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UCRL-CONF-231032

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June 19, 2007

NASA Science Technology Conference 2007
NSTC-2007
Adelphi, MD

Paper Number D11P1

NASA = National Aeronautics and Space Administration

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Pump Fed Propulsion for Mars Ascent and Other Challenging Maneuvers

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***Abstract*-Returning Mars geology samples to Earth within science mission budgets requires a miniature launch vehicle (100-200 kg) for ascending from Mars to an orbital rendezvous. A Mars Ascent Vehicle must deliver a velocity change exceeding 4 km/s within minutes, entirely outside the capabilities of satellite propulsion. A possible solution is to scale down liquid launch vehicle principles to achieve stage propellant mass fractions near 90 percent. Feeding a high-pressure engine from thin-walled low pressure tanks permits stage hardware to be sufficiently lightweight and compact, if very high performance pumps can be made available. NASA's Mars Technology Program has funded refinement and testing of a miniature piston pump, powered by reacted propellant. A pump-fed bipropellant rocket stage remains to be developed. The technology could also benefit other future lunar and planetary science programs.**

INTRODUCTION

Space science missions have benefited greatly from the fact that their propulsive maneuvers are often similar to what is done routinely with communications satellites. Reaching geostationary transfer orbit (GTO) is almost as challenging for launch vehicles as Earth escape is. Interplanetary course corrections need small velocity changes at low acceleration, which is analogous to satellite orbit maintenance. Orbit insertion at planets is done with burns that last on the order of an hour to change velocity by roughly 1 km/s, much like the apogee burn from GTO into geostationary Earth orbit (GEO).

Consequently, it has been easy for the community to become accustomed to the generalization that the needs of science spacecraft can be met readily by adapting hardware from the satellite propulsion industry. The epitome of this phenomenon is perhaps that low-thrust electric propulsion, ideal for satellite station keeping, has been applied to heliocentric course changes. However, launching from a planet presents an entirely different challenge, so propulsion technology should also be advanced in another direction.

The primary difference is that ascent and descent maneuvers must be accomplished within minutes, not hours or longer. Simply adding more engines to increase acceleration does not solve the problem, because the extra rocket hardware mass reduces either the velocity change capability or the payload. Instead of an evolutionary advance from satellite propulsion, a revolutionary technology advance is needed to permit ascent and descent maneuvers on a scale small enough for science missions.

This paper reviews the requirements for launching from Mars, with comparisons to descent (or ascent) at Earth's moon and Europa. The importance of a high propellant fraction is then quantified, followed by a discussion of launch vehicle design principles. A concept for a miniature pump-fed rocket stage is introduced. Achieving a 90 percent stage propellant fraction is thought to be possible on a 100-kg scale, including sufficient thrust for lifting off Mars or visiting the various moons of interest to future solar system exploration.

The second half of this paper describes recent progress at LLNL toward a miniature pump-fed rocket capability. Pump testing was supported by NASA's Mars Technology Program, with oversight by the JPL Propulsion Technology Program. The project was very successful and a comprehensive final report was written [1] in addition to detailed technical papers [2,3].

A 300-gram reciprocating piston pump was fabricated and powered by high pressure gas at elevated temperatures, while pumping water over a duration similar to what Mars ascent requires. Demonstrated pressure and flow are sufficient to feed a 1000-N bipropellant rocket engine which, by virtue of high chamber pressure, would be 5 times more compact and lightweight than a conventional 490-N engine used for orbit insertion. A pair of the pumps delivering hydrazine and nitrogen tetroxide would be powered by decomposed hydrazine representing only 2% of the total propellant mass. This measurement indicates that a pump-fed system would be efficient enough to obtain a net increase in specific impulse, compared to low-pressure satellite engines.

REQUIREMENTS FOR ASCENT AND DESCENT

Table 1 lists velocity changes and times permitted for some familiar maneuvers and anticipated future needs. Compared to the three orbit insertion maneuvers, the next three maneuvers of interest are more challenging. Beyond those, ascending from

TABLE 1.
ASCENT AND DESCENT ARE CHALLENGING MANEUVERS.

Maneuver	ΔV, km/s	Burn duration
Satellite apogee burn into GEO	1.5	2 to 5 hr
Cassini Saturn orbit insertion (in 2004)	0.6	1.5 hr
MRO orbit insertion at Mars (in 2006)	1.0	0.5 hr
Earth return from low Mars orbit	1.5 to 2	0 to 10 min
Lunar descent from orbit (or ascent)	1.8 to 2	0 to 5 min
Europa descent from orbit (or ascent)	1.6 to 1.8	0 to 5 min
Mars ascent to a low orbit (500 km)	4.2	4 min

Mars is in a class by itself, considering the need to deliver more than half the velocity of an Earth orbit, within a few minutes.

A higher Δv requires a greater amount of propellant relative to the rest of a spacecraft or launcher stage, while a shorter maneuvering time requires higher thrust. Given that the mass of larger engines tends to displace propellant, it is extremely challenging to obtain large velocity changes quickly.

Solid rocket motors naturally have high thrust due to the burn rate of solid propellant, so extra engine mass is not needed. However, conventional solid rocket technology does not offer an ideal solution. The specific impulse of solid rockets (~285 s) tends to be lower than that of liquids (~315 s). A solid rocket motor needs an additional means for steering, e.g. the Surveyor of the 1960's used a 620-kg solid motor combined with a 110-kg liquid vernier system for landing on the moon. For much smaller vehicles, it is less straightforward to include two separate propulsion systems. Moreover, the high thrust of solid motors requires any directional control system to be more powerful than would be needed for an equivalent liquid rocket.

For the particular case of Mars ascent, a short burn time has another disadvantage. A Mars ascent vehicle (MAV) having excessive thrust would experience significant aerodynamic drag while reaching high speeds low in the Mars atmosphere. Fig. 1 compares preferred trajectories (least Δv) for solid and liquid Mars ascent [4]. While higher thrust always improves maneuvering efficiency in a vacuum, acceleration on the order of Earth gravity is better for a MAV.

In rough numbers, Mars science missions cost a million dollars per kilogram of useful apparatus landed. The largest Mars landing system designed to date will deliver almost a ton (Mars Science Lab, to be launched in 2009). Considering realistic limits on cost and scale, a 500-kg MAV for example would be prohibitively heavy. From the standpoint of mission mass, the MAV is no different from a science payload delivered to the surface of Mars. In particular, a smaller MAV permits a larger rover or drill for sample collection. For all these reasons, dedicated technology development should be guided by the notion of a 100 to 200 kg MAV.

PROPELLANT FRACTIONS AND STAGING

The rocket equation expresses velocity change as a function of specific impulse and the mass ratio over a propulsive burn. Some algebra permits expressing payload mass as a function of the number of stages and the propellant mass fractions of each stage. Fig. 2, adapted from [4], shows the staging trade for a small MAV which has a total mass of 100 kg at Mars departure.

Each location on the graph of Fig. 2 represents a particular combination of first stage propellant fraction and second stage propellant fraction, defined as useable propellant mass divided by the total mass of propellant plus propulsion hardware. One set of contours shows the mass of payload. The latter includes navigation, communication, etc. for the purpose of this rocket trade calculation. Another set of contours shows the mass of upper stage rocket hardware.

The graph was plotted this way because it will be more difficult to achieve a high propellant fraction for a much smaller upper stage than for the first stage. The upper stage actually vanishes in the triangular region at the right edge of the graph, and there is an additional region in which the upper stage would be exceedingly small. If it happens that a first stage can be 90 percent propellant, while an upper stage can't even reach 70 percent, then a single stage MAV would be the practical choice. A single stage solid is not an option. The main lesson from Fig. 2 is that a MAV needs extremely good propellant fractions, beyond the limits for conventional technology.

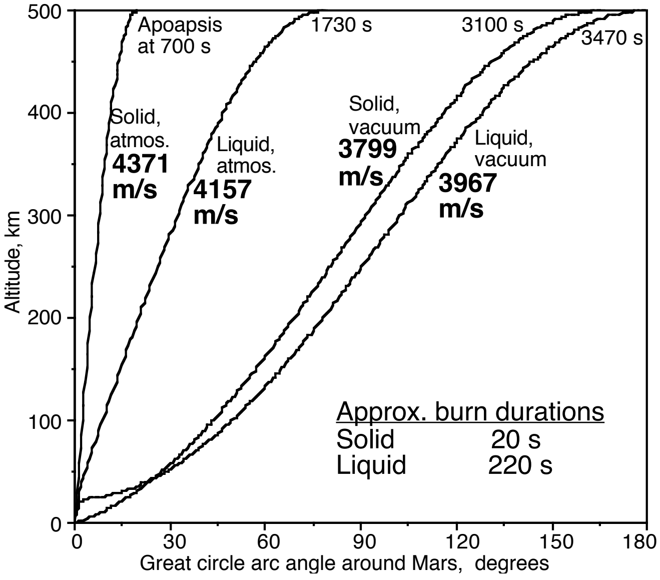

Fig. 1. Mars ascent at moderate thrust offers a more efficient trajectory than a fast-burning solid rocket.

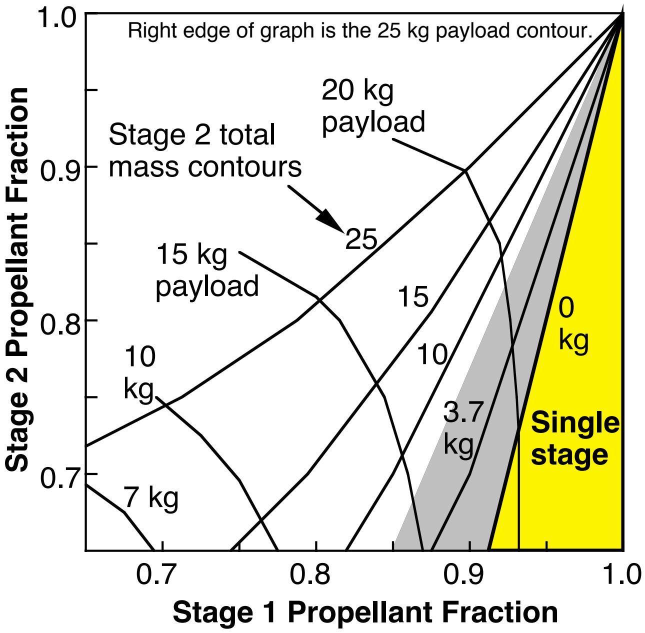

Fig. 2. Staging trade for a 100-kg liquid MAV.

PUMP FED PROPULSION

The stages of Earth launch vehicles routinely achieve 90 percent propellant fractions on a scale of 100 tons. All that is needed is to implement the same principles three orders of magnitude smaller in mass, or one-tenth linear scaling. Launch vehicle stages use thin-walled low-pressure tanks as the primary structure for the vehicle. Typically, such tanks operate at a few atmospheres pressure and hold 100 times their own mass in propellants. This mass efficiency is theoretically independent of scale, so the only difficult question is whether the walls can be made thin enough. The answer is probably yes for a 100-kg MAV (0.2 to 0.5 mm tank walls).

Launch vehicle engines use chamber pressures ranging from 50 to hundreds of atmospheres. Such high pressures permit thrust chambers to be compact and weigh less than one percent of thrust. In contrast, satellite engines operate at or below 10 atmospheres in the combustion chamber. They are fed directly from tanks pressurized to about 20 atmospheres.

Both of these pressures require a mass compromise for the respective components. Small satellite tanks and their supporting structure weigh roughly 10 percent of the propellants. Engines used for the orbit insertions in Table 1 tend to weigh about 5 to 10 percent of thrust.

Luckily, the propulsion industry has built small scale high-pressure engines for other purposes. Weight can be less than one percent of thrust [5], but only if propellants are supplied at pressures exceeding 50 atmospheres (735 psi).

Launch vehicles use high performance centrifugal pumps spun by turbines, powered by reacted propellant gases at high temperatures and pressures. Such pumps weigh less than one percent of the thrust they support, so that a complete pump-fed engine weighs about 1 to 2 percent of thrust.

Fig. 3 shows a conceptual design for a rocket stage sized for a 100-kg MAV. Table 2 lists mass goals, assuming that a single stage vehicle is feasible. Turbopumps become inefficient and relatively heavier when scaled down, so a key challenge is to develop a small high pressure pump that can be driven by propellant energy. As on launch vehicles, any other motive power source such as batteries and electric motors would be prohibitively heavy. In Fig. 3, each pump is shown closely associated with its propellant source tank, because that arrangement is the best way for liquid stored at low pressure to refill the chambers of a reciprocating piston pump.

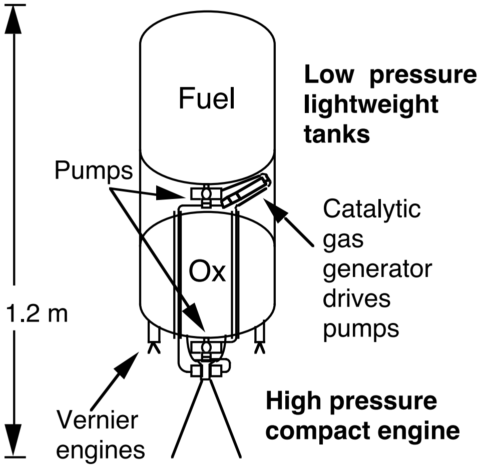
Fuel
Low pressure lightweight tanks
Pumps
Catalytic gas generator drives pumps
Ox
1.2 m
Vernier engines
High pressure compact engine

Fig. 3. Pump-fed stage concept.

TABLE 2.
SUGGESTED MASS BUDGET FOR A SINGLE STAGE MAV.

Component	Mass, kg
Tanks (50 psi) and vehicle structure	4.0
Pumps, gas generator, and heat exchanger	1.1
Bipropellant engine (~500 psi)	1.0
Attitude control thrusters	0.7
Tank pressurization, fill valves, misc.	1.2
Thermal insulation	0.5
Total propulsion hardware mass	8.5
Propellants (1 kg can remain unused at ISP = 315 s)	75.0
Non-propulsion (sample capsule, avionics, batteries, etc.)	15.0
Margin	1.5
Total mass	100

In Table 2, it is assumed that the MAV would produce a total velocity of 4157 m/s per Fig. 1, which includes 257 m/s to circularize the orbit with an engine restart after coasting to 500 km altitude. The tanks and other structure are conservatively estimated at 5 percent of the propellant mass, to allow for thicker tank walls. It is hoped that the tanks can be lighter, which would offer a little more margin elsewhere. The component mass list here would have to be updated as the technology is refined. The whole mass budget could be scaled up some if necessary, e.g. to carry heavier avionics etc.

RECIPROCATING PUMP

A reciprocating pump powered by pressurized gas is conceptually very simple. In Fig. 4, four pumping cylinders surround a central liquid manifold section which houses pairs of check valves (not shown) oriented for inlet and outlet flow directions. A larger gas power cylinder is attached to each pumping chamber, and pistons separate the gas and liquid.

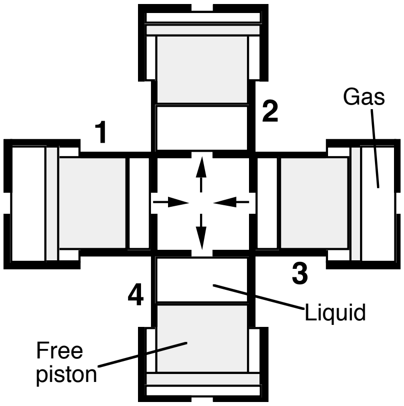

Fig. 4. Cross section diagram of four-chamber pump.

There is no need for shaft power or any other rotating parts. In the absence of such inertia, a reciprocating pump can start and stop very quickly. At the instant depicted in Fig. 4, piston numbers 1 and 3 are moving toward each other as they transfer the gas pressure load to the liquid being pumped. Pumping chambers 2 and 4 are being refilled at low pressure as their pistons move outward in the direction of the arrows. Gas valves (not shown) control intake and exhaust flows with pneumatic switching linked to piston position. The result is that pairs of opposite cylinders exchange roles with their adjacent cylinders to deliver continuous high pressure liquid flow from a low pressure tank. While only two pump chambers would work, opposed pistons balance moving mass to greatly reduce vibration.

By virtue of the larger gas cylinder diameter, propellant is pumped to a pressure which exceeds the gas pressure. This amplification feature accommodates pressure losses as some of the hydrazine fuel flows through a catalytic reactor that generates the gas which powers both pumps in Fig. 3.

The hardware development challenge has been to translate Fig. 4 into a reliable, efficient, lightweight, powerful machine that can be compatible with both hydrazine fuel and nitrogen tetroxide oxidizer. The most relevant measure of fluid power (pressure times volume flow) relative to mass is the ratio of engine thrust to pump weight, at a pressure that permits the thrust chamber to be small. The relevant measure of efficiency is the quantity of decomposed hydrazine (including leakage) needed to power the pump, relative to the pressure and flow of pumped liquid. The density and hence consumption of the gas varies inversely with temperature, so operating the gas power cylinders at elevated temperatures is highly desirable. High temperature functionality and longevity of the gas intake-exhaust valves has presented the most difficulty to date.

Fig. 5 shows the 300-gram pump that was built and tested at LLNL with NASA Mars Technology funding. The large port at the center is the liquid inlet. The straight tubes on top distribute gas to the four intake valves. The thin bent tubes convey pneumatic signals to control the 3-way gas intake-exhaust valves. Other than the various gas tubes, moving valve parts, and seals, the entire pump is made of aluminum for low mass. The liquid cylinders are all machined as one piece with the central manifold block that contains eight check valves. The high heat conductivity of aluminum is taken advantage of for cooling leaktight soft seals, by conducting heat into the pumped liquid. In particular, the gas power cylinder walls remain cool enough for elastomeric gas piston seals. This latest leak-tight pump is an evolved version of earlier designs, one of which functioned in a monopropellant hydrazine flight experiment, albeit with undesirable warm gas leakage past hard uncooled piston rings [6].

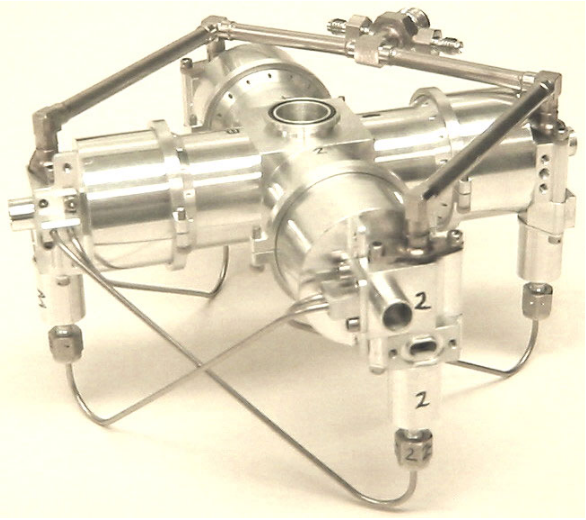

Fig. 5. Pump assembly (5.5 inches across) includes all valves.

PUMP TEST RESULTS

The test article shown in Fig. 5 has pumped water, while powered by room temperature helium and separately by a mixture of steam and oxygen for elevated temperature testing. The cold gas testing was initially done in order to verify functionality and low leakage, then the capabilities and limits of the pump were mapped. Fig. 6 shows mean pressures as a function of flow, which indicate an acceptably small reduction in the pressure amplification ratio as flow increases. The design point for a 1000-N engine is near the middle of the graph, so these data indicate a comfortable margin. The pump delivered more than its own mass in water each second near 800 psi, and the fluid power at the highest flow exceeds 1.75 kW (upper right test point plotted in Fig. 6).

A major functional difference compared to turbopumps used on large rocket engines is that a reciprocating pump has no efficiency losses when throttled down. This pump cycled at just over 12 Hz at the highest flow, and frequency falls smoothly and continuously as a downstream valve is gradually shut. The pump maintains full pressure at zero flow, with virtually no gas consumption. Besides the roll-off of discharge pressure, the maximum flow is limited by the refill rate from the tank. Given that it is critical to maintain a positive tank pressure margin, the threshold tank pressure was characterized as a function of flow. Fig. 7 indicates a comfortable margin for feeding a 1000-N engine from 50 psi tanks.

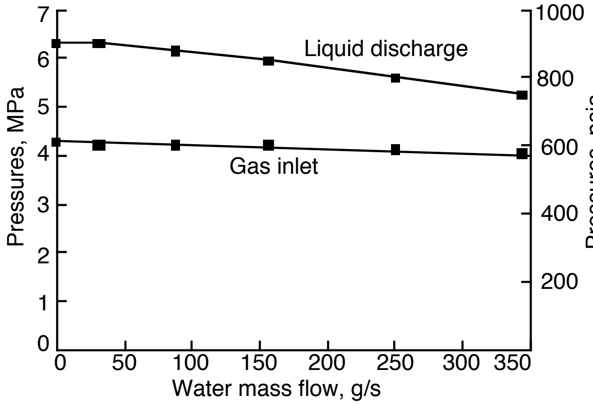

Fig. 6. Mean pressures fall slightly as flow increases.

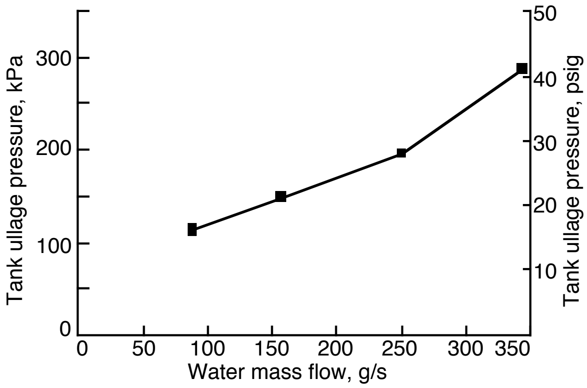

Fig. 7. Minimum tank pressures required.

Decomposed hydrogen peroxide was used for warm gas tests, because its low toxicity permits a faster cycle of test runs followed by hardware changes and further testing, compared to hydrazine. It was possible to do benchtop-type testing in a wet lab. Hardware was rinsed when necessary, but no special decontamination procedures were required. An unintended advantage of using steam is that its heat of condensation on the piston and cylinder walls provided a worst-case demonstration of thermal loading, approximately 5 kW.

Fig. 8 is the data from one of a series of five identical test runs. In addition to recording temperatures (top traces) and pressures (middle traces), the apparatus was configured to precisely measure the quantity of liquid H_2O_2 remaining in the gas generator feed tank (lower trace). The state of the gas was analyzed to determine the mean working molecular weight, including the fractional steam condensation in the pump's power cylinders. Translating the data to the lower molecular weight of decomposed hydrazine indicates that 98 percent of the propellant would reach the thrust chamber in the proposed system, i.e. only 2 percent is needed to drive the pumps.

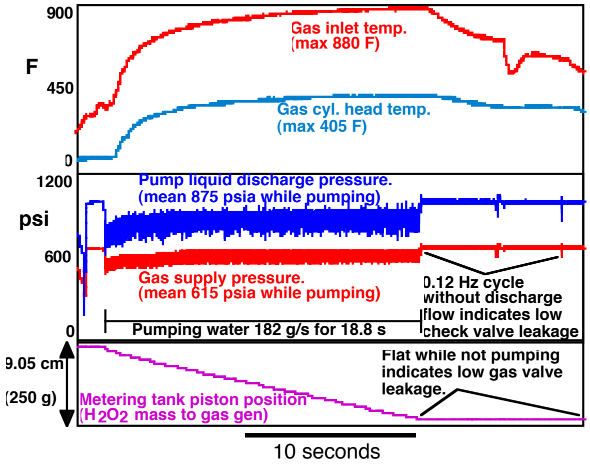

Fig. 8. Warm gas powered pump test.

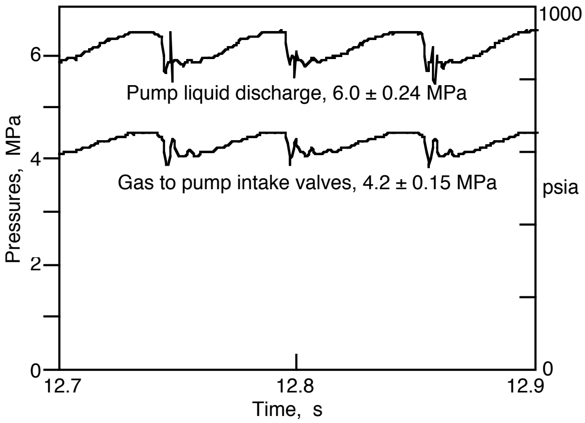

Fig. 9. Close view of pressure data from Fig. 8.

Fig. 9 is a close view of pressure data from Fig. 8. The standard deviations are only 4 percent of the mean pressures. Both pressure waves are close to square when helium is used. Thus it is expected that smoother pressures (than Fig. 9) will result from the low density of decomposed hydrazine, compared to the steam-oxygen mixture used here.

During a separate series of thermal limit tests, gas temperatures as high as 1100 F were measured at the feed port (fitting at upper right in Fig. 5). The most vulnerable part of the design turned out to be the warm gas intake-exhaust valves, so several design iterations were tested. After various changes were made over the course of a year of testing, each major aspect of the pump design had accumulated a different total demonstrated lifetime.

Based on the assumption that two pumps in a 100-kg MAV would each deliver 40 liters of liquid, the cylinders, pistons, check valves, and piston seals experienced 2.4 Mars ascent lifetimes. Of that total, 1.1 lifetimes were run at elevated temperatures. Fig. 10 shows the gas end of one of the pistons after the first 1.3 lifetimes, which included the thermal limit tests. The main warm gas seal showed negligible wear.

The last design iteration of the gas intake-exhaust valves survived 0.8 lifetimes running hot, plus an additional 0.2 lifetimes powered by cold gas. At the end of the test program, the pump remained functional with some degradation in the pressure traces. The cause has not been determined, but most probably some further improvements in the gas valves would be advisable.

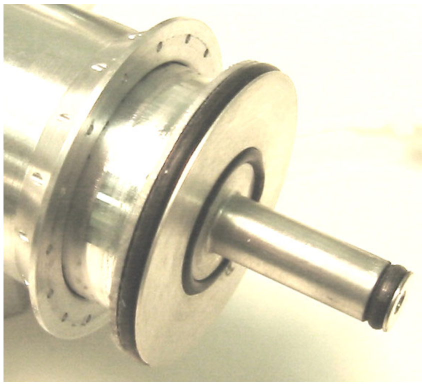

Fig 10. Piston with gas seals after 2000 cycles.

The end result of the NASA-funded testing is that a flight-like pump design was declared to have reached Technology Readiness Level (TRL) 4. Testing in a vacuum with propellants is needed to reach TRL 6. This next phase will most likely be more costly but less technically risky than what has already been accomplished.

REMAINING DEVELOPMENT CHALLENGES

An overall technology development goal is to demonstrate a complete integrated pump-fed rocket stage on the scale of interest, using hardware that is essentially flight weight. Separate component and subsystem efforts remain to be accomplished before that last step. First, the pump needs to be run in a vacuum, powered by a hydrazine gas generator. A heat exchanger would cool the gas to deliver decomposed hydrazine at about 900 F. Flight-weight gas supply components would ultimately be needed.

Pumping liquid hydrazine is considered straightforward, but pumping oxidizer needs to be demonstrated. Technical issues to be addressed for NTO include both its physical properties and chemical compatibility with selected seals. The oxidizer tank pressure may need to be a little higher to accommodate a greater vapor pressure than hydrazine. Preliminary testing of special seals indicates that the oxidizer pump design might need to differ slightly from the fuel pump.

A compact high-pressure engine needs to be developed. It has been known for decades that this is possible, but there has been no application for such a satellite engine in the absence of pumps, because high pressure tanks would be prohibitively heavy.

Designing and testing flight-like low-pressure tanks and their connecting stage structure would be a low-risk hardware effort to support the feasibility of the mass budget in Table 2, while building confidence in the system concept.

Today's satellite and spacecraft propulsion systems are rarely tested at the system level before flight. Component tests are sufficient, both because conventional technology is highly refined, and because a pressure-fed system is usually predictable enough to be functionally just the sum of its parts.

In contrast, the proposed miniature pump-fed stage has system-level complexities above and beyond the functionality of individual components. Satellite engine test facilities isolate propellant source tanks from engine test cells, both for safety and to reduce facility damage in the event of a mishap. The pumps need to be closely integrated with their low-pressure propellant tanks, so extra facility costs may be associated with development testing of a complete pump-fed propulsion system.

DISCUSSION AND CONCLUSIONS

A miniature pump design has been extensively tested and refined. The flight-weight hardware survives its own challenging thermal environment, and flow is sufficient for two such pumps to feed a 1000-N bipropellant engine at high pressure.

While the size tested would work for a notional 100-kg Mars ascent vehicle, the technology could be scaled up or down significantly. Other possible mission applications include lunar descent and ascent, Europa descent and ascent, and Earth return from Mars orbit. Per Table 1, these three have very similar maneuvering requirements, so it is possible that one propulsion system design could be used for three flight programs.

While Mars descent needs only a small velocity change, its high thrust-to-weight requirement could justify a pump-fed system that would reduce the total weight and size of engines. A rocket-powered Mars airplane might benefit from similar pumps immersed within wing tanks that are just slightly pressurized.

Technology development for ascent and descent on a small scale has been an orphan problem in the past, in the sense that there has been no dedicated technology development funding. It is hoped that this situation can continue to improve, because the maneuvering needs for high-profile future missions are beyond being merely challenging for today's proven propulsion technology.

ACKNOWLEDGMENT

Thanks to Ron Reeve of the JPL Propulsion Technology program. Encouragement and suggestions from other individuals too numerous to name is greatly appreciated.

This work was performed under the auspices of the U.S. Department of Energy, by the University of California, Lawrence Livermore National Laboratory under Contract W-7405-Eng-48.

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