

Technology Requirements for Mars Sample Return using CO₂/Metal Powder Propellants



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Technology Requirements for Mars Sample Return Using CO₂/Metal Powder Propellants

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Abstract

Mars sample return missions, carrying a selection of rocks, regolith and compressed atmospheric gases, have been discussed in numerous research papers since the days of the Apollo landings but all sample return vehicle propulsion concepts employed conventional bi-propellant, composite solid propellants or cryogenic in-situ propellants. An alternative propellant combination is the use of carbon dioxide, which constitutes about 95% of the Martian atmosphere, as the oxidiser and a metal that reacts with carbon dioxide, such as magnesium, mined/collected and processed into powder, as the fuel. Establishing in-situ propellant production from Martian resources is seen as a prerequisite to any future long duration human expedition to the red planet.

This study introduces the areas that will have to be addressed and mastered before such a propulsion system can be realised. Carbon dioxide will have to be compressed over one thousand times from its ambient temperatures and pressures to a level where it can liquefy to simplify storage. Metal powder, most probably magnesium, will have to be obtained, purified and converted into a pre-defined powder particle size for use as the propulsion system fuel. The various fuel and oxidiser combinations were fed into a basic propellant evaluation program in order to determine the performance characteristics but impurity of the ingredients can, as well as condensed phase effects and two-phase flow loss, contribute to a serious degradation of performance. Metal powder feed issues have been studied on and off for over three decades and this study reviews the trials and tribulations that were experienced during engine tests by previous researchers. Based on these initial findings, a basic propulsion system is proposed to fulfil the requirements of a predefined mission scenario.

Four different sample return mission scenarios are presented and for each concept, a multitude of different propellant combinations are used. The reason for using such a diverse set of propellant combinations is to afford the reader an opportunity to compare conventional bi-propellant, in-situ and cryogenic propellants. The sample return mission presented is simply a one-off mission and since mobility on Mars is limited to landing, sample procurement and take-off, the conventional bi-propellant proved to be superior in terms of mass savings. As the necessity for Mars mobility increases, so will the ΔV requirement and hence the greater amount of propellant from Earth. This is when the lower performing in-situ propellant will prove to be more superior. However, since an initial mission is intended to act as a technology demonstrator, then the selection of ISRU would have to be one that is comparable in trans Mars injection (TMI) mass to a conventional bi-propellant otherwise selecting the in-situ option would not be justifiable. A two stage Mars ascent vehicle (MAV) using ISRU for the lower stage shows that the total TMI mass of the SRV would be no more than 65 kg heavier than a sample return vehicle (SRV) using conventional bi-propellant for both stages of an MAV. This is an acceptable mass penalty for the test and evaluation of a system that could prove essential to the survival of future human expeditions to Mars.

Executive Summary

The Aurora program, which was approved during an ESA ministerial council meeting in Edinburgh, UK in the autumn of 2001, is intended to be a 30-year roadmap, which includes planned human expeditions to the Moon, Mars and possibly even further a field. Aurora is presently in a preparatory period where relevant missions as well as accompanying technologies are being defined. This report is intended to contribute to a better understanding of 'in-situ resource utilisation' or ISRU technologies for employment on Mars.

Long duration human spaceflight beyond CIS-Lunar orbit will eventually require the utilisation of extraterrestrial resources for the production of propellant for propulsion systems and breathable oxygen for life support. For one-off missions, this may not prove economically viable but with increased mobility requirements at far away destinations comes the necessity for additional propellant and with increased mission duration comes the need for oxygen for life support. This translates to an exponential increase in mission cost. One way to reduce mission costs while at the same time encouraging the development of local infrastructure would be to 'live off the land', a process of which is referred to as ISRU.

For almost four decades, interplanetary probes and series of landers have provided information about the red planet via remote sensing imagery and in-situ examination, respectively. However, in order to better determine the characteristics of the samples, the rocks, regolith and atmosphere should ideally be examined in a terrestrial laboratory. Mars sample return missions have been promoted ever since the days of the Apollo lunar landings where the Mars ascent vehicles (MAV) use either storable liquid or conventional solid rocket propellants or in-situ cryogenic propellants. The difficulty with the latter is the additional mass required to produce the propellant on Mars but the performance of the propulsion systems are far superior than conventional, storable propellants. There is an additional problem of how to manage the production of cryogenic propellants, in-situ. An alternative ISRU concept is available and uses semi-conventional technologies. Here, the oxidiser would simply be carbon dioxide (CO₂), which constitutes approximately 95% of the Martian atmosphere and the fuel would be metal powder, which can be mined and processed from the Martian rocks and soil although for initial missions, it is envisaged that the metal powder fuel will initially be brought from Earth, rendering this proposal a "partial in-situ" propulsion concept. The selection of metal fuels is limited primarily to those found in the Martian soil and the choice is determined by its reactivity with CO₂. Although with this ISRU option, the problem of cryogenics is solved, the concept presents a series of unique technological hurdles, which must be overcome before such an engine can be realised.

This Phase 0 is a basic introduction the various fields of study that need to be addressed in order for a propulsion system using CO₂/metal powder fuels to become a reality.

Section 2 addresses the Martian environment and presents the various gases that constitute the Martian atmosphere as well as the elemental composition of the Martian regolith. Knowledge of geology is a requirement to decipher most, if not all, of the mineralogical research papers written on Mars, the depth of which is beyond the scope of this study. Data acquired from orbiting satellites do not provide sufficient information to be able to determine if there are mineralogical concentrations of metals, such as magnesium and aluminium, to direct future mining efforts. Based on current information, such minerals are distributed equally throughout the surface of Mars. A short guide to minerals containing high concentrations of magnesium is provided for future reference.

Section 3 introduces the various ways in which liquid CO₂ can be produced on the Martian surface.

Magnesium was found to be the most reactive metal for use in CO₂/metal powder engines so the focus of attention in **Section 4** which addresses mining and metal powder production, was on this element. The majority of terrestrial magnesium is produced from brine or high concentration salt water but there is one terrestrial process that produces magnesium from ores, similar to one that can be employed for acquiring magnesium from Martian rocks and regolith via a process known as beneficiation, smelting and refining. Once the metal is refined, the metal is fed, in a molten state, into a chamber where an inert compressed gas is directed towards it. The process produces metal powder ready for loading into the fuel tank.

Safety issues pertaining to the production and use of metal powder are also addressed in this section.

CO₂ and liquid oxygen (LOX) were chosen as the most likely oxidisers for use in ISRU metal powder propulsion systems and in **Section 5**, these fluids are combined with a series of ISRU and non-ISRU fuels in order to determine their performance under similar engine operating and atmospheric conditions. A major engineering hurdle to be addressed is the feeding of metal powder from the fuel tank to the combustion chamber. However, the metal fuel need not necessarily be in powder form and these other options are explored in this section.

Section 6 briefly addresses condensed phase effects, which is another problem experienced by metal powder fuels. The incomplete combustion of metal powder particles and two phase flow imply that prior to absolute combustion of the metal, the fuel will be ejected out of the combustion chamber and this leads to a decrease in efficiency in an already poorly performing system. The stoichiometric value for CO₂/magnesium was addressed and it was shown that the CPE was 50%. A method of reducing CPE is to increase the oxidiser content in the formulation however this will reduce overall specific impulse.

While most research papers concentrate on promoting specific concepts and technologies, **Section 7** provides a comparative assessment between a whole series of potential MAV propellants. In addition to addressing the propellants, four different sample return scenarios are presented and for each and every concept, a summary of the trans-Mars injection (TMI) mass is provided to afford to reader an opportunity to compare all the systems in terms of mass and estimated cost of launch.

Section 8 discusses the various methods that the metal fuel can be stored and transported into the combustion chamber. Assuming that metal fuel will be in powder form, a review of previous efforts to transport and inject metal powder is also included.

With all of this information at hand, **Section 9** presents a proposed engine design, together with accompanying dimensions for the 1st stage of a Mars ascent vehicle (MAV) employing a "Martian" in-situ propulsion system using CO₂/magnesium propellant.

Metal powder engines, particularly those that burn with LOX, generate a tremendous amount of heat and **Section 10** introduces the methods in which such heat can be contained to tolerable levels. The minimum chamber wall thickness of the CO₂/magnesium rocket engine plus the type and thickness of the ablative material is calculated and presented for a single use, 1st stage Mars ascent vehicle.

Section 11 summarises the concept's deficiencies and possible solutions, which is followed by **Section 12**, conclusions to this Phase 0 contract.

The in-depth appendix section provides details of magnesium rich minerals, a European directory of gas manufactures and metal mining companies, a complete list of propellant evaluation program (PEP) printouts and thermo-ablation data as well as extensive bibliography on Martian and metal powder research papers.

When compared with one-off, high ΔV missions, ISRU propulsion systems using CO₂/magnesium are much more inferior than their conventional bi-propellant counterparts but if one were to use ISRU for the lower stage of a two stage MAV, then the total TMI mass of the sample return vehicle (SRV) would be no more than 65 kg heavier. This option would afford an opportunity for technology demonstration, which will assist with the development of Martian infrastructure using ISRU.

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Glossary of Terms

amu	Atomic Mass Unit
AOC	Attitude and Orbital Control
AP	Ammonium Perchlorate
APXS	Alpha Proton X-ray Spectrometer
BYOP	Bring Your Own Propellant
Cab-o-Sil	Silicon Dioxide (SiO ₂)
CO ₂	Carbon Dioxide
CPE	Condensed Phase Effects
CPF	Condensed Phase Fractions
DME	Direct Mars Entry
DER	Direct Earth Return
EDLS	Entry, Descent & Landing System
EOI	Earth Orbit Insertion
ERV	Earth Return Vehicle
ESA	European Space Agency
GEO	Geostationary Orbit
GDS	Global Dust Storms
GRS	Gamma Ray Spectrometer
HST	Hubble Space Telescope
IRIS	Infra Red Spectrometer
ISPP	In-Situ Propellant Production
ISRU	In-Situ Resource Utilisation
ISS	International Space Station
K _{st}	Dust Deflagration Index
LCH ₄	Liquid Methane
LCO	Liquid Carbon Monoxide
LEO	Low Earth Orbit
LH ₂	Liquid Hydrogen
LMO	Low Mars Orbit
LOX	Liquid Oxygen
MACDOF	Mars Atmospheric Carbon Dioxide Refrigerator
MAV	Mars Ascent Vehicle
MDV	Mars Descent Vehicle
MCC	Mid-Course Corrections
MMH	Monomethylhydrazine
MOI	Mars Orbit Insertion
MOR	Mars Orbit Rendezvous
NASA	National Aeronautics and Space Administration

NRAO	National Radio Astronomy Observatory
NTO	Nitrogen Tetroxide
OOE	Out-of-Orbit Entry
PEP	Propellant Evaluation Program
ppm	Parts per millimetre
RWGS	Reverse Water Gas Shift
SNC	shergottite-nakhlite-chassignite
Sol	1 Martian Day
SRB	Solid Rocket Boosters
SRV	Sample Return Vehicle
SSMO	Single-Stage-to-Mars Orbit
STP	Standard Temperature and Pressure
STS	Space Transportation System
TEI	Trans Earth Injection
TES	Thermal Emission Spectrometer
TMI	Trans Mars Injection
TSMO	Two-Stage-to-Mars Orbit
UDMH	Unsymmetrical Di-methyl Hydrazine
UV	Ultra-Violet
WSPC	Wickman Spacecraft & Propulsion Company

Software

The software used during the course of this contract were either written or adapted by John H. Wickman, and come as part of a Book and Software set entitled "How to Make Amateur Rockets", CP Technologies, Casper, Wyoming, USA (<http://www.space-rockets.com>). Their functions are described below.

- CHEM-II** This program, calculates the thermochemistry parameters associated with particular propellant formulations. The code is a GUI adaptation of the US Navy's Propellant Evaluation Program. Input parameters are in Imperial (old English units) but the Output is displayed in both Imperial and SI Units.
- FLIGHT** This program is a simple one-dimensional code to calculate the trajectory and altitude of a single stage rocket. The program integrates the forces (gravity, thrust and aerodynamic) on the rocket over time to calculate the altitude. The drag coefficient remains constant with time. Input and Output parameters are in Imperial Units.
- THERM-II** This program calculates the temperature profiles and ablation of the rocket engine assembly. It is a one-dimensional transient thermal conduction program that includes ablating surfaces and the method of determining the amount of ablation is based on the melting temperature and heat of fusion or heat of ablation, depending on what material is used. Only the surface material can ablate or melt and it will not start to melt or ablate until it reaches the melt or ablation temperature. Input and Output parameters are in Imperial Units.

1.0 Introduction

The development of in-situ resource utilisation (ISRU) concepts is considered a prerequisite to embarking on any long duration, manned mission beyond CIS-Lunar orbit. Engaging in ISRU on extraterrestrial bodies such as the Lunar surface affords the possibility of produce breathable oxygen for life support but the oxygen will first have to be separated from the abundant source of metal oxides and then processed for human consumption. The wide variety of metals could be processed and used for manufacturing outposts and spacecraft or, for some metals, be used as a fuel for employment in propulsion systems. Extracting metal from metal oxides is a relatively simple task but trying to capture the oxygen component is not as easy. This need not be the case for Mars. Here, the oxidiser would simply be carbon dioxide (CO₂), which constitutes approximately 95% of the Martian atmosphere and the fuel, like on the Lunar surface, would be metal powder, which can be mined and processed from the Martian rocks and regolith. A wide range of metal powders is available and the selection is determined by its reactivity with CO₂.

Initially, ISRU may not prove to be economically viable but as mobility requirements increase at distant planetary destinations, so will the necessity to take additional propellant from Earth increase. The same applies for an increase in mission duration for human expeditions where oxygen for life support is essential. This translates to an exponential increase in mission injection mass and hence, cost. One way to reduce mission cost while at the same time encouraging the development of local infrastructure would be to 'live off the land'.

Even though numerous spacecraft and landers have visited Mars and examined the planet's environment by both remote sensing and in-situ techniques, a more detailed series of examinations of samples of the soil, rocks and atmosphere can only be performed in a terrestrial laboratory. Samples will therefore have to be returned to Earth from Mars. Mars ascent vehicle (MAV) concepts using storable liquid, composite solid or in-situ cryogenic propellants have been addressed over the past twenty-five years. The difficulty with the latter is that additional mass in the form of a manufacturing facility is required to produce the propellant on Mars but the performance of the propulsion systems are far superior than conventional, storable propellants. The problem then is how one would manage the production of cryogenic propellants, in-situ. An alternative ISRU concept is available and uses semi-conventional technologies; CO₂ in combination with metal powder fuels. Here the problem of cryogenics is avoided, but this concept presents a series of unique technological hurdles, which must be overcome before such an engine can be realised. This Phase 0 is a basic introduction the various fields of study that need to be addressed in order for a "Martian" propulsion system using CO₂/metal powder fuels to become a reality.

The first part of this study concentrates on the Martian environment and its resources. From this information, it was possible to determine which chemicals could be combined and used as a source of propellant. Methods to produce these chemicals were introduced and then the theoretical performance of numerous fuel and oxidiser combinations was examined.

The second part of the study addresses propulsion system related problems, in particular the storage, transport and injection of the metal powder fuel into the combustion chamber.

The third and final part of the report provides a comparative assessment of four separate scenarios and within each scenario, a series of propellant combinations and affords the reader an opportunity to compare in-situ, conventional liquid and cryogenic propulsion systems. Those systems that produce reasonably sized numbers are presented in a trans Mars injection (TMI) mass chart, including the propellant required to send the sample return vehicle (SRV) on its way. The mass allows one to select the most appropriate and cost effective launch vehicle for the mission. ISRU propulsion systems are much more inferior in performance than their conventional bi-propellant counterparts but if one were to use ISRU for the lower stage of a two stage MAV, then it was found that the total TMI mass of the SRV would be no more than 65 kg heavier than a system using bi-propellant for both stages. This option would afford an opportunity for technology demonstration which will assist with the development of Martian infrastructure using ISRU.

2.0 The Martian Environment

2.1 Introduction to the Martian Environment

The Hubble space telescope (HST) and National Radio Astronomy Observatory (NRAO) show that Mars is more often cloudy than dusty and experiences 'mood' swings between dusty/hot and cloudy/cold.

Three drivers dictate the shifts in climate;

1. Mars' thin atmosphere
2. Its elliptical orbit around the sun
3. Strong climatic interactions between dust and water ice clouds.

There are large annual changes in the sunlight falling on Mars' surface due to the planet's elliptical orbit which varies ~20% over the period of one Martian year. During perihelion (closest point of Mars' orbit from to the Sun and summer in the southern hemisphere), the planet receives 44% more surface radiation than during aphelion (the farthest point of Mars' orbit from the Sun and summer in the northern hemisphere). The surface of the planet is highly irradiated with ultra violet (UV) light since there is insufficient ozone in the atmosphere to absorb the incoming solar light. Mars' atmosphere is ~1% of Earth's atmosphere and with no oceans to 'soak' up the heat energy from the Sun, the planet's temperature responds more

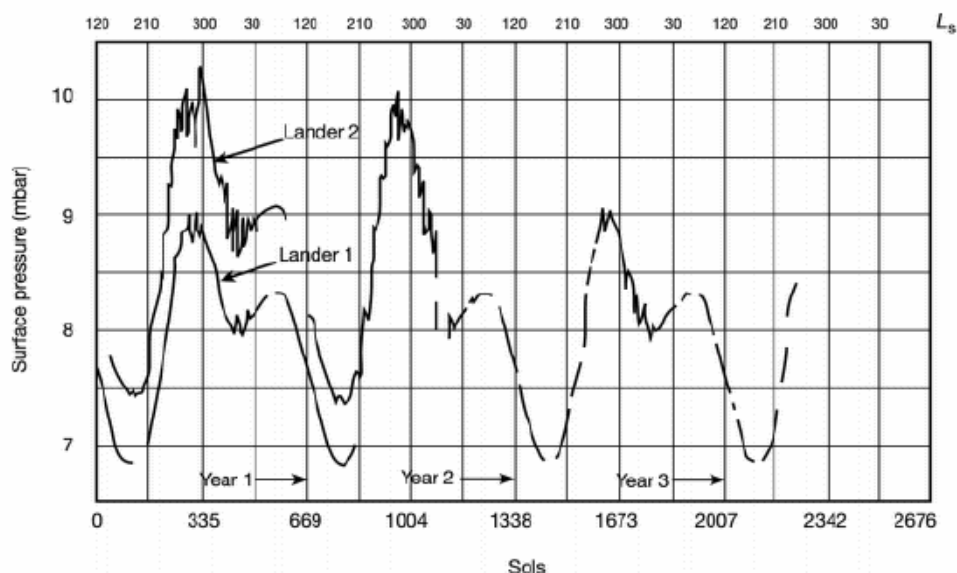


Figure 2.1: Seasonal Changes of Martian Surface Pressure¹

quickly and intensely due to atmospheric heating. This results in a dramatic increase in temperature, forcing continental scale dust storms, referred to as great dust storms (GDS). The dust is swept up, via convection, to altitudes of 10's of kilometres where it disperses globally, absorbs the sunlight and contributes to additional heating of the entire atmosphere. These conditions were observed by Viking, Mariner 9 and NRAO observations in 1992, 1994 and 1996. The annual atmospheric pressure variation is 20 to 30% across the planet¹, producing a rush of air from summer to winter poles at high latitudes. Winds, hurtling from south pole, head toward the warmer aim near the equator and due to the planet's relatively flat southern terrain, the wind flow can accelerate to hundreds of kilometres per hour, picking up fine dust grains and carrying them high into the atmosphere on vertical currents. The airborne dust succeeds in absorbing solar radiation that heats up the atmosphere and lends a hand to further strengthening the winds, ripping more dust from the ground. Eventually, the

¹ the pressure is also dependant on the topography of the measurements. For example, Hellas Planitia is 6km below the Martian ground level whereas Olympus Mons stands 25km above ground level.

blanket caused by the dust storm, which by now is over 24 km deep, blots out the sun. The lower layers of the atmosphere cool and stabilise and the cycle dismantles. This cycle is repetitive and these huge dust storms during perihelion dominate the climate until aphelion.

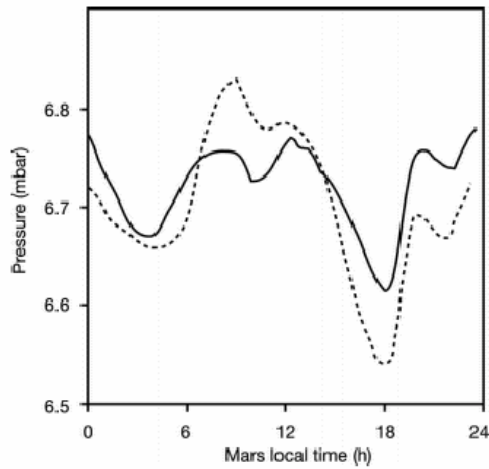


Figure 2.2 : Pathfinder Diurnal Pressure Variations

Between perihelion and aphelion, the sky is salmon pink and cloudless, filled with windblown dust from the Martian surface. The atmosphere has optical depths of 0.45 and dust composed of micron-sized particles. As the planet approaches aphelion, the temperature is 243.15K and when dust is swept up from the atmosphere; temperatures drop another 22K. On average, temperatures range from 273.15K on a warm day near the equator and 148.15K at the winter poles. Volatiles evaporate and mix with the atmosphere in the northern summer and part of the atmospheric CO₂ condenses to form seasonal CO₂ ice cap,

causing seasonal surface pressure change of 20 to 30%. Figure 2.1 shows pressure data versus Martian solar days (Sol) from both Viking 1 and Viking 2 landers. Figure 2.2 gives an indication of the diurnal pressure variations from the Pathfinder landing site. The results presented here are over the period of one Martian day, from Sol = 9 and Sol = 19. Measurements later on in the mission show an increase in Pressures, Figure 2.3.

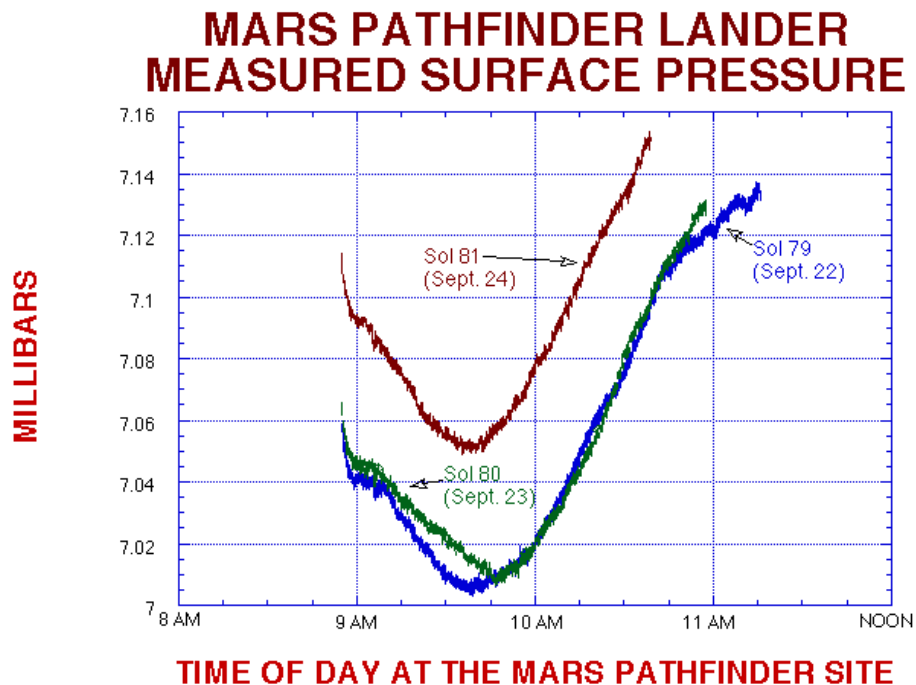


Figure 2.3: Pressure Measurements from Pathfinder

Measurement of temperature above the poles shows that there is a large condensation flow of air towards the poles each winter, and that ~25% of the atmosphere freezes out on to the

polar ice caps each year. Eventually, the dust in the atmosphere freezes over and becomes too heavy to maintain an airborne state.



Figure 2.4: Martian Polar Ice Caps

Mariner 9's (infrared spectrometer or IRIS) detected large seasonal ice caps composed of CO₂, while the perennial ice caps were considered to be formed of CO₂ in the south and possibly CO₂/H₂O ice in the north, Figure 2.4. IRIS provided global surface/atmospheric temperature maps and winds calculated from resultant temperature fields pointed to an intense polar vortex at the winter poles

The distinctive climate of Mars at aphelion is just as breathtaking with its planet wide belts of water ice clouds, which is stimulated by the distance from the sun. These 'clouds' cover the planet, especially around higher topography, at altitudes of 3 to 10 km. During this period, dust storms are confined to

altitudes of less than 10 km and without sunlight; the dust freezes and falls to the ground.

Table 2.1: Earth-Mars Comparison^{2,3}

Data	Earth	Mars
Distance from Sun	1 AU (149.6*10 ⁶ km)	1.524 AU (227.9*10 ⁶ km)
Equatorial Radius ⁴	1 (6378 km)	0.532 (3398 km)
Mass ⁴	1	0.1074
Surface Gravity ⁴	1 (9.78 ms ⁻¹)	0.38 (3.72 ms ⁻¹)
Surface pressure (millibar)	1013.25	6.9 – 10.2
Mean Density (g cm ⁻³)	5.52	3.94
Sidereal Day (hour–min)	23 hr 56 min	24 hr 37 min
Obliquity of the Ecliptic	23.4°	25.2°
Sidereal Period (year)	1 (365.256 days)	1.881 (686.980 days)
Synodic Period	-	780
Inclination to Ecliptic	0.0	1.850
Eccentricity	0.0167	0.0933
Equatorial Escape Velocity (kms ⁻¹)	11.2	5.02

² Kaye, G. W. C. and Laybe T. H., "Tables of Physical and Chemical Constants and some Mathematical Functions", Longman Group Limited, New York, 15th Edition, ISBN 0-582-46354-8, p156

³ Papike, J. J. (Editor), "Planetary Materials", Reviews in Mineralogy, Volume 26, Mineralogical Society of America, ISSN 0275-0279, p1-6

⁴ Earth = 1

Typical wind velocities of between 1 ms^{-1} to 10 ms^{-1} were measured by the Viking lander (1.6 m above ground) and were highly dependant on both diurnal and seasonal variations. Wind measurements at higher altitudes were acquired from infrared atmospheric sounding based on the temperature profiles and indicated velocities of 20 to 100 ms^{-1} at altitudes ranging from 5 to 40 km during GDS.

2.2 Atmospheric Properties

Earth based near infrared spectroscopy, fly-by missions, orbiters and landers have added to the wealth of data on the dynamic nature of the Martian environment. Between 1947 and 1963, carbon dioxide was the only known constituent of the Martian atmosphere. Shortly afterwards, the presence of water vapour was confirmed. Within ten years came the advent of ground based high resolution spectroscopy and spacecraft observations which provided evidence of CO , O_2 and O_3 as well as indications that the atmospheric pressure was more than 100 times lower than the pressure on Earth. The Viking landers expanded on the knowledge of atmospheric content by adding N_2 , NO , Ne , Ar , Kr and Xe . Isotopic ratios in all gases, except for H , were also present.

The first set of in-situ readings were provided during the descent of the Viking lander. On-board mass spectrometers on both Viking aeroshells detected molecular oxygen, atomic oxygen, carbon monoxide and nitric oxide in the upper atmosphere.

The initial pressure readings from the first Viking lander indicated 7.65 millibar (mbar) but that figure decreased with time. The maximum readings were found to have reached 9 mbar at the first Viking site and 10 mbar at the second Viking site. Precise readings of the minor constituents were difficult to ascertain. This variation was attributed to the deposition and sublimation of CO_2 which itself is dictated by the season.

Table 2.2 shows the composition of the lower atmosphere of Mars, as detected and recorded by the Viking landers.

Table 2.2: The Composition of the Lower Martian Atmosphere⁵

Ranking	Gas	Abundance
1	CO_2	95.32%
2	N_2	2.7%
3	^{40}Ar	1.6%
4	O_2	0.13%
5	CO	0.07%
6	H_2O	0.03 ⁶ %
7	$^{36+38}\text{Ar}$	5.3ppm
8	Ne	2.5ppm
9	Kr	0.3ppm
10	Xe	0.08ppm
11	O_3	0.04-0.3ppm

⁵ Owen, T., "Composition and Early History of the Atmosphere", p824, in Kieffer, H. H., Jakosky, B. M., Snyder, C. W., Matthews, M. S., (Editors), "Mars", The University of Arizona Press, ISBN 0-8165-1257-4, 1992, pp1498

⁶ Variable with season and location

The primary gases of interest to CO₂/metal powder propulsion engineers will be the acquisition and storage of CO₂ and N₂; CO₂ as the oxidiser component of the propellant and N₂ as a possible pressurising gas. Ar is also of interest to the production of metal powders. This is addressed in a later section.

The gases in the homopause, which is the region of the atmosphere up to ~125 km, are well mixed due to the eddy diffusion. The average molecular weight is 43.4 amu and this figure can be used for barometric formulas. The gases in heterosphere, which is the atmospheric region above the homopause, follow their own barometric formula and as a result, the lighter gases such as hydrogen and helium are found in the upper atmosphere, above 500km⁷.

Important factors associated with the Martian atmosphere;

- 1) the primary constituent of the atmospheric gas, CO₂, takes part in a process of condensation and sublimation
- 2) diurnal temperature swings can differ up to ~100K
- 3) there is no maximum temperature in the middle of the atmosphere (on Earth, such a maximum is caused by the ozone)
- 4) the aerosols (dust and water ice particles) have a greater stimulus on atmospheric heating
- 5) evaporation and condensation of water has little effect of the redistribution of heat

2.3 Surface Mineralogy

Two spectrally dissimilar regions dominate the Martian surface;

- 1) high albedo regions understood as consisting of extremely eroded aeolian dust.
- 2) low albedo regions interpreted as dark soils, blocks of bedrock and in-situ bedrock.

For a reason that is not yet fully understood, craters of varying sizes dominate the features of the southern hemisphere. The northern hemisphere includes several large volcanoes and volcanic plains. The first three Mariner fly-bys (4, 6 and 7) returned images primarily of the southern hemisphere, presenting a false impression of the entire surface geology. Mariner 9 provided a more realistic characterisation. The Viking orbiters and landers, Pathfinder, Mars Global Surveyor (MGS) and Mars Odyssey have all added to the wealth of knowledge on Martian geology and ESA's Mars Express promises to continue the work of its predecessors.

The data required to determine the mineralogical distribution over the Martian surface is being acquired by the gamma ray spectrometer (GRS) on board the Mars Odyssey, which is, at the point of this study, mapping the surface of Mars. The pertinent information will be available towards the end of 2003. Initial indications show that most elements, apart from Hydrogen, are more or less evenly distributed on the surface of Mars⁸. The most recent data available on chemical make-up of the Martian surface was obtained by Pathfinder's alpha proton X-ray spectrometer (APXS) instrument.

The previous orbiter, the Mars Global Surveyor, employed a thermal emission spectrometer (TES), which afforded an opportunity to map certain surface areas of mineralogy, but not the entire face of Mars. Also, the TES instrument could only read mineralogical content of the top 10 microns of the surface and it was not able to provide sub-surface measurements. Thus, data obtained to date is not sufficient to present an accurate picture of the mineralogical distribution over the surface of Mars, but general geological knowledge does present an opportunity to gauge the types of minerals that can be found in specific locations.

Prior to Pathfinder, two sources of data existed on the surface chemistry of Mars; 1) Viking 1 and Viking 2 landers and 2) Shergottite-nakhlite-chassignite (SNC) meteorites. The Viking landers set down ~6500km from each other and data recorded proved to be relatively similar. The Pathfinder data was acquired with an instrument known as the APXS and conducted in-

⁷ Moroz, V., "Mars : Atmosphere", Encyclopaedia of Astronomy and Astrophysics", Nature Publishing Group, 2001, p1

⁸ Boynton, William V., Principal Investigator, GRS Instrument, Personal Communication, 14th August 2002

situ readings of six soil and five rock samples. After readings of the pertinent elements were made, the total concentrations were normalised to 100%.

Table 2.3: Mineralogical Composition at the Pathfinder Sites

Name	Na ₂ O	MgO	Al ₂ O ₃	SiO ₂	SO ₃	Cl	K ₂ O	CaO	TiO ₂	FeO	Org. sum
Soils											
A-2 After deploy	2.3±0.9	7.9±1.2	7.4±0.7	51.0±2.5	4.0±0.8	0.5±0.1	0.2±0.1	6.9±1.0	1.2±0.2	16.6±1.7	68.6
A-4 Next to Yogi	3.8±1.5	8.3±1.2	9.1±0.9	48.0±2.4	6.5±1.3	0.6±0.2	0.2±0.1	5.6±0.8	1.4±0.2	14.4±1.4	78.2
A-5 Dark next to Yogi	2.8±1.1	7.5±1.1	8.7±0.9	47.9±2.4	5.6±1.1	0.6±0.2	0.3±0.1	6.5±1.0	0.9±0.1	17.3±1.7	89.1
A-8 Scooby Doo	2.0±0.8	7.1±1.1	9.1±0.9	51.6±2.6	5.3±1.1	0.7±0.2	0.5±0.1	7.3±1.1	1.1±0.2	13.4±1.3	99.2
A-10 Next to Lamb	1.5±0.6	7.9±1.2	8.3±0.8	48.2±2.4	6.2±1.2	0.7±0.2	0.2±0.1	6.4±1.0	1.1±0.2	17.4±1.7	92.9
A-15 Mermaid Dune	1.3±0.7	7.3±1.1	8.4±0.8	50.2±2.5	5.2±1.0	0.6±0.2	0.5±0.1	6.0±0.9	1.3±0.2	17.1±1.7	98.9
Rocks											
A-3 Barnacle Bill	3.2±1.3	3.0±0.5	10.8±1.1	58.6±2.9	2.2±0.4	0.5±0.1	0.7±0.1	5.3±0.8	0.8±0.2	12.9±1.3	92.7
A-7 Yogi	1.7±0.7	5.9±0.9	9.1±0.9	55.5±2.8	3.9±0.8	0.6±0.2	0.5±0.1	6.6±1.0	0.9±0.1	13.1±1.3	85.9
A-16 Wedge	3.1±1.2	4.9±0.7	10.0±1.0	52.2±2.6	2.8±0.6	0.5±0.2	0.7±0.1	7.4±1.1	1.0±0.1	15.4±1.5	97.1
A-17 Shark	2.0±0.8	3.0±0.5	9.9±1.0	61.2±3.1	0.7±0.3	0.3±0.2	0.5±0.1	7.8±1.2	0.7±0.1	11.9±1.2	78.3
A-18 Half Dome	2.4±1.0	4.9±0.7	10.6±1.1	55.3±2.8	2.6±0.5	0.6±0.2	0.8±0.1	6.0±0.9	0.9±0.1	13.9±1.4	92.6
Calculated "soil-free rock"	2.6±1.5	2.0±0.7	10.6±0.7	62.0±2.7	0	0.2±0.2	0.7±0.2	7.3±1.1	0.7±0.1	12.0±1.3	

The results in Table 2.3 were normalised to 98% due to variations in the measurement geometry and data for P, Cr and Mn exhibited large errors that were not factored into the analysis. It was found that the data acquired from the Pathfinder site, Ares Vallis, was similar to the Viking measurements, but with slight differences. In order to provide a comparison, each element was recalculated to 100% by weight and was then normalised to 44%, the weight of silica, Figure 2.5.

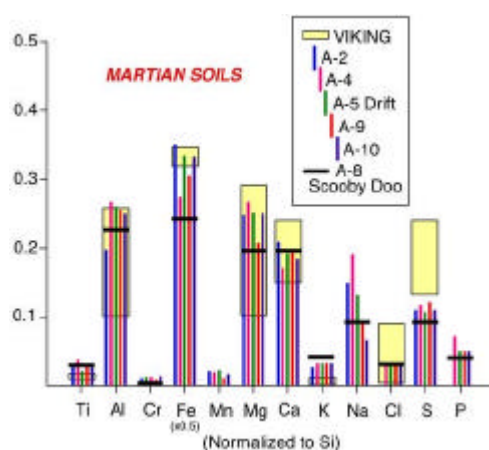


Figure 2.5: Mineral Content Comparison between Pathfinder and Viking

2.4 Introduction to Martian Geology

The focus of interest regarding potential in-situ fuels on Mars is the metals that burn energetically with CO₂ oxidiser. Previous studies on CO₂/metal powder combustions have shown that the two metals which exhibit the most promise are magnesium and aluminium.

Table 2.4: Common Minerals in Volcanic Rocks

Light-coloured minerals (formula)		
Feldspar:		
Plagioclase		(CaAl,NaSi)AlSi ₃ O ₈
Orthoclase		KAlSi ₃ O ₈
Quartz		SiO ₂
Nepheline		NaAlSi ₃ O ₄
Dark-coloured minerals (formula)		
Olivine		(Mg,Fe) ₂ SiO ₄
Pyroxene:		
Hypersthene		(Mg,Fe)SiO ₃
Pigeonite		(Mg,Fe)SiO ₃
Augite		Ca(Mg,Fe)Si ₂ O ₆ (Al,Fe) ₂ O ₃
Biotite		K(Mg,Fe) ₃ AlSi ₃ O ₁₀ (OH) ₂
Magnetite		Fe ₃ O ₄
Hematite		Fe ₂ O ₃

Spectral signatures of igneous rock were partially hidden by layers of dust making it difficult for remote sensing and in-situ observations to gain an accurate reading. However, it is generally realised that basaltic lava covers the surface of Mars, which is rich in aluminium, calcium, iron, magnesium and silicon⁹. Still there is enough data to provide a general indication on the mineralogical content of Martian regolith and rocks.

Silicate minerals dominate the surface of Mars. Due to their high magnesium content, olivine and pyroxenes are of most interest. Olivine, (Mg,Fe)₂SiO₄, is a rock forming mineral which is

believed to have been formed at great depth. The forsterite series contain primarily magnesium and fayalite, primarily iron. Olivine in the majority of the stony meteorites that have fallen to Earth contain on average 60 to 85% forsterite and the remainder, fayalite. Pyroxenes are predominantly dark coloured rock-forming minerals, which are universally found on the Earth and in meteorites. The spectral variation acquired from the TES is graphically illustrated in Figure 2.6.

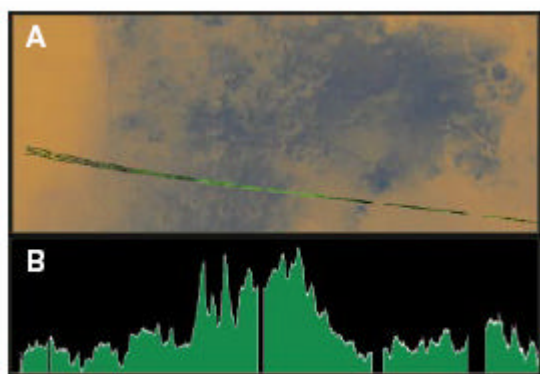


Figure 2.6: Spectral Imaging by TES (MGS) of Pyroxene Content on the Surface of Mars

Figure 2.6A indicated the area examined during the pass of the MGS and Figure 2.6B shows the intensity of the pyroxene absorption. Orthopyroxenes are a subcategory of pyroxenes and are mainly found in meteorites. The magnesium rich minerals that future mining operations will find of interest, in descending order, are enstatite Mg₂Si₂O₆, bronzite (Mg,Fe)SiO₃, hypersthene (Mg,Fe)SiO₃, orthoferrosilite FeSiO₃ and clinopyroxenes which are in the form of pigeonite (Mg,Fe,Al)(Mg,Fe)Si₂O₆, diopside CaMg(SiO₃)₂ and augite (Ca,Na)(Mg,Fe,Al)(Si,Al)₂O₆.

Bronzite, one of the minerals containing a high quantity of magnesium, has been detected by the TES instrument during studies of the Hebes Chasma and was found to constitute¹⁰ 7% of the reading. The remaining minerals detected during TES data acquisition were augite (45%), plagioclase feldspar (12%) and atmospheric dust (36%).

⁹ Ruff, S., Mars Spaceflight Facility, Department of Geological Sciences, Arizona State University, Personal Communication, 5th September 2002

¹⁰ Christensen, P. R., et al., "Results from the Mars Global Surveyor Thermal Emission Spectrometer", Science, Volume 279, 13 March 1998, p1694

3.0 CO₂ Production

3.1 Introduction to CO₂ Production

Literally every 'Martian' in-situ oxidiser production method initially involves the manufacture of CO₂. The CO₂ is acquired, filtered and then compressed to relevant temperatures and pressures where it is condensed into a liquid state. Here it is either stored in a liquid form or is fed to either;

1. a reverse water gas shift (RWGS) device in collaboration with a zirconia or water electrolyser where the device thermally disassociates the CO₂ in order to produce carbon monoxide (CO) and oxygen (O₂). These devices entail high-energy inputs, function at very high temperatures, are not very efficient in terms of CO production and are subject to single point failures¹. The RWGS is also still in its experimental stages and it will be some time before a fully operational device is ready for assignment.
2. a sabatier reactor catalytically converts CO₂ and H₂ into CH₄, H₂O and heat and practical experimentation has proven it to operate at high efficiencies and required no additional heat input after initial start-up². CH₄ and LOX manufacture is the most widely discussed in-situ propellant because of its high specific impulse value.

These methods were developed for the production of CO-LOX or CH₄-LOX propellant combinations and therefore the compressor designs tend to deliver a pressure a fraction of what is actually necessary for the liquefaction of CO₂.

As mentioned in the previous section, the pressure and temperature on Mars fluctuates quite extensively, in extreme situations. The temperature varies from 148 K to 296 K, and the pressure can swing from about 6 millibar to 10 millibar, depending on the location. One proposed method of CO₂ production is to employ an adsorption pump which takes advantage of this diurnal temperature fluctuation to draw in CO₂ and process the gas through an adsorbent. The production rate is dependant on the adsorbent material plus the quantity of the material and can be quite low, which makes it suitable to longer duration stays on the Martian surface. The second method, but one that is far more energy dependent, is the mechanical roughing pump which compresses the CO₂ to relevant temperatures and pressures. Its production rate is suited for short duration missions where a lot of CO₂ is necessary in a short space of time and it can operate around the clock, if required, but this is subject to the method of energy production. The use of solar arrays would limit a pump's use to daytime operation and this method of power generation would not be included in a mission scenario where large quantities of CO₂ would be required in a limited period of time. A third method is a refrigerator which collects CO₂ from the Martian atmosphere by freezing it, collecting it in a container and then heating up the contents until the CO₂ liquefies.

3.2 Atmospheric Dust

The Martian atmosphere contains minute dust particles ranging from 1 µm to 10 µm in diameter and such impurities can not only damage the compressor's components or clog up the adsorption bed, but also lead to a degradation of performance by introducing impurities to the propellant formulation. A filter would ideally be connected to the intake of either the roughing pump or the adsorption bed. Bruckner et al. propose placing a pleated filter prior to the entrance of the intake in order to remove the larger dust particles and a membrane filter at the rear end to remove the sub-micron particles³. A method for removing the obstructing particles from the filter will also have to be devised otherwise the overall CO₂ production system's performance will diminish with time, more so if the landing site of CO₂ plant experiences a dust devil or worse, a GDT.

¹ Bruckner, A. P., Thill, B., Anderson, S., Pasco, D. and Willman, C., "Mars In Situ Propellant Technology Demonstrator Mission", Journal of the British Interplanetary Society, Volume 48, pp338, 1995

² Kleiner, G. N., and Cusick, R. J., "Development of an Advanced Sabatier CO₂ Reduction System", ASME 81-ENAS-11, Intersociety Conference on Environmental Systems, San Fransisco, CA, 1981, pp7

³ Bruckner, A. P., Thill, B., Anderson, S., Pasco, D. and Willman, C., "Mars In Situ Propellant Technology Demonstrator Mission", Journal of the British Interplanetary Society, Volume 48, pp33, 1995

Ash et al., state that for their in-situ propellant production concept, it was estimated that for 140 days of operation, ~200 grams (60 cc's) or ~1.43 grams per day, of dust will have to be removed from the atmospheric intake. The pressure drop through the atmospheric filter in the Martian environment can be estimated by⁴;

$$\Delta P = 0.12 \left(\frac{Q}{A_F} \right) \text{ in millibar} \quad (3.1)$$

where Q - flow rate of the atmospheric fluid, litres/second

A_F - surface area of the filter, m²

A filter with a surface area of 22 m² will experience a pressure drop of ~0.4 mb. This pressure drop is relatively small and acceptable and coupled with the low atmospheric pressure, a filter with a throughput volume of 25 litres will result in a filter mass of 5 kg.

3.3 Adsorption Pumps

3.3.1 Definition of Adsorption

"Adsorption is the process of trapping gas and vapour molecules within the pores of a microporous solid such as activated carbon⁵"

3.3.2 Introduction to Adsorption

Adsorption techniques find employment in many terrestrial applications such as the use of bone char for decolourising foods to the removal of nerve gas from battlefields where activated carbon was used as the adsorbent material. Nowadays, adsorption techniques are used for industrial processes such as water purification. In fact, the common household water filter is but one example of an everyday use of adsorption where chlorine molecules are trapped and 'stored' in the adsorbent material, releasing only pure H₂O and, of course, bacteria. For Mars based applications, the common adsorbents are zeolite and, less often, activated carbon. Such devices have a finite lifetime.

3.3.3 Adsorption Techniques

When uncontaminated sources of fluids are required, the adsorption technique is an ideal process. The diurnal temperature fluctuations, on average 70 K, provide the ideal setting for a process that separates gases during the night and then rejuvenates during the day. The difference in loading, i.e. quantity of gas that can be stored in the adsorption material during the day and night, is close to the theoretical limit for specific adsorbents and the devices can be considered to operate at very high efficiencies. Adsorption pumps are generally regarded as comparable, if not superior, to single stage compressors operating in the same environment⁶ but it is unlikely that a single stage compressor will be able to compress atmospheric CO₂ from ambient pressures, 1000 times to 10 bar.

⁴ Ash, R. L., Richter, W., Dowler, W. L., Hanson, J. A. and Uphoff, C. W., "Autonomous Oxygen Production for a Mars Return Vehicle", IAF-82-210, 33rd Congress of the International Astronautical Federation, September 27 – October 2, 1982, p8

⁵ <http://www.sparkstech.com/products/defadsorp.html>

⁶ Finn, John E, Sridhar, K. R. and McKay, Christopher P., "Utilisation of Martian Atmosphere Constituents by Temperature-Swing Adsorption", Journal of the British Interplanetary Society, Volume 49, pp 423-430, 1996

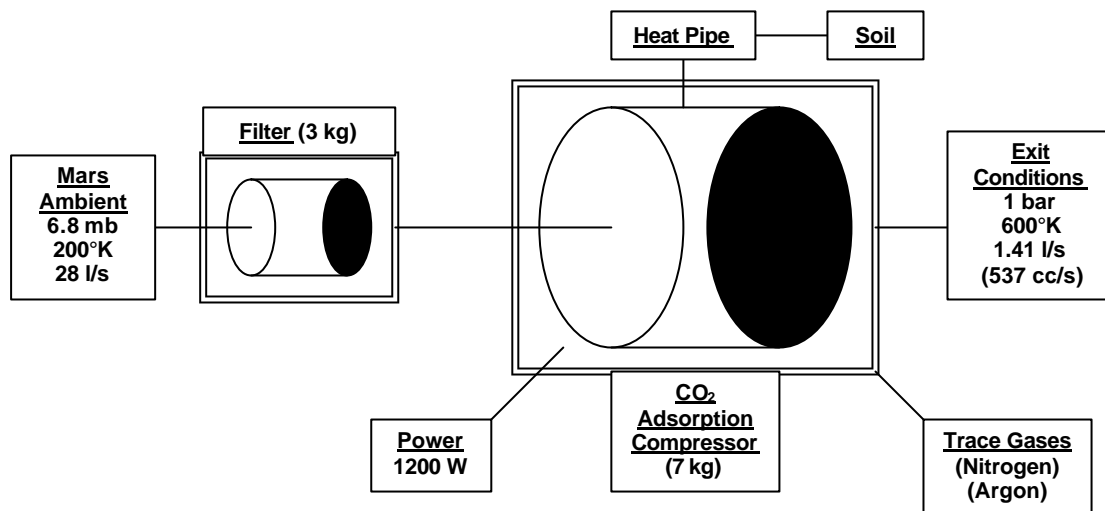


Figure 3.1: Filter and Compressor Schematic⁷

Studies by Martin Marietta, now Lockheed Martin, showed that an activated carbon 'sorption' pump could theoretically adsorb ~40% of the mass of the sorbent material. Actual experimental figures show adsorption to equal between 5 to 15% of the sorbent mass. Therefore, it was determined that in order to produce 1 kg of CO₂ per day, 2.5 kg of sorbent material would be required and that 50 W of power would be required to encourage out gassing at high pressure⁸. By employing Solar Arrays, the available power is dependant on the position of Mars. At Aphelion, the Solar Flux is ~500 Wm⁻² but at Perihelion, the Solar Flux is ~730 Wm⁻². The CO₂ could then either be fed directly into a refrigerated storage tank. The ~5% trace gases such as Nitrogen, Argon etc. will not liquefy under the same environmental conditions as CO₂ and could therefore be disposed off by occasional venting of the CO₂ storage tank. If the CO₂ is compressed to values of 10 bar and then allowed to condense to ambient Martian temperatures, then these trace gases can be bled off relatively easily. Nitrogen is another in-situ fluid that could find a useful application on CO₂/metal powder fuel engines in the form of a pressurant to feed the powder from the fuel tank to the chamber. However, the minimal mass and therefore cost saving of transporting a small quantity of Nitrogen for use in a propulsion system may not warrant the additional hardware required to collect and store the gas. This trade off is only possible once the engine and its required propellant quantities has been sized. Argon is also a useful in-situ gas as it is often used instead of helium as a pressurised compressing gas for the production of magnesium metal powder and therefore could also be retained.

One negative aspect of this system is that the solid sorbent material is brittle and will have to be well protected during entry, landing and descent. Another problem that is envisaged with the operation of such a system is that with the increasing CO₂ requirement, the adsorption bed will have to increase in size. The adsorbent mass will therefore take longer to cool down. A way around this problem would be to connect multiple smaller adsorption beds together but this will increase the inert mass.

3.4 Mechanical Roughing Pumps

The triple point of CO₂ is 5.2 bar at STP. Under Martian conditions, the liquefaction parameters imply that the pressure must be raised to ~10 bar and then allowed to equilibrate to ambient temperature where it will condense into a liquid state. The liquid CO₂ will then be

⁷ Frisbee, R. H., French Jr., James R., Lawton and Emil A., "A New Look at Oxygen Production on Mars – In-Situ Propellant Production (ISPP)", AIAA 87-0236

⁸ Zubrin, R., "Diborane/CO₂ Rockets for use in Mars Ascent Vehicles", Journal of the British Interplanetary Society, Volume 48, p389, 1995

redirected to the MAV oxidiser tank where a limited amount of energy input will allow it to maintain its liquid state. In order to purify the CO₂ and rid the fluid of impurities, i.e. remaining dust particles, the CO₂ could be vaporised and then distilled but this adds complexity to an already complex system.

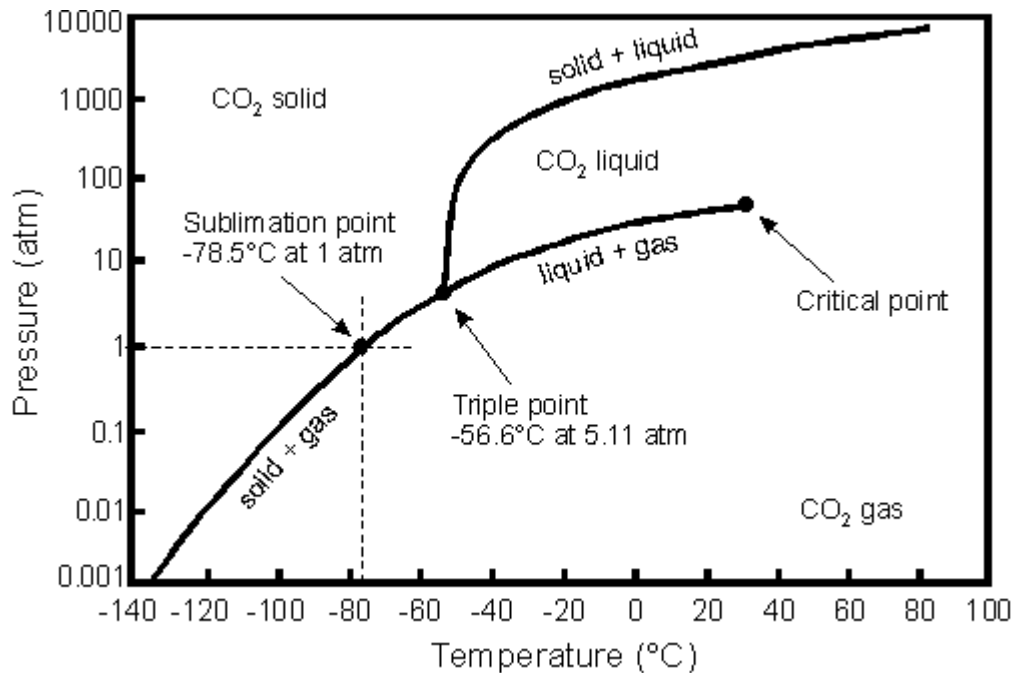


Figure 3.2: P-T Phase Diagram for CO₂

Ash et al. state that in order to collect 1 kg of CO₂ per day, 1.22 litres of atmosphere must be collected per second. They mention that laboratory vacuum pumps with a power consumption of ~700 W are capable of this throughput but the mass of such terrestrial devices are on the order of 50 kg.

In addition to the task of raising pressures from 6.9 mb to ~10 bars, the pump's compressor will also have to contend with overcoming the pressure drop through the filter, Equation 3.1. The quantity of stages for the compressor is dependant on;

1. compression ratio
2. compressor materials
3. allowable tolerance levels
4. efficiency

As an initial assumption, each compressor stage is considered to be four times the volume of the rotor.

$$V_r = 0.385(Q_s^{1.5}) \quad (3.2)$$

where Q_s - volume flow rate at the stage inlet (litres/second)

If the compressor's volume consists of 75% solid material with an estimated density of 7.5 g/cc, the mass of the compressor will be on the order of 6 kg.

One of the concerns about this system is the fact that involves the uses of turbomachinery, i.e. rotating components, which raise issues with regards to reliability. Still, it is the only device with heritage, albeit not space based, that could provide the required atmospheric throughput to produce quantities of CO₂ in a short space of time. But, one must be realistic. The rotational velocity, at which the compressor must operate in order to draw in and compress the very thin atmosphere while keeping in mind that minimising size hence mass and

durability over the required period of operation are essential, will prove to be a task within itself. The upper level density of the atmosphere on Mars at ground level is equivocal to the density of the atmosphere on Earth at about 100,000 ft (30,480 m). Military jets, except one or two advanced spy planes, don't even fly at this altitude and even then, those that do travel at supersonic velocities where in-situ propellant production (ISPP) plants will remain static, on the surface of Mars. Suffice to say, military jet engine manufacturers will be one of the best sources to engineer such products. In order to bring the Martian ambient pressures to CO₂ liquefaction temperature and pressures, the pressure ratio must be almost 1000. A multistage compressor and a resulting large power requirement will only achieve this. It is important to note that such technologies do not exist outside of the military environment.

3.5 CO₂ Freezer/Collector

An alternative concept is the CO₂ freezer/collector but this concept was designed as a component of a CH₄ manufacturing system. This proposed system could be adapted for a CO₂-collection-only scenario. During aphelion winter nights, the temperature can drop as low as 148 K. Using a baseline average temperature as 210 K, the Ash et al. calculated that ~80 W is required to cool down CO₂ gas into the form of a solid, ~76 W of which is required for phase change. The power estimates were deducted from the assumption that the wind speed is 0.1 ms⁻¹ and the known value for the latent heat of sublimation (598.6 kJ/kg).

Other assumptions and operating conditions were as follows;

1. the collector tube diameter is 1cm
2. heat transfer coefficient for convection is between 1 and 8 Wm⁻² K
3. flow rates vary from 1m⁻¹ to 10m⁻¹
4. full range of pressures and temperatures
5. radiation of 60Wm⁻²
6. atmospheric gas is homogenous

The system uses a coolant, which passively freezes the CO₂. The choice of coolant and its temperature as well as the collector's density and thermal conductivity are strong factors that affect the collector's performance. Since the concept was initially designed to produce Methane, the researchers simply used a small quantity of the Methane they produced to feed through the collector's coils. The boiling heat transfer for Methane is ~2500 Wm⁻² K and assuming that it boils off at 130 K, a calculated 91 W could be produced from a 1 ms⁻¹ wind speed. The CO₂ can then be scraped off the cooling coil, compressed to a pressure of ~70 bars fed into a storage tank, provided of course the CO₂ is required to be stored in the form of a solid. During the CO₂ frost extraction process, the introduction of heat could raise the temperature to ~218 K where the frost would liquefy and subsequently be redirected to a liquid CO₂ tank. Approximately 40 W of power would be required to heat the frost to liquefaction temperatures.

3.6 Mars Atmospheric Carbon Dioxide Freezer (MACDOF)

Mechanical roughing pumps require power, moving parts and must operate without maintenance over long periods of time. Depending on the method of generating power, i.e. solar arrays, the pump may only be operable for 50% of the Martian day. Sorption pumps use a bed of active carbon or zeolite, on the other hand, are heavy, brittle and can operate once a day.

Faced with these problems, Pioneer Astronautics, headed by Dr. Robert Zubrin, developed a system which they referred to as a Mars atmospheric carbon dioxide freezer or MACDOF. This system was designed as a component of a larger system to produce CO and LOX with the baseline NASA Johnson Space Center's human mars mission as a reference. This mission calls for the production of 100 kg of CO₂ per day. Therefore, their logic in some cases does not apply to the production of just CO₂.

In principal, the MACDOF actively freezes carbon dioxide from the atmosphere and produces on the order of 700 grams per day. It is less massive than a *passive* adsorption pump and yet, it has twice the capacity and 40% the storage capability. The experimental module was built out of steel whereas the adsorption pump was made from aluminium. It was determined that if the MACDOF was also made out of aluminium, then the mass would have been ¼ of the adsorption pump.

An Adsorption bed can store on the order of 10% of the adsorbent mass whereas the MACDOF is capable of storing 100% dry ice. Based on the logic of equal capacities, the freezer can store up to five times that of an adsorption bed. The freezer material is either diffuse copper or aluminium wool is much less dense than the adsorbent material. For example, copper wool is ~240 grams per litre which is over three times less dense than activated carbon as a adsorbent material. The MACDOF is capable of acquiring a consignment of CO₂ every five hours which, based on the use of solar arrays, affords an opportunity to collect CO₂ twice per day, at best. The hours after sunrise and prior to sunset may reduce the solar flux so ten hours, or twice a day, may be somewhat of an over exaggeration. This system would more likely be able to produce CO₂ 1.5 times per day, which is still a 50% increase over passive or active adsorption pumps.

Pioneer takes the position that a nuclear reactor will power the in-situ propellant production (ISPP) system and therefore such a system will be able to operate at least four times a day. So, with 1/5th the volume, 1/3rd the interior density and 1/4th the cycle, this works out to be 1/60th the mass of an adsorption pump and the shell is assumed to be 1/20th the mass of equivalent capacity. For of a low-cost sample return mission, the employment of a nuclear power plant is out of the question and solar arrays will be the most likely form of power generation. At worst case, the production rate of one times a day relates to 1/15th the capacity of the adsorption pump of equal capacity; which is still a remarkable improvement.

During Pioneer's experiments, 300 grams of CO₂ was produced in a 1-litre bed, which was subsequently rejuvenated over the five-hour period. Approximately 400 W of power was required for this operation. The adsorption pump was found to require a 20 litre capacity bed to produce the same quantity. For a CO₂ production capacity of 300 gram per day or less, it was found that the adsorption pump was less massive than the MACDOF. But, for CO₂ quantities of up to 4 kg/day, the copper wool filled, solar powdered MACDOF proved to be the superior performer. During these comparisons, the supporting hardware for the MACDOF, such as refrigerators, heat sinks and heating units were not included. Also, the experiments were developed to test systems and were not optimised for mass and power consumption.

As with terrestrial refrigerators, electric power is used to dispose of thermal energy from the freezer to ambient temperature. The refrigerator will also have to be thawed, to clear its contents following one production cycle. This could be performed via solar heating during the Martian day but active heating would increase the turn-around time. The power required for freezing depends on the heat rejection temperature;

$$b = \frac{T_C}{T_H - T_C} \quad (3.3)$$

where β Coefficient of Performance
 T_H Heat Rejection Temperature
 T_C Freezing Temperature

4.0 Metal Mining

4.1 Introduction to Metal Mining

Extracting metal, in-situ, for subsequent processing into metal powder for use in propulsion systems will be an extremely difficult and costly task, in terms of energy requirement. As mentioned previously, the most likely metal fuel for use in CO₂ engines is magnesium so the focus of work in this section is on the production of this element.

Terrestrial magnesium oxide is produced from three sources;

1. Magnesite (magnesium carbonate)
2. Seawater
3. Magnesium Chloride-rich Brine

Magnesite is found in concentrations of sedimentary deposits. When heated to 973.15K – 1273.15K, the magnesium carbonate undergoes thermal decomposition to produce magnesium oxide and carbon dioxide. Both seawater and underground sources of brine, located at depths of more than 760 metres below terrestrial ground, contain high contents of magnesium chloride.

As one can see, terrestrial technologies for the production of magnesium on Mars are totally defunct and alternative methods will have to be sought.

Magnesium is estimated as being 4 to 5% content of the Martian regolith and rocks on Mars, see Table 4.1, but to date, there is no evidence of large concentrations like those found in natural deposits on Earth. Perhaps, the GRS instrument on the Mars Odyssey which is presently mapping the surface of Mars or ESA's Mars Express will prove this initial assumption wrong but for now, future production of magnesium on Mars will have to make do with current knowledge that there is an equal distribution of the metal throughout the surface of Mars.

Table 4.1: Elemental Percentages from Pathfinder's Alpha Proton X-ray Spectrometer

Element	A-2, Soil	A-4, Soil	A-5, Soil	A-3, Rock "Barnacle Bill"	A-7, Rock "Yogi"
	weight %	weight %	weight %	weight %	weight %
Carbon [C]	-	-	-	-	-
Oxygen [O]	42.5	43.9	43.2	45.0	44.6
Sodium [Na]	3.2	3.8	2.6	3.1	1.9
Magnesium [Mg]	5.3	5.5	5.2	1.9	3.8
Aluminium [Al]	4.2	5.5	5.4	6.6	6.0
Silicon [Si]	21.6	20.2	20.5	25.7	23.8
Phosphorus [P]	-	1.5	1.0	0.9	0.9
Sulphur [S]	1.7	2.5	2.2	0.9	1.7
Chlorine [Cl]	-	0.6	0.6	0.5	0.6
Potassium [K]	0.5	0.6	0.6	1.2	0.9
Calcium [Ca]	4.5	3.4	3.8	3.3	4.2
Titanium [Ti]	0.6	0.7	0.4	0.4	0.5
Chromium [Cr]	0.2	0.3	0.3	0.1	0.0
Manganese [Mn]	0.4	0.4	0.5	0.7	0.4
Iron [Fe]	15.2	11.2	13.6	9.9	10.7
Nickel [Ni]	-	-	0.1	-	-

Based on current knowledge, magnesium can only be mined and processed in two ways on Mars.

1. Collection and processing of rocks and
2. Collection and processing of regolith

The process used to produce magnesium from magnesite is probably the closest type of technology that could be employed for extracting magnesium from ores for subsequent processing into magnesium metal powder on the Martian surface. Consultation with terrestrial specialists in producing magnesium from magnesite, for example Grecian Magnesite¹, will be required in order to define the best process for producing magnesium from ores.

4.2 Magnesium Manufacturing & Processing

Rocks containing the desired metal, in our case magnesium, are referred to as 'ores' and terrestrially, the selection of ores is dependant on the percentage concentration of the metal plus the economical and technical feasibility of mining and processing the ore. This logic is non-applicable in a Martian mining environment due to the fact that there are no known concentrated deposits of specific minerals. It is therefore safe to assume that so long as the ore contains the average quantity of the metal of interest, then it will have to suffice. Metals are usually found in the form of minerals, not in their natural chemical state and can be a compound in the form of either sulphides or oxides. A complete list of minerals containing magnesium is listed in Appendix A. It is more than likely that in specific ores, a multitude of other minerals will be available but will have no direct value for use as a fuel. In terrestrial applications, such minerals are disposed off but in the case for a mining operation on Mars, it could prove worthwhile to stockpile the other minerals for future use. Intensity of the mineral content can vary from ore to ore. For example, if one were seeking titanium, then the ore could contain ~1% of the element, whereas for aluminium, that ore could hold on average 25% and iron ores could include over 60%.

The three stages required to 'convert' the ore to a mineral are as follows;

1. Beneficiation: This is where the ores are crushed to fine grains and the specific techniques are employed to separate the 'useful' minerals from those that are considered 'useless'.
2. Smelting: The minerals of interest are then fed into a furnace where, at extremely high temperatures, the metal is removed from the slag, or waste.
3. Refining: This process is used to remove impurities from the metal, resulting in a near-pure product.

The sheer complexity of producing terrestrial magnesium from magnesite is shown in Figure 4.1 but without additional knowledge of which minerals can be found in the ore as well as the concentrations, it is not possible to design a plant to manufacture a specific metal².

An alternative methodology to the first part of beneficiation, i.e. crushing ores, is to simply collect the fines, or regolith, on the Martian surface, which have already been crushed by nature in readiness for collection. This approach would be favourable due to the fact that it would alleviate the energy requirements to crush and produce fine powder. A mobile collection device could house a detector to search for magnesium rich areas prior to regolith collection. This would avoid the system having to process 'redundant' metals.

¹ <http://www.grecianmagnesite.com/>

² Friese-Greene, T., Technology Group Executive, Rio Tinto, Personal Communication, 6th September 2002

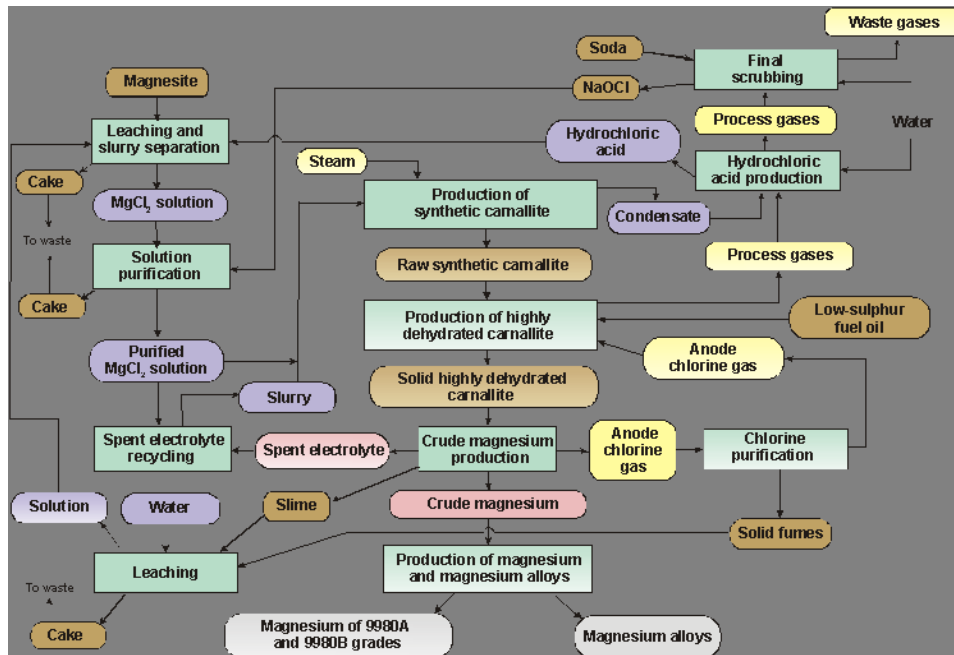


Figure 4.1: Terrestrial Processes for Mass Production of Magnesium³

No doubt, additional work will be required in this area but only once the data from present and future spacecraft and landers have provided the pertinent information required to interpret the information to a form that can be employed for the purpose of mining resources on the Martian surface. To date, only educated estimates and speculations on mineralogical content can be made from present information, which was intended for planetary geology and not mining. Specialists in this field are few and far between, but recently, a space resource roundtable was established to encourage a community of interested persons in order to share and disseminate knowledge in this area⁴.

4.3 Metal Powder Production and Safety

The basic characteristics of metal powders are;

1. Chemical composition
2. Chemical purity
3. Particle size and size distribution
4. Particle porosity
5. Particle microstructure

The standard sieve sizes are shown in Table 4.2.

Table 4.2: Comparison between Imperial (Old English) and SI Sieve Sizes

Sieve No. (Mesh/Linear Inch)	Aperture (Micron)
100	149 ± 6%
120	125 ± 6%
140	105 ± 6%
170	88 ± 6%
200	74 ± 7%
230	62 ± 7%
270	53 ± 7%
325	44 ± 7%

³ http://www.vami.ru/processes/magnesium/sposob_proizvod_magnia_is_oksidnogo_siria.htm

⁴ Duke, M., Colorado School of Mines, Personal Communication, 2nd September 2002

4.3.1 Chemical Characteristics of Metal Powder

It is not a technological challenge to produce metal powders of 90% purity. In some cases, 97% purity may be preferred to 99% purity because the latter have been known to be more abrasive in nature than the former. This is an undesirable feature for engines that require long duration or multiple burns and should be avoided simply because the engine's components would suffer internal degradation resulting in a reduction in performance.

4.3.2 Preparation of Metal Powders

There are multiple ways of producing metal powders;

1. Reduction of oxides and other compounds
2. Electrolysis
3. Atomisation
4. Mechanical comminution (crushing, milling etc.)
5. Coldstream Process

For terrestrial applications, atomisation and reduction techniques are used for high tonnage production whereas the other methods are used for small quantities of specialist powders.

4.3.2.1 *Reduction of Oxides and other Compounds*

This is the most widely used terrestrial method of producing metal powder. The metal is heated to temperatures just below the melting point and the reaction takes place just below the melting points of the oxides and metals. Carbon monoxide is introduced into the procedure and acts as a reducing agent. The powders produced by this process are spongy agglomerates and are usually angular and granular shaped. The agglomerates can be comminuted to fine powders by mechanical means.

This method exhibits a lot of flexibility with regards to the delivery of the final product. By altering the shape and size of the oxide, the reaction temperature (lowering temperature reduces particle size), the type of reducing agent and the rate of flow of the reducing agent, the final particle's shape, size and density can be affected. Oxides in general are brittle in nature and can be easily pulverised to produce very fine metal powder. The major disadvantage of this technique is that there is no refining involved and therefore any impurities will remain in the powder.

4.3.2.2 *Electrolysis*

This method is where metal powders are produced by electro deposition from a solution of fused salts. It is the method that is used for commercial production of iron, copper, nickel and zinc. By controlling the current density, temperature, composition and circulation of the solution, the bath's size and electrode arrangement, powder characteristics can be controlled but not as much as the method of reduction. The metal deposits will require removal from the cathode either at regular intervals or continually, otherwise the density of the current will decrease. The metal powder removal will require the use of scrapers or stiff bristle brushes. Powders produced by this method are fern-like shaped and although the powder batches are relatively uniform in shape, they are of low apparent density and will require densification. An additional complexity with this process is that the metal powder will require a thorough wash before being allowed to dry and exposed to the surrounding environment.

4.3.2.3 *Atomisation*

With this method, the metal will have to be heated to melting temperature where it is drop fed through a small orifice or apex, hence producing a stream of molten metal. The molten metal is then broken up by a jet compressed inert gas is directed towards it at an impingement angle, up to 90°. The compressed gas leaves the nozzle at high velocity but slows down during its flight path to the apex. This attenuation is increased the further the compressed gas

has to travel and results in decreasing the effectiveness of being able to break up the molten stream. When the high the molten metal stream leaves the mouth of the orifice the sudden expansion causes a cooling effect. This, coupled with the cooling of the velocity inert gas,

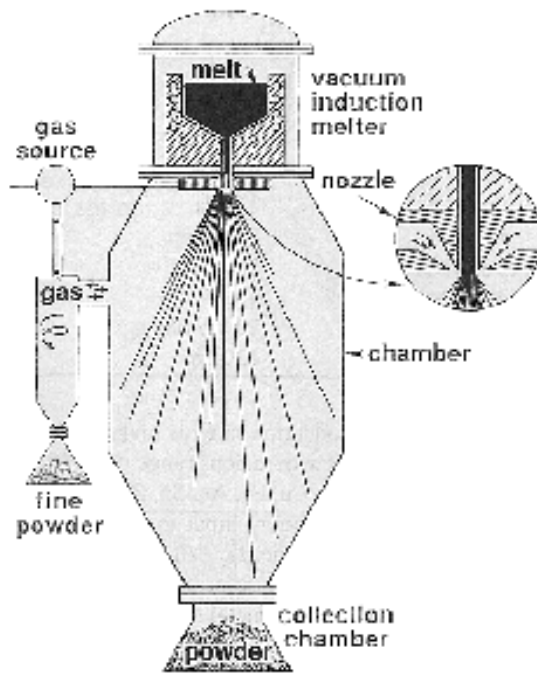


Figure 4.2 : An Example of an Atomiser

causes instantaneous solidification of the metal. Nitrogen, the second most abundant gas in the Martian atmosphere is the least expensive compressing gas but with magnesium, it experiences undesired reactions. Magnesium powder producers tend to use either helium or argon⁵ as the compressor gas, the latter of which is available in-situ, albeit in small quantities. Helium however delivers a solidification rate of one order of magnitude over argon. Subsonic gas atomisation achieves very low efficiencies and therefore results in utilising much more inert gas but this problem could be addressed by re-circulating the gas. A problem regularly experienced with gas atomisation is satellite accumulation. This is where, during the powder production process, small fine particles become attached to the larger, coarser particles. Satellite accumulation reduces packing density by several percent. Particle sizes from between 20 to 400 microns can be obtained from this method of particle

production. The ultrasonic gas atomisation technique can produce ~22 μm spherical particles of aluminium alloy and the satellite population are relatively small. However, very high pressures, between 68.94 to 137.86 bar (1000 to 2000 psig)⁶, are required.

The atomisation technique has multiple controlling factors;

1. Molten metal properties (viscosity, surface tension, composition, surface heat)
2. Molten flow geometry (metal flow rate, stream length, flight path, manifold geometry)
3. Compressed gas jet geometry (apex angle, number of jets, jet location)
4. Compressed gas jet flow (pressure, mass flow rate, viscosity, density)

The nozzle dimensions, temperature and rate of flow of the molten metal and compressing gas are all contributing factors to the resultant particle size. Ideally, this system is designed for temperatures less than 973.15K, due to the limited longevity of the molten metal orifice. Higher temperatures can be obtained by using advanced materials or by subjecting the orifice to cooling. The oxide content of metals particles produced by atomisation are as low as 0.2 to 0.3%. This is attributed to the rapid cooling and possible formation of a thin oxide layer than prevents additional oxidation.

In order to fully comprehend the mechanics of atomisation, a thorough understanding of complex fluid dynamics, heat transfer, energy conservation, chemical kinetics and thermodynamics is required.

An alternative but more costly method is to pressurise the molten metal with a soluble gas which is then introduced into a vacuum chamber via a narrow orifice. Upon entering the vacuum chamber, the pressurised molten metal will erupt into particles of near standard sizes.

⁵ Wilks, T., Magnesium Elektron, Personal Communication, 7th October 2002

⁶ Grant, Nicolas J., "Powder and Particular Production of Metallic Alloys", Chapter 1 in Chin, Gilbert Y., "Advances in Powder Technology", American Society for Metals, ISBN 0-87170-142-1, p6

4.3.2.4 Mechanical Commutation

This method of powder production is ideal for friable, brittle, malleable and ductile metals, which includes both magnesium and aluminium and their alloys. Mechanical or pneumatic gyratory crushers are employed to break the metal into pieces. Foreign bodies can be removed prior to milling by screening, de-stoning, pneumatic separation and magnets. The powder is then fed to a ball, disk or impact mill, which disintegrates the metals into powder form. The resulting powders are irregular in shape and sizes. The problem with this method is that the metal powder is exposed to oxides during its manufacture and grinding surface material can contaminate the powder. Placing the crushing and milling process in an inert atmosphere can avert the aforementioned problem of oxidation and by attaching abrasive resistant material on the grinding surface can reduce contamination. A device known as an 'eddy mill' can eliminate both oxidation and contamination. This technique involves the use of two high-speed fans, placed at opposing ends of a casing filled with an inert gas. The fans rotate in opposite directions and produce high velocity gas streams that carry the powder. The resulting collisions between the particles cause them to pulverise. The resultant powders, tend to be 'saucer' shaped.

4.3.2.5 Cold Stream Process

This process is ideal for brittle metals at low temperatures. The coarse metal produced from a grinder is fed via a very high-pressure carrier gas into a blast chamber. As soon as the particles leave the nozzle, the pressure drops dramatically causing a resultant temperature drop. The raw, brittle metal shatters against the target, forming irregular shaped powders. The plus factor with this concept is that the resultant powder has very little surface contamination and excellent compressible qualities.

4.3.3 Removal of Dust

Once the powder has been produced, it will be fed through a series of sieves in order to group the required sizes for metal powder fuel.

By slowly feeding the metal powders into a container with a vertically rising current, the dust particles (impurities) that have settling velocities less than the velocity of the current will be carried up and are subsequently removed from the container. The velocity of the vertical current can be adjusted in order to afford the disturbance required to separate an agglomerates and any dust that may be trapped between them.

4.3.4 Production of Alloy Powders

The problem experienced with the production of alloys is that due to the difference in the density of constituent metals, homogeneous powders are not produced.

The first step toward the production of alloy powders is to adjust the apparent densities of the alloy's components, in this case magnesium and aluminium. Once the percentage of mixtures is known, the less dense metal (magnesium) content is increased and the higher density material (aluminium) is decreased. The metals are then heat-treated and then the alloy is formed by diffusion.

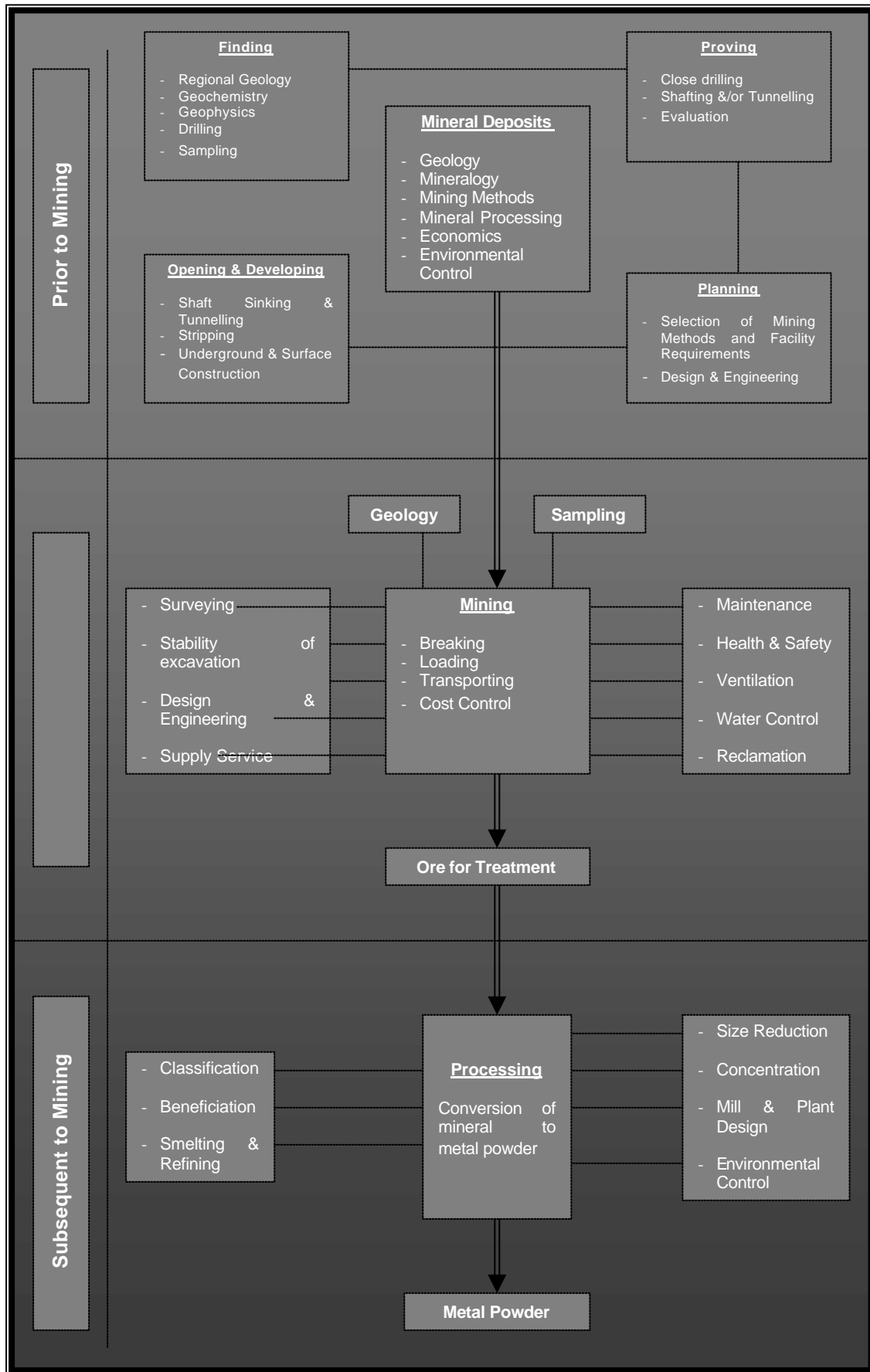


Figure 4.3: Metal Powder Production, from Start to Finish

4.3.5 Metal Powder Production Process

Figure 4.3 outlines the entire process that is used by terrestrial metal mining companies and the mining process on Mars will not be much different. In essence, the process can be split up in to three sub-categories;

1. Prior to Mining

This section requires a detailed analysis of the surface features in order to determine mineralogical concentrations of ores with high content magnesium, such as bronzite. A remote sensing spacecraft with a dedicated instrument for detection of surface and sub-surface mineralogical content will be required to map the entire face of Mars which will be followed by careful scrutiny of the wealth of information. Surface sampling will then, in the case for Mars, be performed by either manned or unmanned methods. The samples will have to be examined for quality before the intricate process of mining the ores takes place.

2. Mining Proper

Minerals containing high concentrations of magnesium are obtained from a predefined site of magnesium rich ores. Once the quality is determined, the design of the site for excavation is established and the pertinent engineering equipment is brought in to procure the ores. Continuous geological sampling and assessment of the process will be required to ensure quality of the product.

3. Subsequent to Mining

Once the ores have been obtained, they are crushed into powder form and then separated for magnesium rich minerals content. The minerals are fed into a furnace and heated to melting temperatures. Although the minerals will contain several elements, the melting point of magnesium will determine the point where the metal can be easily separated from the 'slag' and impurities. The molten metal powder is then drop-fed into a chamber with high pressure compressed directed towards the molten stream. This, as described earlier is one process to produce magnesium metal powder.

4.3.6 Safety Issues

Dust explosions are probably the main threat to the production of metal powder. Such incidents can occur in any environment with oxygen content where;

1. The dust concentration exceeds an explosive limit
2. The presence of an ignition source with sufficient energy

Dust explosions are not just applicable to metal powder production. In the catering industry, dust explosions can result from flour, custard, instant coffee, sugar, dried milk, potato powder and soup powder and have been responsible for the destruction of machinery, property as well as the injury and deaths of a number of people in the UK alone. The processes known to be responsible for producing explosive dusts are milling, grinding, spraying, conveyance and storage of food products. Suffice to say, every one of these processes are used for metal powder production and the only way to eliminate the possibility of a catastrophic explosion is to ensure that either one of both of the conditions mentioned above are prevented.

The majority of the dusts produced in the catering industry are classified as St 1. The following table was based on confinement within a sphere of 20 litres but can be scaled up for experiments in any sized vessel. Adequate measures for the protection of equipment can be based on determining the dust's explosive capability. The dust deflagration Index or K_{st} value is calculated by determining the ratio of the maximum pressure and the maximum rate of pressure rise.

Table 4.3: Dust Explosion Intensity Chart ⁷

Dust Explosion Class	K_{st} (Bar ms ⁻¹)	Characteristics
St 0	0	No Explosion
St 1	>0 <200	Weak Explosion
St 2	>200 < 300	Strong Explosion
St 3	>300	Very Strong Explosion

4.3.6.1 Metal Powder Production

The pressure rise can, in some part, be attributed to a sudden increase in erratic atmospheric flow. In relatively steady conditions, flame speed in a dust cloud can be several metres per second whereas in a turbulent environment, the flame speed can rise to several hundreds of metres per second. This sudden increase in pressure, in an enclosed space, will result in a catastrophic explosion. The metal powder must first be airborne and if ignited it will cause an explosion, the intensity of which is dependant on its apparent density. If the environmental temperature within the enclosed area is high enough to sustain a burn, friction between the metal particles could cause an arc and ignite the powder, resulting in a controlled burn. The density of the flow of metal particles during metal powder production must therefore be kept to a minimum and the entire procedure of powder production should be manufactured either in a partial vacuum or in the presence of a non-reactive inert gas.

4.3.6.2 Rocket Engines

In order to alleviate potential problems during powder feed via either pneumatic or piston fed concepts, sensors which detect a drop or rise in pressure should be strategically placed throughout the system and carefully monitored. The feed pressure could then be adjusted accordingly or the engine shut down and vented in the case of an emergency. To prevent possible arcing as a result of friction between the metal pipes and the metal powder particles, a facility to remove static electricity should be installed.

Having the ability to accurately predict the conditions where a flame can propagate within a confined space may prove to be of value to a metal powder fuel engine design. An increase apparent density of the dust cloud plus a rapid circulation of fluid flow has proven to increase the burn rate of the metal powder fuel. From this, the deduction is made that the fuel should be injected in a way that maximises turbulence within the thrust chamber. Increasing the metal powder density towards the injection end could, hypothetically, increase particle burn rate resulting in a reduction of two-phase flow loss and incomplete combustion.

4.3.6.3 Safety Summary

Addressing and solving the problems of powder production from a manufacturing standpoint and subsequent pneumatic feed will not only benefit an in-situ propulsion system using metal powder fuels, but will also assist with improving the machinery used in the manufacture of food products. As mentioned earlier, such technological advancement could result in the elimination of costly damage to European industry in the form of productivity, equipment and most importantly, the health and safety of European workers.

⁷ Health and Safety Authority, "Dust Explosions in the Food Industry", <http://www.hse.gov.uk/pubns/food2.htm>

5.0 Propellant Combinations using Martian Resources

5.1 Introduction to Propellant Combinations

The majority of previous “Martian” in-situ propellant studies have concentrated on LCO/LO₂ or LOX/LCH₄ engine concept utilising ingredients from the atmosphere alone and in the case of the latter; a small amount of LH₂, brought from Earth, would be required to produce LCH₄. The detection of pockets of hydrogen over the surface of Mars, once confirmed, could eliminate the necessity of bringing the LH₂ from Earth. The LCH₄ option provides a very reasonable performance and the storage temperature is a lot more manageable than LH₂. Russian, Canadian, European, Japanese and American researchers have also advocated the use of metal powders, mined and processed from the Martian regolith combined with CO₂ from the atmosphere of Mars, as a potential source of propellant.

Based on the research from the previous section on the available gases and metals from Martian resources, numerous propellant combinations were identified employing oxidisers derived from the Martian atmosphere and combined with a selection of metal fuels. The two oxidisers were carbon dioxide and cryogenic (liquid) oxygen. LOX was included for several reasons. The first is because the production of LOX is an extension of LCO₂ production and although more complex to generate in terms of manufacture and storage, the combination of LOX with metal fuels delivers higher values of specific impulse and affords the opportunity to address the issue of a lunar soil propellant as well. The difference of course being that the LOX on Mars is extracted from the Martian atmosphere following the production of CO₂ whereas on the Lunar surface, it would have to be extracted from the metal oxides. This study focuses on the use of metal powders, which are found in extraterrestrial bodies, as well as in Martian regolith and rocks. The metals available in quantity on Martian surface are magnesium, aluminium, silicon, titanium, manganese, iron, potassium, sodium, chromium and magnalium. Non-ISRU metals and their hydrides that are known to react well with CO₂ and LOX were also examined. These fuels were zirconium, lithium, beryllium and boron¹. Calcium and silane were not available as propellant ingredients in the software used and were not examined. The others chemicals were² beryllium hydride, magnesium hydride, aluminium borohydride, diborane, pentaborane, methane, ethane, butane, propane and methanol.

5.2 Computer Code

The propellant evaluation program (PEP) used is adaptation of the US Navy's PEP code³ and is available in the public domain. Originally in FORTRAN format, Wickman modified the program for the personal computer (PC). The input parameters are in Imperial units but the output delivers Imperial and SI units. Prior to proceeding with the lengthy procedure of identifying stoichiometric values for each propellant combination, a known chemical combination was used to ensure that the PEP provided a realistic output. The main engines from the US space transportation system (STS) deliver a sea level specific impulse of 363 seconds. The PEP provides a value of 363.4 seconds with a chamber pressure of 500 psi (34.47 bar) and exit pressure of 14.5 psi (1 bar). For the purpose of this short study contract, it was determined that the PEP code, producing similar in results, would suffice.

The method of ascertaining the stoichiometric value was an arduous task. In the absence of an automated iterative technique, the values were entered in manually. The oxidiser and fuel were given values of decreasing and increasing values, respectively. For example, the oxidiser value would start at 90% of the propellant combination and the fuel with 10%. The results were noted and then the figures adjusted to 80% and 20%. If the results showed a peak between the deciles, then the process would continue to values such as 85% and 15% and so on. It was then possible to manually iterate and find the stoichiometric value for that propellant combination.

¹ Shafirovich, E. Ya., Shirayev, A. A. and Goldshleger, U. I., “Magnesium and Carbon Dioxide : A Rocket Propellant for Mars Missions”, Journal of Propulsion and Power, Volume 9, Number 2, March-April, 1993, p198

² Shafirovich, Evgeny Ya. and Goldshleger, Udo I., “Comparison of Potential Fuels for Martian Rockets using CO₂”, Journal of Propulsion and Power, Volume 13, Number 3, May – June 1997, p395

³ PEP Thermochemistry Computer Program, Cruise, D. R., “Theoretical Computations of Equilibrium Compositions, Thermodynamics and Performance Characteristics of Propellant Systems”, NWC TP-6037, April 1979

A more sophisticated code written by Gordon and McBride from NASA Glenn Research Centre is available by formal request only but its complexity supersedes the level of work required in this study.^{4,5} It is, however, recommended that Gordon and McBride's program is used for a more detailed analysis of combustion in future work on CO₂/metal powder engines.

5.3 Propellant Combinations

Table 5.1 outlines a list of propellant combinations that were addressed in this study for 200 psi (13.79 bar) and 500 psi (34.47 bar) chamber pressures, respectively. Some chemicals were not available in the PEP program and therefore could not be calculated. They were included in the table to highlight that such combinations exist. Previous studies on theoretical specific impulse based on increasing chamber pressure and expansion ratio with a Martian ambient exit pressure of 6.9 mb showed little improvement in performance for the added mass required to pressurise the propellant tank. 13.79 bar was therefore chosen for the pressure fed concept. The pump fed concept, even though not addressed in detail in this study, was still examined in order to provide a comparison in performance.

Table 5.1: Propellant Combinations Examined

Oxidiser	ISRU Metal Fuels	Other potential ISRU Fuels	Non-ISRU Metal Fuels	Other non-ISRU Fuels (Hydrides)
Carbon Dioxide (CO ₂) or Liquid Oxygen (LOX)	Magnesium (Mg)	Carbon Monoxide (CO)	Zirconium (Zr)	Beryllium Hydride (BeH ₂)
	Aluminium (Al)	Liquid Hydrogen (LH ₂)	Lithium (Li)	Magnesium Hydride (MgH ₂)
	Silicon (Si)	Methane (CH ₄)	Beryllium (Be)	Silane (SiH ₄)
	Calcium (Ca)	Ethane (C ₂ H ₆)	Boron (B)	Aluminium Borohydride (Al(BH ₄) ₃)
	Titanium (Ti)	Propane (C ₃ H ₈)		Diborane (B ₂ H ₆)
	Manganese (Mn)	Butane (C ₄ H ₁₀)		Pentaborane (B ₅ H ₉)
	Iron (Fe)	Methanol (CH ₃ OH)		
	Potassium (K)	Dinitrogen Dihydride (N ₂ H ₂)		
	Sodium (Na)			
	Chromium (Cr)			
	Magnalium (Mg/Al)			

5.4 Chemical Properties

Initially, the metal powder fuels were selected for their availability on the Martian surface and others for their performance when combined with CO₂. However, it is unlikely that a number of these fuels will be employed in a propellant combination due to undesirable pre-combustion and post-combustion properties.

Although beryllium and beryllium hydride, when combined with CO₂, delivers the highest values of specific impulse and the hydride yields a low condensed phase fraction (CPF) in the nozzle, there is little or no knowledge of its combustion dynamics. Shafirovich et al. states that research to date on diborane as a potential fuel for use with CO₂, present no data on the ignitability and two-phase flow losses. It was shown that the mass fraction in condensed phase effects (CPE) was twice as with CO₂/magnesium reactions. Also, following combustion of CO₂ and diborane, B₂O₃ is formed which contributes to additional losses. Boranes in general have high CPF, some up to 0.8 with mixture ratios of 2.5⁶. Silane, although not

⁴ Gordon, S. and McBride, Bonnie J., "Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications", Part I : Analysis, NASA Reference Publication 1311, October 1994, pp55

⁵ Gordon, S. and McBride, Bonnie J., "Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications", Part II : Users Manual and Program Description, NASA Reference Publication 1311, June 1996, pp177

⁶ Shafirovich, E. Ya. and Goldshleger, U. I., "Comparison of Potential Fuels for Martian Rockets Using CO₂", Journal of Propulsion and Power, Volume 13, Number 3, May – June 1997, pp395 – 397

available as a compound in the PEP program, could potentially be manufactured on Mars. Aluminium borohydride and silane have comparable specific impulses although with $\text{Al}(\text{BH}_4)_3$, the CPF is higher and it is an unstable compound. Silane, however, is stable, and has a comparatively low CPF but it is a toxic compound and there is no data on the ignition and combustion of silane in a CO_2 environment.

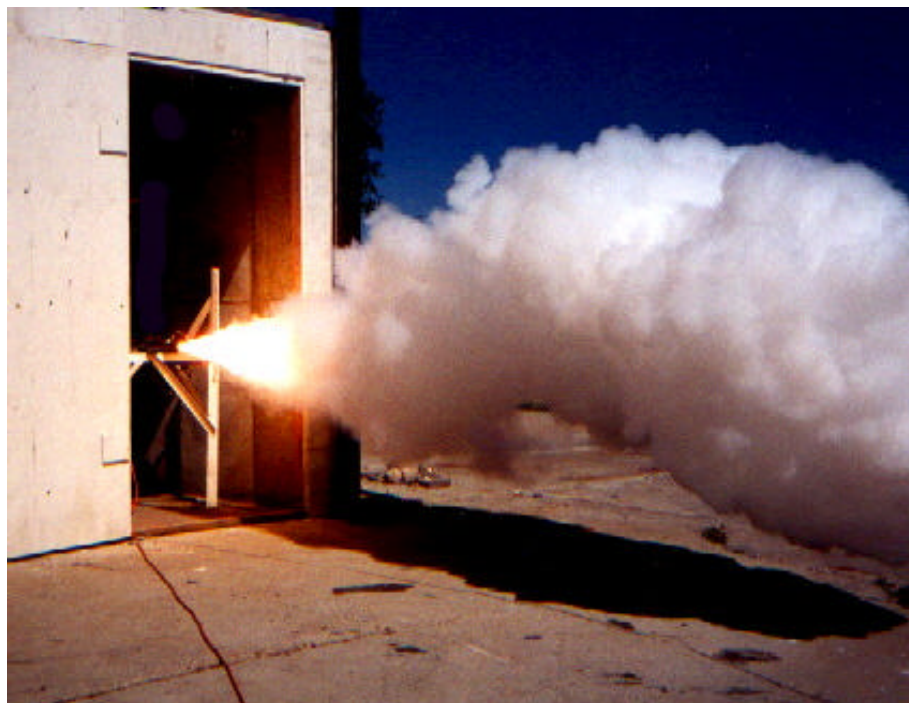


Figure 5.1: Static Firing a CO_2 /Magnesium Powder “Martian” Rocket Engine⁷

The delivered performance of aluminium in a CO_2 environment, when compared to other metals found on the Martian surface, is far superior. However, its ignition temperature is also higher and a permanent ignition source is required to maintain combustion, as it cannot on its own. Experiments on single particles of lithium, boron, silicon, titanium and zirconium showed that the mechanics of combustion were not suitable for employment as fuels in CO_2 /metal powder rocket engines. Lithium experienced no problems with ignition but the combustion time in the vapour phase was lengthy. This would result in a larger than standard engine dimensions in order to cater for the total residency time during the combustion phase. This may not be warranted for the additional 10 seconds of theoretical specific impulse. Boron, like aluminium and lithium has poor ignition capabilities in CO_2 . Silicon, titanium and zirconium do not burn in the vapour phase and result in very poor combustion characteristics.

Magnesium has time and again been able to combust in CO_2 environments, either with single particle experiments or demonstrator engine tests, Figure 5.1. It is also storable and non-toxic. It can be ignited at relatively low temperatures and exhibits a high rate of combustion in the vapour phase. Magnalium is one possibility of attaining a steady burn rate, higher performance yet maintains a relatively low ignition temperature.

⁷ Wickman, J. H., “In-Situ Mars Rocket and Jet Engines Burning Carbon Dioxide”, AIAA-99-2409

Table 5.2: PEP Stoichiometric Values for 13.79 bar (200 psi)

Chemical	Carbon Dioxide (CO ₂)	Oxidiser/ Fuel Ratio	Liquid Oxygen (LOX)	Oxidiser/ Fuel Ratio
Non-ISRU Metals				
Zirconium (Zr)	193.9	49/51	232.9	55/45
Lithium (Li)	213.7	50/50	315.3	41/59
Beryllium (Be)	257.3	83/17	316.7	58/42
Boron (B)	192.1	84/16	298.6	70/30
ISRU Metals				
Magnesium (Mg)	215.7	72/28	263.2	64/36
Aluminium (Al)	228.6	71/29	275.5	68/32
Silicon (Si)	200	76/24	246.9	75/25
Calcium (Ca) ⁸	-	-	-	-
Titanium (Ti)	180.1	65/35	241.9	64/36
Manganese (Mn)	-	-	-	-
Iron (Fe)	99.5	70/30	178.5	65/35
Potassium (K)	104.6	65/35	151.3	40/60
Sodium (Na)	-	-	184.9	60/40
Chromium (Cr)	125.8	50/50	211.1	59/41
Magnalium (Mg/Al)	225.2	71/(5M/24A)	-	-
Other non-ISRU Chemicals				
Beryllium Hydride (BeH ₂)	331.2	50/50	439.6	56/44
Magnesium Hydride (MgH ₂)	239.7	41/59	328.8	35/65
Silane (SiH ₄) ⁹	-	-	-	-
Aluminium Borohydride (Al(BH ₄) ₃)	283.1	33/67	369.6	60/40
Diborane (B ₂ H ₆)	291.1	45/55	380.3	64/36
Pentaborane (B ₅ H ₉)	263.2	40/60	355.1	66/34
Other potential ISRU Chemicals				
Carbon Monoxide (CO)	-	-	256.9	70/30
Liquid Hydrogen (LH ₂)	-	-	443.6	79/21
Methane (CH ₄)	-ve temp	-ve temp	341.4	26/74
Ethane (C ₂ H ₆)	-	-	-	-
Propane (C ₃ H ₈)	-ve temp	-ve temp	-	-
Butane (C ₄ H ₁₀)	-	-	-	-
Methanol (CH ₃ OH)	162.7	1/99	315.5	55/45
Dinitrogen Dihydride (N ₂ H ₂) ¹⁰	-	-	-	-

⁸ Calcium (Crystalline) was not available as a chemical in the Propellant Evaluation Program database

⁹ Silane was not available as a compound in the Propellant Evaluation Program database

¹⁰ Dinitrogen Dihydride was not available as a compound in the Propellant Evaluation Program database

Table 5.3: PEP Stoichiometric Values for 34.47 bar (500 psi)

Chemical	Carbon Dioxide (CO ₂)	Oxidiser/ Fuel Ratio	Liquid Oxygen (LOX)	Oxidiser/ Fuel Ratio
Non-ISRU Metals				
Zirconium (Zr)	201.4	49/51	244	55/45
Lithium (Li)	240.7	50/50	330.7	41/59
Beryllium (Be)	268.7	83/17	330.9	58/42
Boron (B)	201.8	84/16	313.9	70/30
ISRU Metals				
Magnesium (Mg)	224.9	72/28	277.2	64/36
Aluminium (Al)	238.7	71/29	290.3	68/32
Silicon (Si)	208.2	76/24	260.4	75/25
Calcium (Ca) ¹¹	-	-	-	-
Titanium (Ti)	101.4	65/35	254.7	64/36
Manganese (Mn)	-	-	-	-
Iron (Fe)	103.8	70/30	184.3	65/35
Potassium (K)	109.4	65/35	167.4	40/60
Sodium (Na)	158.9	-	193.2	60/40
Chromium (Cr)	131.4	50/50	220.8	59/41
Magnalium (Mg/Al)	234.1	71/(5M/24A)	-	-
Other non-ISRU Chemicals				
Beryllium Hydride (BeH ₂)	346	50/50	460.2 ¹²	56/44
Magnesium Hydride (MgH ₂)	252.1	41/59	344.8	35/65
Silane (SiH ₄) ¹³	-	-	-	-
Aluminium Borohydride (Al(BH ₄) ₃)	293	33/67	386.1	60/40
Diborane (B ₂ H ₆)	303.1	45/55	395.9	64/36
Pentaborane (B ₅ H ₉)	273.2	40/60	371.1	66/34
Other potential ISRU Chemicals				
Carbon Monoxide (CO)	-	-	267	70/30
Liquid Hydrogen (LH ₂)	-	-	454.8	79/21
Methane (CH ₄)	120.4	-	354	26/74
Ethane (C ₂ H ₆)	Very Low Isp	-	Very Low Isp	-
Propane (C ₃ H ₈)	158.9(?)	-	346.7	-
Butane (C ₄ H ₁₀)	Temp=0	-	Temp=0	-
Methanol (CH ₃ OH)	169.1	1/99	327.5	55/45
Dinitrogen Dihydride (N ₂ H ₂) ¹⁴	-	-	-	-

5.5 Impure Gases and Powders

Here, impurity is not just related to physical impurities such as contaminants within the oxidiser or tainted quality of the metal powder. Impurities are also recognised as any chemical other than the chosen oxidiser and the fuel. This implies that the carrier fluid that is employed to transport the metal powder particles from the storage tank to the combustion chamber is also referred to as a contaminant because, depending on the fluid, it could contribute to a degradation of performance.

If one were to assume that during the manufacture of liquid CO₂, the trace gases were **not** removed, then this would result in an automatic reduction of ~5% of the CO₂'s contribution to the propellant formulation. Since previous research has determined that magnesium or magnalium would be the most likely candidates for a CO₂/metal powder engine, only these two metals were addressed for 'degradation of performance' using either impure metal

¹¹ Calcium (Crystalline) was not available as a chemical in the Propellant Evaluation Program database

¹² Density in PEP result = 0 (?)

¹³ Silane was not available as a compound in the Propellant Evaluation Program database

¹⁴ Dinitrogen Dihydride was not available as a compound in the Propellant Evaluation Program database

powders, contaminants in CO₂ and carrier fluids which may or may not increase Specific Impulse.

During rocket engine tests at Wickman Spacecraft and Propulsion Company (WSPC) in 1998, a series of gases were used as the carrier fluid; hydrogen, helium and nitrogen. The effect of those fluids on specific impulse can be seen in Figure 5.2. The results presented in tables 5.2 and 5.3 do not take into account any carrier fluid 'impurities' in the formulation. The reason for the omission was because experiments on the effect of adding carrier fluids in general provided a small increase in performance. So, the results in table 5.2 and 5.3 show a pessimistic result of the true specific impulse.

The selection of carrier fluid for engine tests during WSPC experiments were selected not just storage for considerations but also availability of Mars. Nitrogen is an in-situ gas, constituting ~2.7% of the atmosphere. A 7% weight fraction of hydrogen delivers over 25% increase in specific impulse but it is less dense than the other fluid that leads to the requirement of larger storage tanks for the same volume of gas. These experiments were conducted before the detection of hydrogen on Mars and so nitrogen was selected as the carrier fluid to transport the magnesium powder. The slight discrepancy in specific impulse values used during the WSPC experiments and those calculated here come down to the fact that during the former, the chamber pressure was set at a lower value, the fuel/oxidiser stoichiometric value was estimated after several runs and no detailed iterations to optimise the propellant combinations were performed. One possibility that was not addressed with the work performed at Wickman was the use of CO₂ as the metal powder carrier gas. Employment of the oxidiser as the carrier fluid would result in a gaseous/metal powder monopropellant, which is addressed in a later section.

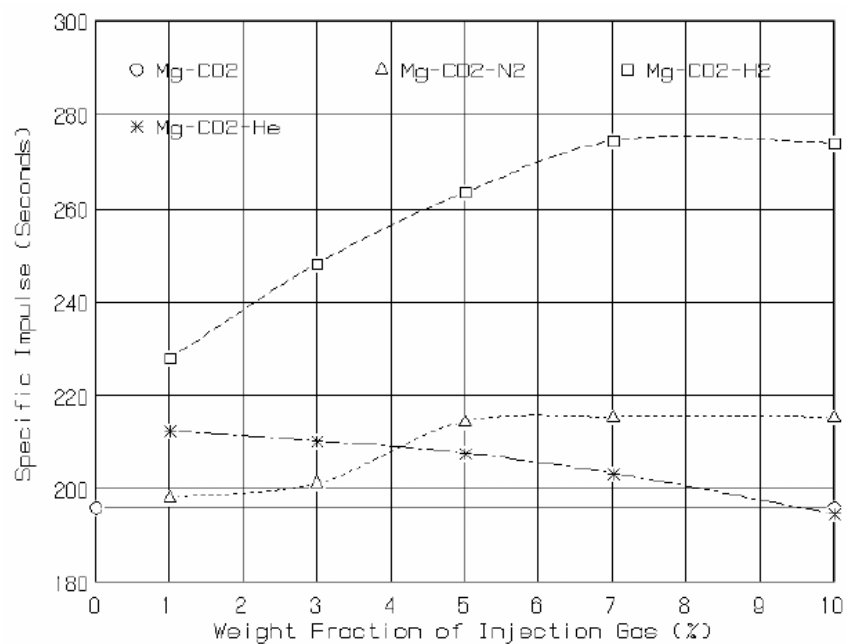


Figure 5.2: Carrier Fluid Influence on Specific Impulse¹⁵

Impurity distribution within the powder is an important contributory factor towards not just the combustion process, which results in degradation of performance, but also to the longevity of the engine. Although this section focuses of performance related output, one must keep in mind that reducing the purity of the metal powder from 99.5% by several percent could increase the lifetime of the engine feed lines and injectors. This is because the more pure the powder, the greater capacity for abrasion and wear within the engine's feed tubes.

¹⁵ Wickman, John H., "In-Situ Mars Rocket and Jet Engines Burning Carbon Dioxide", AIAA 99-2409

6.0 Condensed Phase Effects

Shafirovich et al.¹ provides solutions to condensed phase effects (CPE) for a number of metal powders when ignited in CO₂. These results are presented in Table 6.1 where T_c is the chamber temperature (in K), T_e is the temperature (in K) at the nozzle's exit and z_e is the condensed phase fraction (CPF).

Table 6.1: CPF as a Function of CO₂-Fuel Mass Ratio

	$\phi=2$			$\phi=5$			$\phi=8$		
	T_c	T_e	z_e	T_c	T_e	z_e	T_c	T_e	z_e
Li	2210	1458	0.58	2096	1403	0.35	2155	1215	0.36
Be	2969	2070	0.71	3001	2281	0.46	2851	1456	0.31
B	2153	1564	0.65	2101	1471	0.55	1802	828	0.36
Mg	3100	1975	0.55	2073	894	0.28	1533	703	0.19
Al	2730	1851	0.52	2327	1022	0.31	1726	735	0.21
Si	2266	1622	0.59	1996	848	0.36	1479	737	0.25
Ca	2869	1384	0.47	1606	905	0.33	1322	704	0.28
Ti	2257	1334	0.56	1504	733	0.29	1123	631	0.21
Zr	2314	989	0.45	1263	643	0.24	951	417	0.16

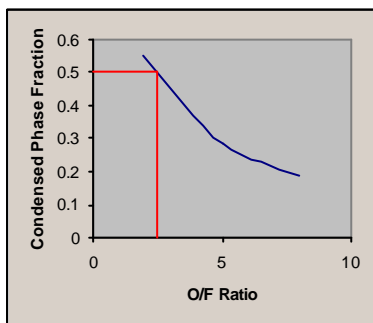


Figure 6.1: Magnesium CPF

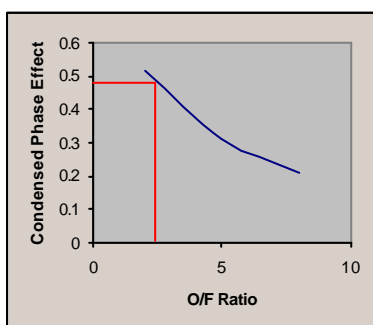


Figure 6.2: Aluminium CPF

In order to identify the CPF for the optimum specific impulse values calculated via the PEP program, a graph is drawn from the three points provided by table 6.1 for the magnesium values with the CPF as the Y-axis and the oxidiser-fuel ratio on the X-axis. The optimum O/F mixture ratio for CO₂-magnesium was 2.57:1. A vertical line was drawn to the point where it meets the curve and then a vertical line is drawn towards the Y-axis. This point on the Y-axis determines the CPF for the PEP values and those used in the MAV propulsion system. For maximum specific impulse, the CPF value for magnesium is 0.5 or 50% and for aluminium, it is 0.48 or 48%. In reality, these figures will have to be reiterated in order to reduce CPE, which effectively reduces the engine's overall performance. CPE decreases by increasing the O/F mixture ratio, which is desirable since additional CO₂ production will minimise the fuel required and the resultant TMI mass but at the expense of performance. An additional benefit of increasing the O/F mixture ratio is that the combustion temperatures would be lower, lessening the requirements of the cooling system. However, reducing the temperature too much will mean that ignition will be harder to achieve and combustion will be more difficult to sustain.

¹ Shafirovich, E. Ya., Shirayev, A. A. and Goldshleger, U. I., "Magnesium and Carbon Dioxide : A Rocket Propellant for Mars Missions", Journal of Propulsion and Power, Volume 9, Number 2, March – April, 1993, p198

7.0 Sample Return Mission Scenarios

7.1 Introduction to Mission Scenarios

In this report, four different mission scenarios are addressed and for each scenario, a selection of different propellant combinations employing cryogenic, storable liquid, composite solid and in-situ (CO₂/metal powder) were examined. The purpose of investigating non-ISRU as well as ISRU propulsion systems was to afford a comparative assessment between competing technologies.

The four possible mission options are;

1. Direct Mars entry (DME) / direct Earth return (DER)
2. Out-of-orbit entry (OOE) / direct Earth return (DER)
3. Direct Mars entry (DME) / Mars orbit rendezvous (MOR)
4. Out-of-orbit entry (OOE) / Mars orbit rendezvous (MOR)

7.2 Definitions

- i) Direct Mars Entry (DME): This is probably the simplest of scenarios in terms of technological know-how but it lends itself to 'hoping for the best' with regards to determining a landing site on Mars. The propulsive burn for trans Mars injections (TMI) is initiated at a predefined time and is based on intricate calculations so the estimated time/date of arrival and landing site can be accurately predicted. The inclusion of an additional amount of propellant for small propulsive burns to aid in directing the spacecraft/lander towards the preferred entry vector could assist with the selection of a more appealing landing site. The benefit of DME is that the spacecraft/lander will enter directly into the Martian atmosphere without entering into Martian orbit. Therefore, additional propellant or extra mass for an aerobrake will not be required, hence contributing to an overall reduction in injection mass. This mass saving is only applicable for the portion of the Earth orbit to Mars orbit (TMI) segment of the mission.
- ii) Out-of-Orbit Entry (OOE): TMI will proceed as explained above but, compared to DME, this is an expensive option as additional mass, in the form of either propellant or an aerobrake, will be required to slow down the spacecraft in order for it to enter into a low Mars orbit (LMO). However, as Mariner 9 experienced almost four decades ago, very high speed winds accompanied by kilometre high dust storms can mask the entire planetary surface for months at a time, totally obscuring any potential landing site. This option affords an opportunity for mission planners to keep the lander in orbit, wait for any unforeseen natural events such as dust devils or GDS to subside and then pre-select a desired landing spot. Two options exist for OOE.
 - a) *Propulsive Capture (PC)*

This is the more expensive of the two options in the form of mass requirement but is also the most versatile. The use of a propulsion system allows the applied thrust to be varied to suit the moment rather than having to plan manoeuvres month, possible years, prior to TMI.
 - b) *Aerobrake Capture (AC)*

This procedure has become increasingly popular amongst mission planners due to the considerable savings in injection masses but the Aerobrake sequence must be defined prior to TMI. Mid Course Corrections (MCC) during TMI can provide mission planners the opportunity to adjust the spacecraft's entry vector, hence altering its final orbit around Mars.
- iii) Direct Earth Return (DER) : With this option, the Upper (2nd) Stage of the Mars ascent vehicle (MAV) is designated the task of not only reaching Martian orbit but also providing the necessary ΔV burn to return to Earth. Once the MAV arrives at

Earth, it is assumed that it will enter the Earth's atmosphere directly and land in a pre-designated site. The benefit of such a concept is that an ERV will not have to spend months, possibly years, trying to locate and rendezvous with the sample container. However, this option causes concern due to issues of planetary quarantine but is realistically the only option available for a sample return mission at this time.

- iv) Mars Orbit Rendezvous (MOR) : This reduces the propulsive requirements of the MAV Upper (2nd) Stage. An Earth return vehicle (ERV) must remain in Martian Orbit and then locate and capture the sample canister prior to returning to Earth. Mission planners will then have to decide how the sample is retrieved from the ERV. The ERV could enter Earth's atmosphere directly unless it has enough propellant or uses an aerobraking manoeuvre in order to enter into an orbit and with an inclination that is either accessible by either the US space transportation system/STS (Space Shuttle) or the International Space Station (ISS).

7.3 Mars Ascent Staging

The intention was to endeavour to minimise complexity of the system with the use of a single stage to Mars orbit (SSMO) ascent vehicle. However, initial calculations showed that only systems delivering very high specific impulse, such as LOX/LH₂, were found to be capable of single stage to Mars orbit with a reasonable inert mass fraction. In-situ propulsion systems using CO₂/metal powder fuels were not capable of achieving Mars orbit with a single stage. As a result, two stage to Mars orbit (TSMO) options were employed for all propellant combinations. LOX/LH₂ was also included in the TSMO for the purpose of comparison.

7.4 Propellant Combinations

An attempt was made to ensure maximum use of in-situ propellants. For initial missions, it is envisaged that the metal powder would be taken from Earth to Mars so, increasing the required CO₂ in the propellant formulation would result in reducing the injection mass from Earth. Oxidiser rich propellant, although reduces performance, would decrease condensed phase effects encountered with CO₂/metal powder combustion but efficiency is increased.

The combinations of propellants used for all four mission scenarios are shown in Table 7.1, where both 1st (lower) and 2nd (upper) stage use the same propellant combination. Although the study is centred on in-situ propulsion, cryogenic propellants were used to provide baseline guidance. NASA sample return planners tend to favour the use of storable liquid propellants or solid rocket motors and so, in order to provide a comparative assessment between in-situ and conventional technologies, three storable liquid propellant combinations and one SRB propellant were also considered.

Table 7.1: Propellants examined for Mars Ascent Vehicle (MAV)

Two Stage Surface-to-Mars Orbit Propellant Options				
Cryogenic	Storable	Solid	Cryogenic In-Situ	Storable ¹ In-Situ
LOX/LH ₂ (STS)	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	AP/Al (SRBs)	CH ₄ /O ₂	CO ₂ /Mg
LOX/RP1 (Zenit 2 nd Stage)	MMH/NTO		CO/O ₂	CO ₂ /Al
	N ₂ O ₄ /UDMH (Proton)			CO ₂ /Be
				CO ₂ /BeH ₂
				CO ₂ /Si

The small contingent of in-situ proponents in the US advocates the use of liquid methane (LCH₄) or cryogenic carbon monoxide (LCO) as the fuel element with liquid oxygen (LOX) as the oxidiser. These propellants were also included to provide a comparison between the 'performance' of cryogenic in-situ and storable in-situ. It is important to note, however, that in this report, although the results show that the total injection mass for LCH₄/LOX engines outperform LH₂/LOX systems, calculations were performed with an assumption that 75% of this (LCH₄/LOX) propellant is manufactured on Mars where the remaining 25% LH₂, required to produce LCH₄, is brought from Earth. The LH₂/LOX system is based on a "bring your own propellant" (BYOP) scenario, which results in an increased TMI injection mass when compared with the lower performing LCH₄/LOX option. An additional factor that was omitted in the LCH₄/LOX calculations is the mass of the system required to manufacture the LCH₄ and LOX. Although this is inconsistent and in contradiction to the objective of this report, the LCH₄/LOX is included in the final results primarily to compare with the LH₂/LOX and is intended to highlight the tactical omission sometimes used when comparing in-situ versus conventional systems for Mars sample return. This omission is used to sell the idea that in-situ is superior but in actual fact, there is no real comparison simply because the choice of the propulsion system will depend on the overall objective of the mission. If the objective is a one-off Mars sample return, then conventional propulsions systems will probably be selected. If the objective is to develop an infrastructure for future human exploration of the Martian surface based on a 25-year program such as ESA's Aurora program, then in-situ propellant production is a necessity.

In addition to the combinations shown in Table 7.1 which use the same propellant combinations for both stages, another set of configurations were calculated using conventional propulsion systems for the upper (2nd) stage and CO₂/metal powder propulsion systems for the lower (1st) Stage. These combinations are shown in Table 7.2.

¹ Only CO₂, Magnesium, Aluminium and Silicon can be found on Mars but for initial missions, it is envisaged that all 'metal' powder fuels or their Hydrides will be taken from Earth. Reference here is made to In-Situ because the CO₂ will be manufactured on the Martian surface.

Table 7.2: Upper Stage (Conventional) and Lower Stage (In-Situ) Propellants

2 nd Stage	LOX/LH ₂ (STS)	LOX/RP1 (Zenit 2 nd Stage)	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	MMH/NTO	N ₂ O ₄ /UDMH (Proton)	AP/Al (SRBs)
1 st Stage	CO ₂ /Mg					
	CO ₂ /Al					
	CO ₂ /Be					
	CO ₂ /BeH ₂					
	CO ₂ /Si					

7.5 Parameters

Numerous research papers and system components from Pathfinder and Beagle2, Table 7.3, were used as a general guideline to estimate system components and masses.

Table 7.3: Total Ejected Mass for Beagle2²

Beagle2 : Total Ejected Mass – 69kg	
Component	Mass (kg)
Entry and Descent System	36
• Structure and Mechanisms	17.2
• Airbags and Gas Generator	15
• Parachutes	3.8
Lander	33
• Instruments ³	8.7
• Structure ⁴	14.2
• Sub-systems	8
• Arm	2.1

In reality, sizing components are dependant on the specific mission where linear scaling laws are not applicable. Many figures used were educated estimates based on previous landers, current designs and proposed vehicles for sample return. Mars 2003 Rover engineers performed linear scaling based on Pathfinder data and this caused numerous problems in the interfacing resulting in re-design⁵ and cost over runs.

The inert mass fractions for the upper stage were set at 10% and 12% for the lower stage. These percentages were only applicable for conventional propulsion systems. Since no flight ready in-situ propulsion systems have been developed, the inert mass fractions were estimated and the values were fixed at 20% for both upper and lower stages.

Table 7.4 shows the constants used in the calculations. Figures that were assumptions are displayed in italics.

² Sims, M., Beagle2 Mission Manager, Personal Communication, 16th August 2002

³ electric support, computer, power, battery, transceiver, harness, etc.

⁴ including solar panels (1.5kgs) but not cells

⁵ Stephenson, Robert J., JPL Structural Engineer, Personal Communication, 5th August 2002

7.6 General Equations

In order to determine the overall trans Mars injection (TMI) mass, the masses from the following sequence of events were calculated;

1. Propellant mass required to return samples from Mars orbit to Earth, referred to as trans Earth injection (TEI)
2. Propellant mass required to lift sample from Mars surface to Mars orbit
3. The entry, descent and landing system (EDLS) mass
4. TMI mass

Table 7.4: Table of Constants

Constant	Value
?V (Surface to Mars Orbit) (m/s) 140 km Parking Orbit	4300
?V (Mid Course Corrections - MCC) (m/s) including plane change	300
?V (Mars Orbit Insertion - MOI) (m/s)	2500
?V (Trans Earth Injection - TEI) (m/s) including mid course corrections and plane change	2800
?V (Earth Orbit Insertion - EOI) (m/s) including mid course corrections and plane change	2500
Gravitational Acceleration (m/s ²)	9.80065
TMI/(ERV)TEI Specific Impulse (s) ⁶	320
TMI Propulsion System Inert Mass Fraction	0.10
ISRU Real Specific Impulse Factor ⁷	80%
Mass of Sample (kg)	0.5
Mass of Sample Container (kg)	3
MAV Fairing (kg)	5
Communications ⁸ (Upper Stage) (kg)	5
Stage 1 Separation Mechanism (kg)	2.4
Stage 2 Separation Mechanism (kg)	12.8
CO ₂ Compressor or Sorption Bed (kg)	12
Lander and Surface Support Factor	0.12
Lander Aeroshell Factor ⁹	0.25

⁶ This value also applies for TEI for the MOR/ERV option

⁷ this percentage is multiplied by the theoretical values of the in-situ propellants obtained from the results of the propellant evaluation program in order to provide a realistic specific impulse, after two-phase flow loss and incomplete combustion

⁸ valid for direct return only

⁹ factor includes airbags and parachutes

7.6.1 Trans Earth Injection (TEI)

As mentioned earlier, there are two methods to return samples to Earth from Mars. The first is by use of an Earth return vehicle (ERV), which captures the sample that is launched into Mars orbit by a two stage MAV which then initiates a propulsive burn that brings the sample back to Earth. The second option is to launch the sample container directly to Earth from the surface of Mars where the upper (2nd) Stage of the MAV will have to provide the necessary ΔV burn for TEI without an MOR. The ERV option is addressed first.

7.6.1.1 Earth Return Vehicle (ERV)

The equation generally used to estimate propellant mass is¹⁰,

$$m_{propellant} = \frac{m_{payload} \left(e^{\left(\frac{\Delta V}{I_{sp} g_0} \right)} - 1 \right) (1 - f_{inert})}{(1 - f_{inert}) \left(e^{\left(\frac{\Delta V}{I_{sp} g_0} \right)} \right)} \quad (7.1)$$

where;

$$m_{payload} = m_{sample} + m_{container}^{sample} \quad (7.2)$$

The specific impulse (I_{sp}) is dependant on the propellant used for that specific stage. As mentioned earlier, the inert mass fraction (f_{inert}) depends on whether an in-situ or conventional propulsion system is employed, the values of which can range from 10 – 20%.

The total trans Earth injection (TEI) ΔV equals 5.7km/s where TEI = 2.5 km/s, mid-course corrections (MCC) = 0.3 km/s, Earth orbit insertion (EOI) = 2.5 km/s and attitude/orbital control (AOC) = 0.4 km/s. These ΔV values are based on a type III or type IV trajectories. If a type II launch opportunity exists for Earth return, then the figures will be more on the order of TEI ~ 4.5 km/s, MCC = 0.3 km/s and AOC = 0.9 km/s. The type II Earth return trajectory does not include sufficient propellant for EOI and will have to either enter directly into the Earth's atmosphere or aerobrake for a rendezvous with the US STS or ISS.

The ERV will also have to expend propellant during its search for the sample container. The ERV increases velocity by decreasing its altitude which, coupled with aerobraking, will allow orbital manoeuvring. A small quantity of propellant is included for AOC.

The inert mass fraction was assumed to be 15% of the total ERV mass.

7.6.1.2 Direct Earth Return (DER)

Propellant mass required for DER is addressed in Section 7.6.2 where the upper stage of the MAV will act as the ERV.

7.6.2 Mars Ascent Vehicle (MAV)

In order to ascertain the optimum ΔV scenario for the various combinations of propellant, the 1st and 2nd stages were divided into percentiles as shown in Table 7.5.

¹⁰ Humble, Ronald, W., Henry, Gary N. and Larson, Wiley J., "Introduction to Space Propulsion", Chapter 1 in 'Space Propulsion Analysis and Design', Revised Edition, Humble, Ronald W., Henry, Gary N. and Larson, Wiley J., Space Technology Series, ISBN 0-07-031320-2, p15

Table 7.5: Percentage distribution of ?V

?V _{2nd Stage} (%)	10	20	30	40	50	60	70	80	90
?V _{1st Stage} (%)	90	80	70	60	50	40	30	20	10

Equation 7.1 provides a useful guide to estimate the propellant mass required based on a predefined payload, ?V requirement, propulsion system specific impulse and inert mass fraction. However, the inert mass fraction only takes into account the structural mass of the propellant tank(s) and feed lines.

The following sections will build on Equation 7.1 in order to produce a more accurate result.

7.6.2.1 2nd (Upper) Stage

Initial samples with a mass of about 250 grams consisting of a 1 to 2 metre long core with an additional 50 gram mass of soil with approximate 1 litre of atmospheric gases, which could be compressed and would have an insignificant mass, would suffice¹¹. But, without specific information from the scientific community at large, the mass of the sample was preset at 500 grams and the sample container was assumed to have a mass of 3 kilograms. The samples of interest will be determined at a later date but it is assumed that the payload will consist of;

- Regolith
- Rock/ores
- Sub-surface specimens
- Compressed atmospheric gases
- Atmospheric dust

Equations for estimating the mass of propulsion, power, electronics, avionics, cables and thermal control were based on factors presented by Guernsey, et al.¹². Each factor is a function the wet mass of the stage. Here, ? refers to either the upper (2nd) or lower (1st) stage.

$$m_{power}^{propulsion} = m_{Mass}^{Wet}(x) * 0.05 \quad (7.3)$$

$$m_{avionic\&cables}^{electronics} = m_{Mass}^{Wet}(x) * 0.01 \quad (7.4)$$

$$m_{control}^{thermal} = m_{Mass}^{Wet}(x) * 0.013 \quad (7.5)$$

and

$$m_{dry-mass}^{2nd-stage} = m_{sample} + m_{container}^{sample} + m_{power}^{propulsion} + m_{avionic\&cables}^{electronics} + m_{control}^{thermal} + m_{MAVFaring} \quad (7.6)$$

$$m_{wet-mass}^{2nd-stage} = m_{dry-mass}^{2nd-stage} + m_{propellant} + m_{communications}^{upperstage} \quad (7.7)$$

where the mass of the propellant, $m_{propellant}$, is calculated from equation 7.1. The mass of the upper stage communications, in equation (7.7) is only applicable for the DER scenario, since The communications system for the MOR is located on the ERV.

Since equations 7.3 to 7.7 are dependent on the wet mass of the stage and the latter is dependent on the former so the spreadsheet generates a circular reference and will continue to iterate to as many significant figures as required.

¹¹ Pillinger, Professor C., Beagle2 Project Manager, Personal Communication, 5th August 2002

¹² Guernsey, Carl S., et al., "Evaluation of Some Candidate Propulsion Technologies for Mars Ascent", 36th Aerospace Sciences Meeting, AIAA 98-0651, 1988, p4 - 5

7.6.2.2 1st (Lower) Stage

The 1st stage calculations differ from 2nd stage in several areas. For the DER scenario, the 2nd stage will include communications, which was estimated at a mass of 5 kg. During the vehicles' residence on the surface of Mars, the communication system is sufficient to converse with ESA's Mars Express. It is assumed that the Mars Express will relay any transmission from the lander to Earth, hence the very low mass assumption for on-board communications.

The 1st stage dry mass is;

$$m_{dry-mass}^{1st-stage} = m_{wet-mass}^{2nd-stage} + m_{propulsion}^{power} + m_{avionics\&cables}^{electronics} + m_{control}^{thermal} + m_{mechanism}^{2ndStageseparator} \quad (7.8)$$

Notice that the 1st stage dry mass does not include the mass of the fairing, sample and sample container simply because the entire 2nd stage wet mass is the payload of the 1st stage.

7.7 Entry, Descent and Landing System (EDLS)

The entry, descent and landing system (EDLS) consists of an aeroshell and heat shield as well as a drogue chute and primary parachute. A schematic of the entry procedure is shown in Figure 7.1.

The total MAV wet mass is simply the summation of the wet mass of the two stages;

$$m_{wet-mass}^{MAV} = m_{wet-mass}^{2nd-stage} + m_{wet-mass}^{1st-stage} \quad (7.9)$$

The total MAV wet mass, minus the CO₂ oxidiser which is envisaged to be produced on the Martian surface prior to Mars ascent, is referred to as the total MAV landed mass;

$$m_{Landed}^{MAV} = m_{wet-mass}^{MAV} - m_{Oxidiser}^{Liquid-CO_2} \quad (7.10)$$

The upper stage will be retained to the lower stage and the lower stage retained to the body of the lander by separator mechanisms. The main body of the lander will support the surface subsystems, instruments to collect the samples and the CO₂ 'manufacturing plant'. The surface support factor, such as batteries, solar cells, cables and sample acquisition equipment, was estimated as being 12% of the lander mass;

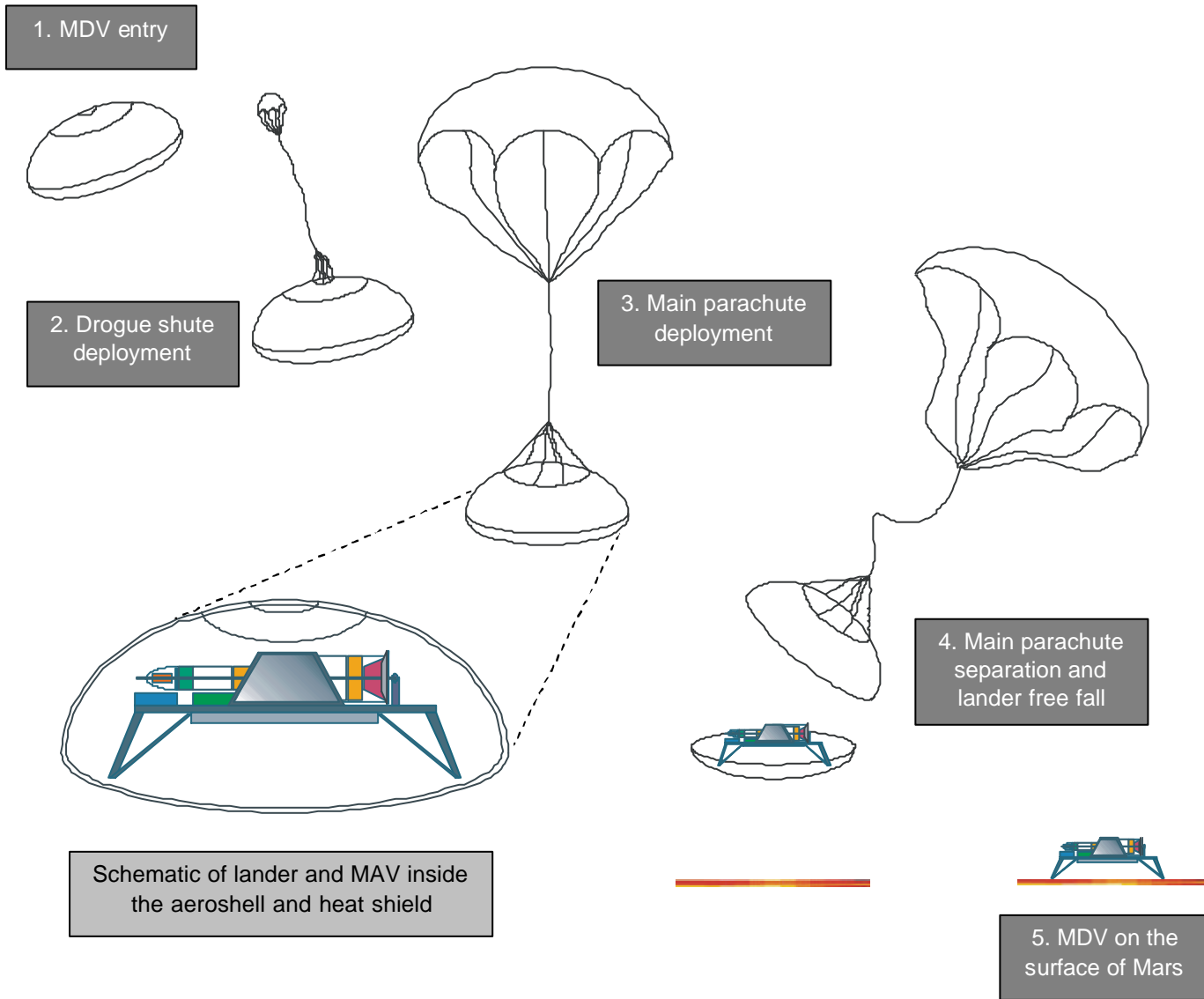
$$m_{Lander} = f_{Support}^{Surface} * (m_{Landed}^{MAV} + m_{mechanism}^{1stStageseparator} + m_{plant}^{CO_2}) \quad (7.11)$$

Figure 7.2 provides a graphical explanation of the various MAV and lander elements.

The mass of the aeroshell, which is intended to protect the vehicle during the descent phase is a factor, estimated as being 25% of the MAV wet mass plus lander mass but minus the CO₂ oxidiser;

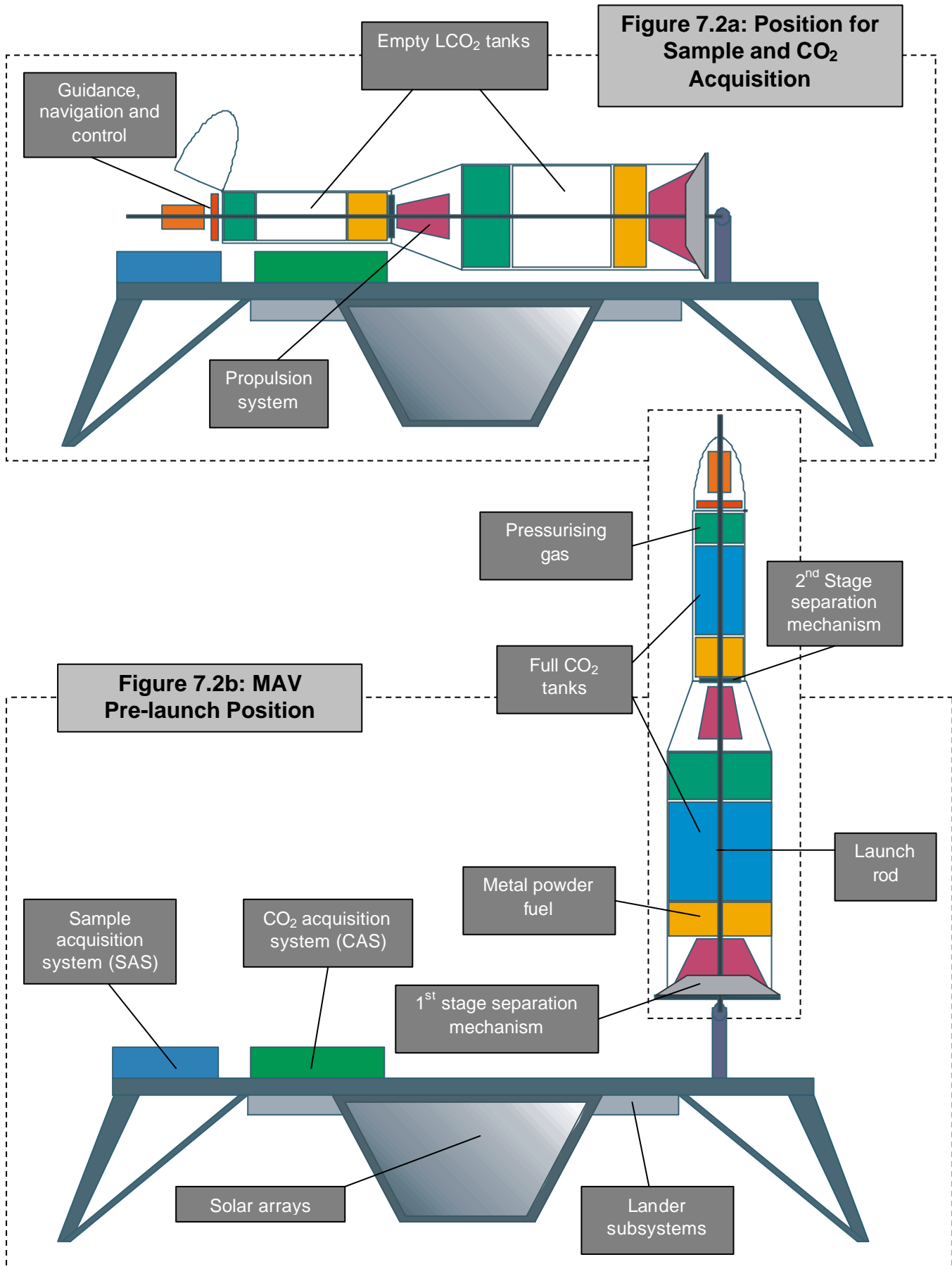
$$m_{Aeroshell} = f_{Aeroshell} * (m_{Landed}^{MAV} + m_{Lander}) \quad (7.12)$$

Figure 7.1: MDV Entry and Descent Schematic



1. The Mars descent vehicle (MDV) is seen descending through the Martian atmosphere at very high velocities, the heat shield absorbing most of the energy.
2. The drogue chute is deployed which affords a minimal reduction in airspeed, in preparation for the deployment of the main parachute.
3. Main parachute deployment.
4. The main parachute and the upper segment of the aeroshell detaches with the remaining section allowed to freefall.
5. The heat shield absorbs the energy on impact and shatters.

Figure 7.2: MAV and Lander Elements



7.8 Trans Mars Injection

The total Mars descent vehicle (MDV) mass is therefore a summation of the aeroshell mass, the MAV mass and the lander mass minus the mass of the sample;

$$m_{MDV} = m_{Aeroshell} + m_{Landed}^{MAV} + m_{Lander} - m_{Sample} \quad (7.13)$$

For the direct Mars entry-direct Earth return (DME-DER) scenario, total sample return vehicle (SRV) mass required for launch to LEO is;

$$m_{SRV} = m_{MDV} + m_{Propellant}^{Total} \quad (7.14)$$

where the propellant required for trans Mars injection (TMI) can be calculated from equation 7.1, but substituting the MDV mass in place of the payload mass and by using pertinent γ 's.

Equations for the other three scenarios, DME-MOR, OOE-DER and OOE-MOR, are addressed later. Most of the equations used in this section, with specific reference to 7.1 to 7.14 are the same, except with some minor differences.

Although the energy required for Earth orbit escape decreases as one raises orbital altitude, the actual energy required to escape from LEO is less than the energy required to raise a spacecraft from LEO to GEO¹³, but is only applicable for a type IV, low energy, trajectory. Thus, in order to keep launch costs to a minimum, the SRV, which includes the TMI propellant, will be launched into LEO. The intention here is to use either a small dedicated launch vehicle or a secondary payload on an Ariane 5 or Proton launch vehicle.

Table 7.6: Free Return Trajectories between Earth and Mars¹⁴

Departure Velocity (km/s)	Orbit Period (years)	Time to Earth Return (years)	Transit to Mars (days)	Mars Aero entry
3.34	1.5	3	250	Easy
5.08	2.0	2	180	Acceptable
6.93	3.0	3	140	Dangerous
7.93	4.0	4	130	Impossible

It is important to note that unless stated, all TMI results shown in the following sections **exclude** the propellant required to send the sample return vehicle (SRV) on its TMI. This is so the numbers are kept in a readable form. The all-inclusive TMI values are addressed in Section 7.15, launch vehicle selection.

7.9 Trajectories

The launch opportunities for a Mars bound vehicle using low energy, non-gravity assist trajectories are shown in Table 7.7, where C_3 is the square of the departure hyperbolic excess velocity.

¹³ Guernsey, Carl S., JPL Propulsion and Thermal Systems Section, Personal Communication, 6th September 2002.

¹⁴ Zubrin, R. and Wagner, R., "The Case for Mars", A Touchstone Book, ISBN 0-684-82757-3, p84

Table 7.7: Earth-Mars Outbound Trajectories^{15,16}

	Type	Earth Departure	Mars Arrival	C_3 (km^2/s^2)
1.	I	7 th June 2003	25 th December 2003	10
2.	IV	14 th November 2003	6 th February 2007	9
3.	III	21 st December 2003	18 th December 2006	11.9
4.	II	7 th August 2005	19 th July 2006	17.9
5.	I	20 th August 2005	26 th March 2006	17.8
6.	IV	21 st November 2006	18 th January 2009	10.3
7.	II	14 th October 2009	6 th September 2010	11.6

The Earth return opportunities are shown in Table 7.8 and the difference between Mars arrival and Earth departure indicates the shortest possible length of stay on the surface of Mars. The first assumption made here is that it will take 24 hours to land and also 24 hours to reach an orbit and inclination in preparation for TEI. The second assumption is that there are no GDS in progress and that landing can happen upon arrival. This residency time on the surface of Mars dictates the production rate of oxidiser and hence this determines the selection of an adsorption bed, pump/compressor or refrigerator to produce liquid CO₂, in-situ. An assumption is made that once the sample is launched into LMO, the ERV will acquire it immediately.

Table 7.8: Mars-Earth Inbound Trajectories^{15, 16}

	Type	Mars Departure	Earth Arrival	Stay Time (days)
1.	IV	24 th January 2006	2 nd August 2008	~759
2.	IV	8 th April 2006	6 th September 2008	13
3.	III	14 th April 2006	22 nd June 2008	19
4.	II	21 st July 2007	29 th April 2008	~415
5.	I	30 th July 2007	29 th February 2008	~424
6.	IV	11 th February 2008	1 st September 2010	~684
7.	II	7 th July 2009	21 st May 2010	~173
8.	II	12 th August 2011	9 th July 2012	~339

Since the objective of this unmanned sample return mission is to keep costs down to a bear minimum, yet reside on the surface of Mars for sufficient time to manufacture enough CO₂ oxidiser for Mars ascent and collect the pertinent samples, type IV missions will be the most energy efficient and therefore the most appropriate. The next available opportunity for a type IV mission is 21st November 2006 and the trajectory will take the SRV on a 2 year and 2 month flight to Mars. Upon arrival on the Martian surface, the earliest possibility that one can depart Mars for a TEI is 7th July 2009 on a type II trajectory for an arrival on Earth on 21st May 2010.

The total stay time on the Martian surface will be approximately 173 days.

An alternative scenario would entail a type II trajectory departure on the 7th August 2005, arriving at Mars on the 19th July 2006. Earth return on a type II trajectory will depart Mars on 21st July 2007 and will arrive at Earth on 29th April 2008 resulting in a residency time on the

¹⁵ Guernsey, Carl S., et al., "Evaluation of Some Candidate Propulsion technologies for Mars Ascent", 36th Aerospace Sciences Meeting, AIAA 98-0651, 1988, p6

¹⁶ Wadel, Mary F., "Benefits of In Situ Propellant Utilization for a Mars Sample Return Mission, NASA Technical Memorandum 106243, prepared for the 29th AIAA, SAE, ASME, ASEE Joint Propulsion Conference, Monterey, California, June 28-30, 1993, AIAA 93-2244, p7

Martian surface for ~365 days. This option, however, will only be possible with aerobraking both at Mars and at Earth.

7.10 Direct Mars Entry – Direct Earth Return (DME – DER)

The results for TMI masses for the DME-DER scenario are summarised in Table 7.9.

Table 7.9: Summary of DME-DER SRV Masses (kg) with Oxidiser Produced on Mars

		1 st (Lower) Stage					
		CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)
2 nd (Upper) Stage	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	2836.02	1930.13	1014.11	684.35	3872.19	536.04

Theoretical specific impulse for CO₂/metal powder combinations delivers ~200 to 330 seconds. But, realistically, without additional research and empirical data, an efficiency of 80% was assumed which reduced the overall performance of the ISRU system to values of ~160 to 265 seconds. This 20% loss was attributed to incomplete combustion, condensed phase effects, two phase flow losses and impure propellant ingredients. The BYOP storable liquid engines use higher performing propellants and did not have to bring along the additional 12 kg mass to produce CO₂ so that value was omitted from the BYOP calculations. Metal powder engine technology is a relative unknown and therefore, the inert mass fraction was set to 20% whereas in comparison, the storable liquid engines, which have enjoyed successful operation for over three decades were assumed to be 10% for the upper stage and 12% for the lower stage. BYOP storable liquids will probably use pump fed engines that minimise the structural requirements of the tank whereas the metal powder will most probably use a pressure fed system that increases the storage tank mass. However, in order to provide an accurate comparison, all engines were assumed to use chamber pressure of 200 psi (13.79 bar).

Based on the information provided in section 7.8, the residency time on the surface of Mars is 173 days. The total amount of CO₂ oxidiser, for various fuel combinations, required for this mission is shown in Table 7.10.

Table 7.10: DME/DER Propellant Requirements

	CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
CO ₂ Mass (kg)	2621.90	1598.24	1000.47	269.76	4293.99
CO ₂ Production Mass/Day (kg)	15.16	9.24	5.78	1.58	24.82
Metal Powder Fuel Mass (kg)	1020.19	652.34	205.02	272.90	1354.57

If the metal powder fuel is also produced on Mars, then the TMI reduces even further. Note that neither beryllium nor beryllium hydride are 'Martian' in-situ propellants and will still have to be brought from Earth. These results are presented in table 7.11.

Table 7.11: Summary of DME-DER SRV Masses (kg) if Propellant Produced on Mars

		1 st (Lower) Stage					
		CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)
2 nd (Upper) Stage	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	1430.55	1031.43	1014.11	684.35	2006.06	536.04

It is evident that for the DME-DER scenario, even if the both fuel and oxidiser were manufactured on Mars, Table 7.11, round trip vehicles from Earth orbit to the Martian surface cannot compete with conventional liquid bipropellants. However, developing an infrastructure using a self-sufficient source of propellant, in-situ, need not necessarily encompass the requirement to return to Earth. Future human expeditions to Mars will require unrestricted mobility that BYOP could deliver, but at a very high cost. Conventional chemical propulsion systems do, at first seem appealing for one-off DME-DER mission scenarios but will require continual refuelling from Earth for missions that require long duration exploration of the Martian surface. So long as the ISRU propulsion systems are not required to deliver high ΔV manoeuvres and remain within the reach of the CO₂ refuelling station, the, BYOP will not be able to compete. This is not exactly a revelation for it proved more economical during the early days of terrestrial exploration and the same logic and rationale for ISRU applies for long-duration space missions.

7.11 Out-of-Orbit Entry – Direct Earth Return (OOE – DER)

The OOE scenario has two options;

i) Propulsive Capture

Here, an assumption is made that the MOI ΔV budget will equal 2500 m/s which results in the total ΔV being 2800 m/s. So, the total ΔV will cater for mid course corrections (MCC) and MOI. Equation 7.1 was used to calculate the total TMI.

ii) Aerobrake Capture

The additional MOI ΔV of 2500 m/s mentioned above is omitted as the aerobrake assumes the role of MOI. The mass of the aerobrake is assumed to be a factor of 25% of the mass of the sample return vehicle (SRV) dry mass;

$$m_{SRV} = m_{MDV} + m_{Aerobrake} + m_{Propellant}^{TMI} \quad (7.15)$$

where;

$$m_{Aerobrake} = m_{SRV} * 0.25 \quad (7.16)$$

The propellant required for MCC was calculated using equation (7.1) but with the SRV mass as its payload;

The results for the aerobrake option show a ‘dramatic’ decrease in the propellant required for TMI when compared to the propulsive capture option. The result is an approximate 45% saving in TMI injection mass.

Table 7.12: Summary OOE-DER SRV Masses (kg)

		1 st (Lower) Stage						
			CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)
2 nd (Upper) Stage	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	Propulsive Capture	7543.15	5133.42	2696.75	1819.57	10299.4	1373.07
		Aerobrake	3545.04	2412.67	1267.65	855.44	4840.25	645.63

The values in table 7.10 provides, which an indication of the amount of CO₂ oxidiser and its required daily production rate for the mission, also applies to the OOE-DER scenario.

7.12 Direct Mars Entry – Mars Orbit Rendezvous (DME – MOR)

The ΔV required for the MAV in the MOR scenario is far less demanding than the ΔV required for the MAV in the DER missions. It was therefore possible to use in-situ propellants for both upper (2nd) and lower (1st) Stages. The performance is so poor that they simply cannot compete but are addressed, irrespective.

7.12.1 In-Situ Lower Stage/Conventional Liquid Upper Stage (DME – MOR)

The MOR scenario introduces the use of an Earth return vehicle (ERV). Although this specific scenario calls for a direct Mars entry, the only way that an ERV can enter Mars orbit is via an OOE manoeuvre, the propellant or aerobrake mass of which must be included in the TMI calculation. So, the DME-MOR is an adaptation of the OOE-MOR, which will be discussed in the following section. The ERV will remain in orbit around Mars, while the MDV lands on the surface, collects the samples and launches the sample container into an orbit where the ERV can locate and collect it. Due to concerns for planetary quarantine, the ERV holds sufficient propellant to cater for an Earth Orbit Insertion (EOI) for subsequent rendezvous with the STS or the ISS, but only if low energy trajectories is used.

With the DER scenarios described in section 7.9 and 7.10, the entire ERV is a part of the MAV upper (2nd) stage and must include the propellant required to send the sample on a TEI. The MAV lower (1st) stage will therefore have the added task of not only launching the sample into Mars orbit, but must also lift the propellant required for TEI which results in a phenomenal propellant mass requirement for both stages of the MAV. This explains why the DER scenarios have such high TMI masses. The MOR option alleviates this necessity since the ERV, which contains the propellant required for TEI, remains in orbit and the MAV need only achieve lower Mars orbit. The ΔV requirements for the MAV in the MOR scenario need only hold a sufficient amount of propellant to launch the 3.5 kg sample container into a low Mars orbit (LMO).

The ERV mass will require sufficient propellant not only to locate the sample container, for TEI, mid-course correction (MCC), but also EOI. The ERV's inert mass fraction is assumed to be 15%, which includes all the necessary sub-systems and uses a liquid bipropellant propulsion engine with a specific impulse of 320 seconds. The ERV mass is calculated by equation 7.1 but with the mass of the sample and the sample container as its payload.

Table 7.13: Summary of DME-MOR SRV Masses (kg) with Oxidiser Produced on Mars

		1 st (Lower) Stage					
		CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)
2 nd (Upper) Stage	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	860.18	853.48	809.07	819.10	858.17	832.86

The first thing that one will notice is that all the TMI masses for in-situ propellants are now comparable to storable liquid TMI masses. The second noticeable feature is that the performance for beryllium results in a lower TMI injection mass than its higher specific impulse delivering hydride. This is due to one reason. Although the CO₂/BeH₂ engine was optimised in such a way that 30% of the MAV's ΔV would be powered by the lower (1st) Stage and the specific impulse is much higher than the other CO₂/metal powder combinations, its stoichiometry value produces a 1:1 Oxidiser/Fuel ratio whereas the CO₂/Be has a oxidiser/fuel ratio of 4.88:1. The total amount of CO₂ oxidiser required for this mission is shown in table 7.14 and the CO₂/Be concept requires ~93% more oxidiser than its hydride. The CO₂/Be concept therefore requires less fuel and hence, a lower TMI mass.

Table 7.14: DME/MOR Propellant Requirements

	CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
CO ₂ Mass (kg)	29.42	42.96	39.51	20.48	32.94
CO ₂ Production Mass/Day (kg)	0.17	0.25	0.23	0.12	0.19
Metal Powder Fuel Mass (kg)	11.45	17.53	8.10	20.48	10.39

The daily production rate of CO₂ is so low that a very small adsorption pump or compressor system requiring a minimal power will suffice. If both fuel and oxidiser is produced on Mars, the TMI mass favours the ISRU concept. As mentioned earlier, beryllium and beryllium hydride are not available on the Martian surface and will, like the storable liquid propellants, have to be brought from Earth.

Table 7.15: Summary of DME-MOR SRV Masses (kg) if Propellant Produced on Mars

		1 st (Lower) Stage					
		CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)
2 nd (Upper) Stage	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	795.56	789.22	809.07	819.10	800.09	832.86

7.12.2 In-Situ Both Stages (DME – MOR)

For results up to and including table 7.14, only the lower (1st) stage employed CO₂/metal powder fuel engines. For MOR scenarios, the relatively low ΔV required to attain low Mars orbit (LMO) presents an opportunity to employ CO₂/metal powder systems for both stages.

Initial missions to Mars will utilise CO₂ produced in-situ on Mars but will use metal powder brought from Earth. So, literally, the concepts presented here are merely partial in-situ. The ultimate objective will be to employ both CO₂ and metals powders, mined and processed from the Martian regolith and rocks. In order to provide a comparison of what could be, the TMI masses for full in-situ propellant production is addressed forthwith.

Table 7.16: Summary of DME-MOR SRV Masses (kg) with Oxidiser Produced on Mars using In-Situ on both Stages

		1 st (Lower) Stage				
		CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
2 nd (Upper) Stage		5930.83	2709.85	1136.60	886.32	119881.94

The results for the two stage to mars orbit (TSMO) vehicle with both stages using CO₂/metal powder propellants simply cannot compete with one off sample return missions using storable liquid propellants for both stages or even with the upper (2nd) stage using storable liquid propellants and the lower (1st) stage using CO₂/metal powder propellants. Beryllium and beryllium hydride are the only fuels that can combine with CO₂ to deliver a respectable performance but their properties, either before or after combustion, have a lot to be desired. The CO₂ and fuel requirements together with the CO₂ production rate are shown in table 7.17.

Table 7.17: DME/MOR Propellant Requirements with Oxidiser Produced on Mars (kg) using In-Situ on both stages

	CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
CO ₂ Mass (kg)	2139.58	806.45	299.02	55.42	55213.69
CO ₂ Production Mass/Day (kg)	12.37	4.66	1.73	0.32	319.15
Metal Powder Fuel Mass (kg)	828.63	329.16	61.28	55.42	17417.51

Even with both fuel and oxidiser manufactured on the Martian surface, the mass that requires injection to the Martian surface is still beyond the capabilities of a small launch vehicle or an auxiliary payload of an Ariane 5 vehicle. The rationale and justification for choosing in-situ propulsion for both stages of an MAV for sample return will have to be one other than a technology demonstration because the values for CO₂/magnesium for example, are almost 3.5 times the injection mass of the storable liquid engine systems.

Table 7.18: Summary of DME-MOR SRV Masses if Propellant Produced on Mars (kg) using In-Situ on both stages

	1 st (Lower) Stage				
	CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
2 nd (Upper) Stage	2894.19	1503.59	1136.60	886.32	56052.81

7.13 Out-of-Orbit Entry – Mars Orbit Rendezvous (OOE – MOR)

OOE provides the advantage of selecting a landing site and to avoid a possible GDS but also minimises the mass requirements to return the sample back to Earth. An examination of the results show that the MAV using lower (1st) stage CO₂/metal powder propellant with an upper stage (2nd) storable bipropellant is comparable, if not superior, in performance to an MAV using storable bipropellant on both stages.

7.13.1 In-Situ Lower Stage/Conventional Liquid Upper Stage (OOE – MOR)

The explanation for the lower performing CO₂/beryllium delivering a lower TMI mass than the higher performing CO₂/beryllium hydride is provided in section 7.11.1. Even before calculations were performed, it was obvious that the propulsive capture option would deliver an unfeasible TMI mass but these results are included in order to present the aerobrake option's superiority and logic behind why this method was chosen in recent Mars missions, i.e. Mars Odyssey.

Table 7.19: Summary of OOE-MOR SRV Masses (kg) with Oxidiser Produced on Mars

		1 st (Lower) Stage						
			CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)
2 nd (Upper) Stage	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	Propulsive Capture	860.18	853.48	809.07	819.10	858.17	832.86
		Aerobrake	350.20	347.05	326.18	330.90	359.51	337.36

At first glance, one will notice that the propulsion capture values are identical to the values presented in section 7.11.1 (in-situ lower stage/conventional liquid upper stage – DME/MOR). The reason for this is because, as mentioned in a previous section, although section 7.11.1 deals with a DME scenario, it also uses an MOR for Earth return and the ERV will therefore require propellant to enter into Mars orbit.

Table 7.20: OOE/MOR Propellant Requirements

	CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
CO ₂ Mass (kg)	29.42	42.96	39.51	20.48	32.94
CO ₂ Production Mass/Day (kg)	0.17	0.25	0.23	0.12	0.19
Metal Powder Fuel Mass (kg)	11.45	17.53	8.10	20.48	10.39

The MOR option is far less demanding on the MAV's ΔV requirement. As a result, the quantity of propellant required is far less than for the DER options. This low production rate makes the in-situ propellant much more attractive as such rates should be easily achievable with CO₂ production equipment of relatively small mass.

Table 7.21: Summary of OOE-MOR SRV Masses with Propellant (kg) Produced on Mars

		1 st (Lower) Stage						
			CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)
2 nd (Upper) Stage	N ₂ O ₄ /(50% N ₂ H ₄ - 50%UDMH)	Propulsive Capture	795.56	789.22	809.07	819.10	800.09	832.86
		Aerobrake	319.84	316.86	326.18	330.90	321.97	337.36

CO₂/magnesium powder propellant is the most likely Martian in-situ propellant combination yet its mass is still 13 kg heavier than the in-situ lower stage/conventional liquid upper stage option. If, however, both propellants were manufactured on Mars, then we see by the results presented in table 7.20, that CO₂/magnesium is now much more economical than conventional liquid bipropellant propulsion systems, by over 17 kg.

7.13.2 In-Situ Both Stages

Table 7.22: Summary of OOE-MOR SRV Masses with Oxidiser Produced on Mars (kg) using In-Situ on both stages

	1 st (Lower) Stage					
		CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
2 nd (Upper) Stage	Propulsive Capture	5930.83	2709.85	1136.60	886.32	119881.94
	Aerobrake	2732.98	1021.91	480.10	362.49	56032.86

The use of in-situ for both stages is, at present, unjustifiable. Although the logic of choosing an in-situ propulsion option for Mars mobility is valid, the rationale for selecting a system for Mars ascent that is inferior in performance is not. At such an early stage of ESA's Aurora Program, it would be wise to attempt to initiate programs that will assist in the development of future infrastructure to support long duration human expeditions to Mars and, as a result, propulsion systems for an unmanned sample return using an in-situ propulsion system for the lower stage and a conventional liquid bi-propellant upper stage, irrespective of a small mass penalty, is a worthwhile risk. After all, the Aurora program is strategy related and does not simply focus on a one-off mission to return 500 grams of samples back from Mars. If the latter is the main objective of a sample return mission, then conventional bi-propellants would be the most logical option.

Table 7.23: OOE/MOR Propellant Requirements with Oxidiser Produced on Mars (kg) using In-Situ on both stages

	CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
CO ₂ Mass (kg)	2129.58	806.45	299.02	48	55213.7
CO ₂ Production Mass/Day (kg)	12.31	4.66	1.73	0.28	319.15
Metal Powder Fuel Mass (kg)	828.63	329.16	61.28	48	17417.57

The quantity of CO₂ required for an MAV using CO₂/magnesium propellant for both stages is over 70 times the quantity required for the MAV using CO₂/magnesium for the lower stage only. It is therefore advisable to attempt the in-situ option that is less taxing at first, in order to demonstrate the technology.

Table 7.24: Summary of OOE-MOR SRV Masses (kg) with Propellant Produced on Mars using In-Situ on both stages

	1 st (Lower) Stage					
		CO ₂ /Mg	CO ₂ /Al	CO ₂ /Be	CO ₂ /BeH ₂	CO ₂ /Si
2 nd (Upper) Stage	Propulsive Capture	2894.19	1503.59	1136.60	886.32	56052.81
	Aerobrake	1306.02	455.07	480.10	362.49	26088.58

Even when both metal powder fuel and CO₂ oxidiser are produced on Mars, the TMI mass cannot compare with conventional liquid propellants, for one-off missions. However, as Shafirovich et al. have quite rightly pointed out, Mars mobility is a requirement for any program of exploration and with an increasing ΔV , one will find that the conventional propellants – which must be brought from Earth – will not be able to compete with in-situ (CO₂/magnesium metal powder) propulsion systems. The ISRU concept will require several fully operational CO₂ production and metal mining and powder processing facility, strategically located over the surface of Mars.

7.14 Mission Scenario Conclusions

A summary of all of the tables is reproduced in table 7.25, below. Propulsion systems using in-situ propellants, CO₂/magnesium and CO₂/aluminium, for both upper and lower stages in the DER options are simply not possible. The conventional storable liquid propulsion system option can achieve a DME/DMR with a TMI mass, excluding TMI propellant, for just over 515 kg. But, this all depends on the logic of the space agency or agencies that wish to embark on a Mars sample return mission. If a 'grab and go' mission is all that is required and the objective is to simply bring back a sample from Mars as soon as possible, then the logical choice would be to choose the conventional liquid bi-propellant option for both stages of the MAV. This type of mission, however, does cause some concern with regards to the selection of landing site and the possible loss of the spacecraft/lander if it arrives at Mars and attempts a landing during a GDS. The only competitor to this mission option, in terms of mass savings, is with the use of the same propellants for both upper and lower stages but in an OOE/MOR scenario that, in comparison, has a TMI mass of just over 337.36 kg; excluding TMI propellant mass. The OOE with aerobrake option affords the opportunity to arrive at Mars, enter into a LMO and then pre-select a landing site with prior consultation with planetary geologists. Also, the ERV remains in orbit around Mars and awaits the launch of the sample container into a LMO. The MAV will have to seek out and rendezvous with the sample container, which could take months or years to locate – extending the mission's length, which will increase the mission's cost. However, the fact that the ERV remains in orbit around Mars alleviates the necessity for the upper stage of the MAV to store, and carry the total propellant mass (required for Earth return) during Mars ascent. Carrying the Earth return propellant from the surface would increase the propulsive requirements and therefore mass of the lower stage and is the reason why the DER options are so inefficient in terms of mass when compared to the MOR scenarios.

Even though a MAV propulsion system using in-situ for both the upper and lower stages is achievable, the fact remains that it will be difficult to justify the selection of such a system based solely on the logic of the development of infrastructure. An in-situ propellant option for both stages on the OOE(aerobrake)/MOR scenario is over 8 times the mass of a system that uses conventional liquid bi-propellants for both stages in the same scenario. However, introducing the a CO₂/magnesium lower (1st) stage with a conventional liquid bi-propellant upper (2nd) stage does afford a comparable performance with the former having a TMI mass, excluding TMI propellant, of only 13 kg higher and could provide the justification to develop the required technology while increasing the chance to succeed in a sample return mission using ISRU.

Table 7.25: Sample Return Vehicle Dry Mass¹⁷ (from LEO) Comparison (manufacturing Oxidiser on Mars)

	2 nd Stage	LOX/CH ₄ ¹⁸		CO ₂ /Mg		CO ₂ /Al		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)	
	1 st Stage	LOX/CH ₄ ¹⁸		CO ₂ /Mg		CO ₂ /Al		CO ₂ /Magnesium		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)	
Direct Mars Entry/Direct Earth Return (DME/DER)		140.88	50%	-		-		2836.02	70%	516.49	50%
			50%						30%		
Out-of-Orbit Entry /Direct Earth Return (OOE/DER)	Propulsive Capture	373.93	50%	-		-		7543.15	70%	1372.07	60%
			50%						30%		40%
	Aerobrake	176.12	50%	-		-		3545.04	70%	645.63	60%
			50%						30%		40%
Direct Mars Entry/Mars Orbit Rendezvous (DME/MOR)		685.86	60%	5930.83	50%	2709.85	50%	860.18	80%	832.86	60%
			40%		50%				50%		20%
Out-of-Orbit Entry/Mars Orbit Entry (OOE/MOR)	Propulsive Capture	685.86	60%	5930.83	50%	2709.85	50%	860.18	70%	832.86	60%
			40%		50%				50%		30%
	Aerobrake	268.29	60%	2732.98	50%	1021.91	50%	350.20	70%	337.36	60%
			40%		50%				50%		30%

¹⁷ Excluding Carbon Dioxide which assumed will be manufactured on Mars

¹⁸ Assumes 25% of Propellant is LH₂, brought from Earth but does not consider the CH₄ and LOX production plant mass required to manufacture the chemicals

Table 7.26: Sample Return Vehicle Dry Mass¹⁹ (from LEO) Comparison (manufacturing entire Propellant on Mars)

	2 nd Stage	LOX/CH ₄ ²⁰		CO ₂ /Mg		CO ₂ /Al		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)	
	1 st Stage	LOX/CH ₄ ²⁰		CO ₂ /Mg		CO ₂ /Al		CO ₂ /Magnesium		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)	
Direct Mars Entry/Direct Earth Return (DME/DER)		140.88	50%	-		-		1430.55	70%	516.49	50%
			50%						30%		
Out-of-Orbit Entry /Direct Earth Return (OOE/DER)	Propulsive Capture	373.93	50%	-		-		3804.51	70%	1373.07	60%
			50%						30%		40%
	Aerobrake	176.12	50%	-		-		1788.20	70%	645.63	60%
			50%						30%		40%
Direct Mars Entry/Mars Orbit Rendezvous (DME/MOR)		685.86	60%	2894.19	50%	1503.59	50%	795.56	80%	832.86	60%
			40%		50%		50%		20%		40%
Out-of-Orbit Entry/Mars Orbit Entry (OOE/MOR)	Propulsive Capture	685.86	60%	2894.19	50%	1503.59	50%	795.56	70%	832.86	60%
			40%		50%		50%		30%		40%
	Aerobrake	268.29	60%	1306.02	50%	455.07	50%	319.84	70%	337.36	60%
			40%		50%		50%		30%		40%

¹⁹ Excluding Carbon Dioxide and Metal Powder Fuel which assumed will be manufactured on Mars

²⁰ Assumes 25% of Propellant is LH₂, brought from Earth but does not consider the CH₄ and LOX production plant mass required to manufacture the chemicals

7.15 Launch Vehicle Selection

This section provides a brief insight into the possible launch vehicles that would fulfil the task of carrying the SRV into LEO in readiness for TMI. Unlike the first 14 sections of this chapter, section 7.15 will include the propellant mass required to send the SRV on a TMI.

Table 7.27: Launch Vehicle Options²¹

Country	Vehicle	LEO capability (kg)	€value (in millions)
Europe	Ariane 4 series	5,000 – 10,200	65 – 125
	Ariane 5	18,000	150 – 180
China	LM-2E	9,500	45 – 55
	LM-3B	11,200	50 – 70
India	PSLV	3,700	15 – 25
	GSLV	5,000	35 – 45
Japan	HII	10,060	165 – 170
	HIIA202	9,940	75
Russia	Proton / K Block DM	19,760 / 21,000	90 – 112
	Rocket	1,800	13 – 15
	Soyuz U / ST	7500 / 7800	30 – 50
Ukraine	Dnepr 1 / M	4500 / 4140	10 – 20
	Tskilon 2 / J	3350 / 4100	20 – 25
	Zenit 2	13,500	35 – 50
United States	Athena	2,065	22 – 26
	Atlas series	7,316 – 20,050	75 – 110
	Delta series	5,140 – 13,600	45 – 110
	Titan II	1,900	30 – 40
	STS	24,400	300

7.15.1 DME-DER

Out of all of the options, a conventional liquid bi-propellant propulsion system for both stages proves to be the only realistic option with an all-inclusive TMI mass of 2129.48 kg. At this mass, two launch vehicles are available. The first is the Ukrainian Tsiklon 2 which can lift a total of 3350 kg to LEO for €20 to 25 million and the second option is the Indian PSLV which is capable of lifting 3700 kg to LEO for €15 to 25 million.

7.15.2 QOE-MOR

Here we have two possible propellant options. The first being the conventional liquid bi-propellant in both stages and the second being the conventional bi-propellant for the upper stage with a CO₂/magnesium propulsion system for the lower stage. The all-inclusive TMI masses are 1683.04 kg and 1747.16 kg respectively; a 64 kg difference.

²¹ Isakowitz, Steven J., et al., "Space Launch Systems", 3rd Edition, AIAA Publication, ISBN 1-56347-353-4, 1999, pp549

The Russian Rockot, with a maximum lift capability of 1800 kg to LEO could lift both of these masses in a dedicated launch for €13 to 15 million. The Ukrainian Dnepr can lift twice the payload and is more cost effective at €10 to 20 million.

Table 7.28: Sample Return Vehicle Wet Mass²² (from LEO) Comparison (manufacturing Oxidiser on Mars) including TMI Propellant

	2 nd Stage	LOX/CH ₄ ²³		CO ₂ /Mg		CO ₂ /Al		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)	
	1 st Stage	LOX/CH ₄ ²³		CO ₂ /Mg		CO ₂ /Al		CO ₂ /Magnesium		N ₂ O ₄ /(50% N ₂ H ₄ – 50% UDMH)	
Direct Mars Entry/Direct Earth Return (DME/DER)		582.44	50%	-		-		11754.80	70%	2139.48	50%
			50%						30%		50%
Out-of-Orbit Entry /Direct Earth Return (OOE/DER)	Propulsive Capture	-	50%	-		-		-	70%	-	60%
			50%						30%		40%
	Aerobrake	877.65	50%	-		-		17704.44	70%	3222.72	60%
			50%						30%		40%
Direct Mars Entry/Mars Orbit Rendezvous (DME/MOR)	-	-	60%	-	50%	-	50%	-	80%	-	60%
			40%		50%		50%		20%		40%
Out-of-Orbit Entry/Mars Orbit Entry (OOE/MOR)	Propulsive Capture	-	60%	-	50%	-	50%	-	70%	-	60%
			40%		50%		50%		30%		40%
	Aerobrake	1338.02	60%	13648.41	50%	5891.02	50%	1747.16	70%	1683.04	60%
			40%		50%		50%		30%		40%

²² Excluding Carbon Dioxide which assumed will be manufactured on Mars

²³ Assumes 25% of Propellant is LH₂, brought from Earth but does not consider the CH₄ and LOX production plant mass required to manufacture the chemicals

8.0 Propellant Feed Issues

Propellant feed issues are entirely dependant on the configuration of engine. There are several possible schematics for a CO₂/metal propellant propulsion system, and not all of them use metal fuel in powder form.

8.1 Fuel System

8.1.1 Liquid Metal Injection

The principal of this design is to liquefy the metal powder prior to transferring and injecting the fuel into the combustion chamber. The advantage of such a system would be that the fuel could be atomised to droplet sizes smaller than the metal particles that what one would find in a solid rocket motor. This would result in a faster burn rate and subsequent combustion. The objective is to minimise two-phase flow losses and incomplete combustion hence contributing to an overall increase of efficiency of the propulsion system. Previous analysis of chamber cooling shows that a pump fed system will have to be employed for this task.

Unfortunately, serious problems face this concept. Due to the high temperatures envisaged in the metal fuel feed system, plus the pertinent cooling, any residual liquid metal fuel may be subject to solidification in the feed pipes and injector interface. Designers envisage the use of inserts to protect the injector tubes and engine nozzle against erosion but then one would have to address issues of maintenance, especially if one wishes to reuse the propulsion system for multi-mission scenarios. But, probably the greatest technological challenge associated with this concept is the energy required to produce molten liquid metal and sustain the metal in that chemical state. It is also worthy to note that even though the design is intended to circumvent the problem of metal powder feed by giving it liquid form, this does not entail that the engine will work in the same way as a conventional liquid propulsion system. Canadian researchers note that the boiling point of magnesium (1373.15 K) is considerably higher than conventional liquids (623.15 K)¹ which implies that the combustion chamber will have to be larger to cater for the longer combustion process otherwise one would have to contend with a low efficiency propulsion system.

8.1.2 Hybrid Engine

This configuration is probably the most simplistic, in terms of technological know-how. If one were to assume that the metal powder fuel were taken from Earth to Mars in a pre-defined mould within the combustion chamber, similar to composite rocket propellant in solid rocket boosters, then only the CO₂ oxidiser will require acquiring, storage, feeding and injection, the technology of which already exists. The fuel grain can take two forms;

1. The metal could be premixed with a HTPB binder and set with a curing agent to maintain integrity. Adding HTPB will lower the percentage fuel content and will reduce specific impulse so the addition of an energetic plasticiser could increase the performance close to the original 'pure state' performance of the fuel.
2. The metal powder could be pneumatically pressed into a predefined shape.

The risk factor associated with this configuration is therefore minimised with the omission of one (fuel) tank and the associated feed lines. As with all hybrid concepts, the system can be throttled by altering the oxidiser flow rate and shut down during operation in case of an emergency.

There are however, several inherent problems associated with this concept. Studies on LOX/aluminium engines have shown that the rate of regression of the metal could be difficult to control². The aluminium fuel grain would act as a heat sink due to its high conductive properties and this would result in melting more metal than required which could result in over

¹ Goroshin, S., Higgins, A. J. and Lee, J. H. S., "Powdered Magnesium-Carbon Dioxide Propulsion Concepts for Mars Missions", AIAA 99-2408, p6

² Meyer, Michael L., "Design Issues for Lunar In-Situ Aluminum/Oxygen Propellant Rocket Engines", NASA Technical Memorandum 105433, AIAA-92-1185, p10

pressurisation leading to engine failure. Magnesium has a lower melting point and therefore would be an even greater concern.

8.1.3 Monopropellant Engine

This process employs a method where the metal powder fuel is premixed and stabilised in a liquid oxidiser to form a 'monopropellant'. Only one tank, plus its associated feed lines will be required reducing costs and increasing simplicity of maintenance.

Experimental studies by Wickman et al. in the 1990's showed that liquid oxygen can be safely mixed with metal powder fuels³. The monopropellants were subjected to various tests to determine if the propellant combinations would detonate, how the burn rate would perform under ambient and pressurised conditions and to gauge the viscosity of the propellant. There are in effect two possibilities for feeding monopropellants from the propellant tank and into the combustion chamber; by use with or without a stabilising agent, which in this case was Cab-O-Sil (SiO₂). The first option requires the addition of a chemical additive of about 0.02 microns in spherical diameter to the propellant formulation to prevent settling; but the integrity of the solution would only be temporary. Although a stabilising agent was shown to prove beneficial, settling of the monopropellant does eventually occur. These tests however were on a propellant formulation under static 'launch pad' conditions. Acceleration forces during launch would 'force' the heavier particles within the propellant solution in the opposing motion to the acceleration, i.e. towards the rocket engine end of the propellant tank. Without an adequate device to continually mix the monopropellant, the initial monopropellant that enters the combustion chamber will be very fuel rich, if not a sludge and the last quantity of monopropellant leaving the tank will probably be close to 100% liquid oxidiser. The second option would entail the use of a minimal amount of liquid oxidiser, which is stored separately, sufficient to fluidise the metal fuel to provide a flowing motion. The propellant would be stored as a fuel rich monopropellant and the remaining oxidiser would be injected into the combustion chamber, under a different feed mechanism. The latter concept resembles the 'bipropellant' option but has one major difference. This option uses the propellant's oxidiser to provide the flow of propellant and is in liquid form whereas the concept described in section 8.1.5 uses a benign carrier gas. It was shown that for monopropellant formulations, metal particles less than 60 microns tended to bundle together whereas this tendency was all but eliminated for sizes of 100 microns and above.

Burn rate tests would have to be performed under both ambient and pressurised conditions to determine if a concern with 'flash back' exists. If for any reason, the combustion chamber pressure spikes above normal operating conditions, then there is possibility that the flame front would work its way back through the feed lines and into the propellant tank, causing catastrophic failure. It seems only logical that a high feed pressure system is used but that would increase mass of the propellant tank (if pressure fed) plus the mass of the feed lines. One area that has not been addressed with monopropellant concepts is cooling. Since the metal powder is premixed with the oxidiser, it seems unlikely that regenerative cooling will be employed due to safety reasons. An alternative method would be to store the monopropellant at a high fuel to oxidiser mixture ratio, i.e. fuel rich. So, if regenerative cooling were to be used, then one way would be to have a separate tank of CO₂ liquid to cool temperature critical components and then, prior to injection, be used to adjust the mixture of the monopropellant to a stoichiometric value, if required. This proposal was addressed for a LOX/aluminium powder 'Lunar soil propellant' concept⁴. The metal particles would have a predefined size to ensure total combustion and will therefore not have to be atomised. However, if the CO₂ remains in its liquid state, it will require atomisation and this will complicate the injector design. The metal powder will be premixed with the CO₂ and therefore it may prove difficult to atomise the oxidiser to the required size. Clogging in the feed lines and injector will be a major technological hurdle and is one that will have to be addressed if such a concept is to see the light of day.

³ Wickman, John H. and James, E. "Gelled Liquid Oxygen/Metal Powder Monopropellants", AIAA 92-3450

⁴ Wickman, John H., "Design of a LOX/Aluminum Lunar Propellant Rocket Engine", 30th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, June 27-29, 1994, Indianapolis, p3

The concepts reviewed thus far assume that the CO₂ will be in liquid form. Of course, for reasons of density and therefore, storage, this would be beneficial. It is more than likely that the liquid CO₂ will convert to a gaseous state, not long after leaving the storage tank and could therefore be used to pressurise the metal powder tank and assist the flow of powder from the tank. An additional feed line of CO₂ is envisaged to meet up with the metal powder fuel line in order to act as a regulator and increase or decrease the CO₂ content in the 'CO₂ gas/metal powder fuel' monopropellant.

Using CO₂ as the carrier fluid as well as the oxidising fluid could lead to the possibility of flashback and will have to be empirically determined. With sufficient safety precautions such as pressure transducers strategically positioned in the feed lines, for example prior to the injector, intricate monitoring of the pressure data could instruct valves to cut off both the fuel and oxidiser flow followed by purging the feed lines of any excess metal powder and CO₂. Figure 5.2 shows that N₂, which was the inert gas used for Wickman's CO₂/magnesium metal powder engine tests is just as reactive with magnesium metal powder so one could conclude that because flashback never occurred during any one of the series of Wickman's tests, then it is unlikely that the use of CO₂ as the carrier fluid and oxidising fluid would cause flashback.

8.1.4 Pneumatic Feed

This was the preferred method of powder feed by Loftus et al. and subsequently by Linne and Meyer. The fuel tank was a modified hydraulic cylinder. With the fuel tank valve turned off, a small amount (~1% of fuel component mass) of carrier or fluidising gas such as nitrogen was introduced into the fuel tank to fluidise the metal powder fuel. The pneumatic piston is compressed slightly so when the valve is opened, the carrier fluid and metal powder will flow into the feed lines and at the same time the pneumatic piston ensures positive feed pressure on the powder. The use of a pneumatic piston in conjunction with a small quantity of fluidising gas reduces the amount of carrier gas requirement by ~5%.

The use of a pneumatic feed for delivery of the metal powder in to the combustion chamber is advantageous for several reasons. Fine powder would prove extremely beneficial to the combustion process, which would limit two-phase flow loss however, particles smaller than 10µm have a tendency to stick together and have proven difficult to transport⁵. Throttling as well as engine shutdown and restarts are possible with this configuration but the feed lines will have to be purged with an inert gas prior to any restart.

The negative aspect of the pneumatic feed system is a separate component from the pump or pressure fed oxidiser and therefore adds inert mass to the system as well as additional power for its operation. The friction from the metal powder particles and the metal feed tubes would cause undesirable heating and if not addressed by adequate cooling methods, could result in metal particles melting in the feed tubes. There is also the added problem of feed line erosion due to friction which is also a problem experienced by pressure fed feed systems.

8.1.5 Pressure Feed

This was the method used by Wickman during test firing of a 'Martian' in-situ propulsion system under a Phase II SBIR contract for NASA Marshall Spaceflight Center/Jet Propulsion Laboratory. A number of carrier fluids were examined, including gases and liquids. The criteria for trade-off for the carrier fluid selection were as follows;

- 1) Evaporation rate
- 2) Compatibility with magnesium
- 3) Low molecular weight
- 4) Influence on combustion temperature
- 5) Influence on specific impulse

Liquids were found to resemble the problems experienced with the monopropellant configuration, i.e. clogging etc. so attention was focussed on the gases. Two categories of

⁵ Meyer, Michael L., "Design Issues for Lunar In-Situ Aluminum/Oxygen Propellant Rocket Engines", NASA Technical Memorandum 105433, AIAA-92-1185, p11

fluidising gas exist; those that improve performance and those that tend not to react with the metal fuel. The three gases that were examined were hydrogen, helium and nitrogen. All three improved the performance (specific impulse) substantially, nitrogen was eventually chosen as the carrier fluid for three reasons. 1) gases such as hydrogen and helium could afford ignition in the feed lines, 2) due to the low density of both hydrogen and helium, these fluids would require storage in cryogenic liquid form and additional system mass is not warranted and 3) the eventual objective is to use readily available resources for 100% in-situ propulsion systems which leaves the choice of carbon dioxide, nitrogen or argon. Initially, it is envisaged that the metal powder fuel plus the carrier fluid will be taken to Mars from Earth, but if we were to assume that we should only use resources available on Mars, then that would automatically disqualify helium as a potential candidate. Nitrogen is the most suitable candidate for its benign properties and storability. Hydrogen has been found to exist in the subsurface of the Martian regolith but this has not yet been confirmed, in-situ.

Linne and Meyer documented one not-so-reassuring observation regarding this method of feeding powder. Inspections showed that with the pressurising of the metal powder within the fuel tank, occasional air pockets formed which caused 'bubbling'. This resulting in a variation in feed pressure resulting in inconsistent flow. Although Wickman made no such observations during experimental test firings, one should not rule out that this could be a possible explanation for a contributing factor for the fluctuating chamber pressure observed in the results of the engine tests.

8.1.6 Pump fed

Pressure fed propulsion systems require thick wall propellant tanks in order to withstand the high feed pressures and so the principal of this concept is to reduce structural mass of the propulsion system. The principal requirement would be to ensure high efficiency to justify the employment of such a system. Japanese researchers performed subscale experiments in the early 1970's on metal powder turbojets and they found that additional research in this area was not advisable due to clogging in the blades⁶. Wickman et al. came to the same conclusion in the late 1990's when working on a concept to propel Mars aircraft by a turbojet using CO₂ and magnesium metal powder propellant combination.

8.2 ***Oxidiser System***

The primary technological concern is to address the feed issues associated with the metal fuel. The oxidiser, however, is a far less taxing issue, as the technology to feed liquid oxidisers already exists. As mentioned in Section 3, the carbon dioxide will be processed from the atmosphere of Mars via a sorption bed, mechanical roughing pump or a refrigerator. It is more than likely that the CO₂ will be stored in the form of a liquid and for the sake of simplicity, the system selected to feed the oxidiser will more than likely be based on the option chosen for the fuel. Since the pump fed system cannot be used for feeding metal powder, the system will have to be pressure fed.

8.3 ***Metal Powder Feed Issues***

Pneumatic and pressure fed concepts mentioned above employ a non-standard propulsion technology component; the method of transporting metal powder from the fuel tank into the combustion chamber. The feed of metal powder or the flow rate is a function of the method of storage, injection and combustion. It is also dictated by the particle size distribution and shape but also the properties of the metal powder such as coefficient of friction and in some cases, by the metal's electrostatic and magnetic characteristics. Adsorbed moisture can also be a contributory factor associated with the inhibition of free flowing particles.

The following sub-sections are dedicated to addressing these issues.

⁶ Tamura, K., Kohno, M. and Akiba, R., "A Feasibility Study of a Jump Pump for Powder Rockets", Proceedings of the 11th International Symposium on Space Technology and Science, June 30th – July 4th, 1975, pp117-122

8.3.1 Fuel Tank

The fuel tank design will differ depending on the chosen method of metal powder feed. Hybrid concepts will employ standard liquid feed systems and liquid metal and monopropellant concepts, although unique in their own way, will involve adaptations of current liquid engine feed systems and will not be addressed here. The concepts of specific interest in this report are metal powder storage, feed and injection. Linne and Meyer proposed four methods of metal powder feed;

1. Fluidised bed
2. Screw/worm feed
3. Fluidised hopper
4. Fluidised piston

The fluidised bed is probably the simplest and it was the method used by Wickman in all 'Martian' in-situ engine experiments. No known problem regarding clogging was encountered but this is probably due to the fact that the standard liquid injectors were not used. A screw/worm feed uses a helical system to transport the powder from one location to another but the feed rate is extremely slow. The fluidised hopper is similar to the fluidised bed except it draws the metal powder fuel from the centre of the tank. The fluidised piston is another term for the pneumatic piston feed method.

8.3.2 Injectors

Three types of injector designs have been considered. Loftus et al. used two concepts, both of which were modelled on liquid engine technologies. The third method was introduced by Wickman, the design of which was based on a compressor gas/metal powder manufacturing technique. For all three designs, cold flow tests provided the necessary feedback for injector calibration and the quantity of inert gas required to fluidise the metal powder within the fuel tank. The results established the metal powder flow rates to obtain optimum conditions for engine operation. All the injector designs underwent performance and stability tests to determine its disposition. This was achieved by the use of benign objects with similar characteristics as metal powder in terms of particle size and density. Once the injectors had been calibrated, the motion of live firings took place to determine its efficiency. It was found that fluctuation in pressure occurred for $\pm 88\%$ in the experiments of Loftus et al. for ammonium perchlorate/aluminium propellants, Wickman, $+15.6\%$ / -8.6% for CO_2 /magnesium propellants and Meyers, $+30\%$ / -27% for gaseous oxygen/aluminium propellants. Loftus and his team at Bell Aerospace introduced methane as a carrier fluid and pressure oscillations decreased from $\pm 88\%$ to $\pm 12\%$. The pressure variance was a result of low flame temperature experienced during the combustion process. It was determined that with a constant feed pressure, the initial low chamber pressure allowed an excess amount of metal powder into the chamber, which was followed by a large pressure spike above mean chamber pressure, due to 'over' combustion, which inhibited the entry of additional powder that was being fed at a predefined feed pressure. This resulted in a pressure drop below mean chamber pressure. The result was a rapid fluctuation of pressure, which was visible when observing the video footage of the engine tests. The variation of pressure during combustion was confirmed after examining the readings acquired from the pressure transducers, which displayed a sinusoidal pattern. The curves, however, were not smooth but were more like a jagged. In all cases, the use of small particles or the introduction of a reactive fluidising gas resulting in near-smooth combustion, albeit the former did prove detrimental to free particle flow and showed a tendency to want to clog in the injector manifold. In this study, we assume the use of only in-situ gases; carbon dioxide, nitrogen and argon.

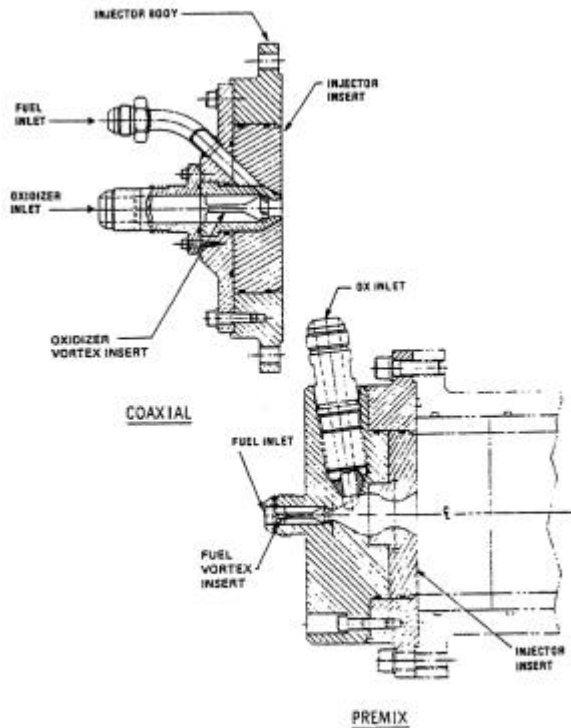


Figure 8.1 : Coaxial and Premix Powder Injectors

oxidiser prior to injection. These injectors were designed for ammonium perchlorate and aluminium powder but could be used to minimise the two-phase flow experienced between the carrier fluid and the metal particles and to enhance the mixing process between the oxidiser and metal particle fuel. Feed pressure drops and pressure oscillations for the coaxial injector were high. This was attributed to the relative uneven mix of fuel and oxidiser and irregular combustion. However, the manifold in the premix injector introduced flow paths of changing internal diameters, which encouraged pressure variations. This resulted in a reduced flow rate and a larger pressure drop than experienced with the coaxial injector. Akiba found that the carrier gas flow rate was directly proportional to the orifice pressure drop;

$$\dot{m}_{Gas}^{Carrier} \propto \Delta P \quad (8.1)$$

and the gas flow rate was directly proportional to the factor of 0.7 of the orifice pressure drop;

$$\dot{m}_{particle} \approx \propto \Delta P^{0.7} \quad (8.2)$$

It should be stated that every known metal powder engine test has never operated more than 5 seconds in duration. Loftus et al.'s engines operated for 2 seconds, Akiba's for 5 seconds, Linne and Meyer's for ~1 seconds and Wickman for 8 seconds but with ~3.5 seconds of continued combustion. Due to the limited burn time plus combustion fluctuation experienced, it was not possible to obtain any detailed data to analyse injector performance. It is possible that a staged combustor or pre-burner, as proposed by Goroshin, will enhance the combustion process but additional experimentation will be required to prove this suggestion. An additional method for improving combustion would be to apply reverse flow, which is where the fuel or oxidiser is injected from the nozzle end towards the rear end of the combustion chamber. The latter method would have to fight against the force of the MAV's gravitational acceleration lending the design to requiring additional feed pressure; an undesirable option because it would necessitate increased pressure for the resulting carrier gas and component mass for pressurised tanks and feed lines, which will have to be engineered to cater for the higher pressure requirements.

Besides serving as a device to mix the fuel and the oxidiser, the metal powder injector requires no intricate design for atomising the metal particles since the size will be predefined for flow characteristics, burn rate and subsequent combustion chamber sizing. The liquid CO₂ will require atomisation but if a regenerative cooling system is used, then there is a possibility that the configuration can be designed so that the heat flux will afford phase changes from CO₂ liquid to CO₂ gas, prior to injection. This would simplify matters greatly and could allow the simplistic injection option as presented by Wickman. However, it is more than likely that CO₂ phase change will occur as soon as the fluid leaves the oxidiser tank.

The coaxial and premix injectors used by Loftus et al. are shown in Figure 8.1. The fuel and oxidiser feed lines on the coaxial injectors simply converge towards the mouth (orifice) of the injector, i.e. prior to the injector meeting the combustion chamber face plate. The fuel and oxidiser feed lines on the premix injector introduces a manifold (a small area with a miniature throat) and nozzle which encourages premixing of the fuel and

8.4 Summary of Metal Particle Fuel Sizes

The advantages of metal powders are that the systems can be easily throttled as well as stopped and restarted, if required. The mass flow rates are insensitive to extreme environmental conditions, which for case of Mars will prove advantageous. The particle sizes are induced by the method of packing and combustion. As mentioned earlier, combustion efficiency increases as the particle sizes decrease but packing density has been proven to have a greater result with a mixture of two sizes, one large and the other small. However, sufficient breathing space must exist between the particles if one is to entrain fluidising gas into the powder fuel tank. If incorrect sizes of metal powder are chosen, then there is a danger of packing the metal powder so tight, that it will exhibit flour like tendencies and stick together which is thought to be caused by electrostatic and adhesive forces. Magnesium combustion temperature will also have a hand to play in the selection of fuel particle sizing but the combustion temperature is also a function of the particle resident time within the combustion chamber. It is therefore safe to conclude that the combustion chamber parameters will have a direct effect on the choice of particle size and vice versa. Suffice to say, past researchers were either content with -325 mesh or 44 micron (Wickman), a 70/30 mix of 30 μ /3 μ (Loftus et al.) or >10 μ m (Meyer), with all researchers agreeing on the particle configuration as being spherical in nature.

9.0 Proposed Engine Design

9.1 Propellant Tank Sizing

In order to size the engine and also determine what type of cooling mechanism is used, it was necessary to find the duration of the burn. But, several other steps were necessary before the burn duration could be realised.

The diameter of the rocket is an important parameter required to calculate the rocket's flight. For the chosen scenario, the MAV 1st stage is required to reach an altitude of 28 km and it was determined that 29.42 kg of liquid CO₂ and 11.45 kg of magnesium is necessary to fulfil this task. The density of the chemicals are 1014 kgm⁻³ (at 256.15 K) and 1738 kgm⁻³, respectively, and with these figures it was possible to calculate the following volumes, presented here as series of heights and diameters, Table 9.1.

Table 9.1: Propellant Tank Dimensions

LCO ₂ Tank Dimensions		Metal Fuel Tank Dimensions		Rocket Body Diameter
Height (m)	Diameter (m)	Height (m)	Diameter (m)	Diameter (m)
0.1	0.61	0.1	0.41	1.22
0.2	0.43	0.2	0.29	0.86
0.3	0.35	0.3	0.24	0.70
0.4	0.30	0.4	0.21	0.61
0.5	0.27	0.5	0.18	0.55
0.6	0.25	0.6	0.17	0.50
0.7	0.23	0.7	0.16	0.46

Since both tanks have relatively small diameters, they were put side by side instead of in tandem. The rocket body diameter was the summation of these two diameters with an additional 20% to allow for the rocket body wall thickness and allowing for a passage for the feed lines from the pressure tank to the rest of the rocket engine. Since the MAV's centre of pressure must be at least one body diameter from its centre of gravity, the decision was made to choose the tanks with the 0.7 metre height. This selection had an added benefit since the diameter was the smallest and this meant that the MAV would experience less drag during ascent.

9.2 Flight Burn Time

A simple 2 dimensional flight trajectory program was used to ascertain the rocket engine's burn duration for a given thrust to weight ratio, which was assumed to be 3:1. A conservative drag coefficient of 0.4 was assumed for a conical nosecone. Since the parameters such as the MAV wet mass, MAV inert mass, propellant mass were known, the burn duration was simply altered until the altitude of 28km was reached. The results are shown in table 9.2 and graphically presented in figure 9.1.

Table 9.2: Results from Flight Program for MAV 1st Stage

Statistic	Result
Maximum Velocity (m/s)	697.02
Maximum G Force	3.742
Maximum Altitude (m)	28614.59 (+572.99)
Time of Maximum Altitude (s)	90.95
Engine Burn Duration (s)	25.45

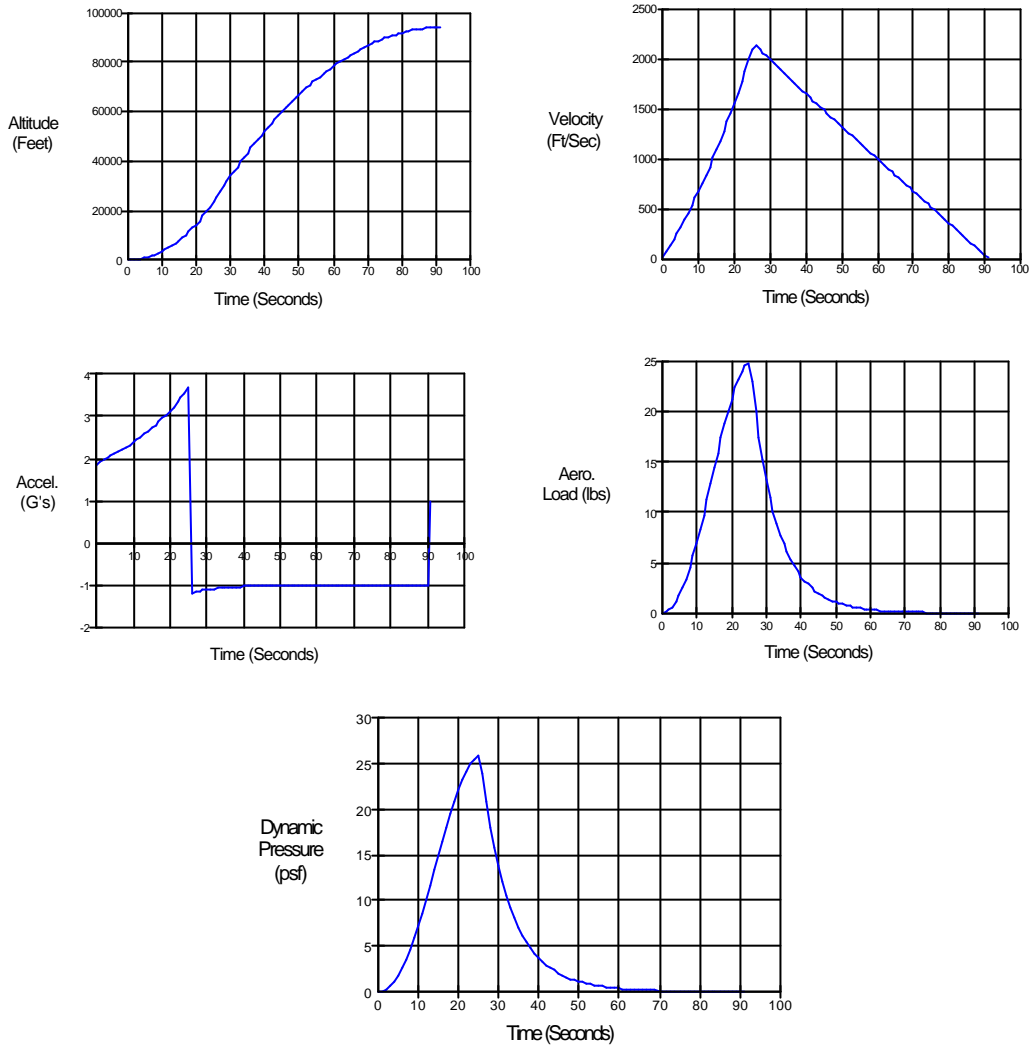


Figure 9.1: Graphical Representation of the MAV Flight Statistics

9.3 Rocket Engine Sizing

Since the software used in this section uses imperial (old English) units, all of the data was converted into imperial, calculated and then reconverted to ensure consistency throughout the document.

Table 9.3: Data Required for Rocket Engine Sizing

Variable	Data	Variable	Data
Required Thrust (lbf)	610.92	Exit Pressure (psi)	0.1
Chamber Pressure (psi)	200	Fuel/Oxidiser Ratio	2.57
? = cp/cv	1.1093	Universal Gas Constant (ft-lb/mol-°R)	1545.32
Chamber Temperature (°F)	5038	Earth Gravity (ft/sec ²)	31.174
Characteristic Length or L* (in)	106	Nozzle Coefficient, N _c	1.75
Molecular Weight	60.21	C* (ft/sec)	3365.3

$$A_t = \frac{F}{P_c \times N_c} \quad (9.1)$$

where A_t = Throat area, in²
 F = Thrust, lbf
 P_c = Chamber pressure, psi
 N_c = Nozzle coefficient

With the area of the throat now known, the chamber contraction ratio, R_c , can be determined from the following equation;

$$R_c = (8 \times A_t^{-0.6}) + 1.25 \quad (9.2)$$

This equation is commonly used to gauge the chamber contraction ratio for liquid bipropellant engines and is based on historical data. However, since metal powder fuels take a greater length of time to burn, this figure calculated for R_c was doubled in order to provide sufficient chamber volume for complete combustion to occur.

The area of the chamber, A_c , is therefore;

$$A_c = \frac{A_t}{R_c} \quad (9.3)$$

The velocity of the exhaust gas, in terms of Mach number, is;

$$M_e = \sqrt{\left[\left(\frac{2}{1-g} \right) \left\{ \left(\frac{P_c}{P_{atm}} \right)^{\frac{g-1}{g}} - 1 \right\} \right]} \quad (9.4)$$

With this value, the area of the Nozzle, A_n , can be calculated;

$$A_n = \left(\frac{A_t}{M_e} \right) \left(\frac{1 + \frac{g-1}{2} M_e^2}{\frac{g+1}{2}} \right)^{\frac{g+1}{2(g-1)}} \quad (9.5)$$

The volume of the chamber, V_c , is based on the characteristic length or L^* and the area of the throat, A_t . The value for L^* was taken from a previous study by Loftus et al. who used ammonium perchlorate and aluminium powder. This was the same value used by Wickman for the CO₂/magnesium powder engines.

$$V_c = 1.1 \times (L^* \times A_t) \quad (9.6)$$

The chamber length, L_c , is therefore;

$$L_c = \sqrt{\frac{V_c \times 4}{D_c \times \rho}} \quad (9.7)$$

The results from these calculations are presented in table 9.4.

Table 9.4: Engine Dimensions and Operating Parameters

Parameter	Results, English Units	Results, SI Units
Throat Area	1.75 in ²	
Throat Diameter	1.49 in	0.038 m
Chamber Contraction Ratio	13.95	
Chamber Area	24.36 in ²	
Chamber Diameter	5.57 in	0.141 m
Exhaust Gas Velocity (Mach)	4.52	
Nozzle Area	318.1 in ²	
Nozzle Diameter	20.12 in	0.511 m
Chamber Volume	203.52 in ³	
Chamber Length	6.82 in	0.17 m
Chamber Wall Thickness ¹	0.25 in	0.635 cm
Nozzle Ablative Material Thickness ²	0.20 in	0.508 cm
Engine Burn Duration, (s)	25.45	25.45
Propellant Weight Flow Rate	3.54 lb/s	1.606 kg/s
Fuel Flow Rate	0.992 lb/s	0.45 kg/s
Oxidiser Flow Rate	2.55 lb/s	1.16 kg/s

With these values, it was possible to present an engine schematic, figure 9.2.

¹ Calculated in Section 10 : Thermodynamic Performance Characteristics

² *ibid.*

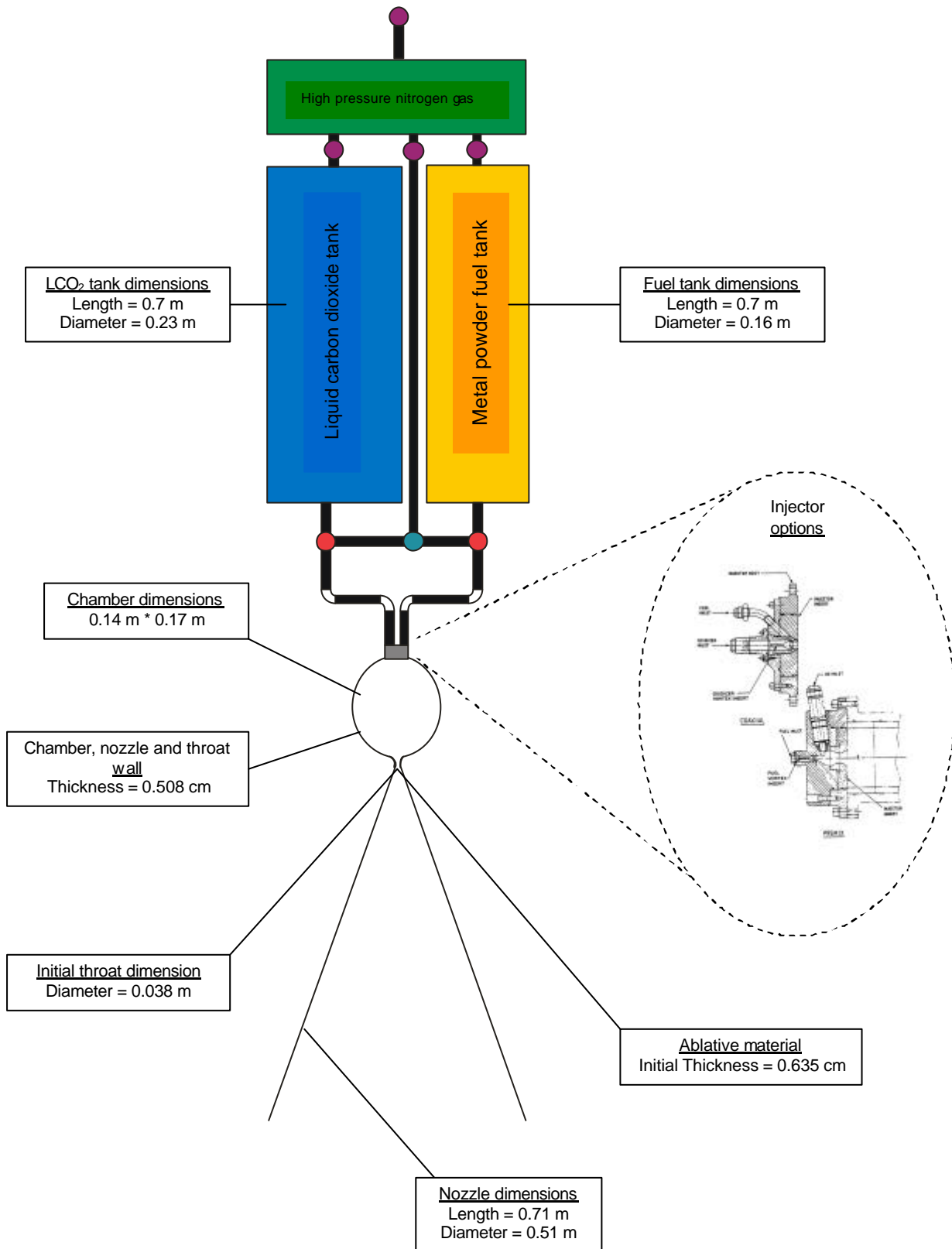


Figure 9.2: MAV 1st Stage (CO₂/Magnesium Powder Fuel) Engine Schematic

10.0 Thermodynamic Performance Characteristics

10.1 Introduction to Thermodynamic Performance

This section is concerned, not with the chemistry of the thermodynamic reactions taking place during the combustion process but how to cater for the heat produced from the reaction and identify ways to provide engineering solutions to avoid engine failure due to overheating.

There are in essence four techniques used to cool conventional rocket engines;

1. Regenerative cooling
2. Ablative cooling
3. Radiation cooling
4. Film/boundary layer cooling

In order to enhance performance, the first three techniques usually employ film/boundary layer cooling as well.

10.1.1 Regenerative Cooling

Regenerative cooling typically employs the liquid fuel component as the coolant and although oxidisers have been known to be used, this is usually avoided due to the concern of 'oxidising' the coolant channels¹. If regenerative cooling is to be used for this engine concept, then this is problem that will have to be overcome. Since the fuel will most likely be in powder form, either the CO₂ or an additional fluid will have to act as the coolant.

The standard principal of regenerative cooling predominantly applies to mid to high thrust levels because the heat flux increase with the mass flow rate. This implies the use of pump fed engines, which are not the case with metal powder engines, where the primary choice of feeding the propellant is via high pressure.

Knowing the thermo physical properties of the fluids will allow the determination of the fluid flow and heat transfer characteristics. The different thermo physical properties will dictate the heat transfer capability and therefore the coolant channel optimisation. The thermo physical properties of the thrust chamber, most importantly its melting point and thermal conductivity, limit the wall temperature and it is important that the material is kept within its elastic limit so as not to permanently deform its physical form. In addition to severe temperatures, the thrust chamber must also be able to cater for the pressure loads produced during the combustion process.

10.1.2 Ablative Cooling

With ablative cooling, the heat energy from the combustion gases vaporise the ablative material, which coats the inner combustion chamber walls, throat region and nozzle. Typical materials tend to be silica, quartz, carbon cloth or resin composites. Ablative materials are often employed for one-off, short burn duration or low to mid thrust level motors (i.e. moderate mass flow rate) where the complexity of a regenerative cooling system is not warranted. The lifetime of an engine using ablative materials is on the order of 2000 seconds of firing². With metal powder engines, this lifetime will most probably be reduced due to the high heat flux and the wear afforded by the impingement of micro-particles of metal powder that have not fully combusted in the combustion cycle.

¹ Humble, Ronald, W., Lewis, D., Bissell, W. and Sackheim, R., "Liquid Rocket Propulsion Systems", Chapter 5 in 'Space Propulsion Analysis and Design', Revised Edition, Humble, Ronald W., Henry, Gary N. and Larson, Wiley J., Space Technology Series, ISBN 0-07-031320-2, p235

² ibid 1, p236

10.1.3 Radiation Cooling

Radiation cooling is a method whereby the heat from the combustion process is transferred through the chamber walls, which is a highly conductive metal, by convection and radiation. The metal will have to have the necessary thickness to withstand the pressures produced by the combustion process plus maintain its elastic integrity. This method of cooling is ideally suited for a low-thrust option, in-space propulsion option and is probably not the model candidate for a Mars ascent vehicle. This does not rule out the possibility of introducing radiation cooling in combination with other methods of cooling to form a hybrid-cooling system.

10.1.4 Film/Boundary Layer Cooling

It is standard practice to use film or boundary layer cooling in conjunction with the previous three methods of cooling. Injectors are placed at pre-designated positions within the combustion chamber where the coolant, typically the fuel component of the propellant, is injected along the chamber wall increasing thermal resistance and reducing the wall temperature affording higher engine operating temperatures. The throat region, however, experiences standard temperatures simply because most if not all of the film coolant arrives at that position in the form of a vapour. Here especially, the chamber wall material should possess high thermal conductivity in order for the heat flux to radiate axially, away from this temperature sensitive segment of the engine. It is important to stress that unlike conventional cooled engines.

Assuming that we wish to employ a regenerative or film/boundary layer cooling system, and limit the selection of coolant fluid to those available from Martian resources, then the three obvious contenders would be carbon dioxide, oxygen and nitrogen. Nitrogen, if used as a pressurant, would probably be available in too small a quantity to be of any practical use as a coolant it would have to be in liquid form but then the energy required to produce it would not be justified. Nitrogen was therefore eliminated from any further investigation.

10.2 Heat Transfer Coefficient

10.2.1 Chamber

Since the CO₂/magnesium engine burn duration for the chosen MAV scenario will last for just over 25 seconds, the objective here is to try and maintain simplicity, therefore trying to avoid complex cooling mechanisms such as regenerative cooling. Without a doubt, if one were to consider continual reuse for Mars mobility, then regenerative cooling would be the first and foremost option. The first point that one will notice is the significant difference in chamber temperature. LOX/LH₂ propulsion systems deliver 2½ times the specific impulse of CO₂/magnesium engines yet the chamber temperature of the latter is 250 K (450°F) higher. This is one of the major drawbacks with metal powder engines. If future infrastructure were to employ LOX/aluminium in order to allow refuelling on the Lunar and Martian surface as well as the Martian moons and asteroid belt, then the sophisticated cooling will be required to cater for temperatures of 4460 K, or over 1650 K higher than the heat experienced in present day LOX/LH₂ engines. The temperatures mentioned here were based on a chamber pressure of 13.79 bar (200 psi), see table 10.1.

The software used to determine thermo-ablation uses Imperial (old English) units and, like in the previous section, all pertinent values were converted from SI units to Imperial and then reconverted back to SI units. The equations used in this section were adapted from "How to Make Amateur Rockets".³

³ Wickman, John H., "How to Make Amateur Rockets", 2nd Edition, published and distributed by CP Technologies, Casper, Wyoming, USA, pp8.1-8.6

Table 10.1: Chamber Stagnation Temperatures (in SI Units: Kelvin & bar)⁴

Chemical	CO ₂ Pressure Fed @ 13.79 bar	CO ₂ Pump Fed @ 34.47 bar	LOX Pressure Fed @ 13.79 bar	LOX Pump Fed @ 34.47 bar
<i>Non-ISRU Metals</i>				
Boron (B)	2155	2279	4478	4710
Beryllium (Be)	3134	3139	4919	5277
Lithium (Li)	2000	2000	2788	2877
Zirconium (Zr)	3304	3316	4074	4260
<i>ISRU Metals</i>				
Aluminium (Al)	3249	3260	4461	4710
Calcium (CaO) ⁵	-	-	-	-
Chromium (Cr)	1854	1868	3617	3719
Iron (Fe)	844	853	2374	2377
Manganese (Mn)	-	-	-	-
Magnesium (Mg)	3054	3097	3759	3984
Magnalium (Mg/Al)	3068	3097	-	-
Potassium (K)	894	913	2000	2000
Silicon (Si)	2508	2501	3512	3708
Sodium (Na)	-	-	2292	2364
Titanium (Ti)	2467	2467	4125	4348
<i>Other non-ISRU Chemicals</i>				
Aluminium Borohydride (Al(BH ₄) ₃)	1773	1797	3616	3762
Beryllium Hydride (BeH ₂)	2547	2602	3823	3978
Diborane (B ₂ H ₆)	1723	1770	3459	3582
Magnesium Hydride (MgH ₂)	2196	2310	3301	3462
Pentaborane (B ₅ H ₉)	1806	1819	3730	3890
Silane (SiH ₄) ⁶	-	-	-	-
<i>Other potential ISRU Chemicals</i>				
Carbon Monoxide (CO)	-	-	3255	3367
Liquid Hydrogen (LH ₂)	-	-	2804	2836
Methane (CH ₄)	350	364	3260	3357
Ethane (C ₂ H ₆)	-	-	-	-
Propane (C ₃ H ₈)	649	659	-	-
Butane (C ₄ H ₁₀)	-	-	-	-
Methanol (CH ₃ OH)	761	781	3088	3174
Dinitrogen Dihydride (N ₂ H ₂) ⁷	-	-	-	-

⁴ Kaye, G. W. C. and Laby, T. H., "Tables of Physical and Chemical Constants", ISBN 0-582-46354-8, p11

⁵ Calcium (Crystalline) was not available as a chemical in the Propellant Evaluation Program database

⁶ Silane was not available as a compound in the Propellant Evaluation Program database

⁷ Dinitrogen Dihydride was not available as a compound in the Propellant Evaluation Program database

The chamber temperature (5038°F) had to be converted to absolute temperatures by adding 460. The Prandtl number and gas conductivity were estimated at 0.49 and 0.218 Btu/Hr-Ft-R, respectively. The gas constant was calculated since the molecular weight was realized from the PEP program.

The gas density is therefore;

$$\mathbf{r}_g = \frac{P_c}{C_g \times T_g} \quad (10.1)$$

where $\rho_g =$ gas density, lb/ft³
 $P_c =$ chamber pressure, psi
 $C_g =$ gas constant
 $T_g =$ gas temperature, °R

Since the chamber and throat diameters are known, the area ratio, A_R , is;

$$A_R = \left(\frac{D_c}{D_t} \right)^2 \quad (10.2)$$

The area ratio of 13.95 implied that the gas velocity, v_g , would be on the order of 100 ft/sec.

The viscosity of the gas, μ_g , was one value that the PEP program could not provide and as a result, an estimated value of 0.000062 lb/ft-sec was used. This value was actually used for composite rocket propellant using magnesium powder and hence, it was assumed that for this first iteration that this value would suffice.

The Reynolds number could now be calculated with the following equation;

$$Re = \frac{\mathbf{r}_g \times v_g \times D_c}{\mu_g} \quad (10.3)$$

With all of these values now known, it was possible to calculate the heat transfer coefficient;

$$h = \left(\frac{k}{D_c} \right) \times 0.023 \times Re^{0.8} \times Pr^{0.4} \quad (10.4)$$

This value is presented in Btu/Hr-Ft²-R and was converted to Btu/sec-Ft²-R before entry into the thermo-ablation program.

10.2.2 Throat

The heat transfer coefficient for the throat region uses slightly different equations to those used above, but with the substitution of pertinent diameters and temperatures.

The gas temperature at the throat, T_{gt} , is calculated by;

$$T_{gt} = T_f + Pr^{0.33} \times (T_c - T_f) \quad (10.5)$$

where T_f is the free stream temperature, which is assumed to be 90% of the chamber temperature. The gas density at the throat is estimated as being 62% of the chamber gas density. The Reynolds number at the throat is;

$$Re = \frac{\mathbf{r}_t \times v_s \times D_t}{\mu_g} \quad (10.6)$$

where $\rho_t =$ density of gas at throat, lb/ft³
 $v_s =$ sonic gas velocity, ft/sec
 $D_t =$ diameter of the throat, ft
 $\mu_g =$ viscosity of the gas, lb/ft-sec

The heat transfer coefficients are calculated by using equation 10.4 but by substituting the values for the Reynolds number from equations 10.6 and the throat diameter in place of the chamber diameter.

10.3 Results

Since the heat transfer coefficient at the throat is over a factor of 10 higher than that of the chamber, the throat region was examined first. The input data is shown in table 10.3 and table 10.4.

Table 10.2: General Data for Nozzle and Chamber Analysis

Final Time (sec)	Time Increment (sec)	Number of Materials	Initial Temperature (°R)	Gas Temperature (°R)	Heat Transfer Coefficient (Btu/sec-ft ² -°R)	Heat Transfer Coefficient (Btu/sec-ft ² -°R)
25.45	0.01	2	530	5498	0.480376	0.033521

The number of nodes multiplied by the node thickness constitutes the thickness of the material. Since the software did not determine the required thickness based on the heat flux automatically, the process was performed manually. The minimum wall thickness for any part of the chamber was calculated with a simple equation;

$$t_{cw} = \frac{P_c \times D_c}{S_y} \quad (10.7)$$

where P_c = chamber pressure, psi
 D_c = chamber diameter, in
 S_y = yield strength for type 304 stainless steel, psi

Table 10.3: Material Properties for Nozzle (only Material 2 is used for Chamber)

	Material 1	Material 2
Material Type	G-90 Graphite	304 Stainless
Node Thickness (in)	0.05	0.05
Number of Nodes	5	4
Density (lbm/ft ³)	118	501
Melting Temperature (°R)	4260	3210
Specific Heat (Btu/lbm-°R)	0.4	0.12
Heat of Fusion (Btu/lbm)	2750	200
Conductivity (Btu/hr-ft-°R)	36	9.4

It was found that even with a minimum wall thickness, which includes a safety factor of 50%, the material disintegrates within a matter of seconds of engine operation due to the tremendous heat flux generated from the combustion process. For the throat region, stainless steel up to ½ inch thickness, on its own also fails before engine cut-off so rather than increasing the engine's mass by increasing the wall thickness with stainless steel, an ablative material with a lower density was added to the configuration. The stainless steel around the throat was 1/5 of an inch (0.508 cm) thick but an additional thickness ¼ inch (0.635 cm) of G-90 graphite was required to allow the throat region to maintain its structural integrity yet minimise mass.

It is important to note that the engine's performance was based on a fixed diameter nozzle. As one will see by the results in appendix E, the total amount of erosion in the throat is about 0.635 cm and this will obviously alter the performance of the engine. Since all the calculations were performed using either spreadsheets or with software requiring manual inputs, it was not possible to provide a more accurate performance with numerous iterations in the duration of this short contract.

The process was repeated for the chamber but with varying material thickness and quantities of material as well as the use of the pre-calculated specific heat. In order to maintain consistency, the engine chamber was also kept at a thickness of 0.508 cm. The chamber walls did not require ablative protection since the software showed that no erosion occurred throughout the 25-second engine burn time.

It was therefore found that for the purpose of a single-use, 1st stage MAV using CO₂/magnesium engines, that ablative cooling at just the throat region would suffice.

11.0 Concept Deficiencies and Possible Solutions

11.1 CO₂ Manufacture

In most, if not all, research papers, the manufacture of CO₂ is taken for granted. However, in order to produce liquid CO₂, the pressures must be raised from ~6.9 millibar over 1000 times to ~10 bar. This is not a simple task and will have to be addressed. Three possible CO₂ production opportunities present themselves;

- a) Mechanical compressor
- b) Adsorption pump
- c) Refrigerator

The most efficient is dependant on the duration the MDV/MAV is required to stay on the surface of Mars.

Production of liquid CO₂ is also a prerequisite for the production oxygen for life support or liquid oxygen for use in cryogenic propulsion systems. By adding hydrogen to the carbon monoxide component of CO₂, liquid methane could also be produced. LOX/LCH₄ promises to be the most versatile propellant combination, in terms of storage and performance, for ISRU on Mars. Therefore, one should not look at the production of CO₂ just for the sake of CO₂.

Due to these additional benefits of CO₂ production, methods for CO₂ acquisition and separation storage on the Martian surface should be given a high priority.

It should be stressed, however that CO₂ in its gaseous form is relatively benign and can be used in combination with almost any material however, liquid CO₂ has the tendency to form carbonic acid and therefore the liquid oxidiser tank plus the CO₂ acquisition equipment will have to use materials resistant to this acid.

Other trace gases that are resident in the Martian atmosphere, such as nitrogen has uses as the pressurising gas for the propulsion system whereas Argon, an inert gas can be used to produce metal powder. Methods should therefore be developed to bleed off gases and separate them from the other trace gases for subsequent processing and storage.

11.2 Metal Mining

There is an inherent lack of information on the mineralogical distribution on the surface and sub-surface of Mars. The GRS instrument on the Mars Odyssey is capable of providing such data but firstly this data will not be ready until the end of 2003 and secondly, mineralogical distribution for future mining efforts may not be the primary source of interest to planetary researchers. Irrespective of the mineralogical distribution, the ores or perhaps the regolith will have to be gathered for subsequent processing. Mining metals from ores or fines (regolith) is common practice with terrestrial mining techniques but additional work will be required to determine the exact process to be used on the Martian surface.

Two areas will require attention;

- a) The mapping of minerals with high contents of magnesium, aluminium, silicon, iron and titanium
- b) The processes in which they can be extracted with minimal energy and autonomously in a hostile environment.

Dedicated mineral mapping sensors, similar to those used to detect minerals for terrestrial mining would be recommended for inclusion on future Mars Orbiters.

11.3 Feed Issues

Even though the issue of feeding metal powder has been addressed on and off for the past three decades, the associated problems such as clogging in the feed lines and injectors as well as incomplete combustion still haven't been resolved. In order to maximise metal powder storage density a trade off must be performed between a mixture of large and small powder

particle sizes when compared to the feed and injection of varying particle sizes. A concerted effort to solve these problems is required if this propulsion concept is to become a reality.

The catering industry currently use pneumatic feed systems as one option to transport food products such as flour, sugar, coffee powder and salt from production to packing. This is considered the safest way and preferred method of transporting the powder since dust explosions can be avoided. It also has the added benefit of avoiding contamination since the powder is transported in an enclosed environment. However, it does have a tendency of damaging the food product, and is rarely used when compared to other methods. A possibility therefore presents itself to team up with terrestrial industry for mutual benefit. A solution to the feed of metal powder may lie with the decades of experience from feeding powders in the catering industry. Any success with the design of metal powder rocket engines could then be spun-off for the benefit increasing safety in industry.

11.4 Combustion

To date, only CO₂-magnesium or CO-magnesium studies have taken place and even then, only single particles were used. In reality, before the complete combustion of particles takes place, additional CO₂ and magnesium powder will be injected in to the combustion chamber so reactions involving CO₂-MgO and CO-MgO will also have to be addressed. Detailed analysis with the use of Monte Carlo type simulations will be required to model the true performance of this type of propulsion system.

11.5 Propulsion Technologies

This report's primary focus was that of CO₂/metal powder engines however, propellant combinations using LOX/metal powder were also presented. The primary reason for introducing propellant combinations using LOX/metal powder was to highlight the fact that the development of such propulsion technology could not only be used for missions to Mars but vehicles using such technology could also be refuelled with Lunar soil propellant (LSP) or on the asteroid belt where such minerals are plentiful, albeit not in the form of regolith. Additional work in LOX/metal powder systems will also be advised. Unlike Mars, the Lunar surface and asteroid belt contain large quantities of oxygen but in the form of metal oxides. Terrestrial mining technologies exist where fuel is extracted from the metal oxides but the oxygen content is disposed off. The problem for the case of ISRU is that problem is not to extract the metal, but capture the oxygen so work in this field is also recommended. This would afford the possibility of using propellant combinations for all CIS-asteroid belt transportation and with one generic propulsion system. The tremendous heat flux given off by CO₂/metal powder and LOX/metal powder combustion is much higher than those experienced from present day chemical propulsion systems. The cooling of such engines will prove to be a challenge but one that must be mastered if such propulsion technologies are to become a reality.

12.0 Conclusions

The intention of this report was to introduce to the layman, the process in which in-situ propellant production can be used for the benefit of supporting future infrastructure on Mars. In order to accomplish this wide-ranging task, an introduction to the Martian environment was followed by a summary of techniques that could be used to obtain oxidisers and fuels from Martian resources. The various propellant combinations were then entered into a propellant evaluation program to determine performance within a Mars ascent propulsion system based on local ambient conditions. Four different sample return scenarios were also considered in order to afford an opportunity to provide a comparative assessment of technologies. The focus of attention was on CO₂/magnesium metal powder engines but an entire series of propellants were examined. The engineering problems experienced with the past three decades of research in metal powder fuel engines were addressed. Based on the information at hand, a proposed engine design was presented.

Knowledge on the Martian atmospheric content is quite extensive. CO₂ is the most abundant chemical constituent making up about 95% of the content. Nitrogen and argon, which present on the order of 2.7% and 1.6% of the atmosphere, respectively, were also found to be of benefit for either the manufacture of in-situ propellants or for use in in-situ propulsion systems. However, there is a fundamental lack of knowledge of mineralogical distribution throughout the surface and below the sub-surface of Mars. Detailed information on mineralogical concentrations is a prerequisite to any proposed mining and metal powder processing operation. An alternative would simply be to ignore the traditional mining techniques and use the regolith, which there is a plentiful amount of and has already been processed by nature. This would eliminate the necessity to have to mine for ores and then crush them to obtain the necessary minerals. The majority of terrestrial mineral processing technologies use a combination of automated and human participation. Small production facilities would still have to be designed and tested based on an entirely autonomous system.

When presenting in-situ propellant production or propulsion related papers, many researchers skip over the production of CO₂. Adsorption pumps and refrigerators are two technologies that have been addressed for CO₂ manufacture on Mars but little or no work has been done on the design mechanical compressors. In order to liquefy CO₂ from ambient temperature and pressures, the latter must be raised by a factor of 1000 times. Such a huge pressure increase will only be probable with multistage compressors but even then, the power requirements and production rates are not known. Adsorbent materials are inherently brittle and lend themselves to being easily damaged, especially during landing when the entire bed could shatter. Also, the larger the quantity of CO₂ required for a limited period of time, the larger the Adsorption material and hence overall mass. With current knowledge, it seems as though the most likely method of producing liquid CO₂ on Mars would be with the method of refrigeration or mechanical roughing pump/compressor.

Apart from the feeding of metal powders, the other technological challenge associated with such engines is the sheer heat generated from the combustion products as well as the micro particles of metal that would eventually wear away and degrade the propulsion system components. Single use engines, such as the design of the 1st stage of an MAV presented in this report would survive short burn durations but a concerted effort to solve these problems is a requirement especially if one were to design the engine for continual use for Mars mobility.

A Mars sample return mission is presented in order to ascertain if this technology proves to be more beneficial than conventional chemical propulsion technology. Here the ISRU propellant is used as the 1st stage of the MAV with liquid bi-propellant as the 2nd stage. The MAV will launch the sample container in to a LMO where it will rendezvous with an ERV. This ISRU concept, when compared to an MAV using bi-propellants for both stages, has only 65 kg more mass for TMI. Most CO₂/magnesium concepts fare poorly against a comparison with conventional liquid bi-propellants however this configuration does come close and may warrant funding for a Mars sample return technology demonstrator for a type IV Earth departure on 21st November 2006, a stay time on the Martian surface for 173 days and then launched on a type II Mars departure for an arrival at Earth on 21st May 2010.