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Mars Ascent Propulsion on a Minimum Scale

J. C. Whitehead
C. S. Guernsey

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Lawrence Livermore National Laboratory

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John C. Whitehead
Lawrence Livermore National Laboratory

Carl S. Guernsey
Jet Propulsion Laboratory

A concept is presented for a single stage vehicle intended to lift a Mars sample to an orbital rendezvous. At 200 kg liftoff mass, it can potentially be delivered by a Mars Pathfinder size aeroshell. Based on launch vehicle design principles, propellants are pumped from thin-walled low pressure tanks into compact high pressure thrusters. Technical risk is reduced by using non-cryogenic propellants, and by driving piston pumps with heated helium.

Introduction

The cost of transportation to and in space will continue to drive exploration price tags for the foreseeable future. Achieving low cost therefore requires reducing transported mass. For Mars sample return, the production of propellants on Mars has been extensively studied. A different approach considered here is to relentlessly miniaturize all mission hardware. The goal is to deliver a 30 kg payload to Mars orbit, with a 200 kg liftoff mass. Recent efforts[1&2] are combined to emphasize storable bipropellants in a risk-reduced pump system.

Regardless of the propellant source and degree of miniaturization, Mars ascent is an unsolved engineering problem. Reaching Mars orbit, let alone earth, is beyond the capability of all spacecraft ever built. Figure 1 defines the problem. The choice of axes is justified because achieving high velocity (x-axis) and high acceleration (y-axis) are in direct physical conflict. Specifically, tanks of propellant must compete with engines for a share of the same mass budget. It is far easier to obtain the required Δv at low acceleration (Deep Space-1) or conversely (Viking lander).

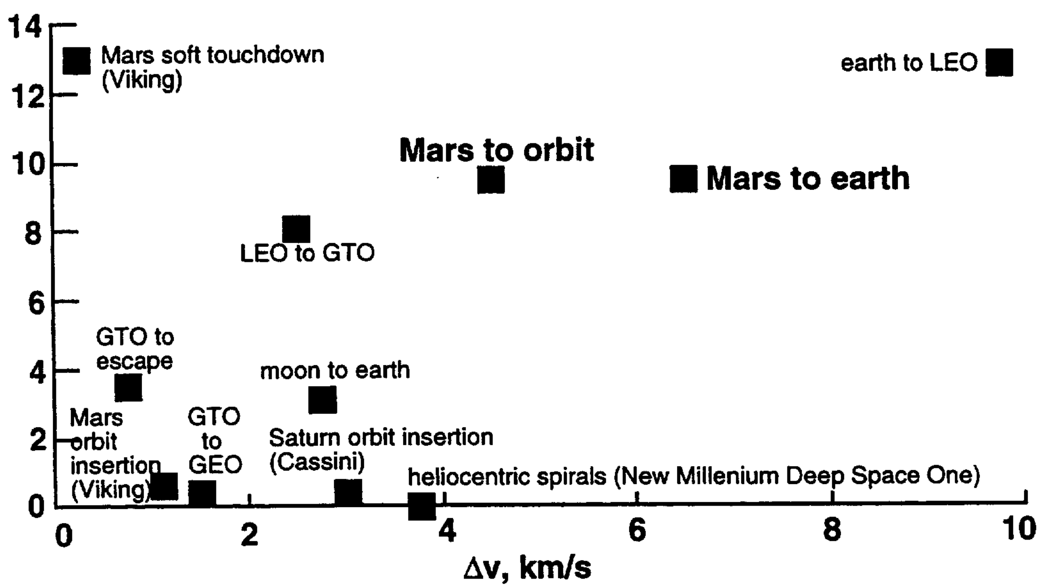


Figure 1. Maneuvering requirements compared.

Maneuvering capability along both axes depends on the relative sizing of payload and propulsion. For a closer look at the problem, corresponding performance ratios for propulsion systems alone were chosen as the axes of Figure 2. Points labeled "stage" include structure in the denominators, whereas "components" refers to data which excludes structure. Additional points labeled with bold type represent the goal for either one or two Mars ascent stages.

Figure 2 shows that both required ratios are easily exceeded by individual first stages of earth launchers. However, small upper stages and the largest spacecraft are barely capable. Those plotted here are Delta's bipropellant upper

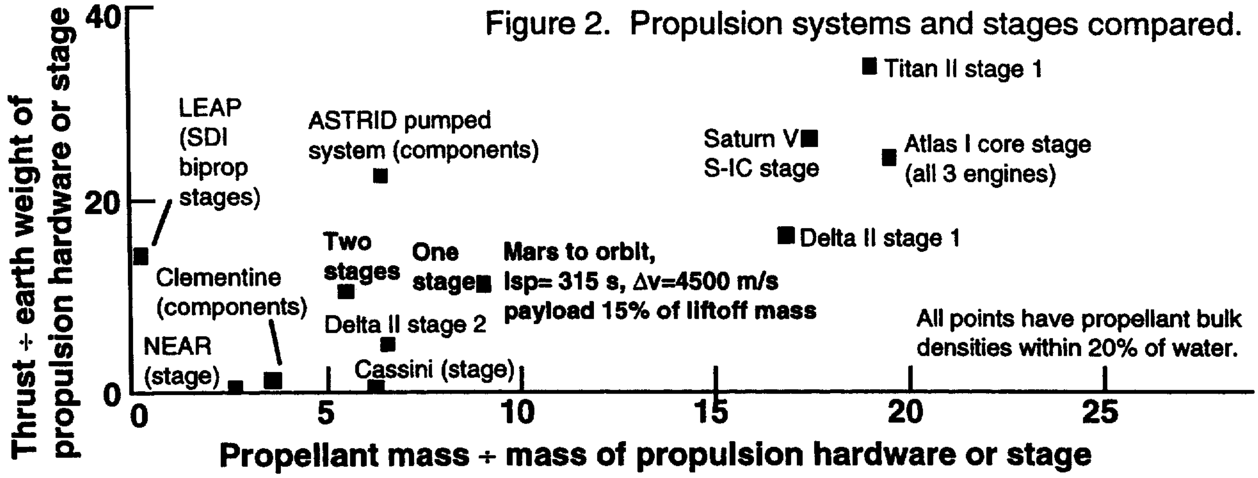


Figure 2. Propulsion systems and stages compared.

stage (7.0 metric tons wet) and Cassini's propulsion module (3.6 tons wet). The graph indicates that similar systems having additional thrust would suffice for a 2-stage vehicle. By a separate calculation, the Delta stage with an extra engine could reach Mars orbit alone, if the payload fraction is reduced to 10%.

Recent deep space bipropellant systems approaching the 200 kg scale of interest include Clementine (255 kg wet) and NEAR (438 kg wet). Per Figure 2, the hardware of both is roughly 100% overweight. Regarding possible upper stages, the only smaller bipropellant systems known to the authors are those built for missile defense at the left edge of the graph (<10 kg wet). They are tailored for a significant acceleration, but at very low Δv.

It is easy to conclude that absolute size is the determining factor, in which case Mars ascent would require a large vehicle. This is true if existing capability must be used. However, it is significant that downsizing rocket hardware is limited more by practical difficulties than by fundamental scaling laws.

In addition to being large, all four stages on the right side of Figure 2 use pump fed rocket engines. This is standard because low pressure tanks can be lightweight, whereas high thrust chamber pressure permits engines to be compact and light. The existing systems in the lower left region of the graph are all pressure fed. Their tanks operate above thrust chamber pressure, which compromises pressure levels and increases the mass of both. The ASTRID flight experiment point in Figure 2 represents 450 N thrust and 12.7 kg of hydrazine, with 2 kg of wetted components (21 kg total at liftoff).[3] It shows that pump fed operation can offset some of the extra mass due to scaling difficulties.

Reciprocating pump concepts

Large rocket engines use centrifugal pumps that are shaft-driven by turbines. As these rotating dynamic pumps are scaled down, they become heavier and less efficient. Fortunately, reciprocating pumps are suited to small scale applications. Moreover, their positive displacement can accommodate wide ranges of pressure and flow.

Pressure fed expulsion requires strong tank walls as indicated on the left side of Figure 3. The right sketch illustrates the reciprocating pump principle. A much lighter tank is pressurized only enough to refill the pump chambers through a short large pipe and check valves. Three way valves control the flow of high pressure gas into and out of the pump chambers. The latter alternately expel at high pressure and refill at low pressure. The thermodynamic principle is the same in both sketches. The gas cannot know whether it is displacing liquid from a large tank or from small pump chambers. However, total hardware mass on the right can be much less than on the left.

The thin tank still needs some pressurant, and the pump wastes a little gas for valve pilots and to deliver a small fraction of the liquid back to the tank through check valves that don't close instantly. If the pump chambers have pistons, there is a pressure loss to friction. The pump discharge pressure has switchover transients, but they are minimized by overlapping the expulsion phases.

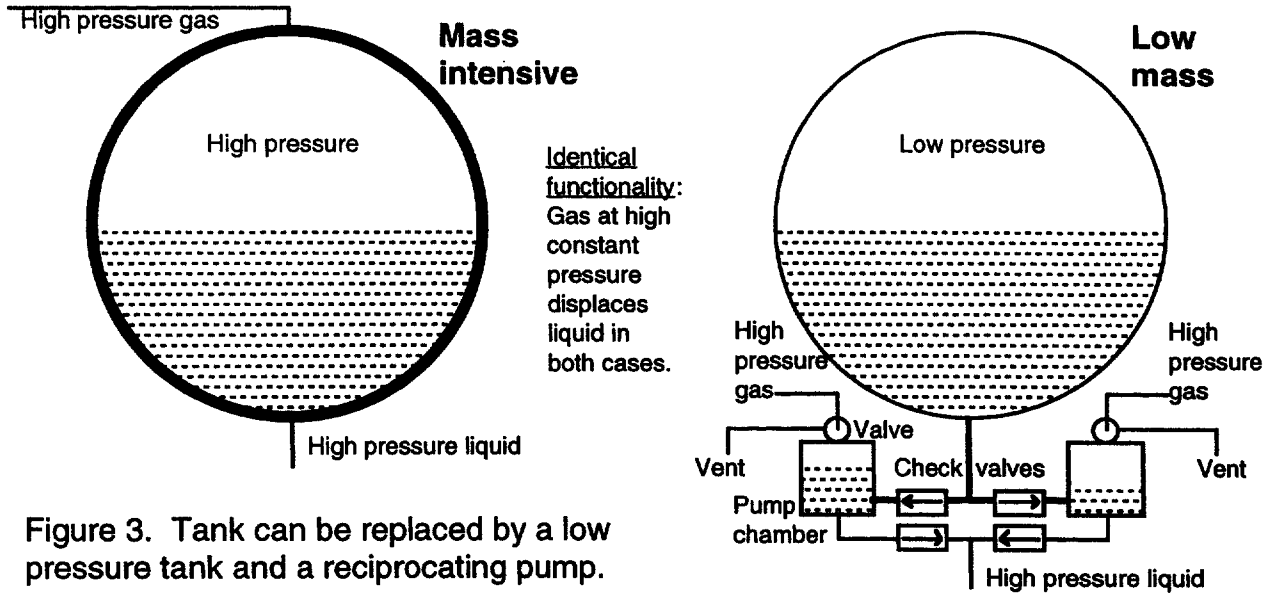


Figure 3. Tank can be replaced by a low pressure tank and a reciprocating pump.

A difference which is favorable to pumping is that the gas is vented, and can even produce thrust. In the left sketch pressurant is retained as inert mass. This effect increases if the gas is warmer than the propellant or surroundings. Gas used to drive pumps has little time to cool, whereas cooling in the left tank requires more pressurant.

In general, the gas which drives pumps may be reacted propellant, but this complicates both the pumps and the system. A simpler type of pump-fed rocket uses a separate source of gas. It is notable that all piloted flights to orbit outside the U.S. have relied on such simplified engines. Specifically the RD-107 and RD-108 engines, on the Vostok and Soyuz launch vehicles, power their turbopumps with decomposed monopropellant.

The ideal pump drive gas would have a low density, and it would be inert to prevent reactions in the pumps. This suggests consideration of helium, as in Figure 4. The primary disadvantage of using stored gas is its tank mass.

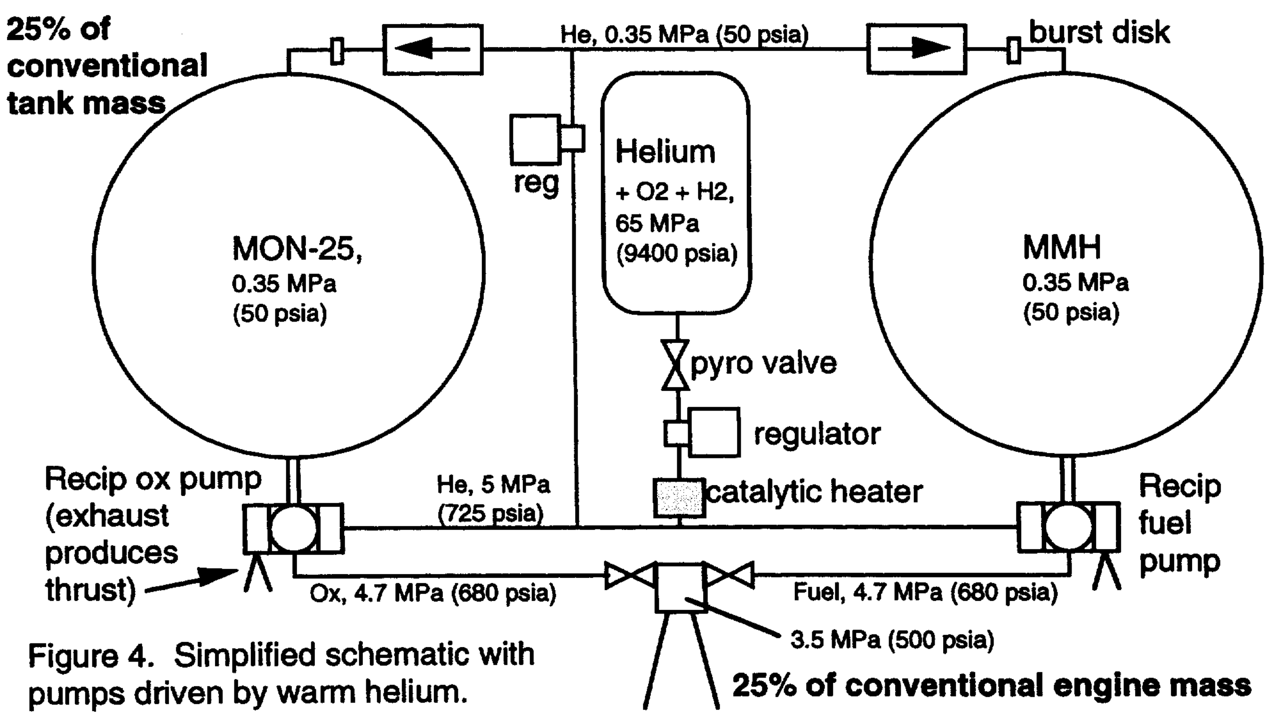


Figure 4. Simplified schematic with pumps driven by warm helium.

Helium tanks were once heavy like conventional propellant tanks, due to similar PV products. However, gas vessels now benefit from fiber materials, since thick walls allow for layers of overwrap on a leaktight liner. As examples, the helium tank mass is well under half the propellant tank total on both Clementine and NEAR.

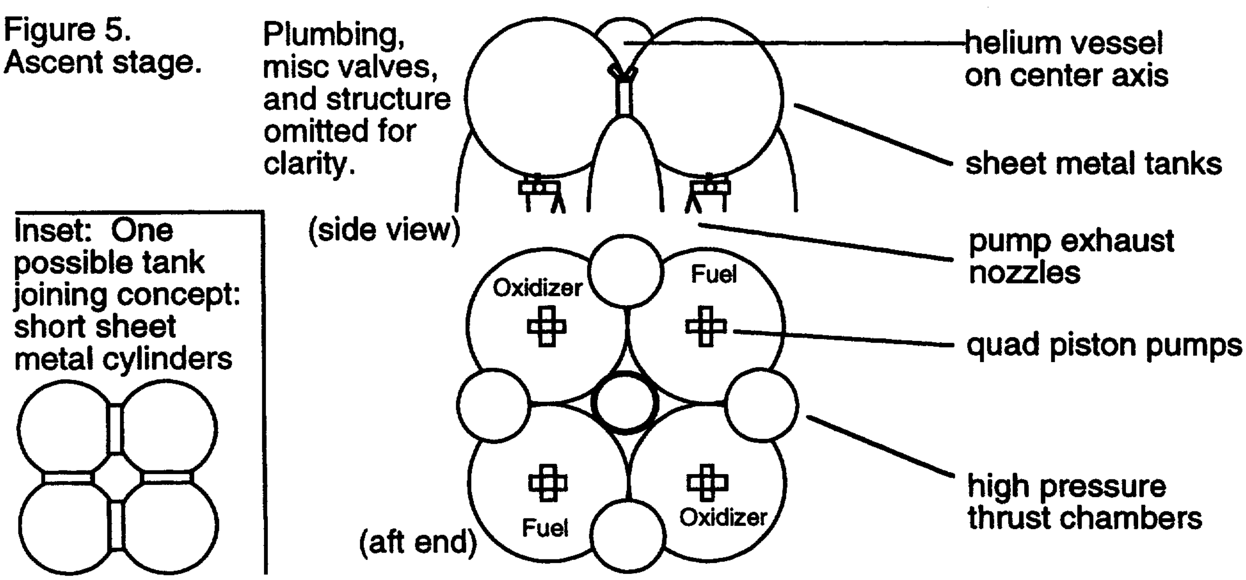
As in Reference 2, small amounts of oxygen and hydrogen in the helium would react in a catalyst bed. Doubling the absolute temperature of a conventionally-sized helium load permits engine injectors to be fed at twice conventional pressure. System pressure levels and resulting mass improvements are approximated in Figure 4. The sum of the major component masses falls to within acceptable limits, so the concept is worthy of careful consideration.

Stage design

Reference 1 suggested carrying both propellants within a single partitioned sphere to reduce structure and plumbing. Such a common-bulkhead tank is employed with hypergolics on the Delta 2 upper stage. To reduce risk, the traditional 4-tank arrangement for balancing fuel and oxidizer is considered here.

In spite of the mass of extra structure, there are advantages to the packaging shown in Figure 5. The source of gas for tank pressurization and pump drive fits within the center space, so balance is maintained without multiple tiny tanks. It is recognized that this rules out a single center engine, and that the single sphere design would need only three engines. However, four separate engines nested between four tanks can control rotations on all three axes if mounted at appropriate small angles. The stage's size makes it especially desirable to do without liquid attitude thrusters, due to scaling difficulties. Tiny valved gas jets would use residual helium for on-orbit attitude control.

Figure 5. Ascent stage.



As with car engines, a large number of pump cylinders will smooth pulsations. The four-tank design is consistent with this, as it is convenient to locate a multi-chamber pump below each tank. This pump location is ideal for minimizing plumbing from the tanks and to the aft-directed exhaust nozzles. Both passageways are critical, since the pump chambers must rapidly vent then refill from the low pressure tanks. The pumps are not actively controlled, so their flows may differ slightly. Tank balance would then require small shunt tubes which connect each tank pair.

Low pressure sheet metal tanks may permit a unique structural approach (Figure 5 inset). The Atlas launch vehicle has examples of component brackets spot welded to sheet tanks. There are fracture and fatigue issues. A detailed analysis, which needs to include deceleration load paths to the lander, would require resources not available at present.

A single stage ascent vehicle is attractive, since complexity is relatively mass-intensive on a small scale. An upper stage would have to be miniaturized further by at least a factor of three, with only a small relaxation in stage propellant fractions, per Reference 1. Fitting two stages in the permitted envelope can be difficult, per Reference 2.

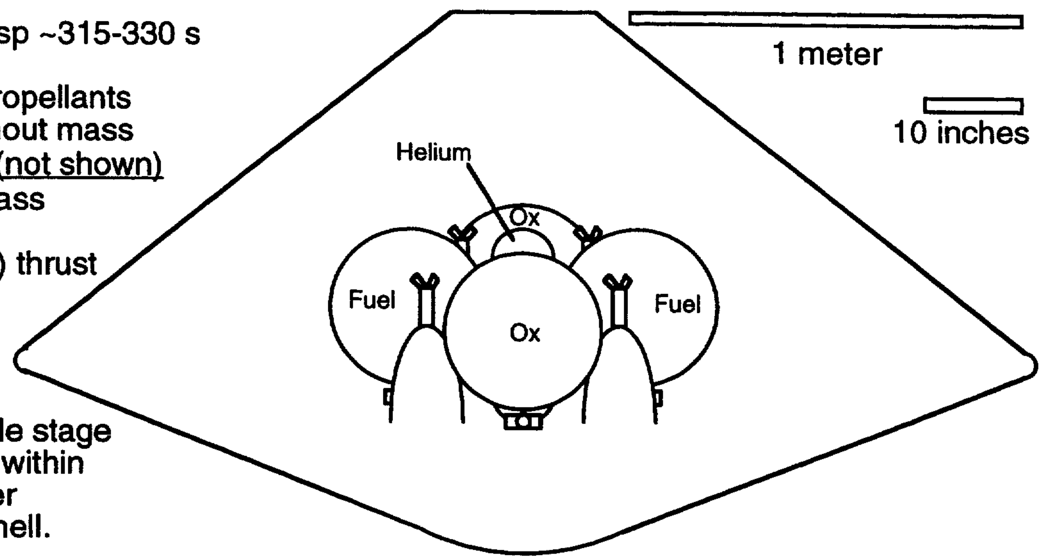


Figure 6. Single stage ascent vehicle within Mars Pathfinder descent aeroshell.

Figure 6 compares the design to Mars Pathfinder's aeroshell outline. Reference 2 assumptions applied to Pathfinder's 535 kg at entry yield 398 kg landed and 303 kg delivered, or nearly 100 kg each for the lander and sample collection.

Propellants and components

Propellants and thrusters

Many spacecraft use nitrogen tetroxide (NTO) to combust monomethylhydrazine (MMH) at a 1.65 mass ratio, which conveniently requires equal volumes. This mixture ratio also provides excess fuel for film cooling of thrust chamber walls, which enables multi-hour burn lifetimes for inserting large satellites into geostationary earth orbit (apogee burn). Typically, Isp approaches 315 s, which can be raised by increasing the mixture ratio or pressure.

While MMH freezes at 221 K, NTO does so at 262 K. Adding nitrogen monoxide (NO) to depress the freezing point would reduce the heating need on Mars. MON-25 (mixed oxides of nitrogen with 25% NO) freezes at 218 K, but combustion instability is an issue. At a mixture ratio of 2.1, performance could exceed 330 s at pump-fed pressures. Herein, the full ranges of both NO fraction (0-25%) and mixture ratio (1.65-2.1) are kept under consideration.

Conventional apogee engines have the required thrust (490 N, 110 lb), but at 0.7 MPa (100 psia) chamber pressure, they are too heavy (~3.6 kg) and quite large (0.5 m long). Missile defense programs during the 1980's demonstrated the same thrust level from compact short-lived hardware over 10 times lighter, at greatly increased pressures. Several rocket companies have begun to develop "compact apogee engines," with chamber pressures near 3.5 MPa (500 psi). Their goals are to quarter the mass, halve the size, and increase Isp above 325 s without sacrificing the multi-hour lifetime. The most optimistic mass considered here (0.9 kg) is consistent with this, but it could be above 1.2 kg.

Liquid tanks

The 0.4 m (16 in) diameter tanks shown in Figures 5 & 6 are sized to carry 96 kg of NTO and 58 kg of MMH (equal volumes) with a few percent ullage. Increasing the mixture ratio would raise bulk density, and increasing Isp reduces propellant mass. The resulting volume reduction is assumed to provide mass compensation for the structural overhead of accommodating unequal tanks. Equal tanks might still make sense, since the penalty of pressurizing extra fuel ullage is minimized by the pump-fed system.

In a deviation from spacecraft tradition, the tanks are fabricated from commercially pure titanium sheet (possibly 15-3-3-3- alloy), rather than being forged from 6Al-4V-Ti and subsequently machined. Weaker metal permits cold forming of hemispheres. The wall thinness is limited by fabrication capability, so the reduced strength is acceptable on the scale of interest at 0.35 MPa (50 psi) internal pressure. Rolled sheet has far less thickness variation than machined forgings, and fabrication of sheet tanks is cheaper once tooling exists. Technology heritage began circa

1960 with the SR-71 aircraft (c.p. Ti), and the Atlas and Centaur stages (steel sheet tanks). The latter were as thin as 0.25 mm (.010 in) for a 10 ft diameter! Of more particular relevance, 0.2 mm (.008 in) thick titanium was butt-welded by a Nd-YAG laser to make the 15 liter hydrazine tank in Reference 3. The proposed tanks would use this demonstrated wall thickness, at an operating stress of only 175 MPa (25 ksi). The mass of each sphere is 0.5 kg.

Helium subsystem

Helium containing small amounts of oxygen and hydrogen is kept at 233 K, in thermal equilibrium with the propellants. The best masses for the main regulator and catalytic heater sum to 1.4 kg, as scaled from Reference 2 assumptions for 500 lb thrust (technology advances are required). Also per assumptions of ongoing Mars ascent studies, the gas temperature is raised by 600 C, half of which is lost. The effective temperature in the pump cylinders is thus 533 K (500 F), which requires 147 moles of gas to displace all the propellant (130 liters) at 5 MPa.

The tank pressurant regulator operates at low flow and a low imprecise pressure, so its mass is estimated at 100 grams. Helium entering the propellant tanks will cool to a lower temperature than in the pumps. In particular, heat from the helium would initially vaporize some of the volatile oxidizer, until stopped by thermal stratification within the ullage. In consideration of the boiling points (289-323 K) of candidate oxidizers at the tank pressure, 333K is assumed to be the average ullage temperature for the system. Tank pressurant totals 16 moles.

Including helium tank residual and pump losses, 200 moles of gas is budgeted. It must be stored at 300 K for earth launch. The Reference 2 assumption for composite overwrapped pressure vessels (COPV's) is PV/W = 1.3 M inch at burst with a 1.5 safety factor. This 7.8 liter 65 MPa tank would mass 2.3 kg (heavier with current capability).

Pumps

Figure 7 shows some details of the quad piston pump. It is a lightweighted modified version of the hydrazine pump from Reference 3, specially designed for warm helium operation. All liquid manifolding with eight integral check valves is tightly packaged in a central aluminum block, from which four cylinders protrude. Gas and liquid are separated by pistons (1 of 4 shown), which have rods to automatically switch the gas valves depending on piston position. Each 3-way intake-exhaust valve is integral with its gas cylinder head (1 of 4 indicated in the sketch).

The original hydrazine pump quad used bolt-on liquid cylinders. Here, they are machined as part of the block, which eliminates flange mass and shrinks overall dimensions while preserving the one inch bore and half inch stroke. A numerical solid model indicates a 90 gram mass for the liquid block and cylinders, whereas this subassembly mass was previously 187 grams.

For operation in a gas-generator cycle rocket engine, the hydrazine pump required a differential-area piston and bolted-on hot gas cylinders larger than the liquid cylinders. The use of inert gas, at a moderate temperature, from a separate

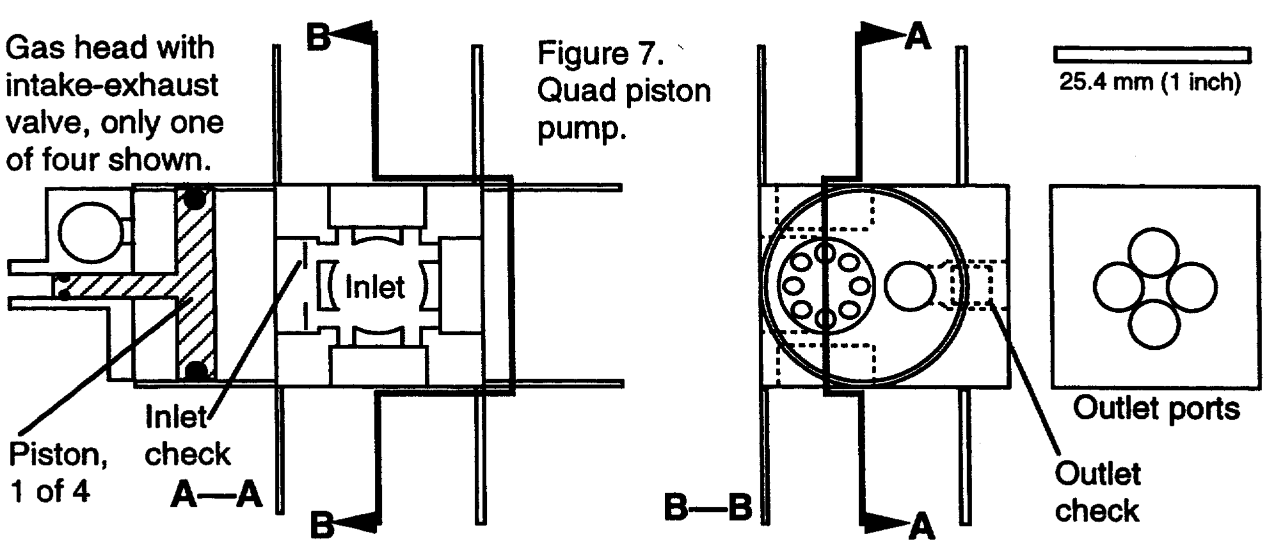


Figure 7. Quad piston pump.

source, avoids these extra cylinders. The gas temperature and short life is consistent with the use of aluminum and fluoroelastomer seals throughout the pump, which further reduces mass and eliminates the previous gas leakage past solid graphite seals. Heads are welded on in the present design. As a result of all changes, the original 365 gram pump mass is expected to fall to 200 grams for the same pressure and flow.

In operation, opposite pistons stroke toward each other to minimize vibration. If downstream valves are modulated, pump speed shifts, and the pistons stop at full pressure if all TCV's are shut. The original quad was bench tested to 370 cc/sec of water flow at high pressure. The Mars ascent stage requires each pump to deliver only 130 cc/s from its tank, at 6 Hz (24 cylinder expulsions/s for each quad). The lower flow greatly reduces pressure drops through the valves, in addition to providing margin relative to maximum flow (limited by refill time from low tank pressure).

Additional flow components

Thrust control valves, pump valves, and helium regulators were accounted for above. Miniature gas jet thrusters for attitude control on orbit can be 20 grams each. Additional parts are required for propellant and pressurant filling and isolation, for earth safety and long term storage. A particular requirement is to isolate any pump seals which lack long-term compatibility with the oxidizer. This may require frangible seals in the pump cylinders, which are broken by the high pressure helium. One kilogram is assumed for the mass of miscellaneous valves and burst disks, as well as tubing, filters, and fittings.

Electrical items

In accordance with ongoing Mars ascent studies at JPL, the 30 kg payload includes avionics, electrical power, and power electronics to drive the propulsion valves. Per Reference 2, 1% of the wet stage mass would be wiring, based on established practice. The absence of liquid attitude thrusters may permit this to be halved.

Structure, fairing, and insulation

Spacecraft structure is typically 5% of gross mass, but this is not true of stages which use tanks as primary structure. For example, the S-IC had 35 tons of non-tankage structure, 1.3% of the entire Saturn V stack's liftoff mass. This included aerodynamic fins, engine fairings, an intertank, and forward skirt. Some large stages have less structure. Herein, the range from 2 kg (1%) to 10 kg (5%) is considered. Detailed structural studies supported by development tests are required to narrow this range of uncertainty. Insulation is estimated at 1 kg.

Mass and performance summary

Table 1 lists components in the order discussed above, along with their mass goals and ranges for some. Numbers in square brackets represent an implementation with off-the-shelf pressure-fed spacecraft propulsion components. Table 2 presents Δv calculations for different assumptions of specific impulse, with and without expending helium.

Table 1. Mass, kg	goal	high	[press-fed]	
Thrusters	3.6	5.0	[~14]	
Liquid tanks	2.0		[~8]	
He tank	2.3	3.0	[~4]	
He regulators	0.5	1.5		blanks =
He heater	1.0			same as at
Pumps	0.8		[0]	immediate
Flow comp.	1.0			left
Wiring	1.0	2.0		
Structure	2.0	10.0		
Insulation	1.0			
Total dry	15.2	27.3	[~43]	
Helium mixture	0.9	(only 0.3 is residual)		
Residual liquid	1.0	3.5	(mostly mixture ratio reserve)	
Stage at burnout	16.5	31.1	[~47]	
Payload @ 315 s	30	15	[~0]	(153 kg expended liquids)
Payload @ 330 s	33	18	[~3]	(150 kg expended liquids)

Table 2. Calculated performance, 200 kg liftoff mass.

Liquid expended, kg @ Isp	Gas expended, kg @ Isp	Δv, m/s
153 @ 315 s	none	4470
153 @ 315 s	0.6 @ 100 s	4498
153 @ 330 s	none	4683
153 @ 330 s	0.6 @ 100 s	4712
150 @ 330 s	none	4483
150 @ 330 s	0.6 @ 100 s	4510

The optimistic assumptions are consistent with the 30 kg payload goal (Table 1 left column). Of great significance is the uncertainty in inert mass, particularly that of non-tankage structure. All the high estimates with pump-fed operation would halve the payload (or liftoff mass could be doubled to 400 kg). If conventional pressure fed parts are used along with these other high mass estimates, the payload is essentially eliminated (Table 1 right column).

Per Table 2, increasing specific impulse from the near-conventional 315 s up to 330 s would yield a relatively modest improvement, by permitting inert mass to grow by 3 kg (or ~200 m/s more Δv). An even smaller improvement results from pump exhaust thrust (but it could be significant with heavier gas).

Discussion

The authors have not reached agreement on what constitutes reasonable mass goals for items listed in the middle column of Table 1. For example, the high estimate for residual liquids is typical of exploration spacecraft at 2.5%, but earth launcher stages do much better. The mass of structure is driven by Mars atmospheric entry decelerations of ~30 earth g, which earth launch vehicles never face. While ascent dynamic pressure and aero heating peak at only ~240 newtons per square meter (~5 lb per square foot) and 3 suns respectively, shock wave impingement is a concern which requires fairing mass to be carried part way to Mars orbit. The latter is not accounted for in the smallest structural allotment above. However, if structure must be 5% of liftoff mass, something else needs to be tried.

This paper considered the application of reduced-risk pump technology to an accepted stage configuration. The result is a major reduction in inert mass, but most likely not enough. More aggressive options can help to reduce burnout mass. For example, all helium component masses would shrink to roughly one fifth, if a 2.5 liter tank of hydrazine is provided for driving pumps (~2 kg net advantage).

Additional ideas beyond this paper require broadening the scope of the problem. Designing a lander to accommodate an improved ascent vehicle aspect ratio would be a worthy endeavor. A cylindrical shape on a tilt-up mechanism could save ~3/4 of the ~400 m/s drag loss, for a ~4 kg inert mass advantage. With stacked cylindrical tanks doubling as primary structure, it may be easier to reach structure & fairing fractions achieved by earth launch stages.

There would be numerous advantages to sending the propellants in separate, dedicated "Mars GSE" tanks. This would greatly reduce Mars atmospheric entry loads for the tanks and structure. There would be no earth launch range safety concerns for the thin tanks, so a common bulkhead integrated design could be implemented. Long term propellant isolation devices, as well as tank insulation and heaters with wiring, would not burden the Mars ascent. The automated propellant transfer capability would be relevant to other proposed missions.

Traditionally, propulsion designs are based on proven capability, and sized (with margin) for a given payload and maneuvering requirement. Total mission mass and cost are calculated results of this process. The different philosophy of the present work is to ask, "What needs to be done to permit low-cost Mars sample return?" In this context, the highly successful Mars Pathfinder mission of 1997 provides a working definition of "low cost." Note that a stage developed to meet the Mars return need would enable other low-cost exploration such as lunar return. Even smaller missions could reach the lunar surface from GTO, or interplanetary trajectories from LEO.

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