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The conceptual design of a Mars nuclear landing and ascent vehicle utilizing indigenous propellant

Zubrin, Robert M., Ph.D.

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U·M·I
300 N. Zeeb Rd.
Ann Arbor, MI 48106

The Conceptual Design of a Mars Nuclear Landing and Ascent Vehicle Utilizing Indigenous Propellant

by

Robert M. Zubrin

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(Chairperson of the Supervisory Committee)

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Abstract

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Robert M. Zubrin

Chairperson of the Supervisory Committee:
Professor Reiner Decher
Department of Aeronautics and Astronautics

Interplanetary travel and exploration can be greatly facilitated if indigenous propellants can be used in place of those transported from Earth. Nuclear thermal rockets offer significant promise in this regard, as in principle, any gas at all can be made to perform as a propellant to some extent.

In particular, the Martian atmosphere is composed of 95% CO_2. Under Martian conditions, this gas can be liquified by simple compression to about 100 psi, and remains storable without

refrigeration. When heated to 2400 K and exhausted out of a rocket nozzle, a specific impulse of about 226 s can be achieved. This is sufficient for flights from the surface to orbit or from one point on the Martian surface to any other point on the planet. Because the power requirements for acquiring CO_2 are quite low, the propellant acquisition system can travel with the vehicle, allowing it to refuel itself each time it lands. Thus this vehicle concept, which is termed a NAV (Nuclear Ascent Vehicle), offers unequalled potential to achieve planetwide mobility, allowing complete global access for the exploration of Mars, and potentially can reduce the initial mission mass in LEO as well.

This dissertation presents the results of a extensive study which centered on the conceptual design of a NAV vehicle with surface to orbit capability. Carbon dioxide was the propellant of choice, with some examination of alternate concepts using other propellants. The NAV configuration defined by the conceptual design was used as a basis for defining engine performance requirements, and a detailed study of a potential NAV engine that could meet these requirements was then conducted. The resulting NAV/engine combination was then examined in a series of trade studies to determine its potential merit in assisting in the exploration of Mars.

It is found that the use of a NAV vehicle can increase the exploratory effectiveness of a manned Mars mission by about a factor of 10, and reduce the Earth to orbit launch mass required to sustain an ongoing program of Mars exploration by about a factor of 2.

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Dedication

This dissertation is dedicated to my step-children, Eliot and Sarah, and all like them, who still know that the universe is infinite, and Hope all powerful.

Introduction

Interplanetary travel and exploration can be greatly facilitated if indigenous propellants can be used in place of those transported from Earth. Nuclear thermal rockets offer significant promise in this regard, as in principle, any gas at all can be made to perform as a propellant to some extent.

In the past, nuclear thermal rockets were developed and extensively investigated under the NERVA[1] program for use with hydrogen propellant to achieve medium thrust with high (900 s) specific impulse. What was not investigated however, was the potential benefits of utilizing propellants, which while offering much lower engine performance than hydrogen, are nevertheless readily available at planetary destinations.

In particular, the Martian atmosphere is composed of 95% CO_2. Under Martian conditions, this gas can be liquified by simple compression to about 100 psi, and remains storable without refrigeration. When heated to temperatures equivalent to those accomplished with hydrogen in the NERVA program and exhausted out of a rocket nozzle, a specific impulse of about 220 to 260 seconds can be achieved. This is sufficient for flights from the surface to highly energetic orbits or from one point on the Martian surface to any other point on the planet. Because the power requirements for acquiring CO_2 through atmospheric compression are quite low (about two orders of magnitude less than those required to produce chemical fuels) the propellant acquisition system can travel with the vehicle, allowing it to refuel itself each time it lands. This vehicle concept[2], which is termed a NAV (Nuclear Ascent Vehicle), offers unequalled potential to achieve planetwide mobility, by allowing complete global access for the exploration of Mars. In addition, by eliminating the necessity of transporting ascent propellant to Mars, the NAV can significantly reduce the initial mass in LEO of a manned Mars mission. The NAV concept is illustrated in Fig. 1.

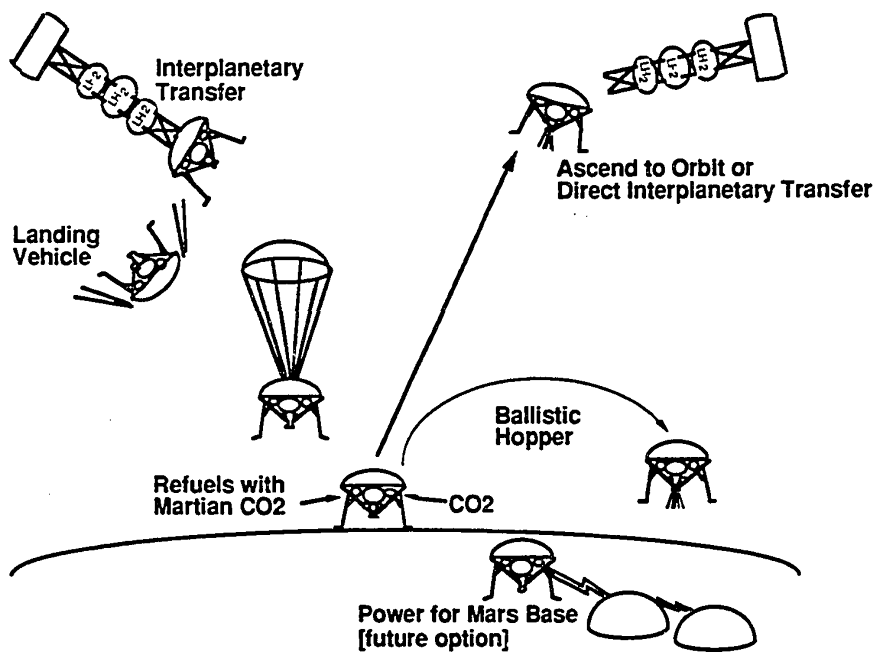


Fig. 1. The NAV concept in a variety of modes.

Alternative propellants for use in a NAV vehicle that are either indigenous to Mars or can be manufactured out of Martian materials include water, methane, nitrogen, carbon monoxide, and argon. While not as accessible as CO_2, some of these offer other advantages. For example both water and methane yield high specific impulse (350 and 600 seconds, respectively) allowing direct ascents from the Mars surface to trans-Earth injection orbits.

This dissertation presents the results of a extensive study which centered on the technical issues associated with the conceptual design of a NAV vehicle capable of launch from the surface of Mars to orbit. Carbon dioxide was the propellant of choice. A limited examination of alternate concepts using other propellants was also carried out. The NAV conceptual design was used as a basis for defining engine performance requirements, and a detailed study of a potential NAV engine configuration that could meet these requirements was then conducted. The resulting NAV/engine combination was then examined in a series of trade studies to determine its potential merit in assisting in the exploration of Mars.

An overall outline of this study is as follows:

Introduction.

Chapter 1. Definition of the operating scenarios, mission requirements, and design constraints followed in the study. Scenario and mission requirements include definition of such issues as payload, capability, and endurance. Design constraints include such issues as engine temperature limits as they effect specific impulse, the selection of technologies assumed available, and crew radiation limits.

Chapter 2. Conceptual design of rocket ballistic NAV vehicle. The design includes sizing of engine, shield, tankage, structure, habitat, and propellant acquisition system.

Chapter 3. Analysis and Design of the NAV engine based on the requirements developed from Chapter 2.

Chapter 4. Performance evaluation of the NAV.

Chapter 5. Discussion of an alternative chemical rocket ballistic hopper system based on CO/O_2 combustion.

Chapter 6. Discussion of methods of propellant production on Mars for both the NAV and the CO/O_2 ballistic hopper.

Chapter 7. Studies comparing the merit of the NAV and the CO/O_2 ballistic hopper in achieving global access on Mars and in reducing the initial mass in LEO of manned Mars missions.

Chapter 8. Examination of alternative NAV concepts using propellants other than CO_2 and their potential merit in accomplishing either or both of the two objectives of achieving global access on Mars and reducing the initial mass in LEO of manned Mars missions. Issues affecting possible commonality of NAV engines to more than one type of propellant are identified.

Chapter 9. Conclusions.

Following the main body of the dissertation there are two appendices. Appendix A, "A Primer on Rocketry and Mars Missions," provides a basic introduction to the science and terminology of these areas for those readers who are new to the field. Appendix B provides a glossary of the acronyms used throughout the text.

Chapter 1. Performance Objectives

Prior to undertaking the design of the nuclear ascent vehicle (NAV) using indigenous Martian propellant, it is useful to state the operational modes and performance objectives intended for the vehicle. The following is therefore stated as a list of design goals.

1.1 The nuclear ascent vehicle (NAV) analyzed in this study shall operate as Mars surface to orbit and surface to surface transportation for a crew of 3 astronauts. The vehicles shall also function as living quarters and mobile base for the crew for periods up to 1 (Earth) year, and be capable of operating for that period without resupply or external logistic support.

1.2 The design orbit (maximum expected capability) for the NAV shall be a 250 km altitude circular orbit around Mars. The vehicle shall be designed so as to be able to ascend to such an orbit using nuclear thermal rocket (NTR) heated CO_2 propellant, and to be able to descend from such an orbit using aerobraking.

1.3 The vehicle shall be designed for a lifetime (with resupply and orbital maintenance) of at least 3 descents and ascents from its design orbit, and at least 15 suborbital hops and landings.

1.4 The vehicle shall be able to provide a sufficient power supply to allow it to refuel for an ascent to its design orbit in no more than 60 days.

1.5 While a propellant temperature limit of 2800 K appears attainable, in this study a nominal value of 2400 K is adopted in order to introduce an element of conservatism into the design. Calculations based on JANNAF[3] thermochemical data assuming equilibrium flow indicate that for a realistic nozzle with 98% efficiency and an expansion ratio 100, the expected Isp of CO_2 propellant at this temperature will be 224 seconds at a chamber pressure of 8 MPa, while for an expansion ratio of 200 it will be 227 seconds, and these values will be used as the basis for the NAV approximate sizing.

1.6 The NAVs will incorporate no technology or material that can not be expected to be available for use by the year 2010.

1.7 The NAVs will be so designed that the crew shall not receive a whole body radiation dose from the NAV greater than 5 Rem over the course of one descent from and ascent to the design orbit.

1.8 The operating scenario for the NAV will be as follows:
The NAV shall be delivered to Mars on a conjunction class orbit and braked by a disposable separate aerobrake into its design

orbit. Astronauts arrive on a separate manned vehicle and board the NAV, using it to descend from orbit, hop around Mars, and finally ascend for rendezvous with their manned spacecraft. The astronauts shall then depart Mars, leaving the NAV in its design orbit, ready to be resupplied and reused.

Chapter 2. Conceptual Nuclear Ascent Vehicle Design

2.1. Engine Sizing

As a starting point in the design process, it is assumed that the NAV will have a dry mass of 50 metric tons (t, or tons) and has the capability to take off from the mid-latitude Martian surface for a 250 km altitude orbit. The orbital velocity is 3.6 km/s, from which a planetary rotational velocity gain of 0.2 km/s (corresponding to a latitude of 34 degrees) is subtracted. Multiplying this result by 1.2 to account for gravity losses, a total ascent ΔV of 4.08 km/s is obtained, to which a landing ΔV of 0.3 km/s is added, for a total of 4.38 km/s. Assuming an Isp of 224 seconds, the vehicle has an effective exhaust velocity, V_e, of 2195 m/s. Using the rocket equation:

$$\text{Mass Ratio} = \frac{M_i}{M_f} = \exp\left(\frac{\Delta V}{V_e}\right) \tag{2.1}$$

where M_i is the vehicle initial or "wet" mass (including propellant), M_f is the vehicle final or "dry" mass (with all propellant exhausted), it is found that the NAV requires a mass ratio of 7.35 which is rounded off to 7.5 to allow for extra margin. Thus the ground lift off mass is 375 t. If an initial thrust/weight ratio of 1.5 is desired, the thrust required then is 2080 kN, and the power required is 2513 MWth.

Using a modified version of the "Chi scaling"[4] for updated NERVA derivative NTR engines developed during under the NASA LeRC/SAIC/Martin Marietta Task Order Contract, approximate masses can be derived for the NAV engine. More exact results will be derived in chapter 3, when the NAV engine will be designed from first principles and its mass calculated. The scaled engine mass presented in this chapter will only lead to an approximate total vehicle mass, and thus an approximate vehicle performance which will be refined as the design process proceeds. The scaling relationships used are:

$$M_r = 810 + 13(T/P) \tag{2.2}$$

$$M_n = 0.023(E^{1.1})(T/P) \tag{2.3}$$

$$M_h = 160 + 0.74(T) \tag{2.4}$$

where M_r is the mass of the reactor (kg), M_n is the mass of the nozzle (kg), M_h is the mass of the non-nuclear hardware associated with the engine (kg), E is the nozzle expansion ratio (dimensionless), T is the engine thrust (kN), and P is the chamber pressure (MPa). On the basis of these scaling relationships, it is estimated that a NAV engine (Fig. 2) of this power and thrust will have the characteristics given in Table 1.

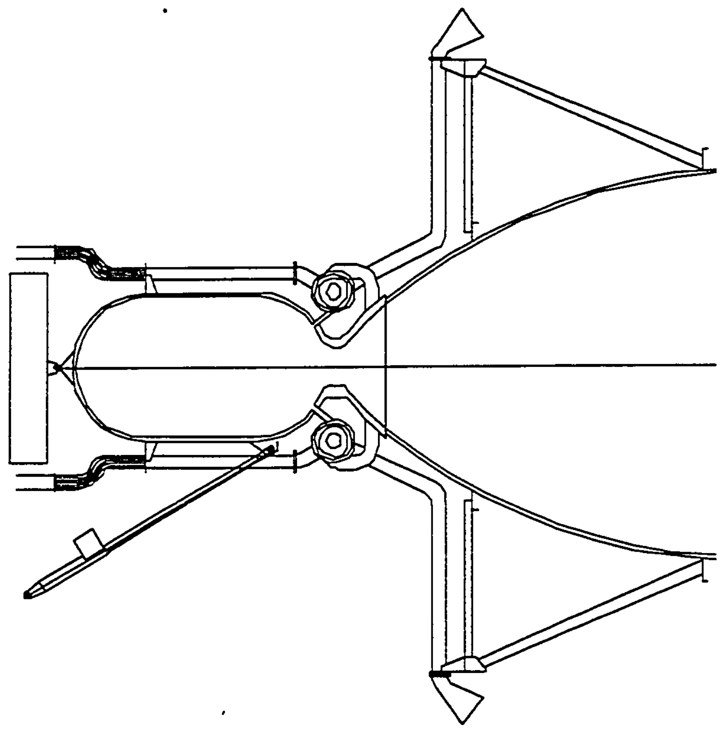


Fig. 2 The NAV Engine

Table 1. NAV Engine

Power	2513 MWth
Thrust	2080 kN
Chamber Pressure	8.0 MPa
Isp	224 s
Expansion Ratio	100
Reactor Mass	4190 kg
Shield Mass	8580 kg
Auxiliary Mass	1700 kg
Nozzle Mass	960 kg
Total Mass	15430 kg

The reactor mass given in Table 1 includes the pressure vessel and the mass of the boron-aluminum-titanium-hydride "BATH" reactor internal shield. The shield mass given is external shield and the basis for its estimate is given in section 2.2. The power to mass ratio of the reactor is scaled from NERVA engines, taking into account the improvement in power density possible at increased pressures. (NERVA engines operated between 3.1 and 4.3 MPa, but the 8.0 MPa (1170 psi) is not excessive for a modern engine.) Since the NAV engine would probably use a derivative of a particle[5] or pebble bed[6] design, and these are capable of much higher power densities than NERVA derivatives, the use of NERVA scaling may be seen as a very conservative design assumption. (The more compact reactors made possible by a particle bed approach would also greatly reduce the shield mass.) The auxiliary mass given in Table 1 includes the mass of the turbopumps, propellent lines, thrust structure and gimbal, valves, actuators, instrumentation and electronics.

2.2. Shield Sizing

In studies of application of NTR propulsion to manned space missions, the propellant tanks are always placed between the astronauts and the reactor in order to use the mass of the propellant as shielding material. Since the propellant mass is very large, it is an extremely effective shield, and extra shadow shielding is actually required only during a critical final stage of the burn when the propellant supply is nearly exhausted. Furthermore if the propellant drainage is arranged so that the portion remaining during the final stages of the burn is constrained to form a shadow shield, and if this final portion is only used during reactor cooldown when power levels are dropping exponentially (and the decay gamma radiation spectrum is softer than the prompt fission gamma spectrum), then the requirements of the solid external shadow shield are further reduced.[7] For unmanned missions the doses are low enough that the external shield may be discarded altogether, and some authors have even argued that this may be possible in manned configurations.[8] However in detailed studies of the shielding problem of manned missions using the NERVA engine, it was deemed necessary to have an external disc shield, the design of which is shown in Fig. 3. This shield was taken as the basis for a disc shield for the NAV, except that since a typical NERVA manned application would position the crew about 3 times as far from the reactor as is practical in the relatively compact NAV, it was decided to make the NAV's disc shield thicker than the NERVA disc shield in order to reduce the gamma ray flux by at least an order of magnitude. Greater distance from a source reduces the radiation dose received in inverse proportion to the square of the distance. Thus extra material shielding is required of the NAV in order to compensate for the stronger $1/r^2$ distance shielding effect available to a NERVA space transfer vehicle configuration. The NAV disc shield is depicted in Fig. 4. It is the same as the NERVA shield except that 7.62 cm of stainless steel has been added. The decrease of gamma radiation flux caused by a material shield can be described by:[9]

$$F(x) = F_i \exp(-\mu x) \tag{2.5}$$

where F(x) is the gamma flux at a given point, x, F_i is the initial flux entering the shield from the side where the source is located, and μ is a constant describing the shielding capability of the material. Assuming that the typical gamma ray of interest is 1 MeV, then the μ value[9] for iron is 0.48/cm. The gamma ray flux will then be reduced by an extra factor of exp(-(0.48)*(7.62)) = 0.026. This more than compensates for the loss of a factor of 0.11 in $1/r^2$ distance shielding that a NERVA driven crew would obtain by virtue of being located 3 times as far from their engine as the crew of the NAV.

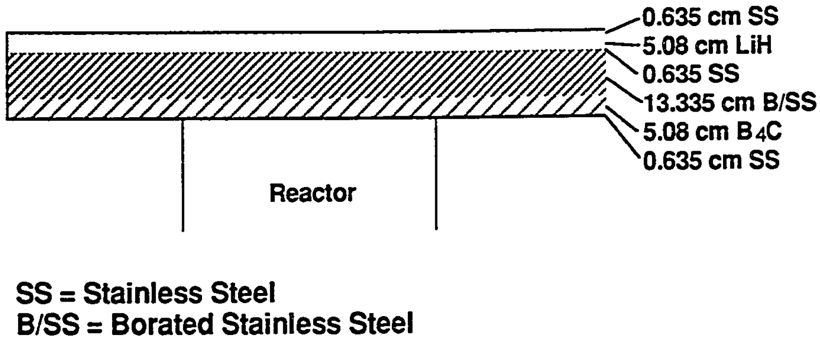


Fig. 3 NERVA Disc Shield

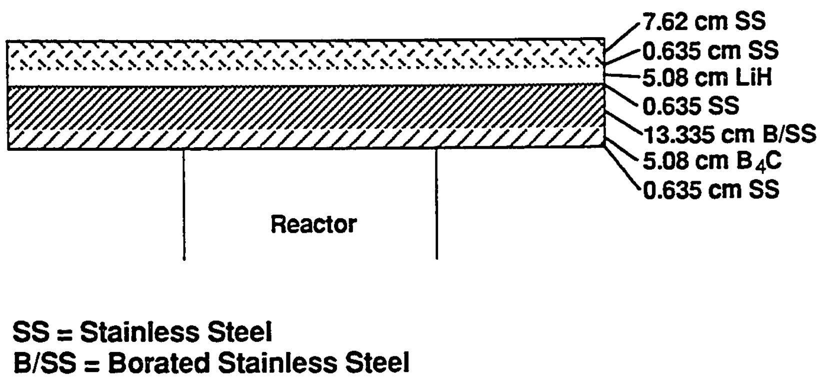


Fig. 4 NAV Disc Shield

The radius of this top disc shield is 1.0 m and its mass is 6208 kg. It is 34 cm thick.

In order to prevent neutron escape from the reactor, which might cause activation of some of the parts of the NAV structure located near the engine, it was decided to surround the reactor completely with a side shield, consisting of 30% B_4C, 60% LiH, and 10%

stainless steel, by volume. The NAV reactor is moderated internally by LiH (with its Li^6 content depleted), and so its neutron spectrum is essentially thermal. Thermal neutrons are very effectively absorbed by B_4C. In fact it has been shown experimentally[10] that a 0.1 cm thick coating of B_4C paint will absorb over 90% of an incident thermal neutron flux. The side shield contains the equivalent of a layer of B_4C 3.9 cm thick, so the possibility of thermal neutron leakage through this shield is essentially zero. The side shield is 13 cm thick and has a mass of 2372 kg. The two dedicated shielding elements directly associated with the reactor have a combined mass of 8580 kg, as given in Table 1.

It was decided to further add two extra shielding features to the design. One of these is a coaxial propellant tank wrapped around the reactor, surrounding it with a layer of liquid CO_2 1 m thick during ground operations, when humans may be operating in close vicinity to the shut down reactor, as well as during takeoff when the reactor is at maximum power and tangential leakage of gammas might result in a reflected groundshine dose to the astronauts located on the control deck. (During takeoff the crew is completely protected against any direct (i.e. non reflected) radiation by the large mass of propellant in the main tank.)

During landing, however, the coaxial tank would ordinarily be empty, and the main propellant tank would be nearly empty. The shielding problem is thus most critical during landing, with reflected groundshine being the major threat. The exposure danger is mitigated somewhat by the brevity of the landing maneuver and the fact that the engine is operating at less than 10% of maximum power during the final landing phase. It was decided that the best way to eliminate this problem was to position an additional shield on the control deck itself, directly below the astronauts. In order to circumvent the protection of this shield, reflected radiation would have to be transmitted to and from distances several hundred meters away, with a $1/r^4$ relationship reducing the dose from such "trick shots" to negligible levels. The shield itself is a square 2 m by 2 m and is made of lead 13.2 cm thick. It has a mass of 6000 kg. Assuming an average gamma energy of 1 MeV, the μ value of lead is 0.87/cm, and so using equation 2.3 it can be seen that this shield will reduce the transmitted dose by about 5 orders of magnitude.

2.3. Tankage Sizing

On the basis of the considerations given in section 2.1, above, it is apparent that the NAV should have a propellant capacity of about 325 t. Assuming a 3% ullage factor, 301.7 t of liquid CO_2 can be carried in a spherical tank of 4 m radius, and thus the NAV main propellant tank was sized. (The density of liquid CO_2 is 1.16 t/m^3.) The size of the coaxial secondary tank was determined by geometric, engine gimballing, and shielding considerations as having an inner radius of 1.395 m, and outer radius of 2.418 m, a cylindrical length of 2.418 m, and top and bottom hemispherical radii of 0.511 m, resulting in a CO_2 capacity (assuming 3% ullage) of 44.4 t. The total propellant capacity is thus 346.1 t.

Since the NAV is launched from Earth with its propellant tanks nearly empty, and lifts from Mars with accelerations less than 0.6 Earth g's, it is apparent that the thickness of the NAV's tanks will be determined not by launch loads, but by the hoop stress caused by the necessity to store the CO_2 at pressure. This can be seen by the fact that when the tank is fully fueled with CO_2, the total stress resulting from the launch load is equal to the net downward force divided by the area of the strip of material around the tank's equator that must react it. This is given by:

$$\text{stress} = \frac{\text{force}}{\text{area}} = \frac{\frac{4}{3}\pi r^3 \rho g}{2\pi r t} = \frac{2\rho g r^2}{3t} \qquad (2.6)$$

where ρ is the density of CO_2, g is the acceleration, r is the radius of the tank, and t is the thickness of the tank wall. Since the launch load pressure of the fluid column, P_L, on a small area at the bottom of the tank is given by:

$$P_L = 2\rho r g \qquad (2.7)$$

equation (2.6) can also be written:

$$\text{stress} = (1/3)P_L(r/t) \qquad (2.8)$$

Equation (2.8) is in similar form to the standard equation for the hoop stress exerted on a tank by a pressurized gas (a form of which is given in equation (2.9) below.) If the tank is full and the upward acceleration is 0.6 Earth g's (6 m/s^2), then P_L is equal to 111 kPa, or 16 psi. This is an order of magnitude less than the gas pressure, and its significance as a cause of tank stress is reduced further by the factor of 1/3 which appears in equation (2.8).

At an average ambient Martian temperature of 215 K, CO_2 liquifies under 100 psi[11], but to put a large amount of margin into the system the tanks are designed as if they contained propellant at 150 psi. (Martian temperatures vary between 170 and 280 K, but thermal inertia can be used to keep the tank temperature near the average value. For example if the main tank is coated with a 1 cm thick layer of foam insulation with a thermal conductivity of 0.026 W/m-K, a temperature difference of 50 K maintained for 12 hours between the full tank and the Martian environment will cause the propellant temperature to vary by 2

K. Even a 10% full tank will only result in a temperature change of 20 K for the contents.)

The tanks are made of weldalite aluminum-lithium alloy which has a yield stress of 85,000 psi and a density of 2.7 t/m^3. Using the standard formula for a pressure induced hoop stress (which can be derived in an analogous manner to equation (2.8)):

$$s = P(1.25)r/(xt) \qquad (2.9)$$

where s is the yield stress, P is the tank pressure, r is the tank radius, x=2 for the spherical tank or 1 in the case of the coaxial tank, t is the thickness of the tank wall, and (1.25) is a safety factor, the thickness of the spherical tank wall is found to be 4.41 mm, while the coaxial tank wall is 5.33 mm thick. Calculating the surface areas of these two vessels and multiplying by the density of weldalite and a factor of 1.15 to account for welds and insulation, it is found that that the total mass of the spherical tank is 2753 kg while the coaxial tank has a mass of 1595 kg. Thus it is seen that the use of dense CO_2 monopropellant in a large spherical tank gives the NAV a low tankage mass fraction, which to some degree compensates it for the large inert engine and shielding masses that it must carry.

In order to prevent large shifts of the vehicle center of gravity during flight, the main tank is divided into five sections by three slosh baffles. Two of these baffles divide the major portion of the tank "north-south and east-west" into four major regions, all of which drain into a small disc like region at the bottom of the tank (see Fig. 5). This bottom region, which contains the last 18 t of propellant before the NAV has emptied its tanks, forms the final portion into a liquid disc shield supplementing the solid disc shield discussed in section 2.2.

2.4. Hull Sizing

The overall size of the NAV hull is determined by a number of considerations. First, the maximum diameter of an object liftable by a future heavy lift launch vehicle is assumed to be 12 m (39 ft). Second, the NAV must contain an 8 m diameter spherical tank, and a 2500 MWth engine with a nozzle expansion of 100, as discussed in sections 2.1 and 2.3. Third, there is a requirement for a stable aerodynamic shape. Past analyses[12] have shown that a conic with an L/D of about 0.8 is satisfactory from an aerodynamic viewpoint while also providing excellent packaging efficiency. Finally it is necessary to provide an adequate amount of living and storage space for the crew and their supplies.

The NAV vehicle is designed to function as living quarters and mobile base for a crew of 3 astronauts for a period up to one Earth year. It was thought best, therefore, to provide the crew with two complete habitation decks with a shirt-sleeve environment area which also contains the control area and life support equipment. Each of these decks was allocated 2.2 meters of ship length, resulting in 2.05 m of useful "headroom" within each deck. The upper of these two decks is called the "control deck", since that is where the controls are; while the lower is called the "habitation deck." During the brief moments when the reactor is actually operating, the astronauts position themselves above the crew shield on the control deck. A third deck that can be pressurized was provided. This deck contains the CO_2 compressors and other auxiliary equipment and supplies. This machine deck is shielded from the reactor by the main propellant tank, and during extended stays on the surface or in orbit can be made suitable for shirt-sleeve environment maintenance of the compressors. A fourth pressurizable space is the storage dome above the control deck. The overall vehicle layout is shown in figs. 5 and 6.

Given the conical shape chosen, this design gives the crew 63.7 m^2 (695 ft^2) of floor space, plus an additional 65.04 m^2 (699 ft^2) of supplementary pressurizable area. While by no means luxurious, this is more space per capita than that enjoyed by many apartment dwelling families living in modest financial

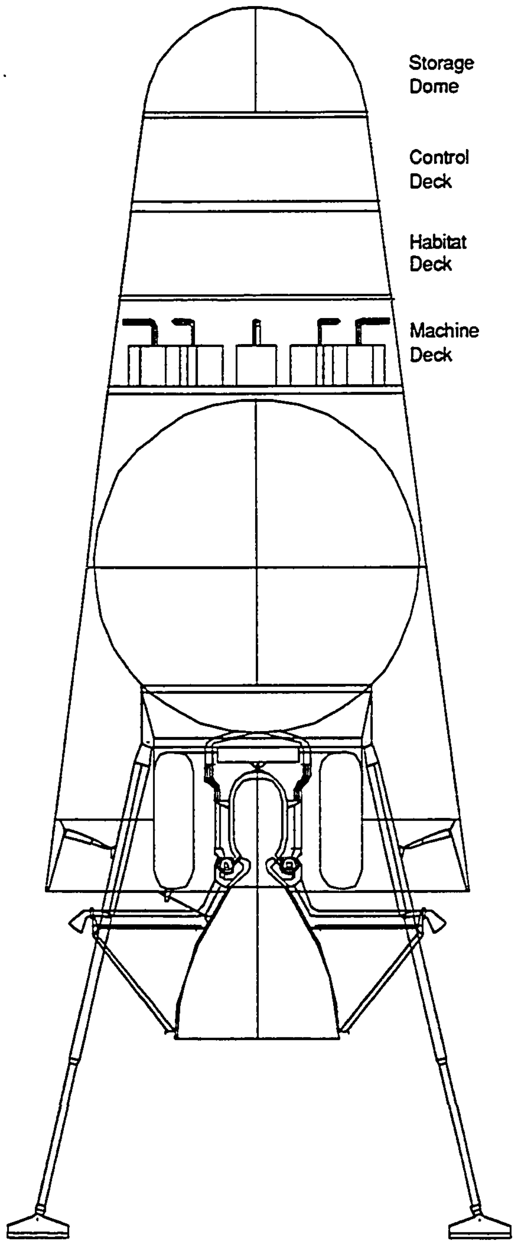


Fig. 5 NAV Overall Vehicle Layout

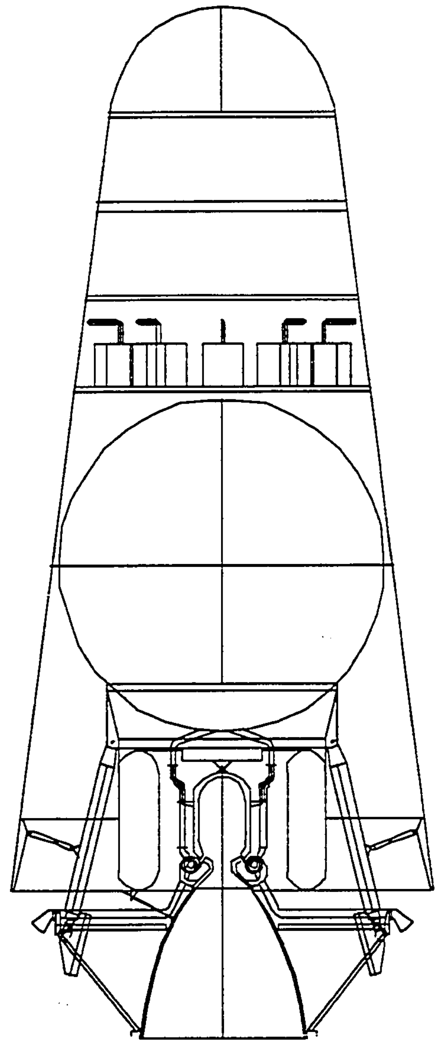


Fig. 6 NAV Overall Vehicle Layout with Legs Withdrawn

circumstances in the United States today, and is far more than that endured by common seamen on merchant ships, exploration, and military vessels throughout human history.

The surface area of this hull is 494 m^2. Assuming that the hull is 2 mm thick and made of titanium alloy with a density of 4.51 t/m^3, the hull itself will have a mass of 4454 kg. Allowing for various structural reinforcements, the four decks, and a layer of "frisi" flexible outer surface insulation (such as that used on the dorsal surface of the Space Shuttle) for thermal protection, it is estimated that the NAV hull and main structure elements can be constructed for a mass of less than 8000 kg.

The design objectives outlined in chapter 1 of this study specified that the NAV need not be designed to aerocapture itself from a trans-Mars injection orbit, but only to de-orbit at least three times from a 250 km circular orbit around Mars, as well as endure 15 suborbital hops through the Martian atmosphere. The heating loads associated with these two modes are mild enough that the frisi coated titanium alloy hull of the NAV is expected to have an unlimited lifetime. An additional layer of insulation is placed inside the hull to prevent an excessive degree of internal heating during atmospheric entry. If it is desired to aerocapture the NAV during its initial passage to Mars, a separate expendable aerobrake is used.

2.5. Control Mechanisms

In order to maintain control during orbital maneuvers, glide, and powered flight, the NAV employs three separate and partially redundant control mechanisms. First, the NAV possesses a reaction control system (RCS) using cold CO_2 gas stored in small tanks at 3000 psi. Two of these tanks are located in the storage dome above the control deck, while two others are located above the reactor shield deck near the bottom of the main propellant tank. The primary purpose of these cold gas RCS jets is to allow vehicle control during zero gravity maneuvers on orbit with the reactor shut down. This allows the NAV to co-operate actively when it is performing a rendezvous with an Earth-Mars interplanetary spacecraft. These cold gas jets, with a thrust of several thousand newtons and a specific impulse of 80 s, may also be of some assistance in fine tuning control during ascent and landing. CO_2 cold gas was selected for this application in spite of its modest performance because the supply is replenishable on Mars, allowing the NAV to continue to function without resupply of RCS propellant from Earth.

The second control mechanism is a set of aerodynamic control surfaces. There is a set of aerodynamic body flaps, four of which are positioned with 90 degree spacing around the circumference of the vehicle hull, hinged at the reactor control deck. These four flaps allow for full maneuverability and control during supersonic flight. In addition, a pair of small flaps is hinged in the fore-and-aft direction to form a set of extendable canards which open along the side of the storage dome near the top of the vehicle. These canards are extended during the final phase of aerodynamic deceleration of the NAV during landing, to allow it to bring its bow around and turn the stern of the vehicle into the wind so as to align its main engine for rocket braking and landing maneuvers.

The third control mechanism is the main nuclear engine, which can be used to control pitch and yaw during powered ascent and terminal descent. The method chosen to allow the engine to accomplish this is simply to gimbal the engine on a ball and socket joint positioned on the base of the reactor disc shadow shield. The engine can then be gimballed in two dimensions by two sets of battery operated actuators which can push or pull on the engine nozzle bell from positions at the base of the coaxial propellant tank. Roll control is accomplished by directing engine turbine bleed through two small aimable nozzles (Fig. 7).

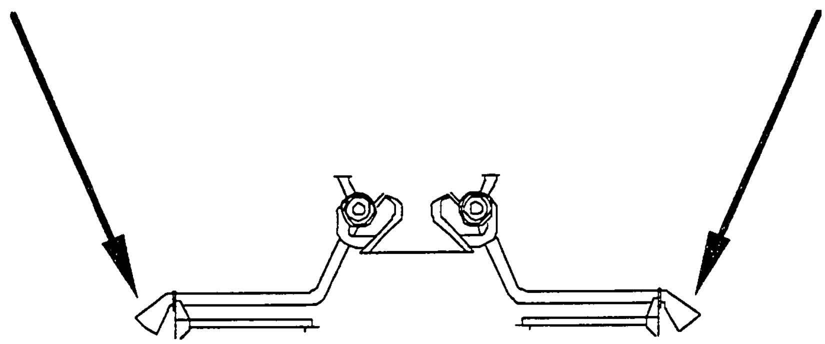


Fig. 7 Roll Control Nozzles (Location shown on Figs. 5 and 6)

An alternative method of thrust vector control is to fix the reactor in place below the disc shield, and simply gimbal the nozzle through the use of a flex seal, such as are sometimes used in solid rockets (Fig. 8). Solid rocket flex seals regularly endure much more extreme conditions than those in the NAV engine, however they are not reusable. A possible method of maintaining a reusable flex seal would be to introduce a flow of cold propellant into the lower nozzle and then release it into the engine chamber directly above the flex seal joint. This would protect the flex seal with a pocket of relatively cool gas at the cost of a very small reduction in engine specific impulse. Thrust vector control could then be exercised by moving the lightweight nozzle and leaving the much heavier reactor in place. This is advantageous, but such a system has never been operated, and since the system of gimballing the entire engine, while more cumbersome, is common practice, it was decided not to use a flex seal nozzle on the NAV.

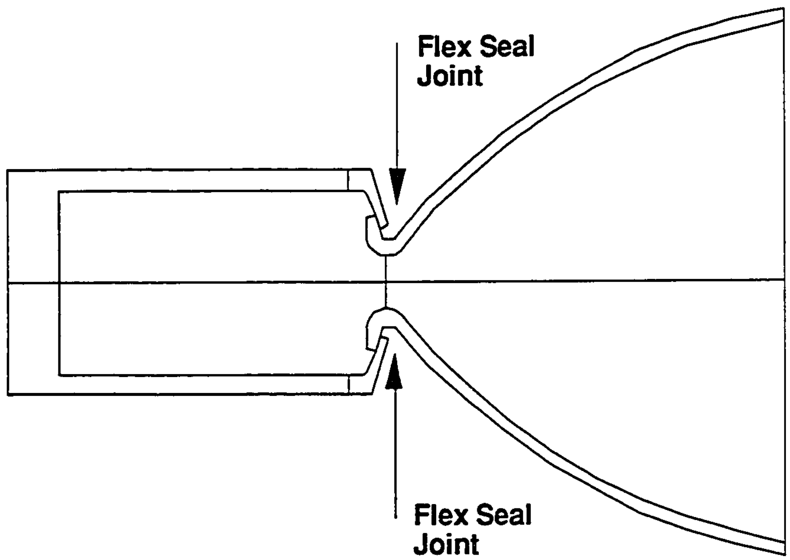


Fig. 8 Flex Seal Gimballed Engine

2.6. Propellant Acquisition System

At a typical Martian tropical daytime temperature of 233 K, CO_2 can be compressed and liquefied out of the Martian atmosphere for an energy cost of 84 kW-hrs per metric ton. This number assumes an ideal isothermal compression process with 100% of electric power provided converted to pump work. The ideal requirement under typical nighttime conditions is about 60 kW-hrs per ton. Assuming an efficiency of 75%, it is therefore found that the average energy requirement is about 100 kW-hrs per ton.

The design requirements developed in chapter 1 specified that the NAV be able to collect enough propellant to ascend to a 250 km circular orbit in no more than 60 days. Since 325 tons of propellant is required for such an ascent, this means that the NAV must be able to liquefy at least 5.5 tons of CO_2 a day, which implies a need for about 25 kWe of power. This amount of power could be provided either by a nuclear reactor, a solar array, or a dynamic isotope power source (DIPS).

The use of a nuclear reactor for propellant acquisition causes some inconvenience, as a critical reactor emits gamma rays which would preclude operating outside of the habitation decks. If a reactor is chosen, it would probably be advisable to utilize it at a power level of 100 kWe or more, so as to conclude the propellant acquisition phase after a span of 14 days or less. The reactor would then be shut down for the remainder of the stay at the given site. Since the crew could not stay directly above the crew shield (on the control deck) during this entire span, the amount of groundshine generated by the reactor (especially prior to the filling of the coaxial tank) would have to be calculated and an appropriate amount of side shielding incorporated into the design to reduce the groundshine dose to allowable levels. The reactor to provide the 100 kWe could be a "dual mode" reactor,[13,14,15,16] i.e. the NAV's main propulsion engine with separate cooling channels rigged in to allow it to be run closed cycle; or it could be a separate small dedicated power reactor. The latter alternative would certainly be simpler from an engineering standpoint, although probably somewhat heavier. The power reactor would be prevented from exchanging neutrons with the engine reactor by a thick layer of boron carbide, and could conceivably be placed inside the engine reactor's solid disc shield. This would provide it with massive peripheral shielding to reduce its groundshine dose, while allowing it to do double duty by using it as a shield against the engine reactor (which operates at a much higher power level). The two reactors would not operate simultaneously, so that neutronic coupling would not be a problem. A 100 kWe reactor power system using a closed Brayton cycle could probably be built for 3000 kg, with the mass of the turbines, radiators, power conditioning, and some extra shielding included. The radiator, which is a system of titanium heat pipes built into the lowest 4 meters of the hull skirt (which is not covered with frisi), and which comprises 72 square meters in area, can dispose of the 500 kW of thermal power that this 100 kWe system generates by radiating at a temperature of 600 K, assuming a 50% view factor (i.e. taking no credit for heat removal through radiation from the inward surface and out through the bottom of the skirt, and no credit for natural convection.)

An alternative is a solar array. An average power of 25 kWe would be needed, which requires a peak daytime power of about

75 kWe. Given that the average solar insolation at the surface of Mars[17] is about 180 W/m^2, and assuming a 16% conversion efficiency, this implies a solar array of about 2600 square meters area. Such an array would involve about 650 lightweight panels, each measuring 2 m by 2 m and with a mass of about 10 kg each. If a man can set up 20 of these in an hour, then a 3 man crew working 5.5 hour days could set up the entire array in 2 days. A certain amount of time may have to be spent each morning brushing off any accumulated dust or frost, and the entire array would have a mass of about 6.5 tons. All in all, the solar alternative appears feasible as a fallback option, but is not especially attractive.

A third alternative power source for the propellant acquisition system would be a dynamic isotope power source (DIPS). DIPS utilizing Pu 238 heat sources and closed Brayton cycle conversion systems are currently being designed for the U.S. Department of Energy by Rocketdyne[18] and are expected to be operational by 1999. Each DIPS unit will generate 6 kWe and have a mass of about 800 kg. Given the slow decline in power generating ability that occurs with time in an isotope power source, 5 such DIPS units will be needed to assure adequate amounts of power during the useful lifetime of the NAV. The DIPS system can thus be expected to have a mass of about 4000 kg. The radiator, as in the case of the reactor driven system described above, is a system of titanium heat pipes built into the lowermost 4 meters of the hull skirt, covering 72 square meters. With a 50% view factor, this radiator can dispose of all the thermal energy generated by the DIPS by radiating with a surface temperature of 410 K, which is well below the operating limits of titanium.

The amount of Pu 238 required to generate 25 kWe would be quite expensive[19], approximately $200 million at current prices. This may not be prohibitive, however, given the overall cost of a manned Mars mission, which may be expected to be at least an order of magnitude greater. An alternative would be to use Sr 90, which costs about 1/10 as much as Pu 238, as a heat source for the DIPS. While Pu 238 has been the isotope in general use in recent years, Sr 90, in the chemically inert form of $SrTiO_4$, was used in radioisotope power sources in the 1960s[20]. The problem with strontium radioisotope thermoelectric generators (RTGs) and DIPS is that Sr 90's daughter product produces a prompt gamma, which creates a shielding problem for most missions. The NAV, however, already carries substantial shielding in any case. The strontium heat source could be placed inside the engine disc shield, and be shielded to an adequate degree without any additional mass penalty. Thus either a Pu 238 or Sr 90 DIPS may be considered viable, with the Sr 90 system offering some cost reduction benefit.

The best choice then for the NAV propellant acquisition system power supply appears to lie between a DIPS and a small reactor, either of which would be located within the engine disc shield. The reactor allows for more rapid fueling, but prevents manned operations in the vicinity of the vehicle while propellant acquisition is underway and may create a requirement for additional shielding. Since the DIPS will accomplish the mission without raising such problems, it was decided to use a set of five 6 kWe DIPS to drive the NAV propellant acquisition system.

With 25 kWe of pumping power available, the NAV pumping system needs to be sized to liquefy 250 kg of CO_2 an hour. If the velocity with which atmosphere can be driven into the intakes by fans is 20 m/s, then at an average Martian surface atmospheric density of 0.014 kg/m^3, 0.25 m^2 of intake area will be required. For the sake of redundancy, the propellant acquisition system is designed to comprise five pumps, each with a power rating of up to 6 kWe (about 7.5 horsepower) and an intake area of 0.06 m^2, and each driven by its own electric motor. Thus the design rate of propellant acquisition can be accomplished if any one of the pumps fails to function. Each of the five intakes has a radius of 14 cm (5.5 in). The pumps themselves are three-stage centrifugal compressors, with each stage increasing the pressure of the gas by a factor of 10. Fans are used to provide forced convective cooling with Martian atmosphere to maintain the compression process close to ambient temperature conditions. Two additional small high-pressure pumps are also available for use in providing restoration propellant to the cold gas RCS system. The total mass of the pumps, motors, and compressor system is estimated at 200 kg.

2.6. Consumables

The NAV life support system recycles air and water using physical/chemical means. Since no water cycle is 100% closed, makeup is provided by using whole food (instead of dehydrated). The mission requirement is to maintain a crew of 3 for one year. If 2 kg of whole food is provided per person per day, this results in a consumable requirement of 2200 kg. For added margin, 2500 kg of food is carried.

The atmosphere of Mars is composed of 95% CO_2, 2.7% nitrogen, 1.6% argon, 0.3% water, 0.13% oxygen, and 0.07% carbon monoxide[21]. (Table 2.) Thus is may be noted that in the course of producing 300 t of CO_2 propellant from the Martian atmosphere, the NAV will also produce 5.4 t of nitrogen, 4.6 t of argon, 390 kg of water, and 300 kg of oxygen. If desired, these can be collected and stored. The nitrogen and argon can be used as makeup for any buffer gas lost during the numerous airlock operations, and the oxygen and water can be used directly to supplement the recycling capability of the life support system. If the nitrogen is not needed for such purposes, it could be stored under high pressure for use as a cold gas RCS system with somewhat better performance than that using CO_2.

Table 2. Composition of the Martian Atmosphere (mole percent)

Carbon Dioxide	95.0
Nitrogen	2.7
Argon	1.6
Water,	0.3
Oxygen	0.13
Carbon Monoxide.	0.07

2.7. Landing Gear

The landing gear of the NAV consists of 6 pneumatic telescoping legs positioned symmetrically around the vehicle. The legs, which are divided into three 3.5 m tube segments are hinged on the circular ring support on the reactor shield deck. This ring support is connected by a circular truss to a second ring support encircling the main propellant tank along its 45 degree parallel of south latitude (Fig.5). The loads from both the landing gear and the engine are thus transmitted through this structure to the main propellant tank and thence to the upper hull of the vehicle. The landing gear tubes are made of weldalite aluminum alloy and are filled with nitrogen gas pressurized at 4 MPa (580 psi). (Weldalite, a modern aluminum-lithium alloy with a density equal to standard high strength aluminum (2.7 gm/cm^3), but with a yield strength more than twice as great (700 MPa verses 300 MPa for standard aluminum alloys) is advantageous for this and other applications that require high strength but not high temperature.) The bottom tube segment is the narrowest, and has an inner radius of 17 cm. This allows the weight of the vehicle fully fueled to be supported by four of the six legs. At the moment of touchdown during landing, the six legs will decelerate a 50 t NAV at an initial rate of 45 m/s^2 (about 4.5 Earth g's). As the legs contract, the gas pressure reacting the vehicle will increase, which causes the rate of deceleration to increase to 68 m/s^2 when the leg has contracted to 2/3 of its full length. This allows the NAV landing gear to accommodate a NAV falling with an initial contact velocity of 18 m/s. This is the velocity reached during an unpowered drop from an altitude of 43 meters. One-way compression valves are positioned within the legs to prevent bounce. The NAV engine power can thus be cut safely at an altitude of 30 m above the ground, greatly reducing groundshine shielding problems during landing. Both groundshine and landing leg shock can reduced even further by using cold CO_2 to accomplish terminal deceleration. In fact, 3 tons of cold CO_2 gas

(Isp = 80 s) applied as main thrust during terminal descent can soft land the NAV from a main engine cutoff altitude of 200 m.

The landing gear tubes are 0.4 cm thick, which gives them an operating safety factor of 2.3 times their yield stress. This thickness results in a mass of 730 kg for the landing legs themselves, and a mass of 1000 kg is allocated for the landing gear system.

2.8. Mass Breakdown

The total mass breakdown for the NAV is given in Table 3. With the addition of 1000 kg for a ground rover, 1000 kg in habitation furnishings and life support equipment, and 240 kg for the crew, the total vehicle mass is 44.7 tons. There is thus a contingency mass of 5.3 tons, or 12%, before the NAV exceeds the initial design assumption of 50 tons unfueled mass.

Table 3. Nuclear Ascent Vehicle Mass Breakdown

Component	Mass (kg)	Distance From Bow (m)
Reactor	4190	19.46
Disc Shield	6208	18.05
Side Shield	2371	19.46
Engine Auxiliary	1700	20.54
Nozzle	960	23.61
Main Tank	2753	13.45
Coaxial Tank	1595	19.46
Hull	8000	14.18
Landing Gear	1000	19.46
Pumps	200	8.11
Radiators	200	19.11
RCS	1000	10.27
Ground Rover	1000	6.48
Habitation	1000	4.86
Crew Shield	6000	4.64
Crew	240	3.98
Consumables	2500	5.95
Main Propellant	0	13.45
Coaxial Propellant	0	19.46
DIPS	3800	18.50
Total	44717	14.30

Chapter 3. NAV Engine Design

In chapter 2, the conceptual design of a manned NAV vehicle with a dry mass of 44.7 tons was presented. The engine of the vehicle had a required thrust of 2080 kN, which at an assumed propellant temperature of 2400 K, translated into a power level of 2513 MWth. Reactor masses were estimated using scaling relationships based upon updated NERVA derivative nuclear thermal rocket concepts[4], while non-nuclear engine component masses were scaled using standard algorithms applied in the Advanced Launch System program[22]. Together these scaling relationships produced an estimated engine mass of 6850 kg (without shield) for a thrust to weight (T/W) ratio of 31.0 (The use of high molecular weight of CO_2, while resulting in a nearly fourfold decrease in specific impulse relative to hydrogen, increases thrust in the same proportion for a given power level.) Since the engine required 8580 kg of shielding, a total engine/shield system mass of 15430 kg and a T/W of 13.76 are obtained.

In this chapter the estimates of NAV capabilities made in chapter 2 are refined by abandoning scaling relationships based upon hydrogen NTR reactor power/weight ratios. Instead the results are presented of an assessment of the potential performance of an NTR engine using CO_2 propellant based directly upon computational thermal-hydraulic studies of CO_2 driven NTR engines themselves.

3.1 Chemical Compatibility and Propellant Temperature

When CO_2 is heated to temperatures in excess of 2000 K, it partially dissociates into CO and O_2, and the resulting mixture behaves as an oxidizer. This fact makes it unlikely that CO_2 could be used in a reactor employing the same carbide coated-graphite based fuel elements that were used in the NERVA program, although SiC is chemically compatible with CO_2 at temperatures close to 2000 K. To resist an oxidizing medium at high temperatures, oxide coated fuel elements will have to be used. Table 4 shows data on various materials taken from References 11 and 23. The reaction temperatures given in in the center column of Table 4 are rough lower limits below which CO_2 does not react with the solid materials cited on time scales of interest to the NAV engine application.

Table 4. High Temperature Properties of CO_2 Compatible Materials

Material	Reacts with CO_2	Approximate Melting Point
ZrO_2	> 2700 K	3000 K
ThO_2	> 2900 K	3500 K
MgO	> 2400 K	3100 K
BeO	> 2400 K	2800 K
HfC	> 2000 K	4150 K
SiC	> 2000 K	3000 K
ZrC	> 1400 K	3800 K

It can be seen that both ThO_2 and ZrO_2 have melting points in excess of 3000 K and are capable of resisting corrosion by CO_2 at temperatures in excess of 2600 K. Examining these data, it can be seen that there will be a large design margin on chemical compatibility in the present study if propellant temperatures are limited to 2400 K, and fuel element surface temperatures to 2600 K or less. This design choice is made, and the NAV's fuel elements are selected to be spheres with a kernel of UO_2/ThO_2 mixture, surrounded by a coating of ThO_2, since, as shown in Table 4., this material offers the best high temperature properties.

3.2 Thermal Hydraulics Code

For the present study, a thermal hydraulics code was written to analyze the performance of CO_2 cooled reactors. Since it is clear that the thrust to weight (T/W) ratio of the system is critical, which in turn will depend upon the attainable reactor power density, the code was written to allow for analysis of such high power density options as the pellet bed[6] and the particle bed[5] nuclear thermal rocket configurations. In these configurations, a cold propellant enters a pellet or particle bed from the outermost of two coaxial cylindrical "frits" and then moves from this cold frit through the bed, exiting at the inner coaxial hot frit. The propellant is then piped to a chamber whence it is expanded through a throat and nozzle to produce thrust. Both the particle bed and pellet bed systems work on this principle, the difference being that the pellet bed uses a single fuel bed and fuel spheres that have a diameter on the order of 0.5 to 1 cm, while the particle bed reactor employs multiple fuel beds and utilizes fuel spheres with a diameter on the order of 0.5 to 1 mm.

The calculation program, called BEDCODE, works as follows. First, a reactor configuration is proposed, and parameters are defined. These include power level, inlet and outlet propellant temperature, input propellant pressure, pellet size, and bed inner and outer diameter. Given the predetermined rise in propellant temperature in the core and the power level, the propellant mass flow rate is also defined. The bed is divided into n concentric segments, each defining a ring of material which produces a certain predetermined amount of fission power which must be removed by the propellant. At the outermost position, the initial propellant temperature and pressure, and thus density, flow speed, and Prandtl and Reynolds numbers are known. The heat transfer coefficient in that region can then be computed from a relation broadly accepted within the particle bed and pellet bed reactor communities[24]:

$$\varepsilon \frac{Nu}{Pr^{0.33}} = 2.876 + 0.3023 Re^{0.65} \tag{3.1}$$

where ε is the bed void fraction, Nu is the Nusselt number, Pr is the Prandtl number, and Re is the Reynolds number whose characteristic length is the particle diameter.

The pressure drop in that segment can also be calculated using the von Karman correlation for packed particle beds[25]:

$$\Delta P = 4K_{ef}q\frac{L}{d} \tag{3.2}$$

where ΔP is the pressure drop across the bed segment in question, q is the dynamic pressure of the fluid in that section, L is the path length through the section, and d is the pellet diameter.

The factor K_{ef} in equation (3.2) is given by:

$$K_{ef} = \frac{f(1-\varepsilon^3)}{\varepsilon^3} \tag{3.3}$$

where

$$\begin{aligned} f &= 75/Re && \text{for } Re < 10 \\ f &= 0.875 + 75/Re && \text{for } 10 < Re < 1000 \\ f &= 0.875 && \text{for } Re > 1000 \end{aligned} \tag{3.4}$$

Since the power in the first segment and the total surface area, A, of the particles in the bed are known, the ΔT between the surface temperature of the fuel and the propellant gas can be calculated in accordance with an energy conservation statement.

$$\text{Power} = hA(\Delta T) = \frac{k(Nu)}{D} A(\Delta T) \tag{3.5}$$

Here h is the heat transfer coefficient and k is the fluid thermal conductivity.

Thus having obtained ΔT, the fuel temperature in the first segment of the bed is known. Since the power and the mass flow are also known, the temperature of the fluid as it exits the first segment can also be calculated. Equation (3.2) then gives the pressure drop of the fluid as it passes through segment 1, and so all fluid state variables are known or can be computed. The procedure outlined above is then repeated and the fluid marched through successive positions in the bed, to generate profiles of fluid temperature, pressure, and fuel element temperature at every position throughout the bed, as well as the bed total pressure drop.

3.3 Results of Exploratory Thermal Hydraulic Analysis

A series of exploratory studies of pellet and particle bed CO_2 NTR configurations were done using BEDCODE. The pellet beds systems (Fig.9) employ a single large bed generating all 2500 MW of the engine power. This is possible in the pellet bed because the large pellet size causes a smaller bed pressure drop per unit length and thus allows a total bed flow path to be longer. The particle beds (Fig.10), whose smaller pellet diameter allows for a much larger power density in the beds themselves, consist of a series of beds interspersed among a reactor moderating material. Each of these beds produce a fraction of the total power. The subdivision of the particle beds is necessary in order to avoid an excessive pressure

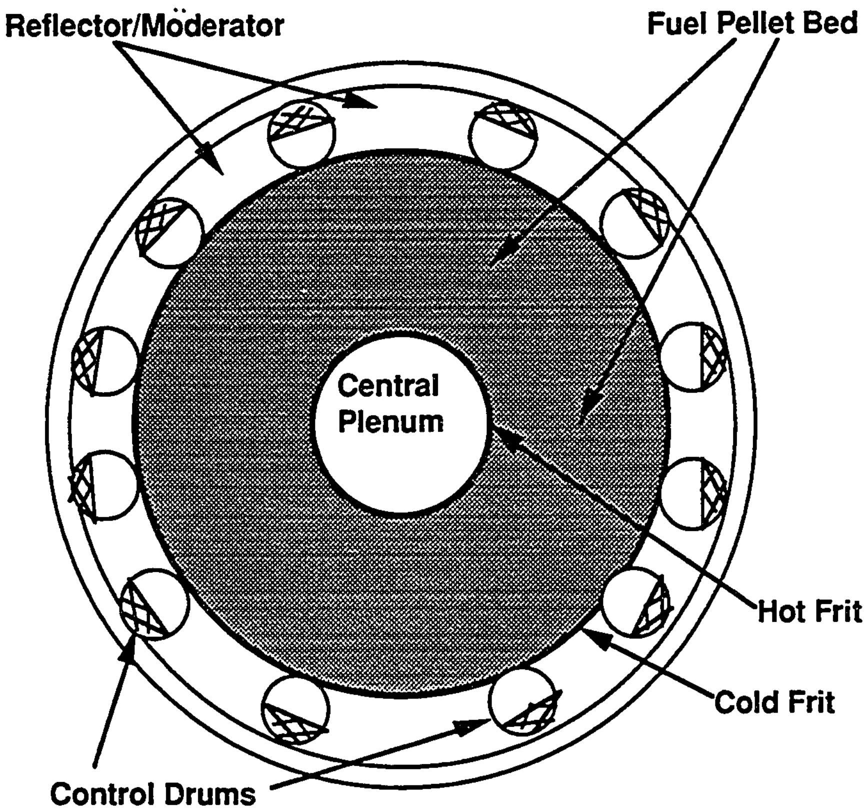


Fig. 9 Pellet Bed Reactor

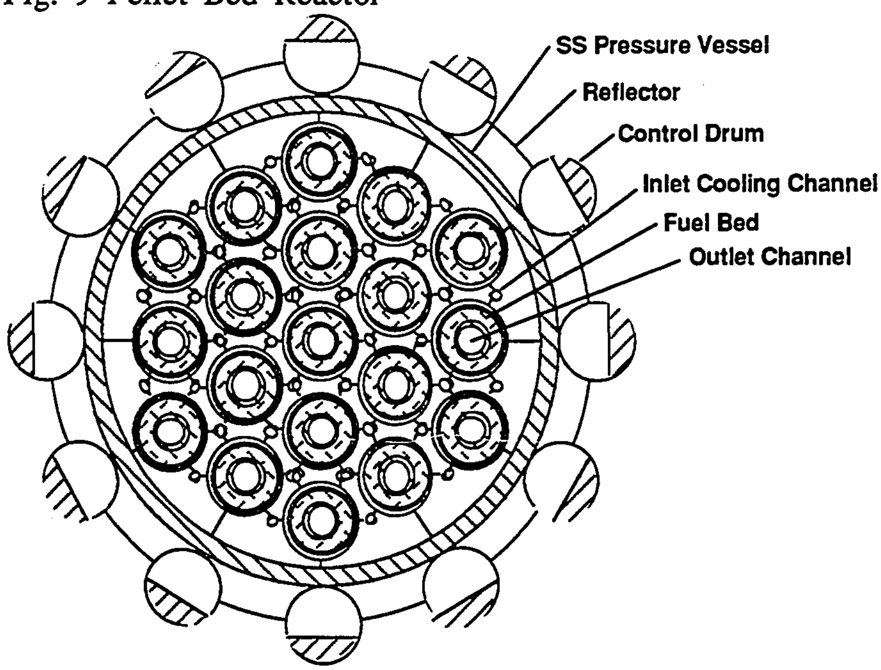


Fig. 10 Particle Bed Reactor

drop in the core. All reactors examined had assumed void fractions of 0.35, input gas pressure of 80 atmospheres, inlet gas temperature of 300 K and outlet temperature of 2400 K. All systems had a total power of 2500 MW and a core length of 1.5 m. The results of these studies are given in Table 5.

In Table 5 all dimensions are in meters, the power of each bed is 2500 MW divided by the number of beds, the pressure at outlet is the pressure of the gas as it exits the bed in atmospheres, and temperature of wall outlet is the temperature of the fuel at the hottest (exit) portion of the bed in degrees K.

Cases 1 through 10 are pellet beds (single beds/large pellets) and in examining them, the difficulty that the pellet bed concept has in providing a favorable result for NAV application can be seen. Case 1 fails because of excessive exit fuel temperature, attempting to rectify this in case 2 by reducing the pellet size (as pellet size decreases, total fuel surface area within the bed increases linearly) results in a pressure drop that is greater than the total input pressure. This indicates that the required mass flow rates are not possible in the case 2 bed under the given conditions. In case 3 the pressure drop is corrected by shortening the path length across the bed, but the thinner bed now has a much greater power density than before and so even with the smaller pellet size, the exit fuel temperature is too great. The combined problem

Table 5. Particle and Pellet Bed Exploratory Studies

Reactor radius	Number of beds	Inner bed radius	Outer bed radius	Pellet radius	Pressure at outlet	Temp. of wall outlet
1. 0.6	1	0.3	0.6	0.006	24.6	3213
2. 0.6	1	0.3	0.6	0.005	< 0	----
3. 0.6	1	0.4	0.6	0.005	55.6	3423
4. 0.6	1	0.4	0.6	0.004	47.7	3147
5. 0.6	1	0.4	0.6	0.003	30.9	2890
6. 0.6	1	0.5	0.6	0.002	55.4	2976
7. 0.6	1	0.5	0.6	0.001	< 0	----
8. 0.8	1	0.7	0.8	0.001	48.2	2577
9. 1.0	1	0.9	1.0	0.001	61.2	2559
10. 1.0	1	0.9	1.0	0.0006	36.3	2461
11. 0.6	7	0.1	0.16	0.001	65.8	2656
12. 0.6	7	0.1	0.16	0.0006	49.4	2516
13. 0.6	7	0.1	0.16	0.0005	36.8	2482
14. 0.6	7	0.11	0.16	0.0004	35.1	2465
15. 0.6	19	0.05	0.1	0.0004	55.6	2458
16. 0.5	19	0.04	0.08	0.0003	28.4	2443
17. 0.5	19	0.04	0.08	0.0004	52.3	2480

of meeting the demand for a limited exit fuel temperature and a moderate pressure drop can only be met by enlarging the bed, and acceptable results are not found until the core radius is 1.0 meters (e.g. case 10).

In cases 11 through 17 the strength of the particle bed concept is illustrated. In cases 11 through 14, the core has been divided into 7 beds (1 central surrounded by 6 others in a hexagonal pattern), each producing 360 MW. Since a number of beds are employed, the flow path length can be shortened without decreasing the total reactor volume, and so the pellet size can be reduced, allowing substantial increases in overall power density. Thus acceptable cases (for example cases 13 and 14) are seen with core radii of 0.6 meters. If this principle of core subdivision is extended to 19 beds (1 central, 6 surrounding in an inner ring, and 12 more surrounding those in an outer ring) each producing 136 MW, an acceptable configuration (case 16 or 17) can be found with a core radius of 0.5 meters. Since the power density of the system goes as the inverse square of the core radius, the 19 bed small particle system is capable of a power density about 4 times as great as the single bed large pellet option. The design decision in favor of the multiple particle bed over the single pellet bed option is clear.

In addition to offering 4 times the core power density of the single pellet bed, the multiple particle bed offers the additional advantage of allowing the placement of hydrogenous moderating material, such as LiH or ZrH_2 within the core. These materials cannot be used above 950 or 800 K, respectively. However since the moderator in the multiple particle bed core only sees the cold frit side of the hot fissioning beds, it can be kept within acceptable temperature limits despite the fact that the propellant gas is exiting the system at 2400 K. Because of its low atomic mass, hydrogen is the most effective element for moderating fast neutrons created in fission reactions down to thermal energies where the neutron's probability of inducing a uranium fission is high. By using such hydrogenous moderators, a very thermal reactor system (one whose neutron energy spectrum approximates a Maxwellian distribution for the reactor's temperature) can be obtained, which is highly advantageous for maximizing the efficiency of neutron utilization in the reactor.

3.4 Neutronic Analysis

In order to complete the thermal hydraulic analysis of the reactor a power distribution must be obtained. Since, for reasons given above, the use of a particle bed system for the NAV reactor has been chosen, the attractive option of placing hydrogenous moderator in the core is also available. Both ZrH_2 and LiH have hydrogen densities comparable to that of liquid water, and so a very thermal neutron spectrum is obtained. In such a reactor, one-group neutron diffusion theory using thermal cross sections is a valid form of analysis, and so a code utilizing one-group diffusion theory was written to analyze the neutron flux distribution (and thus power distribution) in the NAV reactor.

Bare cores with homogeneous materials have well known analytic solutions are available defining both system criticality and flux distribution. The flux distribution so obtained is:

$$\Phi = \cos\left(\frac{\pi z}{h}\right) J_0\left(\frac{v_0 r}{r_0}\right) \tag{3.6}$$

where Φ is the neutron flux, and r_0 is the reactor radius and h is the reactor height. Examining equation (3.6), it can be seen that this solution is strongly peaked near the central regions of the reactor. It is therefore desirable to use both a neutron reflector around the core and a non-uniform uranium enrichment within the core in order to realize a more uniform core power density. For such configurations analytic solutions are not available and numerical methods are necessary.

A code, FISSION, was written to solve the one-group neutron diffusion equation in r-z cylindrical co-ordinates, assuming azimuthal symmetry. Since exact criticality (i.e. steady state operation), cannot be assumed for a given material composition and geometry, an iterative technique must be used to solve simultaneously for the time-independant neutron flux spatial form and for the reactor neutron generation time multiplication factor k_{eff}. The procedure is to assume an initial neutron flux distribution Φ_1 and then to solve the neutron diffusion equation for Φ_2 keeping Φ_1 fixed. The neutron diffusion equation then becomes:

$$-\nabla \cdot D\nabla\Phi_2 + \Sigma_a\Phi_2 = \nu\Sigma_f\Phi_1 \qquad (3.7)$$

In equation (3.7), ν is the average number of neutrons born in each fission event, D is the neutron diffusion coefficient, Σ_a is the macroscopic absorption cross section, and Σ_f is the macroscopic fission cross section of the reactor material. Since equation (3.7) is an elliptic partial differential equation, the method of successive over relaxation (SOR) can be used and will converge to a solution (for a given Φ_1). Values at a certain point in the reactor, say r=0, z=0 can then be used to renormalize Φ_2. Φ_1 was equal to 1.0 at this point; if Φ_2 is divided everywhere by Φ_2 at this point to renormalize, the renormalization constraint $\Phi_2(0,0)$ will be a first estimate for k_{eff}. Φ_2 is now substituted for Φ_1 on the right hand side of equation (3.7) and the procedure is repeated until both the flux shape and the estimate for k_{eff} converge on fixed values. According to Duderstadt and Hamilton[26], this procedure will always converge.

The FISSION code was written and verified by testing it on a bare homogeneous core for which analytic solutions for both flux distribution and k_{eff} could be obtained. After 900 outer (k_{eff} determining) iterations (with 5 SOR inner iterations for each outer iteration), it was found that the flux distribution given by FISSION matched the analytic solution to within 1%, while FISSION's estimate of k_{eff} was equalled the analytic result within 0.01%.

A reactor model was then set up for analysis using FISSION. Fuel particles were assumed to have a kernel consisting of a 50/50 mixture of UO_2 and THO_2 out to half the radius, and this in turn was surrounded by a layer of pure ThO_2 out to the particle's spherical surface. Enrichment was varied according to location (lowest at the center of the core). ThO_2 was assumed to be the primary material of the hot frit, as it is very refractory and strong in compression. LiH was chosen for the moderator, as it has less than 1/5th the density of the alternative ZrH_2. Since Li^6 has a thermal absorption cross section of about 1000 barns, while Li^7 only has 0.033 barns absorption cross section (for an average in natural Li of 71 barns) it is important to use Li with its Li^6 isotope largely removed. The variability of the isotopic enrichment of the U^{235} and the Li^7 are the two primary controls for creating a core which is only slightly supercritical at the geometric size which is required for adequate heat removal (i.e. variable enrichment will allow the setting of the geometric buckling of the reactor equal to its material buckling.[26]) In addition to LiH, the moderator region was assumed to include a homogenous mixture of 5% ThO_2 (by volume) and 15% Fe (i.e. steel.)

The composition of the core used for neutronics analysis is given in Table 6. The isotopic composition of U^{235} and Li7 varied in

Table 6. Reactor Composition Used in Design Neutronics Model

Fuel Assembly	Moderator	Reflector
2.188% UO_2	80% LiH	80% LiH
32.813% ThO_2	15% Fe	10% Fe
65.000% Void	5% ThO_2	10% Ti

Core Composition:	40% Fuel assembly
	60% Moderator
Core Dimensions:	69 cm Radius
	138 cm Height
Reflector Thickness	3 cm
U^{235} enrichment	40% at Core Center
	90% at Core Surface
Li^7 enrichment	99.84% at Core Center
	99.39 % at Core Surface
	99.84% in Reflector

linear fashion with distance from the core center (r=0, z=0) to the extreme limit of the core (r=69 cm, z=69 cm). The enrichment of Li^7 proposed is high, but it should be noted that the composition of natural lithium is 93% Li^7.

The results of the neutronics analysis code are shown in Figs. 11 and 12. In Fig. 11 the neutron flux distribution in the core and reflector is shown, and the usual strongly peaked flux function can be observed. In Fig. 12 the power distribution within the core is shown, and it can be seen how the increasing enrichment of fissile U^{235}, as we move outward in the core, has served to partially reduce the peaking effect. Thus, for example, while the flux at position r=24, z=24 is about 50% that of the flux at the core center (r=0, z=0), the power at r=24, z=24 is about 75% as great as that at r=0, z=0. (Because the neutron diffusion equation is invariant with respect to flux magnitude, the flux and power values in Figs. 11 and 12 are given in arbitrary units.)

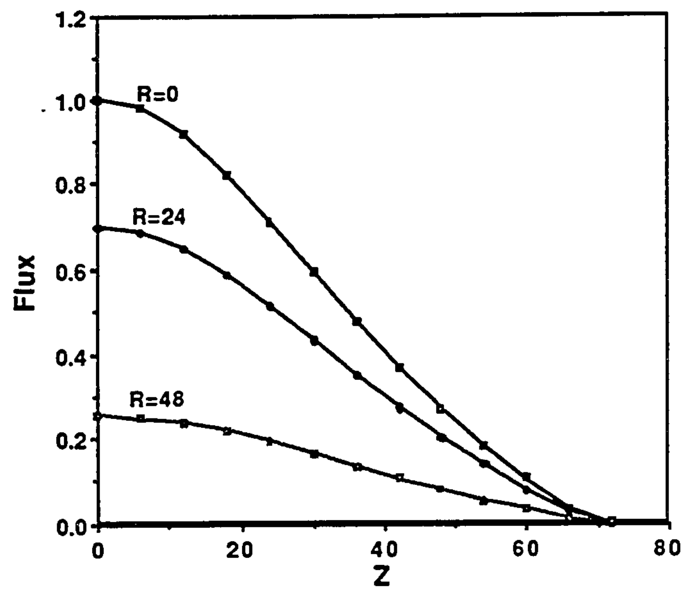


Fig. 11 Distribution of Reactor Neutron Flux (R, Z in cm)

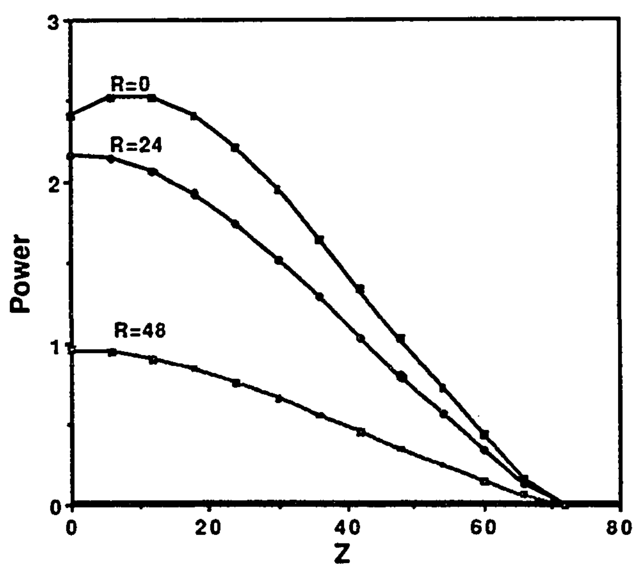


Fig. 12 Reactor Power Distribution (R, Z in cm)

3.5 Reactor Design Point

As a result of the exploratory thermal hydraulics analysis described above, it was decided to base the design of the NAV reactor upon the particle bed concept. A system with a central fuel assembly, surrounded by an inner ring of 6 fuel assemblies and an outer ring of 12 more fuel assemblies (Fig.13) was chosen. The fuel assemblies each have a hot frit radius of 5 cm and a cold frit radius of 10 cm. The assemblies of the inner ring have their centers positioned on the reactor core's 24 cm radius circle, while the outer ring fuel assemblies have their centers located on the core 48 cm radius circle. As can be seen from Fig. 12, each of the fuel assemblies on the inner ring will produce a total power about 90% that of the central fuel assembly, while each assembly on the outer ring will have a power about 40% of that of the central assembly. For the total power requirement of 2500 MW, the central assembly will have a power of 223 MW, the inner ring assemblies will produce 201 MW each, while each of the outer assemblies will produce 89 MW.

Examining Fig.12 further, it can be seen that the peak power density in the central fuel assembly is about 60% greater than the average power density within the central fuel assembly. The entire reactor configuration must be designed to tolerate the power density in this, the hottest region within the reactor. The

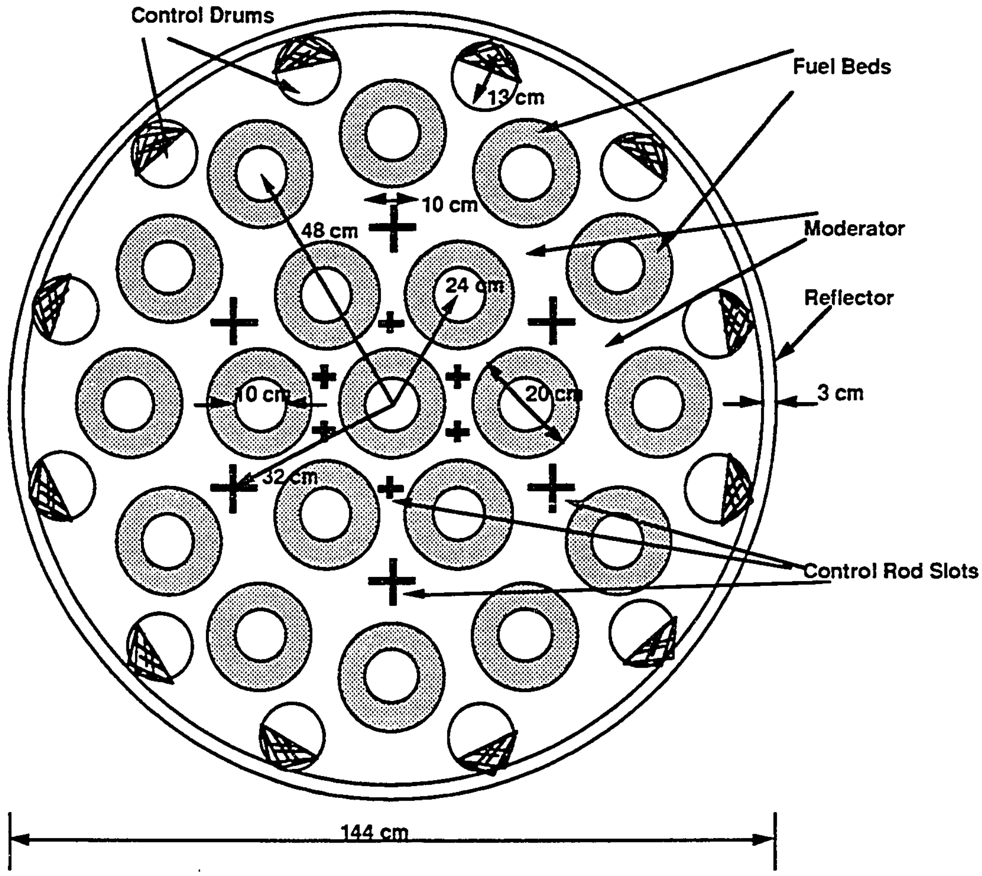


Fig.13. NAV Reactor Design Point Geometry

power density here is considerably higher than the average densities considered in Table 5, and it forces the selection of input pressures considerably higher than the 80 atmospheres examined earlier. (The power density at the center is equivalent to 250 MW per meter of length, compared to 88 MW per meter in the cases numbered 15 through 17 in Table 5.)

In Table 7. we see the results of thermal hydraulic analysis of fuel assemblies operating at the reactor's peak power density.

Table 7. Thermal Hydraulic Analysis of Central Fuel Assembly Operating at Peak Power Density

	Input pressure	Pellet radius	Temp. of wall outlet	Outlet pressure
1.	80	0.001	2771	29.9
2.	80	0.0009	----	<0
3.	100	0.001	2781	66.2
4.	100	0.0009	2723	60.6
5.	100	0.0008	2672	52.6
6.	100	0.0007	2619	39.9
7.	100	0.0006	----	<0
8.	120	0.0008	2677	84.3
9.	120	0.0007	2627	76.5
10.	120	0.0006	2579	64.5
11	120	0.0005	2531	41.5

On the basis of the above results, it was decided to design the reactor using pellets of 0.0006 m radius with a peak inlet pressure of 120 atmospheres (case 10, above). The resulting chamber pressure is about 64 atmospheres (940 psi), which is higher than that used in the NERVA program (630 psi on the

Phoebus engine), but lower than that used on many modern engines such as the RL-10 (1100 psi)[27] or Space Shuttle Main Engine (3000 psi)[28]. The turbopump power required to feed propellant to the engine at this pressure is calculated to be 8.6 MW.

The rise in temperature and the drop in pressure as the gas moves from the outer cold frit to the hot inner frit of the peak power region of the central fuel assembly is shown in Fig. 14.

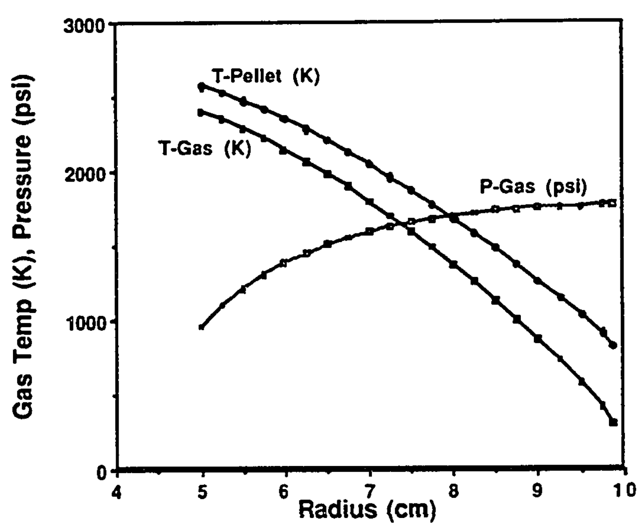


Fig.14 Variation of Gas Temperature and Pressure in Reactor Peak Power Region

3.6 NAV Engine Startup and Control

A nuclear reactor functions by having its population of neutrons reproduce themselves through fission. Each fission event produces on the average about 2.4 neutrons, however many neutrons do not produce a fission event. Instead they may be absorbed into the nuclei of various reactor materials or leak from the system. The average number of neutrons produced in the next generation by each neutron in the present reactor population is known as the effective neutron multiplication factor, k_{eff}. In order for a reactor to maintain a constant level of power, k_{eff} (henceforth simply "k") must equal 1.0. If k is greater than 1.0, the reactor is said to be supercritical, and the power level will increase. If k is less than 1.0, the reactor is subcritical and the power level will decrease in time.

In a thermal reactor, such as that in the NAV engine, each generation of neutrons has a lifetime of about 0.0001 seconds. It would thus appear at first glance that a reactor would be extremely difficult to control, since if all neutrons had such a short lifetime, a reactor with k=1.001 say, would have its neutron population increase by a factor of k^{10000} (= 21900) in a single second. In reality, however, a certain fraction of all neutrons produced in a given generation are not given off as the immediate result of fission events, but rather are yielded by fission products after a significant delay, ranging from about 0.3 to 80 seconds. In a thermal reactor such as that on the NAV, the fraction of neutrons belonging to this delayed release category, known as β, is equal to about 0.0065. As long as $1.0 < k < 1.0 + \beta$, the reactor will be supercritical but its population will multiply on time scales comparable to the delayed neutron fraction release time, typically tens of seconds. This provides the margin that allows the reactor to be subject to operator control. On the other hand, if $k > 1.0 + \beta$, the reactor will become "prompt supercritical," and its power level will increase on a time scale of about 0.1 milliseconds. Such a reactor could not be controlled by human operators or mechanical processes, and thus is impractical.

The value of k in a reactor may be changed by the manipulation of control mechanisms which introduce or withdraw neutron absorbing substances from the reactor. The degree of control authority exercised by such mechanisms is measured in units of "dollars." A control mechanism whose manipulation across its operating range induces a change in the value of k equal to β is said to be "worth" $1.00, while those with less authority than that have their value measured in an appropriate number of cents. The k value of a reactor may also be changed by feedback mechanisms caused by changes in system temperature. Reactor safety is improved if such feedback mechanisms are negative, i.e. $dk/dT < 0$, because in such a case a power excursion that results in a temperature rise will cause k to decrease to 1.0, at which point the system will tend to stabilize itself. On the other hand, if the reactivity decreases as the temperature rises, a system with sufficient reactivity to start at zero power may find itself unable to achieve operation at its design point temperature. Such systems must have reserves of reactivity to draw on if they are to be made able to overcome negative feedback and achieve the desired operating temperature.

The k value of the NAV reactor design point was calculated by the FISSION code to be 1.024 with all control mechanisms positioned so as to maximize system reactivity. The system reactivity is a strong function of the enrichment of Li^7 within the moderator. The potential variation in maximum value of k within the NAV reactor with Li^7 enrichment is shown in Fig. 15. It can be seen that if it were found that greater reserves of reactivity were necessary or desirable, k values greater than 1.25 can be created in the NAV reactor by an appropriate increase in Li^7 enrichment.

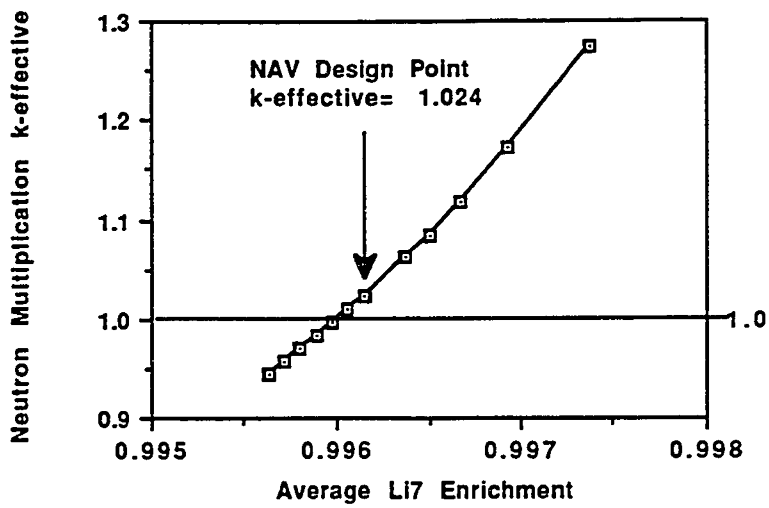


Fig. 15 Variation of $k_{effective}$ with Li^7 Enrichment

As shown in Fig. 13, the NAV reactor has two independent set of of control mechanisms; an outer ring of 12 control drums and two rings of 6 cruciform control rods.

Each control drum is a cylinder, one half of which is composed of moderator material and the other of B_4C, a very strong neutron absorber. When the drums are rotated so that the B_4C halves are moved towards the center of the reactor, reactivity is suppressed. Using the fission code, the difference in reactivity between a condition in which all control drums were rotated B_4C side "in" as opposed to that in which all drums had their B_4C side facing "out" was evaluated. It was found that all of the control drums working together were able to create a Δk of 0.003, or about 46 cents. According to data presented in reference 9, this is sufficient to swing a reactor from a constant power output to one whose power level is increasing exponentially with a period of about 6 seconds. Slower power increases could be obtained by partial manipulation of the drum system. The drum system thus appears to be well suited for controlling reactor power level during flight.

The NAV, however, with a maximum k value of 1.024, has a reactivity reserve of $3.69. With a worth of only 46 cents, the control drums alone have insufficient authority to control the NAV reactor. Thus sets of control rods are necessary.

The NAV control rods are of the cruciform type widely used in the commercial nuclear industry. They are stainless steel blades which have sandwiched in between them a row of tubes containing B_4C, making the rods strong neutron absorbers. The rods can be withdrawn from the core into a plenum within the reactor pressure vessel above the core, or held in the plenum for insertion within the core. The worth of the control rods was calculated using a procedure outlined in reference 9. In this procedure, the rate of absorption of the calculated reactor neutron flux impinging upon the rod surface is calculated, and compared with the rate of absorption of the neutron flux throughout the reactor with absorbing materials throughout the reactor system. The ratio of total absorption rate of the rod plus the reactor to that of the reactor alone then yields an approximate value for the worth of the rod. It was found that the total system of 12 rods could create a Δk of 0.07, thus giving them a value of about $10.76. Each of the large (10 cm blade) control rods in the outer ring was found to have a worth of $0.92, while each of the smaller (6 cm blade) rods in the inner ring was found to have a worth of $0.87. The reason why the worth of the small and the large rods have values that are so nearly equal is because the flux at the inner ring location is about 60% greater than that at the outer ring, and thus their effective areas are weighted accordingly.

With 2 inner rods plus two outer rods fully inserted, plus the control drums positioned with B_4C "in," the reactivity of the system is cut by $4.04, making it subcritical by $0.35. Rotating the control drums can add $0.46, making the system supercritical by $0.11, allowing a slow rise in reactor power. Rapid shutdown of the reactor can be achieved either by drum rotation or by the insertion of any one of the 10 remaining control rods.

In order to avoid thermal shock that might damage reactor components, the rate of temperature rise within the system should be restricted. During the NERVA program, a startup sequence that allocated about 3 minutes to raising the system to its desired operating temperature was found sufficient to avoid thermal shock.[29] Such a slow rise in system temperature can be achieved by bringing the reactor to a critical condition at very low power and supply a small amount of propellant (coolant) as needed to govern the temperature rise.

Due to its construction as a particle bed system, the only portions of the NAV reactor which experience large temperature increases during operation are the fuel particle beds and the neighboring hot frits. This gives the system a negative temperature coefficient of reactivity due to doppler broadening of the absorption cross section of the thorium which comprises the dominant component of the fuel particles.

The change in reactivity of a reactor is frequently described by by a coefficient α_T defined as:

$$\alpha_T = \left(\frac{1}{k}\right)\frac{\partial k}{\partial T} \tag{3.8}$$

Methods of calculating α_T in a reactor with UO_2/ThO_2 fuel pellets are given in reference 9. Using those methods, it was found that for the NAV;

$$\alpha_T = \frac{-0.00028}{\sqrt{T}} \tag{3.9}$$

While the maximum temperature rise in the fuel bed is 2100 K, the average rise in the fuel bed is around 800 K. If the value of α_T

described in (3.9) is substituted into (3.8), and then (3.8) is integrated from T=300 K to T=1100 K, it is found that:

$$\ln\left(\frac{k_{1100}}{k_{300}}\right) = -0.0088 \tag{3.10}$$

Exponentiating (3.10) it is found that Δk = - 0.0088, for a negative reactivity feedback of $1.35. The NAV's reserve reactivity of $3.69 is thus found to be sufficient to overide negative feedback.

3.7 NAV Engine Design Performance

In Table 8 the results of the present NAV reactor design point study are compared to the assumed (scaled) characteristics of a NAV engine used in chapter 2.

In Table 8, T/W ratios are computed on the basis of total thrust divided by engine mass, not including the mass of the shield within the latter. The T/W ratio of 21.7 computed for the present design is quite low for a particle bed reactor, considering that dense CO_2 is the propellant. For purposes of comparison, Reference 5 claims a T/W of 30 for a particle bed reactor using hydrogen as a propellant. The use of CO_2 would be expected (to first order) to quadruple the thrust for a given power density compared to hydrogen; these results are thus conservative compared to the existing particle bed literature.

Table 8. NAV Reactor Design Point vs Previous Scaled Model

	Present Design	Chapter 2 Model
Power	2500 MWt	2513 MWt
Thrust	2083 kN	2080 kN
Chamber Pressure	6.4 MPa	8.1 MPa
Chamber Temp.	2400 K	2400 K
Isp	226 s	224 s
Expansion Ratio	100	100
Engine T/W Ratio	21.7	31.0
Reactor Mass	6870 kg	4190 kg
Shield Mass	8580 kg	8580 kg
Auxiliary Mass	1700 kg	1700 kg
Nozzle Mass	1215 kg	960 kg
Total Mass	18365 kg	15430 kg

In chapter 2 a design of a ballistic NAV vehicle was presented with a total dry mass of 44.7 tons and a propellant capacity of 346.1 tons. If the configuration is kept the same but uses the present engine design point in place of the scaled model used in chapter 2, the total dry mass is increased to 47.6 tons. The maximum fully fueled mass ratio of the vehicle is thus 8.27, somewhat less than the 8.74 possible in the chapter 2 configuration. With a mass ratio of 8.27 and a specific impulse of 226 s, the present NAV design will be able to achieve a maximum ΔV of 4.68 km/s, which compares to the 4.76 km/s ΔV capability of the chapter 2 design. However, since the ΔV (including a gravity loss factor of 1.15) to attain low Mars orbit is only about 4.0 km/s, and the ΔV required to land is about 0.5 km/s, our present design retains the capability to ascend to orbit and land anywhere on the planet. Thus despite the decreased engine performance of the revised design, the NAV's unique capability for global mobility on Mars is retained.

Chapter 4. Performance of NAV Vehicles

4.1 Performance of NAV Vehicles as a Function of Temperature

The performance of the ballistic NAV is shown in Fig. 16. While the design of this vehicle assumes a propellant temperature of 2400 K, for purposes of sensitivity analysis it is useful to vary the propellant temperature across its possible range while keeping the rest of the design fixed. In Fig. 16 the rocket ΔV's given for the vehicle are their ideal free space ΔV's, while the ΔV's required for the specified mission has been increased by a factor of 1.15 to account for gravity losses. For the missions to low Mars orbit (LMO), the 250 by 33000 km (250 by 1 sol) elliptical orbit (HMO), and trans-Earth injection (TEI), a planetary rotational assist of 0.2 km/s has been assumed, and the ΔV required to land again at Mars is not included. Thus for these missions, if the vehicle is to land again without refueling propellant or a parachute delivered from Earth, about 0.5 km/s must be added to the mission requirement indicated. The ΔV required to land is included in the requirement given for the 2000 km hop mission, and no planetary rotational gain is assumed, since the hop can be in any direction (and obviously no refueling from an exterior source is possible prior to landing).

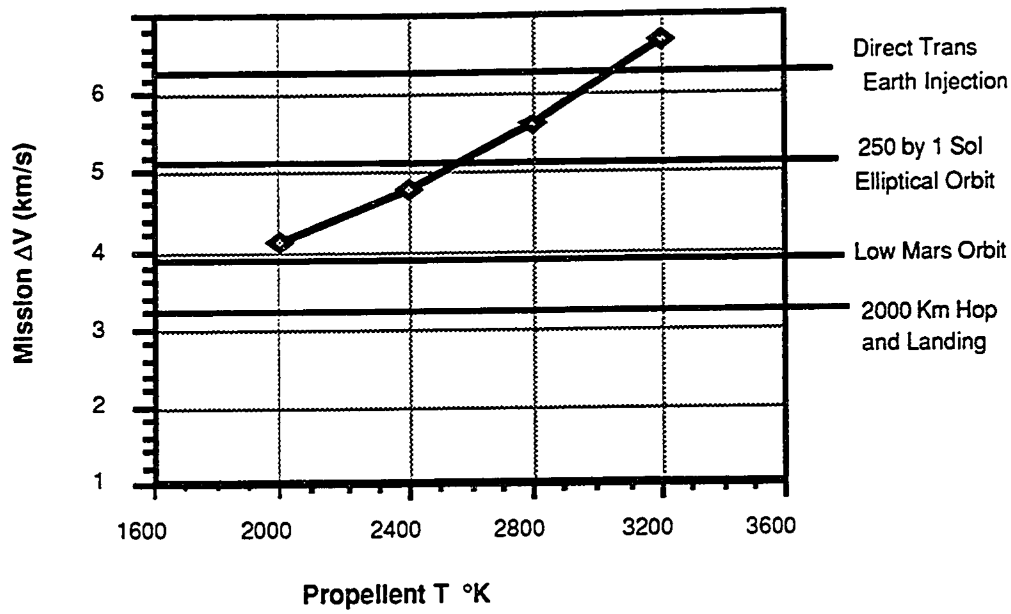


Fig. 16 Performance of the NAV as a function of temperature

It can be seen that at the assumed design temperature of 2400 K, the ballistic NAV can fly from the Mars surface to its design orbit (LMO) and land without additional propellant brought from Earth. It can also be seen that even at the very modest temperature of 2000 K (about 800 K lower than that achieved on NERVA test engines), the ballistic NAV still can attain LMO. If a temperature of 2700 K is attained, then the NAV can achieve a 250 km x 33000 km orbit, which is favored for many missions because its orbital period equals the length of a Martian day and because its ease of entry and exit leads to low mass missions. If temperatures in excess of 3100 K can be achieved (3200 K was achieved in electric furnace tests late in the NERVA program with carbide fuels; the uranium-thorium oxide fuel pellets that would most likely be used in a NAV have a melting point of 3300 K) then the ballistic NAV can achieve direct trans Earth injection on a minimum energy orbit (8 months flight time) after which it could aerobrake into LEO. Operation at such high temperatures will probably lead to degradation of the fuel elements due to reactions between the fuel and the propellant gas, but since such performance would only be required on the final (Earth return) burn of the mission, it may be considered a real, if speculative, possibility.

4.1. Ballistic NAV Simulations Using the FLYIT Code

In order to obtain a more rigorous assessment of the capability of the NAV, a set of computer simulation studies were performed using the FLYIT code[30]. FLYIT is a Martin Marietta proprietary code which simulates the surface to orbit performance of any flight vehicle with three degrees of freedom. Gravity and aerodynamic losses are calculated explicitly for launch from Earth or Mars by directly integrating the primitive equations of motion for the vehicle. A preset program of pitch change is used in an iterative manner in an attempt to reach a targeted orbit. FLYIT has been shown to be accurate by being benchmarked against the industry standard POST[31] code. Its advantage over POST is that it runs much faster. Unlike POST, however, it does not optimize the trajectory. Rather all trajectory optimization must be done manually through sequential runs. Thus, the FLYIT simulations may be regarded as an "existence proof;" a FLYIT simulation shows that a certain performance is possible, but it is not necessarily the best performance possible.

Figure 17 gives the output for a FLYIT simulation of the design base NAV ascending from the Martian surface.

In Table 9 the summary results for FLYIT simulations are given for the NAV assuming propellant temperatures of 2400 K (baseline), 2800 K (optimistic enhancement of the baseline), and 3200 K (speculative.) It can be seen that if the propellant temperature is limited to the baseline of 2400 K, the NAV is capable of reaching a 250 km by 3074 km orbit around Mars. For an enhanced propellant temperature of 2800 K, the NAV can reach a 250 by 35768 km elliptical orbit around Mars. This is somewhat more than that required for the widely admired 250 km by 1 sol (synchronous with Mars' daily rotation) orbit[32] (which is a 250 km by 33000 km orbit). If the propellant temperature is 3200 K, then the FLYIT simulations show that a minimum energy direct trans-Earth injection trajectory can be achieved with a hyperbolic excess velocity, V_h, beyond Mars escape of 2.71 km/s. (A hyperbolic excess velocity of 2.38 km/s is required for a Hohmann transfer from Mars to Earth, assuming an average (circular orbit approximation) condition.)

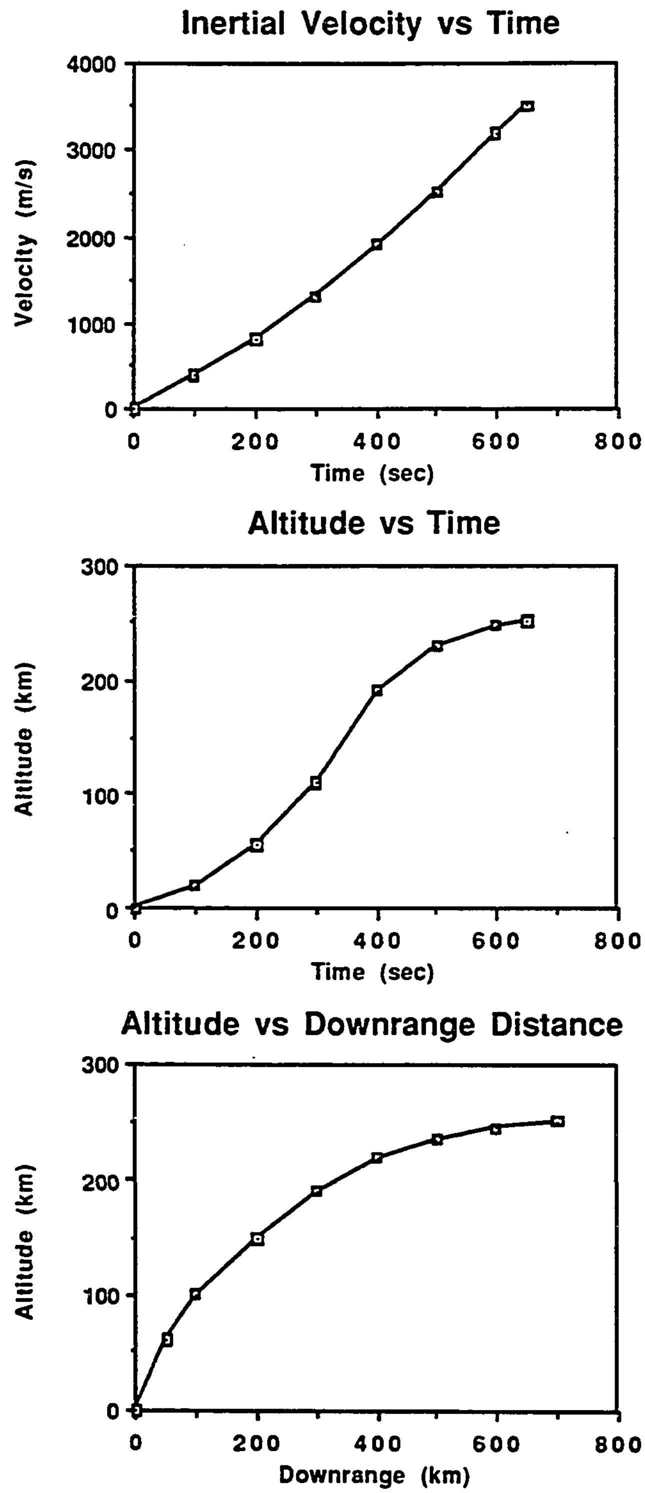


Fig. 17 FLYIT simulation of the NAV ascending to orbit .

Table 9. FLYIT Results for Variable Temperature NAV Ascents

Propellant Temperature	Isp	Final Orbit
2400 K	226 s	250 km x 2348 km
2800 K	269 s	250 km x 35768 km
3200 K	318 s	Escape, V_h= 2.71 km/s

In the FLYIT results reported in Table 9, a burnout mass of 45.1 tons was assumed for the 2400 K case, while the 2800 K case used 44.1 tons. This is 2.5 tons and 3.5 tons, respectively, less than the full mass, the differences being represented by the 2.5 tons of consumables and the 1 ton rover that are included in the NAV total mass of 47.6 tons. On the final ascent to the design orbit, the consumables would no longer be present on board, while the rover could be abandoned to increase ascent performance if necessary. A burnout mass of 45.6 tons for the 3200 K NAV was used, resulting from leaving behind the ground rover (1 ton) and 1 ton of consumables (as up to 1.5 tons will be needed during the trans Earth injection flight). The specific impulses reported for the 2800 K and 3200 K results in Table 9 require nozzles with an expansion of 200, instead of 100 which has been used for the 2400 K baseline.

The FLYIT simulations thus confirm the analytic assessment given in chapter 3 that the ballistic NAV is capable of meeting its design performance objectives.

Chapter 5. Carbon Monoxide Hopper Definition

It has been suggested[33,34] that global mobility can be achieved on Mars through the use of rocket powered hopping vehicles using a carbon monoxide/oxygen bipropellant combination. The propellant for such a vehicle can be manufactured from Martian CO_2 via the reaction:

$$CO_2 \rightarrow CO + \frac{1}{2}O_2 \tag{5.1}$$

Since its propellant can be derived from the Martian environment, such a vehicle represents an alternative concept for performing the same functions as the NAV. In order to analyze the relative merit of the NAV in the mission analysis chapter that follows, an alternative CO/O_2 hopper configuration is defined in this chapter.

5.1 Carbon Monoxide Hopper Design

The carbon monoxide hopper (CMH) defined for the purposes of this study uses the same basic hull structure and overall arrangement as the ballistic NAV with a number of important changes.

The upper storage dome, the control deck, and the habitation deck remain the same, but the compressors have been removed from the compressor deck which now functions as a storage area. (It is also used for storage on the NAV.) The large spherical CO_2 tank of the NAV has been replaced with a heavily insulated bipropellant system consisting of an elliptical liquid oxygen tank with a radius of 3.5 m and a half dome height of 1.75 m, and a single large liquid CO tank with an equatorial radius of 4.39 m and a hemispherical half dome height of 3.1 m. Assuming an ullage factor of 5% and a tankage mass comprising 4% of the propellant contained, these tanks contain 279 t of bipropellant and have a mass of 11.16 t. The nuclear engine is replaced with 6 CO/O_2 engines each with 356 kN (80 klb) thrust. These give the fully loaded vehicle a lift-off thrust/weight ratio of 1.84 with all engines firing, or 1.23 if an engine should go out (and its symmetrical opposite shut off to compensate.) Assuming an engine T/W of 40, the set of chemical engines has a mass of 5.454 t. The bipropellant is burned with a weight mixture ratio of 1:2 LOX/CO (stoichiometric would be 4:7) and produces a specific impulse of 265 s. The overall CMH vehicle layout is depicted in Fig. 18.

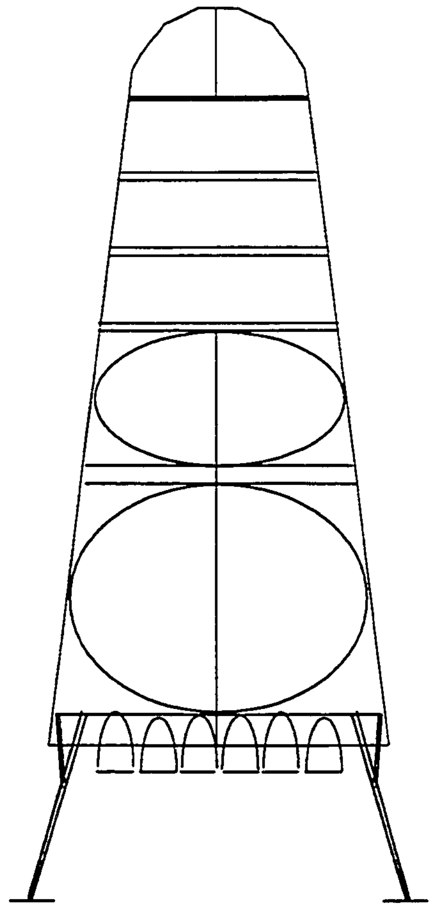


Fig. 18 The Carbon Monoxide Hopper (CMH) overall layout.

The 25 kWe DIPS system of the NAV has been removed and replaced with a deployable (by astronauts) set of solar panels with a peak daytime power output of 22 kWe, or an average power level of about 7 kWe. The CMH thus has a considerably smaller portable power supply than the NAV, but in general it needs less, since it does not do its own propellant acquisition.

Some refrigeration equipment has been added in order to reliquefy the cryogenic boiloff. The crew, consumables, ground rover, and landing gear are all the same as that used in the ballistic NAV.

A mass breakdown of the CMH is given in Table 10. The total mass is 33.554 t. This is about 25% less than that of the ballistic NAV, with the difference primarily caused by the elimination of the radiation shielding.

Table 10. Carbon Monoxide Hopper Mass Breakdown

Component	Mass (kg)
Engines	5454
Tankage/feed	11160
Coaxial Tank	1595
Hull	8000
Landing Gear	1000
Refrigerators	200
Solar Panels	2000
RCS	1000
Ground Rover	1000
Habitation	1000
Crew	240
Consumables	2500
Propellant	0
Total	33554

5.2 CMH Support Requirements

We shall see in chapter 6. that the energy cost of producing CMH propellant is about 50 times greater than that for the NAV, and the process is much more complex. This makes it impractical for the CMH to carry its propellant acquisition system in flight. For example, it will be shown that with a 100 kWe nuclear power source (SP-100) operating around the clock at its full rated power exclusively for propellant production, it would take about 500 days to produce enough propellant to fully fuel the CMH. (By comparison, the same power source could fuel the NAV in 12 days.) Since an SP-100 cannot be relied upon to operate at anything like full rated capacity for this length of time, two SP-100 reactors enplaced at a base are required to insure production of a full load of propellant for the CMH in the course of a 400 day mission stay time at Mars. Attempting a short stay/quick trip would obviously make the situation much worse for the CMH. These reactors can be transported one way to Mars along with the CMH, and be deployed away from it, along with a LOX/CO production and liquefaction unit, with the help of a CO/LOX fueled light truck. Each SP-100 has a mass of 4 tons, and with the weight of the light truck and chemical production unit included it is estimated that this entire assortment has a mass of 10 tons. Once these reactors have been used on the Martian surface, they cannot be moved by the CMH as it carries no shielding and they will be highly radioactive. Thus the CMH must operate from a fixed base.

5.3 CMH Performance

With a dry mass of 33.6 t, a propellant loading of 279 t, and an Isp of 265 s, the CMH is capable of an ideal ΔV of 5.8 km/s. This compares with 4.68 km/s for the NAV. Thus the CMH is capable of reaching higher orbits than the NAV, and, in fact, can reach the 250 x 1 sol elliptical orbit, while the baseline 2400 K NAV is limited to little more than low Mars orbit. However, if used in an exploratory mode to perform a sortie from its base to allow astronauts to visit a second site, it is severely penalized by the fact that it must carry with it enough propellant for the round trip. On the other hand, the NAV need only carry propellant for a one way hop (as it can make the return propellant at the destination.) This means that the ΔV required for the CMH to accomplish a given sortie is twice that required by the NAV. As shall be shown in chapter 7, this will greatly limit the CMH range. The total mass required to be delivered to Mars to put the CMH in operation is 43.6 t, somewhat less than the 47.6 t required for the NAV.

Chapter 6. Propellant Production on Mars

6.1 Carbon Dioxide Acquisition

The only raw material required from Mars for the manufacture of propellant for either the NAV or the CMH is carbon dioxide. The atmosphere of Mars, as measured by the 2 Viking landers, is composed of 95.3% CO_2, 2.7% nitrogen, 1.6% argon, and trace quantities of water, oxygen, and carbon-monoxide[35]. CO_2 is thus the most plentiful material in the Martian atmosphere and can be acquired "free as air" anywhere on the planet. The atmospheric pressure measured at the 2 Viking sites varied over a Martian year between 7 and 10 mbar, with a year round average of about 8 mbar (6 torr) observed at the higher altitude Viking 1 landing site on Chryse Planitia. The acquisition of a gas at 7 to 10 mbar does not represent a significant technological challenge. In fact, pumps which can acquire gas at this pressure level and compress it to a workable pressure of 1 bar or more were first demonstrated by the English physicist Francis Hawksbee in 1709. Better pumps are available today[36].

In an optimized Mars atmospheric acquisition system that was the subject of an extensive design study[37], it was recommended that the compression system be composed of a three stage centrifugal pump, with each pump compressing the gas by a factor of 10.

Assuming an isothermal compression system, in which the heat of compression is released to the environment, the amount of work required to liquefy CO_2 in this manner varies between 60 and 100 kWe-hrs per ton (depending on the outside temperature - Mars surface temperatures vary over the day-night cycle between 180 K and 300 K.) In order to insure quality control in the propellant production process, it is desired that no substances of unknown composition, to wit, Martian dust, be allowed to enter the chemical reactors. This can be accomplished by first placing a dust filter on the pump intake to remove a major portion of the dust, and then compressing the CO_2 to about 7 bar pressure. When CO_2 gas is brought to this pressure and then allowed to equilibriate to ambient Martian temperature conditions, it will condense to the liquid state. Any dust which evaded the pump filters will follow into solution, while nitrogen and argon will remain gaseous and thus can be removed by venting. If CO_2 is vaporized for use from the holding tank it will be distilled 100% pure, since any dust will be left behind in solution.

An alternative method of CO_2 acquisition that has been proposed[38] is to take in Martian atmosphere at low pressure and then refrigerate it below the freezing point of CO_2. The energy cost of this operation is about 60 kWe-hr per ton of CO_2 acquired, slightly less than the compression method. The dry ice would be produced on a batch basis in a small, strong holding tank.

Periodically the tank would be heated and the dry ice melted into liquid CO_2 at high pressure. The high pressure gas at the top of this reservoir would then be pumped into the main tank and compressed slightly, causing it to liquefy, with dust left behind in the holding tank. While this method offers an apparent advantage in energy economy, the problem is that as dry ice accumulates in the holding tank it will insulate the interior of the tank from the refrigeration apparatus. This, in-turn, will slow the propellant acquisition process and degrade the efficiency of the process. In addition, solid CO_2 could block up various intake or outlet feeds, damage the intake fans, or may cause other problems. For these reasons the dry ice scheme is set aside in favor of three stage compression in the current design of the NAV.

The acquisition and liquefaction of pure CO_2 completes the manufacture of propellant as far as the NAV is concerned. In the case of the CMH it provides the essential first step of building up feedstock with predictable chemical properties for further chemical process operations.

6.2 Carbon-Monoxide/Oxygen Bipropellant Manufacture

Once pure CO_2 is acquired, there are a number of methods that may be utilized to produce CO/O_2 bipropellant. Three such methods are summarized in Fig. 19.

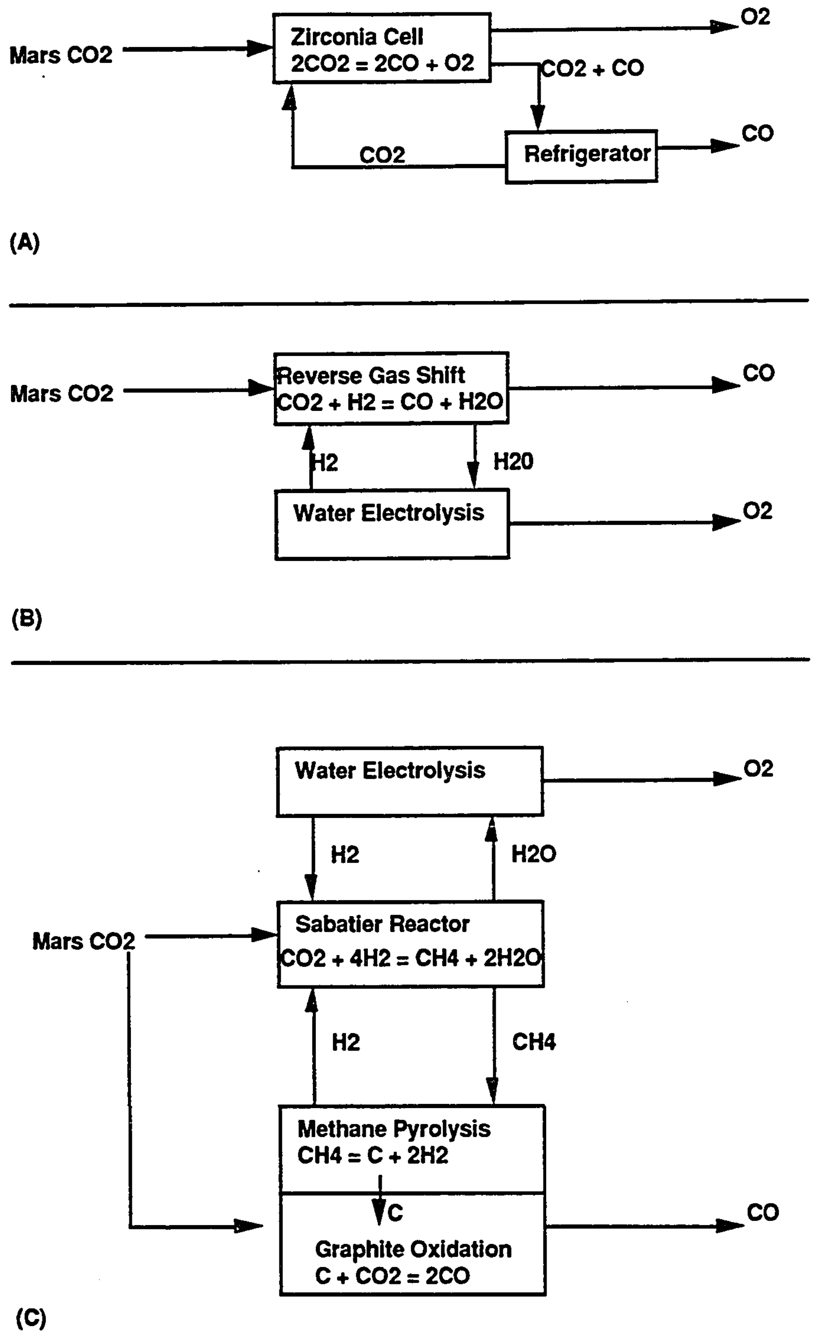


Fig. 19 Alternative Methods of CO/O_2 Production. (A) Direct Reduction, (B) Reverse Gas Shift, (C) Sabatier-Pyrolysis Method

One possible method is the direct reduction of CO_2.

$$CO_2 \rightarrow CO + \frac{1}{2}O_2 \tag{6.1}$$

This reaction can be forced by heating CO_2 to about 1300 K, which will cause the gas to partially dissociate, after which the free oxygen so produced can be electrochemically pumped through a zirconia ceramic membrane by applying a voltage. The use of this reaction to produce oxygen on Mars was first proposed by Ash[39] in the 1970s, and since then has been the subject of ongoing research by both Ash, Ramohalli and Sridhar[40], and Suitor[41]. The oxygen diffuses across the membrane and can then be liquified for an energy cost of about 4.3 MWe-hr per metric ton. Ash does not consider processing the CO out of the CO_2/CO waste gas as this is considerably more complex. Instead he suggests burning the LOX with methane brought from Earth. It is estimated that the CO can be obtained for an energy cost of 6 MWe-hrs/ton. Thus, at a mixture ratio of 1:2 (LOX/CO), the energy cost for propellant production of the CMH is 5.43 MWe-hr/ton.

The advantage of the direct reduction process is that it is completely decoupled from any other chemical process, and an unbounded amount of oxygen can be produced without requirement for additional feedstock. The disadvantages are that the zirconia tubes employed are brittle, and have very small rates (order of grams per day) of oxygen output so that very large numbers of tubes (on the order of 10^5) would be required for the CMH application. These tubes would have to be manifolded in groups, say 200 groups of 500 tubes each, and if a single tube were to break, the entire group of 500 would have to be shut down. Improved yields have recently been reported[42], so the process may be regarded as potentially practical.

An alternative that would keep the set of processes employed firmly within the world of 19th century industrial chemistry, would be to run the well-known water-gas shift reaction in reverse. That is, react some hydrogen with CO_2 in the presence of an iron-chrome catalyst as follows:

$$CO_2 + H_2 \rightarrow CO + H_2O \tag{6.2}$$

This reaction is endothermic to the right and proceeds at 400 K, which is well within the temperature range of common engineering materials. This reaction can be carried out in a simple steel pipe, making the construction of such a reactor quite robust[43]. The disadvantage is that, in the temperature range of interest, it has an equilibrium constant of only about 0.1. This means that in order to drive it to the right it is necessary to overload the left hand side of the equation with extra CO_2 and condense water out to remove it from the right hand side[44]. Once the water is produced, it is split into hydrogen and oxygen by the familiar electrolysis reaction:

$$\text{Electric current} + 2H_2O \rightarrow 2H_2 + O_2 \qquad (6.3)$$

The oxygen produced is refrigerated and stored, while the hydrogen can be recycled to the reverse gas shift reaction (6.2).

Modern electrolysis units are extremely compact and robust, composed of sandwiched layers of electrolyte impregnated plastic separated by metal meshes, with the assembly compressed at each end by substantial metal end caps. Such solid polymer electrolyte (SPE) electrolyzers[45] have been brought to an advanced state of development for use in nuclear submarines, with over 7 million cell-hours of experience to date. Testing for ruggedness has included subjecting cells to loads of up to 200 g's. Both the Hamilton Standard and the Life Sciences companies have also developed light weight electrolysis units for use on the Space Station.[45] The SPE units that Hamilton Standard has supplied for use by Britain's Royal Navy have an output level of sufficient size to support the propellant production requirements of the CMH.[45] These units have operated for periods of up to 28,000 hours without maintenance, about twice the utilization required for the fueling of the CMH. The submarine SPE electrolysis units are very heavy, as they are also used for ballasting purposes. SPE electrolysis units designed for space missions would be much lighter.

Water electrolysis requires about 5 MWe-hrs per ton electrolysed. Reaction (6.2) requires about 3 MWth-hrs per ton to drive, but only 70% of the output material is CO, which makes the effective cost about 4.3 MWth-hrs per ton of CO produced. The fact that reaction (6.2) can be driven by thermal energy seems to be attractive, but it will probably prove to be difficult to place a chemical synthesis plant in the immediate vicinity of the reactor, in which case the 4.3 MW<u>th</u>-hrs/ton CO turns into 4.3 MW<u>e</u>-hrs/ton CO. In any case, a further 0.5 MWe/ton CO will be required to refrigerate the output gas. The net result is that the combination cycle of reactions (6.2) and (6.3) leads to CO/O_2 propellant production for an energy cost of about 4.9 MWe-hrs per ton.

Hydrogen exists on Mars in the form of water in permafrost, vapor (in very dilute concentrations), soil hydrates, and water ice.[35] The acquisition of water to produce hydrogen to carry out processes (6.2) and (6.3) is likely to be an operation of considerable complexity. However, since hydrogen is only a cycling reagent in these processes, it is sufficient to bring an initial supply from Earth to start the process and provide makeup for inevitable leakage loss.

A third possibility which is favored by many practical industrial chemical engineers,[43,46] as it avoids the difficult to drive reverse gas shift (6.2) is to react hydrogen with CO_2 in a Sabatier reactor via:

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O \qquad (6.4)$$

This reaction is exothermic to the right and will occur spontaneously in the presence of a nickel catalyst[47]. The equilibrium constant is very effective in driving the reaction to the right, and production yields of greater than 99% utilization with just one pass through a reactor are routinely achieved[47]. In addition to having been in wide scale industrial use for about 100 years, the Sabatier reaction has been researched by the National Aeronautics and Space Administration, the United States Air Force, and their contractors for possible use in Space Station and Manned Orbiting Laboratory life support systems. The Hamilton Standard company, for example, has developed a Sabatier unit for use on Space Station Freedom, and has subjected it to about 4200 hours of qualification testing[45].

The water produced in reaction (6.4) is condensed, transferred to a holding tank, and finally pumped into an electrolysis cell. Here it is subjected to the electrolysis reaction (6.3) to produce oxygen, which is stored as propellant, and hydrogen, which is recycled to the Sabatier reactor (reaction 6.4). The methane produced there is taken to a pyrolysis reactor where it is decomposed into carbon and hydrogen.

$$\text{Heat} + CH_4 \rightarrow C + 2H_2 \qquad (6.5)$$

The hydrogen so produced would then be cycled back to react with more Martian CO_2 via reaction (6.4). In the course of operation, a graphite deposit will build up in the chamber in which reaction (6.5) is being carried on. (This reaction is actually the most common method used in industry to produce pyrolytic graphite.) When the graphite buildup reaches a certain point, the methane input to the reactor would be shut off and instead the chamber would be flushed with hot CO_2 gas. The hot CO_2 would react with the graphite to form CO, which would then be pumped off to be refrigerated and stored, cleaning out the chamber.

$$CO_2 + C \rightarrow 2CO \qquad (6.6)$$

Such a plan, incorporating two chambers, with one carrying out pyrolysis while the other is being cleaned, has been suggested to the author as the simplest solution to the CO/O_2 production problem by a group of researchers at Hamilton Standard[46].

The Hamilton Standard group also obtained mass estimates for propellant production systems for both a manned CMH and an unmanned sample return mission based on a system combining reactions (6.3), (6.4), (6.5), and (6.6). These estimates are given in Table 11.

Table 11. Hamilton Standard Mass Estimates for CO/O_2 Plant

Reactor	Sample Return	Manned CMH
Sabatier	36 kg	218 kg
Electrolysis	90 kg	636 kg
Pyrolysis	105 kg	600 kg
Total	231 kg	1454 kg
Requirement	3.6 kg/day	540 kg/day
Capability	7.2 kg/day	718 kg/day

The mass estimates in Table 11. assume two complete units, each with 100% mission capacity for the system used in the sample return mission, and four units each with 33% capacity for the system employed on the manned CMH. The reason for the different approach to redundancy on the two missions is that the mass of small scale units used on the sample return mission is fixed regardless of capacity, while the much larger units used on the manned CMH scale in a roughly linear manner with capacity.

The power required to drive the CMH system at the required capacity is 100 kWe, which translates into 4.44 MWe-hr/ton of propellant. When refrigeration power is added in, this becomes about 5 MWe-hr/ton of propellant, which is consistent with the power requirements of the other methods of propellant production. The plant described above, operating at its required level (75% of absolute maximum capacity) can produce the 279 tons required to fuel the CMH for an ascent to its maximum orbit in 517 days. This duration is consistent with the length of a typical conjunction class mission (see Appendix A) surface stay.

In summary, the methods required to produce pure CO_2 propellant on Mars are simple, with energy requirements less than 0.1 MWe-hrs per ton. There are a number of alternative processes available to produce CO/O_2 bipropellant once CO_2 has been acquired. All of these processes involve chemical synthesis plants of some complexity, and require about 5 MWe-hrs/ton of propellant produced. This 50-fold difference in power requirement is that which allows the NAV to produce its own propellant, but forces the CMH to operate out of a fixed base where such infrastructure is available.

Chapter 7. Mission and Trade Studies

7.1 Reduction in Earth to Orbit Mass Trade Study

In order to examine the benefit of the NAV in reducing the required Earth to Orbit (ETO) mass of a manned Mars mission a trade study is conducted. In the study, three different Mars Descent/Ascent Vehicle (MD/AV) technologies are employed; the NAV, the carbon monoxide hopper (CMH), and a conventional two stage MD/AV (CMD/AV) utilizing a set of cryogenic LOX/H_2 engines to land (Isp=450 s) and a set of storable propellant engines (Isp=320 s) for ascent. The CMD/AV incorporates a 6 ton crew cab and 2 tons of engines and tanks in its ascent stage, 1 ton of engine, tanks and landing gear in its descent stage, and has a total wet mass of 47.6 tons. All missions use a 25 ton Mars Transfer Vehicle (MTV) as crew habitation during the minimum energy Hohmann interplanetary transfers from Earth to Mars and return. The conventional mission lands a 25 ton habitation (including supplies, ground rover, and a 2 ton solar array) module one way to the Martian surface, while the CMH and NAV missions use the CMH or NAV vehicles themselves as the required surface habitation. The CMH and NAV have an ETO mass of 53 and 57 tons, respectively, as they include the propellant required for their first landing. After the first mission, the CMH and NAV remain in Mars orbit, and their reuse merely requires the sending of 19.5 or 9.6 tons, respectively, to supply them with consumables, some additional landing propellant, and in the case of the CMH, a new set of SP-100 reactors so it can open up a new landing site. In all cases the Mars Transfer Vehicle is considered reusable.

Space transfer propulsion options considered include cryogenic LOX/H_2 (cryo, Isp=465 s) and nuclear thermal rockets (NTR, Isp=900 s). Aerobrakes with a mass of 15% of the mass they decelerate are employed as options on both types of propulsion system, either at Earth and Mars, Earth only, or neither. NTR empty tank masses are taken as 10% of the propellant enclosed, while cryo tank mass fractions are assumed to be 7%. The NTR engine is a 334 kN thrust unit with a mass of 5 tons, while the cryo engines had a mass of 2 tons and a thrust of 890 kN. The ΔVs employed were 3.8 km/s for trans Mars injection (TMI), 1.5 km/s for Mars orbital capture (MOC) and trans Earth injection (TEI) by the CMH and CMD/AV missions, 2.5 km/s for MOC and TEI for the NAV missions, and 3.8 km/s for Earth orbital capture (EOC). A summary of all the assumptions used in the study is given in Table 12.

Table 12. Reduction in Earth to Orbit Mass Study Assumptions

Space Transfer Propulsion:	NTR	Cryo	Aerobrake
Specific Impulse (seconds)	900	465	-------
Engine Mass	5 ton	2 ton	15% of cargo
Tankage Mass (% of prop.)	10	7	0

Excursion Vehicle:	CMDAV	CMH	NAV
Specific Impulse (seconds)	450/320	265	226
Vehicle Dry Mass	9.0 ton	30.0 ton	44.1 ton
Consumables	2.5 ton	2.5 ton	2.5 ton
Propellant From Earth	38.6 ton	7.0 ton	9.4 ton
Aux. Surface Equipment	25.0 ton	1.0 ton	1.0 ton
Transfer Vehicle Mass	25.0 ton	25.0 ton	25.0 ton
Parking Orbit Apoapsis (km)	33000	33000	250
Parking Orbit Periapsis (km)	250	250	250
Mission ΔVs (km/s)			
Trans-Mars Injection	3.8	3.8	3.8
Mars Orbit Capture	1.5	1.5	2.5
Trans-Earth Injection	1.5	1.5	2.5
Earth Orbit Capture	3.8	3.8	3.8

The reason for the difference in ΔV requirement for the MOC and TEI burns between the different mission types is that the CMD/AV and CMH missions can use the 250 x 33000 km staging orbit (HMO), while the NAV missions must stage in low Mars orbit (LMO). The thrust levels employed were in general large enough to accomplish these minimum energy missions with the indicated ΔVs with a single TMI burn; however, in a few of the heaviest all-propulsive (no aerobraking) cases two perigee burns would be needed. ΔVs used after aerocapture were 0.1 km/s at Mars, and 0.2 km/s at Earth. Space Station orbit (circular at an altitude of 400 km) was assumed for the initial and final mission LEO orbit. The NTR was considered reusable when employed in the all propulsive mode, but expended i.e. not returned to Earth) in the NTR/AB modes. The results of the trade study are given in Table 13.

In Table 13 the column labeled "1st" under each option is the ETO mass required to launch the first mission, while that under "2nd" is the mass required for the second and all follow-on missions. "Cryo" and "NTR" entries use cryogenic or NTR propulsion, respectively, to accomplish all mission ΔVs. Entries such as NTR/AB(E) or NTR/AB(EM) use aerobraking to accomplish the required ΔVs at Earth (AB(E)), or Earth and Mars (AB(EM)), and NTR propulsion to accomplish the remaining mission ΔVs.

Table 13. ETO Masses of Manned Mars Missions (tons)

Propulsion	CMD/AV 1st/2nd	CMH 1st/2nd	NAV 1st/2nd
Cryo	616/591	546/399	842/598
Cryo/AB(E)	431/406	360/214	540/296
Cryo/AB(EM)	355/330	297/174	348/182
NTR	258/228	220/124	278/148
NTR/AB(E)	226/201	188/ 93	234/103
NTR/AB(EM)	216/191	179/ 91	199/ 84

It can be seen that if first missions only are considered, then all the missions for a given space propulsion option are generally about the same in ETO mass (except for the cryo all-propulsive NAV mission, which faces a severe disadvantage due to the NAV's inability to ascend to the 250 x 33000 km orbit). However if follow-on missions are included, the NAV is typically about 10% more than the CMH but in the neighborhood of 50% less than the conventional CMD/AV mission. Furthermore it can be seen that with ETO masses ranging from 148 tons (NTR) to 84 tons (NTR/AB), a follow-on manned Mars mission using the NAV can be accomplished using a single launch of a Shuttle-Z[48] heavy lift launch vehicle (ETO lift capability of 145 tons), an ALS[49] (100 tons ETO capability) or an in-line Shuttle-C[50] (95 tons ETO lift capability).

Roughly half the mission mass of the NAV mission is required to send the NAV one way to Mars, while the other half is required to send the MTV round trip from LEO to LMO and back. This suggests an interesting mission plan. Using NTR and Earth aerocapture (NTR/AB(E)), the ETO mass for the first mission is 234 tons, so that the entire mission can be launched with an in-line Shuttle-C and a Shuttle-Z. (An in-line Shuttle-C consists of a Space Shuttle External Tank with 4 Space Shuttle Main Engines mounted below, an Advanced Solid Rocket Booster attached to either side, and a payload fairing with a 12 meter diameter mounted above. A Shuttle-Z is an in-line Shuttle-C that has been structurally reinforced to accommodate a cryogenic upper stage with about 2000 kN of thrust. The addition of this upper stage increases the launch vehicle's LEO delivery capability 50%, and also gives it the capability to directly inject substantial payloads onto Trans-Mars or Trans-Lunar trajectories.) The Shuttle-C lifts the NAV and an expendable NTR stage (which can use a 3 ton 134 kN thrust NTR) that sends the NAV on Trans-Mars Injection, while the Shuttle-Z lifts the MTV and its propulsion stage. The two parts of the mission thus fly out to Mars in convoy and no on orbit assembly is required. When the MTV returns to Earth it aerocaptures into LEO and is refurbished at the Space Station, after which it mates with a 88 ton propulsion unit lifted from Earth by an in-line Shuttle-C and sets off for Mars again.

A NAV on an in-line Shuttle-C launch stack that can send it directly to Mars is depicted in Fig. 20. If the landing propellant and consumables are brought over with the MTV, then the required ETO lift capability is 95 tons. If the thermal protection on the NAV is upgraded to allow it to aerobrake itself (eliminating the need for a separate expendable aerobrake), then the required ETO capability to send the NAV to Mars orbit is only 84 tons. Alternatively the NAV could be sent with its landing propellant but no consumables for an ETO mass of 95 tons. The MTV component of the mission (which does not aerocapture at Mars, but includes the NAV consumables) is then 105 tons, for a total initial mass in low Earth orbit (IMLEO) of 200 tons. The convoy mission could thus clearly be launched by two near-term heavy lift boosters with no rendezvous, docking or assembly of any type being required in LEO.

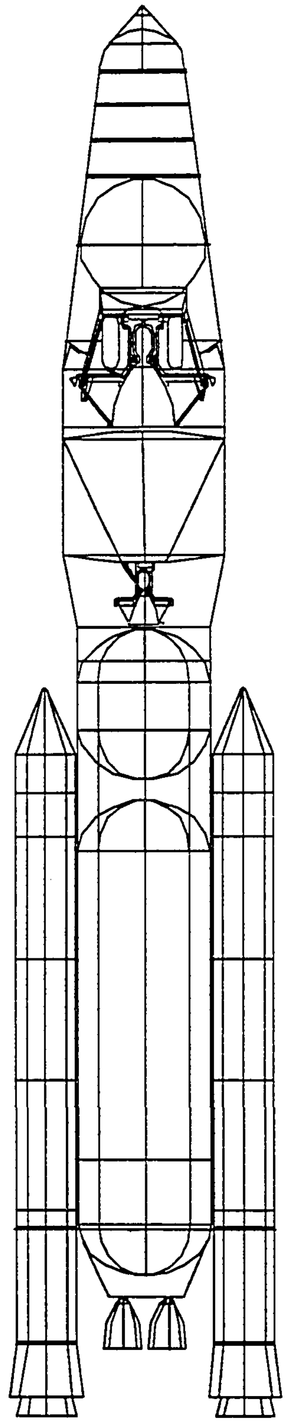


Fig. 20 NAV on an in-line Shuttle-C launch vehicle

Since no piece of equipment is indefinitely reusable, a lifetime limit on all reusable components of five missions is imposed. The average ETO mass per mission for each of the options considered can then be calculated over the course of a five mission program. The results are shown in Fig. 21. In all of the more interesting options (cryo/AB, NTR, NTR/AB) the average ETO mass of the NAV missions are about a factor of two less than that of the conventional CMD/AV baseline, and about 0 to 10% more than the CMH.

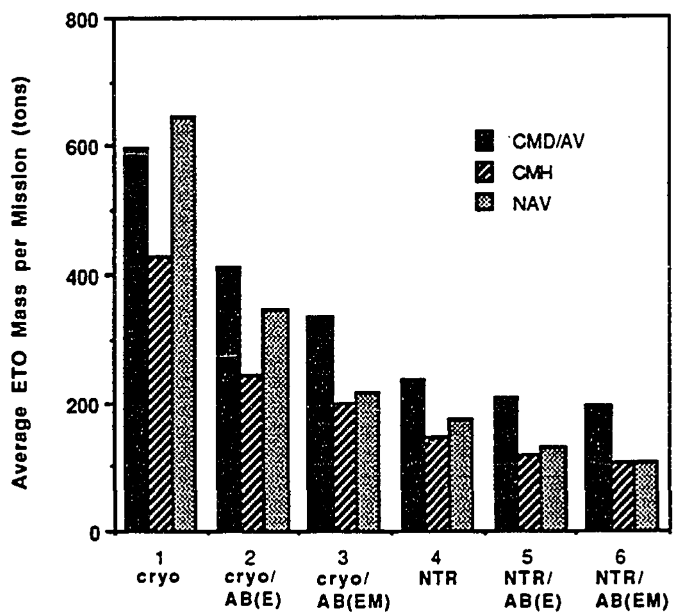


Fig. 21 Average Earth to Orbit Masses of Mars Missions

7.2 A Figure of Merit for Evaluating Mars Missions

In any enterprise with many variables, it is useful to have a global figure of merit for comparing the efficiency of one plan to another. The following dimensionless figure of merit for evaluating manned missions to the Martian surface is proposed.

$$m = (n)(t)(p) \tag{7.1}$$

In equation 7.1, "m" is the dimensionless figure of merit, "n" is the number of discrete landing sites visited by each unit of dry payload delivered to Mars, "t" is the time spent on the Martian surface divided by the total time in transit, and "p" is the payload (excluding propellant) delivered to the Martian surface divided by the total ETO mass. Thus "n" is a science return efficiency factor whose value is governed by the mobility of the system at Mars, "t" is a time efficiency factor (useful time/transit time), and "p" is a mass efficiency factor (useful mass/total mass). For any mission, it is desired that the merit factor, "m," be as large as possible.

For the missions considered, t=1.0 in all cases, as 250 days are spent in transit each way and 500 days are spent on Mars. For the CMD/AV system n=1.0 for all elements, while for the CMH system n=5.0 for the CMH hopper and 1.0 for its reactor/fuel factory elements. For the NAV, n=50.0 over the course of the 5 mission program (10 sites can be visited per mission). The "p" values are computed for all cases and vary widely, but generally are about a factor of 1.9 lower for the NAV and the CMH than the CMD/AV, because of the much larger total mass actually delivered to the Martian surface over the course of 5 missions by the latter system. The resulting merit factor for all options is shown in Table 14.

Table 14. Merit Factor "m" for All Manned Mars Mission Options

Propulsion	CMD/AV	CMH	NAV
cryo	0.055	0.102	0.692
cryo/AB(E)	0.080	0.179	1.293
cryo/AB(EM)	0.099	0.220	2.082
NTR	0.139	0.296	2.437
NTR/AB(E)	0.160	0.378	3.479
NTR/AB(EM)	0.167	0.394	4.177

It can be seen that the CMH missions have a figure of merit about a factor of 2 larger than the CMD/AV missions, while the NAV missions have a merit value 6 to 10 times greater than the CMH or 12 to 20 times greater than the CMD/AV. For purposes of comparison, the Mars mission designed for the NASA 90 Day

Study[51] effort and its sequel has an "m" value of about 0.004, three orders of magnitude less than the NAV missions described here. The 90 Day Study mission is extremely inefficient partly because it uses a CMD/AV lander and partly because it uses an opposition class trajectory which increases the ETO mass and forces the ratio of the productive part of the mission time to the transit time to an absurdly low level (t=0.067). The rationale given for such a mission strategy is that it reduces the "mission time", and thus presumably the mission hazard, but in actuality the total time spent in the zero gravity and radiation environment of interplanetary space flight is the same as for the conjunction class missions employed here, while the useful exploratory time at Mars is reduced. Furthermore, the crew hazards of the opposition class missions are actually much greater than for the conjunction missions because they fly as close to the Sun as Venus, where the radiation field is much stronger, and they encounter the atmosphere of both Earth and Mars with a much greater velocity, exposing the crew and craft to extraordinary acceleration and thermal loads. It is thus recommended that such mission strategies be abandoned from further consideration.

7.3 A Manned Mars Mission with a Single Vehicle

Since the days of the Apollo program it has always been assumed that a manned mission to a major planetary body must include an orbiting mother ship. See for example the MTV in the studies described in section 7.1. The mother ship would be coupled with a small descent/ascent vehicle and a sizeable surface habitation, particularly if the stay at the planet is to be substantial. The reason for this is that if all the propellant is brought from Earth, the penalty associated with lifting a given amount of return payload out of a planetary gravity well is very substantial. Indeed, as we have seen in section 7.1, even lifting a 6 ton crew cab back to HMO results in an increase of the TMI mass of the mission by 48 tons. If lifting a 25 ton spacecraft off the Martian surface using propellants transported from Earth had been attempted, the ETO mass of the mission would have tripled. On the other hand, we have seen that by using the indigenous Martian propellants, CMH or NAV landers with masses of the order of 40 tons can be lifted to Mars orbit and this results in a lower ETO mass than the baseline CMD/AV mission.

The question then arises, can the mission mass be further reduced by abandoning the concept of the orbiting mother ship altogether and simply use the NAV as the habitation during interplanetary transfer. The NAV has a mass of 47.6 tons, compared to the MTV's

25, but set against this we have reduced the three primary pieces of mission hardware (MTV, MDAV, surface hab) to one. It thus seems clear that while the follow-on mission masses using such a mission strategy will be greater than the approach adopted in Section 7.1, the ETO mass of the first mission can be made significantly less, perhaps small enough that the entire first manned Mars mission can be launched "all up" with a single flight of a Shuttle-Z class vehicle.

Such a mission is discussed here, using the space propulsion systems described in Section 7.1, but assuming that the thermal protection of the NAV was suitable to allow the NAV to aerobrake itself, and that additional aerobrakes with a mass of 15% of their payloads are only required for any separate propulsion units whose aerocapture at Mars or Earth is desired. The results of this study are given in Table 15.

In Table 15, options such as "NTR/(LOX/CH4)/AB(EM)" mean that NTR is being used for trans-Mars injection, aerocapture is being used at both Earth and Mars, and LOX/CH_4 is being used for trans Earth injection and raising the orbit periapsis after aerocapture. The idea behind such hybrids is to avoid the necessity of storing large amounts of hydrogen in Mars orbit during the 500 day stay time at Mars. LOX/CH_4 propulsion was assumed to be based upon the technology of the RS-44 engine[52] with a specific impulse of

390 s, while nitrogen tetroxide/monomethylhyrazine (NTO/MMH) was assumed to have a specific impulse of 330 s.

Table 15. NAV "Direct" Missions to Mars

Propulsion	ETO Mass	
	1st Mission	Follow-On Missions
cryo	1081	1036
cryo/AB(E)	478	433
cryo/AB/(EM)	287	242
NTR	280	235
NTR/AB(E)	186	141
NTR/(NTO-MMH)/AB(EM)	233	188
NTR/(LOX/CH_4)/AB(EM)	204	159
NTR/(LOX/H_2)/AB(EM)	184	139
NTR/AB(EM)	144	99

It can be seen that the NTR/AB(EM) option can accomplish a manned Mars mission in a single Shuttle-Z launch, while either of the NTR/AB(E), NTR/(LOX/CH_4)/AB, NTR/(NTO-MMH)/AB or NTR(LOX/H_2)/AB options can accomplish one with in a single Shuttle-Z launch plus a single Shuttle-C. Such single or dual launch missions must be considered a profound advance in capability when compared to the baseline 90 day study mission which requires 5 Shuttle Z launches and can only visit one site at Mars while the NAV mission can visit 10.

If the NAV propellant temperature can be elevated to 2800 K it becomes capable of reaching the 250 x 33000 km orbit, and the mass of the NTR/(LOX/CH_4)/AB mission drops to 145 tons. A complete manned Mars mission configuration using a NAV, NTR for TMI, a space storable LOX/CH_4 stage for TEI and Mars perigee raise, all on an in-line Shuttle-Z launch vehicle is shown in Fig. 22.

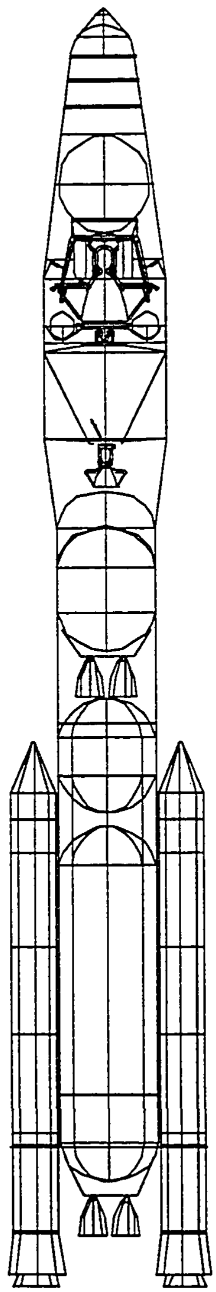
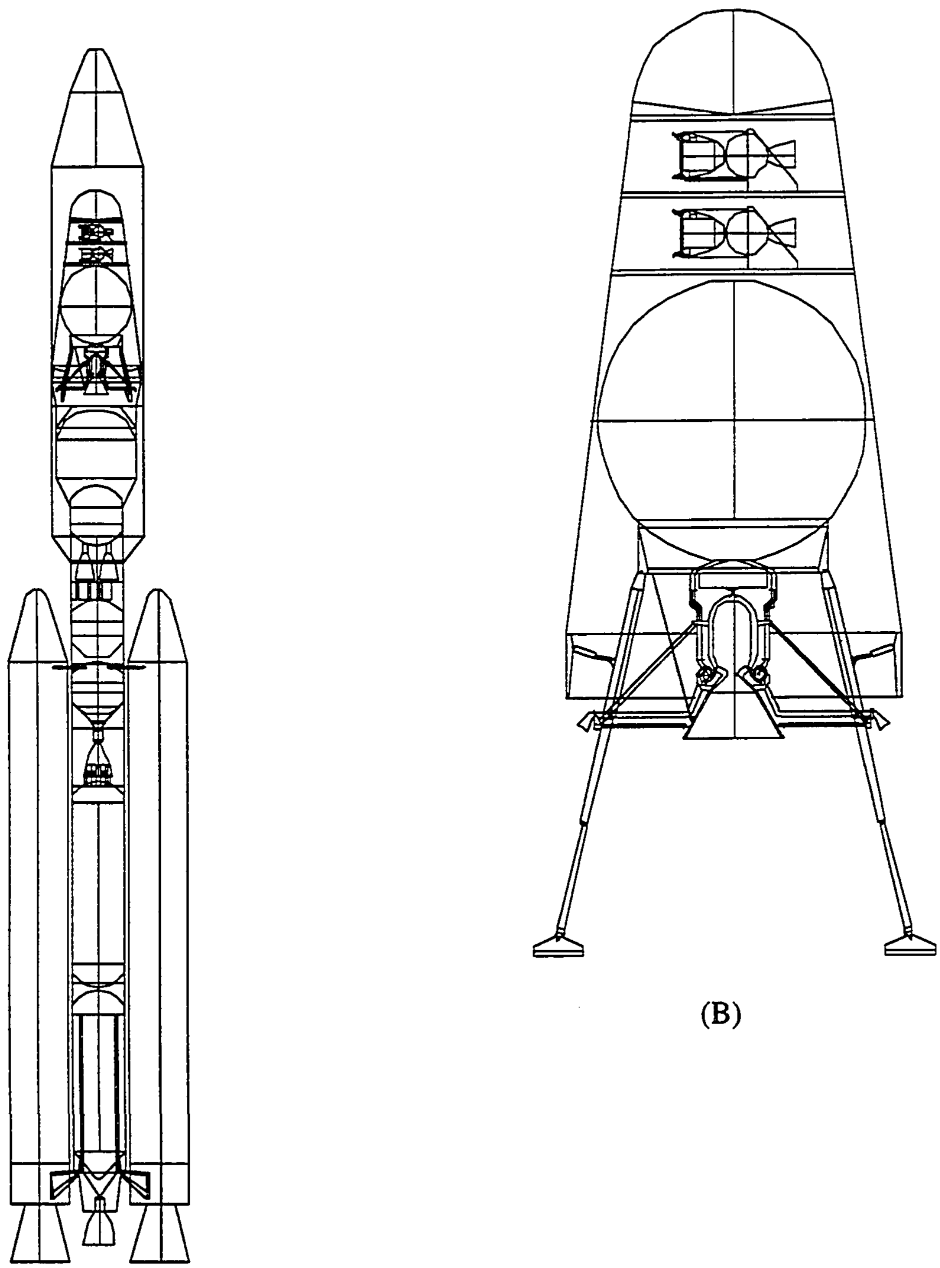


Fig. 22. Fully integrated NAV manned Mars mission on an in-line Shuttle-Z

7.4 Unmanned Mars Rover Sample Return Using a Mini-NAV

There is currently great interest in launching an unmanned mission to Mars that would deliver a roving surface vehicle to the Red Planet and return a set of soil samples to Earth for analysis. This mission, known as the Mars Rover Sample Return (MRSR) mission[53], has been the subject of numerous studies in which conventional chemical propulsion techniques are used. These missions, which bring the return propellant from Earth, are only able to deliver a very small surface payload from a single site area (per lander), and return a very small sample payload to Earth. It is of interest, therefore, to examine whether a scaled down version of the NAV designed in the present study could be utilized to advantage as part of the MRSR mission.

Figure 23 depicts a design for a Mars Rover Sample Return (MRSR) mission utilizing a small unmanned NAV vehicle. The mission is lifted to LEO by a Titan IV rocket and then is propelled onto a minimum energy Trans-Mars injection orbit by an expendable cryogenic LOX/H_2 upper stage. Arriving at Mars, the mini-NAV aerobrakes and lands. Utilizing a 3 kWe DIPS to drive its propellant acquisition pumps, the NAV spends 40 days at its initial landing site filling its 4 m diameter propellant tank, while simultaneously deploying small robots to collect samples and



(A) (B)

Fig. 23 MRSR NAV on Titan IV Centaur (A), and Mars surface (B) emplace instruments. At the conclusion of 40 days, the NAV launches itself on a long distance suborbital hop to land at a second exploration site, where the process is repeated. thus during the 400 days that must elapse before a minimum energy Earth return trajectory becomes available, the NAV can visit 10 widely separated sites on Mars, collecting samples and emplacing instruments for long term global data acquisition. At the conclusion of the 400 day period, the NAV launches itself into orbit around Mars and fires back to Earth 55 kg of Mars surface samples in one of its two torpedo like Sample Return Vehicles (SRVs). The SRV is propelled by a Star 24 solid rocket engine onto a minimum energy trajectory towards Earth, where it makes an Apollo style direct entry into the atmosphere 8 months later. After scientists have examined the samples, the NAV can be redirected to the areas that are the most interesting to collect a second consignment of samples that can then be launched back to Earth 2 years after the first on the second SRV. A total of 110 kg of samples can thus be returned from 10 or more widely separated sites by a single NAV MRSR mission. This compares quite favorably to a conventional MRSR mission which will only return about 5 kg of samples from a single site. A summary of the mission parameters is given in Table 16.

Table 16. Parameters for Mini-NAV MRSR Mission

Mini NAV Mass	5.30 tons
Engine Power	300 MWth
SRVs (2)	0.56 tons
Science Payload	1.00 tons
Cryogenic Stage	14.40 tons
Sample Returned	0.11 tons
Launch Vehicle	Titan IV
Number of Sites Visited	10
Number of Sample Shipments	2

Such an unmanned Mini-NAV MRSR mission would not only provide a rich science return, but might well be an optimum path to proving out the NAV technology in an unmanned mode prior to its implementation on a manned vehicle.

7.5 Comparison of NAV and CMH in Achieving Global Mobility

A key requirement for any space transportation architecture designed for the exploration of Mars is that it be able to achieve "global access," which means planetwide mobility for scientific exploration and for long distance transportation of indigenous materials.

In the past, it has been frequently suggested that the mission of global access could be performed by a chemical rocket ballistic hopper burning CO and O_2 (such as the carbon monoxide hopper, CMH). The viability of such an alternative depends upon a large number of both quantitative and qualitative factors. It is useful, therefore, to draw up a list comparing the merit of such a system to the NAV in performing this mission.

7.5.1 The NAV utilizing CO_2 propellant as a working fluid at 2400 K produces a specific impulse of 226 s, while the chemical vehicle burning CO and O_2 obtains a specific impulse in the neighborhood of 265 seconds. Neither engine is developed technology today, but the physical principles underlying both are well understood. The NAV, however, represents a more formidable development challenge. In both these respects, therefore, the NAV is at some disadvantage in comparison to the chemical vehicle.

7.5.2 The NAV can acquire propellant by compressing it out of the Martian atmosphere at an energy cost of about 84 kWe-hrs per ton. The CO/O_2 fuel for the chemical vehicle must be produced by a chemical processing facility on the surface of the planet at energy cost of about 5,000 kWe-hrs per ton. In this respect, then, the NAV is about 60 times superior to the chemical vehicle. Indeed, opting for the chemical vehicle might be compared to buying a car which can only use gasoline costing $70.00 per gallon. Actually however, the situation is worse than that, because in this case you also have to buy not only the gas, but the gas station, and the oil company too. That is to say, the chemical vehicle will not be able to operate until there is a manned base with a nuclear reactor and a significant chemical processing capability. In other words, no long distance exploration will be possible until after the infrastructure is established. Furthermore, even after the infrastructure is built, the production of fuel for the chemical exploratory vehicle will be an overhead on the base power supply that will be in competition with other demands.

7.5.3 The chemical vehicle must be fueled at a base, while the NAV can fuel itself. This means that when the chemical vehicle takes off it must carry sufficient fuel for both the outbound and return trips, whereas the NAV need only carry sufficient propellant for the hop one way. In effect, this difference in operation cuts the real specific impulse of the chemical vehicle in half relative to the NAV, thus severely limiting its operating range.

In Fig. 24 the mass ratios are given for both a NAV and a chemical ballistic hopping vehicle, assuming that both use aerodynamically decelerated landing with a terminal rocket deceleration requirement of 500 m/s.

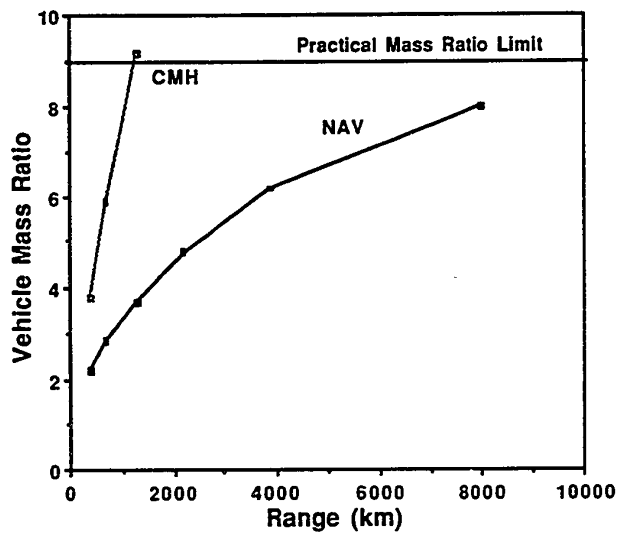


Fig. 24 Comparison of sortie range of NAV and CMH

It can be seen that the maximum effective range of the CMH is about 1260 km (788 miles). That is hardly global access. The NAV, on the other hand, can reach any point on the planet in a single hop. Thus, we see that in this respect the NAV is greatly superior to the chemical vehicle.

7.5.4 Because of its lower performance, the chemical vehicle will have to be built much lighter and carry much less payload than the NAV. This means that it will be structurally less safe, carry fewer scientific instruments, fewer supplies, and have less endurance for an extended visitation to an exploratory site than the NAV. It also means that the chemical vehicle is incapable of performing any role in global transport of indigenous materials (such as transporting water from the Martian polar caps to a base at the equator, or bringing a useful high grade ore from a distant mining site to the base), while the NAV can do the job (as discussed in section 7.6).

7.5.5 The NAV carries a much larger source of electrical power than the CMH (25 kWe DIPS on the NAV compared to a round the clock average of 7 kWe solar on the CMH.) This means that the NAV can recharge the hydrogen/oxygen fuel cells for electric land roving vehicles used locally by the exploration party much quicker than the CMH can. The poverty of electric power faced by a group of explorers utilizing the chemical vehicle my also limit the use of their instruments, and together with their small supply capability, may put them in peril if a minor malfunction should delay their intended return to base.

7.5.6 The CO_2 carried by the NAV is a storable monopropellant under Martian conditions, while both the CO and the O_2 carried by the chemical vehicle are cryogens, and would boil off over time. The boiloff of these cryogenic propellants would itself limit the stay time of the chemical vehicle at an exploratory site. If the boiloff outgassing or other leakage were to occur in any enclosed space (for example that created by an attempt to vacuum jacket the tanks to reduce heat leak or an enclosing fuselage to reduce aerodynamic drag) a flammable (possibly explosive) and toxic mixture could result.

7.5.7 When the chemical vehicle returns to base it must land in the immediate vicinity of the fueling station or it will become useless, as there will be no way to haul it overland through the rough Martian terrain if it lands a kilometer or two away. As the vehicle must use an aerodynamically decelerated landing and Martian winds can be high, the chemical vehicle's requirement for precision landings may prove difficult to meet. The NAV, on the other hand, can land anywhere. If it is only off by a few kilometers the astronauts can walk or return to bases by rover, if it is hundreds of kilometers off, it can just pump itself some more fuel out of the atmosphere and make an additional hop to get home.

7.5.8. Because it refuels after it lands, the NAV can land empty of fuel. The chemical vehicle, on the other hand, must land filled with enough fuel to return home. This means that it is much heavier than the NAV when it is landing, putting increased demands on the engineering of its deceleration system and landing gear. If it hits the ground hard enough to crack its fuel tanks, it may explode.

7.5.9. Set against all these advantages for the NAV is the fact that the NAV carries a nuclear reactor. However the NAV reactor carries a radioactive inventory about 6 orders of magnitude less than a typical commercial power reactor. This small radioactive inventory represents a small hazard compared to that presented by the chemical alternative to the NAV, which will be a lightly built structure filled with toxic gas and chemical explosive.

To summarize, if exploration is desired, what is needed is an exploratory vehicle, a self-contained world that is free to roam at will. The great voyages of exploration of the 15th through 19th centuries were only possible because of the long range capability, independence, endurance and versatility of the full rigged sailing ship. If Columbus had had a coal fired steam paddle wheeler he never would have made it to America. The NAV like the Santa Maria, the Endeavour, and the Beagle before it, derives her motive power from the air about her, and thus must it ever be with true explorers.

7.6 Use NAV Vehicles to Achieve Long Distance Transport

As we have seen in Section 7.5, the NAV offers to unique ability to provide astronaut explorers with complete global mobility on Mars. The question arises as to whether it also offers the potential to provide long distance transport of cargo between two points on the surface of Mars. For example, if a Martian base is located near the equator and it is desired to bring water collected from the north polar ice cap 4000 km away, how much can be transported by a NAV vehicle? This is a very formidable mission, and can only be attempted by a NAV. After all, the CMH cannot even perform a 2000 km round trip empty, let alone a 4000 km round trip returning with cargo.

Let us consider a 4000 km cargo mission with the NAV. The flight out to the pole is no problem, since the vehicle can fly empty between any two points on Mars in a single hop. The question is how far can it fly and land with cargo once it refuels. The NAV reduces its initial propellant loading by an amount equal to the cargo loading (cargo compartments can be created in the skirt, in the void regions above and below the main tank, and on the compressor deck) to preserve liftoff T/W and obtain the results shown in Fig. 25.

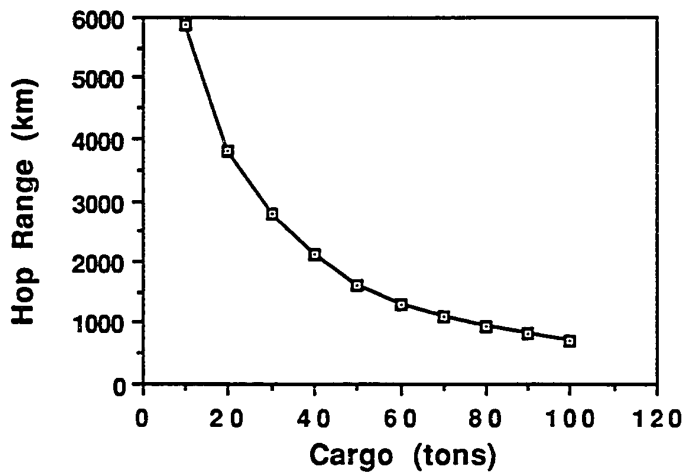


Fig. 25 NAV Cargo Delivery Capability

The results shown in Fig. 25 assume ballistic orbits over a spherical Mars, with the ascent ΔV increased by a factor of 1.15 to account for gravity losses and a landing ΔV of 0.5 km/s to account for a terminal descent burn after aerodynamic deceleration. It can be seen that the NAV can transport cargos of 20 tons over distances of 4000 km, and cargos of 100 tons over distances of 700 km across the Martian surface. This is an impressive capability, which cannot be remotely approached by any chemical system. With the assistance of such a capability a manned Mars base at a single location would be able to access raw materials in large quantities from all points of the planet.

Chapter 8. NAV Alternate Concepts

8.1 Alternative NAV Propellants

In addition to its 95% CO_2, the atmosphere of Mars includes 2.7% nitrogen, 1.6% argon, 0.3% water, 0.13% oxygen, and 0.07% carbon monoxide. With the exception of oxygen, all of these may be regarded as candidate propellants for a NAV. The use of water would depend upon discovering and harvesting ice or permafrost in useful quantities. If ice or permafrost is discovered, the water can also be reacted with carbon dioxide to yield methane and oxygen, using the nuclear reactor as the energy source to drive this endothermic chemical reaction. The methane so obtained can either be burned with the oxygen as a chemical fuel, or used as a propellant gas in the NAV. Carbon monoxide utilization would be based on stripping carbon dioxide of oxygen through one of the methods discussed in chapter 6.

The specific impulse obtainable with each of the above propellants at various temperatures is shown in Fig. 26. A nozzle with an expansion ratio of 100, an efficiency of 0.98, and equilibrium flow is assumed.

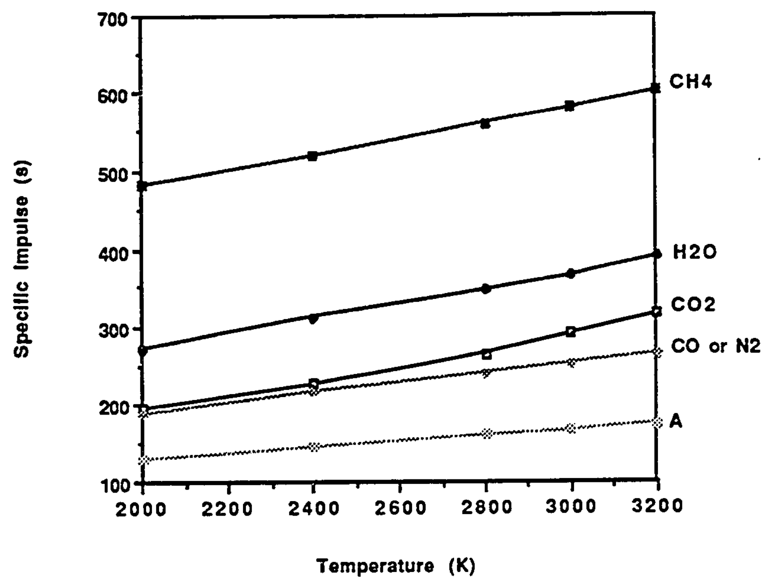


Fig. 26 Specific Impulse of Alternative NAV Propellants

In the following, the characteristics of each of the alternative propellants is examined.

8.2 Water

The characteristics of water as a NAV propellant are similar to those of CO_2 in that there is a low temperature regime where dissociation is negligible, and a high temperature regime where dissociation effects benefit us by providing a significant boost to the specific impulse. The equilibrium constant for H_2O dissociation in the high temperature regime is given approximately by[11]:

$$K_P = \frac{(H_2)(O_2)^{0.5}}{H_2O} = 5.3\exp\left(\frac{-15181}{T}\right) \qquad (\sqrt{bar}) \tag{8.1}$$

If the pressure is 20 atmospheres, this gives a dissociation fraction of 0.038 at 2800 K, increasing to 0.166 at 3500 K. Even a small dissociation fraction, makes high temperature water a strongly oxidizing medium. Thus the use of H_2O, like CO_2, would require oxidation resistant fuel elements.

The key advantage of H_2O over CO_2 is the higher specific impulse obtainable with water. Even at the low temperature of 2000 K, the use of water propellant allows a NAV vehicle easy access to low orbit (with a mass ratio of 5.5), and possible access to highly energetic (250 km by 1 sol) orbits (with a mass ratio of about 8.8). Very high mass ratios may be obtainable with water since the required fuel tank containment pressure is very small.

If the high temperature regime is attained, the use of water propellant gives remarkable performance. For example, a 2800 K water propelled NAV taking off from the Martian surface for a direct minimum energy flight to Low Earth Orbit would have a mass ratio of 6.44. Using the same 2500 MW engine and 47.6 ton dry mass of the baseline CO_2 propelled ballistic NAV presented in chapter 2, a propellant mass of 259 tons, and a weight of 1134 kN on the Martian surface would be required. The engine would have an Isp of 345 seconds, and would generate a thrust of 1478 kN, for a lift off thrust to weight ratio of 1.3.

The main problem with the use of water is finding it, and the second is harvesting it. It is believed by many planetary scientists that vast quantities of water may exist on Mars in the form of permafrost covered by a few feet of sand[35,54]. After all, the planet once had flowing rivers. The existence of such quantities of water on Mars may be verified by the unmanned probes planned by the U.S. and the Soviet Union for the 1990s. At the present, the only large source of water known for certain to exist on Mars, is a several meters thick layer of ice in the north polar cap. This could be harvested rapidly by a NAV utilizing a double hosed device, (Fig. 27) one hose of which pipes down hot CO_2 or steam heated by the NAV's reactor to create a pocket of pressurized hot liquid water under the ice (at a safe distance from the NAV), while the other hose carries the water up to the NAV's propellant tanks. This scheme, however, suffers from the fact that restriction of the NAV's operation to the Martian poles would be very inconvenient. Of course, if the NAV can use both CO_2 and water, it could operate all over the planet using CO_2 and go to the pole to gather water for its final direct injection flight to Earth. A manned Mars mission using such a system would be very attractive. It would consist of the crew being directly sent to Mars in the NAV by an expendable TMI stage, aerobraking and landing on Mars, hopping to several

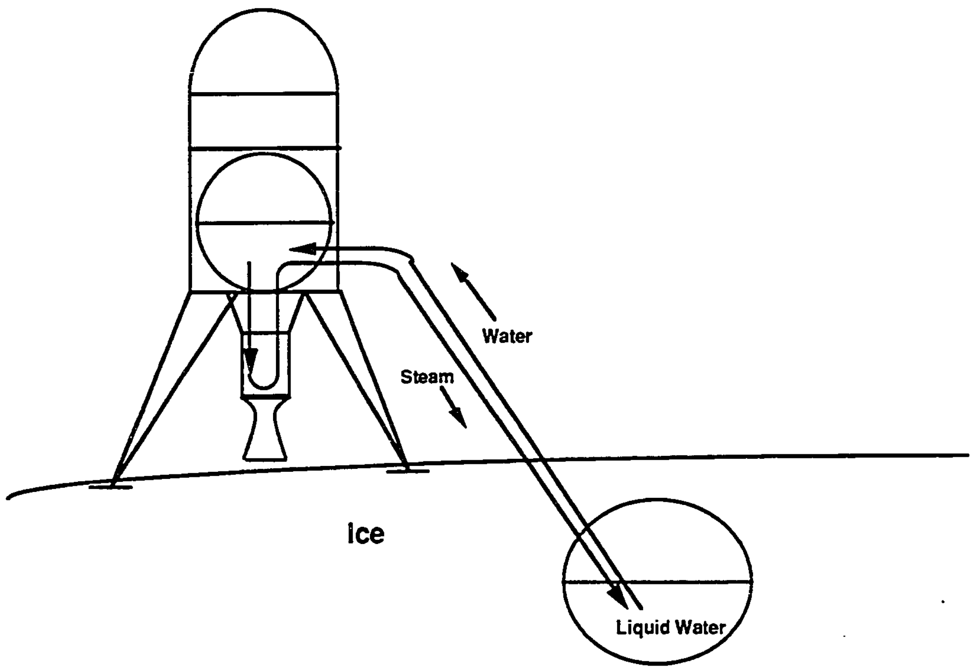


Fig. 27 Water Driven NAV Harvesting Propellant on Ice Field

sites, and then a direct return to LEO from the Martian pole. If the TMI stage was a NTR, initial mass in LEO for the entire mission would be about 95 tons.

8.3 Methane

If water is discovered on Mars, then methane can be produced (along with oxygen) by electrolysing the water into hydrogen and oxygen, and then reacting the hydrogen (using a nickel based catalyst) with CO_2 to produce methane and water. Even if the use of nuclear rockets is not contemplated, it would certainly be worthwhile to transport a nuclear reactor to Mars to drive this process and produce an excellent supply of multipurpose chemical fuel. However, as can be seen from Fig. 26, methane is an excellent candidate propellent for a NAV vehicle, with a specific impulse of 550 seconds achievable. Furthermore, since it does not contain oxygen, the use of methane eliminates one of the major problems associated with either CO_2 or H_2O, namely oxygen corrosion of fuel elements. There is therefore nothing to preclude operation with methane in the high temperature regime using carbide fuel elements similar to those developed for the NERVA program. (Methane would in fact be more compatible with NERVA carbide fuel elements than hydrogen is, as the presence of carbon in the methane propellant reduces its proclivity to react with the carbon in the fuel. For this reason, during the NERVA program, experiments were conducted in mixing gels composed of small quantities of methane in hydrogen, so as to produce a propellant which would be less corrosive to the engine materials.[55])

Methane owes some of its high specific impulse to the fact that it dissociates almost completely at temperatures as low as 1200 K. The specific impulse calculated for methane given in Fig. 22 is based on the assumption that the dissociated monatomic carbon atoms remain as a monatomic gas until such time as they recombine with hydrogen to form methane again in the exhaust. If they were to remain as monatomic carbon and not recombine, Isp would drop to near 515 seconds, which would still be an effective propellant. If, instead, the carbon were to precipitate onto the surfaces of the reactor, the majority of the thrust would be lost and methane would have to be ruled out as a candidate propellant. If the carbon precipitated is only a thin layer, however, the result would actually be highly beneficial, as the carbon layer would protect the fuel and nozzle from corrosion. Such an experience was seen on the Saturn F-1 engine, which burned kerosene.[56,57] More information is needed for this case before a conclusion can be drawn.

Liquid methane would have to kept refrigerated on Mars, but this is not expected to present significant difficulties. Methane liquifies at 135 K (166 K), at 5 (20) atmospheres pressure, respectively. About 11 MWe-hrs/ton are required to produce the methane, so a methane propelled NAV would require the support of a Mars surface base with substantial electric power capability

Liquid methane has a specific gravity of 0.46, which means that our baseline ballistic NAV design, if converted to methane, would have a mass ratio of only 4.07. With an Isp of 565 s (T=2800 K) this would give it a total ΔV capability of 7772 m/s, enabling it to perform a round trip sortie of 3600 km from the base. This is about 3 times as far as the CMH, but it falls short of the surface mobility capability of the CO_2 NAV. The real value of the methane NAV is in performing a direct high energy trans Earth injection from the Martian surface. Taking our baseline ballistic NAV above as an example, the methane NAV would be able to perform a direct return from the Martian surface, leaving Mars with a hyperbolic excess velocity of about 5.2 km/s. This would enable it to arrive at Earth between 120 and 180 days later, depending on the mission opportunity.

8.4 Nitrogen and Carbon Monoxide

The primary advantage of both nitrogen or carbon monoxide as NAV propellants is their lack of chemical reactivity with fuel or cladding materials. Diatomic nitrogen has a dissociation constant of 2.24×10^{-6} at 3000 K, while that of CO is 1.77×10^{-9} at 3500 K. This implies that it will be safe to operate with nitrogen up around 3000 K, provided unprotected carbide fuels are avoided. (Nitrogen tends to substitute for carbon in uranium carbide, forming low melting point nitrides.) Cermet or uranium nitride fuels should be used instead. Carbon monoxide will be benign at any temperature the NAV engine's solid fuel and cladding materials can be made to withstand. Therefore a practical specific impulse greater than 230 seconds may be expected with these propellants, which is more than adequate for ascent to low orbit or ballistic hops. Both of these propellants would require significant refrigeration on Mars, of comparable amount. Nitrogen liquifies at 94 or 116 K and carbon monoxide liquifies at 102 or 123 K at 5 or 20 atmospheres pressure, respectively.[11]

Of the two, nitrogen is probably the simpler to obtain, as it can be compressed and refrigerated directly out of the Martian atmosphere. It has by shown by Meyer and McKay[37], that a small scale compression and refrigeration system can be designed that will liquify a gas mixture of nitrogen and argon from the Martian atmosphere at a cost of about 9.4 MWe-hrs per ton. A larger scale operation should be more efficient[37], however, and it is estimated that about 4 MWe-hrs per ton of buffer gas should be achievable. It will then take about one additional MWe-hr per metric ton for refrigeration to separate the nitrogen from the argon. Since the buffer gas is 54% nitrogen by weight, it is found that liquid nitrogen should be obtainable at an energy cost of about 9.3 MWe-hrs per ton. Carbon monoxide requires about 7.5 MWe-hrs/ton to reduce from CO_2 and liquefy. This is higher than the 5 MWe-hrs/ton cited in chapter 6 for the production of CO/O2 bipropellant, since 36% of the propellant produced in the processes described in chapter 6 is oxygen. If only the CO produced is used as propellant, the process is less efficient.

Such energy requirements for propellant manufacture clearly imply the need for a base power system. Moreover, since the power requirements for propellant manufacture are actually greater than that for manufacturing chemical bipropellants, and the performance is lower, we find that the design of NAV vehicles using nitrogen or CO is an idea which has little merit.

It should be noted that if water is available, it is possible to combine hydrogen from the water with nitrogen to form ammonia, a non-cryogenic, oxygen-free propellant capable of yielding an Isp of around 450 seconds at reactor temperatures of 2800 K. Since the ammonia is relatively dense, performance comparable to the methane NAV may be obtainable. However, since the acquisition of both water and nitrogen is required, the propellant production processes involved may be excessively complex and energy expensive.

8.4 Argon

The primary advantage of argon as a NAV propellant is that it is chemically inert and will not react with the fuel elements or other rocket components at any temperature. As can be seen from Fig. 26, however, the specific impulse available from argon is low, being only 166 seconds at the extreme temperature of 3200 K. Thus its performance is not significantly better than that of the much more available CO_2 at a temperature of 1400 K where dissociation is negligible. Because of its low specific impulse, a argon driven NAV would be limited to sub-orbital ballistic hops.

Argon liquifies at 106 K (132 K) at 5 (20) atmospheres pressure.[11] Based on the results of the Meyer and McKay study[37], it is estimated that it would require about 10.9 MWe-hr per ton to produce from the Martian atmosphere. Since this is more than that required for much higher performing chemical bipropellants, it is clear that the use of argon as a NAV propellant is an idea unworthy of further examination.

8.5 Materials Considerations

The performance attainable by a NAV depends critically upon finding fuel element and propellant materials which are chemically compatible at high temperatures. The earliest NERVA test engines experienced severe corrosion due to high temperature reactions between the hydrogen fuel and the graphite fuel pellets. This problem was later solved by coating the graphite with a layer of zirconium carbide. Such pellets, consisting of a core of uranium carbide surrounded by ZrC coated graphite, would be highly vulnerable to attack by oxygen from partially dissociated CO_2 or H_2O. Improving the pellet by making the entire fuel pellet out of a solution of uranium carbide and zirconium carbide (melting point 3600 K) would allow for operation with hydrogen, methane, or CO in the 3200 K range, but would still leave the pellet vulnerable to corrosion by CO_2 or water.

As discussed in chapter 3, the surest option for a high performance NAV using CO_2 or water propellant is to make the fuel elements entirely out of oxides. Fuel pellets composed of a combination uranium-thorium oxide have been made with melting points above 3300 K. If coated with another oxide (ThO_2) to prevent direct interaction between the water and uranium oxide, such pellets may be able to sustain CO_2 or water driven NAV engines with propellant temperatures of 3000 K. The disadvantage of such oxide fuel pellets is that they would probably not be compatible with hydrogen or methane. It has been suggested by Storms[58] that by doping the hydrogen propellant with a small amount of water, the tendency for the hydrogen to attack the oxygen in the oxide could be suppressed. This would lead to a dual use NAV-interplanetary engine. Experimental data are needed to verify such a possibility.

An alternative approach is that of the cermet fuels[59], which have been investigated by GE, among others. A cermet fuel element is a continuous refractory metal structure surrounding individual uranium oxide or uranium nitride fuel particles each of which are 40 to 75 microns in diameter. This arrangement provides excellent fuel and fission product retention, high strength, and good effective thermal conductivity to the fuel. The cermet is then clad with a refractory metal. In tests at GE, a UO_2-W cermet was found to bond well to a W-Mo-Re alloy cladding. Cermet fuels have been tested for up to 50 hours (and in hundreds of cycles) at 2860 K and briefly at 3020 K without observable damage. In general, cermet fuel elements are expected to be compatible with hydrogen, methane, nitrogen, carbon monoxide, or argon. If used with carbon dioxide or water, a protective coating of either a ceramic or chrome type will be required. Cermet reactors may also suffer from the defect of being heavy for NAV applications.

A considerable amount of experimental research will be needed to determine the optimal fuel/cladding material for CO_2 or H_2O driven NAV spacecraft. However it appears highly probable that appropriate combinations will be found that will allow the NAV to operate in the high temperature (>2400 K) regime using either CO_2 or water. It also seems highly probable that engines can also be developed that can utilize both hydrogen and methane, although work studying the aerothermodynamics of methane propellant will be needed. The prospect of finding a single fuel form that can operate with all four of these propellants at high temperature appears to be rather remote, however.

8.6 Exotic Missions Made Possible Using NAV Propulsion

In addition to its primary purpose as a facilitating technology for manned and large scale unmanned Mars missions, the NAV concept is also an enabling technology for a number of exotic missions whose impossibility without the NAV has caused them to be largely ignored. For example, a winged automated NAV utilizing atmospheric acquisition of CO_2 propellant could accomplish a Venus surface sample return, collecting ground samples and low altitude aerial reconnaissance from every part of the planet before returning to orbit. A methane propelled NAV could use Titan as a base for repeated sorties to Saturn's other moons, and return to Earth with ground samples and low altitude observations of every one of Saturn's satellites. Similarly, a water fueled NAV could use the ice bearing worlds of Europa, Ganymede and Callisto as bases from which to carry out a similar multiple sample return exploration of the Jovian system. Water ice also exists in nearly pure form on Saturn's moons Enceladus, Tethys, Rhea, and Dione, as well as on Uranus' moons Ariel, Umbriel, Oberon, and Titania[60]. Neptune's moon Triton could provide a ready source of methane propellant for NAV exploration of the outer solar system. The asteroid Ceres has ice deposits on its surface, and it is believed that many other asteroids especially in the outermost belt and Trojan regions may also contain large amounts of water ice, thus giving water fueled NAVs multiple bases from which to carry out the prospecting of the asteroid belt. NAVs can extract propellant from the icy cores of comets, and could use comets as staging bases for missions to Pluto and beyond.

If equatorial planetary rotation is taken advantage of, the velocity required to attain Saturn orbit is 14.9 km/s, while that required for Uranus or Neptune is 12.2 km/s. A winged hydrogen fueled NAV with an Isp of 950 seconds could descend into the atmospheres of these planets and collect gas samples (or ground samples, if solid ground exists) refuel itself out of the hydrogen atmosphere, and reascend to orbit. Using the rocket equation (2.1), it can thus be seen that a pure rocket (i.e. not jet intake augmented) NAV would require a mass ratio of about 5.0 to accomplish a Saturn atmosphere return mission, while the mass ratio required for a Uranus or Neptune atmosphere return would be about 3.7. If jet intake augmentation is used during thrust, these mass ratios could be substantially reduced. Such missions may be of great interest because of the high abundance of He^3 fusion fuel in the atmosphere of these planets, as well as their interesting organic chemistry.

If lunar processing facilities are established, sulfur-dioxide, a dense storable liquid could be produced from indigenous lunar materials.[61] A NAV utilizing oxide fuel elements with SO_2

propellant could attain a specific impulse of over 200 s. A vehicle with this performance could attain Lunar orbit with a mass ratio of 2.4 or fly directly into a trans-Mars injection orbit with a mass ratio of about 5.0. Since an oxide-fueled reactor could probably be designed to use both SO_2 and CO_2, and possibly water as well, this means that a Moon-Mars shuttle could be set up that would require no propellant from Earth at all. Such an SO_2 propelled NAV could also function as a long distance ballistic hopping vehicle for travel between widely separated bases on the Earth's Moon. Since the creation and maintenance of Lunar facilities capable of producing SO_2 out of regolith is likely to be an expensive proposition, however, the possible benefits of such an operation (compared to simply launching a fueled trans-Mars injection stage to LEO from Earth) would have to be quantified in a future study.

Chapter 9. Conclusions

This dissertation has reported the results a study from first principles of the potential performance of a Mars ascent vehicle deriving its propulsion from a nuclear thermal rocket engine using Martian CO_2 as propellant.

In spite of the mass penalty associated with a nuclear engine and its shielding, the high density of CO_2 monopropellant was found to allow for the design of a vehicle with a mass ratio greater than 8.

The particle bed reactor concept, utilizing 1.2 mm diameter fuel particles held in 19 fuel beds, each of 5 cm inner radius and 10 cm outer radius, was found to offer the best potential. In the design selected, engine thrust to weight ratios of 22 and specific impulses of about 226 s at a propellant temperature of 2400 K were attained.

These results taken together are sufficiently good to enable a self-fueling vehicle with surface to orbit capability, complete global mobility, and significant surface to surface long distance cargo transport capacity.

These capabilities are unique among current concepts. They would enable a team of astronauts to visit ten or more widely separated sites on the surface of Mars during a typical 500 day conjunction class mission surface stay. Thus, if mission effectiveness is measured by the number of regions of interest that can be explored per mission, the use of a NAV can result in an improvement of the cost-effectiveness of a Mars exploration mission by about an order of magnitude.

It was also found that NAV vehicles employing CO_2 propellant offer high leverage in reducing the launch requirements of manned Mars missions. The use of NAV vehicles can reduce the ETO mass requirements of a series of manned Mars missions by a factor of 2, relative to that required by vehicles using a conventional MD/AV utilizing propellants of terrestrial origin. The NAV's mass benefits are matched, however, by those that accrue to the mission via the utilization of an ascent vehicle employing a CO/O_2 chemical bipropellant mixture produced from indigenous CO_2. The primary and unique benefit of the NAV is thus seen to be its ability to enable global access on Mars, independent of any surface infrastructure.

It was also found that an unmanned Mars Rover Sample Return (MRSR) mission using a scaled down version of the ballistic NAV is extremely attractive. Flown to Mars by a Titan/Centaur launch vehicle, the NAV MRSR mission returns 110 kg of samples from 10 widely separated sites. This is 22 times as much sample mass drawn from 10 times as many sites as the current conventional MRSR baseline. Such a mission is an ideal first application for the NAV, proving out the technology in an unmanned mode while simultaneously returning a very rich scientific harvest.

While NAV vehicles capable of global mobility on Mars can be designed using CO_2 propellant at temperatures as low as 1600 K, surface to low orbit capability requires propellant temperatures greater than 2200 K. Mission performance can be further enhanced if propellant temperatures of 2800 K can be obtained, allowing the NAV to achieve highly elliptical staging orbits around Mars, while a propellant temperature of 3200 K would allow a NAV to perform direct trans-Earth injection from the Martian surface. Experimental work to identify and characterize materials that are compatible with CO_2 at temperatures above 2400 K is therefore greatly to be encouraged. Leading candidates for material surface coatings that may be compatible with CO_2 at highly elevated temperatures include ZrO_2 and ThO_2.

Of the alternative indigenous propellants examined, the most promising (after CO_2) were found to be water and methane. Because of their higher specific impulse, NAV vehicles employing these substances as propellants would be capable of direct trans Earth injection ascents from the Martian surface even when operating at moderate propellant temperatures. This allows for reduction in mission ETO mass by eliminating the need for a trans Earth injection stage. Unfortunately, however, NAV vehicles limited to these propellents would not possess the capability for unlimited Mars surface mobility which is the primary purpose and unique advantage of the NAV. Engine chemistry considerations indicate that it may be possible to design NAV engines that are compatible with both CO_2 and water propellant, and this should be investigated experimentally, as such engines would combine the advantages of both systems.

In summary, it is therefore therefore concluded that the NAV concept holds great promise as an aid to Mars exploration, and should be made the object of further study and a significant national research and development effort.

9.1 A Final Remark

The challenges that will have to be overcome to develop the NAV are not trivial, but nothing worthwhile is ever easy and the benefits to be gained are so large that it would be folly not to proceed. To quote Shakespeare[62]:

Our doubts are traitors,
And make us lose the good we oft might win
By fearing to attempt.

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Appendix A. A Primer on Rocketry and Mars Missions

The following brief discussion of elementary aspects of rocketry and interplanetary missions is provided for the benefit of those readers who are new to the field or whose background can use refreshment. A much more complete introduction to rocketry can be found in the excellent texts by Oates[63] and Sutton[64]. No equivalent texts explaining the multi-faceted art of manned Mars mission design is known to me, however further reading explaining basic methods of calculating interplanetary trajectories can be found in the widely used book by Bate, Mueller and White[65].

A.1 Basic Rocketry

A rocket works by taking advantage of Newton's Third Law of Motion, which states that any reaction has an equal and opposite reaction. Thus if a stream of gas with a mass flow rate dM/dt is ejected from a vehicle in a given direction with an exhaust velocity V_e, the vehicle, of mass M and velocity V, will be accelerated in the opposite direction with an acceleration dV/dt described by the equation:

$$\left(\frac{dM}{dt}\right)V_e = -M\left(\frac{dV}{dt}\right) \tag{A.1}$$

Equation A.1 can be rewritten:

$$\frac{dM}{M} = \frac{-dV}{V_e} \tag{A.2}$$

If equation (A.2) is integrated from the vehicle's initial to final conditions, the result is:

$$\ln\left(\frac{M_i}{M_f}\right) = \frac{\Delta V}{V_e} \tag{A.3}$$

Exponentiating both sides of equation (A.3) leads to:

$$\frac{M_i}{M_f} = \exp\left(\frac{\Delta V}{V_e}\right) \tag{A.4}$$

Equation (A.4) is known as the "rocket equation." The ratio of initial to final mass of the vehicle, M_i/M_f, given on the right hand side is known as the rocket's "mass ratio." Examining equation (A.4) it can be seen that increasing the ability of a rocket vehicle to implement a velocity change, ΔV, requires either increasing the vehicle's mass ratio or increasing the gas exhaust velocity.

The mass ratio required in a rocket vehicle, which is essentially the ratio of the mass of the vehicle (including payload) with propellant tanks full to that with tanks empty, thus increases exponentially as the velocity ratio, $\Delta V/V_e$. Since the maximum mass ratio attainable is not infinite, but is limited by the mass of tanks required to hold a given amount of propellant and mass of the engines required to lift the vehicle, increasing the exhaust velocity is the prime lever for improving system performance.

The exhaust velocity of a rocket can be given in meters per second or any other velocity unit, however in practice it is almost universally given in units of seconds of "specific impulse." The specific impulse of a rocket propellant, abbreviated Isp, is defined as the number of seconds a pound (mass) of fuel can be made to deliver a pound (force) of thrust. The relationship between specific impulse (in seconds) and exhaust velocity is given by:

$$V_e = g(Isp) \tag{A.5}$$

where g is the gravitational acceleration at the surface of the Earth in whatever system of units the exhaust velocity is given in. Thus for example, if a rocket has a specific impulse of 300 s, multiplying 300 times 9.8 gives the exhaust velocity in m/s, while multiplying 300 times 32 gives the exhaust velocity in ft/s.

The thrust, or amount of force that a rocket engine exerts on the vehicle, is given by:

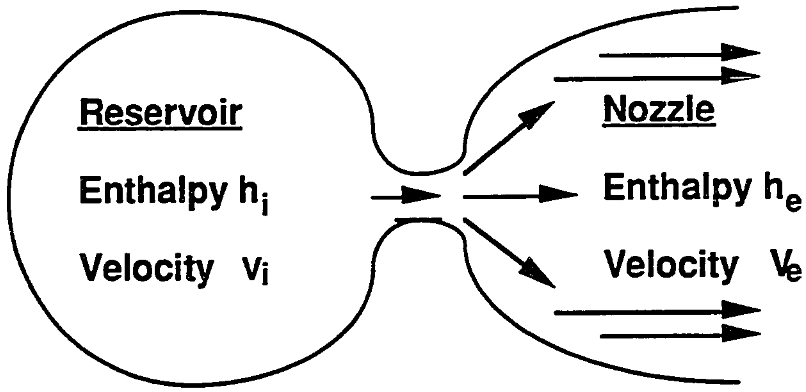
$$\text{Thrust} = T = \left(\frac{dM}{dt}\right) V_e \tag{A.6}$$

The power of a rocket engine is given by:

$$\text{Power} = \left(\frac{dM}{dt}\right)\frac{V_e^2}{2} = \frac{TV_e}{2} \tag{A.7}$$

Thus a rocket whose power is fixed will actually generate more thrust if its exhaust velocity is decreased. This is because while the exhaust velocity has been decreased, the mass flow of such a system will be increased in proportion to the inverse square of the exhaust velocity.

The potential exhaust velocity obtainable from a rocket engine may be calculated on the basis of the principle of conservation of energy. Consider a reservoir of hot gas with a small hole, allowing the gas to exit from the reservoir into a nozzle, as shown in Fig. 28.



$$h_i >> h_e$$

$$V_i << V_e$$

$$h_i + V_i^2/2 = h_e + V_e^2/2$$

Fig. 28 Basic Rocket Thermodynamic Relationships

The gas in the reservoir has enthalpy h_i and velocity V_i, the gas at any point in its travel down the nozzle has enthalpy h and velocity V, and at the nozzle exit plane has enthalpy h_e and velocity V_e. These quantities are then related by the equation:

$$h_i + \frac{V_i^2}{2} = h + \frac{V^2}{2} = h_e + \frac{V_e^2}{2} \qquad \text{(A.8)}$$

Equation (A.8) can be simplified by realizing that the gas within a reservoir with a small exit is nearly static, so V_i is negligible. Furthermore, as the gas is expanded in the nozzle, its temperature, and thus enthalpy drops. For an infinite expansion then, h_e goes to zero and equation (A.8) reduces to:

$$V_e = \sqrt{2h_i} \tag{A.9}$$

Thus in this idealized, infinite expansion, case, the exhaust velocity is found the be simply the square root of twice the enthalpy. Since enthalpy is energy per unit mass, it will be seen that the exhaust velocity is inversely proportional to the square root of the gas molecular weight. Thus CO_2 with molecular weight of 44 may be expected to have a much lower exhaust velocity when heated to a given temperature than hydrogen, with a molecular weight of 2. However, as equation (A.7) shows, a CO_2 gas system heated by the same power source as a hydrogen gas system will also produce proportionately greater thrust.

While a gas released into the vacuum of interplanetary space may indeed by considered to undergo an infinite expansion, only the expansion that occurs while the gas is undergoing unidirectional motion within the confines of a rocket nozzle can be translated into useful thrust. Thus in order to achieve results which approximate the ideal rocket performance given by equation (A.9), large nozzles are needed. The size of such nozzles is ordinarily given in terms of its expansion ratio, ε, which is the ratio of the area of the nozzle exit plane to the area of the gas reservoir exit, which is known as the "throat."

In the case of an ideal gas, a set of relations can be derived which yield the exhaust velocity of the gas for any finite expansion ratio. An extended derivation of these relations is given by Oates.[63] From those relations it can be shown that a nozzle with an expansion ratio of 100 employing CO_2 propellant will yield an exhaust velocity about 95% of that predicted by equation (A.9), for infinite expansion, while an expansion ratio of 200 will yield an exhaust velocity about 97% of that predicted for infinite expansion.

However, even the finite expansion results are still idealized, as they do not take into account energy losses through friction and heat transfer between the gas and the nozzle surface. Practical experience has shown[66], that with proper design to minimize such losses, rocket nozzles can be built capable of achieving exhaust velocities 98% of those predicted by the ideal finite expansion relations. Thus a realistic rocket engine with a nozzle expansion ratio of 100 could be expected to achieve about 93% of the exhaust velocity predicted by equation (A.9), while one with an expansion ratio of 200 could be expected to achieve 95% of the ideal infinite expansion results. Some further improvement could be obtained by going to even larger expansion ratios, but the inconvenient size and weight of very large expansion ratio high-thrust engines generally makes such options undesirable.

A.2 An Introduction to Mars Missions

Chemical and nuclear thermal rockets generate the ΔV required to initiate a given orbit transfer maneuver on a time scale that is very short compared with the length of time required to actually perform the maneuver. For this reason, orbital maneuvers using these high-thrust systems can be calculated to high accuracy by assuming that the rocket ΔV occurs instantaneously, causing the spacecraft to change from one trajectory to another. The path of the trajectories themselves are determined in accordance with Newton's laws of motion and gravitation, and correspond to orbits described by Kepler's laws of planetary motion.

Consider a spacecraft of mass M and velocity V, orbiting a body of mass m at a distance r. The energy per unit mass, E, of the spacecraft is given by:

$$\frac{\text{Energy}}{M} = E = \frac{V^2}{2} - \frac{Gm}{r} \qquad \text{(A.10)}$$

where G is the universal gravitation constant. If the spacecraft is in a circular orbit, all of the quantities in equation (A.10) are constant. If the spacecraft is in an elliptical orbit, G, M, m, and E will be constant, but V and r will vary inversely, as dictated by equation (A.10). While a spacecraft's energy is determined by the square of its velocity, the amount of propellant required to accomplish a given ΔV is independent of the spacecraft velocity. It is therefore the case that a given amount of propellant can be expended most effectively to increase a spacecraft's energy if the rocket engine is fired when the craft is moving its fastest. As equation (A.10) shows, this will occur when the vehicle is at the point in its orbit where r, its distance from the center of the body it is orbiting is the smallest. This point of the orbit is known as the periapsis, and in the special case of a body orbiting the Earth, is also known as perigee. If a rocket engine has insufficient thrust to accomplish the entire required ΔV while the craft is close to periapsis, the efficiency of the orbit transfer maneuver can be sustained by firing the rocket engine only while the vehicle is close to periapsis, and then shutting down the engine until the vehicle cycles around its orbit to return to periapsis for another engine burn. This orbit transfer strategy is known as "perigee kicks" or "perigee burns."

If a spacecraft is gravitationally bound to the body it is orbiting, the energy per unit mass, E, given by equation (A.10) will always be negative. If the spacecraft is not bound to the body, E will be positive. For a spacecraft at a given distance, r, from the central body, the value of V required to set E=0 is known as the escape velocity, V_{esc}.

Consider a spacecraft in low Earth orbit (LEO) that is required to be sent on a trajectory that will take it to Mars. The spacecraft must not only escape the Earth, it must have enough velocity left over after Earth escape to allow it to transfer through interplanetary space to Mars. This extra velocity the craft will have after escaping the Earth is known as its "hyperbolic excess velocity," V_h. Since after Earth escape has occurred, r goes to infinity, the quantity Gm/r drops out of equation (A.10). By definition, $V_{esc}^2/2 = Gm/r$, and thus we obtain :

$$E = \frac{V_h^2}{2} = \frac{V^2}{2} - \frac{Gm}{r} = \frac{V^2}{2} - \frac{V_{esc}^2}{2} \qquad \text{(A.11)}$$

Equation (A.11) can also be written:

$$V^2 = V_{esc}^2 + V_h^2 \qquad \text{(A.12)}$$

The velocity of a spacecraft in a circular orbit about a planet can be determined by setting the acceleration due to its circular motion equal to the acceleration caused by the planet's gravity. Thus for a body in circular orbit:

$$\frac{V_{circ}^2}{r} = \frac{Gm}{r^2} \qquad \text{(A.13)}$$

It follows from equation (A.13) and the definition of escape velocity that:

$$V_{esc}^2 = 2V_{circ}^2 \qquad \text{(A.14)}$$

Using equations (A.13) and (A.14) and textbook values[65] for G and the Earth's mass m, it can be shown that a spacecraft orbiting the Earth in a circular orbit must have at an altitude of 300 km will have a velocity of 7.74 km/s and an escape velocity of 10.95 km/s. Thus a ΔV of 10.95 - 7.74 = 3.21 km/s would be required to transfer the spacecraft from a circular orbit to a trajectory that will escape the Earth.

In 1925, the German mathematician W. Hohmann demonstrated that the trajectory requiring the least ΔV to transfer a spacecraft between the Earth and Mars is an ellipse with its periapsis at Earth and its apoapsis at Mars.[65] In order to attain this "minimum energy" transfer orbit, a spacecraft co-orbiting the sun with the Earth but outside of its gravity well would have to generate a ΔV of about 3.0 km/s. Thus if the spacecraft in LEO were to attempt to transfer to Mars in two burns, one to escape Earth and the other to initiate a Hohmann transfer ellipse to Mars, the total ΔV required would be about 6.2 km/s.

This would be very inefficient, however, as demonstrated by applying equation (A.12) with V_h = 3 km/s. The resulting calculation is $V^2 = (10.95)^2 + (3)^2 = 128.9$ km^2/s^2, or V = 11.35 km/s. Thus by generating both the energy for Earth escape and the required hyperbolic excess velocity with a single rocket burn done in LEO, the total ΔV required would be 11.35 - 7.74 = 3.61 km/s. This ΔV is only 58% of that required if the maneuver were done using the escape-then-transfer scheme, and is only 12% more than that required for Earth escape only.

So after a rocket ΔV of 3.61 km/s the spacecraft is on its way to Mars, where it will arrive 8.5 months later. Arriving at the Red Planet, the spacecraft could use rocket propulsion again to brake itself into orbit, and for reasons that are clear from the above discussion, the least amount of rocket propellant would be expended by braking the spacecraft into a highly elliptical orbit around Mars. Such an orbit would require the smallest ΔV to both enter from an Earth-Mars transfer trajectory, and also will minimize the ΔV required to resume such a trajectory for a return transfer to Earth. One example of such a highly elliptic Mars parking orbit is the widely favored orbit with a periapsis altitude of 250 km and a apoapsis altitude of 33000. This orbit is known as the "250 by 1 sol" orbit because it has a period of 24.6 hours, which is equal to one Martian day, or "sol." Compared to other highly elliptical orbits, the 250 by 1 sol orbit is favored because it would allow an orbiting spacecraft to pass over a base on the surface every day at the same time.

A spacecraft approaching Mars on a Hohmann transfer orbit would have a hyperbolic excess velocity of 2.7 km/s. The velocity at perigee of an craft in a 250 by 1 sol orbit around Mars is 4.7 km/s, and its escape velocity would be 5 km/s. Applying equation (A.12), it is found that the total ΔV required to enter or leave this orbit from a Hohmann transfer interplanetary trajectory would be about 1.0 km/s.

While elliptical parking orbits can be used to minimize the ΔV required to capture a spacecraft around a planet, an alternative technique, known as aerocapture or aerobraking, is available. Aerobraking can almost eliminate rocket propellant required for the maneuver. In performing an aerobraking maneuver, a spacecraft approaching a planet is allowed to fly into the planet's atmosphere and uses the resulting friction with the planetary atmosphere to produce the ΔV required to decelerate the craft from its interplanetary trajectory into a bound orbit about the planet. Since the velocities of atmospheric entry during an aerobraking maneuver are very high (always greater than the planet's escape velocity from low orbit) the heating loads created are very large, and special thermal protection is required. This thermal protection would be provided by using a device known as an aerobrake, or aeroshield, which acts both as a shield to protect the craft from the hypersonic gas flow and as an aerodynamic surface to create the necessary drag and lift to perform and control the aerobraking maneuver. Immediately after aerocapture the spacecraft would find itself in an orbit whose periapsis is within the planet's atmosphere. This situation is intolerable since, after repeated orbits, it would result in de-orbiting the spacecraft. A small rocket burn, generally less than 0.1 km/s is thus required at apoapsis to raise the orbit's periapsis out of the planet's atmosphere. This burn is known as a "periapsis raise" maneuver.

While safe aeroentry from orbit has been done many times on Earth and twice (during the Viking program) on Mars, and aeroentry from trans-Lunar orbit was done during the Apollo program, no spacecraft has ever been aerobraked from an interplanetary orbit into a bound orbit around a planet. The technology, however, is considered a high priority for development by the National Aeronautics and Space Administration, and is currently the focus of a significant research effort.

Mars missions are divided into two categories, "conjunction class" and "opposition class." "Conjunction" and "opposition" are astronomical terms of great antiquity, referring to whether a planet appears to be on the same side of the Earth as the Sun (i.e.

in conjunction with the Sun), or on the opposite side of the Earth from the Sun (i.e. in opposition to the Sun.) Thus a Mars mission employing a Hohmann transfer would be a conjunction class mission, since the transfer is effected with Mars on the far side of the Sun as seen from the Earth, as shown in Fig. 29.

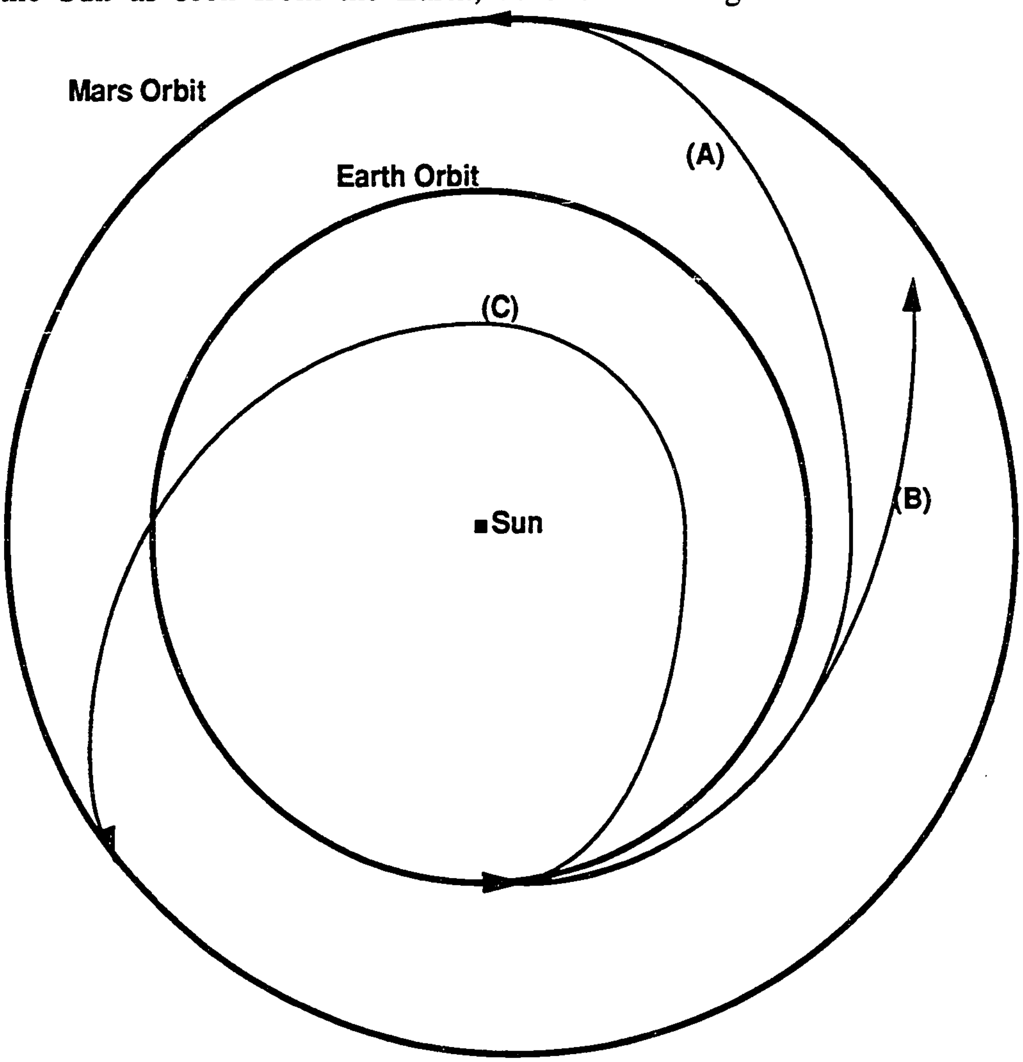


Fig. 29 Alternative Mars Transfer Orbits: (A) Hohmann Transfer, (B) Type I Conjunction, (C) Opposition.

While Hohmann transfer orbit represents the minimum energy option, it may desirable on manned Mars missions to choose an alternative orbit, for example to reduce either interplanetary transit time or total round trip time. If it is desired to reduce transit time, the best option is to take a conjunction class orbit with a somewhat higher energy than the Hohmann transfer, allowing the spacecraft to transfer from Earth to Mars traveling less than the Hohmann transfer's full 180 degrees around the Sun in the process. Such an orbit is known as a "Type I" conjunction class orbit. Such an orbit would have a transit time less than the 8.5 months required by a Hohmann transfer. A "Type II" conjunction orbit would involve the spacecraft traveling more than 180 degrees around the Sun during its Earth-Mars transfer, resulting in a long transit time that would be undesirable for a manned Mars mission.

The use of Type I conjunction transfer orbits results in Mars missions with short interplanetary transits, but waiting for the return launch opportunity entails a surface stay greater than 500 days. If such an extended stay is not desired, an alternative would be to fly the mission on an opposition class trajectory. Such trajectories require the spacecraft to fly into the inner solar system to about Venus's distance from the Sun (with Venus gravity assist maneuvers as an option) in order to swing the spacecraft more than 270 degrees around the Sun during either the outward or inward bound legs of the mission. The Earth-Mars, Mars-Earth transits of such a mission are usually unsymmetrical in length of time, with one leg usually requiring more than a year in flight while the other may be as short as 5 months. The total flight time is thus comparable to a Hohmann transfer. However the time of stay at Mars may be as little as 30 days. The total mission time away from Earth in an opposition mission is thus typically about 1.5 years, compared with 2.5 years for a conjunction mission.

While the short time way from home nature of the opposition class mission has earned it some favor with those concerned with the psychology of small crews on extended missions, the opposition mission has many disadvantages. These include the much greater ΔV's required for opposition missions, which increase the mass of rocket propellant required and thus mission cost, while constraining the amount of exploration and life support equipment that can be sent. The high energy opposition trajectories also make aerocapture maneuvers much more risky, with aero-heating loads greatly increased and crews subjected to more than double the deceleration g loads as would be encountered decelerating a conjunction class mission. Because the transit times of the opposition mission are long, the exposure of the crew to the debilitating effects of zero gravity and the space radiation environment are significantly greater on the opposition mission than on the conjunction class alternative. The thermal problems associated with flying a manned spacecraft through the inner solar system also add risk to the opposition class mission. Perhaps worst of all, however, is that the opposition mission reduces total trip time solely by reducing the stay time at Mars, which is the productive part of the mission. If the weather is bad when an opposition mission reaches Mars, the crew may not be able to land at all before the appointed time to leave. Even in the best of circumstances, an opposition mission would have less than 10% of the available time for surface exploration as a conjunction class mission.

A.3 Surface to Surface Ballistic Trajectories on Mars

The equations of orbital motion can also be used to calculate the range of a ballistic vehicle traveling from one point on the surface of a planet to another. Such a vehicle travels in an ellipse with one focus at the center of the planet. As depicted in Fig. 30, the ellipse has orbital elements including its eccentricity, e, semi-latus rectum, p, and semi-major axis, a. The ellipse intersects the surface of the planet of mass m and radius R at two points, x_i and x_f, which are distributed symmetrically on either side of the major axis of the ellipse. The angle at the center of the planet between the vector to the periapsis of the ellipse and the vector to x_i is ν. The points x_i and x_f represent the initial and final points of the journey, and they are separated by $\phi = 2\pi - 2v$ radians of the planets circumference. The vehicle has velocity V_i when it is at point x_i.

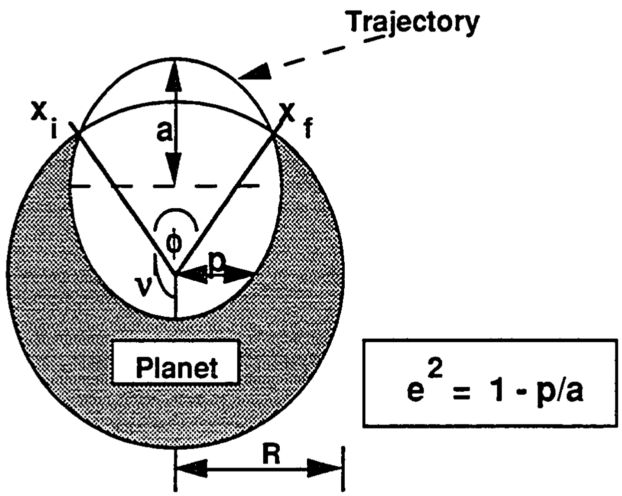


Fig. 30 Elements of a suborbital ballistic trajectory

The quantities a, e, R, m, V_i and p are linked by the following equations[65]:

$$p = R(1 + e(\cos v)) \tag{A.15}$$

$$p = a(1 + e)(1 - e) \tag{A.16}$$

$$E = \frac{V^2}{2} - \frac{Gm}{r} = \frac{V_i^2}{2} - \frac{Gm}{R} = -\frac{Gm}{2a} \tag{A.17}$$

Equations (A.15) and (A.16) can be combined to yield:

$$\cos \nu = \frac{\left(a\left(1 - e^2\right) - R\right)}{eR} \tag{A.18}$$

Equation (A.17) shows that, because R is a constant, "a" is a function of V_i only. So to find the eccentricity, e, that will yield the greatest arc travelled for a given V_i, cosv is differentiated with respect to e in equation (A.18) and the resulting derivative is set equal to zero. After some manipulation, the result that follows is:

$$e^2 = R/a - 1 \tag{A.19}$$

Combining equation (A.18) with (A.19) we obtain:

$$\cos \nu = 2\left(\frac{a}{R} - 1\right)\sqrt{\frac{a}{R - a}} \tag{A.20}$$

If we substitute (A.17) into (A.20), and define $W = V_i^2 R/Gm$, we obtain:

$$\cos \nu = \frac{-2\sqrt{1 - W}}{2 - W} \tag{A.21}$$

It should be noted that since Gm/R equals the square of the low orbital velocity, V_{circ}, $W = (V_i/V_{circ})^2$.

Equation (A.21) gives the maximum range of travel of a ballistic vehicle under the idealized case of a spherical planet with no atmosphere and an initial velocity generated instantaneously at the planet's surface. The results it yields for a vehicle hopping on Mars (R= 3380 km, Gm= 43050 km^2/s^2) are given in Table 17.

Table 17. Ideal Maximum Range for a Ballistic Vehicle on Mars

V_i (km/s)	W	cosv	ϕ (radians)	Range=ϕR (km)
0.5	0.0196	-0.99995	0.020	68
1.0	0.0785	-0.99913	0.083	281
1.5	0.1766	-0.9950	0.200	676
2.0	0.314	-0.9825	0.375	1266
2.5	0.491	-0.9456	0.663	2240
3.0	0.707	-0.837	1.157	3911
3.5	0.962	-0.377	2.367	8000

The artillery equation for the maximum range of a projectile fired on a flat planet is;

$$\text{Range} = \frac{V_i^2}{g} \tag{A.22}$$

where g is the gravitational acceleration at the surface of the planet (g = 3.77 m/s^2 on Mars). It can be seen that for small V_i, equation (A.22) predicts ranges that are only slightly less (66.3 km for V_i = 0.5 km/s, 265 km for V_i = 1.0 km/s) than those predicted by equation (A.21). However as Vi and range are increased, the curvature of the planet becomes more important and the results of (A.22) and (A.21) diverge. When V_i is set equal to V_{circ} (3.57 km/s), the vehicle is orbital and the range given by equation (A.21) goes to 11097 km, which takes the craft to the opposite point on the surface of the planet, while that predicted (very inaccurately) by equation (A.22) is only 3380 km.

As stated above, equation (A.21) is idealized in the sense that it assumes instantaneous acceleration. However a rocket vehicle accelerating in a gravity field will experience "gravity losses" due to the necessity of accelerating the craft in a direction opposite the gravity vector for at least part of the trajectory. Calculation of gravity losses with precision requires the use of a simulation code. However, useful approximate results can be obtained for vehicles with a lift-off thrust-to-weight ratio greater than 1.4 by assuming that gravity losses increase the total ΔV required to attain a trajectory that corresponds to a given V_i by a factor of 1.15.

On Mars, a ballistic vehicle can take advantage of the atmospheric drag to reduce the ΔV required to land from a ballistic trajectory to about 500 m/s, regardless of the ascent ΔV. On an airless planet, the required ΔV for descent and landing would be about equal to the ascent ΔV.

Appendix B. Glossary of Acronyms

AB	Aerobrake
ALS	Advanced Launch System
CMD/AV	Conventional Mars Descent/Ascent Vehicle
CMH	Carbon Monoxide Hopper
DIPS	Dynamic Isotope Power Source
E	Earth
EM	Earth and Mars
EOC	Earth Orbital Capture
ETO	Earth to Orbit
GE	General Electric
HMO	Highly elliptical Mars Orbit
IMLEO	Initial Mass in Low Earth Orbit
JANNAF	Joint Army Navy NASA Air Force
LEO	Low Earth Orbit
LMO	Low Mars Orbit
LOX	Liquid Oxygen
MDAV	Mars Descent/Ascent Vehicle
MOC	Mars Orbital Capture
MRSR	Mars Rover Sample Return
MTV	Mars Transfer Vehicle
NAV	Nuclear Ascent Vehicle
NERVA	Nuclear Engine for Rocket Vehicle Applications
NTO-MMH	Nitrogen Tetroxide-Monomethyl Hydrazine

NTR	Nuclear Thermal Rocket
POST	Program for Optimizing Simulated Trajectories
RCS	Reaction Control System
SPE	Solid Polymer Electrolyte
SRV	Sample Return Vehicle
SSME	Space Shuttle Main Engine
TEI	Trans-Earth Injection
TMI	Trans-Mars Injection
T/W	Thrust to Weight ratio

Vita

Robert M. Zubrin

B.A. Mathematics, University of Rochester, 1974.

M.S. Nuclear Engineering, University of Washington, 1984.

M.S. Aeronautics and Astronautics, University of Washington, 1986.