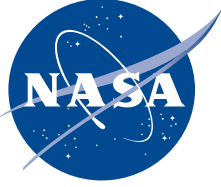


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Powdered Magnesium—Carbon Dioxide Rocket Combustion Technology for In Situ Mars Propulsion

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September 2007

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LIST OF ACRONYMS AND SYMBOLS

Al	aluminum
Al ₂ O ₃	aluminum oxide
Ar	argon
B	boron
B ₂ H ₆	diborane
B ₂ O ₃	boron oxide
Be	beryllium
BeH ₂	beryllium hydride
C ₃ H ₈	propane
Ca	calcium
CO	carbon monoxide
CO ₂	carbon dioxide
H ₂ O	water
ISRU	in situ resource utilization
Kr	krypton
Li	lithium
LOX	liquid oxygen
MEPAG	Mars Exploration Program Analysis Group
Mg	magnesium

LIST OF ACRONYMS AND SYMBOLS (CONTINUED)

MgH ₂	magnesium hydride
MgO	magnesium oxide
MSFC	Marshall Space Flight center
MSR	Mars sample return
N ₂	nitrogen
N ₂ O ₄	nitrogen tetroxide
Ne	neone
O ₂	oxygen
O ₃	ozone
PDFB	positive displacement fluidized bed
Si	silicon
SiH ₄	silane
SiO ₂	silicon dioxide
Ti	titanium
UDMH	unsymmetrical dimethylhydrazine
Xe	xenon
Z	zirconium

NOMEMCLATURE

I_{sp}	specific impulse
m_{CO_2}	first-stage carbon dioxide mass
m_{d_1}	first-stage dry mass
m_{d_2}	second-stage dry mass
m_{ex}	mass of CO ₂ extraction system
m_{f_1}	first-stage propellant mass
m_{f_2}	second-stage propellant mass
m_{Mg}	first-stage magnesium mass
m_p	payload mass
m_{tmi}	trans-Mars injection mass
O/F	oxidizer-to-fuel mixture ratio
t	time
$V_{e,i}$	exhaust velocity for the i th stage
ΔV_i	velocity increment for the i th stage
ΔV_1	first-stage velocity increment
ΔV_2	second-stage velocity increment
ε	structural coefficient

TECHNICAL PUBLICATION

POWDERED MAGNESIUM—CARBON DIOXIDE ROCKET COMBUSTION TECHNOLOGY FOR IN SITU MARS PROPULSION

1. INTRODUCTION

It is widely recognized that sustained, long-term exploration of Mars will be highly dependent upon in situ resource utilization (ISRU) technologies.^{1–3} Most martian ISRU concepts focus on propellant production techniques and compatible propulsion devices that would enable both long-range surface mobility and planetary ascent. These concepts typically involve processing of the martian atmosphere, the composition of which has been well established through prior landing missions. The composition of the lower atmosphere, as deduced from Viking lander measurements, is as follows: 95.32 percent carbon dioxide (CO₂), 2.7 percent nitrogen (N₂), 1.6 percent Argon (Ar), 0.13 percent oxygen (O₂), 0.07 percent carbon monoxide (CO), and 0.03 percent water (H₂O), with trace amounts of neon (Ne), krypton (Kr), xenon (Xe), and ozone (O₃).

The predominance of CO₂ in the atmosphere has led to several intriguing possibilities. One long-standing suggestion has been to transport a small chemical processing plant to Mars for the manufacture of methane and O₂ from in situ CO₂ and H₂O resources.⁴ A variation on this scheme, which avoids the need to extract H₂O from the arid martian environment, would require transporting a small feedstock of hydrogen from Earth for the production of methane and water in a Sabatier reactor.⁵ The O₂ required for methane combustion would be derived from electrolysis of the manufactured H₂O.

Implementation of such an approach, although fundamentally sound, would be extremely challenging from an engineering perspective, primarily due to the difficulties encountered with long-term storage and deep space transport of hydrogen. Moreover, the power required to produce and liquefy cryogenic O₂ and methane would likely be prohibitive under most mission scenarios.

A more contemporary approach to martian ISRU focuses on the production of CO and O₂ directly from the atmosphere via a zirconia electrolyzer, which requires no feedstock material. In this case, CO could be burned directly with liquid oxygen (LOX) yielding a specific impulse (I_{sp}) around 290 seconds. Alternatively, the LOX could be burned with methane delivered from Earth as a means of achieving higher propulsion performance. Despite these potential advantages, zirconia electrolyzer technology remains a risky proposition for most Mars mission scenarios. For instance, the devices must operate at very high temperatures (>1,000 K), consume prodigious amounts of power, and have yet to demonstrate a high level of reliability.

In considering the various alternatives, it should be noted that any approach based on in situ chemical processing faces a critical dilemma. That is, production of meaningful quantities of propellant during a

time-limited mission often leads to prohibitive surface power requirements; whereas, lowering propellant production rates to obtain more practical surface power levels can greatly extend mission duration (more than a year in most cases) and raises major reliability concerns.

A conceptually simpler approach would be to burn CO_2 directly as an oxidizer with some suitably energetic fuel, which could be delivered from Earth.⁶⁻¹⁹ For bipropellant rocket applications, it would only be necessary to collect and liquefy the CO_2 , thereby avoiding the major complications encountered with chemical processing plants. For the special case of an endoatmospheric propulsion system such as a ramjet, the imported fuel could be injected and burned directly with the resident CO_2 .

The use of CO_2 as an oxidizing agent may appear counterintuitive, but there are certain fuels that can remove the bound oxygen with a substantial release of energy. The most promising candidates are metals and their hydrides and mixtures with hydrogen compounds. Shafirovich et al. have evaluated the equilibrium thermodynamic rocket performance of a wider range of these candidates including lithium (Li), beryllium (Be), boron (B), magnesium (Mg), aluminum (Al), silicon (Si), calcium (Ca), titanium (Ti), zirconium (Z), beryllium hydride (BeH_2), magnesium hydride (MgH_2), diborane (B_2H_6), and silane (SiH_4).^{11,15} The results of these calculations, based on a chamber pressure of 10 atm and expansion to an ambient pressure of 0.01 atm, showed that the most promising fuels were Be, BeH_2 , Al, and Mg. Thermodynamic equilibrium rocket performance calculations by the authors, as shown in figure 1 for representative fuels, confirm these predictions.

Beryllium and its hydride yielded the highest theoretical I_{sp} , but the practical viability of these propellants is dubious due to extreme toxicity and unknown combustion characteristics. Al, Li, and B all had higher I_{sp} s than Mg for large oxidizer-to-fuel mass ratios, but are known to have poor ignitability characteristics and low combustion rates. Combustor slagging would also be more severe when burning Al due to the lower melting point of aluminum oxide (Al_2O_3) (2,315 K) compared to magnesium oxide (MgO) (3,098 K). As a result, Shafirovich et al. concluded that Mg was the most promising metal fuel for an engine utilizing CO_2 as an oxidizer, despite its relatively poor theoretical I_{sp} (220 s peak).^{11,15} It is readily ignitable in CO_2 at a relatively low temperature (around 1,000 K), exhibits a rapid combustion rate, and is known to burn to near completion when in particle form. As a means of improving performance while retaining these favorable characteristics, Shafirovich suggests that Mg/Al alloys should be investigated as a potential fuel as well.

These theoretical performance calculations also show that some liquid hydrides, such as B_2H_6 and SiH_4 , could serve as viable energetic fuels for CO_2 combustion, a point of significant practical importance since they could be injected and burned in a conventional bipropellant rocket configuration. This simplification in engine design led Zubrin to propose B_2H_6 as a fuel for a Mars ascent rocket.¹⁴ However, a potential difficulty with this propellant combination is the high mass fraction of condensed phase combustion products and the resulting impact on I_{sp} performance. Moreover, the condensed-phase boron oxide (B_2O_3) is in liquid form, introducing a potential risk for slag formation in the throat region.

However, for SiH_4 combustion with CO_2 , the mass fraction of condensed phase products is much lower and the I_{sp} performance falls between Mg and B_2H_6 . The largest contributor to the condensed phase mass loading in this case is silicon dioxide (SiO_2), which forms only inside the combustion chamber.

Therefore, SiH_4 fuel does not suffer nozzle condensation performance penalties and is apparently more attractive than B_2H_6 , despite its lower theoretical I_{sp} . Although SiH_4 is toxic and its ignitability and combustion characteristic with CO_2 are unknown, it is stable and storable and should be retained as a potential fuel candidate pending the acquisition of reliable combustion data.¹⁵

Nevertheless, current knowledge and understanding of performance and combustion characteristics indicate that Mg remains the most promising fuel candidate for use with a CO_2 oxidizer. This general conclusion has led to a few basic laboratory studies on the combustion of Mg pellets and particles in CO_2 gas,^{6,7,9,10,13,16} clearly demonstrating ignitability and vigorous vapor-phase burning. Although these results are not fully informative with respect to the actual burning process in an engine, they do indicate that the Mg fuel would need to be injected as a small-diameter powder (on the order of 100 μm) in order to achieve a practical combustion time scale (≈ 1 ms).

In an attempt to obtain information more relevant to the operating characteristics of a practical rocket engine, Goroshin et al. investigated flame stabilization and measured the laminar burning velocity for Mg particle clouds dispersed in CO_2 .¹⁷ By using mesh graded Mg powder (maximum particle size of 44 μm) suspended in CO_2 at a concentration of 800 g/m^3 , these experiments were able to demonstrate self-sustained Mg/ CO_2 flame propagation through a 1-bar pressure flame tube with an indicated laminar flame speed of ≈ 1 m/s.

Results such as these provide general encouragement for the basic concept, but many questions and issues remain unresolved. Clearly, further progress will require moving beyond simplified laboratory experiments and engaging in the engineering development of a practical rocket configuration for the combustion of powdered Mg with liquid CO_2 .

As a step in this direction, Wickman built and operated a small Mg/ CO_2 rocket engine, but the maximum firing time was limited to less than 5 seconds due to poor injector performance and severe clogging.²⁰ These results serve to highlight the difficult engineering challenges associated with the development of a practical engine configuration for this unusual propellant combination. The authors are not aware of any experimental work in which Mg/ CO_2 mixtures were burned in a sustained combustion process and detailed measurements were acquired.

To begin addressing this shortcoming, we outline a multiphase research and development approach aimed at resolving some of the critical technological issues associated with Mg/ CO_2 rocket engines. The engineering aspects of major concern include transport/injection of compacted powder, ignition, combustion efficiency, combustion stability, dilution effects, lean burn limits, and slag formation issues. It is suggested here that useful initial results could be obtained by adapting an existing dump-configured atmospheric pressure burner, a device previously developed and utilized in the investigation of powdered Al/ H_2O combustion for torpedo propulsion.²¹ This device could be effectively used to develop the critical injector technology and to evaluate basic combustion characteristics. Follow-on phases would involve the development and testing of a pressurized research combustor and technology demonstration tests of a prototypical rocket configuration.

2. MISSION RELEVANCE

It is of interest to consider the mission relevance of Mg/CO₂ propulsion in some detail. The most near term mission that could potentially utilize this particular in situ technology would be a Mars sample return (MSR), a highly ranked science priority with profound implications for planetary geology and biology. To increase the scientific return and enhance the odds for obtaining samples of special significance, it should be noted that NASA's Mars Exploration Program Analysis Group (MEPAG) has recommended that an MSR mission should include modest mobility in order to ensure access to diverse samples and sedimentary deposits at a site known to be scientifically compelling.

It is anticipated that the MSR mission will require several descent-ascent vehicles or a rover that can traverse hundreds of kilometers about the landing site. However, inclusion of this mobility requirement can increase launch payload mass to impractical levels, making the mission nonviable with existing technology. As an alternative, in situ propellant production and utilization should be examined as a possible means of circumventing this limitation.

For instance, Zubrin has recommended a ballistic descent-ascent vehicle/hopper utilizing a nuclear thermal engine that could process CO₂ propellant.²² The basic idea would be to hop from site to site by refilling the propellant tank with liquid CO₂ after every landing. The final fill would then be used to boost a sample return module into Mars rendezvous orbit or directly into a minimal energy orbit to Earth.

Alternatively, Shafirovich et al.^{11,12} recommend consideration of Mg/CO₂ rocket propulsion for the descent-ascent vehicle/hopper, as illustrated in figure 2. In this case, the Mg fuel transported from Earth may have less mass than a nuclear reactor with radiation shielding. These studies estimate that about 2 tons of storable unsymmetrical dimethylhydrazine (UDMH)/nitrogen tetroxide (N₂O₄) propellant would need to be transported from Earth for the surface-to-orbit ascent of a vehicle having a final mass of 1 ton. An Mg/CO₂ engine operating with a CO₂/Mg mass ratio of 4 (I_{sp} = 190 s) would require 5.55 tons of propellant, but only 1.11 tons of this total would need to be transported from Earth. The real gain could be less if the final mass of the vehicle increased.

We briefly examine this issue by considering a two-stage Mars ascent vehicle in which the first boost stage uses either Mg/CO₂ propellant or UDMH/N₂O₄ storable propellant. The second upper stage is always considered to use UDMH/N₂O₄ storable propellants. We assume a surface stay time of 500 days and include the required mass for CO₂ extraction equipment.

Application of the rocket equation, including a structural coefficient (ϵ) to account for the dry weight of the stages, yields the following relation for the upper stage propellant mass per unit mass of ascent payload:

$$\frac{m_{f_2}}{m_p} = \left[\frac{1 - e^{-\Delta V_2/V_{e,2}}}{e^{-\Delta V_2/V_{e,2}} - \epsilon} \right] (1 - \epsilon) . \quad (1)$$

This equation can then be used to obtain a similar expression for the boost stage propellant mass per unit mass of ascent payload:

$$\frac{m_{f_1}}{m_p} = \left\{ \left[\frac{1 - e^{-\Delta V_2/V_{e,2}}}{e^{-\Delta V_2/V_{e,2}} - \varepsilon} \right] + 1 \right\} \left[\frac{1 - e^{-\Delta V_1/V_{e,1}}}{e^{-\Delta V_1/V_{e,1}} - \varepsilon} \right] (1 - \varepsilon) . \quad (2)$$

Here, ΔV_i is the velocity increment for the i th stage and $V_{e,i}$ is the exhaust velocity for i th stage. These results can be used to deduce the dry mass of each stage per unit mass of ascent payload according to the standard definition of the structural coefficient

$$\frac{m_{d_1}}{m_p} = \left(\frac{\varepsilon}{1 - \varepsilon} \right) \frac{m_{f_1}}{m_p} \quad \frac{m_{d_2}}{m_p} = \left(\frac{\varepsilon}{1 - \varepsilon} \right) \frac{m_{f_2}}{m_p} . \quad (3)$$

The relative amounts of Mg and CO₂ in the boost stage depends on the oxidizer-to-fuel mixture ratio (O/F)

$$\frac{m_{Mg}}{m_p} = \left(\frac{1}{1 - O/F} \right) \frac{m_{f_1}}{m_p} \quad \frac{m_{CO_2}}{m_p} = \left(\frac{O/F}{1 - O/F} \right) \frac{m_{f_1}}{m_p} . \quad (4)$$

To estimate the mass of the CO₂ extraction equipment, we consider a solar powered adsorption compressor.²³ Following a similar approach by Shafirovich, we assume that this type of acquisition unit can produce 0.1 kg of CO₂ per unit mass of the acquisition unit per day.²⁴ Because the specific power of this CO₂ extraction unit is roughly equal to that of a martian solar power plant (≈ 15 W/kg), we deduce that the mass of the complete CO₂ extraction system per unit mass of collected CO₂ is

$$\frac{m_{ex}}{m_{CO_2}} = \frac{1}{0.05t} , \quad (5)$$

where t is the collection time in days. It follows that the mass of the CO₂ extraction equipment per unit mass of ascent payload is given by

$$\frac{m_{ex}}{m_p} = \left(\frac{1}{0.05t} \right) \left(\frac{O/F}{1 - O/F} \right) \frac{m_{f_1}}{m_p} . \quad (6)$$

A good parameter of merit for this mission is the trans-Mars injection mass (m_{tmi}) per unit mass of ascent payload, which may be expressed in the form:

$$\frac{m_{tmi}}{m_p} = \frac{m_{ex}}{m_p} + \frac{m_{Mg}}{m_p} + \frac{m_{d_1}}{m_p} + \frac{m_{f_2}}{m_p} + \frac{m_{d_2}}{m_p} + 1 . \quad (7)$$

By specifying the ϵ , the total maneuver velocity increment, and the exhaust velocity for each stage, this parameter can now be evaluated as a function of the velocity increment split between stages. Here, we examine both Mars orbit rendezvous ($\Delta V = 4,300$ m/s) and direct minimum energy return to Earth ($\Delta V = 7,000$ m/s) assuming CO_2/Mg mass ratios of 2 ($I_{sp} = 220$ s), 4 ($I_{sp} = 190$ s), and 6 ($I_{sp} = 170$ s) and structural coefficients of 0.10, 0.12, and 0.14. Representative results are shown in figure 3 (Mars orbit rendezvous) and figure 4 (direct Mars return) for a CO_2/Mg mass ratio of 4, with $\epsilon = 0.1$ for all stages. Note the occurrence of an optimum ΔV split between stages to obtain a minimum value for the m_{tmi} . These results demonstrate that the in situ boost stage provides a slight advantage for ascent to Mars orbit but leads to an inferior system for direct return to Earth. In the latter case, the poor I_{sp} of the in situ propellant translates into excessive Mg and structural mass for the boost stage. Note that the optimum ΔV split occurs when about 20 percent of the total velocity increment is provided by the Mg/ CO_2 boost stage.

The baseline performance characteristics for two-stage systems with all storable propellant are listed in table 1. The optimal performance results for two-stage systems using Mg/ CO_2 for boost-stage propulsion are summarized in table 2. From this data, we observe that the optimal CO_2/Mg mass ratio is about 4 and that the maximum m_{tmi} savings is somewhat less than 12 percent when ascending to a Mars orbit rendezvous, assuming equal structural coefficients for all stages. When ascending directly into an Earth return orbit, the in situ system is unable to provide any mass savings under any operating conditions.

The preceding analysis assumes a simple grab-and-go MSR mission that requires no mobility for sample procurement. Thus, the vehicle is only required to land and ascend once, and in situ propellant utilization yields no significant mass savings over conventional storable propellants transported entirely from Earth. In this case, there is little incentive to take on the additional risks associated with new vehicle architecture. However, it has been argued that an ISRU architecture should remain in consideration for the purpose of demonstrating the technology and preparing infrastructure for long-term sustained exploration of Mars.¹⁹

As the need for mobility increases, the mass savings of the in situ architecture becomes more substantial and beneficial. Detailed analyses by Shafirovich et al. have shown that a Mars descent-ascent vehicle that undertakes two to three ballistic hops with interhop CO_2 refills before the final ascent to orbit has a clear mass savings over conventional storable propellants.^{11,12,24}

In situ propellant and utilization technologies will be even more essential from a longer term exploration perspective. Light metals and elements are generally abundant in the crust of terrestrial planets, and methods for processing the planetary regolith and extracting these elements appear feasible. Therefore, burning of metals with indigenous oxidizers may have broad applicability across the solar system and warrants further technological development.

3. ROCKET ENGINE CONFIGURATION ISSUES

The practical question of how to configure and design an Mg/CO₂ rocket remains an open issue. Several alternative engine design concepts have been suggested, but rudimentary evaluation and engineering judgment leans heavily toward direct injection of powdered Mg and CO₂ in a bipropellant engine.^{11,17} This conclusion is largely motivated by the favorable combustion characteristics of Mg particles in CO₂, as observed and confirmed in previous laboratory experiments.

For instance, data from these experiments indicate that the minimum ambient temperature for particle ignition is very close to the melting point of Mg ($\approx 1,000$ K), implying easy engine ignitability. Furthermore, these experiments confirm the existence of two spatially separated chemical reaction zones in which $\text{Mg} + \text{CO}_2 \rightarrow \text{MgO} + \text{CO}$ within a gas phase region displaced from the particle surface, while $\text{Mg} + \text{CO} \rightarrow \text{MgO} + \text{C}$ occurs, essentially heterogeneously, near the particle surface. As a result, Mg particles undergo vigorous vapor-phase burning in CO₂, which leads to an expectation of high engine combustion efficiency.

The primary engineering challenge of this type of engine appears to be the development of a reliable and effective transport/injection system for metal powder. Such a system must be capable of pumping, injecting, and dispersing Mg powder in a fluid-like manner without jamming or clogging. The consensus view on this matter is that a pneumatic feed system based on fluidization techniques represents the most promising technological option for this type of engine, an opinion supported by the authors' own informed experience with similar devices. An additional simplification resulting from this approach is that the Mg powder may be utilized in the same state in which it is stored and transported to Mars as a dry compacted powder.

4. RESEARCH AND DEVELOPMENT APPROACH

Since the ultimate objective is to develop a practical Mg/CO₂ rocket engine and since basic feasibility issues have been more or less resolved through simplified laboratory experiments, we suggest that the most appropriate strategy is to proceed with the development of a practical engine configuration. In this way, it will be possible to begin addressing the important engineering aspects of a powdered metal combustion device such as transport/injection of compacted powder, ignition, combustion efficiency, combustion stability, dilution effects, lean burn limits, and slag formation issues.

Along these lines, we recommend a multiphase research and development approach aimed at resolving critical technical issues in a cost effective manner. For example, useful initial results could be obtained by adapting an existing dump-configured atmospheric pressure burner, a device previously developed and utilized in the investigation of powdered Al/H₂O combustion for torpedo propulsion.²¹ With some simple low-cost modifications, this device could be readily modified to serve as a test bed for the development of a fluidized-bed powdered metal transport system and for evaluation of basic combustion characteristics. Follow-on phases would involve the development and testing of a pressurized research combustor and technology demonstration tests of a prototypical rocket configuration.

4.1 Phase I Atmospheric Pressure Burner Configuration

The basis for phase I development activities is an existing refractory-lined, atmospheric-pressure dump combustor designed for investigation of powdered Al combustion with steam. This device, shown schematically in figure 5, is currently being installed in the NASA Marshall Space Flight Center (MSFC) Propulsion Research Laboratory, and modifications are being implemented for conversion to Mg/CO₂ operation.

The combustion chamber consists of a 4-in diameter circular passage with a 3-in thick high-alumina, heavy-castable, refractory lining inside of two 24-in sections of 10-in diameter carbon steel pipe. Eight sets of diametrically opposed ports are distributed at 6-in intervals along the combustor axis for optical access and extraction of combustion product samples. The combustor is cooled by running water over an external wick and has a removable head section to simplify cleanout of deposits between test runs.

Powdered metal and gaseous oxidizer are injected through a central annular injector. The powder is supplied to the injector as a fluidized stream using a small quantity of inert carrier gas and enters the combustion chamber with the gaseous oxidizer through an annular injector. A portion of the oxidizer may be injected through an annular shroud surrounding the injector tube to reduce premature melting of the incoming fuel particles and buildup of recirculated condensed phase combustion products at the injector exit.

For illustrative purposes, figure 5 also includes a photograph of the device in operation with Al/H₂O propellants. This particular propellant combination required significant combustor preheating to promote ignition; but, once started, stable and efficient combustion could be maintained for an extended

duration (several minutes) without difficulty and without injector clogging. Because the ignition temperature is lower for Mg particles in CO₂ than for Al particles in H₂O and because burning rates of Mg particles are faster in CO₂ than for similar sized Al particles in H₂O, it is highly probably that a practical Mg/CO₂ combustor can eventually be developed.

A metal powder feed system that can supply a steady, repeatable flow of fuel to the combustor is a critical component for any metal powder rocket engine. As part of the previous Al/H₂O combustor development activities, a Positive Displacement Fluidized Bed (PDFB) feeder was designed, developed, and successfully demonstrated. The PDFB concept was originated at Bell Aerospace during the late 1960s in an attempt to supply powdered metal fuels to rocket and missile combustors.^{25,26}

In the PDFB system, metal powder is contained in a pneumatic cylinder with porous metal plates on the head and on the piston face. Fluidizing gas is supplied through the porous plates while the piston is used to maintain uniform powder density at the feed line inlet as powder is being removed. Previous experience indicates that this type of system can supply a steady and stable stream of powdered metal to a combustion chamber with a high degree of reliability.

Our PDFB system was constructed by modifying a standard 3-in diameter × 32-in stroke stainless steel pneumatic cylinder. Porous metal plate distributors for fluidizing gas were added in the piston face and at the discharge, and the piston seal was modified to keep it from binding on the Al particles. A separate 3-in diameter pneumatic cylinder supplies the force needed to maintain uniform density in the powder. A schematic of this PDFB system is provided in figure 6 along with a photograph of the assembled unit.

During testing of the feeder system with 17-μm mean diameter spherical Al particles, it was found that smooth and repeatable flow rates in the range of 5–10 g/s could be supplied through a 1/16-in inside diameter × 20-ft long dense phase-feed line with feed line pressure drops ranging from 40–100 psi. Reliable flow was achieved flowing into both atmospheric and pressurized receivers.

Representative calibration results for the PDFB feeder are summarized in figures 7 and 8. Figure 7 shows a typical result for PDFB piston displacement as a function of time, which clearly demonstrates the achievement of steady and stable powder flow. Figure 8 summarizes the correlation with feed line pressure drop for a wide range of powder flow rates.

As part of the combustor modifications for Mg/CO₂ operation, a new powdered metal injector is being sized, designed, and fabricated. The fundamental design configuration for the Mg/CO₂ injection system follows a conceptual approach previously established during burner operations with Al and steam. The basic conceptual design is illustrated in figure 9 where powdered metal is introduced to a mixing chamber through a central tube surrounded by two annular injectors for oxidizer and propane (C₃H₈), which serves as an ignition and flame stabilization source. The mixing tube feeds into the main dump combustor and is shrouded by an annular bypass injector for the oxidizer. During previous burner operations with Al and steam, the annular shroud was found to be an effective means of preventing slagging and clogging in the main injector. This capability is illustrated in figure 10, which shows a photograph of the main injector through the burner bore following extended operation with Al and steam. Note that the injector ports are clear and free of any residue.

Following modifications of this apparatus for Mg/CO₂ operation, experiments are planned to determine how variations in particle size, fuel flow rate, and O/F impact combustion efficiency and stability. A process and instrumentation diagram for these experiments is shown in figure 11. The start-up procedure is to first introduce a C₃H₈/O₂ pilot flame into the combustor. Then, Mg powder flow is started and C₃H₈ flow is stopped. At this point, the O₂ flow may be slowly reduced until the combustor is sufficiently warmed to support unaided Mg/CO₂ combustion. Detailed combustion performance measurements could then be implemented.

Subsequent experiments would determine the range of stable combustion and required residence times for Mg/CO₂ mixtures for different particle sizes and O/Fs. Flame radiation characteristics, including temperature and spectral intensity, would be measured and evaluated, and combustion product samples would be extracted and analyzed to acquire information on combustion efficiency. As part of this experimental testing, it is also highly recommended that a parallel computational analysis capability be developed and validated. This activity would be essential for interpreting the experimental data, understanding device operation, and for perfecting engine design to obtain optimal performance attributes.

At the conclusion of the phase I effort, hardware required for practical rocket engine operation would be developed and proven, particularly the metal powder feed system and injector. Moreover, the technical feasibility of the engine concept would be convincingly demonstrated, and sufficient engineering data and adequate analysis tools would exist to support the design and development of a pressurized combustor configuration in a follow-on phase II effort.

4.2 Phase II Pressurized Combustor Configuration

In the follow-on phase II effort, data from the phase I atmospheric pressure tests would be used to design a pressurized combustor that could operate in the pressure range of interest for practical rocket applications. It has been suggested that a Mars ascent rocket should operate at a chamber pressure in the range of 10 atm, since this would allow an expansion ratio $P_C/P_E = 1,000$ at Mars surface pressure (about 10 mbar) and would also be low enough to allow the CO₂ oxidizer to be pressure-fed.

The preliminary concept for the pressurized combustor would employ a fuel-rich preburner section followed by a dilution section, where additional CO₂ would be added to raise the overall O/F to the range of 4 to 6. The combustion chamber would have a linear configuration with a water-cooled metal jacket and graphite liner, as illustrated by the conceptual sketch in figure 12. Optical ports would be installed along the combustor axis, and experiments similar to those planned for the atmospheric combustor would be conducted. Computational analysis tools would also be refined and validated as part of this broad-based technology development effort.

4.3 Phase III Prototypical Rocket Configuration

The results of the phase II pressurized combustor experiments would provide the engineering data required for the design of a phase III prototype rocket engine. The purpose of this effort would be to demonstrate a flight-like engine configuration including all major components and subsystems. Tests with this engine would fully demonstrate the technology in a relevant fashion as a risk reduction activity leading to a mission application.

5. CONCLUSIONS

Powdered Mg combustion with CO₂ has been examined as a potential in situ propellant combination for Mars propulsion and power. Mission analyses indicate that this combination can have merit over storable bipropellants transported from Earth despite its relatively poor propulsion performance. This advantage becomes more pronounced as surface mobility requirements increase.

Basic laboratory studies have established proof of principle, and it is recommended that further progress will require the development of practical combustor configurations. A multiphase research and development strategy has been outlined as a cost-effective approach to technology maturation. It is argued that this strategy could be initially implemented by adapting an existing atmospheric pressure-dump combustor, previously developed for investigation of powdered Al combustion with steam. This would permit the establishment of basic engineering feasibility while addressing fundamental engineering aspects of a practical engine design. Follow-on phases would involve the development and testing of a pressurized research combustor and technology demonstration of a prototype rocket engine.

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Table 1. Baseline two-stage ascent vehicle performance.*

Mars Orbit Rendezvous ($\Delta V = 4,300$ m/s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	2,150	2,150	2,150
ΔV_2 (m/s)	2,150	2,150	2,150
m_{mi}/m_p	4.71	4.97	5.27
Direct Return to Earth ($\Delta V = 7,000$ m/s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	3,500	3,500	3,500
ΔV_2 (m/s)	3,500	3,500	3,500
m_{mi}/m_p	14.16	16.12	18.64

*UDMH/N₂O₄ propellants for all stages ($I_{sp} = 330$ s).

Table 2: In situ two-stage ascent vehicle performance.*

Mars Orbit Rendezvous ($\Delta V=4,300$ m/s)			
1st-Stage CO ₂ /Mg Mass Ratio = 2 ($I_{sp}=220$ s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	1,760	1,700	1,650
ΔV_2 (m/s)	2,540	2,600	2,650
m_{imi}/m_p	4.27	4.59	4.97
Mars Orbit Rendezvous ($\Delta V=4,300$ m/s)			
1st-Stage CO ₂ /Mg Mass Ratio = 4 ($I_{sp}=190$ s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	1,660	1,570	1,500
ΔV_2 (m/s)	2,640	2,730	2,800
m_{imi}/m_p	4.17	4.52	4.93
Mars Orbit Rendezvous ($\Delta V=4,300$ m/s)			
1st-Stage CO ₂ /Mg Mass Ratio = 6 ($I_{sp}=170$ s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	1,500	1,420	1,350
ΔV_2 (m/s)	2,800	2,880	2,950
m_{imi}/m_p	4.21	4.59	5.03
Direct Return to Earth ($\Delta V=7,000$ m/s)			
1st-Stage CO ₂ /Mg Mass Ratio = 2 ($I_{sp}=220$ s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	2,510	2,520	2,550
ΔV_2 (m/s)	4,490	4,480	4,450
m_{imi}/m_p	14.24	17.35	21.84
Direct Return to Earth ($\Delta V=7,000$ m/s)			
1st-Stage CO ₂ /Mg Mass Ratio = 4 ($I_{sp}=190$ s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	2,260	2,270	2,300
ΔV_2 (m/s)	4,740	4,730	4,700
m_{imi}/m_p	14.39	18.06	23.68
Direct Return to Earth ($\Delta V=7,000$ m/s)			
1st-Stage CO ₂ /Mg Mass Ratio = 6 ($I_{sp}=170$ s)			
ϵ	0.10	0.12	0.14
ΔV_1 (m/s)	2,060	2,070	2,100
ΔV_2 (m/s)	4,940	4,930	4,900
m_{imi}/m_p	15.06	19.38	26.36

*Mg/CO₂ ascent stage & UDMH/N₂O₄ upper stage.

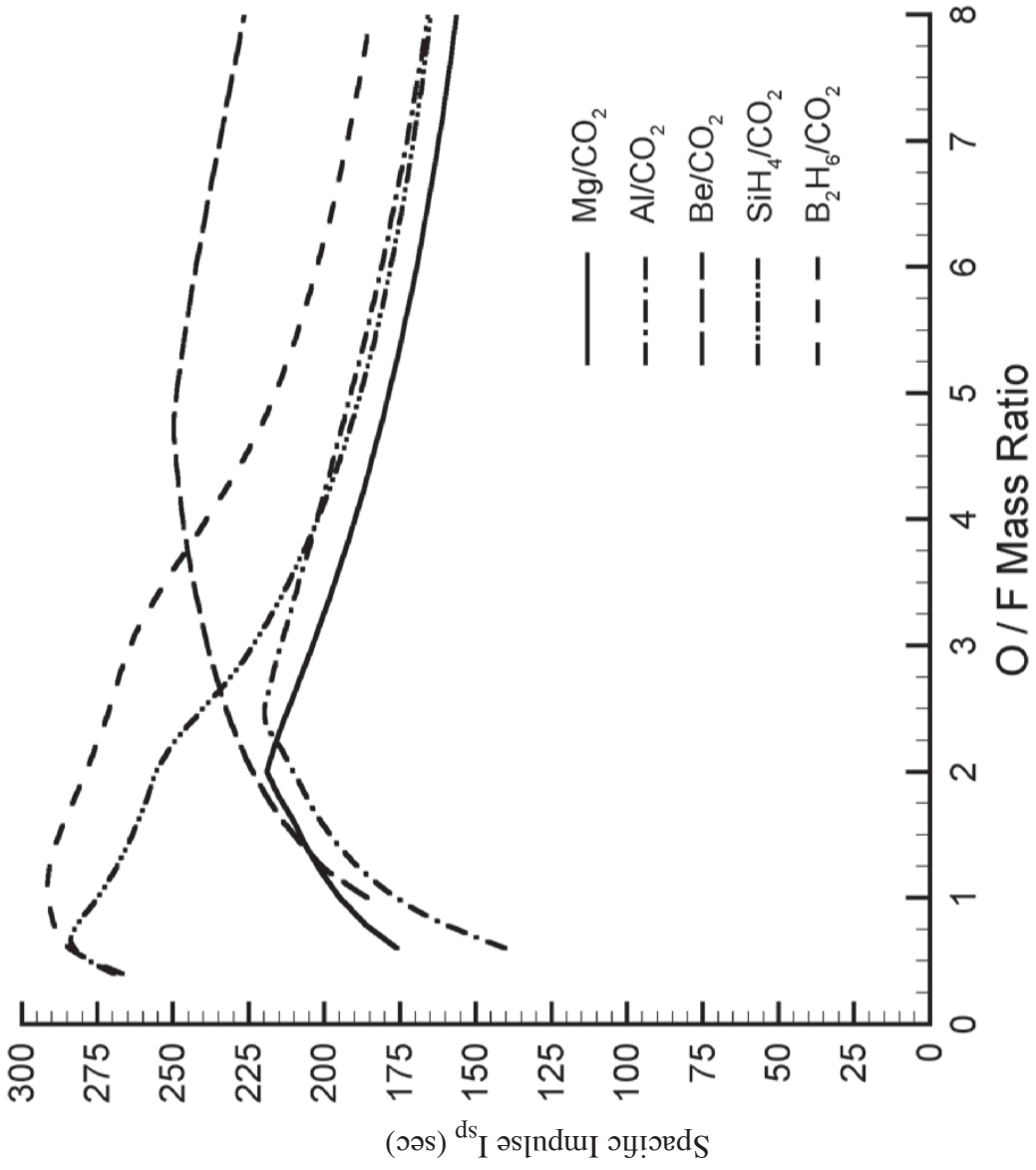


Figure 1. Ideal rocket I_{sp} for the combustion of representative fuels in CO_2 . Results are from thermodynamic equilibrium calculations assuming a chamber pressure of 10 atm and a nozzle exit pressure of 0.01 atm.

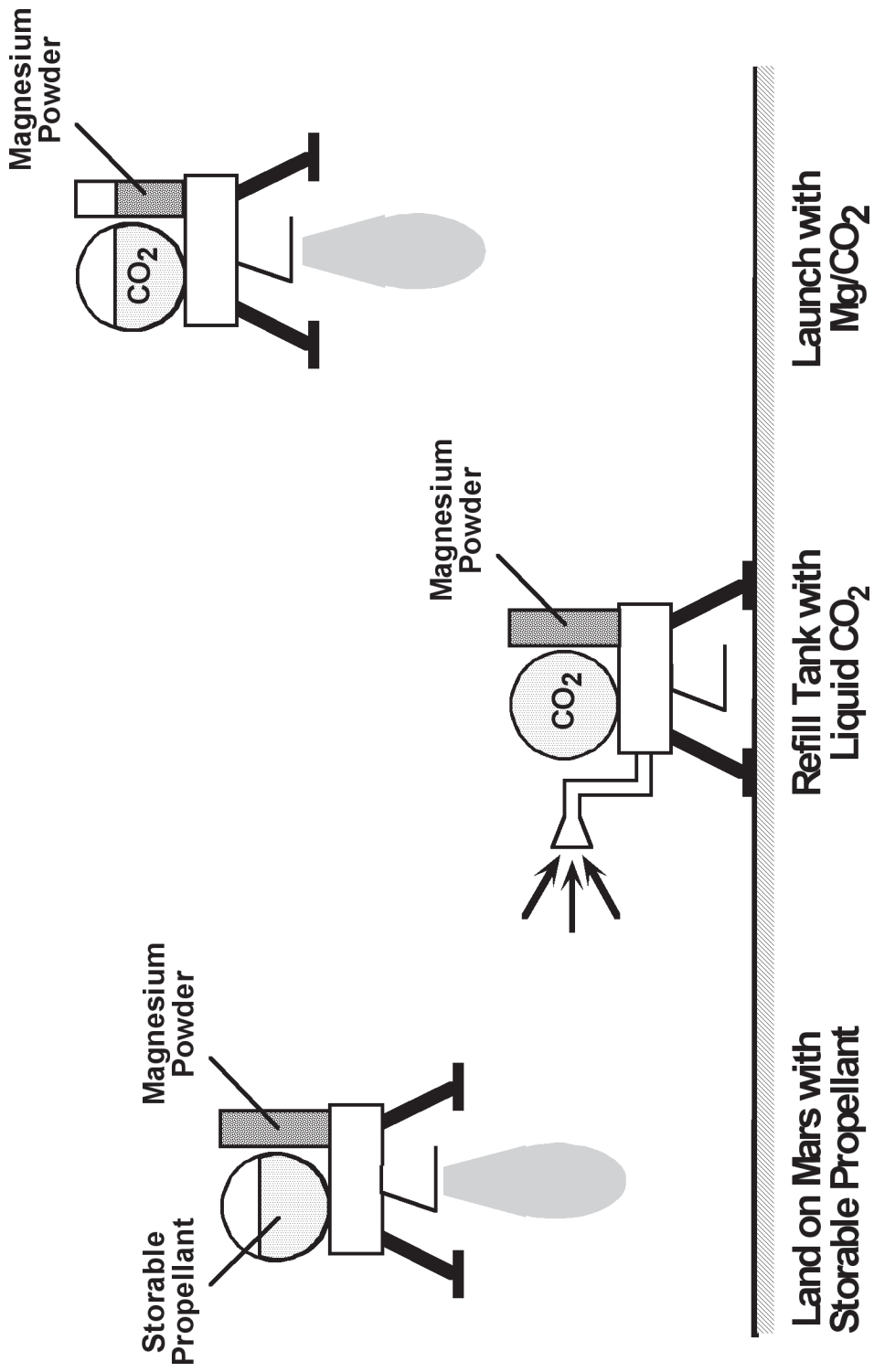


Figure 2. Illustration of a Mars descent-ascent vehicle/hopper using Mg/CO₂ rocket propulsion. CO₂ is collected and liquefied between each landing and launch event.

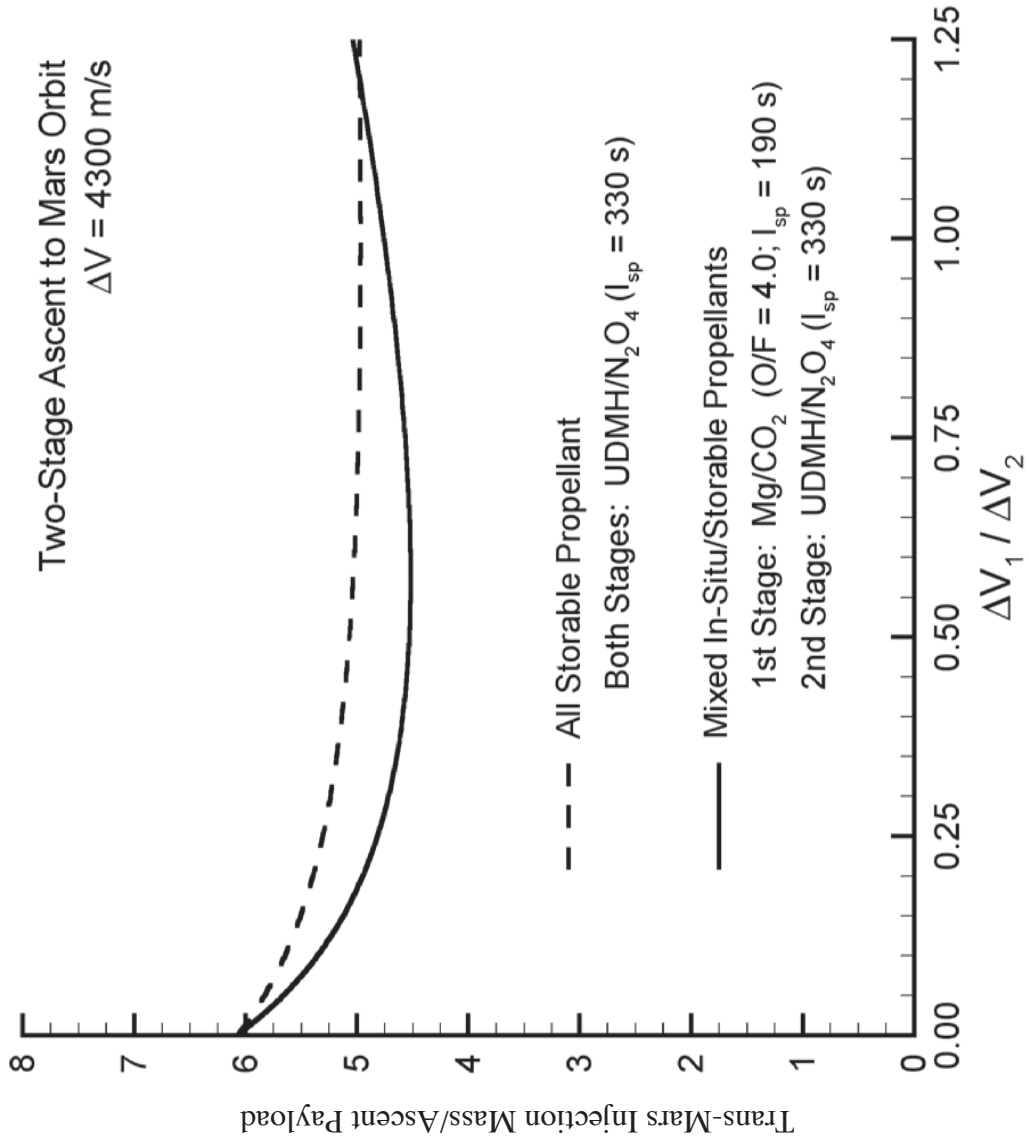


Figure 3. Trans-Mars injection mass per unit mass of ascent payload as a function of the velocity increment split for a two-stage ascent to Mars orbit.

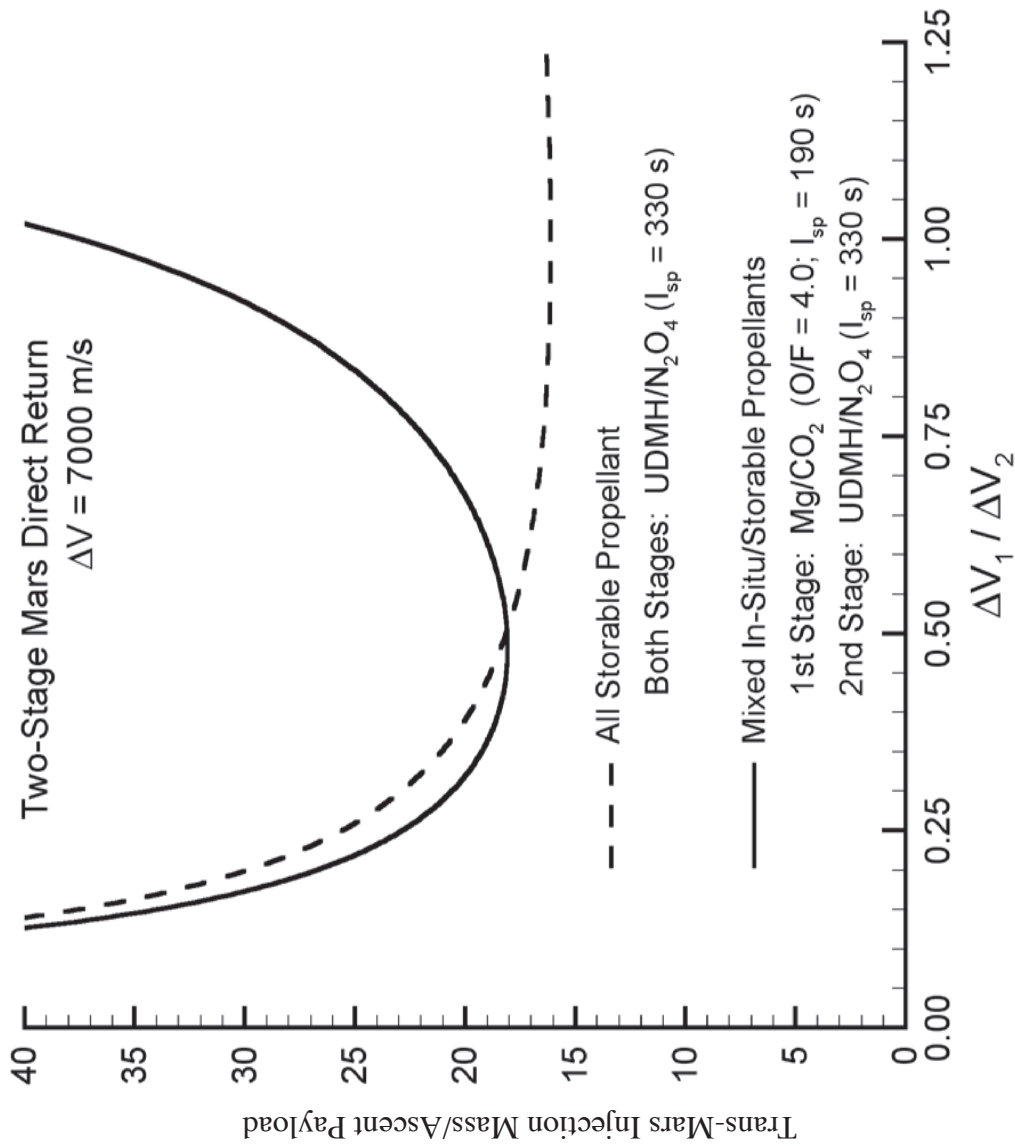


Figure 4. Trans-Mars injection mass per unit mass of ascent payload as a function of the velocity increment split for a two-tage direct return to Earth.

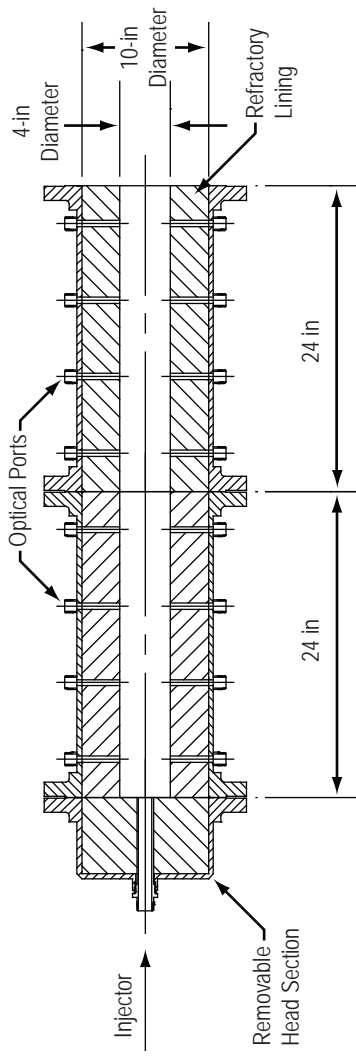


Figure 5. Schematic of refractory-lined atmospheric dump combustor designed for investigation of powdered metals combustion. Lower photograph illustrates previous combustor operation with Al/H₂O propellants.

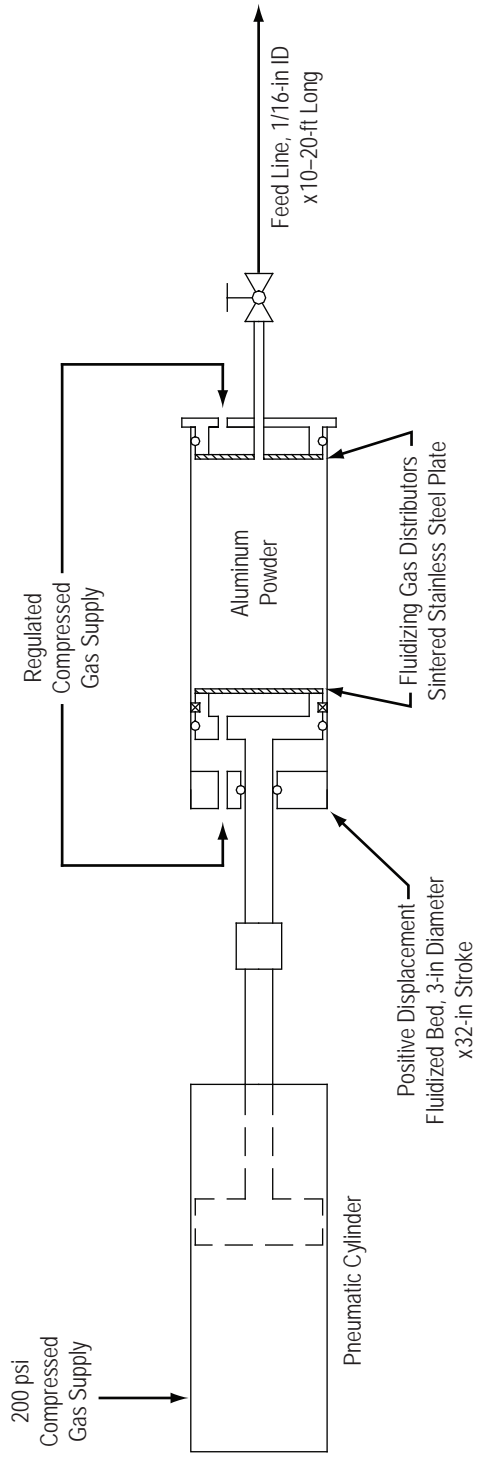


Figure 6. Schematic of PDFB powdered metal feed system and photograph of assembled unit.

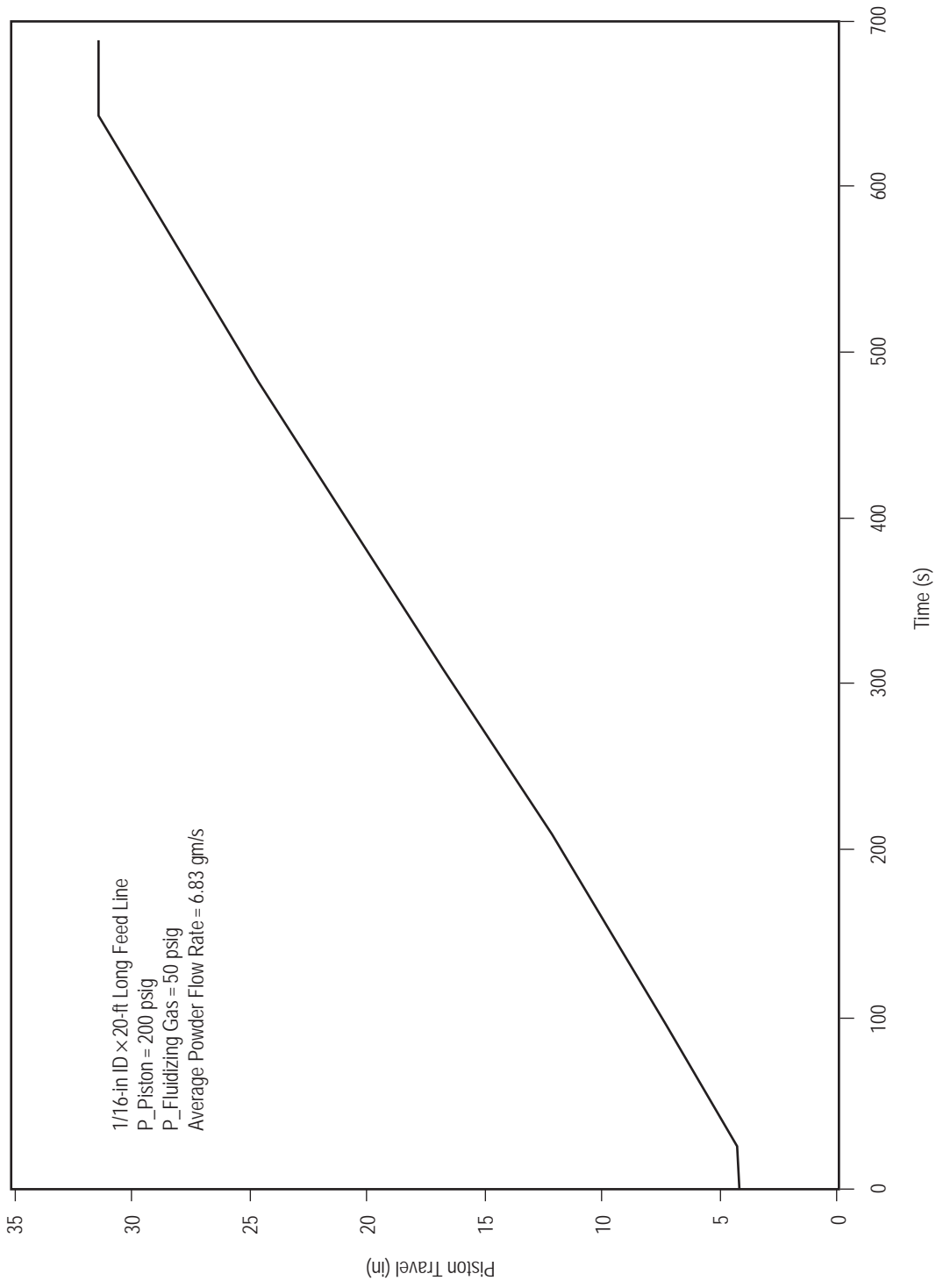


Figure 7. Representative calibration results for the PDFB feeder showing piston travel as a function of time.

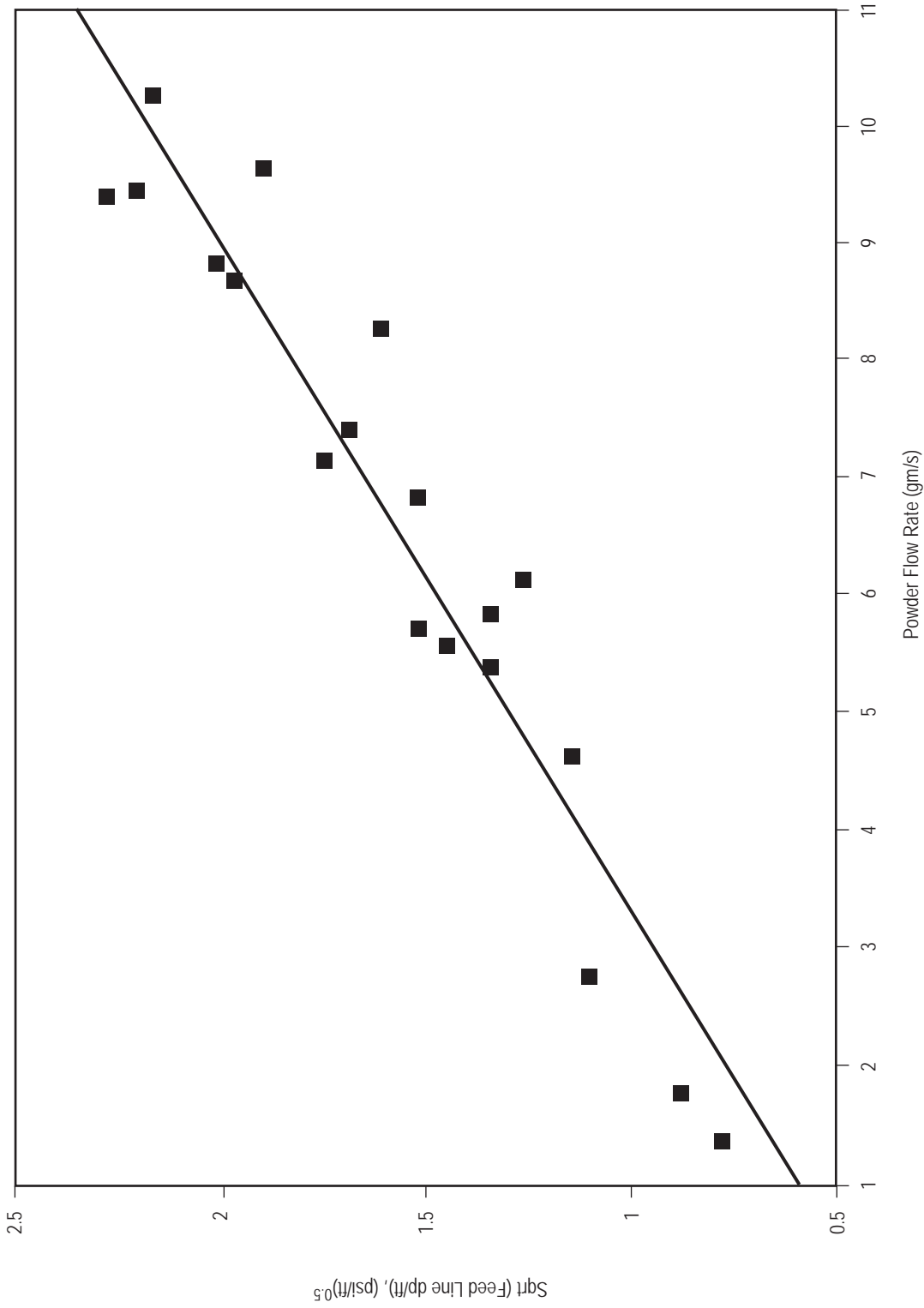


Figure 8, Representative calibration results for the PDFB feeder showing feed line pressure drop as a function of flow rate.

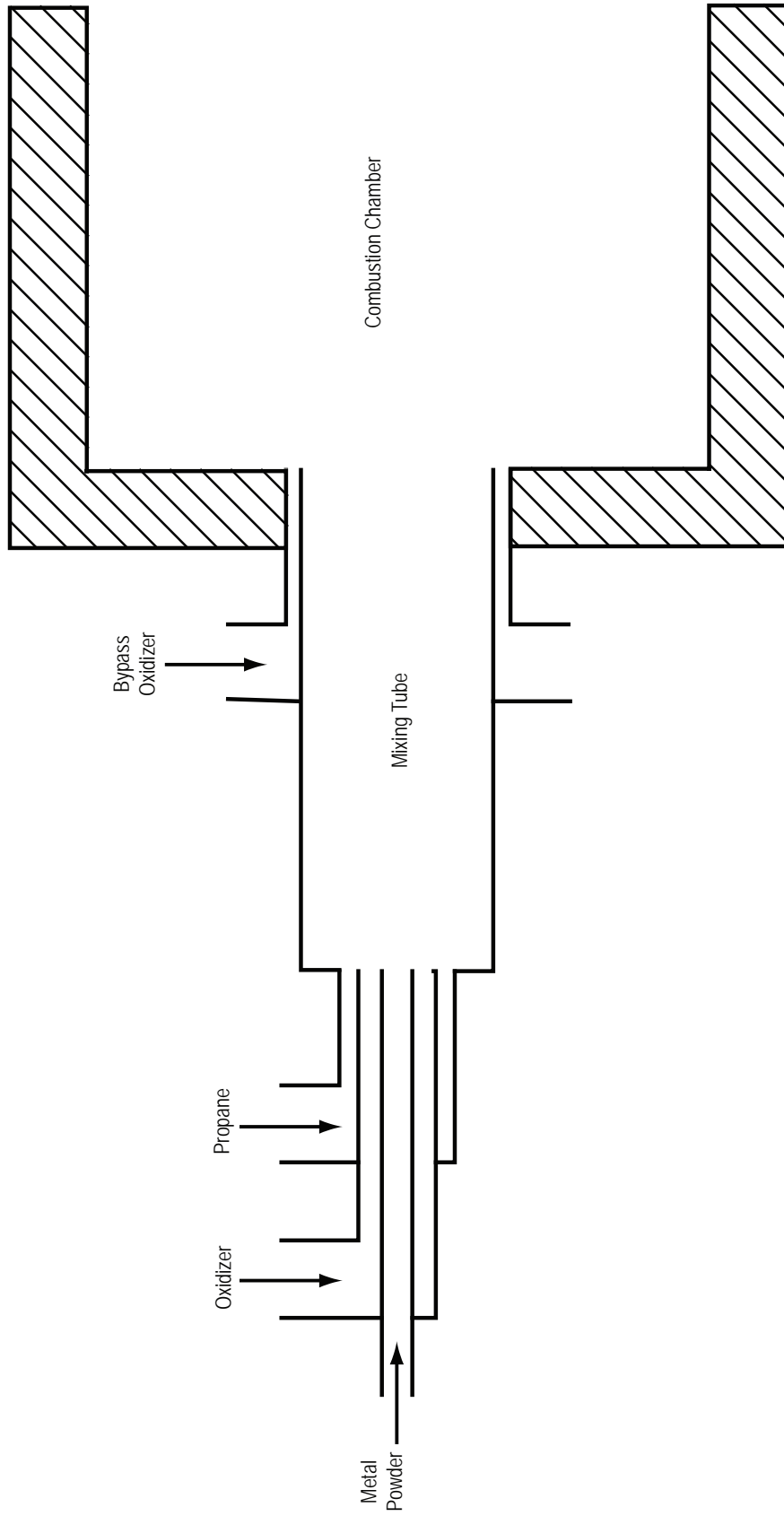


Figure 9. Conceptual design of powdered metal injector for Mg/CO₂ research combustor.

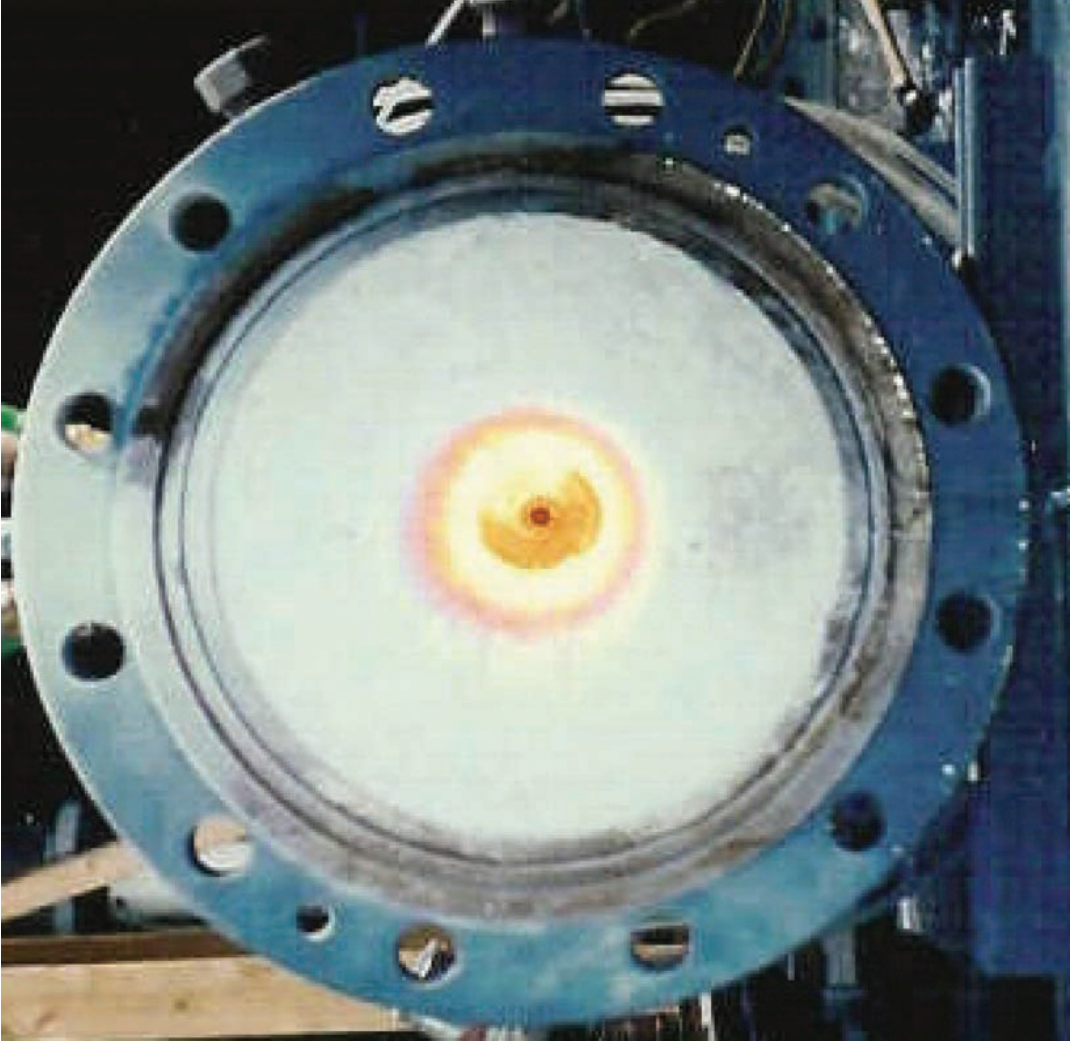


Figure 10. Photograph of main injector through burner bore following extended operation with AI and steam.

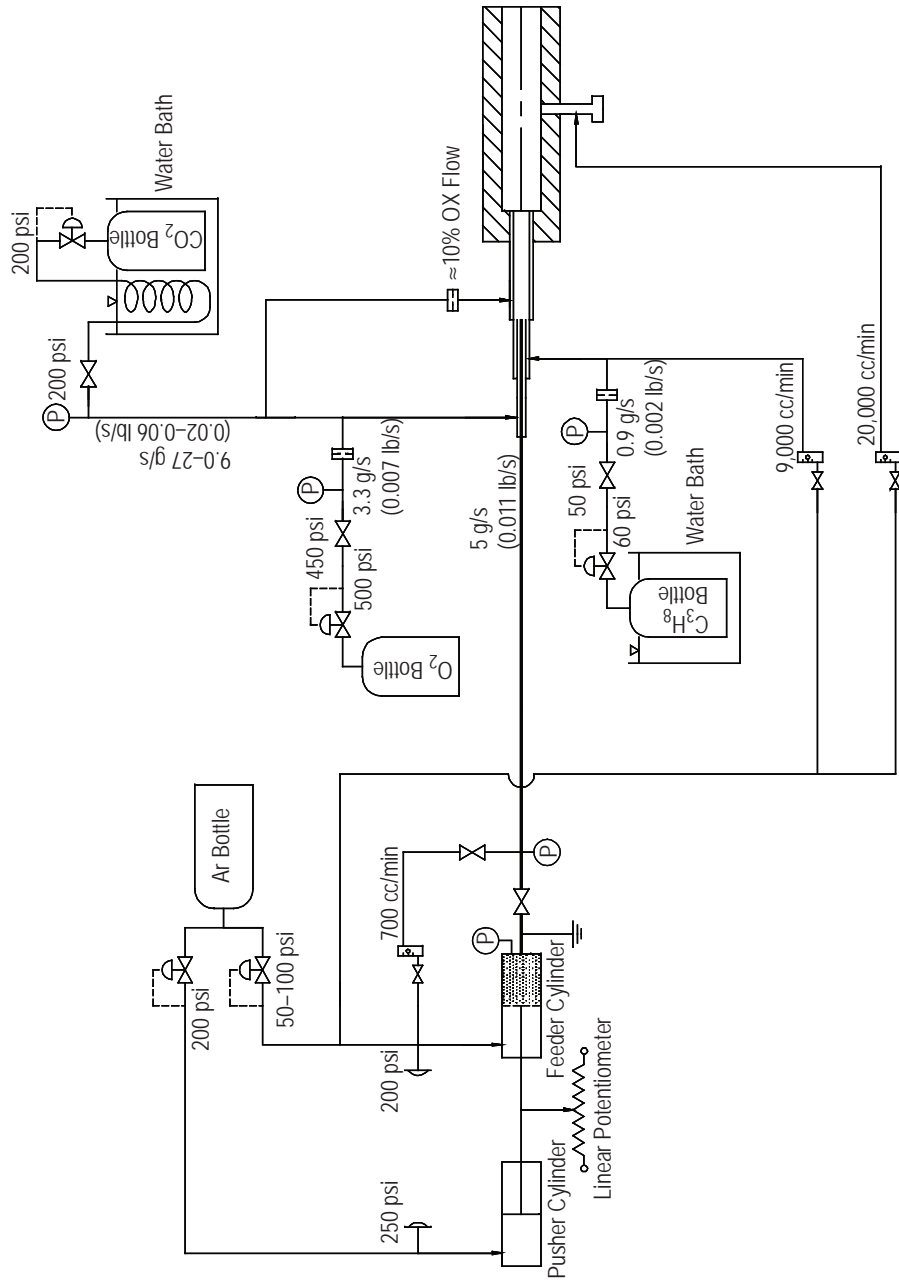


Figure 11. Process and instrumentation diagram for Mg/CO₂ research combustor operation.

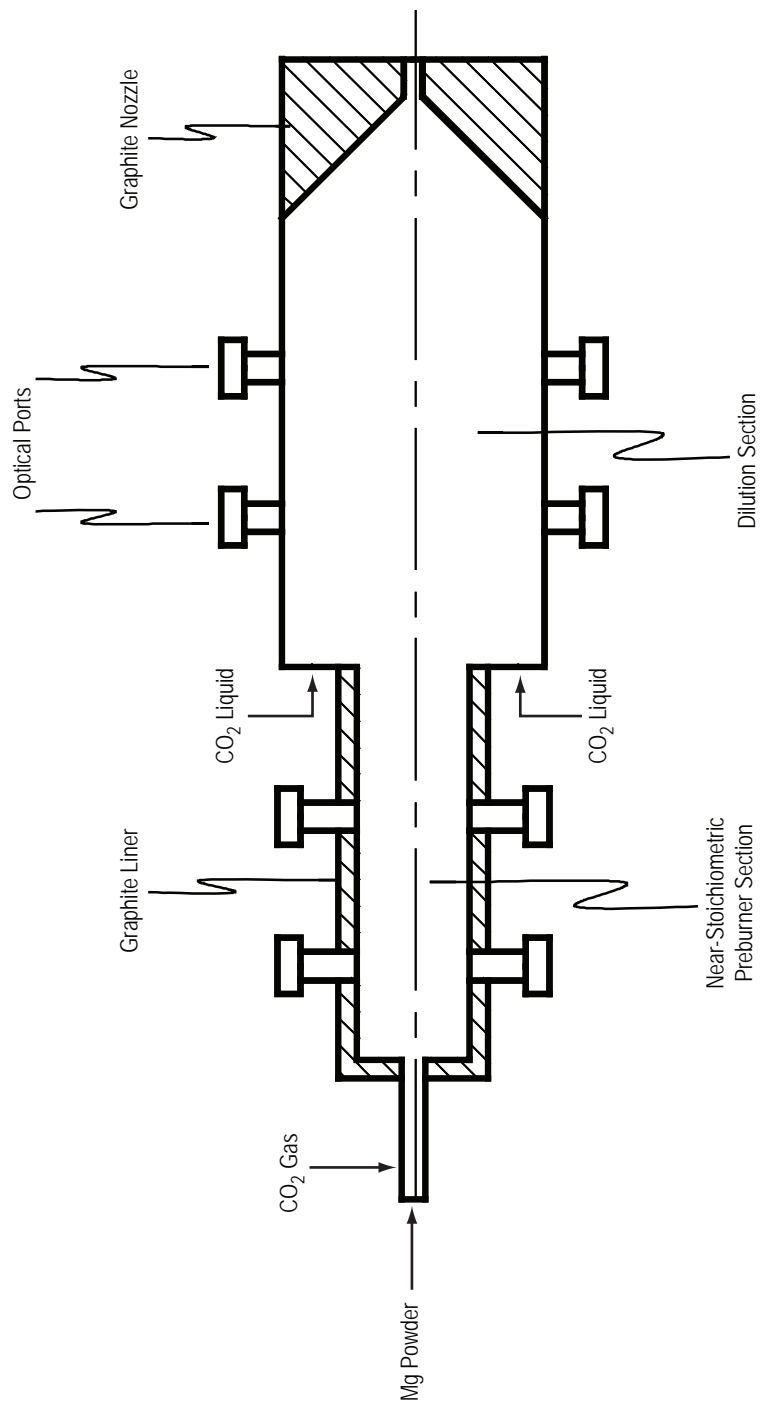


Figure 12. Conceptual sketch of a pressurized configuration for investigation of Mg/CO₂ rocket combustion.

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13. ABSTRACT <i>(Maximum 200 words)</i> Powdered magnesium (Mg)—carbon dioxide (CO ₂) combustion is examined as a potential in situ propellant combination for Mars propulsion. Although this particular combination has relatively low performance in comparison to traditional bipropellants, it remains attractive as a potential basis for future martian mobility systems, since it could be partially or wholly manufactured from indigenous planetary resources. As a means of achieving high mobility during long-duration Mars exploration missions, the poorer performing in situ combination can, in fact, become a superior alternative to conventional storable propellants, which would need to be entirely transported from Earth. Thus, the engineering aspects of powdered metal combustion devices are discussed including transport/injection of compacted powder, ignition, combustion efficiency, combustion stability, dilution effects, lean burn limits, and slag formation issues. It is suggested that these technological issues could be effectively addressed through a multiphase research and development effort beginning with basic feasibility tests using an existing dump configured atmospheric pressure burner. Follow-on phases would involve the development and testing of a pressurized research combustor and technology demonstration tests of a prototypical rocket configuration.			
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