale oxygen production plant.[4] The 2001 Surveyor lander mission was canceled as a result of the program changes following the loss of the Mars Polar Lander and Mars Climate Observer; however, the flight hardware for this propellant demonstration was built and qualified.

The proposed Mars In-Situ Propellants Rocket (MIPR) vehicle will demonstrate the use of the ISPP technology in a small vehicle designed to fly on a Mars Surveyor class mission. It will also demonstrate a cryogenic propulsion system for Mars ascent vehicles, lightweight space engine technology, and other innovative technologies for both Mars and Earth-based missions.

Mars Hopper: New Vehicle for Mobility on Mars

Mobility on Mars has a high science value. Invariably, wherever a lander may touch down, we will always want to know what is beyond the next ridge, on top of the nearby hill, or just over the horizon. The Mars *Pathfinder* mission (July 1997) convincingly demonstrated the value of mobility on a planetary surface, and even though the *Sojourner* rover crawled at less than half a meter per second and wandered no more than a maximum of 12 m from the lander, the scientific (and public outreach) value of the *Sojourner* rover was incalculable.

Today the concept of surface mobility of Mars is defined by the use of rovers. This challenges that assumption. Surface rovers are limited by terrain and cannot explore many of the most interesting territory on Mars. If a vehicle were to rise above the surface, it could traverse "impassible" chasms and hop over "uncrossable" cliffs.

A valuable surface explorer would be a "hopper" vehicle able to take off and land repeatedly, carrying a suite of science instruments over hundreds of meters per hop. The rocket-powered hopper is designed to achieve the following objectives: 1) refuel itself autonomously for multiple hops by using solar power to react atmospheric CO_2 into oxidizer and fuel; 2) achieve an altitude of several hundreds of meters and traverse a distance of several hundreds of meters during each hop; and 3) carry a suite of scientific instruments to a soft landing at the conclusion of each hop.

Figure 1 shows an artist's sketch of a hopper vehicle taking off from a Mars lander.

Description of Mission

In situ production of methane-oxygen propellant via the "Sabatier" reaction has been proposed for Mars sample return missions and for human Mars exploration.[1,2,5] However, methane-oxygen propellant production requires hydrogen, typically from Earth. Use of a consumable for propellant production will limit the range of a hopper vehicle. For a hopper vehicle without such a limitation, we chose the carbon monoxide/oxygen (CO/O_2) propellant combination, produced by zirconia electrolysis. Because of its low specific impulse ($\sim$250 s), CO/O_2 is impractical as a propellant for an Earth return vehicle for a sample return mission (although it may be usable for the first stage of such a vehicle, where performance is largely insensitive to specific impulse). For the hopper vehicle proposed, however, because this propellant requires no consumable hydrogen for production, CO/O_2 is ideal.

The hopper will be situated on the science deck of a Surveyor-class Mars lander. Once the lander sets down on Mars, the solar arrays will begin to produce power to operate its propellant production plant. The available power will determine the production rate.

Our preliminary designs indicate that the production plant will be at least half of the hopper's dry mass. The distance achieved during a hop is a function of launch angle, quantity of propellants, thrust, and dry mass. For initial planning purposes we have assumed a launch angle of 45 deg to maximize range. As a technology goal, we want

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to demonstrate an engine large enough that it can be scaled up for a Mars sample-return mission,[6] where required thrust is expected to be 1700–2200 N (400–500 lbf). However, it is also important to keep hopper thrust levels low to minimize mass and to allow a soft landing after each hop. For this mission we anticipate engine thrust to be 200–700 N (50–150 lbf). An engine thrust of 335 N (75 lbf) was used for calculations in this paper.

The nature of the hop is determined by the dry mass of the vehicle, the O_2 and CO production rates [measured in standard cubic centimeters per minute (sccm)], and the length of time between hops.

Figure 2 shows typical trajectories for a vehicle flight of 500 m and 1, 2, and 4 km. For these profiles the dry mass of the vehicle was held constant at 18.9 kg. Flight duration varies from 32 s for the 500-m hop to 48 s for the 4000-m hop. The longer hops need a larger amount of propellant and hence require a longer time between flights to produce propellant. The propellant production time ranges from 50 days for the 500-m hop to 155 days for a 4-km hop.[7]

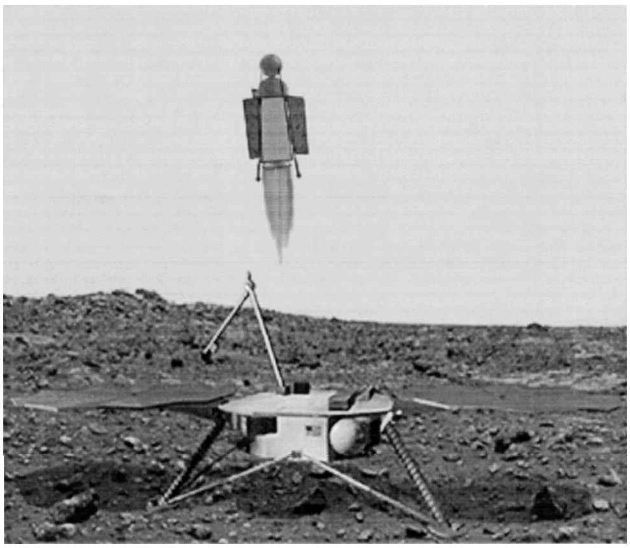

Fig. 1 Sketch of hopper vehicle launching from Mars lander.

A 500-m flight was chosen for the design, using the assumption that more frequent hops (and hence a larger number of surface sites visited) were of higher science value than longer travel distance.

Figure 3 shows how the required propellant mass and time between hops varies for a 500-m flight for the various values of initial (launch) mass. The same 335-N engine was assumed for all profiles; the lighter vehicle actually achieves a slightly higher maximum altitude. Flight duration varies between 32 s for the 20-kg vehicle to 27 s for the 35-kg vehicle.

With a smaller initial mass the production plant is smaller and has a lower fuel production rate. This results in a longer wait between hops. To determine the time between hops, the mass of the propellants, tanks, science, engine, power system, cables, and structure are subtracted from the initial mass. The remaining mass is that which is available for the production plant, which in turn determines the production rate. The time between hops is 50 days for the lightest (20-kg initial mass) vehicle and decreases as the vehicle size increases, reaching 28 days for an initial mass of 35 kg.

An example of the mass allocations is shown in Table 1 for a 30.5-kg initial mass vehicle.

Aerial photographs, first from the lander and then from previous hops, will help determine a direction for each hop. This will allow the vehicle to be directed toward and over interesting scientific sites and to target a relatively benign landing site.

MIPR Flight Demonstration

Objectives

The flight demonstration has several technology objectives: to demonstrate a suborbital Mars launch vehicle, to demonstrate storage of cryogenic propellants on the surface of Mars, to demonstrate a pressure-fed cryogenic propulsion system for Mars ascent vehicles; to demonstrate a lightweight space engine; and to use for the first time propellants manufactured in situ.

The overall flight demonstration consists of both the MIPR demonstration and the end-to-end ISPP demonstration. Together, they have four major components: hopper vehicle, propulsion system, propellant production system, and science payload.

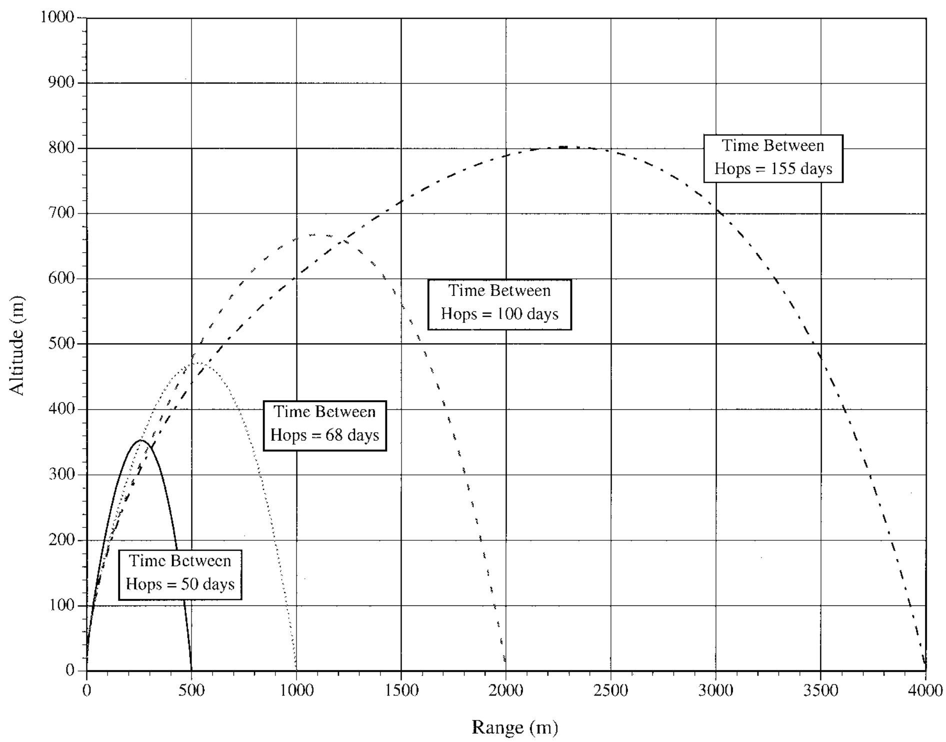

Fig. 2 Trajectory for hop distances of 500 m and 1, 2, and 4 km.

Table 1 Mars In-Situ Propellant Ballistic Hopper: Mass summary[a]

Component	Mass, kg	Subsystem mass, kg	Comments
O tank	0.76	——	2000-psi storage, 15.7-cm diam
CO tank	1.72	——	2000-psi storage, 20.7-cm diam
Subtotal for tanks	——	2.48	All metal or composite wrapped metal tanks; mass est. from Arde
Science	——	1	Pressure, temps, wind for meteorology; aerial photos
Engine	——	3	Pressure fed; based on existing engines
Solar arrays	0.25	——	20 W for 8 h daytime; $\sim$1 m^2 at operating average specific power of 80 W/kg
Solar array structure	0.25	——	Assumed equal to solar array mass
Batteries	0.5	——	Lithium-ion; 24 W-h at 100 W-h/kg assuming 50% depth of discharge
Electronics package	1	——	——
Subtotal for power	——	2	——
Propellant production	——	18.4	Based on scale-up of MIP project (production rate = 30 sccm)
Guidance, navigation, and control	0.5	——	——
Cables	——	0.29	1% of dry mass
Oxygen	0.49	——	——
Carbon monoxide	0.89	——	——
Subtotal for propellants	——	1.38	——
Structure	——	1.52	5% of wet mass
Total vehicle mass	30.52 kg (fueled) 29.0 kg (empty)		

[a]Single hop range, 0.50 km; maximum altitude, 350 m; engine thrust, 335 N (75 lbf); engine specific impulse, 250 s; propellants, O_2/CO; duration between hops, 31 days.

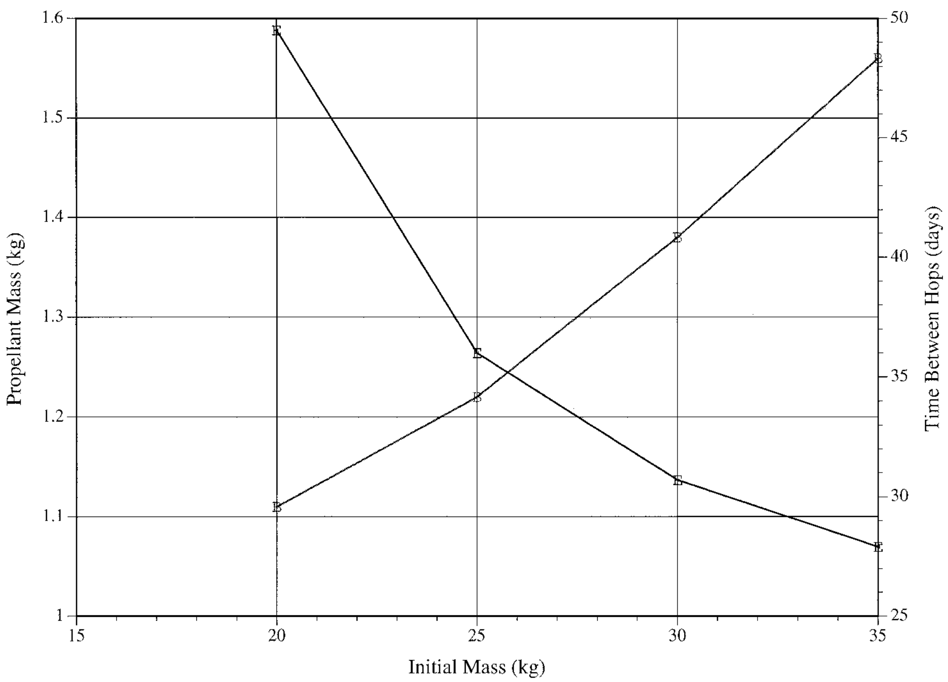

Fig. 3 Example of relationship between initial mass and time between hop, shown for the case of 335-N engine thrust, hop range of 500 m.

Hopper Vehicle

Because the hopper will be reused many times over a long period of time, the vehicle design will emphasize robustness and durability. As a result, vehicle mass may be increased to achieve the desired reliability.

The vehicle structure will be comprised of composites and lightweight metal alloys. The vehicle's physical configuration will be driven mainly by 1) lander packaging, 2) payload requirements, and 3) aerodynamic and mass property considerations. A key feature of the hopper structure will be the landing gear that will 1) have to absorb the shock of landing multiple times on the Martian surface and 2) have to have a reasonable chance of ensuring the hopper ends up in a position that allows another hop.

The hopper's design will also incorporate self-righting capability, using a design similar to that used on the Pathfinder vehicle, where the deployment of solar array panels will automatically set the vehicle into an upright position if adverse terrain at the landing site tips it over. This self-righting capability is shown in schematic in Fig. 4.

Attitude Determination and Control

An inertial attitude determination and control system will be used on the hopper vehicle. Momentum wheels or gyros will be "spun up" using electrical power while the hopper is on the ground. After each hop excess momentum will then be removed from the system by using electrical or electromechanical brakes to slow down the wheels. We will also examine the possibility of integrating the gyro system with energy storage into an attitude control/energy storage combined unit.

Command, Communications, and Data Handling

Communications will be via a UHF link to a relay on either the lander or an orbiting vehicle. Because communication lags between the Earth and Mars make it impractical to control the ISPP plant directly from Earth, autonomous adaptive control software is necessary for the plant to operate over a long period of time. It must be able to handle component failures and chemical process degradation. Examination of Mars mission designs makes it clear that this type of autonomous control will be needed in many different systems.

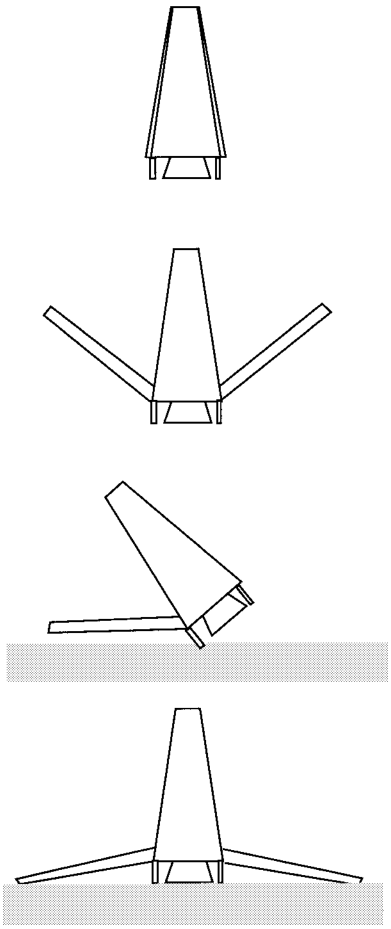

Fig. 4 Deployment of solar arrays: a) launch configuration, arrays retracted against vehicle, b) descent, arrays partially deployed to stabilize attitude, c) self-righting, and d) propellant manufacture, arrays fully deployed.

Power

Propellant production requires a significant amount of power. The power system consists of three main components: solar arrays, storage system, and power management and charge/discharge control. Operation of each element is different on Mars from that on Earth or in conventional orbital applications.

The Mars environment differs in several critical ways from the Earth orbital environment in which space solar arrays normally operate[8]: a solar intensity and spectrum modified by dust, indirect (scattered) sunlight, low temperature operation, deposited dust on the solar arrays, possible high wind, peroxide-rich soil, radiation environment different from Earth orbit, and low atmospheric pressure.[8]

The solar spectrum at the surface of Mars is modified by the atmospheric dust, making it blue deficient and enriched in red and infrared (IR) compared to the orbital ("air mass zero") spectrum. The reduced transmission of short wavelengths will change the technology choice to make materials that respond most to the red and IR more desirable than cells responding to the blue end of the spectrum.

In addition, the Martian solar spectrum varies with season and time of day. Because the dust particles absorb in the blue end of the spectrum but scatter in the red end, the spectral content of Mars sunlight shifts with the amount of dust in the atmosphere and the path length of the sunlight through the atmosphere, shifting toward the red as dust increases and in the morning and afternoon operation. These differences mean that we need a solar cell adapted to the Martian environment. Mission cost considerations also make extremely lightweight array technology desirable.

The redder spectrum of Mars and the low operating temperature tend to favor lower bandgap solar cell technologies, such as silicon, over high bandgap technologies, such as gallium arsenide or dual-junction cells. Constraints caused by shock and g loading of the landing are also a factor, as well as flexure of the arrays caused by wind, if a lightweight flexible array technology is used. This also favors more robust cell technologies, such as silicon.

The target technology for the test vehicle will be advanced, high-performance silicon solar cells (minimum efficiency greater than 20% under space conditions and a target efficiency of 25% under Mars conditions). This solar array type has been developed for high-efficiency terrestrial systems but has never been used in space. It is expected to be significantly more robust and higher in performance than solar cells currently used in space.

Three solar array panels fold out on hinges at the vehicle bottom and are folded against the vehicle for launch. The solar array hinging is shown in Fig. 4.

Power Electronics

Power electronics, required for battery charge and discharge regulation and for power management and voltage regulation, typically have a restricted range of operating temperatures and require a thermal enclosure (or "warm electronics box") to maintain operating temperature. This can be a significant mass element in the power system, as well as an undesirable power load, if electrical heaters are required for temperature maintenance. For this flight vehicle we will take advantage of the NASA John H. Glenn Research Center program in low-temperature electronics to design a power regulation system that will operate at Mars ambient temperatures, with a target of full performance with no degradation down to $-120°C$ (average night temperature on Mars approximately $-100°C$, varying with latitude and season).

Thermal

The hopper will use passive thermal control (material selection, insulation, and coatings) wherever possible. Low-temperature electronic components being developed at NASA John H. Glenn Research Center will also be used where possible. Where this cannot be done, a small thermal enclosure using aerogel insulation will be provided. This enclosure will be heated with traditional resistance heaters.

The sorption pump for propellant production plant will require radiators to reduce the night temperature of the sorption bed; these are accounted for in the propellant production system.

Propulsion System

The propulsion system for the Mars hopper will be a carbon-monoxide/oxygen (CO-O_2)-fueled rocket engine.[6] This engine will be based on propulsion system research conducted at the NASA John H. Glenn Research Center to develop technology to utilize in-situ produced propellants.[9] Research in the O_2/CO propellant combination has focused on ignition barriers and catalysts for ignition,[9,10] heat-transfer characteristics,[11] kinetics, combustion, and thrust performance.

For the Mars hopper propulsion system the O_2 and CO propellants will be produced as gases and then liquefied and stored as cryogenic liquids. To simplify the propulsion system, a pressure-fed system will be used. The liquid propellants will be stored in small spherical tanks of 15-cm diam. Some of the propellants produced will be used to pressurize the tanks for each hop. Before each hop a small amount of oxygen and carbon monoxide would be drained off into evacuated pressurant bottles and allowed to warm up to Mars ambient temperature, thereby increasing pressure to a preset value. This pressurant gas would then be regulated down to tank run pressure for operating the engine. This option has an advantages over using helium pressurization because it makes the vehicle truly independent of Earth resupply by eliminating the need to bring high-pressure helium gas from Earth. After each hop the residual pressurant gas in each tank can be recondensed, thus reducing production requirements. Calculations for various size tanks indicate that approximately 10% of the propellant mass will be required to pressurize the tanks. Once the pressure in the pressurant bottles has reached the desired pressure, the engine sequence will be initiated. The pressurant will be regulated down to a set pressure, and the propellants will flow through a flow-control device (e.g., a venturi) to control flow rate and then be sent to the engine. A spark torch igniter will be used for ignition.[10]

Engine restart and a self-throttling system will also be utilized. During a hop, the engine will fire two times. For the 500-m flight the first burn will last for approximately 6 s and will lift the hopper off the surface into a ballistic parabola. As the vehicle begins to descend and approach the surface, the engines will restart for a second burn. As the quantity of pressurant gas dwindles, the supply pressure will eventually fall below the regulated pressure. The flow rate, and therefore the thrust, will begin to decrease in a predictable manner (based on the venturis used to control the flow rate). Therefore, using appropriate propellant management as the pressurant bottles begin to empty the engine will naturally throttle down for landing. The hopper's legs will also be designed to absorb some force upon landing, through either flexible legs or cushioned feet.

The engine and injectors will be made of lightweight ceramic materials to minimize vehicle mass. The engine thrust is expected to be between 220 and 550 N (50–125 lbf), depending on the final vehicle mass and selected mission profile. Ceramic engines of this size have already been demonstrated, including multistart operation during tests performed at Aerojet in 1994. This technology continues to be refined. Ceramic injectors, however, still need to be demonstrated. These ceramic materials will let the engine operate near peak efficiency without active cooling systems. It is estimated that ceramic engine technology can reduce the mass of current state-of-the-art engines by 20–65%.

ISPP System

The key subsystems for the propellant production plant are 1) a sorption pump to acquire and compress CO_2 from the Martian atmosphere, 2) a zirconia-electrolysis oxygen generation system to strip out O_2 from the CO_2 feedstock, 3) separators to separate CO from the byproduct stream, and 4) a cryo-cooler to liquefy and store the oxygen and CO. Support subsystems include an electronics package to manage fuel production, purge fan to blow inert gas accumulation from sorption bed, batteries for sorption pump power and cryo-cooler operation at night, and radiator panels to radiate waste heat from the sorption pump and cryo-cooler.

The sorption pump and oxygen generation subsystems will be developed based on those developed for the Mars ISPP Precursor (MIP) experiment package on the (now canceled) Mars 2001 Surveyor lander.[12] The cryo-cooler will be developed from ground tests to be conducted as part of the Mars ISPP Systems Technologies breadboard testbed program at NASA Johnson Space Center. The CO separator will be based upon laboratory research previously performed at NASA John H. Glenn Research Center.

The propellant production system is adapted from elements developed to be used on the MIP experiment, and although they represent technology that is flight ready in 2000 it may not necessarily represent what will actually fly.

Science Payload

We have allocated 1 kg for the science payload, sufficient to let us carry out several scientific objectives:

1) Aerial photography of landing site: The aerial view of the landing site will be invaluable for placing geological investigations in a proper context. We will get high-detail images at a different sun angle and from a different physical perspective than the images taken by the descent imager during landing. Thus, our aerial images will complement the science data obtained from other means. These images will also provide aerial reconnaissance for selecting traverse path and locating interesting targets for rover samples.

2) Meteorology: Studies of Martian climate and meteorology will benefit greatly from an expanded range of altitudes for temperature and wind measurements.

3) Vertical profile of aerosols: The aerosols suspended in the Mars atmosphere are a significant climate and meteorology driver; the hopper scientific payload will measure the vertical profile and investigate the change in optical scattering properties of the dust as a function of altitude.

4) Geological measurements at isolated remote sites: Because the vehicle easily traverses obstacles that rovers cannot, we will be able to sample regions that are geologically interesting but too rugged for surface rovers to reach.

Conclusions

A critical, and enabling, technology for a human mission to Mars is in situ propellant production, the manufacturing of rocket propellants on Mars using indigenous resources. The primary resource on Mars available for ISPP is the atmosphere, which is 95% CO_2. This CO_2 can be converted directly into O_2 and CO, which can be used as rocket propellant.

The MIPR is a reusable hopper vehicle, which would use ISPP to accomplish self-refueling, manufacturing rocket propellant directly out of the Martian atmosphere. The MIPR will demonstrate the feasibility of using the Martian atmosphere to make, liquefy, store, and use propellants in a Mars launch vehicle. The MIPR will demonstrate a new method of mobility for Mars. Future science missions could explore large regions of the surface by deploying one or several hopper vehicles.

Individual components of the MIPR will benefit other space missions as well. Demonstration of in-situ propellant production and use by the MIPR will increase confidence in the concept, allowing the possibility of ISPP for Mars sample return at a significantly reduce mass and cost. A future human exploration mission could use MIPR performance and lifetime data to develop a propellant production plant of the size needed for a human mission to Mars.

The technology of lightweight, high-temperature materials for rocket engine chambers and nozzles can reduce engine mass by 20–65%, representing significant mass savings for larger engines used in near-Earth transportation systems. Lightweight cryogenic tanks are critical for planetary missions and offer significant benefits to all upper-stage and Earth-orbit vehicles. A pressure-fed cryogenic propulsion system with gas-gas injection is very simple and reliable. This type of system will minimize moving components and reduce operational complexity. Aerogel-based insulation systems used as the primary insulation on the cryogenic tanks may offer an order of magnitude increase in insulation performance over standard multilayer insulation in the "soft" vacuum range. And robust, high-performance solar array technology, together with low-temperature batteries and electronics, may benefit many applications with low levels of ambient insolation, particularly in applications such as solar-electric propulsion missions and outer solar-system missions.

The MIPR hopper explores the Martian surface under rocket power and can repeatedly take off and land, carrying a suite of science instruments over a range of hundreds of meters per hop. Such a vehicle would accomplish a range of technology objectives important to both unmanned probes and to future human missions, including demonstration of a suborbital Mars launch vehicle, storage of cryogenic propellants on the Mars surface, a pressure-fed cryogenic propulsion system for Mars ascent vehicles, a lightweight space engine, and use for the first time of propellants manufactured in situ.

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References

[1]Linne, D., and Meyer, M., "Technical Prospects for Utilizing Extraterrestrial Propellants for Space Exploration," 42nd International Astronautical Federation, Paper 91-669, Oct. 1991; also NASA-TM-105263, Jan. 1991.

[2]Hoffman, S. J., and Kaplan, D. I. (eds.), "Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team," NASA SP 6107, July 1997.

[3]NASA Strategic Plan 1998, NASA Policy Directive 1000.1, 1998, p. 27.

[4]Kaplan, D., Ratliff, J. , Baird, R., Sanders, G., Johnson, K., Karlman, P., Juanero, K., Baraona, C., Landis, G., Jenkins, P., and Scheiman, D., "In-Situ Propellant Production on Mars: the First Flight Demonstration," 30th Lunar and Planetary Science Conf., Lunar and Planetary Inst., March 1999, Paper 1797.

[5]Sullivan, T. A., Linne, D. L., Bryant, L., and Kennedy, K., "In-Situ-Produced Methane and Methane/Carbon Monoxide Mixtures for Return Propulsion from Mars," *Journal of Propulsion and Power*, Vol. 11, No. 5, 1995, pp. 1056–1062; also AIAA Paper 94-2846, June 1994.

[6]Linne, D., "A Rocket Engine for Mars Sample Return Using In Situ Propellants," AIAA Paper 97-0893, Jan. 1997; also NASA TM-107396, Jan. 1997.

[7]Landis, G., "Solar-Powered 'Hopper' for Mars," Mars Scouts Concept Workshop, NASA Headquarters Mars Program Office, May 2001.

[8]Landis, G., "Solar Cell Selection for Mars," *2nd World Conference on Photovoltaic Energy Conversion*, Vol. 3, edited by J. Schmid, H. Ossenbrink, P. Helm, H. Ehmann, and E. Dunlop, Joint Research Center, European Commission, Rept. EUR 18656 EN, 1998, pp. 3695–3698.

[9]Linne, D. L., "Carbon Monoxide and Oxygen Combustion Experiments: A Demonstration of Mars In Situ Propellants," AIAA Paper 91-2443, June 1991; also NASA TM-104473, Jan. 1991.

[10]Linne, D. L., "Experimental Evaluation of the Ignition Process of Carbon Monoxide and Oxygen in a Rocket Engine," AIAA Paper 96-2943, July 1996; also NASA TM-107267, June 1996.

[11]Linne, D. L., "Performance and Heat Transfer Characteristics of a Carbon Monoxide/Oxygen Rocket Engine," NASA TM 105897, Feb. 1993.

[12]Sridhar, K. R., Gottmann, M., and Baird, R. S., "Update on the Oxygen Generator System for the 2001 Mars Surveyor Mission," AIAA Paper 2000-1068, Jan. 2000.

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