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High-Pressure-Pumped Hydrazine for Mars Sample Return

John C. Whitehead*
Lawrence Livermore National Laboratory, Livermore, California 94551
and
Gerald T. Brewster†
Primex Aerospace Company, Redmond, Washington 98073

A monopropellant single stage can lift geological samples from Mars to orbit. The proposed propulsion system uses previously demonstrated miniature pump-fed hydrazine technology. Loading the propellant just prior to operation avoids the prohibitive structural, thermal, and safety constraints that existed during earlier mission phases. A 100-kg Mars liftoff mass is 86% hydrazine, <6% propulsion hardware, and >6% payload and guidance. The useful mass delivered to Mars can be increased by also using pump-fed hydrazine for landing.

Introduction

THE rocket problem of ascending from Mars has remained unsolved. Conventional technology for spacecraft manveuvering offers no low-risk options, as it does for planetary flybys, orbiter missions, and Mars landing.[1] Progress toward producing propellants on Mars has been widely acclaimed, but it is less applicable to initial small-scale robotic missions. For any propellant choice, the critical unknown is how to build miniature rocket stages having very high propellant fractions and enough thrust, along with trajectory control capability.

Mars departure is more than twice as hard as leaving Earth's moon, with respect to the two fundamental maneuvering parameters (Δv and acceleration). Earth launch vehicles and their stages stand alone in having the necessary capability. Of course, they are many times too heavy to be affordably placed on a Martian launch pad. Therefore, Ref. 1 discussed the application of launch vehicle design principles on a miniature scale. A hypergolic bipropellant engine fed by piston pumps was schematized.

Other possible solutions under consideration also require new advances. Pressure feeding requires engines, structure, and tanks to all be lighter than the state of the art.[2] A combination of ideas from Refs. 1 and 2 would simplify pump-fed bipropellant engine development by driving pumps with heated helium, but mass estimates are uncertain.[3] Solid propellant rocket motors on a sufficiently small scale have very low inert mass fractions, but including lightweight means for trajectory control is a major challenge.

This paper examines another option. Despite its low specific impulse I_{sp}, hydrazine can propel a Mars ascent vehicle (MAV) to orbit with a small payload. Advantages of this monopropellant choice include the availability of a very lightweight thruster, the simplicity of a single-tank vehicle structure, and the ease of fueling on Mars to reduce ascent hardware mass. Moreover, miniature high-pressure-pumped hydrazine (HPPH) systems have already been tested and flown experimentally.[4–6] This experience lends realism to the MAV propulsion design described here.

Single-Stage Advantages

Although staging relaxes propellant fractions, upper stages must be smaller still, and they require connecting structures with separation hardware. Reference 1 quantified this trade and found no strong preference for staging over the avoidance of further miniaturization. Upper stages on a MAV are undesirable for other reasons. A tall vehicle does not fit naturally into a Mars arrival capsule. Loading fuel automatically on Mars and then subsequently disconnecting is more complicated for an upper stage. From a mission perspective, a larger object in Mars orbit is easier to find than just a tiny upper stage. A single-stage vehicle potentially has higher reliability at a lower cost because there is less hardware to design, build, and test.

Considering the extreme propellant fraction required, it is relevant to compare single-stage Mars ascent with Earth single stage to orbit (SSTO). The specific impulse of a hydrazine engine (~230 s) is just over half that of an oxygen-hydrogen one (450 s). Ideal velocities for Mars escape and orbiting are 0.45 those of the Earth, for example, 3600 m/s for a low-altitude orbit. Drag is less in Mars's thinner atmosphere, even with quicker acceleration that reduces gravity losses.

The result of more than halving Δv and not quite halving I_{sp} is that the MAV requires a less extreme liquid fraction than Earth SSTO. Compared with the density of oxygen-hydrogen, hydrazine's nearly threefold higher density permits relatively lighter tanks. The high specific strength of titanium and an efficient spherical shape can lighten the MAV tank even further. Mars ascent does not need wings, wheels, or reentry thermal protection. Overall, a single-stage HPPH MAV is technically less challenging than a reusable Earth SSTO vehicle.

Fueling on Mars

Sending a fully fueled MAV from Earth to Mars would be consistent with conventional spacecraft methodology, but doing so imposes significant constraints unrelated to the Mars ascent flight. Table 1 lists the earlier mission phases chronologically, along with the design impacts of carrying the propellant from Earth in the MAV itself. Qualification to meet all these requirements would be especially costly for a newly developed ultralightweight propulsion system.

Safety is of primary importance during Earth launch operations. It must be established, to a high statistical degree of certainty, that a spacecraft cannot expel toxic fluids by leakage or rupture. It is similarly necessary to guarantee that thrusters do not operate unexpectedly. Tanks filled on Earth must withstand high structural loads caused by the multiplication of liquid mass by severe acceleration environments. Propellant storage for a year requires extremely low leakage rates.

Addressing these issues requires a hardware mass that could easily displace the payload from a prefueled MAV. Instead, the ascent propulsive stage described herein remains dry from Earth launch through Mars surface operations, while the ascent hydrazine is carried in a separate tank on the lander. The latter tank and its associated components can readily be designed to meet the safety and structural requirements, without adding inert mass to the Mars ascent stage. The propellant is finally transferred into the MAV tank just prior to Mars departure. Fueling on Mars is especially synergistic with pumped hydrazine. Only one fluid at low pressure has to be transferred, and Mars landers typically have tanks of hydrazine already, for landing propulsion.

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*Senior Scientist, P.O. Box 808; jcw@llnl.gov. Senior Member AIAA.
†Senior Project Engineer, P.O. Box 97009; gerry@red.prmx.com.

Table 1 Heavy items to be avoided by fueling on Mars

Mission phase	Full tank requirements
Earth prelaunch	Thicker tank wall to meet fracture mechanics rules; multiple valves in series with independent controls; pressure monitoring
Earth launch	Tank and structure to withstand vibration
Trans-Mars cruise	Nine-month leaktight propellant storage
Mars entry and descent	Loaded tank designed for >200 m/s^2 deceleration
Mars surface stay	Tank heaters and insulation to prevent propellant freezing
Mars prelaunch	Startup pressurization (if no fluid connection to lander)

These considerations are not the only value of fueling on Mars. Even if an Earth-fueled MAV could be feasible, the rocket equation shows that each kilogram of hardware that can be shaved off the MAV increases the mission mass allowance by 7 kg. The leverage comes from avoiding the 6 kg of hydrazine needed to lift the nonessential kilogram to Mars orbit. If a Mars-fueled MAV tank can be just one-seventh lighter than one designed for filling on Earth, it would compensate for the mass of the hydrazine storage tank on the lander. The reality is far better than this break-even point, yielding a net mission mass benefit by fueling on Mars.

A third advantage of fueling on Mars is the improved conservation of precious lander electricity. A storage tank that remains on Mars can certainly be insulated better than the MAV, to reduce heater power. The lander-based tank could even use RHUs (radioisotope heating units), whereas RHUs on the MAV itself would be too heavy ($\sim$100 g/W).

MAV Sizing and Operation

A Mars liftoff mass as low as 100 kg is of interest for small-scale sample return missions. To meet thrust-to-weight and velocity change requirements, the propulsion system delivers 1000 N of thrust in vacuum, and it carries 86 kg of hydrazine. Table 2 lists a mass summary along with calculated capability. A mass margin and residual fluids are included.

The key to holding mechanical stage hardware within the allotted 4 kg is to feed high-pressure thrusters from a low-pressure tank. As on launch vehicles, both are lighter than pressure-fed alternatives, by many times the pump mass.

Figure 1 shows a fluid schematic. This gas generator cycle is based on the results of previous pump-fed hydrazine tests.[4–6] Gaseous decomposed hydrazine powers the pump and then is exhausted externally. The net I_{sp} is 2% below that of the thrust chambers, because this fraction of the total expended mass is pump exhaust. A still smaller fraction of the high-pressure warm gas is regulated down for in-flight tank pressurization.

Extra valves normally included on spacecraft are absent from the system. A tank isolation valve would be a particular burden, as the low-pressure pump inlet tube must be large. Conventional fill valves having redundant sealing features are replaced by a flyaway disconnect. This has a valve that closes itself upon Mars departure.

A system analysis yielded the operating parameters displayed in Table 3. Ideal gas calculations were used because liquid displacement occurs at steady pressures without gas expansion work. Pressure drops through the gas generator circuit require liquid to be boosted to a higher pressure than the gas. This can be accomplished by using differential area pistons running in pump gas cylinders, which are larger than the liquid cylinders. It is assumed here that the piston area ratio is 1.56, as successfully tested previously. The gas volume required to run the pump is taken to be 20% higher to account for flow losses during gas and liquid valve switching.

Once the MAV tank and feedlines contain propellant, only the small valve in the gas generator circuit has to be actuated to start the system. Initially, warm gas at tank pressure enters the pump powerhead. The piston area ratio increases the liquid discharge pressure. This positive feedback loop rapidly raises the system to operating pressure as controlled by the liquid regulator.

Table 2 Mass and performance summary

Parameter		Measurement
Propulsion hardware and structure		4 kg
Batteries and valve electronics		1 kg
Guidance and control		2 kg
Sample package		4 kg
Mass margin ($>$20%)		3 kg
Total dry mass		14 kg
Hydrazine		86 kg
Total Mars liftoff mass		100 kg
System I_{sp}	*Residual fluids*	Δv
235 s	1 kg	4369 m/s
230 s	1 kg	4276 m/s
235 s	1.5 kg	4293 m/s

Table 3 Propulsion operating parameters

	Propellant allocations and mass flows		
Main thrusters	83.2 kg	423 g/s (1000 N at I_{sp} = 241 s)	
Pump power	1.8 kg	9.5 g/s (800 cm^3/s volume flow)	
Tank pressurant	0.15 kg	0.75 g/s (433 cm^3/s into tank)	
Unused liquid	0.85 kg		
Total burn time at full thrust: 85 kg $\div$ 0.433 kg/s = 196 s			
Net I_{sp} = 241 $\times$ (423)/(423 + 9.5) = 235 s			
	Pressures, temperatures, densities, and molecular masses		
Pump liquid cylinder	7.6 MPa	300 K	1001 kg/m^3
Thruster feed	7.2 MPa	306 K	995 kg/m^3
Thrust chamber	4.1 MPa	1380 K	5.4 (15)[a]
Gas generator output	5.7 MPa	1000 K	8.2 (12)
Pump gas cylinder	5.0 MPa	600 K	12.0 (12)
Tank ullage	0.35 MPa	300 K	1.7 (12)
	Heat rates		
Gas to liquid in heat exchanger			10 kW
Pressurant to tank wall and propellant			0.5 kW
	Volume displacements		
Tank, 86 kg hydrazine at $>$990 kg/m^3 requires 87 liters			
Pump, 440 cm^3/s displaced by 4 cylinders $\times$ 11 cm^3 at 10 Hz			
Mission life of the pump is 10 Hz $\times$ 196 s = $\sim$2000 cycles			

[a]Numbers in parentheses are molecular mass.

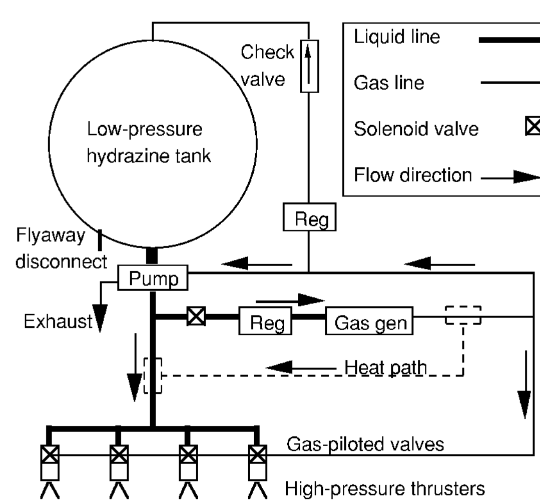


Fig. 1 Pump-fed hydrazine propulsion schematic.

The gas is cooled substantially in a heat exchanger, to reduce the operating temperature of the pump seals. This cooling strategy is reflected in the table and in the schematic. Note that the gas generator temperature is already far below that of the thrust chambers, by virtue of a high fractional ammonia dissociation. However, 1000 K is still too hot for a long seal life with negligible leakage. As indicated, the warm gas is passed through a heat exchanger and then on to the pump at 600 K. The resulting warming of the propellant stream by 6 K enhances thruster performance.

Minimal retained pressurant is another mass advantage of pump fed operation. The tank accumulates only 150 g of gas, or less if the ullage remains above 300 K. The heating load on the low-pressure tank is also comfortably low. This is important because of concerns of hydrazine thermal decomposition above 480 K. Even if the tank wall had to radiate the entire 500 W listed in Table 3, it would remain below 350 K at high emissivity.

Stage Layout and Components

The single-stage propulsion configuration is illustrated to scale in Fig. 2. It comprises one spherical tank, four high-pressure thrusters, and one pump. Components are mounted below the tank for several reasons. Obviously, the pump should be at the tank outlet port. The thrusters are clustered closely around the pole to minimize resulting vehicle torques. Plumbing weight is kept low by locating all other wetted components nearby.

This preferred configuration permits the lander to support the heavy parts of the MAV during Earth launch and Mars arrival. The thin tank wall only has to support itself, aided by a small internal pressure. The tank's smooth upper hemisphere serves as an ascent fairing. A potential drawback of the parts arrangement is that the center of mass is aft of the center of pressure. However, destabilizing aerodynamic moments are low at Mars.

An external tank pressurant tube could potentially interfere with the supersonic flow around the front of the vehicle. Instead of running it outside to the top, the tube enters from below. Its length, as shown in the sketch, keeps bubbles away from the pump inlet.

The single-tank MAV avoids large structural elements entirely. The only nonwetted flight structure is numerous brackets welded to the tank wall, as traditionally done on the sheet-metal tank of the Atlas launch vehicle. For example, threads at the top of the thrust chamber screw into titanium cylindrical shells welded directly to the tank.

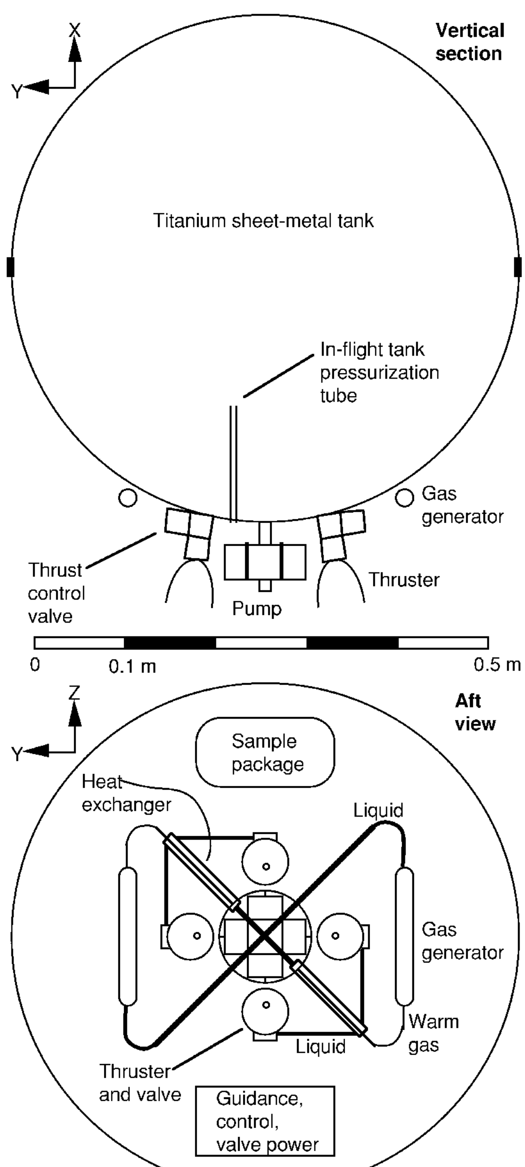


Fig. 2 Locations of major components.

Attitude Control

During ascent, three-axis attitude control is effected by off-pulsing selected thruster pairs. Adjacent pairs control pitch and yaw, and members of diagonally opposite pairs are angled slightly to produce a pure roll torque when shut off together.

Pairs of 250-N thrust vectors offset 0.05 m from the mass center impart pitch and yaw torques of 25 N $\cdot$ m to the vehicle. The worst-case sensitivity occurs when the tank is nearly empty, that is, angular accelerations of 80 rad/s^2 for a $\sim$0.3 kg $\cdot$ m^2 rotary inertia. Off-pulsewidths of 5 ms change the empty vehicle's angular velocity by 0.4 rad/s. Twenty guidance updates per second limit angular excursions to the order of 0.02 rad. The atmospheric part of the flight is much smoother, as the propellant increases rotational inertia.

High-Pressure Thrust Chamber

Beginning in 1988, a high-pressure hydrazine thruster was developed for advanced technology programs. The MR-125 (MR stands for monopropellant rocket) is shown in Fig. 3, with data in Table 4.

The MR-125 is by far lighter and smaller than conventional thrusters, and I_{sp} is improved. These advantages are directly attributable to a high operating pressure, which reduces nozzle area. Also, the density of the decomposition gases is greater than that in conventional low-pressure thrusters, which results in a longer residence time for a given catalyst bed geometry. Weight is additionally reduced by the omission of features included on satellite thrusters. For example, the latter require heaters and thermal shielding to maintain performance for many infrequent short pulses.

Table 4 MR-125 test parameters

Parameter	Measurement
Equilibrium vacuum performance	
Thrust	250 N
Specific impulse	240–245 s
Feed pressure	7.2 MPa
Chamber pressure	4.1 MPa
Chamber temperature	1380 K
Expansion ratio	50:1
Hydrazine grade	Purified
Mass (as pictured in Fig. 3)	100 g
Response times after warmup	
Start response	$<$15 ms to 90% thrust
Stop decay	$<$20 ms to 10% thrust
Shortest pulses tested	5 ms, $\pm$13% repeatability
Life tested thruster	
Total propellant throughput	93 kg
Total on time	994 s
Longest single burn	600 s at nominal conditions
Ambient temperature starts	3
Feed pressure range	0.9–9.7 MPa

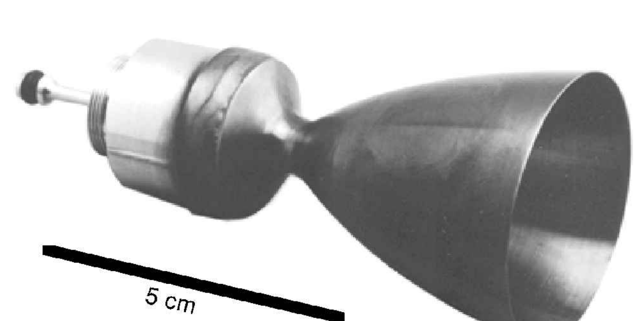


Fig. 3 High-pressure 250-N hydrazine thruster.

More than 30 units were manufactured for use in experimental propulsion system tests, including pump-fed suborbital flight. Lightweight valves were specifically designed to control this high-pressure thruster. A cold-gas piloted valve and a warm-gas piloted one massed under 40 g each. The combined delay of both valve stages was between 2 and 5 ms. These custom valves were used in system tests, and timing data in Table 4 reflect this. The weights of the valve, thruster, and thermal standoff sum to just over 0.5% of thrust.

Given the benign reaction temperature of monopropellant, material thermal limits do not impair lifetime as with miniature high-pressure bipropellant thrusters. As shown in the table, one of the hydrazine thrusters operated several times longer than the 200 s needed for Mars ascent.[7] In addition, pulse performance is adequate to effect directional control during the flight. Short off pulses are necessarily crisp, because the catalyst remains hot.

Development including thrust stand characterization is complete. Flight qualification would include random vibration, pyroshock, and thermal mapping. Because of its maturity and minimum weight, this thruster technology has been attractive to Mars lander system engineers.

Tank

A conventional satellite tank sized for the MAV volume would consist of >5 kg of alloyed titanium in a spherical shape. It would have an average wall thickness near 1 mm and a burst pressure well above 3.5 MPa. A worthy goal here is to diminish the mass by a factor of 5, as a similar reduction in burst pressure is acceptable. Thus the tank wall material need not be stronger, just thinner. Without Earth safety concerns, reducing thickness is mainly a fabrication project. Options are to form hemispheres from thin sheet stock or to use the traditional forge-and-machine approach. It is likely that either can be made to work.

An inner diameter of 0.56 m encloses over 91 liters. Considering hydrazine's thermal expansion, 86 kg of liquid will fit even at elevated temperatures. Worst-case calculations here assume that the tank wall has the density (4.54 g/cm^3) and the moderate strength ($\sim$500 MPa) of unalloyed titanium. The latter facilitates cold forming of hemispheres from sheet material. A wall thickness of 0.2 mm yields a shell mass of 0.9 kg and a hoop stress equal to 700 times the internal pressure. Thus at 0.35 MPa, the tensile stress is very low at 245 MPa. Burst pressure is twice this operating level. This is a far greater margin than human-rated launch vehicle tanks, which are tested to only single-digit percentages above their similar operating pressures.

For alignment tooling to be simplified, the shell halves are welded to an equatorial ring. At a cross-sectional area of 20 mm^2, the mass is 160 g. For residuals to be minimized, 200 g are budgeted for anti-slosh baffles near the outlet. The tank mass goal is therefore 1.3 kg, including a 40-g porting allowance. Note that, if the tank halves are machined from forgings in the traditional manner, wall thickness variations will influence achievable weight. The Ti-6Al-4V alloy is much stronger than the pure metal, and so the minimum wall thickness in this case might be less than that specified here.

Tank technology heritage, at least at the proof-of-principle level, comes from the pumped hydrazine flight in 1994. A titanium 6Al-4V sheet that was 0.2 mm thick was welded with an Nd:YAG laser. Tanks were proof tested at a hoop stress above 620 MPa, and one was cycled hundreds of times to 525 MPa without any failures except in deliberate burst tests.

Quad Piston Pump

Figure 4 shows a quad piston pump assembly. Four chambers are alternately filled from the tank and expelled at a much higher pressure. The cylinder assemblies are bolted to a central liquid manifold block that contains inlet and outlet check valves. This arrangement lowers liquid pressure losses as well as mass. Opposite pistons stroke toward each other, which cancels net mass shifts to greatly reduce vibration. There is no external control, as the gas valves are synchronized pneumatically. Piston speed and switching frequency can vary all the way down to zero, at full pressure. Actual flow depends entirely on thrust chamber valve actuation, just as in a pressure-fed system.

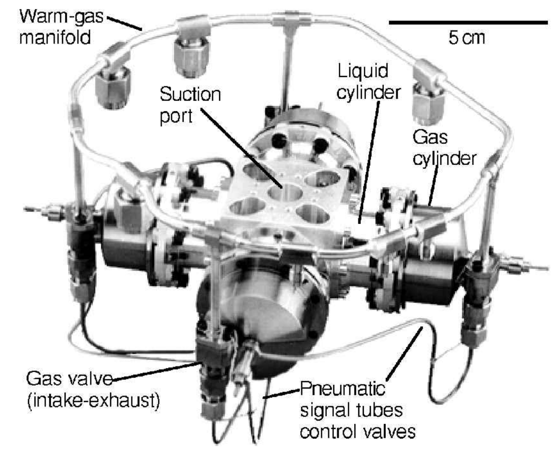
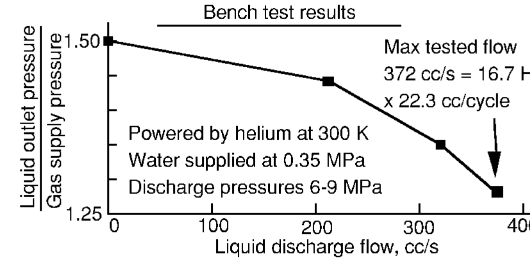


Fig. 4 Four-chamber hydrazine pump tested in 1993.

Sizing information

Mass as tested	365 grams
Liquid cylinder bore	25.4 mm
Gas cylinder bore	31.75 mm
Piston area ratio	1.56
Max piston travel	12.8 mm
Ideal liq displacement	6.5 cc x 4 cylinders = 26 cc/cycle

Bench test results



Tests with hydrazine

Pump-fed engine static test, 520 N thrust at sea level
Pump-fed engine flight test, 250-260 cc/s for 37 s duration
Half quad life test, warm gas & water, >1500 cycles

Fig. 5 Characteristics of the quad piston pump.

Figure 5 indicates sizing and performance for the hardware pictured in Fig. 4. The 365-g assembly delivered its own mass in liquid each second above 6.2 MPa, from a tank at 0.35 MPa. This flow of hydrazine supports vacuum thrust 230 times the pump's Earth weight. Figure 5 indicates that the pump cycle life requirement listed in Table 3 is immediately within reach.

For operation in a gas generator cycle engine, a key performance parameter is the pressure ratio of the liquid discharge to the driving gas. This boost ratio falls in the graph as flow rises. Boost is reduced by pressure losses in passageways during the power stroke, particularly the liquid discharge check valves and the gas intake. Even the static boost ratio (1.50) was below the piston area ratio (1.56) because gas leakage caused intake flow with an associated pressure loss.

The MAV pump has to be rated at up to twice the flow of the original quad assembly. For pressure drops to be held constant, areas must be scaled as flows to leave fluid velocities unchanged. Therefore, the linear dimensions of the pump would increase by 30–50%, and its mass would theoretically scale as the cube, approximately three times. Straightforward measures for reducing mass include machining the liquid cylinders and block as one piece. Dimensions bounded by fabricability (e.g., tubing walls in Fig. 4) need not be thickened. Therefore, the pump mass can realistically be under 1 kg.

The gas valves in Fig. 4 vent directly to ambient with no exhaust manifold. Considering the central aft location of the MAV pump in Fig. 2, it is simple to run the exhaust through aftward-pointed nozzles. No credit was taken for this extra thrust in the system analysis.

A technical obstacle to perfecting a highly reliable system was the need for dynamic warm gas seals on the pump pistons and in the intake-exhaust valves. In Fig. 4, solid graphite seals were used near 1000 K and simply allowed to leak even more gas than was needed for pump power.

Ongoing work includes testing pump designs that have liquid-cooled soft seals to eliminate gas leakage. Preliminary results indicate that the 600 K gas temperature used in Table 3 is a practical compromise. In the worst case, the penalty for cooling the gas further to ∼400 K is heavier heat exchangers. The system I_{sp} would fall by several seconds, requiring an extra few hundred grams of propellant.

Gas Generator

Like the MR-125 thruster, this technology is mature. A long catalyst bed is used to promote endothermic ammonia dissociation, thus cooling the gas and reducing average molecular weight. A high-pressure gas generator was developed specifically to power the pump. At 115 g, it includes a small integral accumulator and a filter. Its typical throughput is 5 g/s at a 0.9-MPa pressure drop. One was tested to a total throughput of 0.95 kg. A pair in parallel satisfy both flow and life requirements in Table 3. As indicated in Fig. 2, they are arranged to feed the circular pump manifold at opposite points for even flow.

Table 5 Mass summary for fluid parts and structure

Component	Mass, kg
Tank	1.3
Pump	1.0
Thrusters (4 at 100 g)	0.4
Thruster valves (4 at 50 g)	0.2
Gas generators (2 at 115 g)	0.23
Gas generator valve	0.05
Liquid regulator	0.05
Heat exchangers (2 at 50 g)	0.1
Gas regulator	0.03
Check valve	0.03
Pressurant tube	0.05
Flyaway fluid disconnect	0.08
Liquid filters	0.1
Liquid lines	0.04
Gas lines	0.03
Fluid joints	0.15
Thruster mounts	0.06
Component brackets	0.10
Propulsion hardware total	4.0

Table 6 Key elements of demonstrated capability

LLNL ASTRID flight from VAFB, 1994[a]
Ground launch of miniature pumped hydrazine rocket
Vacuum Δv equivalent >2000 m/s from 21-kg launch mass
Isolation valves omitted for less mass and complexity
Lightweight quad configuration for piston pump
Titanium sheet metal tank, wall thickness 0.2 mm
Warm-gas pressurized tank, no gas–liquid separator
Thrusters individually pulsed for warmup prior to liftoff
Soft elastomer seals contained warm gas in valve pilots
ASTRID static fire tests at VAFB, 1993[a]
Bootstrap start from only 0.22 MPa tank pressure
Thruster off-pulsing during steady burn, re attitude control
Warm gas up to 450 K pressurized tank without incident
Terrestrial test experience is relevant to MAV testing
Components and systems at Primex test lab
High-pressure long-life thruster proven repeatedly
Smooth pump system operation, well instrumented
Leaky pump seals detracted just 5% from system I_{sp}
Thruster pulses down to 5 ms during pump feed
Laboratory tests at LLNL[a]
High power-to-weight measured for LLNL developed pumps
Thin wall tube used for mini-heat exchanger at 7 MPa, 900 K
Leaktight liquid-cooled warm-gas seals show promise

[a]LLNL, Lawrence Livermore National Laboratory; VAFB, Vandenberg Air Force Base.

Propulsion Mass and Heritage Summary

The masses of the remaining components have similarly been estimated based on functional parts tested previously. As listed in Table 5, most are below 50 g each.

The historical record for system operation is summarized in Table 6. The unique components and system design evolved together and culminated in the flight of a miniature vehicle. This work was documented in detail.[4–6]

Lander Systems

Ground support equipment is included on the lander, just as is required for launch vehicles departing from Earth. Separable interfaces include structure, a fluid joint, electrical connectors, and heat transfer paths. These are located below the tank, and so the vehicle essentially nests within its support equipment.

Temperature-critical items are warmed from below, so a full enclosure over the MAV may not be necessary. Depending on the time interval between fueling and launch, the fuel might be preheated above 320 K to preclude the possibility of prelaunch freezing (275 K). A thin layer of hydrazine ice could be tolerated inside the unheated upper portion of the MAV tank. Aerodynamic heating and the warm pressurant would melt it during ascent.

The hydrazine is supplied from well-insulated conventional tanks on the lander, in a system having all the features listed in Table 1. Propellant transfer is accomplished simply and reliably by blowdown. Conventional tanks operate at up to 3.5 MPa, and so only a 10% initial ullage volume is needed to fill the MAV to 0.35 MPa. A pressure transducer on the ground side is sufficient to indicate when propellant transfer is complete.

The support equipment may include a regulated helium supply to maintain a small pressure inside the MAV during transport to Mars. Before fueling, the MAV is vented. Residual gas at Mars ambient pressure then compresses to well under 0.5 liter, leaving the flight tank 99.5% full. Note that no ullage gas is needed in the flight tank for starting, because the initial pressure is applied by the gas in the lander tanks.

The electrical interfaces transfer guidance data and power until the moment of launch. As this includes solenoid power, valve operation may be verified acoustically before fueling without draining ascent flight batteries. To further reduce ascent electrical loads, the MAV can use normally open valves initially held closed by lander power. In the case of the gas generator feed valve (Fig. 1), the actuating solenoid could even remain on the lander.

Before liftoff, all catalyst beds are raised to operating temperature by individually pulsing the thrust chamber valves. The ground tank connection keeps the MAV tank full, so the consumed warmup propellant does not detract from that available for ascent. As the MAV flies away, the withdrawal of the fueling tube lets a check valve close on the MAV, while an active valve shuts on the ground side.

Landing Propulsion

Mars landers have historically used hydrazine for their final descent propulsion.[8] Reasons include system simplicity and the absence of carbon, oxygen, and water, which could interfere with science measurements. Another reason is that the relative importance of engine mass and I_{sp} differs vastly from the norm in spacecraft propulsion systems.

Landing softly on Mars requires only a low Δv (∼300 m/s) at an unusually high acceleration (∼1 Earth g). Engine mass is nearly twice the tank mass if conventional spacecraft components are used.

Considering the preference for monopropellant hydrazine and the value of reducing engine mass, HPPH is entirely appropriate for Mars retropropulsion. Such a landing system would have conventional tanks filled on Earth, unlike the high-Δv MAV. In addition to reduced engine mass, packaging would be improved because the higher-pressure thrusters are much smaller than the conventional ones. HPPH landing propulsion would permit larger science payloads to be delivered to Mars.

Since 1995, the MR-125 thruster has been of great interest to Mars lander engineering teams. Unfortunately, the improvement would be canceled by the extra weight of high-pressure tanks, and implementing HPPH still requires some development effort. What has not been recognized previously is that, if HPPH is developed for either a lander or a MAV, this advanced technology would then be more readily available for both applications.

Broader Aspects of Mars Ascent Vehicle Development

There are many related mission problems, including sample handling mechanisms and biological contamination concerns. Although a complete MAV design is beyond the scope of this paper, several broader issues are noted as follows.

Scaling to Fit Mission Needs

The achievable mass of guidance and control hardware may determine a minimum MAV size. It would be straightforward to scale the system somewhat; for example, six thrusters would be used for a 150-kg Mars liftoff mass.

Trajectory and Drag

The actual propellant load and control requirements must be based on a detailed trajectory analysis to a particular orbit. From a propulsion system operation standpoint, a direct ascent trajectory is preferred. Otherwise, the flight may require attitude control during coasting and a restart capability for a circularization burn. In this case, microgravity fluid management and miniature gas jets would consume some of the mass margin.

Drag and stability during atmospheric flight depend on the well-known supersonic flow field around a sphere. Aft flow around the exposed components would remain subsonic. At $Cd = 1$, a dynamic pressure near 250 N/m^2 produces 62 N of drag. Assuming this occurs as an average value over the first minute, drag cancels the impulse of 1.5 kg of hydrazine, a Δv loss near 50 m/s.

Testing on Earth

A primary issue for Earth testing is the potential exposure of people to toxic propellant contained inside unusually lightweight hardware. Rupture damage is a lesser concern, because of the small size and low tank pressure. System analogs using benign propellants would undoubtedly be useful for initial system testing. For example, cold gas permits repeated three-axis attitude control system testing. Short-duration flights could be performed with a liquid monopropellant such as hydrogen peroxide, as was done during Apollo lunar module development.

Although expensive, there are numerous potential flight test scenarios for the full-up MAV with its lander-based support equipment. These include a suborbital ground launch, a suborbital high-altitude flight (e.g., from a balloon), a flight to orbit from a booster or reusable launch vehicle (RLV) prototype, and in space.

Perspective on Alternatives

The thin dark bars in Fig. 6 show the single-stage nonexpended mass allowances for three propellant options. For example, the improved performance of bipropellants over hydrazine increases the burnout mass of a 100-kg loaded MAV by 9 kg. This multiplies the ascent payload several fold (or lightens the MAV), but only if propulsion hardware is light enough.

For each propellant, the masses of associated hardware elements are displayed. The leftmost example is the single-stage HPPH MAV. Masses shown next to the center bar are typical of conventional bipropellant technology flown on satellites. Such proven components are too heavy, even with the improved performance of bipropellants.

Both advanced pressure-fed and pump-fed bipropellant technology options have been proposed for Mars ascent. These are not represented in Fig. 6 because their low levels of maturity do not support mass calculations traceable to tested propulsion systems. Numerous miniature high-pressure biprop thrusters even smaller than the MR-125 have been demonstrated. Although sufficient lifetime is an issue, their thrust-to-weight ratios have been exceedingly high, so it would be appropriate to feed them with

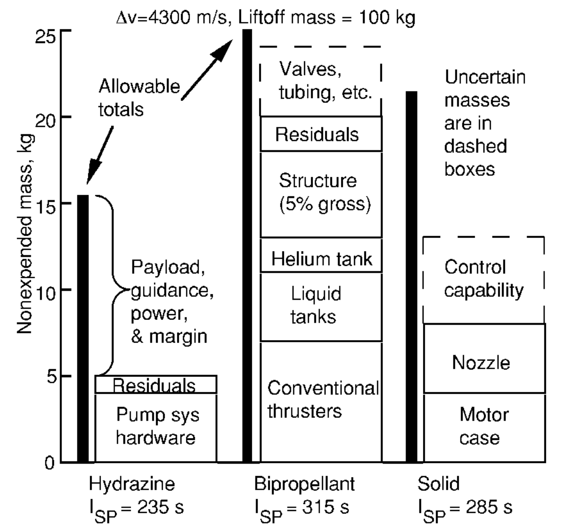


Fig. 6 Propulsion hardware mass is critical.

reciprocating pumps like the ones developed for hydrazine. Further work on advanced miniature bipropellant technology seems appropriate.

The bar chart includes a solid propellant option for completeness. Because multiple burns are needed to achieve orbit, a single-stage solid would really need several grains and igniters. Alternatively, the solid representation in Fig. 6 may be viewed as merely indicative of achievable stage inert fractions. Solid rocket motors consist of titanium spherical shells built to withstand high operating pressures near 7 MPa. Solids also have high thrust levels that are due to inherent burn rates. For these reasons, thick motor cases and large nozzles each mass approximately 5% of the propellant. In principle, it requires less material to build low-pressure liquid tanks and engines sized for the maneuvering requirement.

The graph also indicates that the practicability of a solid MAV depends on the latter's uncertain control system mass. Auxiliary hardware must produce high control moments during the burn. If this consists of nozzle steering actuators, a separate three-axis system may be needed during coasting. Control challenges for solid rockets may also include spin dynamics with precise mass management.

If staging is needed, the extra interstage structural mass required to surround large nozzles might be heavier for solids than for liquids. Last, it is noteworthy that solid propellant cannot be easily transferred on Mars. Both structural hardware and the solid propellant grains themselves must withstand high acceleration environments.

A final consideration is that the engine exhaust of the HPPH MAV will do the least damage to the Mars lander and its science instruments. There is far less gas at lower temperatures, without organics or solid condensibles. Mars landers have used hydrazine partly for this reason.

Conclusion

Affordable Mars sample return requires an unusual miniature rocket vehicle, outside the realm of well-proven technology. From the data presented herein, HPPH is a strong candidate for a practical MAV. Feeding proven high-pressure thrusters from a thin tank by using a pump enables a single hydrazine stage to lift Mars samples to orbit. Based on previously documented technology development efforts, this option offers an acceptably low risk, affordable, and technically defensible solution for Mars sample return.

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J. A. Martin
Associate Editor