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# Technical Prospects for Utilizing Extraterrestrial Propellants for Space Exploration

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**NASA**

# TECHNICAL PROSPECTS FOR UTILIZING EXTRATERRESTRIAL PROPELLANTS FOR SPACE EXPLORATION

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## ABSTRACT

NASA's Lewis Research Center has supported several efforts to understand how lunar and martian produced propellants can be used to their best advantage for space exploration propulsion. A discussion of these efforts and their results is presented. A manned Mars mission analysis study identified that a more thorough technology base for propellant production is required before the net economic benefits of in situ propellants can be determined. Evaluation of the materials available on the moon indicated metal/oxygen combinations are the most promising lunar propellants. A hazards analysis determined that several lunar metal/LOX monopropellants could be safely worked with in small quantities, and a characterization study was initiated to determine the physical and chemical properties of potential lunar monopropellant formulations. A bipropellant metal/oxygen subscale test engine which utilizes pneumatic injection of powdered metal is being pursued as an alternative to the monopropellant systems. The technology for utilizing carbon monoxide/oxygen, a potential martian propellant, has been studied in subscale ignition and rocket performance experiments.

## INTRODUCTION

The Space Exploration Initiative, as proposed by U.S. President George Bush and developed in studies for NASA and the National Space Council (refs. 1-3), outlines an ambitious plan for establishment of a permanent lunar base and for manned exploration of Mars. Other countries are also planning ambitious space exploration missions (ref. 4,5). The leading restraint on exploration possibilities is the cost of launching large masses into orbit. Space transportation studies have shown that significant reductions in launch mass may be realized if propellants produced from indigenous space materials are used (ref. 6). Such a reduction in launch mass would alleviate the economic burden of space exploration.

Unfortunately, the exploration mission transportation studies are limited by incomplete information for technologies proposed for the future missions (ref. 7). Many of the proposed production processes for extracting useful elements from the lunar regolith have not been developed beyond the concept level. Mars' atmospheric processing concepts have been demonstrated only at a laboratory level. Similarly, the operation of rocket engines using the sometimes unconventional propellants has not been demonstrated. Because of the scarce base for these proposed technologies, the accuracy and confidence level for the space transportation study results is uncertain.

To eliminate some of the uncertainties in the propulsion technology, efforts at the NASA Lewis Research Center continue to evaluate the benefits and technical prospects of utilizing in situ propellants. The work has focussed on two areas: expanding and updating the space transportation studies with the most recent data for technology assumptions, and building a data base for the unconventional propellants which may be produced from indigenous lunar and martian resources. This report discusses the results and status of these efforts.

## BACKGROUND

The makeup of the moon and Mars are known to a different extent. The US Apollo and USSR Luna missions returned lunar samples to Earth that allowed for a detailed analysis of the lunar regolith (ref. 8). The US Viking landers to Mars provided an analysis of the martian atmosphere and, to a lesser extent, the composition of the martian soil

(ref. 9). It is from these banks of data that one first looks to determine what is available at the moon and Mars that has potential to be used as a rocket propellant. Once the potential propellant combinations have been identified, mission analyses can be performed to quantify the expected benefits.

Available Resources

The lunar regolith is comprised of minerals, the most common of which are olivine ($(Mg,Fe)_2SiO_4$), pyroxene ($(Ca,Fe,Mg)_2Si_2O_6$), ilmenite ($(Fe,Mg)TiO_3$), and anorthite ($CaAl_2Si_2O_8$). These minerals can be found in varying concentrations in the highlands and the maria. Table I shows the amounts of some elements on the moon based on analysis of the mineral samples returned to Earth. From the table, it is apparent that oxygen is plentiful, and can be used as the oxidizer. Hydrogen and carbon, the two elements that are most commonly used in rocket fuels, are not present on the moon to any appreciable extent. Some of the metals that are present on the moon, however, have been used in solid propellant fuels and therefore have potential as lunar in situ fuels.

Table I. - Elemental Breakdown of Lunar Regolith

| Element | Weight Percent |
|---------|----------------|
| Oxygen, $O_2$ | 42 |
| Silicon, Si | 20 |
| Aluminum, Al | 9 |
| Iron, Fe | 9 |
| Calcium, Ca | 8 |
| Magnesium, Mg | 4 |
| Titanium, Ti | 3 |
| Phosphorous, P, Sulfur, S, Sodium, Na, Potassium, K, Chromium, Cr | ~1 |

Figure 1 shows a qualitative comparison between several propellant options for the moon. Although rocket engine specific impulse is usually a leading factor in the selection of a propulsion system, several other factors play an important role in the selection of an in situ propellant combination. Abundance in the lunar soil, ease of manufacturing, simplicity of the engine system, and technical background and experience with the propellant combination will also be important criteria. Currently, the database for production and propulsion technology is incomplete, and a quantitative selection of the "best" propulsion system to use for future lunar activities cannot yet be made.

The results of the Viking missions show that the martian soil is rich in magnesium, iron, and calcium, and that the polar caps may contain water ice or dry ($CO_2$) ice. In addition, the regolith potentially contains a permafrost of water ice at some depth below the surface, but the depth, quantities, and distribution have not been ascertained. Another important resource is the thin martian atmosphere. It has an average pressure of 0.07 - 0.10 psia and is approximately 95 percent carbon dioxide. It has been postulated that this source of carbon dioxide can be dissociated into oxygen and carbon monoxide to use as rocket propellants. An alternative option is to chemically react hydrogen or water (brought from Earth, obtained from the martian permafrost, or obtained from the martian moons) with the atmosphere directly to obtain oxygen and methane propellants.

Because the carbon monoxide and methane fuels are higher performing than the magnesium, iron, or calcium fuels, and because the processing of the gaseous atmosphere would be a much less complex task than mining and processing minerals from the regolith, the fuel selection at Mars appears to be between the carbon monoxide and methane. A trade-off between the lower performance of carbon monoxide and the necessity to either supply hydrogen from Earth or mine it at or near Mars for the methane appears to be a key factor in the selection of the "best" martian propellant system.

Benefits of In Situ Propellants

The utilization of indigenous space materials for propulsion offers several tangible benefits such as

reduction in initial launch mass, increase in payload delivered, or reduction in Mars trip time. The intangible benefits include the establishment of self-sufficiency, reduction in mission complexity, and development of technologies with potential terrestrial applications.

The most important benefit of in situ propellants is the potential to significantly reduce the mass required in low Earth orbit (LEO). When launch costs to orbit are counted in thousands of dollars per pound delivered to orbit, a reduction in the mass required from Earth can be translated to a cost savings for the overall mission. Figure 2 shows two figures taken from the literature (refs. 10,11) that depict the magnitude of the mass-in-LEO reduction for a lunar and a Mars mission. Figure 2a shows that the use of lunar produced aluminum/oxygen propellants for near-lunar operations can reduce mass-in-LEO requirements by 63 percent over the all-Earth produced hydrogen/oxygen baseline. Figure 2b shows that the use of Mars produced methane/oxygen propellants for Earth return of a 5 kg sample can reduce Mars injected mass by 67 percent compared to the baseline case that used solid propellants for Earth return. Even more importantly, the figure shows that producing the return propellants at Mars allows the mission to be accomplished with only one shuttle flight instead of two. The assumptions used in these two mission analyses can be obtained from references 10 and 11, or from reference 6, which contains a summary of these two analyses and five others taken from the literature that show significant launch mass reduction with in situ propellants.

Another potential benefit of in situ propellant utilization is increased payload capability. This benefit is a direct relationship: for every kilogram of return propellant that does not need to be delivered to the destination planet, a kilogram of payload can be added (for a specified initial mass). Additional payload capability can reduce the total number of missions or can allow for the delivery of larger payloads. Maximum payload capability is achieved when there is no reduction in initial mass in orbit. Alternatively, the two benefits can be combined, with some decrease in initial mass and some increase in payload capability.

Decreasing the trip time to Mars has emerged as a key requirement for manned

exploration, and in situ propellants can also help achieve this goal. Based on a specified mass in LEO, every kilogram of propellant that does not need to be saved for use at Mars can be used to increase the departure energy from the Earth. An increase in the available Earth departure energy will allow the spacecraft to travel a more direct path to Mars, thereby decreasing the trip time. Maximum reduction in trip time is achieved with no decrease in initial mass or increased payload. However, two or all three benefits can be achieved together to a lesser degree.

Other benefits can be obtained with the utilization of in situ propellants. If the ultimate goal is to expand human presence into the solar system and beyond, exploration must evolve into settlement. The ability to utilize resources available at the new settlement and to reduce or eliminate the dependency on the homeland is the ultimate measure of an independent establishment. At Mars, in situ propellant production may provide the ability to perform direct return missions, thereby eliminating the need to perform autonomous rendezvous in martian orbit. Finally, the development of the necessary technology to mine and beneficiate the lunar soil and to autonomously dissociate the martian atmosphere has potential terrestrial applications. The lunar technologies may help to produce metals more efficiently on Earth, and the martian technologies may help control the levels of carbon dioxide in the Earth's atmosphere.

TECHNOLOGY PROGRAM AT LEWIS

The Space Propulsion Technology Division at the NASA Lewis Research Center began investigating the utilization of lunar and Mars propellants in 1989. The objectives of the activity are to identify potential uses of in situ propellants and quantify the benefits, to determine likely propellant combinations and the status of their technology, and to establish the technology database that will be needed to develop engines that use in situ propellants. All work to date has focused on chemical propulsion options. To accomplish these goals, efforts have been concentrated in three areas, mission analysis, propellant characterization, and propellant performance.

## Mission Analysis

A contracted study (ref. 7) was initiated to assess the benefits of in situ propellants for a series of manned Mars missions. The two main objectives of the analysis were to determine the reduction in initial mass in low Earth orbit (IMLEO) for various propellant combinations using a consistent set of groundrules and assumptions, and to determine the cost of delivering and maintaining the in situ production infrastructure. All in situ propellant options were compared to a baseline scenario in which Earth-supplied hydrogen and oxygen was used.

The first objective of the study was to determine the mass of the necessary infrastructure, and the cost of delivery and maintenance. Unfortunately, the concepts for various production methods on the moon have not been developed completely. It was therefore difficult to establish accurate estimates of the mass, power, hardware resupply, and reagent resupply requirements for the lunar production plants. Although some Mars pilot plants have been tested (refs. 12, 13), mass and power estimates for full-scale models are still uncertain. The results of the first phase of the study, therefore, merely emphasized the need for further technology definition of the plant mass, power requirements, and production plant reagent resupply requirements.

Figure 3 (ref. 7) shows the results from the study for steady state missions (i.e. after the production plant has been delivered). The mission assumptions were a 2016 opposition-class flight profile and the delivery of 25 metric tons of useful payload to the surface of Mars. Figure 3a compares the initial mass in LEO for the baseline case with and without aerobraking and several Mars in situ propellant options. (Note that the in situ propellant options do not use aerobraking at Mars.) The results show that in steady-state operation, the in situ propellant options reduce the initial mass in LEO by approximately 50 percent over the all Earth-supplied option. More significantly, the in situ propellant options without aerobraking are comparable to the all Earth-supplied propellant option that uses aerobraking. While the dynamics of aerobraking at the Earth are relatively well understood, questions regarding high entry angles and velocities and variable atmospheric densities during dust storms

makes the success of the maneuver at Mars less certain. In situ propellants therefore offer a potential to replace aerobraking at Mars. If aerobraking at Mars becomes a reliable option, it can be combined with in situ propellant utilization to further reduce launch mass requirements. Figure 3b compares the same baselines with options that travel to the moon to pick up lunar propellants for the trip to Mars and back. Although these options further reduce the IMLEO, these results do not include the propellant production requirements. This further emphasizes the need for accurate determination of the infrastructure requirements before the actual economic benefits of in situ propellants can be determined.

## Propellant Characterization

The presence of metals on the moon that may be used in the solid phase poses unique technology challenges in the development of a rocket engine utilizing completely lunar indigenous resources. Several options exist to inject the metal fuel into the combustion chamber. One option is to suspend powdered metal particles in the liquid oxygen by means of a gellant. This is similar to metallized fuel technology that has been performed in the past (refs. 14,15). The resulting mixture would be a monopropellant with non-newtonian flow characteristics. Safety issues and rheological properties must be resolved before the monopropellant can be tested in a rocket engine. An alternative method to inject the metal fuel into the combustion chamber is to entrain the particles in an inert or fuel gas stream that would carry the particles into the chamber. The small amount of gas stream needed for this method would either be obtained at the moon or brought from Earth. A third method is to use a solid/liquid hybrid where the metal fuel is formed into rods or cylinders through which the liquid oxygen flows. Research into this alternative will not be discussed in this paper.

Monopropellant Hazards Assessment. The monopropellant concept has the potential to be a hazardous material, and hazards assessment and propellant formulation must be completed before any combustion experimentation can begin. The ultimate objective of the hazards assessment activity is to assign an explosive classification to the monopropellant so that the required safe handling procedures will be known. A preliminary goal of the

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hazards assessment is to test small, laboratory-scale quantities for explosive hazards such that formulation research can begin with assurances of safety.

The first phase, conducted at NASA White Sands Test Facility, consisted of mixing tests, where small amounts of aluminum, titanium, silicon, and iron powder were combined with liquid oxygen and then stirred at low speeds (approximately 600 rpm)

tests are only an indication that small scale formulations can be safely mixed.

The second phase of the hazards assessment consisted of mechanical impact tests, where a weight was dropped into a small sample of the monopropellant from various heights to determine the necessary energy to cause a reaction (ref. 17). Because impact test results can vary due to differences in the test apparatus, two well-characterized materials were used to act as points for comparison. The first test material was pentaerythritol tetranitrate (PETN), which is a solid Class A explosive known to be impact sensitive. Nitromethane, which is a flammable liquid, was also used because the oxygen/metal mixtures were

Table 2. - Summary of Results of Mechanical Impact Tests

| Sample | Approx. O/F Ratio | 50% Height (cm) | Impact Energy (Joules) | Impact Energy Density (Joules/cm$^2$) |
|---|---|---|---|---|
| PETN | NA | 51.0 | 45.4 | 19.4 |
| Nitromethane | NA | >123.0[a] | >109.4 | >46.8 |
| Titanium | 2.33 | < 15.2[b] | < 13.6 | < 5.8 |
| Al/Mg | 2.33 | 67.6 | 3.1 | 25.5 |
| Aluminum | 2.33 | >123.0[a] | >109.4 | >46.8 |
| Silicon | 2.03 | >123.0[a] | >109.4 | >46.8 |
| Iron | 2.03 | >123.0[a] | >109.4 | >46.8 |

[a]No reactions occurred at highest height available on apparatus
[b]Reaction occurred 100% at lowest height available on apparatus

while being monitored for any signs of chemical reaction. A total of 63 tests were conducted that varied the metal, metal to oxygen ratio, and presence and type of gelling agent (ref. 16). Figure 4 shows a typical temperature versus time trace taken from an iron/oxygen test. The absence of any large temperature spikes indicates that no chemical reaction occurred. Metal particles were also analyzed chemically before and after mixing to verify that no metal had been oxidized. There were no reactions observed in any of the tests. It must be stressed that the results of these tests can be applied only to the quantities and conditions that were tested. These results can not be extrapolated to larger scale batches, nor can they be used to assess the hazards of mixing performed at higher shear rates. Therefore, these

more liquid than solid and nitromethane is known to detonate under certain shock conditions. The materials tested in liquid oxygen for the monopropellant impact tests were aluminum, titanium, silicon, iron, 80% aluminum/20% magnesium alloy, aluminum/gellant, and titanium/gellant.

The results were reported in terms of a 50 percent height, which is the height at which a reaction occurred 50 percent of the time. Table 2 (ref. 17) lists the results of the mechanical impact tests. For all of the powders except titanium, the results of the impact tests indicate that it is safe to handle the monopropellants in the quantities and manners necessary to begin formulation and characterization of the monopropellant. The Al/Mg

alloy monopropellant should be treated as a secondary class A explosive (the classification of PETN) until more assessment tests can be completed.

Some of the tests that still need to be performed to fully classify the monopropellants include high-speed stirring and rotary friction, electrostatic discharge, water hammer, and detonation sensitivity. In addition, many of the tests would need to be repeated with larger quantities to determine the scaleability of the results.

Monopropellant Formulation and Characterization. With the encouraging results from the first phases of the hazards assessment program, a contracted effort (ref. 18) was initiated to prepare small formulations of the metal/oxygen monopropellant and to test some of the basic physical, chemical, and rheological properties. The objective of the contract is to determine the minimum amount of gellant needed to prevent settling of the metal powder but still allow for acceptable flow properties. The formulation experiments indicate that aluminum and silicon can be stably suspended in liquid oxygen with only one to two weight percent gellant (amorphous fumed silica). With this amount of gellant, very little settling of the metal powder was observed during the first 24 hour period. In general, the majority of the settling in such a gelled mixture will occur in the first 24 hours after formulation.

A second objective of the formulation and characterization task was to determine the ambient and pressurized burn rates of the monopropellants. If the monopropellant burns faster than the injection velocity into the chamber, then burning could propagate into the feed lines and the propellant tank, causing catastrophic failure. The ambient burn rate tests were conducted with the monopropellant submerged in a liquid nitrogen bath to prevent boil-off of the liquid oxygen before the start of the test. During the test, this liquid nitrogen acted as a strong heat sink and absorbed the energy created by the combustion of the monopropellant. Because of this rapid transfer of heat, the monopropellant combustion was unable to sustain itself after the solid propellant ignition charge was removed. Therefore, the ambient pressure burn rate of the monopropellants in the presence of a sufficient heat sink approaches zero, assuring that the flame will not propagate into the feed lines. Further tests will be needed to determine

the monopropellant burn rate when it is contained in an insulated line expected in an operational engine.

Pneumatic Powder Fuel Feed System. An alternative approach to the metal/LOX monopropellant is also being pursued. This option is pneumatic injection of the powder into the thrust chamber. A carrier gas is used to transport powder from the fuel storage device to the injector. Injector designs (impinging or coaxial), which are commonly used for gas mixing, can be used to disperse the powder and mix it with gaseous oxygen. The carrier gas may be either minimally reactive (e.g. nitrogen, helium) for system safety or an additional fuel (e.g. hydrogen, methane) to enhance performance. Since gases other than oxygen are only available in limited amounts on the moon, the carrier gas will probably be brought from earth. Thus, the amount of carrier gas used must be minimized to reduce the system's dependence on earth-based propellant. Related studies suggest the flow rate of gas could be limited to as little as 1.0% of the metal mass flow (ref. 19). However, an empirical data base to predict the physical characteristics of a two-phase gas-solid fuel flow must be developed.

Pneumatic conveying of dust or granular materials is common in commercial industries, but these pneumatic systems cannot be directly applied to a rocket fuel feed system. A novel system will be required to operate at high pressures, minimize the amount of carrier gas used, and accurately control the solids flow rate. Four types of pneumatic feed systems were considered. The simplest device was a fluidized bed. This is a vertical column with a bed of powder supported by a gas distribution plate. Gas is forced up through the powder, mobilizing the particles and giving the powder the appearance of being fluid. Some of the powder is entrained into the flow and carried away from the bed. Another device considered was a screw or worm feed system which would use a helical blade to carry powder from a hopper to either a carrier gas line or the thrust chamber. A third feeder option was a fluidized hopper. Although this is similar to a fluidized bed, it differs in that the powder is drawn from the core of the fluidized particles. In this manner, a more dense stream of particles is extracted. The last entrainment device considered was a fluidized piston. The powder would be placed in a cylinder and pressurized with the piston. Gas injected near the piston exit

would help expel the powder and carry it to the thrust chamber.

A pneumatic feed system for powder metal fuels was evaluated to define powder entrainment issues in a fluidized bed. In a cold flow experiment, the operation of a fluidized bed was studied with nitrogen fluidizing powdered metal oxides. The results indicated that a fluidized bed would require large amounts of gas to entrain a powder fuel, and bubbling and slugging would occur in the particle bed. Bubbling is the formation of small voids in the particle bed which rise to the surface. As the voids sporadically break through the surface, particles are spouted well above the surface. If the voids are large enough to fill most of the column cross section, the behavior is termed slugging. Slugging causes periodic rising and collapsing of the particle bed surface. Both of these behaviors would cause the flowrate of metal to be inconsistent. The fluidized bed was thus eliminated as a candidate for the fuel feed system. However, the cold flow tests did demonstrate like-on-like impingement of two powder/gas streams as a viable propellant injection method.

Because the undesirable attributes of the fluidized bed were largely due to free-surface behavior (bubbling and slugging), current powder feed system work is focussed on fabricating a fluidized piston feed system. A fluidized piston has no free surface, requires very little gas for entrainment, and operates independently from gravity.

Propellant Performance

Because the fuels available at the moon and Mars are nonconventional (i.e. containing no hydrogen or nitrogen), little technology base exists. Under the task of in situ propellant performance, several subtasks have been performed to determine the performance of these nonconventional propellants in a rocket engine. For the lunar propellants, the subtasks include evaluation of theoretical specific impulse performance, metal particle ignition and combustion, and experimental subscale engine performance. For the Mars propellants, the subtasks include evaluation of theoretical specific impulse performance, ignition characteristics, and engine performance and combustion characteristics.

Lunar Metal Propellant. The potential performance of a lunar metal/oxygen rocket engine is the subject of much debate. The Chemical Equilibrium Composition computer program (ref. 20) has been used to calculate the ideal performance of metal/oxygen propellants, but there are many non-ideal losses which cannot be accounted for by this program. From aluminized solid propellant experience, it is known that the metal oxide combustion products will condense into solid particles which do not maintain velocity and thermal equilibrium in the exhaust. In a solid propellant rocket motor, the lack of velocity and thermal equilibrium between the gases and particles results in specific impulse losses of 2 - 5% (ref. 21). In addition, some of the metals of interest are slow burning, a characteristic which may cause incomplete combustion of the fuel and further degrade performance (ref. 21). But, it is not possible to accurately estimate the total effect these and other losses will have on metal/oxygen propellant performance. Therefore, research has been initiated into experimentally determining the performance losses and evaluating techniques to minimize these losses.

Researchers at the University of Illinois are studying the fundamentals of lunar metal particle combustion. Several parameters are being varied to determine the effect on ignition delay, combustion time, and exhaust particle size. The results of this project will provide a basis for choosing a metal fuel.

The use of a pneumatic feed system for the powder fuel provides another tool to positively affect the lunar metal rocket performance. The carrier gas can dramatically affect performance even though only small quantities are used. The equilibrium calculation results presented in Figure 5 show that using hydrogen gas provides not only the added performance as expected but also lowers the combustion temperature. Lower combustion temperatures would reduce cooling requirements--a significant issue which must still be addressed.

Carbon Monoxide/Oxygen. Carbon monoxide and oxygen has been identified as a propellant combination that can be obtained completely from the martian atmosphere. Methane is another fuel which has been advocated for use at Mars, although the hydrogen for the methane will

either need to be delivered from the Earth or obtained from the potential martian permafrost. The advantage of methane over carbon monoxide is a higher specific impulse, which means less propellant to produce. The advantage of carbon monoxide over methane is the ability to obtain all of the fuel and oxygen from a single process (dissociation of atmospheric carbon dioxide) and the elimination of the need to mine hydrogen from the permafrost or transport it from Earth, Phobos or Deimos, or nearby asteroids. To evaluate the performance potential of carbon monoxide and oxygen, a theoretical parametric evaluation was performed.

A one-dimensional equilibrium computer code (ref. 20) was used to calculate vacuum specific impulse as a function of mixture ratio, chamber pressure, and area ratio. Figure 6 shows the results of this parametric study for a mixture ratio range of 0.25 to 2.0, chamber pressures of 1.4 and 20.7 MPa (200 and 3000 psia), and area ratios of 10, 60, 100, 200, and 500. As expected, chamber pressure has a small effect on ideal specific impulse, with only a 5 or 6 sec increase in specific impulse gained with an increase in chamber pressure from 1.4 to 20.7 MPa.

Figure 6 shows that theoretical specific impulses as high as 313 sec are predicted for an engine with a nozzle expansion ratio of 500. This value, however, is an ideal theoretical prediction, and an actual engine would not be expected to deliver this performance. Another computer code (ref. 22) was used to predict performance losses associated with finite-rate kinetics, two-dimensional flow, and boundary layer growth. A specific impulse efficiency was calculated by dividing the $I_{sp}$ values predicted by the various loss mechanisms by the ideal one dimensional equilibrium values. Figure 7 shows the specific impulse efficiencies obtained at the two different chamber pressures (1.4 and 20.7 MPa) when finite-rate kinetics are included in the analysis. The figure shows that while the predicted kinetic losses at the stoichiometric mixture ratio are as much as 8 percent at low chamber pressure, the kinetic losses are only a little more than 3 percent for the high pressure. These efficiencies would reduce the ideal predicted impulse of 313 seconds to 304 seconds at high pressure and 288 seconds at low pressure. Although the kinetic losses were the most significant losses in the theoretical analysis, other losses caused by two-dimensional effects and
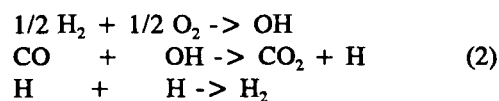
boundary layer growth will further reduce the predicted specific impulse. Reference 23 contains a more complete discussion on the theoretical results.

The theoretical analysis indicated slow kinetic reaction rates may affect the performance of carbon monoxide and oxygen. The slow kinetics may also affect the ignition of the $CO/O_2$ mixture. The equation for carbon monoxide oxidation is written as

$$CO + 1/2 \; O_2 \; -> \; CO_2 \qquad (1)$$
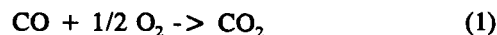
This reaction, however, has a high activation energy. One-dimensional kinetics simulation indicates that at the high temperatures and pressures typical in a rocket engine chamber, the energy barrier would be overcome, and the reaction would be self-sustaining. Therefore, to determine if CO and $O_2$ can be an effective propellant combination, an experimental program was conducted to investigate what ignition methods are required to initiate and nurture the reaction until it produces sufficient energy to overcome the high activation energy barrier and becomes self-sustaining.

One method of lowering the activation energy of a reaction is by the introduction of a catalyst to the system. Some transition metal and noble metal catalysts are known to promote this particular reaction, and are used in the automotive catalytic converter (ref. 24). The presence of small amounts of hydrogen in the system will also act as a catalyst. The key reactions in the mechanism are listed below.

$$
\begin{array}{lll}
1/2 \; H_2 \; + \; 1/2 \; O_2 \; -> \; OH & & \\
CO \quad + \quad OH \; -> \; CO_2 + H & & (2) \\
H \quad + \quad H \; -> \; H_2 & &
\end{array}
$$

The tests in this experimental program concentrated on the use of small amounts of hydrogen as the catalyst for the reaction. Once ignition was initiated, the hydrogen was no longer needed, and the reaction was allowed to proceed as a dry system.

The experiments were conducted in a spark-torch igniter, with small amounts of hydrogen added to the carbon monoxide gas stream (.15 to 1.0 percent of the CO by weight). The following conclusions were drawn from the experiments. Gaseous oxygen and dry carbon monoxide will not

light in a spark-torch igniter. Ignition was achieved, however, with as little as 0.0062 weight percent hydrogen in the carbon monoxide at a mixture ratio of 0.35 with ambient temperature oxygen. At higher mixture ratios and lower oxygen temperatures, more hydrogen was needed to initiate ignition (figure 8). A definite mixture ratio range exists where the carbon monoxide and oxygen will ignite and sustain combustion even after both the hydrogen and the spark are shut off. These mixture ratio boundaries are also dependent on the inlet temperature of the oxygen.

With the success of the ignition tests, another experimental program was started to investigate actual engine performance. Two measures of engine performance were taken during the experimental tests. The first was characteristic velocity, C*, which was calculated based on the measured chamber pressure and propellant flow rates. The second measure of performance was the vacuum specific impulse, which was calculated based on the measured propellant flow rates and measured thrust corrected to vacuum conditions by adding the nozzle exit pressure force. Both of these performance measurements were compared to theoretical values predicted by the computer code. Figure 9 shows the experimental and theoretical vacuum specific impulse efficiencies as a function of mixture ratio. Again, the theoretical ideal specific impulse was used as the basis for the efficiency calculation. Because the expansion area ratio of the test hardware was only 2.36, and because the kinetic losses discussed previously become more significant as expansion continues, the theoretical specific impulse efficiency is higher than that shown in figure 7. The theoretically predicted efficiencies are about 93 to 95 percent, while the experimental efficiencies are 85 to 89 percent. It was noted from the results for C* efficiency (ref. 23) that the difference in theoretical and experimental efficiency was most likely caused by incomplete energy release in the chamber that the computer code does not predict. The difference between theoretical and experimental specific impulse efficiency shown in figure 9, therefore, is likely caused by the same incomplete energy release.

In coordination with the experimental and theoretical engine performance research, analytical evaluations of the potential of carbon monoxide to be used as a regenerative coolant are also being conducted (ref. 25). The results indicate that, in general, liquid carbon monoxide acts as a better coolant than liquid oxygen for a $CO/O_2$ engine. The differences in cooling performance are small enough, however, that experimental testing will be needed to confirm the proper selection of coolant. In addition, engine cycle analyses have been performed to identify most likely engine cycles and operating characteristics of a carbon monoxide/oxygen engine system (ref. 26).

## CONCLUDING REMARKS

The resources available in the lunar soil and in the martian atmosphere contain large quantities of oxygen that can be used as the oxidizer in a rocket engine. Although very little hydrogen exists at either site, other resources, such as lunar metals and martian carbon monoxide, have great potential for use as fuels. Mission analyses have shown that the utilization of these in situ resources for near-planet and Earth-return propulsion can provide significant benefits for space exploration. These advantages include reduced launch mass, increased payload capability, reduced trip time to Mars, and establishment of base self-sufficiency.

Because the in situ propellant combinations are not commonly used, a technical database must be established to support the development of rocket engines that use these propellants. Recent activities have already produced some results towards this goal. Preliminary hazards assessment and formulation research have given strong indications that a liquid oxygen/powdered metal monopropellant is a safe and viable candidate for the moon. Single particle ignition research is focused on methods to reduce the performance losses anticipated with such a propellant. Similarly, technology work in the form of ignition and combustion performance evaluation has been conducted to build the necessary technology base to develop an in situ propellant rocket engine for Mars. This work has indicated that carbon monoxide and oxygen make a viable propellant combination for Mars.

These preliminary investigations into the use of in situ propellants indicate that the rocket engine technology should not be an impediment to achieving the potential benefits of in situ propellants. The magnitude of these benefits, however, is dependent

upon the cost of implacing the necessary mining and production infrastructure and maintaining and resupplying this equipment. Several studies have recently attempted to quantify these costs and deduct them from the total benefits of in situ propellants. These studies have concluded, however, that the state of the technology for in situ production (especially the mining of the moon) needs to be much further developed before the picture for the total mission will become clear.

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| Propellant | Fuel Abundance in Lunar Soil[a] | Simplicity | Specific Impulse[b] | Experience Base |
|---|---|---|---|---|
| $H_2/O_2$ | Very Low | High | High | High |
| $Al/O_2$ | High | Medium | Medium | Medium |
| $Si/O_2$ | High | Very Low | Low | Very Low |
| $Mg/O_2$ | Medium | Very Low | Medium | Low |
| $Ph/O_2$ | Low | Very Low | Low | Very Low |
| $S/O_2$ | Low | Very Low | Low | Very Low |
| $Fe/O_2$ | High | Very Low | Very Low | Very Low |

[a] See Table 1 for actual percentages.

[b] High: >400 sec;  Medium: 300-400 sec;  Low: 250-300 sec;  Very Low: <250 sec

● High     ◕ Medium

◒ Low     ⊖ Very Low

Figure 1. Comparison of Potential Indigenous Lunar Propellants
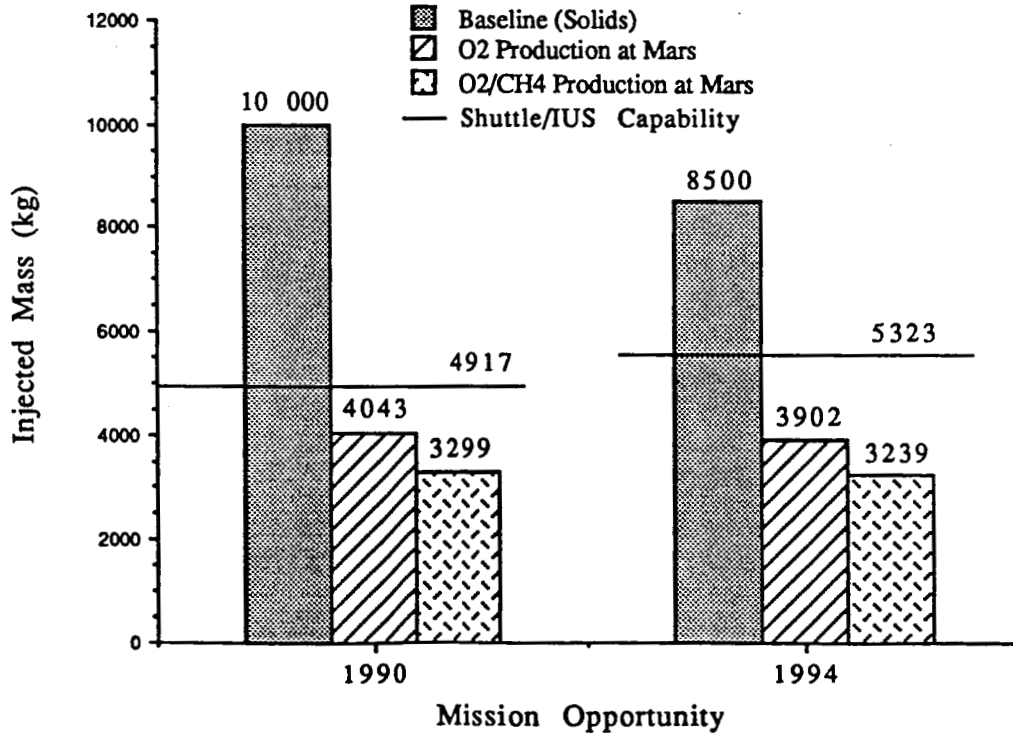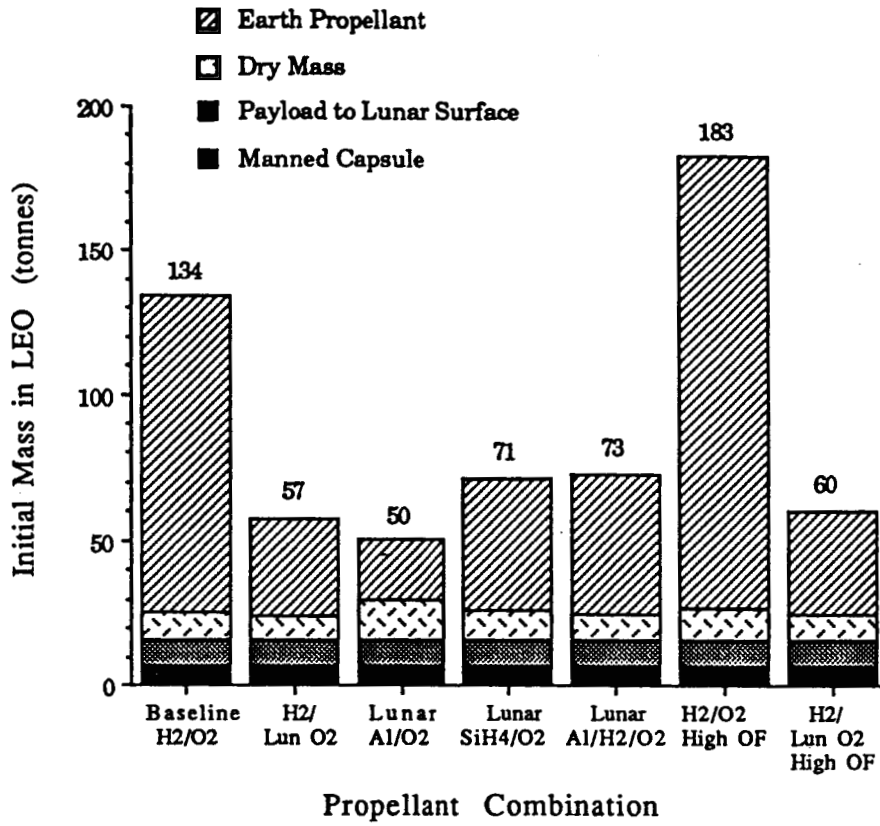
a) - Initial mass in LEO for a single manned lunar mission (ref. 10)
b) - Injected mass to Mars for a direct entry/direct return Mars Sample Return mission using Mars propellants (ref. 11)

Figure 2. - Potential reductions in initial mass in LEO

13
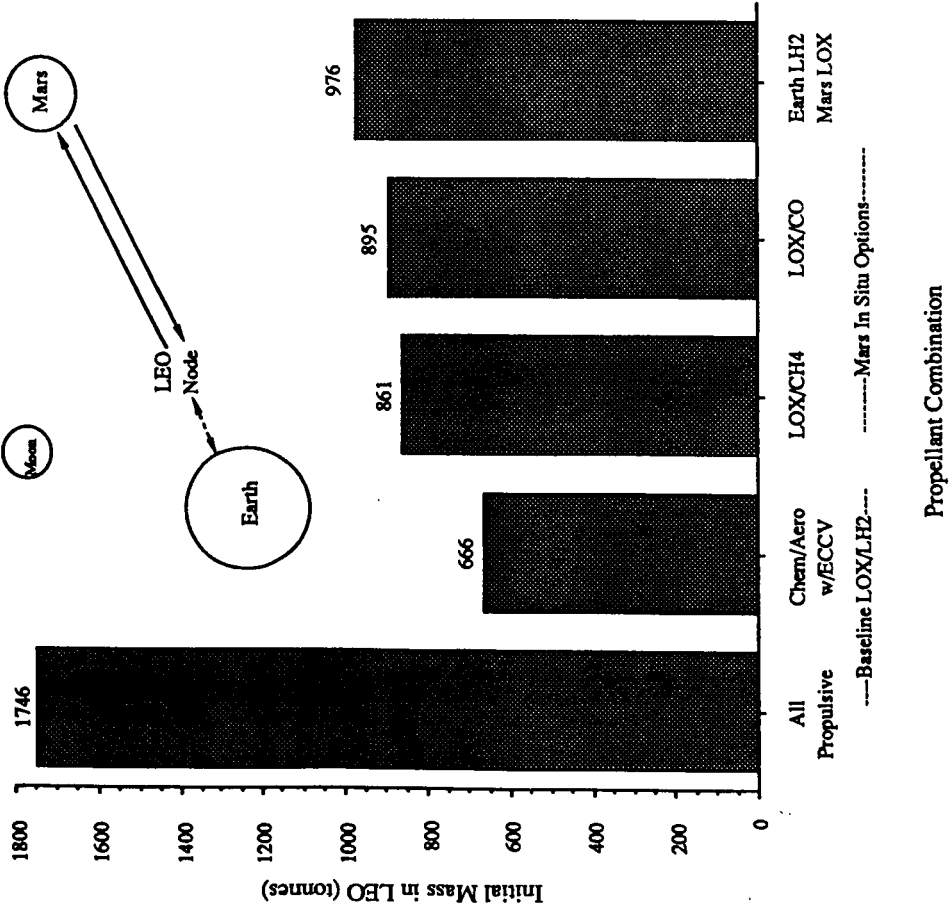
a.) - Mars propellants (in situ options are all propulsive)
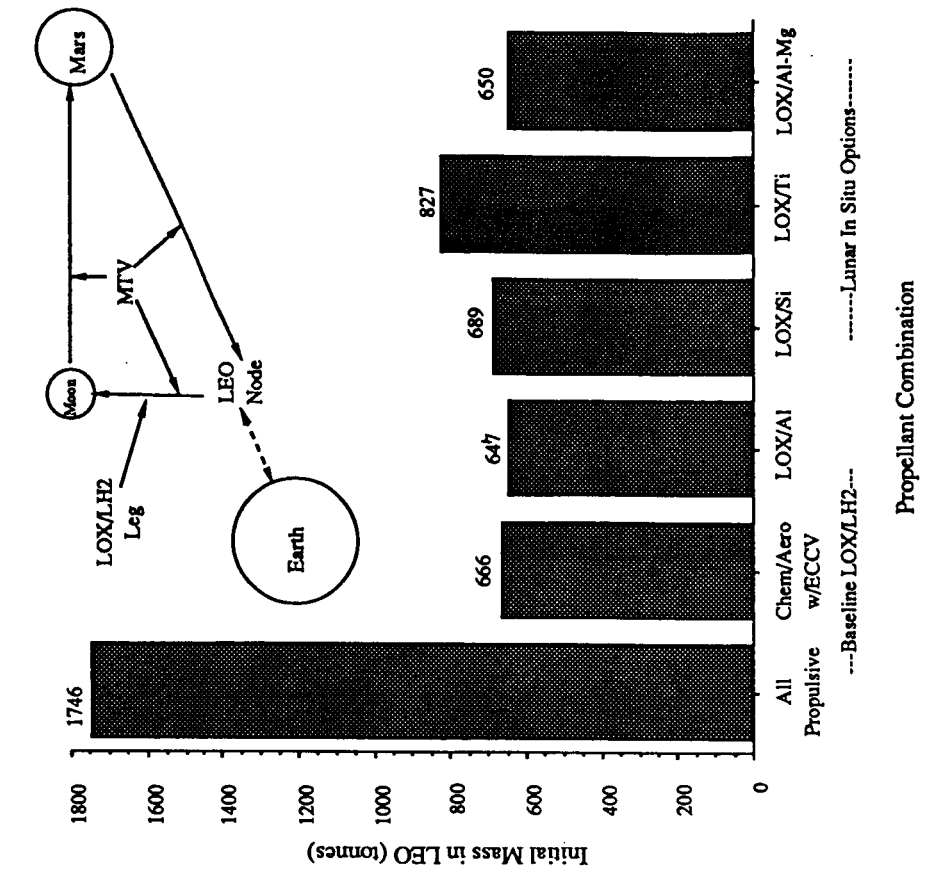


b.) - Lunar propellants (in situ options are all propulsive)

Figure 3. - Steady-state LEO mass requirements for manned Mars missions (ref. 7). Mission assumptions: 25 t payload to Mars surface; opposition-class, short stay time flight profile; 2016 departure.

14
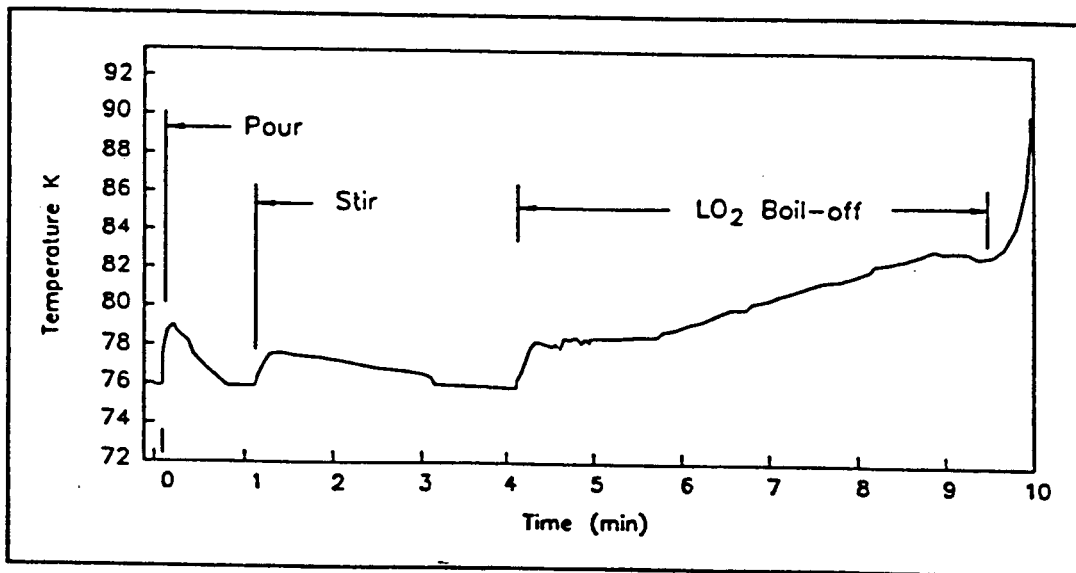
Figure 4. - Metal/Oxygen hazards assessment results: Temperature
trace of Fe/LOX mixing test.
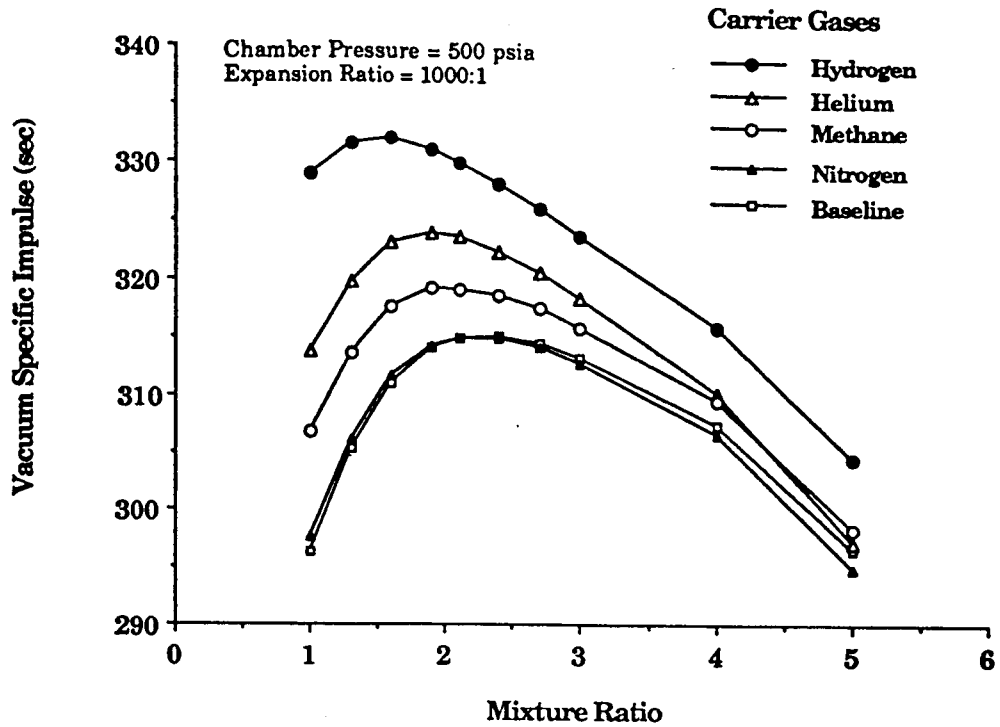
Figure 5a. The vacuum specific impulse performance of an aluminum/oxygen/carrier gas propellant with different carrier gases compared to a baseline using no carrier gas. The carrier gas is 2% of the fuel.
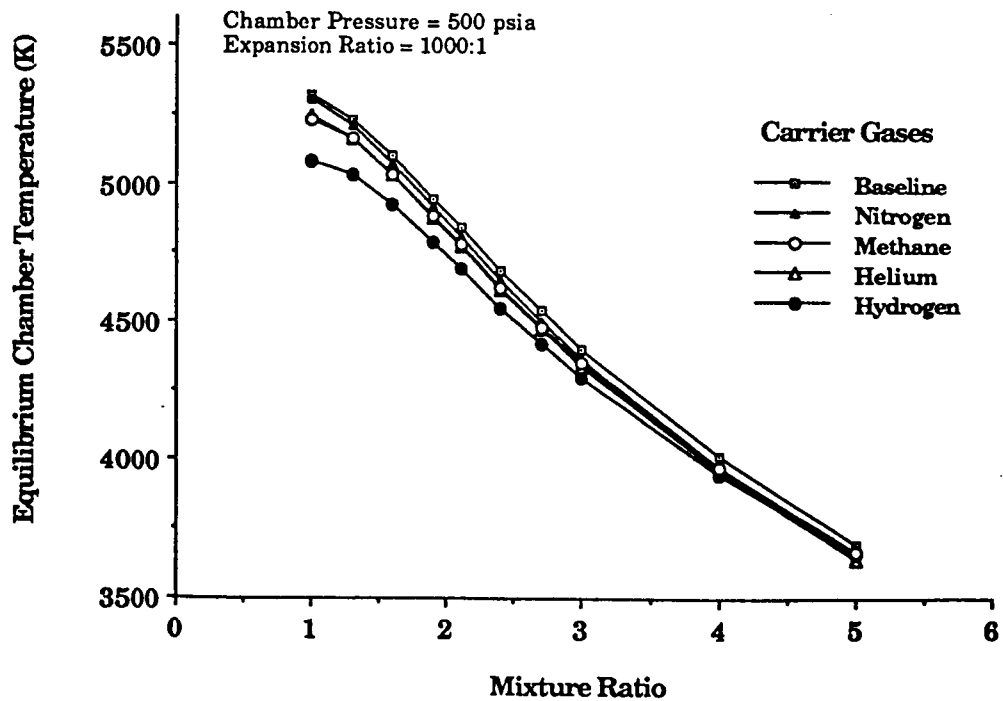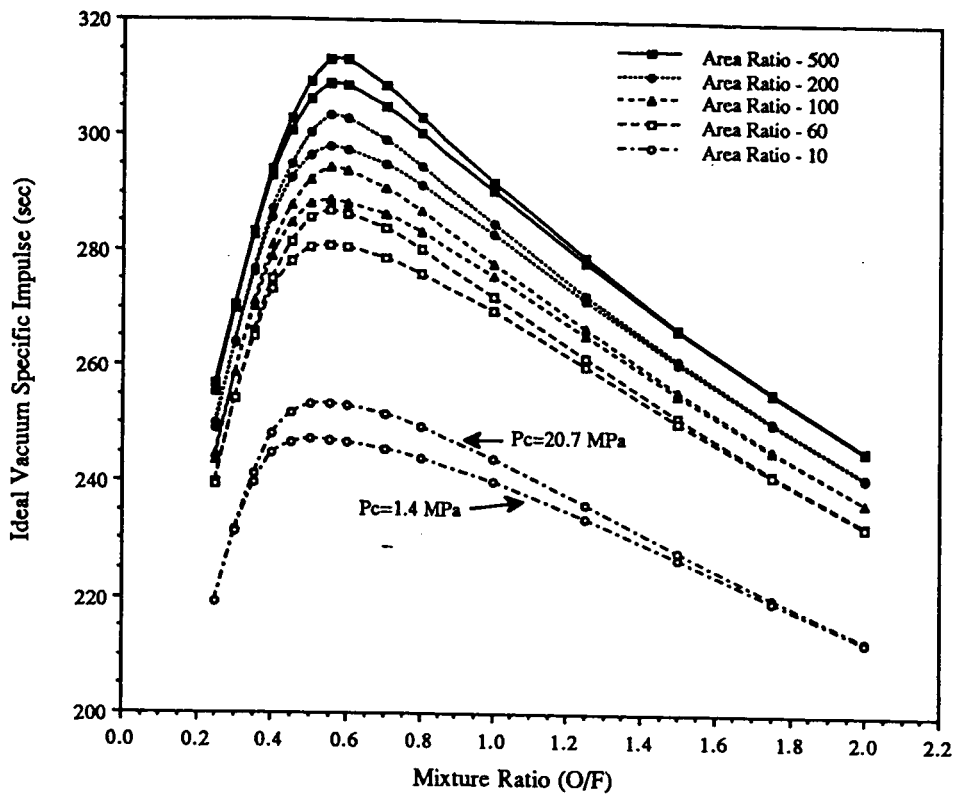


Figure 5b. The thrust chamber equilibrium temperature of an aluminum/oxygen/carrier gas propellant with different carrier gases compared to a baseline with no carrier gas. The carrier gas is 2% of the fuel mass.

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Figure 6. - Theoretical ideal performance predictions for LCO/LO2 combustion
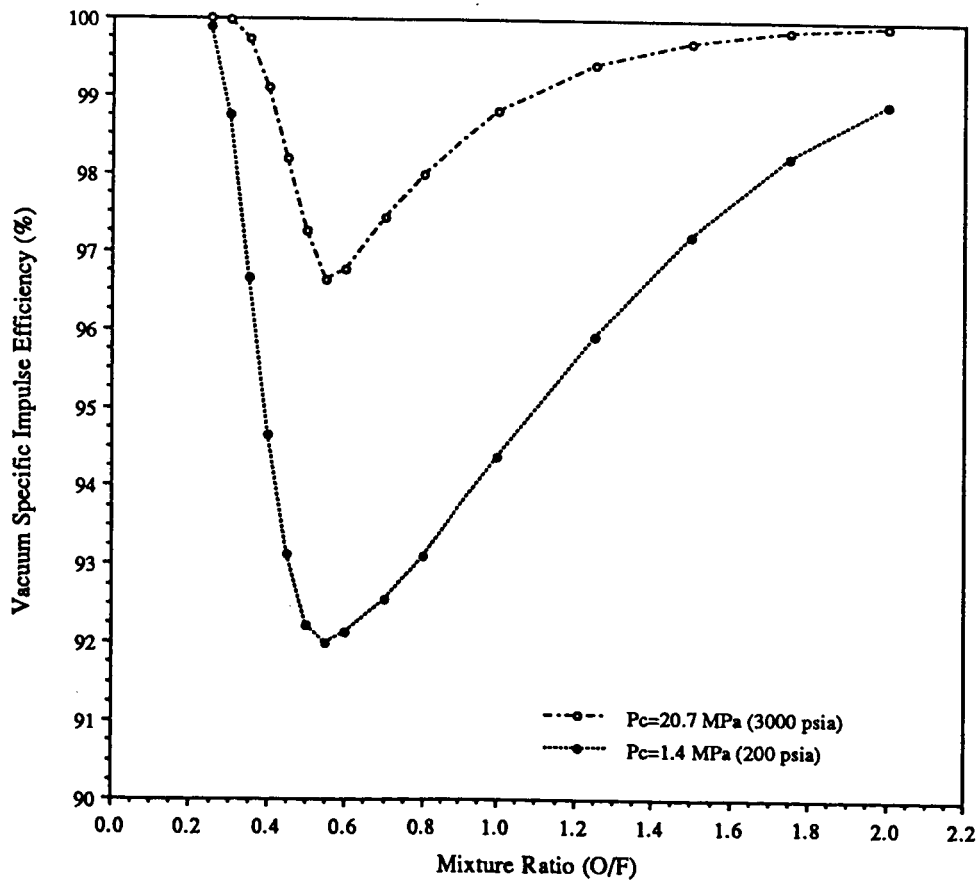


Figure 7. - Theoretical specific impulse efficiency assuming finite-rate kinetics for CO/O2 in an RL10 rocket engine for different chamber pressures (Area Ratio=205, LCO regen. cooled)
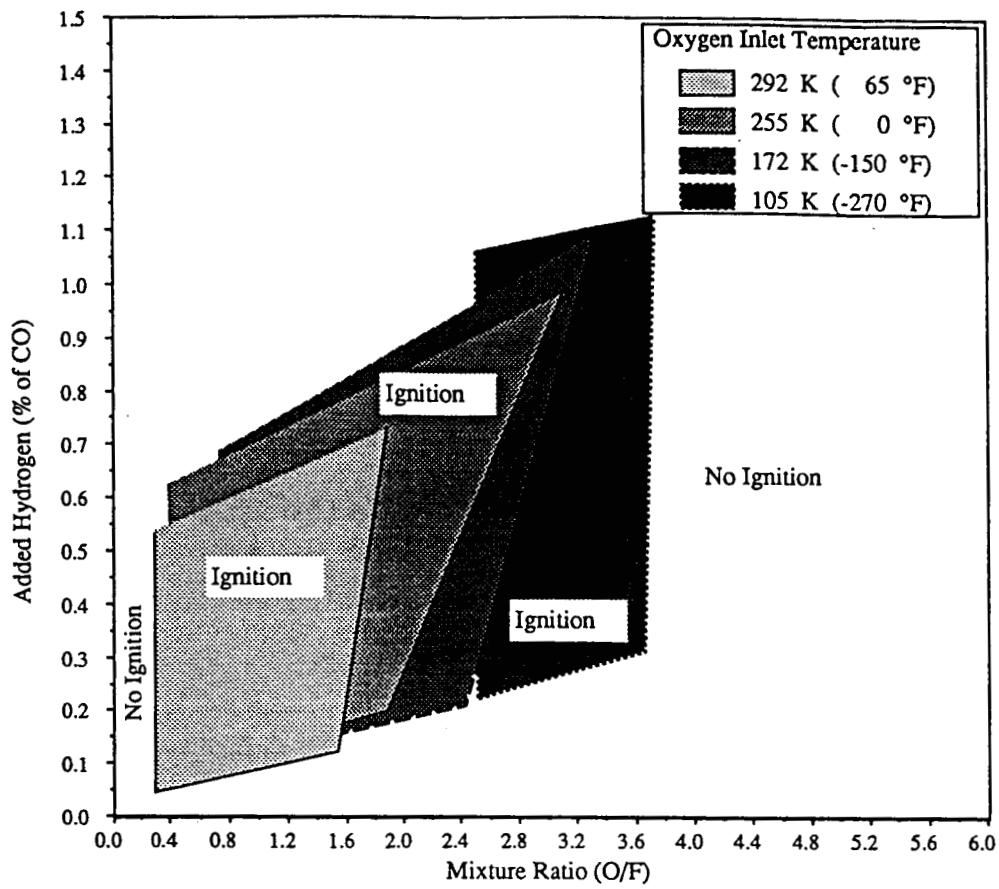
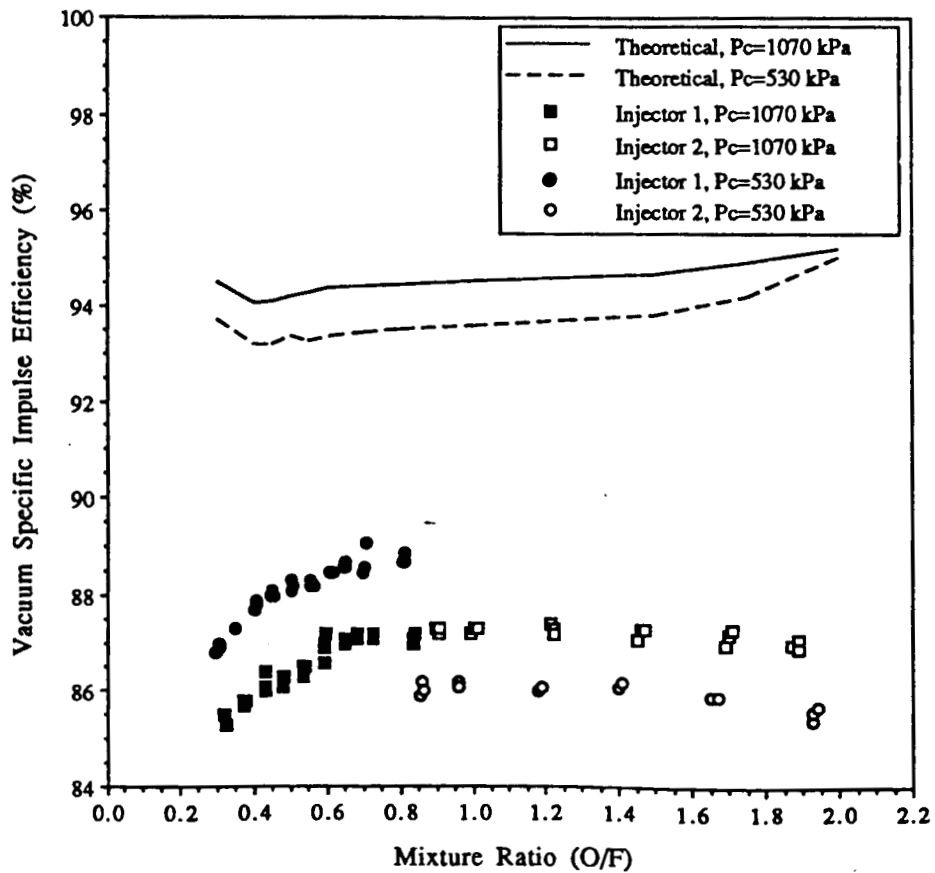Figure 8. - Ignition boundaries for dry CO/O2 ignition in a spark-torch ignitor



Figure 9. - CO/O2 combustion experimental vacuum specific impulse efficiency data (referenced to One Dimensional Equilibrium Theory)

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# REPORT DOCUMENTATION PAGE

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**13. ABSTRACT (Maximum 200 words)**

NASA's Lewis Research Center has supported several efforts to understand how lunar and martian produced propellants can be used to their best advantage for space exploration propulsion. A discussion of these efforts and their results is presented. A manned Mars mission analysis study identified that a more thorough technology base for propellant production is required before the net economic benefits of in situ propellants can be determined. Evaluation of the materials available on the moon indicated metal/oxygen combinations are the most promising lunar propellants. A hazards analysis determined that several lunar metal/LOX monopropellants could be safely worked with in small quantities, and a characterization study was initiated to determine the physical and chemical properties of potential lunar monopropellant formulations. A bipropellant metal/oxygen subscale test engine which utilizes pneumatic injection of powdered metal is being pursued as an alternative to the monopropellant systems. The technology for utilizing carbon monoxide/oxygen, a potential martian propellant, has been studied in subscale ignition and rocket performance experiments.

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Extraterrestrial resources; In situ propellants; Space exploration; Space propulsion

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