University of California, Los Angeles
Mechanical, Aerospace and Nuclear Engineering Department

DESIGN OF A SCIENTIFIC PROBE FOR
OBTAINING MARS SURFACE MATERIAL

FINAL REPORT

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Instructor:

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MSP SPECIFICATIONS

OBJECTIVE: Return 1Kg martian soil sample from the surface of mars to a mothership in a 600km mars orbit.

STRUCTURE:
- overall height: 1.2 meters (touchdown configuration)
- 1.5 meters (stowed/descent configuration)
- overall width: 0.66 meters (touchdown configuration)
- 1.5 meters (stowed/descent configuration)
- maximum gross vehicle mass: 110.25 Kg (after mothership separation)
- mass at return: 2.00Kg (includes 1Kg sample)
- planform: 3-legged layout with hexagonal bus
- materials: Beryllium (launch rings, sample retrieval system, heat shield backing, landing gear main struts)
  Magnesium (shock absorbers, landing gear secondary struts, bus frame)
  Aluminum (bus platform, vehicle skin)

PROPULSION
- deorbit/descent/ascent: 1 single restartable liquid propellant motor, pressure fed, using 50% UDMH, 50% Hydrazine as a fuel and nitrogen tetroxide as oxidizer
- apogee kick (rendezvous): solid propellant motor
- maneuvering: (8) thrusters using UDMH as propellant
MISSION PROFILE

0-1
This mission begins with detachment from the mother ship, the Mars Orbiting Vehical (MOV) of the designed spacecraft, the Mars Surface Probe (MSP). After separation the MSP determines its position and the relative position of the landing sight and initiates a short burn which will cause the MSP to be at our optimum position for deorbit within seven days. During this time the ship is in a dormant mode with occasional systems checks made by a monitoring microcomputer. Adjustments to the precession orbit are periodically made. When the MSP is in the correct location to descend with a minimum fuel expenditure and no plane change, the main motor fires, decelerating the ship.

1-2
The MSP turns around to face its heat shield toward the direction of flight. When the atmosphere is encountered at about 200-150km altitude the ship releases a kind of "drogue chute" for passive stability during descent.

2-3
The MSP sheds velocity during reentry, protected by its heatshield and a multilayer insulation. At 1km above the actual ground surface a parachute is deployed to slow the craft even further. The heat shield is jettisoned with explosive bolts. After slowing to 42 m/s and drifting to within 100 meters of the surface the chute is released and the main motor fires to bring the craft to a standstill 5 meters above the surface.

3-4
After stopping 5 meters above the surface the main motor shuts down and the MSP falls freely to avoid contamination of soil samples. Once safely on the ground a shovel type mechanism deploys and begins collecting sample soil.

4-5
Once the sample has been collected and stored the ship waits for half a martian day until the MOV orbit will be nearly overhead. The MSP ascent stage separates from the hexagonal descent stage leaving spent fuel tanks, the sample collector, the landing gear, and other used parts.

5-6
The ascent stage climbs out of the atmosphere using the main liquid motor for the last time.
The ascent stage burns out. The computer points the rendezvous stage in the right direction, spins it up and releases it. There is no active guidance on the rendezvous stage.

The rendezvous stage makes its solid motor burn and activates its beacon for the MOV to find it. Its orbit differs slightly from the MOV so the two will eventually pass each other.

The MOV catches up to the rendezvous stage and brings it inside the docking bay which originally held the full MSP and then secures it for the return trip.
MSP Mission Scenario

0: MSP released from MOV, main liquid motor burn

1: End of burn, reorientation of MSP

2: Descent through atmosphere, deployment of guide/drogue chute

3: Chute release, main engine retro firing

4: Sample acquisition

5: Ascent stage separation

6: End of ascent burn, separation of apogee stage

7: Solid apogee motor burn and orbital insertion
<table>
<thead>
<tr>
<th>Mass: Final (on orbit):</th>
<th>1.00 (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dirt:</td>
<td>0.10 (kg)</td>
</tr>
<tr>
<td>Sample Container:</td>
<td>0.25 (kg)</td>
</tr>
<tr>
<td>Apogee Engine Case:</td>
<td>0.30 (kg)</td>
</tr>
<tr>
<td>Beacon:</td>
<td>0.15 (kg)</td>
</tr>
<tr>
<td>Batteries/solar cells:</td>
<td>0.20 (kg)</td>
</tr>
<tr>
<td>Structure:</td>
<td>2.00 (kg)</td>
</tr>
<tr>
<td>Total:</td>
<td></td>
</tr>
</tbody>
</table>

| Before Separation:     | 2.00 (kg) |
| Apogee Stage:          | 0.73 (kg) |
| Apogee Fuel:           | 0.90 (kg) |
| Apogee Stage Support:  | 3.10 (kg) |
| Fuel and Helium Tanks: | 0.50 (kg) |
| Valves and Regulators: | 1.80 (kg) |
| Aeroshell:             | 2.00 (kg) |
| Guidance and Control:  | 0.10 (kg) |
| Batteries:             | 2.00 (kg) |
| Liquid Engine and Plumbing: | 0.90 (kg) |
| Engine Structural Support: | 0.50 (kg) |
| Thermal Insulation:    | 0.50 (kg) |
| Attitude Control:      | 15.03 (kg) |
| Total:                 |           |

| Ascent Stage Before Liftoff: | 15.03 (kg) |
| Ascent Stage Dry Mass:       | 44.17 (kg) |
| Ascent Fuel (5% ullage):     | 59.20 (kg) |
| Total:                       |           |

| Mass on Mars:               | 59.20 (kg) |
| Ascent Stage:               | 1.50 (kg)  |
| Descent Tanks and Plumbing: | 1.50 (kg)  |
| Landing Gear:               | 15.00 (kg) |
| Payload Scoop:              | 0.80 (kg)  |
| Batteries:                  | 1.00 (kg)  |
| Laser Range Finder:         | 1.00 (kg)  |
| Landing Computer:           | 0.50 (kg)  |
| Attitude Control:           | 1.50 (kg)  |
| Thermal Insulation:         | 7.55 (kg)  |
| Structure:                  | 89.55 (kg) |
| Total:                      |           |

| Mass at Re-entry:           | 89.55 (kg) |
| Landing Mass:               | 3.00 (kg)  |
| Retro Fuel:                 | 2.00 (kg)  |
| Parachute:                  | 0.30 (kg)  |
| Parachute Canister:         | 5.00 (kg)  |
| Heat Shield and Supports:   | 2.00 (kg)  |
| Drogue Basket:              | 101.85 (kg) |
| Total:                      |           |

| Initial (On-Orbit) Mass:    | 101.85 (kg) |
| Re-entry mass:              | 7.57 (kg)   |
| Descent Fuel:               | 0.83 (kg)   |
| Orbital Maneuvering Fuel:   | 110.25 (kg) |
| Total:                      |           |

Table MSP Mass Estimates
Contents

Section A: **Structural Design and Component Placement**  
Eric Deyerl

Section B: **Thermal Control and Guidance**  
Terrance Yee

Section C: **Propulsion Systems**  
Myles Baker

Section D: **Orbital Mechanics**  
Bob Langberg

Section E: **Specialized Structures**  
Tim Gibson
Section A: Structural Design and Component Placement

Eric Deyerl
1. Goal

To provide the most compact (space and weight efficient) structure to unite, protect, and allow effective operation of all spacecraft components.

2. Development

2.1 Research

Much research was accomplished, mainly through the perusal of large amounts of NASA technical reports (see References). Here, the designer focused his attention on reports that dealt with landing gear and spaceframe studies. Most specifically, there were two NASA projects that provided the most insight in these areas; those were the Surveyor and Apollo programs. (Figures 1 and 2). Although both of these missions were lunar in objective, they had similar parameters in their missions, and so were valuable in the information they provided.

Figure 1: Surveyor spacecraft

Figure 2: Apollo lunar lander
The surveyor three-legged layout was inspirational for the MSP designer. It is simplistic and effective, and had been used successfully in the Viking mission to Mars. (Refer to Figures 3 and 4) With three legs, one can be sure that all footpads will be in contact with the Martian soil upon landing. The three-legged design also allowed nicely for a structural-rigid hexagonal bus structure. Finally, it is also a stable platform whose shape makes it difficult for surface winds to catch a broad side of the craft and topple it.
The surveyor project also gave valuable insight into landing gear and shock absorber design, in conjunction with the placement of crushable honeycomb material. (Figure 5)

Figure 5: Surveyor landing gear leg

Figure 6: Apollo lunar lander landing gear leg

The Apollo Lunar Module’s landing gear was also studied in an effort to incorporate a similar retracting system on the MSP. This was later found unnecessary. A rendition of one of the “Eagle’s” gear is found in Figure 6.
2.2 Preliminary Layout Studies

Prior to much of the research, preliminary sketches were made to seek out viable spacecraft layouts. Two early plans and some notes on each are given in Figures 7 and 8.

Figure 7: First preliminary MSP layout sketch and comments

Figure 8: Second preliminary MSP layout sketch and comments
Later, after more propulsion studies and relative sizing of components had been undertaken, it was possible to produce more accurate and intermediate layouts, as shown in Figure 9.

![Figure 9: Intermediate MSP layout](image)

It was at this point that the designer realized that the critical components, as far as space was concerned, were the fuel and oxidizer tanks. (which is usually the case with light aerospace vehicles). Thus, a fuel tank study was initiated, which (as shown in Figure 10) resulted in a possible toroidal tank configuration. This idea was later discarded due to complexity of purging, pressure containment, and the like.

<table>
<thead>
<tr>
<th>CYLINDERS</th>
<th>TOROIDS</th>
</tr>
</thead>
<tbody>
<tr>
<td><img src="image" alt="Cylinder Diagram" /></td>
<td><img src="image" alt="Toroid Diagram" /></td>
</tr>
<tr>
<td>Volume = $r \times h = 17,200 \text{ cc}$</td>
<td>Volume = $\left( \frac{(r - r')}{2} \right) \times \left( \frac{(r + r')}{2} \right)$</td>
</tr>
<tr>
<td><img src="image" alt="Cylinder Table" /></td>
<td><img src="image" alt="Toroid Table" /></td>
</tr>
</tbody>
</table>

![Figure 10: Fuel tank study](image)
2.3 Final MSP Configuration

After many hours of conference, deliberation, and improvisation, the layout of the MSP was finalized to that shown in Figures 11 and 12.

It is a space efficient design, and yet tries to remain as simple as possible. The design is roughly axisymmetric, with the ascent vehicle able to clear all bus components.

3 Component Design

Following are the steps taken to design the critical components of the MSP, which include the landing gear, shock absorbers, bus frame and platform, and ascent and apogee stage launch rings.

Care was taken to use the lightest and (naturally) strongest materials possible, with thought given to their manufacture and interaction with the environments the MSP would be subjected to. It also was always desired to use a safety factor of at least 1.5. As will be shown, because of the generally small dimensions of the craft, this would never be a problem.

3.1 Landing Gear

It was chosen to use the landing gear configuration shown in Figures 13 and 14, with a main strut containing a shock absorber to cushion the vehicle at landing, and secondary struts for any torsional forces encountered by the gear. The following are the calculations made in the design of these components.

3.1.1 Loads on Gear

The mission scenario calls for the MSP to impact the Martian surface at 5.5 m/s. The force on one gear (considered in the extreme case of a one-legged landing) is given by

\[ F = ma \]

where

\[ m = 73.9 \text{ kg (at landing)} \]

\[ a = \left(\frac{v^2}{2s}\right) \]

where

\[ v = 5.5 \text{ m/s} \]

\[ s = 0.20 \text{ m (vertical gear travel)} \]

Thus, the force on one leg (taken to be applied through the axis of the shock absorber in the case of a lopsided one-legged worst-case landing, depicted in Figure 15) is

\[ \text{Landing force on one leg} = F = 5589 \text{ N} \]
Figure 11.1: MSP Exterior Dimensions
(Descent Configuration)
Figure 11: MSP Cutaway
(Descent Position)
Figure 12.2: MSP Planform Exterior
Figure 12: MSP Planform Cutaway
Figure 12.1: MSP Planform Dimensions
Figure 13: Landing Gear Geometry

Deflected Gear (After Landing)

Crushable Honeycomb

Undelected Gear (Before Landing)
3.1.2 Shock Absorber

The shock absorber chosen was a fluid-filled type, in which the travel of the piston forces fluid through a small orifice in the piston, damping the spacecraft's motion. A cutaway of the MSP shock is given in Figure 16.

**Figure 16: Shock Absorber Cutaway**

From the derivation given in class, which uses Bernoulli's principle to equate mass flow generated by the motion of the piston of area $A$ to the resulting mass flow through the orifice of area $A^*$, the following relation results:

$$\frac{A}{A^*} = \left(\frac{2F}{\rho A}\right)^{1/2} \left(1/v\right)$$

where

- $F = 5589 \text{ N}$ (landing force)
- $\rho = 917 \text{ kg/m}^3$ (assumed from SAE 30W oil)
- $v = 5.5 \text{ m/s}$ (landing velocity)

which results in

$$A^* = (1.57)A^{3/2}$$
Then, for the sake of a compact design, choosing a piston radius of 1 cm gives

| Piston area = $A = 3.14 \times 10^{-4}$ m$^2$ |
| Orifice area = $A^* = 8.74 \times 10^{-6}$ m$^2$ |

From the geometry of the landing gear, it can be seen that to effect a 20 cm vertical deflection of the footpad, it is necessary to have

| Piston stroke travel = 8.4 cm |

To design the dimensions of the shock cylinder for strength, it was necessary to find the maximum pressure developed in the cylinder. Here, it was assumed that immediately upon impact, the maximum force of landing was transmitted to the shock absorber (this is obviously a more-stringent-than-reality assumption). So here, the pressure is given by

$$P = \frac{F}{A}$$

where again
- $F = 5589$ N
- $A = 3.14 \times 10^{-4}$ m$^2$

**Maximum pressure in shock cylinder = 43.76 atm (44.36 x 10$^5$ Pa)**

Finally, to find the maximum stresses occurring in the cylinder, we use the relations from Ugural,

$$\sigma_r = (r_i^2)P \left(1 - \frac{r_o^2}{r_i^2}\right)\left(\frac{r_o^2}{r_i^2} - \frac{1}{P}\right)$$

$$\sigma_o = (r_i^2)P \left(1 + \frac{r_o^2}{r_i^2}\right)\left(\frac{1}{P} - \frac{1}{r_o^2}\right)$$

$$\sigma_z = \frac{F}{(P)(r_o^2 - r_i^2)}$$

where here,
- $r_i = 0.01$ m
- $r_o = 0.014$ m
- $P = 44.36 \times 10^5$ Pa

This results in the following

**Maximum stresses in the shock cylinder**

- $\sigma_r = 4.44$ MPa
- $\sigma_o = 24.59$ MPa
- $\sigma_z = 18.53$ MPa
As for the material chosen, at first Lockalloy, the Be-Al metal looked promising, but proved too heavy for original gear weight estimations, so the following metal was chosen (Bruhn):

<table>
<thead>
<tr>
<th>Material for Landing Gear: AZ61A Magnesium</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ultimate Strength: 96.53 MPa</td>
</tr>
<tr>
<td>Density: 1791 kg/m³</td>
</tr>
</tbody>
</table>

Obviously, our safety factors are well-satisfied, and as long as the Magnesium can be machined to 4 mm, there should be no problem. In fact, the weight savings are tremendous:

**Mass of shock absorber:**

\[
\text{Fluid} = \text{(stroke)} \times \text{(area)} \times \text{(density)} = (0.084 \times 0.000126 \times 917) = 97.1 \text{ g}
\]
\[
\text{Cylinder} = \text{(length)} \times \text{(area)} \times \text{(density)} = (0.10 \times 0.012^2 - 0.01^2) \times (2100)
\]
\[
= \frac{29.0 \text{ g}}{} 
\]
\[
\text{TOTAL} = 0.189 \text{ kg}
\]

In addition, the main strut must be checked for Euler buckling, where the critical load applied to the strut is given by

\[P_{cr} = \frac{\pi^2EI}{L^2}\]

After performing this analysis once, it was found necessary to use Beryllium, where

<table>
<thead>
<tr>
<th>Material for Landing Gear Main Struts: Lockalloy (Be-Al)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Young's Modulus: 2.9 x 10^{11} Pa</td>
</tr>
<tr>
<td>Yield Strength: 3.24 x 10^8 Pa</td>
</tr>
<tr>
<td>Density: 1826 kg/m³</td>
</tr>
</tbody>
</table>

\[
E = 2.9 \times 10^{11} \text{ Pa}
\]
\[
I = \pi r^4/4 = (\pi)(0.01)^4/4 = 7.85 \times 10^{-9} \text{ m}^4
\]
\[
L = 0.40 \text{ m}
\]

And thus

**Critical buckling load for main strut = 140,400 N**

Which is well within the factor of 3 for safety used for most buckling and instability cases, considering that the applied load will be about 5900 N.

**Mass of gear struts:**

\[
\text{Braces (2)} = (2) \times (\text{length}) \times (\text{area}) \times (\text{density}) = (2)(0.335)(\pi)(0.005^2)(1791)
\]
\[
= 94.2 \text{ g}
\]
\[
\text{Main Strut} = (0.422)(\pi)(0.012)(1826) = 229 \text{ g}
\]
\[
\text{TOTAL/leg} = 0.323 \text{ kg}
\]

**Total mass of landing gear = (0.0189 + 0.154 + 0.0942)(3) + footpads = 1.50 kg.**
Figure 17: MSP Bus Construction
3.2 Bus Analysis

The construction of the main structure of the MSP is, as given in Figure 17, a hexagonal frame of I-beams with a center platform made from a composite hexagonal sandwich platform. The following is the analysis done for its design.

Figure 18: Analytical Modelling of Bus Shelf

3.2.1 Platform Design

The bus platform encounters its main loading condition upon landing, when the inertial force of the ascent vehicle stack impinges upon the thrust ring at its center, as shown in Figure 18. Because of its hexagonal shape, the platform was modelled as a circular plate under a distributed loading, as shown in Figure 18. From Urugal, we have

\[ \sigma_{r_{\text{max}}} = (3/4)(P_o)(r^2)/(t^2) \]
where 

$$P_0 = \frac{\text{load/area}}{r_2} = \frac{(45\text{kg})(75.6\text{m/s}^2)}{(\pi)(0.32\text{m}^2)} = 33.2\text{ KPa}$$

$t = \text{thickness of plate} = 0.025\text{ m}$

Thus  

$$\sigma_{\text{max}} = \frac{(0.75)(33,200)}{\pi} = 7.64\text{ KPa}$$

From Bruhn, we are able to find a honeycomb sandwich material as shown in Figure 19:

<table>
<thead>
<tr>
<th>Material for platform: Al 5056 honeycomb</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flexural rigidity: approx 4.5 MPa</td>
</tr>
<tr>
<td>Density: 76.4 kg/m$^3$</td>
</tr>
</tbody>
</table>

So, for our needs, the platform will have a mass of

**Platform mass = 0.71 kg**

And obviously, well within the range of safety.

---

**Figure 19: Platform material construction**
3.2.2 I-Beam Design

The frame of the bus is constructed of I-beams, depicted in Figure 20. Here, the designer assumed a worst-case condition, in which a side beam is loaded with the total vertical force of landing transmitted through the shock absorber into the I-beam at midpoint.

Figure 20: Analytical Modelling of Bus Frame I-Beams
From the classical beam theory analysis, the maximum stress occurs from bending at midpoint in the beam, and is given by

\[ \sigma_b = \frac{M(h/2)}{I} \]

Where \( M = \text{moment applied} = (\text{landing force})(\text{half of beam length}) \)
\( = (5589\text{N})(0.20\text{m}) = 1118\text{ Nm} \)
\( h = \text{height of beam} = 0.20\text{ m} \)

And where \( I \) for the beam is calculated by taking the moments of area each of the portions in Figure 21 separately, as is classically done. So, using the notations in Figure 21, we have

\[ \text{Figure 21: I-Beam Cross-Section Dimensions} \]

<table>
<thead>
<tr>
<th>Portion</th>
<th>Area</th>
<th>Arm</th>
<th>((\text{Area})(\text{Arm})^2)</th>
<th>Own ( i )</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>((t)(b))</td>
<td>((0.5)(a+t))</td>
<td>((t)(b)(1/4)(a+t)^2)</td>
<td>((b)(t^3)/12)</td>
</tr>
<tr>
<td>2</td>
<td>((t)(a))</td>
<td>0</td>
<td>0</td>
<td>((t)(a^3)/12)</td>
</tr>
<tr>
<td>3</td>
<td>same as portion 1</td>
<td>0</td>
<td>0</td>
<td></td>
</tr>
</tbody>
</table>

Inserting all the relevant dimensions gives

\[ I_{\text{section}} = 7.61 \times 10^{-6} \text{ m}^4 \]

Which ultimately leads to

\[ \text{Maximum stress in beams} = 14.7 \text{ MPa} \]

Which is well below the same ultimate stress level of the Magnesium. As for the masses of the beams, we can use that

\[ \text{Mass} = (\text{cross-sectional area})(\text{length})(\text{density}) \]
\[ = [(2)(t)(b) + (a)(t)](L)(\rho) \]
\[ = [(2)(0.005)(0.05)+(0.19)(0.005)](0.40)(1791) \]
\[ = 1.04 \text{ kg/beam} \]

\[ \text{Total mass of frame} = 6.24 \text{ kg} \]
3.3 Launch Rings

The launch rings are located as shown in the cutaway of Figure 11. The lower, or ascent launch ring is designed to join the ascent stage to the bus frame, and the upper, or apogee launch ring is designed to join the apogee stage to the ascent stage. Both rings are to be strong enough to keep the craft together upon the two major vehicle forces of chute deployment and landing. The two rings are depicted in Figure 22.

Naturally, the chute deployment is a condition that puts the rings in tension, while the landing phase puts both in compression. Roughly, both need to withstand joining the vehicle at 10g's in tension and 7.5g's in compression. Because the rings separate parts of the vehicle that have different masses, the forces each experiences will be different under the different conditions.

The resulting applied stress on each ring will be given by

\[ \sigma_{app} = \frac{F}{A} \]

where \( F = ma \)

where \( m \) = mass of the portion of the spacecraft supported by the ring in question under load
\( a \) = acceleration in question

The designer then compared this resulting applied stress to the critical stress for the case of a loaded truncated cone, which is given by

\[ \sigma_{cr} = \left( \frac{\gamma E t t}{(r_1)(r_2)(\cos \alpha)} \right) / (3(1-\nu^2))^{1/2} \]

where
\( \gamma \) = buckling safety factor = 0.33
\( E \) = Young's modulus = 2.9 \times 10^{11} (Beryllium)
\( t \) = thickness of cone = 0.005 m
\( \nu \) = Poisson's ratio = 0.33 (non ferrous)
\( r \) = smaller radius
\( \alpha \) = cone half angle
The results for different loading conditions for the two Beryllium rings of Figure 22 are summarized as follows:

<table>
<thead>
<tr>
<th>Ring</th>
<th>r (m)</th>
<th>α (deg)</th>
<th>Loading (g)</th>
<th>Supported Mass (kg)</th>
<th>( σ_{\text{app}} ) (Mpa)</th>
<th>( σ_{\text{cr}} ) (Mpa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Upper</td>
<td>0.0475</td>
<td>54.8</td>
<td>7.5 comp. (landing)</td>
<td>2.75</td>
<td>0.28</td>
<td>3550</td>
</tr>
<tr>
<td>Lower</td>
<td>0.100</td>
<td>26.6</td>
<td>7.5 comp. (landing)</td>
<td>45.0</td>
<td>2.17</td>
<td>2620</td>
</tr>
<tr>
<td>Upper</td>
<td>0.0475</td>
<td>54.8</td>
<td>10.0 tens. (chute)</td>
<td>87.0</td>
<td>11.44</td>
<td>3550</td>
</tr>
<tr>
<td>Lower</td>
<td>0.100</td>
<td>26.6</td>
<td>10.0 tens. (chute)</td>
<td>34.0</td>
<td>2.12</td>
<td>2620</td>
</tr>
</tbody>
</table>

Very obviously, the applied stresses in the rings are several orders of magnitude below the critical stresses, and therefore acceptable.

Mass of rings

Finally, we can roughly compute the masses of the rings, assuming that they are less in mass than a cylinder of similar radius and thickness.

\[ M = (\text{density})(\text{volume}) < (\rho)(\pi)(D)(t)(h) \]

where for each ring

\[ \rho = 1826 \text{ kg/m}^3 \]
\[ D = 35.0 \text{ cm} \]
\[ t = 5.0 \text{ mm} \]
\[ h_{\text{upper}} = 9.0 \text{ cm} \]
\[ h_{\text{lower}} = 15.0 \text{ cm} \]

**Mass of upper launch ring = 0.90 kg**

**Mass of lower launch ring = 1.50 kg**
4 Summary

This MSP structural design, then, represents a involved attempt to satisfy mission criteria for a spacecraft designed to return a sample of Martian soil for Earthbound analysis.

Although at points overdesigned, and in some areas not detailed enough, the structure (or at least its critical components) have been shown to be able to withstand some of the major loads placed upon them by the harsh and varied environments the craft would encounter on its round trip to and from the Red Planet.
References


Various NASA Technical Reports:


### MSP Structural Designer Quarterly Time Sheet

<table>
<thead>
<tr>
<th>Date</th>
<th>Time</th>
<th>Activity</th>
</tr>
</thead>
<tbody>
<tr>
<td>4/6</td>
<td>2.25* hr</td>
<td>Discussion of layout and propulsion basics, mission scenario</td>
</tr>
<tr>
<td>4/7</td>
<td>0.50</td>
<td>Research on Viking, computer sketches</td>
</tr>
<tr>
<td>4/8</td>
<td>1.00</td>
<td>Preliminary MSP sketches</td>
</tr>
<tr>
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<td>3.00</td>
<td>Stress calc's on bus, gear, platform</td>
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<td>MSP final drawings started</td>
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<td><strong>Hours for quarter</strong></td>
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*Involved group meetings (some not listed)
Section B:
Thermal Control and Guidance

Terrance Yee
SPACECRAFT THERMAL CONTROL

Thermal design for this craft is primarily driven by the need to keep the propellant tanks between +10 and +55 degrees Celsius and the batteries between 0 and +25 degrees Celsius (ref D6). The most critical conditions are during nighttime on the surface and during ascent in the atmosphere. To remain within the allowable temperature range, a variety of surface coatings, insulation, and active cooling and heating units are employed.

Orbital Thermal Control

Thermal control before descent is accomplished through proper surface coatings and orientation of the spacecraft. After release from the MOV and a short burn to precess in its orbit, the MSP assumes a slow "rotissary" mode of rotation about its axis of symmetry. The bottom of the hexagonal bus section is gold coated; the top of the hexagonal bus is painted white and the sides are vertically striped with 50% gold and 50% white paint. This paint scheme together with the shadow of the heat shield allow the spacecraft to alter the effective absorbivity vs. emissivity ratio by changing the spacecraft angle with respect to the sun. The angle of the symmetry axis with respect to the incident radiation is chosen by the onboard computer to give a daylight (non-eclipse) equilibrium temperature of 20°C. If the spacecraft begins to warm, the computer will tilt the top of the craft towards the sun thereby shadowing some of the craft with the heat shield and exposing the white top of the hexagonal bus. This maneuver will also be used prior to reentry to precool the craft to 15°C which will lessen the impact of the aerodynamic heating on reentry. If during orbit the spacecraft is too cool, the top can be tilted away from the sun exposing the gold bottom of the hexagonal bus which will tend to warm the craft.

This flexible effective surface was chosen due to the eccentricity of the martian solar orbit which alters the intensity of solar radiation depending on what time of year the mission is
performed during. It also allows us to be less precise in our calculation of the required mix of coatings. Those calculations were performed by analyzing the steady state radiation equilibrium setting the incident solar radiation and albedo radiation equal to the thermal radiation leaving the craft. Approximating the craft by a sphere of radius \( R \) the following results were obtained for the different mars-sun distances:

\[
Q_{\text{em,s}} = Q_{\text{abs}}, \quad \varepsilon 4\pi R^2 G T^4 = \alpha_s (I_s + I_{\text{alb}}) \varepsilon R^2 \quad I_{\text{alb}} = \frac{q I_s}{8} \left[ 1 - \sqrt{1 - \frac{R_m}{R_m + \text{600 km}}} \right]
\]

\( T = 273.15 ^\circ \text{K} \)

minimum mars-sun distance \( \Rightarrow \frac{\alpha_s}{\varepsilon} = 2.356 \)

maximum mars-sun distance \( \Rightarrow \frac{\alpha_s}{\varepsilon} = 3.429 \)

selecting gold \( (\frac{\alpha_s}{\varepsilon} = 6) \) and Titanium Oxide white paint with Methol (.222)

\( \Rightarrow \) minimum distance : 37% gold, 63% white paint

\( \Rightarrow \) maximum distance : 56% gold, 44% white paint

For an eclipse the following relations were used to find the time of the eclipse:

for a 600 km altitude orbit above Mars the angle that the penumbra makes with respect to the mars-sun radius is only 0.193° so we can approximate this angle as 0°.

therefore we can write

\[
\sin \theta = \frac{R_m}{R_m + \text{600 km}} \Rightarrow \theta = 58.22 ^\circ
\]

\[
\tau = 2\pi \sqrt{\frac{R_m^3}{\mu}} (R_m + \text{600 km})^{3/2}
\]

\[
\tau_{\text{ecl}} = \frac{2\theta}{360 ^\circ} \tau
\]

\( \Rightarrow \tau_{\text{ecl}} = 2986 s = 4 \text{min 26 sec} \)
Using a lumped capacitance approximation for spheres of different radii the change in internal thermal energy is set equal to the energy radiated away during the eclipse. Specifying the allowable change in temperature as ten degrees Celsius allows us to solve for the surface temperature. Once this is known a minimum ratio of insulation thickness to thermal conductivity can be derived from Fourier's law of conduction. The actual thickness of insulation is seen to be greater than the minimum required so the craft will stay within the allowable temperature range. An analysis was also performed assuming no insulation and use of electric resistance heaters during eclipse as a comparison.

$$m \rho \alpha \Delta T = \frac{4 \pi R^2 C(T - T_s)}{L} \Rightarrow K = 0.334 \frac{W}{mK} \quad (\text{for } R = 1m)$$

$$\text{actual } K \text{ used} = 2.9 \times 10^3 K/\text{m}^2$$

Reentry Heating

The allowable temperature change on reentry is specified as 5 degrees Celcius which allows a factor of safety of 2 before the batteries overheat. Originally an ablative heat shield was envisioned but after considerable research and a more accurate determination of the reentry heating it was found that a radiative heat shield using a high temperature multilayer insulation would function quite well in this environment.

The reentry heating of the shield was approximated by calculating the stagnation enthalpy and pressure at the stagnation point for numerous points during the descent then the heat transfer
rate was found using the chart provided. Setting the total heat
transfer rate to the shield equal to the heat radiated away from the
front surface allowed us to solve for the maximum temperature at
this point on the front of the shield. This allowed us to choose the
type and thickness of insulation to give the desired heat transfer
and backface temperature. The beryllium honeycomb on the
backface was not allowed to exceed 700 degrees Fahrenheit. It was
also discovered that the thermal lag for a heat shield of about 45
seconds allowed for an additional factor of safety in the backface
support structure since it will be at a lower temperature than
equilibrium when it experiences the maximum drag force.(ref)

Two computer programs were written for this section. The
first helped collate the atmospheric, trajectory, and heating rate
tables and charts. The second calculated a few stagnation
temperatures outside the boundary layer in front of the shield
assuming no dissociation (not a very good assumption) in order to
get a rough idea of how much error is induced by assuming a 300
degree Kelvin wall in using the given handout since a very high
freestream temperature would make the difference in heat shield
temperature less important.(see appendix D1)

\[ q_{\text{max}} = \varepsilon G T_{\text{max}}^{\text{th}} \Rightarrow T_s = 913K \]

\[ T_{\text{back}} = 644K \Rightarrow q_{\text{allow}} = 11,311 \text{ W for 180 seconds} \]

\[ L_{\text{reg}} = 0.112 \text{ mm of Aluminum fiberglass multilayer blanket (max temp: 1033K)} \]

minimum thickness manufactured: 5mm, \( f = 1311 \text{in}^2 \text{lb}^{-1} \)

so our 5mm thick insulation weighs 2.55 kg and keeps the back
face very cool with almost no heat transfer to the vehicle.

In the recirculation zone behind the heat shield the convective
heat transfer is approximately one tenth of that experienced at the
stagnation point.(ref D3 ) Approximating the temperature of the
recirculation zone as 1400K and modeling the craft as a cylinder we
get the following convective heat transfer relations for the bus
during descent:
In order to maintain stability during descent a reinforced carbon-carbon "drogue chute" is deployed when entering the atmosphere. This cone-shaped body is tethered to the rear of the craft on either side of the motor by high temperature titanium cables. The drag cone is stored as the lid to the main chute canister and when deployed by coil spring will trail 10 meters behind the craft. Since the heat shield stagnation point was at 991K we assumed this temperature as the worst case for the entire cable. To size the drag cone we assumed a 50 m/s gust which is approximately a 2% variation in drag force (about 100N). Letting this disturbance force act at 75cm from the center of the heatshield leads to a 75Nm disturbance torque. If we allow a maximum moment arm of the drag cone force of 1m then we require a minimum force of 75N which translates into an area of 220cm^2 if we assume that the drag force per area of the heatshield is equal to the drag force per area of the drag cone. With a right cone (45 degree half angle) this translates into a surface area of 707cm^2 and a base radius of 15 cm. Adding a factor of safety to this, we increase the base radius to 21cm and the surface area to 980cm^2. Making the cone out of 5mm reinforced carbon-carbon (RCC) leads to a cone weight of 784g. A simple stress analysis of the cable shows that a diameter
of 3.175mm is capable of supporting more than 1000N force and weighs 360g. Therefore a total weight estimate of the stability system is 1.3kg allowing 156g for fixtures such as bolts and a spring.

Thermal Control on the Surface

After reentry the MSP may be warmer than desired, therefore a method of rapid cooling is required. The method chosen to accomplish this is three small louvers which are placed 120 degrees apart on the ascent body above the ascent fuel tanks. The louvers are opened by thermally set springs and held shut by latches which the computer opens upon landing and locks at night. They are top hinged, opening outward to expose an 8cm high, 3cm wide opening for the martian atmosphere to flow through.
Some concern was originally felt for the cooling effect of the expanding helium in the propulsion system. The expansion is solved using the ideal gas law:

\[
\frac{p_1 v_1}{p_2 v_2} = \frac{T_1}{T_2}
\]

\[
p_1 = 20 \text{ MPa} \quad \rho_1 = 4.905 \times 10^{-3} \text{ m}^3
\]

\[
p_2 = 3 \text{ MPa} \quad v_2 = 2.9575 \times 10^2 \text{ m}^3
\]

\[
T_2 = 220.8 \text{ K}
\]

\[
mC_p\Delta T = 75 \text{ kJ} \quad \text{for a .2 kg He tank}
\]

\[
\Delta T_{\text{min}} = -0.15 \text{ °C} \quad \text{per He tank during descent}
\]

\[
\Delta T_{\text{max}} = -1.5 \text{ °C} \quad \text{per He tank during ascent}
\]

This temperature decrease is certainly acceptable during the rocket firings while heat from the rocket motor and aerodynamic heating are offsetting the helium expansion effects.

This brings us to the next problem area, the rocket motor. While the thrust chamber is ablatively cooled and has negligible heat transfer to the rest of the ship, the nozzle and throat are radiatively cooled and must be isolated from the rest of the ship. This is done using a three layered system consisting of a highly reflective outer coating of silver with a protective coating of SiO (150 Angstroms), a layer of beryllium as a heat sink, and a final layer of refrasil, a multilayer high temperature insulation. The analysis of this system follows:
A worst case approximation of the system would be as parallel plates for which

\[ q_{12} = \frac{\varepsilon_1 \varepsilon_2 G (T_1^4 - T_2^4)}{\varepsilon_1 + \varepsilon_2 - \varepsilon_1 \varepsilon_2} \]  

(Ref. 87)

\[ \Rightarrow T_2 = 622 K, \quad q_{12} \tau_{b_{in}} = 611 K \]

thermal resistance to conduction: \[ 3.95 \frac{m^2 K}{W} \]  
thermal resistance to radiation: \[ \frac{1}{4} \frac{m^2 K}{W} \]

\[ q_{rad} = \frac{324}{3.95} = 82 \frac{W}{m^2} \]

\[ \Rightarrow \frac{4}{5} \text{ of energy transferred to spacecraft} \]

\[ \Rightarrow 48.9 K \text{ to craft} \]

\[ \Rightarrow 2.4^\circ C \Delta T \text{ for ascent stage} \]

\[ \Rightarrow +5^\circ C \text{ for deorbit burn} \]

The last ground problem is cooling at night due to radiation loss and convection. It is this area which requires the most insulation and indeed sizes the insulation used. A small UDMH decomposition fuel cell is used to warm the ship at night.

Since pressure stays the same at day and night, \[ \phi_{night} = \phi_{day} \]

\[ \phi_{day} = \frac{1.96 \times 10^{-5} \text{ W/m}^2}{(180 K)} \]

\[ \varepsilon = 0.37 \times 10^{-7} \frac{\text{W}}{\text{m}^2} \]

\[ \Rightarrow q_1 = 5.484 \times 10^4 \frac{m^2}{s} \]

\[ \Rightarrow \frac{1}{\varepsilon} = 0.852 \]

\[ Re = \frac{V \rho \varepsilon}{\mu} = \frac{20 m/s \times 0.34 m}{4.67 \times 10^{-5} \text{ W/m}^2} = 15361 \]

using the relation

\[ N_u = K Re \frac{Pr^{1/3}}{\varepsilon} \]

\[ \Rightarrow N_u = 69.1 \]

\[ \Rightarrow h = \frac{N_u K}{\varepsilon} = 1.68 \frac{W}{K m^2} \]

\[ \Rightarrow 293 - 150 \]

\[ \Rightarrow \frac{1}{2.6 \times 10^{-5}} \ln \left( \frac{150}{147} \right) \]

\[ \Rightarrow 30.6 \text{ W} \]
Ascent Heating

Using the same methods as the descent the ascent trajectory was analyzed but with a nose made of RCC and .96cm of refrasil protecting the skin. Again as in the descent, the craft was allowed to cool to 15C before launch in order to alleviate heating constraints.
Rendezvous Vehicale

The rendezvous vehical has very little on board that is temperature sensitive; the electronics are very simple and can be built to withstand wide temperature variations, there are no batteries or fuel, only superstructure, dirt, solar cells and an omnidirectional radio beacon. The surfaces not covered by solar cells are covered by gold to keep the temperature near 290K.

GUIDANCE AND POWER SUPPLY

Guidance is provided by inertial rate gyros and by sun and star sensors when outside the atmosphere. Final landing guidance is provided by a system which uses a prescanned map of the landing area which is stored in on board ROM computer memory. (see appendix D2) The map records safe landing sites and local landmarks for the entire probable region of landing. Just after the parachute is jettisoned and before the retro maneuver a snapshot image is taken of the terrain and matched by landmarks and distinct features to the map stored in memory. Once the computer determines where it is, it can use the map to direct the craft to the closest known safe landing zone. A laser range finder is used to determine distances to the ground to allow the computer to make the decisions when to engage the main chute, when to jettison the chute, and when to engage and disengage the retro motor.
The sun and star sensors along with inertial rate gyros and an accurate timer allow the ascent computer to determine the correct timing of the firing of the apogee kick motor as well as its correct orientation. Also the ascent computer will set a timer on the apogee motor that will pyrotechnically pierce the combustion chamber to effectively kill the thrust when the correct orbit has been reached.

The on board computer actually consists of three parts. The first part is the descent computer which is the ROM board and the guidance for descent. This part is located in the hexagonal bus section and weighs about 1kg. The second part is the main or ascent computer. It is located in the ascent stage and weighs just under 2kg. Its functions include powered flight guidance, directing the sample collector, and orbital equations. The third part is a subpart of the main computer. Its function is to periodically check all systems, especially thermal, during the dormant periods while the MSP waits for a particular orbital conjunction. It has limited control over the attitude jets and thermal system and will activate the main computer if a serious problem is detected. It has negligible power usage and negligible mass.

Power is supplied to all but the rendezvous stage by zinc-silver oxide batteries with a power density of 80 watt-hr/lb. (ref D2). There are two groups of batteries, one in the ascent stage and one in the hexagonal bus. The batteries for the sample collection motor are subdivided into many smaller cells in order to give a high voltage while the other batteries supply a lower voltage usable by the computers. The main computer system is assumed to operate at 30 watts and the descent computer at 8 watts, the monitor subcomputer is estimated to have such a low power usage as to be negligible.

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<th>Power (W)</th>
<th>Time (min)</th>
<th>Energy (watt-hr)</th>
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<td>30</td>
<td>35</td>
<td>17.6</td>
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<tr>
<td>Sample Collection</td>
<td>330</td>
<td>10</td>
<td>55</td>
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<tr>
<td>Final Descent</td>
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<td>5</td>
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<td>Deorbit and Reentry</td>
<td>30</td>
<td>60</td>
<td>30</td>
</tr>
<tr>
<td>Adjusting Precession</td>
<td>30</td>
<td>40</td>
<td>20</td>
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</table>
Burn to precession | 30  | 10  | 5  | 27.63
unforeseen/reserves | total | 158.4

Total battery weight is 0.9kg, 0.8kg is stored in the hexagonal bus.

The beacon on the rendezvous stage is an omnidirectional beacon broadcasting a narrow band 100W signal of 1 millisecond duration five times every second. To provide this power 145 cm$^2$ cross sectional area of solar cells must be used. On the 10 cm diameter rendezvous stage this translates into a 14.5 cm strip around the cylinder of 11% efficient silicon cells at furthest distance from the sun. If 17% efficient GaAs cells are used the strip will be 9.4 cm wide. The remainder of the surface should be covered in gold to keep the craft warm. Silicon solar cells are currently available and their reliability is high so they were chosen for this mission, giving a weight of 0.15kg to the power system.
REFERENCES


D7. AGARD-CP-411, "Advances in Guidance and Control Systems and Technology", NATO 1986


<table>
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<th>WEEK</th>
<th>HOURS</th>
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<td>25</td>
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<tr>
<td>total</td>
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Figure 1  CONVECTIVE HEATING AT A STAGNATION POINT
(100-PERCENT CO₂)
63-10199

For wall temperature = 300 °K,
chemical reactions are included (multi-component
diffusion is neglected).

q = heat transferred to wall, per unit area and unit time.
Rₙ = nose radius
Hₛ = stagnation enthalpy

\[ Hₛ = \frac{n}{c_p} \]
III. BOUNDARY-LAYER SOLUTIONS

Heat-transfer rates and boundary-layer profiles have been found from a sin solution of the laminar boundary-layer equations. Local similarity assumptions are used to reduce the equations to a set of five first-order total differential equations with variable coefficients. The solution is then found by an iterative technique using 400-point profiles. A detailed description of the solution is also given elsewhere.

A number of assumptions are involved in this boundary-layer solution and must be mentioned for completeness: The atomic composition is assumed to be constant throughout the boundary layer. In essence, this means that multicomponent diffusion is neglected. The local similarity assumptions are exact for the case studied in this report. All fluid properties are continuously variable throughout the boundary layer. The effects of chemical reactions are included by employing the reaction-conductivity concept.

All calculations in the report have assumed a wall temperature of 300°K. Similarly, only stagnation-point solutions are reported here. Other cases with more realistic wall temperatures and other pressure gradients can be obtained by application of this method.
Capture of sample container

1. MOV zeroes in on beacon, waits until closest approach, then maneuvers to docking distance.

2. MOV opens door to bay which housed MSP

3. MOV maneuvers to capture sample container inside bay

4. MOV closes bay doors, sealing container inside

5. Net system inside bay draws container tight against a corner read for return trip to Earth. See diagrams:

   ! as drawstrings pull on beads, the net rises, pulling the sample toward the ceiling. When the net reaches the top the right side of the net is reeled along the ceiling while the left side is reeled through a gap at the top left edge. This process continues until the sample is held firmly against the upper left edge of the docking bay.
STAGNATION PRESSURE ON SHIELD IS 9.324666E-07
X AXIS IS 112.718
VELOCITY(m/s) = 3520
HEATING RATE (W/m^2) = 68.85323
ALTITUDE(km) = 150

STAGNATION PRESSURE ON SHIELD IS 1.227144E-04
X AXIS IS 231.6017
VELOCITY(m/s) = 3562
HEATING RATE (W/m^2) = 93.89644
ALTITUDE(km) = 100

STAGNATION PRESSURE ON SHIELD IS 1.155286E-03
X AXIS IS 203.1755
VELOCITY(m/s) = 3552
HEATING RATE (W/m^2) = 23495.77
ALTITUDE(km) = 80

STAGNATION PRESSURE ON SHIELD IS 3.14366E-03
X AXIS IS 188.1773
VELOCITY(m/s) = 3520
HEATING RATE (W/m^2) = 28372.25
ALTITUDE(km) = 70

STAGNATION PRESSURE ON SHIELD IS 7.70426E-03
X AXIS IS 166.9835
VELOCITY(m/s) = 3425
HEATING RATE (W/m^2) = 45774.96
ALTITUDE(km) = 60

STAGNATION PRESSURE ON SHIELD IS 1.679816E-02
X AXIS IS 139.1813
VELOCITY(m/s) = 3220
HEATING RATE (W/m^2) = 45661.6
ALTITUDE(km) = 50

STAGNATION PRESSURE ON SHIELD IS 2.303351E-02
X AXIS IS 120.0036
VELOCITY(m/s) = 3020
HEATING RATE (W/m^2) = 45548.23
ALTITUDE(km) = 45

STAGNATION PRESSURE ON SHIELD IS 2.919942E-02
X AXIS IS 97.29362
VELOCITY(m/s) = 2750
HEATING RATE (W/m^2) = 46104.9
ALTITUDE(km) = 40

STAGNATION PRESSURE ON SHIELD IS 3.438264E-02
X AXIS IS 74.39153
VELOCITY(m/s) = 2400
HEATING RATE (W/m^2) = 44774.96
ALTITUDE(km) = 35
X AXIS IS 49.35343
VELOCITY(m/s) = 1950
ALTITUDE(km) = 30

STAGNATION PRESSURE ON SHIELD IS 3.192682E-02

X AXIS IS 30.8775
VELOCITY(m/s) = 1500
ALTITUDE(km) = 25

STAGNATION PRESSURE ON SHIELD IS 8.349418E-02
X AXIS IS 49.79182
VELOCITY(m/s) = 1990

HEATING RATE (W/m²) = 15959.39

STAGNATION PRESSURE ON SHIELD IS 2.101189E-02
X AXIS IS 15.34456
VELOCITY(m/s) = 990
ALTITUDE(km) = 20

HEATING RATE (W/m²) = 6649.746

STAGNATION PRESSURE ON SHIELD IS 1.179943E-02
X AXIS IS 8.059934
VELOCITY(m/s) = 600
ALTITUDE(km) = 15

HEATING RATE (W/m²) = 3103.215
10 CLS
20 REM**********PROGRAM TO DO HEAT SHIELD HEATING (MARS)**********
30 COLOR 7
40 INPUT "ALTITUDE (km)";ALT
50 INPUT "STATIC TEMP AT INFINITY(K)";T1
60 COLOR 9
70 INPUT "STATIC PRESS AT INFINITY(mb)";P1
80 COLOR 10
90 INPUT "SOUND SPEED AT INFINITY(m/s)";A
100 COLOR 11
110 MW=43.824
120 COLOR 12
130 INPUT "VELOCITY (m/s)";V
140 COLOR 14
150 PRINT
160 GAMMA=A^2*MW^2/(8314*T1)
170 CP=8314/MW*GAMMA/(GAMMA-1)
175 PRINT "CP";CP."830"
180 HS=CP*T1+V^2/2
190 X=HS/8314*MW*T1
200 M=V/A
210 PRINT "GAMMA";GAMMA
220 PRINT "M";M
230 P2=P1*((GAMMA+1)*(GAMMA-1)/2*M^2*(GAMMA-1)/(GAMMA+1))^{(GAMMA-1)/(GAMMA-1)}
240 PRINT "STAGNATION PRESSURE ON SHIELD IS";P2
250 LPRINT "X AXIS IS";X
260 PRINT "X AXIS IS";X
270 COLOR 13
280 INPUT "INPUT Q*SQR(Rn) IN Btu/(Ft (3/2)*sec)";Q1
290 PRINT "HEATING RATE IN W/m^2 IS";Q1
300 COLOR 15
310 PRINT "VELOCITY(m/s)= ";V. "HEATING RATE (W/m^2)= ";Q1. "ALTITUDE(km)=";ALT
320 LPRINT
340 GOTO 30
<table>
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<tr>
<th>ANNA</th>
<th>VELOCITY</th>
<th>SOUND SPEED</th>
<th>STATIC TEMP UPSTREAM</th>
<th>STA. TEMP</th>
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<td>1.24</td>
<td>3.231</td>
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<td>168</td>
<td>5994.24</td>
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<td>1.3</td>
<td>3.194</td>
<td>193</td>
<td>190</td>
<td>6384.406</td>
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<td>1.34</td>
<td>3.063</td>
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<td>231</td>
<td>6546.158</td>
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<td>1.38</td>
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<td>223</td>
<td>213.8</td>
<td>5322.552</td>
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<td>1.33</td>
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<td>228</td>
<td>5883.902</td>
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<td>1.324</td>
<td>1.420</td>
<td>240</td>
<td>241</td>
<td>417.1375</td>
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<tr>
<td>1.322</td>
<td>522</td>
<td>245</td>
<td>236.2</td>
<td>876.9264</td>
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<tr>
<td>1.323</td>
<td>1000</td>
<td>244</td>
<td>242.3</td>
<td>353.5449</td>
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<tr>
<td>1.32</td>
<td>429</td>
<td>247</td>
<td></td>
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</tbody>
</table>
Section C:
Propulsion Systems

Myles Baker
Propulsion:

1: Requirements:

The Mars Surface Probe places several demands on its propulsion system. From the Mission Scenario, at least five separate impulses are required:

<table>
<thead>
<tr>
<th>Event:</th>
<th>Required Delta V:</th>
<th>Required Thrust:</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial orbital maneuvering</td>
<td>&lt;10 m/s</td>
<td>&lt; 10000 N</td>
</tr>
<tr>
<td>Descent burn</td>
<td>&lt;200 m/s</td>
<td>&lt; 10000 N</td>
</tr>
<tr>
<td>Landing retro burn</td>
<td>&lt;200 m/s</td>
<td>&gt; 500 N</td>
</tr>
<tr>
<td>Ascent from Mars surface</td>
<td>~3920 m/s</td>
<td>&gt; 1000 N</td>
</tr>
<tr>
<td>Apoapsis injection</td>
<td>~830 m/s</td>
<td>&lt; 200 N</td>
</tr>
</tbody>
</table>

In addition, the propulsion system should be as small and light as possible, since mass is at a premium in such a mission. It should be as simple as possible to increase reliability, and it should be very stable due to the approximately two years it will spend in flight to Mars. An additional restriction placed on the propulsion system is that it never cause the vehicle to accelerate at more than 100 m/s², which is reflected in the maximum allowable thrusts.

With these criteria in mind, the decision was to use two separate engines: one for the low-thrust apoapsis injection burn and the other for all other, higher thrust burns. Due to the varying requirements of the first four impulses, it was decided to use a throttleable, re-startable liquid propellant engine. On the other hand, because of the decision to have no active guidance on the final payload capsule, the advantages of a solid motor far outweigh its slightly lower specific impulse. These
advantages include the lack of propellant control devices (valves, etc.), propellant tanks, and the fact that the pre-determined impulse delivered by a solid motor does not require any outside control mechanisms (other than a simple ignitor).

**Liquid Propellant Engine:**

The prime consideration in the design of this engine was its simplicity. There are several choices in the design of a rocket engine, each of which has a great impact on the final design of the vehicle. Some of the options, and selections for this mission, are listed below.

**Thrust:** The primary restrictions placed on the thrust of the engine are the imposed design restriction of less than 100 m/s² maximum acceleration and the need to overcome Mars’ gravity with minimum gravity losses, which requires as high a thrust as possible. From Newton's second law of motion,

\[ F_{t,\text{max}} = M_{\text{min}} a_{\text{max}} \]

With a burn-out ascent mass of 15 kg, a thrust of approximately 1500 Newtons is required.

**Cooling:** There are several methods of cooling a rocket engine, ranging from simple radiation cooling to regenerative cooling by pumping cool fuel through a cooling jacket around the engine. Since this engine is rather small, the extra hardware and complexity required for regenerative cooling eliminate it as a viable option, while radiation cooling is very well suited for such small engines. However, due to the fact that the actual combustion chamber is located inside the vehicle during much of
the mission, some additional cooling is required. Since there is a relatively short total burn time for the engine (approximately 120 seconds), an ablative combustion chamber liner will be used. Such materials are most reliable at chamber pressures of 1 MPa or lower, so 1 MPa is chosen as the maximum chamber pressure for the MSP liquid engine.

**Fuels:** There are many available fuels for liquid rockets, and various parameters for selecting a fuel for a particular mission. In this case, our requirements are a high specific impulse, high density, and easy storeability for up to two years without maintenance. The latter requirement eliminates several of the higher-energy propellants, such as the cryogenics (Liquid Hydrogen) and the fluoridated oxidizers. On this mission it was decided to use Nitrogen Tetroxide as an oxidizer and 50% Unsymmetrical Dimethyl Hydrazine, 50% Hydrazine (50-UDMH) as a fuel. This propellant combination has several advantages, including hypergolic ignition (no separate ignition system is required), extremely high storeability, a relatively high specific impulse, and a broad base of engineering experience with such propellant combinations.

**Propellant feed:** There are two types of propellant feed systems in use today: pump-fed and pressure-fed. The pump-fed system requires some additional machinery (a turbo pump, gas generator, and gas turbine), but it allows lower propellant tank pressures, and hence thinner tank walls, lowering the total mass of a large vehicle. Pressure-fed systems, on the other hand, maintain the entire propellant supply at a high pressure and thus
require heavier tanks, but virtually no machinery other than valves. For a vehicle as small as the Mars Surface Probe, the slight increase in tank mass is greatly compensated by the lack of supporting machinery, so a pressure-fed system is more appropriate.

**Materials:** Recent advances in high-temperature, relatively oxidation-resistant composite materials makes the choice of materials for the motor structure fairly straightforward. Multi-cycle burn times well over the required 100 seconds have been demonstrated with engines constructed almost entirely from a 3-Directional Novoltex\textsuperscript{R} type carbon fiber reinforcement in a silicon-carbide matrix\textsuperscript{[C-1]}. In the previously built engines, the fuels, flame temperatures, and motor sizes were similar to those in the Mars Surface Probe.

**Solid Motor:**

The solid rocket motor for re-insertion into the Mars Orbiting Vehicle's orbit is much simpler. It is only required to ignite, deliver a pre-determined impulse, and extinguish. The payload capsule will already be positioned in the correct attitude and spin-stabilized by the ascent stage, so no attitude control will be necessary. With spin-stabilization, if the spin rate is high enough, it is not necessary to adjust the thrust vector assuming a small uncertainty, so the engine mechanics can be extremely simple. The spin-stabilization of the upper stage will be provided by unwinding a pre-stressed "clockspring" mechanism, in effect spinning the ascent and final stages against
one another.

The fuel used in the solid motor is also well-tested, off the shelf technology. It consists of Aluminum and Ammonium Perchlorate powders in a polyurethane binder. This compound was chosen primarily for its slow burning rate of 6 mm/s, which makes it possible to remain within the 100 m/s² design acceleration limit. To minimize the motor case mass and surface area, a spherical case will be used, with a nozzle very similar in its proportions to that used in the liquid engine. The materials used are also very similar to those used in the liquid engine, with the addition of a 4-Directional Carbon-Carbon Sepcarb® throat insert to minimize erosion in this critical area.

Propellant Tanks:

There are two necessary sets of propellant tanks. The first set is used for the orbital maneuvering, de-orbit, and retro burns and is housed in the main bus of the vehicle, which is left behind on the planet surface. It contains approximately 5 liters each of N₂O₄ and 50-UDMH pressurized to 1.2 MPa, allowing for 5% ullage, and 2 liters of helium gas at 25 MPa to pressurize the tanks. These tanks will be split up into three identical sets of three tanks each, with similar tanks (e.g. N₂O₄, 50-UDMH, and He) arranged symmetrically about the axis of the vehicle. Due to the small size of the tanks, and the relatively low pressure, 2014-T6 Aluminum alloy (chosen for its good machining properties) will be used for the propellant tanks, and Ti 6Al-4V Titanium alloy (for its exceptionally high strength) for the Helium. Unfortunately,
Nitrogen Tetroxide is not compatible with aluminum tanks, so an interior protective coating will have to be used.

The second set of tanks, used only for the ascent burn, must contain about 19 liters of each of the propellants and 7 liters of helium, again allowing 5% ullage. The configuration for the ascent tanks is shown in figure C.7, with the materials similar to those in the descent tanks. The nested configuration was chosen in order to keep the ascent envelope as small as possible, and only weighs about fifty grams more than the optimum spherical tanks.

**Propulsion System Analyses:**

The design of a spacecraft is necessarily an iterative process, with the results from each part of the analysis affecting the parameters the remaining phases are based on. This necessitates a certain number of assumptions in the preliminary vehicle design, with successive refinements as the design progresses. The assumptions used in the analyses below have been through three such iterations, and so are fairly accurate.

**Liquid Propellant Engine:**

There are many parameters of interest in the design of a rocket engine, but perhaps the most important are the thrust ($F_t$), specific impulse ($I_{sp}$), mass ($m$), and dimensions, which will be developed here.

The combustion characteristics of Nitrogen Tetroxide and 50-UDMH are tabulated in several references, each with slightly
different values. The values below are taken from Sutton and Ross\textsuperscript{[c-2]}:

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flame Temperature:</td>
<td>3100 K</td>
</tr>
<tr>
<td>Gas Constant ( (R) ) of Exhaust:</td>
<td>396 J/(kg-K)</td>
</tr>
<tr>
<td>Ratio of Specific Heats (( \gamma )):</td>
<td>1.24</td>
</tr>
<tr>
<td>Density: ( \text{N}_2\text{O}_4 ):</td>
<td>1450 kg/m(^3)</td>
</tr>
<tr>
<td>Density: 50-UDMH:</td>
<td>910 kg/m(^3)</td>
</tr>
<tr>
<td>Mixture Ratio (O:F)</td>
<td></td>
</tr>
<tr>
<td>by Mass:</td>
<td>1.62</td>
</tr>
<tr>
<td>by Volume:</td>
<td>1.01</td>
</tr>
</tbody>
</table>

Table X.X: Combustion Properties of Liquid Fuels

**Nozzle:** Assuming a ratio of \( \frac{A_e}{A_t} = 20 \) between the nozzle exit area \( (A_e) \) and throat area \( (A_t) \), the mach number (obtained through a computer iteration) at the exit plane is found to be \( M_e = 3.93 \).

Given the chamber pressure \( (P_c = 1 \text{MPa}) \), the pressure at the nozzle exit plane \( (P_e) \) can be obtained from the equations for isentropic flow:

\[
P_e = P_c \left[ 1 + \frac{\gamma - 1}{2} M_e^2 \right]^{\gamma/\gamma - 1}
\]

\[
P_e = 10^6 \left[ 1 + \frac{1.24 - 1}{2} (3.93)^2 \right]^{\gamma/\gamma - 1}
\]

\[
P_e = 4439 \text{ Pa}
\]

Given the chamber pressure, the exit pressure, and the back pressure \( (P_b) \), the thrust coefficient, defined as

\[
F_t = P_c A_t C_f
\]

can be obtained from the relation

\[
C_{f,\text{ideal}} = \sqrt{\frac{2\gamma \nu c}{\varepsilon A_c}} \left[ \left( \frac{\rho_c}{\rho_b} \right)^{\gamma - 1/2} \right] + \frac{\rho_c}{\rho_b} \varepsilon
\]

which, for this engine, at the mean surface pressure at the MSP landing site of 700 Pa, is

\[
C_{f,\text{ideal}} = 1.77
\]
Assuming a bell nozzle profile of maximum efficiency (a perfect bell nozzle of expansion ratio 30:1 truncated to 20:1), the nozzle losses (due to friction and nozzle divergence) are approximately 2%\(^2\), so the actual thrust coefficient is

\[ C_f = C_{f,\text{ideal}} \times 0.98 = 1.73 \]

Given a maximum thrust of 1500 N and a maximum \(P_c\) of 1 MPa, the required nozzle throat area is

\[ A_t = F_t/(P_cC_f) = 1500/(1.73 \times 10^6) \]

\[ A_t = 9 \text{ cm}^2 \]

For a throat area of 9 cm\(^2\), the inside diameter at the throat is 3.4 cm, the exit diameter is 15.1 cm, and from the bell nozzle optimization curves\(^3\), the nozzle length is 18.6 cm.

The mass flow through the nozzle can be obtained by a mass balance at the throat, where flow is necessarily sonic, so

\[ m = A_t P_c \sqrt{\frac{Y}{R}} \left( \frac{2}{Y + 1} \right)^{1/(Y-1)} \]

\[ m = (10^6 \text{Pa})(10^{-4} \text{m}^2) \left( \frac{2}{1.4 + 1} \right)^{1/(1.4-1)} \]

\[ m = 0.533 \text{ kg/s} \]

at a maximum thrust of

\[ F_{t,\text{max}} = C_f A_t P_c = (1.73)(10^{-4})(10^6) \]

\[ F_{t,\text{max}} = 1560 \text{ N (at surface)} \]

\[ 1575 \text{ N (in vacuum)} \]

This gives the liquid engine a specific impulse of

\[ I_{sp} = \frac{F_t}{m} \]

\[ I_{sp} = 298 \text{ sec (at surface)} \]

\[ 301 \text{ sec (in vacuum)} \]

**Combustion Chamber Geometry:** In order to minimize pressure losses in the combustion chamber, the velocity just upstream of the throat should be very small, implying a large chamber to
throat area ratio \( (A_c/A_t) \). In this case, \( A_c/A_t \) will be set to 6.0. Thus the radius of the combustion chamber is
\[
    r_c = \sqrt{\frac{6A_c}{\pi}}
\]
\[
    r_c = \sqrt{\frac{0.0054}{\pi}} = 4.45 \text{ cm}
\]
A second parameter in the design of a liquid-fuel combustion chamber is the characteristic length \( (l^*) \), which is defined as the chamber volume divided by the throat area. This length is a useful measure of the amount of time a given element of the fuel-oxidizer mixture remains in the chamber, and must be high enough to ensure complete combustion. In the reaction control thrusters used on many early American launch vehicles, Monomethyl Hydrazine and \( N_2O_4 \) were used as propellants and the characteristic length was approximately 0.5 meters \[^{[2]}\]. This thrust chamber was used as a model, thus requiring a combustion chamber length (approximating the chamber as a cylinder) for the MSP engine of
\[
    l_c = l^{*}(A_t/A_c)
\]
\[
    l_c = 0.5/6 = 8.33 \text{ cm}
\]
Allowing for 5 cm clearance at the injector for fuel lines and the thrust vector control (gimballing) mechanism, the total liquid engine dimensions are:
\[
    D_{\text{max}} = 16 \text{ cm}
\]
\[
    l_{\text{tot}} = 30 \text{ cm}
\]

**Ablative Cooling Insert:** Ablative materials are often used in cooling small, short duration thrust chambers. When exposed to the heat of combustion, the ablator, a fiber reinforcement in an organic matrix, decomposes, leaving a porous, insulating layer of char on the surface. The decomposition of the organic matrix
is an endothermic process, and the pyrolysis gases released tend to insulate the remaining material from the hottest of the combustion chamber gases.

\[ \text{Char layer} \rightarrow \text{Hot gases} \rightarrow \text{pyrolysis gases} \]

**Figure 5.1: Ablative Chamber Cooling**

A silica fiber reinforcement in a phenolic matrix will be used as the ablative liner of the combustion chamber. The depth of the porous char layer, and hence the necessary thickness of the ablative material, is given by the relation

\[ d_c = \sqrt{\text{burn time in seconds}} \text{ mm} \]
\[ d_c = \sqrt{120} \text{ mm} = 11 \text{ mm} \]

This analysis does not consider the fact that the engine will reach ambient (spacecraft) temperature between firings, so the actual char depth should be significantly less. Nevertheless, adding a margin of safety, the thickness of the ablative material should be approximately 1.5 cm.

**Propellant Injectors:** Given the overall mass flow rate, the mixture mass ratio, and the propellant densities, it is easily seen that the necessary volumetric flows for maximum thrust are

\[ Q_f = 0.225 \text{ l/s (50-UDMH)} \]
\[ Q_o = 0.227 \text{ l/s (N}_2\text{O}_4) \]

and, given an orifice-type injector, the volumetric flow rate is
given by the equation \[Q = A_i C_i \sqrt{\frac{1}{2} \frac{\Delta P_i}{\rho}}\]

where \(C_i\) is an experimentally determined discharge coefficient, usually between 0.6 and 0.9 (assume 0.75), and \(P_i\) is the pressure drop across the injector. For a set of \(n\) injectors, the total area is

\[
\sum A_i = \frac{Q}{C_i \sqrt{\frac{1}{2} \Delta P_i}}
\]

which is, for a 200 kPa pressure drop,

\[
A_{i,f} = 1.43 \times 10^{-5} \text{ m}^2
\]

\[
A_{i,o} = 1.82 \times 10^{-5} \text{ m}^2
\]

For efficient combustion, it is desirable to have as many doublet-impinging pairs of injectors as possible in order to atomize the propellants effectively. This is limited, however, by the machining of the injector manifold and the injectors themselves. One possibility is a set of 19 injector pairs arranged as shown in figure 2. The diameter of an individual injector is

\[
D_i = \sqrt{\frac{A_i}{\pi}}
\]

so, for 19 injectors, the diameters for the fuel and oxidizer injectors, respectively, are

\[
D_{i,f} = 0.979 \text{ mm}
\]

\[
D_{i,o} = 1.104 \text{ mm}
\]

The injectors should be positioned at angles such that the net momentum of the fuel and oxidizer streams is parallel to the axis of the engine, so

\[
m_f v_f \sin(\varphi_f) = m_o v_o \sin(\psi_o)
\]

\[
\psi_o = \sin^{-1} \left[ \frac{m_f v_f D_o A_{i,o}}{m_o v_o \sqrt{\frac{1}{2} \Delta P_i}} \right]
\]
Assuming a fuel injector angle of 20° from the engine axis, the oxidizer injector angle should be

\[ \varphi_o = \arcsin \left[ \left( \frac{0.20\text{°}}{0.331} \right)^{\frac{1}{0.450}} \right] \left( \frac{1.72}{0.45^2} \right) \sin 20° \]

\[ \varphi_o = 15.5° \]

**Chamber Stress Analysis:** The stress analysis of the combustion chamber is very straightforward, considering only the hoop stresses. For 3-Dimensional Carbon-Silicon Carbide Novoltex\(^{(R)}\), \( \gamma = 2300 \text{ kg/m}^3 \), \( \sigma_y = 80\text{MPa} \), and for hoop stresses,

\[ t_w = r_c (P_c / \sigma_y) \times \text{S.F.} \]

where S.F. is the safety factor (1.5) and \( r_c \) contains the ablative chamber liner. Substituting in the values,

\[ t_w = (0.0445 + 0.015) \times \frac{10^6}{(80 \times 10^6)} \times 1.5 \]

\[ t_w = 1.1 \text{ mm} \]

A combustion chamber with walls this thin would be extremely difficult to construct reliably, so an additional safety factor will be added to the wall thickness by increasing it to a uniform 2.0 mm.

**Mass Estimates:** Given the material properties and dimensions calculated above, it is very straightforward to compute the mass of the engine as the sum of the masses of the components. The volume of 3-D Novoltex\(^{(R)}\) in the combustion chamber and nozzle shells is

\[ V_{c/c} = (0.002 \text{ m}) (2\pi (r_c + 0.015 \text{ m}) + A_n) \]

Where \( A_n \), the nozzle surface area, is given by\(^{[C-2]}\)

\[ A_n = 70 \times A_t \]

\[ V_{c/c} = (0.002) (2\pi (0.0445 + 0.015) + 70(0.0009)) \text{ m}^3 \]

\[ V_{c/c} = 8.74 \times 10^{-4} \text{ m}^3 \]
Figure C-3 MSP Liquid Engine
So the mass of the chamber and nozzle shells is approximately

\[ M_{\text{shell}} = 2300 \text{ kg/m}^3 \times 8.74 \times 10^{-4} \text{ m}^3 \]

\[ M_{\text{shell}} = 0.20 \text{ kg} \]

The ablative insert mass, assuming a uniform density of 1740 kg/m\(^3\) is

\[ M_{\text{abl}} = 2\pi(r_c + 0.015/2) \times l_c \times \pi \times 0.015 \]

\[ M_{\text{abl}} = 2\pi(0.0445 + 0.0075) \times 0.0833 \times 1740 \times 0.015 \]

\[ M_{\text{abl}} = 0.71 \text{ kg} \]

And the mass of the injector plate, assumed stainless steel (\(\rho = 7600 \text{ kg/m}^3\)), 1 cm thick, and 50% hollow for fuel manifolds,

\[ M_{\text{inj}} = \pi(r_c + 0.015)^2 \times 0.01 \times 7600 \times 50\% \]

\[ M_{\text{inj}} = 0.80 \text{ kg} \]

Thus the entire engine mass is approximately

\[ M_{\text{eng,liq}} = 0.20 + 0.71 + 0.80 = 1.71 \text{ kg} \]

Allowing some excess for throat reinforcement, plumbing, and the gimballing bearing,

\[ M_{\text{eng,liq}} = 2.0 \text{ kg} \]

**Solid Orbital Insertion Motor:**

The analysis of the solid motor is very similar to that for the liquid motor. The properties of the solid Polyurethane/Ammonium Perchlorate/Aluminum fuel, again from Sutton and Ross\(^{[2]}\), are:

- **Flame Temperature:** 3088\(^{\circ}\)K
- **R of Exhaust Products:** 284 J/(kg-K)
- **Y of Exhaust Products:** 1.17
- **Burning Rate:** 6 mm/s
- **Density:** 1750 kg/m\(^3\)

**Dimensions:** Assuming a chamber pressure of 1 MPa and a
nozzle exit area ratio of 20:1, the Mach number at the exit plane is found (again by computer iteration) to be 3.65. At the exit,
\[ P_e = (10^6) \left[ 1 + \frac{1.17 \cdot 3.65^2}{2} \right] \]
\[ P_e = 5452 \text{ Pa} \]

And the thrust coefficient is
\[ C_{f,\text{ideal}} = \sqrt{\frac{2(1.17^2)}{1.17+1} \left[ 1 - \left( \frac{5452}{1000000} \right)^{1.17} \right] + \frac{54520}{1000000}} \]
\[ C_{f,\text{ideal}} = 1.849 \]

Again applying the 98% nozzle efficiency (similar geometry to liquid engine):
\[ C_f = 1.812 \]

Assuming a maximum thrust of 200 N (2.0 kg final mass at 100 m/s²),
\[ A_t = \frac{F_t}{C_f P_e} = \frac{200 \text{ N}}{1.812 \cdot \frac{10^6}{(10^6 \cdot 5452)}} \]
\[ A_t = 1.103 \times 10^{-4} \text{ m}^2 \]
\[ A_t = 1 \text{ cm}^2 \]

So the actual maximum thrust is
\[ F_{t,\text{max}} = (10^{-4})(10^6)(1.812) = 181.2 \text{ N} \]

The mass flow is again given by
\[ m = \rho A \frac{V}{C_{f,\text{ideal}}} = \rho A \sqrt{\frac{T_e}{R_0 \gamma}} \left( \frac{2}{Y+1} \right)^{\frac{1}{Y+1}} \]
\[ m = (10^6)(10^{-4})\sqrt{\frac{1.17}{1.17+1}} \left( \frac{2}{1.17+1} \right)^{\frac{1}{1.17+1}} \]
\[ m = 0.0687 \text{ kg/s} \]

And hence the specific impulse:
\[ I_{sp} = 181.2/((0.0687)(9.81)) = 269 \text{ sec} \]
The specific impulse, from the ideal rocket equation, gives the propellant mass and volume:

\[
M_p = 2.0 \text{ kg} \times \left[ \exp \left( \frac{\Delta V}{g \cdot \rho} \right) - 1 \right]
\]

\[
M_p = 2.0 \times \left[ \exp \left( \frac{2 \cdot 1750}{3 \cdot 0.0006} \right) - 1 \right]
\]

\[
M_p = 0.734 \text{ kg}
\]

\[
V_p = M_p/\gamma_p = 4.19 \times 10^{-4}
\]

The burning area \((A_b)\) for these conditions can be found from

\[
A_{b,\max} = \frac{\dot{m}}{(\rho \cdot r_b)}
\]

\[
A_{b,\max} = \frac{(0.0687)}{(1750)(0.0006)}
\]

\[
A_{b,\max} = 65.4 \text{ cm}^2
\]

**Grain Design:** For a neutral-burning, cylinder-based grain configuration (such as the conventional star grain), the cylinder length and radius, in order to meet the criteria on area and volume, must be

\[
r = \frac{2V_p}{A_b} = \frac{2(4.19 \times 10^{-4})}{6.54 \times 10^{-3}}
\]

\[
r = 17.1 \text{ cm}
\]

\[
l = \frac{V_p}{(\pi r^2)} = 6 \text{ mm}
\]

Obviously this grain configuration is absurd, so another must be sought. Since this motor operates in free fall, a neutral burn is not required, and the options for grain configurations is almost endless. This motor will have a "sphere within a sphere" grain configuration as shown in figure C.4. The burning area as a function of time is

\[
A_b = \frac{\pi (r_c + \xi t)}{\xi} \left[ \xi^2 - (\xi + \xi t - r_c)^2 \right]
\]

and the remaining propellant mass is

\[
M_p = \frac{\pi \xi^2}{3} \left[ 2r_c^3 + \frac{3}{2} \xi (r_c + \xi t)^2 - \frac{1}{3} (r_c + \xi t)^3 \right]
\]

Using the fact that the thrust in vacuum is directly proportional
to $A_b$ and $P_c$, a computer spreadsheet was programmed to calculate the chamber pressure and acceleration as functions of time (figures x.x and x.x), and various dimensions were tested. The final propellant grain dimensions are a total grain radius of 4.65 cm and an initial bore radius of 1.0 cm. As can be seen from the graphs, the maximum acceleration is $90.0 \text{ m/s}^2$ and the maximum chamber pressure is 1.25 MPa. This pressure is slightly higher than initially assumed, but easily feasible.

**Stress Analysis and Mass Estimate:** The stress analysis is nearly a duplicate of that used for the liquid engine except for the fact that this case is spherical rather than cylindrical. Chamber wall thickness is given by

$$t_w = \frac{r_c}{2} \times \left( \frac{P_c}{\sigma_y} \right) \times \text{S.F.}$$

With similar materials,

$$t_w = \left( \frac{0.0465}{2} \right) \left( \frac{1.25}{80} \right) (1.5)$$

$$t_w = 0.545 \text{ mm}$$

This chamber is also impossible to reliably construct, so again, a 2 mm uniform wall thickness is assumed. This gives the total mass of the case and nozzle:

$$M_{\text{tot}} = (0.002)(4\pi(0.0465)^2 + 70(0.0001))(2300)$$

$$M_{\text{tot}} = 0.157 \text{ kg}$$

Allowing 0.093 kg for throat inserts, internal insulation, and attachment hardware, the mass of the solid motor is

$$M_{\text{fuelled}} = 0.984 \text{ kg}$$

$$M_{\text{dry}} = 0.250 \text{ kg}$$

**Solid Motor Spin-up Mechanism (Clockspring):** In order to have stability in a spin-stabilized spacecraft, the rotational
Figure C.4
Solid Insertion Motor

Mass
Initial: 988 g
Burnout: 250 g
Max Thrust: 216 N
Total Impulse: 1880 N-s

D = 4.65 cm
r = 1.0 cm

L = 12.0 cm
L_m = 6.20 cm
D_o = 5.10 cm
Figure C.5: Final Stage Acceleration Profile

Figure C.6: Solid Motor Chamber Pressure Profile
kinetic energy must be much greater than the maximum anticipated disturbing moment. Assuming a 1 mm misalignment of the thrust vector with the center of mass at 200 N thrust, the disturbance torque is

\[ M_0 = (0.001 \text{ m})(200 \text{ N}) = 0.2 \text{ N-m} \]

The rotational kinetic energy of a spinning body is given by

\[ E_K = I \omega^2 / 2 \]

where \( I \) is the moment of inertia about the spin axis and \( \omega \) is the spin rate in radians per second. Assuming the rendezvous stage to be a uniform cylinder of \( r = 5 \text{ cm}, m = 2.7 \text{ kg} \), then \( I \) is given by

\[ I = mr^2 / 2 = (2.7 \text{ kg})(0.05 \text{ m})^2 / 2 \]
\[ I = 0.00375 \text{ kg-m}^2 \]

Assuming a kinetic energy of five times the disturbance torque,

\[ \omega = \sqrt{\frac{2M_0}{I}} \]
\[ \omega = \sqrt{\frac{0.2 \text{ N-m}}{0.00375 \text{ kg-m}^2}} \]
\[ \omega = 23.09 \text{ radians/sec} \]

With a safety factor of over 2 to include the depletion of propellant mass,

\[ \omega = 50 \text{ radians/sec} \]

The angular acceleration of a rotational mass is

\[ \alpha = \frac{T}{I} = \frac{d^2 \theta}{dt^2} \]

and the torque exerted by a rotational spring is

\[ T = -k \theta \]

which implies

\[ \frac{d^2 \theta}{dt^2} + \frac{k}{I} \theta = 0 \]
\[ \theta = -\theta_0 \cos(\sqrt{\frac{k}{I}} t) \]
\[ \omega = \frac{d\theta}{dt} = \omega_0 \sqrt{\frac{k}{I}} \sin(\sqrt{\frac{k}{I}} t) \]
At the maximum rotational velocity, this is
\[
\omega_{\text{max}} = \frac{\theta_0 \sqrt{K}}{I} \\
k = \left(\frac{\omega_{\text{max}}}{\Theta_0}\right)^2 I
\]
Assuming a spin-up angle of 10 revolutions (62.8 radians),
\[
k = \left(\frac{50}{62.8}\right)^2 (0.00375)
\]
\[k = 0.00238 \text{ N-m/rad}
\]
This is a very small torque, and a spring to provide it, along with a bearing and shaft, should weigh no more than 0.1 kg.

**Liquid Propellant Tanks:**

**Descent:** During the descent phase of the mission, there are three separate liquid motor firings: Orbital Maneuvering, De-Orbit, and Retro Landing. The mass of fuel required for each burn is calculated by applying the ideal rocket equation,
\[
M_{\text{fuel}} = M_{\text{veh}} \left[ \exp \left( \frac{\Delta V}{g \sqrt{\rho}} \right) - 1 \right]
\]
using the \( V \)'s and \( M_{\text{veh}} \)'s at the given time. Applying the equation,
\[
M_{\text{om}} = 0.802 \text{ kg} \\
M_{\text{d-o}} = 7.353 \text{ kg} \\
M_{\text{ret}} = 1.850 \text{ kg (max)} \\
M_{\text{desc}} = 10.105 \text{ kg}
\]
Allowing almost 2 kg of fuel for ullage and contingencies such as an extended retro firing, the total descent fuel mass is 12 kg. Using the mixture ratio and densities of the propellants, the necessary tankage is 5.1 liters each of \( \text{N}_2\text{O}_4 \) and 50-UDMH. The volume of Helium necessary to keep these tanks pressurized to 1.2 MPa can be calculated by modelling the process as an adiabatic
expansion of an ideal gas where
\[ pV^\gamma = \text{Constant} \]

For an adiabatic expansion from \( P_1, V_1 \) to \( P_2, V_2 \), the required \( V_1 \) is
\[
V_1 = V_2 \times \frac{P_1^{\gamma/\gamma}}{(P_2^{\gamma/\gamma} - P_1^{\gamma/\gamma})}
\]
\[
V_{\text{He}} = (2 \times 5.1) \times \frac{1.2^{1/1.66}}{(25^{1/1.66} - 1.2^{1/1.66})}
\]
\[ V_{\text{He}} = 2.0 \text{ liters} \]

Taking three identical sets of tanks, the radii are
\[
r_f = 7.4 \text{ cm}
\]
\[
r_o = 7.4 \text{ cm}
\]
\[
r_{\text{He}} = 5.4 \text{ cm}
\]

Using 2014-T6 Aluminum (\( \gamma = 2800 \text{ kg/m}^3, \sigma_\gamma = 386 \text{ MPa} \)) for the propellant tanks and Ti-6Al-4V Titanium (\( \gamma = 4430 \text{ kg/m}^3, \sigma_\gamma = 999 \text{ MPa} \)) for the Helium, the required wall thicknesses are, from an analysis identical to that used in the spherical solid rocket case:
\[
t_f = 0.17 \text{ mm}
\]
\[
t_o = 0.17 \text{ mm}
\]
\[
t_{\text{He}} = 1.01 \text{ mm}
\]

The thicknesses of the fuel and oxidizer tanks are thinner than can be practically manufactured, so assuming a minimum machineability of aluminum to be 0.3 mm, the total tank mass is
\[
M_t = 12 \pi \left( r_f^2 t_f \rho_{\text{Al}} + r_o^2 t_o \rho_{\text{Al}} + r_{\text{He}}^2 t_{\text{He}} \rho_{\text{He}} \right)
\]
\[
M_t = 12 \pi \left( (2)(2800)(0.074^2)(0.0003) + (4430)(0.054)^2(0.00101) \right)
\]
\[ M_t = 0.84 \text{ kg} \]
Adding over 50% for valves and plumbing, the total fuel system mass for the descent stage is \( M = 1.50 \) kg.

**Ascent:** The fuel requirements for the ascent burn cannot be calculated from the ideal rocket equation due to the effects of aerodynamic drag, but since the burn time and thrust are known, the mass of fuel expended can be calculated using the specific impulse:

\[
M_p = \frac{F_t \cdot t_b}{\dot{q}_p I_p}
\]

Where \( t_b \) is the burn time in seconds. For the ascent stage, with a 1500 Newton constant thrust for 82 seconds,

\[
M_p = \frac{(1500)(82)/((9.81)(298))}{((9.81)(298))}
\]

\[
M_p = 42.0 \text{ kg}
\]

Adding 5% for ullage,

\[
M_p = 44.1 \text{ kg}
\]

Using a breakdown similar to the descent tanks, this requires volumes of

\[
V_f = 18.5 \text{ liters}
\]

\[
v_0 = 18.9 \text{ liters}
\]

\[
V_{He} = 7.3 \text{ liters}
\]

In order to minimize the envelope of the ascent vehicle, the tank configuration shown in figure C.7 was selected. The spherical caps and the cylindrical section have a radius of

\[
R = \frac{3V_f}{4\pi R}
\]

\[
R = 18.5 \text{ cm}
\]

and the cylindrical section (containing \( N_2O_4 \)) has a length of

\[
L = \frac{V_0}{\pi R^2}
\]

\[
L = 18.0 \text{ cm}
\]
Figure 11-7
Agent Propellant Flow and Removal
The Helium tank has radius

\[ R_{He} = \sqrt[3]{\frac{3V}{4\pi}} \]
\[ R_{He} = 12.0 \text{ cm} \]

Again using 2014-T6 Aluminum for propellant tanks and Ti-6Al-4V Titanium for the Helium, the required thicknesses are:

\[ t_{sphere} = 0.43 \text{ mm} \]
\[ t_{cyl} = 0.86 \text{ mm} \]
\[ t_{He} = 2.26 \text{ mm} \]

The total mass of the ascent tank structure is

\[ M = \pi ((6R_{sphere}^2t_{sphere} + 2RL_{cyl})(\rho_{Al}) + 4R_{He}^2t_{He}(\rho_{Ti}) \]
\[ M = \pi (6(0.185)^2(0.00043)+2(0.185)(0.180)(0.00086)) \times (2800) + \pi(4)(.120)^2(0.00226)(4430) \]
\[ M = 3.07 \text{ kg} \]

Adding 0.43 kg for plumbing and valves, the final tank mass is

\[ M = 3.50 \text{ kg} \]
References:


Section D:
Orbital Mechanics

Bob Langberg


**Orbital Transfer and Ascent/Descent Trajectories**

**Planetary Characteristics Pertinent to Orbital Calculations**

<table>
<thead>
<tr>
<th>Characteristic</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass</td>
<td>6.4182 $10^{23}$ (kg)</td>
</tr>
<tr>
<td>Universal Gravitational Constant</td>
<td>6.6732 $10^{-11}$ (m$^3$/kg s$^2$)</td>
</tr>
<tr>
<td>Gravitational Constant for Mars</td>
<td>4.2830 $10^{13}$ (m$^3$/s$^2$)</td>
</tr>
<tr>
<td>Equatorial Gravitational Acceleration</td>
<td>3.73 (m/s$^2$)</td>
</tr>
<tr>
<td>Mean Radius</td>
<td>3402 (km)</td>
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<tr>
<td>Sidereal Day</td>
<td>24 hours 37 minutes 23 seconds</td>
</tr>
<tr>
<td></td>
<td>24.623 (hours)</td>
</tr>
<tr>
<td></td>
<td>1477.4 (minutes)</td>
</tr>
<tr>
<td></td>
<td>88643. (seconds)</td>
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<tr>
<td>Angular Velocity</td>
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<tr>
<td></td>
<td>4.2529 $10^{-3}$ (rad/min)</td>
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<tr>
<td>Rotational Speed (Equatorial)</td>
<td>14.468 (m/min)</td>
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<tr>
<td></td>
<td>0.24113 (m/s)</td>
</tr>
<tr>
<td>(28.5° Latitude)</td>
<td>12.715 (m/min)</td>
</tr>
<tr>
<td></td>
<td>0.21191 (m/s)</td>
</tr>
</tbody>
</table>

**Mars Orbiting Vehicle (MOV) Orbital Parameters**

Circular Ascending Node at 250° Longitude
Inclination 30°
Altitude 600 (km) above mean radius
Orbital Radius 4002 (km)
Orbital Velocity 3.2715 (km/s)
Orbital Period 2 hours 8 minutes 6 seconds
2.1351 (hours)
128.11 (minutes)
7686.4 (seconds)

**Landing Site**

Depression Hellas, near Crater 29
Altitude 4.2 (km) Below Mean Radius
Radius of Site 3397.8 (km)
Longitude 283 (Deg)
Latitude -28.5 (Deg)
Descent Analysis

Ideal Descent Analysis, Neglecting Atmospheric Drag

Initially the descent trajectory was modeled without taking the atmosphere into account, using ideal equations for elliptical orbital motion (Kepler Motion). In this analysis there was an orbital plane change, calculated with spherical trigonometry, at the same time as the deorbit burn. This combined burn put the Mars Surface Probe (MSP) on an elliptical orbit that intersects the Martian Surface at the landing site. It was optimized with the constraint that the flight angle at landing could not be less (not shallower) than 15°. Orbits of eccentricity varying from 0.08 (nearly circular) to 0.93 (an extremely narrow ellipse) were examined. The delta v required was least at the more circular ellipses, with an absolute minimum occurring at the ellipse with eccentricity of 0.14, which had the minimum plane change angle (10.1°), unfortunately its angle with the horizon upon landing was 7.83° which was too shallow.

The best case which did meet the 15° requirement was the transfer ellipse with eccentricity of 0.270. It had a velocity at apoapse (4002 km) of 2.795 km/sec, which is 0.476 km/sec less than the MOV’s circular orbital velocity of 3.271 km/sec. This orbit takes the MSP through a 58.7° change in true anomaly, to land with an impact velocity of 3.409 km/sec. The entire deorbit takes 21.8 minutes, during which time Mars rotates 5.31° on its axis. The plane change angle for this case was 11.7°, which when combined with its velocity change for deorbit (0.476 km/sec) required a total delta v of 0.781 km/sec. This was the best that could be done with the plane change while neglecting the atmosphere.
Detailed Descent Analysis, With Atmospheric Drag

Avoiding the Plane Change Maneuver

After the ideal analysis was completed it was decided that a significant fuel savings (305 m/sec) could be attained by avoiding the 11.7° plane change. By leaving the MOV orbit with a small delta v of 8.1 m/s (0.2% of the circular velocity) the MSP would advance or recede by a rate of 2.5° per orbit. After 72 orbits, the MSP may move up to ±180°, allowing it to be positioned anywhere on the orbit. With precise positioning it may then deorbit as Mars rotates and the target point passes below, without the necessity of a plane change. This positioning would take up to 6.2432 Mars sidereal days (6 sidereal days and 9 hours) and would allow the MSP to position itself at any point in orbit to successfully deorbit and land at the specified sampling site. The extended time in orbit presents no apparent thermal problems.

Limits of Atmospheric Effects

To accurately model the effects of the atmosphere, the deorbit trajectory was assumed to be a Kepler ellipse until an altitude of 250 km was attained. The drag force on the MSP at an altitude of 250 km is $1.04 \times 10^{-4}$ N. The maximum drag is $6.92 \times 10^3$N which occurs at an altitude of 30.1 km. This difference, in excess of seven orders of magnitude, shows that not only is it a very reasonable assumption to ignore atmospheric effects above 250 km, but that atmospheric drag probably could have been ignored until about 150 km altitude where the drag is approximately $0.6$ N.

Incremental Time Step Analysis

At 250 km, the velocity and flight path angle was computed for the ideal transfer orbit. These were used as initial conditions for a time stepping analysis. In this analysis, the aerodynamic force on the MSP, which was assumed to be drag only, with no lift is computed. The drag
force was computed based on a Cd of 1.0 which is referenced to the projected area of the MSP. The gravitational acceleration was computed as a position of radius using the standard inverse square law at each position during the descent. These two forces were used to compute the acceleration in the radial and tangential directions and the resultant changes in velocity and position. Time increments ranged from one second to ten seconds, with five second increments used during the period of maximum aerodynamic forces. The MSP mass was changed as the heat shield is dropped as the parachute is deployed. The Cd reference area was changed at the time of parachute deployment, but the Cd is kept at 1.0. Both of these changes take place approximately 1.2 km above the landing site.

Terminal Velocity of Heat Shield and Parachute

Terminal velocity design charts were generated for the heat shield and parachute design analysis. These charts were used in the sizing process only and were not directly part of the incremental time/dynamic analysis. The final dimensions for the heat shield and parachute diameters are 1.5 and 4.0 m respectively.

Atmospheric Model

The atmospheric model used to calculate the variation of density with altitude, used for drag calculations is based on: The Mars Reference Atmosphere, 1982, published by The Committee on Space Research, Chapter 1: "Post-Viking Models for the Structure of the Summer Atmosphere of Mars", written by A. Seiff. Data from this source concerning the atmosphere in the southern hemisphere, during summer, at low altitudes was used to construct an exponentially decaying atmosphere model. The model is based on the density at the mean surface, \(1.78 \times 10^{-2} \text{ kg/m}^3\), and an exponential scale height, \(11.75 \text{ km}\), where the density has fallen off by \(1/e\).

\[\text{density(alt)} = \text{density}(0) \times \exp[-\text{alt}/\text{scale ht.}]\]
This model is most accurate where it has the greatest influence on the MSP, at low altitudes where the density is greatest, and the resulting aerodynamic forces are largest. See graphs for the shape of the atmospheric models. It is significant to note that the density has fallen off the linear scale at 70 km altitude. (0.26% of the density at the surface.

**Optimizing Deorbit Delta V with respect to Eccentricity**

The transfer ellipse that sets the initial conditions for the descent analysis was originally assumed to be the transfer ellipse from the ideal case with an eccentricity 0.27. Due to the drag on the heat shield and parachute, the landing trajectory is nearly vertical near the surface. Thus the 15° restriction on landing angle is no longer a constraint for the real descent case.

By doing a computation comparing the deorbit fuel required vs. the eccentricity of the exoatmospheric transfer ellipse, a simple relationship was developed. It shows that the delta v only depends on the original circular velocity and the transfer ellipse eccentricity, e:

\[
\text{Delta V} = V \text{ Circular} \times \left[ 1 - \sqrt{1-e} \right]
\]

This function is tabulated and graphed. It was found that another significant fuel savings could be realized by making an even shallower approach to the Martian surface, and the shallower the approach, the more fuel saved.

The problem with making the approach more shallow is that the distance traversed becomes greater, and the accuracy of the reentry is reduced. For this reason the transfer ellipse was made only slightly more circular. The original eccentricity was 0.27, requiring a delta v of 0.476 km/sec, and the new eccentricity is 0.10, requiring a delta v of only 0.169 km/sec. This causes the true anomaly traversed in descent to increase from 53.42° to 98.10°, and
the time of descent to increase from 22.37 to 36.04 minutes. However, most of this increased travel is exoatmospheric. The atmospheric true anomaly traversed (250 km to surface) increases from 11.23° to 16.00°, and the time in the atmosphere (250 km to surface) goes from 5.85 to 7.19 minutes.

These two trajectories are essentially the same near the surface where the velocities and flight path angles are: 161.7 vs. 159.9 m/sec and 66.8° vs. 68.8°, for the old and new cases respectively. These values are for an altitude 5 km above the surface. Thus, for this analysis they are essentially compatible, and since the e = 0.10 case is more fuel efficient, (0.169 vs. 0.476 km/sec, a 64% savings) it was chosen as the final descent trajectory. The fuel savings was approximately 10 kg.

Range Error Due to Uncertainty in Deorbit Delta V

Because we enter the atmosphere at a rather shallow angle, (17.6° at 100 km altitude), the accuracy of the trajectory was questionable. Since the vehicle is in free flight with only small attitude control thrusters for orbital corrections, it is of importance to know how sensitive the trajectory is to the only parameter we can effectively control, the deorbit delta v. The same time stepping analysis was run for a 2% decrease in delta v. The new case placed the MSP 13.64 km short of its original target, but this is a small change considering the fact that the deorbit trajectory traverses over 5800 km.
The final descent trajectory has the following characteristics:

<table>
<thead>
<tr>
<th>Characteristic</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Eccentricity</td>
<td>0.10</td>
</tr>
<tr>
<td>True Anomaly Traversed (exoatmospheric/atmospheric)</td>
<td>82.1°/16.0°</td>
</tr>
<tr>
<td>Radius at Apoapse</td>
<td>4002 km</td>
</tr>
<tr>
<td>V Apoapse</td>
<td>3.102 km/sec</td>
</tr>
<tr>
<td>V Terminal (300 m above surface)</td>
<td>&lt; 45 m/sec</td>
</tr>
<tr>
<td>Delta V (deorbit)</td>
<td>168 m/sec</td>
</tr>
<tr>
<td>Delta V (combined deorbit &amp; retrofire)</td>
<td>211 m/sec</td>
</tr>
<tr>
<td>Max. Mach (100 km alt.)</td>
<td>19.3</td>
</tr>
<tr>
<td>Max. Dyn. Press. (30.1 km alt.)</td>
<td>1968. N/m²</td>
</tr>
<tr>
<td>Max. Deceleration (30.1 km)</td>
<td>8.2 earth g’s</td>
</tr>
<tr>
<td>Time during max deceleration (3000/250 m/s)</td>
<td>75 sec</td>
</tr>
</tbody>
</table>
Ascent Analysis

The ascent was modeled the same way as the descent. The assumed Cd was again 1.0. The gravitational acceleration and the atmospheric density varied with radial position. Because the 82 second 1500 N ascent burn cannot be assumed to be impulsive for a ten minute flight, the thrust was considered to act in a constant manner and was just considered to be another force on the body for the duration of the burn. It was assumed that the mass decreased linearly during this period. The resulting acceleration was computed each second during the burn and every five or ten seconds after burn out. These accelerations were used to calculate the new velocity and position in polar coordinates, just as they were in the descent analysis. The basic parameters that were varied were the length of the 1500 N burn and the launch angle. From these initial conditions, the delta v required at 600 km to circularize the orbit at the same velocity as the MOV was computed.

Optimum Ascent Angle and Minimum Insertion Delta V

The required delta v was quite sensitive to launch angle, as measured from the horizon. The optimum angle varied between 60° and 62° in a nonlinear fashion, depending on the duration of the ascent burn. At each burn duration, the optimum angle was determined to within 0.5°. By launching at 0.5° too low an angle, the ascent stage fell approximately 25 km short of its 600 km target altitude. If launched at 0.5° too great an angle, it would reach 600 km with a flight path angle of about 10°, which must be corrected for with a larger delta v for circularization. The optimum launch angle allows the ascent stage to just reach the 600 km mark with a flight path angle as close to zero as possible to minimize the delta v required.
It was also found that with longer launch burns the velocity at 600 km was greater (at the optimum launch angle), requiring less and less insertion/circularization delta v. The penalty here is the increased launch mass for fuel and tanks. There is obviously an optimum distribution of the increased mass between the boost stage and the apoapse stage, but we had a driving design condition other that total mass to consider.

The driving criteria for the launch trajectory turned out to be the apoapse kick solid rocket motor. With its burn characteristics determined by the spherical grain configuration, the maximum delta v it could provide without exceeding 10 g’s was between 800 and 850 m/s. With our original launch mass of 46.4 kg the minimum possible delta v at the optimum launch angle was 1481. m/s, which could not safely be considered. A number of alternate cases were investigated.

The final optimum launch profile increased the launch mass to 56.8 kg (a 10.4 kg increase), neglecting the increase in fuel tank mass. This profile required a launch angle of 61.0° and a burn for 82 sec of 1500 N. The ascent stage reaches 600 km altitude after 10.8 minutes, with a flight path angle of only 2.92°. The required delta v is 828.0 m/sec.

If the flight path angle had been 61.5° the required delta v would have been 930.4 m/sec (a 102.4 m/s increase), and if launched at 60.5° the ascent stage would have fallen about 40 km short of its 600 km target.

Rendezvous Strategy

For rendezvous purposes the ascent stage could be placed into an orbit with a slightly different period than that of the MOV, thus letting the sample return vehicle catch up to the MOV without the MOV having the burn any additional fuel (passive recovery). As discussed for the deorbit positioning, a ±8.1 m/sec delta v would let the
ascent stage advance/recede by a rate of 2.5'/orbit. This allows for rendezvous in less than seven days. The ascent stage is not thermally sensitive at this point in the mission, therefore the rendezvous passage presents no thermal problems.

The final ascent trajectory has the following characteristics:

<table>
<thead>
<tr>
<th>Characteristic</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ascent Thrust (82 seconds)</td>
<td>1500 N</td>
</tr>
<tr>
<td>Delta V Launch</td>
<td>4020.3 m/sec</td>
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<tr>
<td>Maximum Acceleration at Launch</td>
<td>10 g's</td>
</tr>
<tr>
<td>Launch Angle</td>
<td>61.0°</td>
</tr>
<tr>
<td>Max. Mach (75 km alt.)</td>
<td>16.8</td>
</tr>
<tr>
<td>Max. Dyn. Press. (10 km alt.)</td>
<td>2732. N/m²</td>
</tr>
<tr>
<td>True Anomaly Traversed (exoatmospheric/atmospheric)</td>
<td>1.64°/2.37°</td>
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<tr>
<td>Time in Atmosphere (up to 250 km altitude)</td>
<td>196 sec</td>
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<tr>
<td>Flight Path Angle at Insertion</td>
<td>2.92°</td>
</tr>
<tr>
<td>V (600.9 km Altitude)</td>
<td>2456.1 m/sec</td>
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<tr>
<td>V Circular (MOV)</td>
<td>3271.4 m/sec</td>
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<tr>
<td>Delta V insertion</td>
<td>828.0 m/sec</td>
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<tr>
<td>Delta V (combined launch &amp; insertion)</td>
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<td>Maximum Acceleration at Insertion</td>
<td>&lt; 10 g's</td>
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<tr>
<td>Total Delta V: (m/s)</td>
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<tr>
<td>Positioning</td>
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<td>Deorbit</td>
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<tr>
<td>Retrofire</td>
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<tr>
<td>Ascent</td>
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<tr>
<td>Insertion</td>
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<tr>
<td>Rendezvous</td>
<td>5075.3</td>
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</table>
Orbit and Trajectory Supplemental Information

Mission Overview Diagram
Mission Profile
Ideal Descent
  Equations
    Plane Change Calculation
    Time Elapsed During Orbital Motion Calculation
Detailed Analysis
  Force Balance Diagram
  Pertinent Equations
  Iterative Time Stepped Procedure
    Used in Spread Sheet Calculations
  Selected Results From Descent Trajectory, Including:
    Parachute Deployment
    Retro Fire Requirements
Mars Atmosphere Model
  Linear Plot
  Logarithmic Plot
Terminal Velocity Design Charts
  Heat Shield
  Parachute
Deorbit Delta V vs. Eccentricity
  Graph
  Table
Final Descent Profile (e = 0.10) Spreadsheet
  Graphs
    Descent Altitude vs. Time
    Descent Velocity vs. Time (e = 0.10)
    Descent Velocity Vs. Time (e = 0.27)
Ascent Analysis
  Minimum Insertion Delta V vs. Lift-Off Mass
  At Optimum Launch Angle
    Graph
    Chart
Final Ascent Profile Spreadsheet
  Launch Delta V Due To Constant 1500 N Thrust
  Calculation
Mach Number vs. Altitude for Ascent and Descent
  Graph
  Chart (Includes Dynamic Pressure)
0+1 DE-ORBIT BURN (LIQUID)
2 IMMEDIATELY BEFORE PARACHUTE DEPLOYMENT (VERY NEAR SURFACE)
3 IMMEDIATELY PRECEDING IMPACT
4 GROUND CONDITION
5+6 ASCENT BURN (LIQUID)
7+8 APPOX. CICL BURN (SOLID) AND RANGEOVUS
9 RENDEZVOUS

REAL TRAJECTORIES
WITH ATMOSPHERIC EFFECTS

DISTANCES & ANGLES NOT TO SCALE.
MISSION PROFILE

- DETACH FROM M.O.V. AND GIVE SMALL BURN FOR POSITIONING.
  ~ 8.0 m/s
- SPEND UP TO SEVEN DAYS POSITIONING A M.S.P. FOR DE-ORBIT
  BURN (LIQUID ENGINES) - MODIFY POSITION UP TO 180° TO AVOID
  PLANE CHANGE
- DESCENT AND LANDING (PARACHUTE AND LIQUID ENGINE RETRO-FIRE)
  - MUST BE ON SURFACE AT LEAST ~12 HOURS UNTIL
    ORBITAL PLANE PASSES OVERHEAD.
  - COMPLETE SAMPLING MISSION DURING THIS TIME

- ASCEXT WITH LIQUID ENGINE. (SAME AS USED FOR DE-ORBIT
  AND DESCENT RETRO-FIRE) LEAVE BEHIND SIGNIFICANT FRACTION
  OF VEHICLE MASS.
- APOGEE KICK WITH SOLID TO SEPARATE FROM ASCENT
  MOTORS AND TANKS, AND REACH ORBIT NEAR THE
  CIRCULAR ORBIT OF M.O.V. (AT VELOCITY SLIGHTLY
  GREATER OR SLIGHTLY BELOW THAT OF M.O.V., FOR
  SUBSEQUENT RENDEZVOUS.)
IDEAL ORBIT CALCULATIONS

1. ECCENTRICITY \( e \) TABULATED

2. SEMI-MAJOR AXIS \( a \) (km)

3. APOGEE VELOCITY FOR DEORBIT TRAJECTORY (km/sec)

4. ANGULAR MOMENTUM OF DESCENT ORBIT (km^2/sec)

5. PARAMETER FOR POAR EQU. \( p \) of DESCENT ORBIT (km)

6. THETA (\( \Theta \)) TRUE ANOMALY (degrees)

   
   \[ r = \frac{p}{1 + e \cos \Theta} \]

7. "IMPACT" VELOCITY (km/sec)

   \[ \mu \left[ \frac{2}{R_i} - \frac{1}{a} \right] \]

   \[ \text{where} \quad R_i = 35786 \text{ km} \]

8. LANDING (FLIGHT PATH) ANGLE (degrees)

   \[ h = r \cdot \cos \beta \]

   \[ \beta = \cos^{-1} \left( \frac{h}{\text{impact vel.}} \right) \]
\[
\cos 1 = \cos (28.5^\circ) \cos (33^\circ) \Rightarrow 1 = 42.520^\circ
\]

\[
\sin 2 = \frac{\sin (90^\circ)}{\sin (28.5^\circ)} = \frac{\sin (42.520^\circ)}{\sin (28.5^\circ)} \Rightarrow 2 = 44.912^\circ
\]

\[
(3) = (2) - 30^\circ = 44.912^\circ - 30^\circ \Rightarrow 3 = 14.912^\circ
\]

Solve for plane change angle \(\alpha\)

\[
\frac{\sin \alpha}{\sin (42.520^\circ)} = \frac{\sin (14.912^\circ)}{\sin (28.5^\circ)} \quad \text{but} \quad \sin (28.5^\circ) = \sin \Theta
\]

\[
\sin \alpha = \frac{\sin (14.912^\circ) \sin (42.520^\circ)}{\sin \Theta} = \frac{0.17391}{\sin \Theta}
\]

9. Plane Change Angle (deg)

\[
\alpha = \sin^{-1} \left( \frac{0.17391}{\sin \Theta} \right)
\]

10. De-orbit Delta V. (km/sec)

\[
\Delta V = \left[ \frac{V_{\text{circular}}^2 + V_{\text{apogee}}^2 - 2 V_{\text{circular}} V_{\text{apogee}} \cos \alpha}{}\right]^{1/2}
\]

\[
V_{\text{circular}} = \sqrt{\mu / R} = 3.2714 \text{ km/sec}
\]
**TIME ELAPSED IN DEORBIT:**

- **E** = **ECCENTRIC ANOMALY**, ANGLE MEASURED FROM CENTER OF ELLIPSE FROM PERIAPSE.
- **θ** = **TRUE ANOMALY**, MEASURED FROM PERIAPSE AT MASS/FORCE CENTER.

**GIVEN & FIND TIME FROM DEORBIT AT APOAPSE:**

\[
E = \cos^{-1}\left[ e + \frac{c \cos \theta}{a} \right]
\]

WHERE \( a \) IS THE SEMI-MAJOR AXIS AND \( c \) IS THE DISTANCE FROM THE MASS CENTER.

\[
t_r = \frac{E - e \sin E}{\sqrt{\frac{\mu}{a^3}}}
\]

WHERE \( e \) IS ECCENTRICITY AND \( \mu \) IS THE GRAVITATIONAL PARAMETER FOR MARS.

\[
t_{apoapose} = \pi \sqrt{\frac{a^3}{\mu}}
\]

\[
\Delta t = t_{apoapose} - t_r
\]
TAKE \( r, \nu, \beta \) FROM IDEAL ORBIT INITIALLY

CALCULATE INITIAL \( r, \theta, \nu_r, \nu_\theta \)

FROM THESE CALCULATE \( L, D \)

USE THESE TO GENERATE NEXT \( r, \theta, \nu_r, \nu_\theta, L, D, \) etc.

\( L, D \) DEPEND ON BODY (HEATSHIELD) AND/OR PARACHUTE

Polar Coordinates

\[ \begin{align*}
V_r &= v \sin \beta \\
V_\theta &= v \cos \beta \\
a_r &= g + \frac{L \cos \beta - D \sin \beta}{M} \\
a_\theta &= \frac{L \sin \beta - D \cos \beta}{M}
\end{align*} \]

\[\begin{align*}
\theta_2 &= \theta_1 + \frac{(\nu_\theta_2 + \nu_\theta_1) \Delta t}{2} \\
r_2 &= r_1 + \frac{(\nu_r_2 + \nu_r_1) \Delta t}{2}
\end{align*}\]

\[V_{r_2} = V_{r_1} + a_r \Delta t\]

\[V_{\theta_2} = V_{\theta_1} + a_\theta \Delta t\]

\[V_2 = \sqrt{(V_{r_2})^2 + (V_{\theta_2})^2}\]

\[\beta_2 = \tan^{-1}\left(\frac{V_{r_2}}{V_{\theta_2}}\right)\]

1. \( r_1 \quad \theta_1 \quad \nu_r_1 \quad \nu_\theta_1 \quad \nu_1 \quad \beta_1 \quad L_1 \quad D_1 \quad a_r_1 \quad a_\theta_1\)

2. \( r_2 \quad \theta_2 \quad \nu_r_2 \quad \nu_\theta_2 \quad \nu_2 \quad \beta_2 \quad L_2 \quad D_2 \quad a_r_2 \quad a_\theta_2\)

3. CONTINUE TO "STEP DOWN" UNTIL \( r = r_{\text{SURFACE}}\)
ORBITAL PARAMETERS:

1. \( a = \frac{\text{apogee}}{1+e} \)  \text{SEMI-MAJOR AXIS}

2. \( p = a (1-e^2) \)  \text{ORBITAL PARAMETER}

3. \( h = \sqrt{p\mu} \)  \text{ANGULAR MOMENTUM}

POSITION DEPENDENT BASIC VALUES

1. \( \theta = \cos^{-1} \left( \frac{p-r}{re} \right) \)  \text{TRUE ANOMALY AS FUNCTION OF RADIUS}

2. \( v = \sqrt{\frac{\mu}{\frac{2}{r} - \frac{1}{a}}} \)  \text{VELOCITY AS FUNCTION OF RADIUS}

3. \( \beta = \cos^{-1} \left( \frac{h}{rv} \right) \)  \text{FLIGHT PATH ANGLE AS FUNCTION OF RADIUS (AND VELOCITY WHICH IS A FUNCTION OF RADIUS)}

DERIVED QUANTITIES NEEDED FOR ITERATION

1. \( v_\theta = v \cos \beta \)  \text{TANGENTIAL VELOCITY AS FUNCTION OF \( \beta \) AND VELOCITY}

2. \( v_r = v \sin \beta \)  \text{RADIAL VELOCITY AS FUNCTION OF \( \beta \) AND VELOCITY}

1. \( g = g_0 e^{-\frac{\mu}{r^2}} \)  \text{DENSITY FUNCTION (\( g_0 \) SURF. DENSITY, \( \mu \) SCALE HT.)}

2. \( g = \mu \)  \text{GRAVITATIONAL ACCELERATION AT GIVEN RADIUS}

IDEAL ORBITAL EQUATIONS DURING KEPLERIAN PHASE OF DESCENT
ITERATIVE PROCEDURE

TAKE INITIAL VALUES FROM IDEAL ORBIT

RADIUS $r$  Given $m$ mass
ANGLE $\Theta$  Given $\mu$ Gran. Const.
VELOCITY $v$  Given $r$ mass path
COMPONENTS $v_r, v_\Theta$ VARIABLE $m, A, C_D, K, \Delta t$
ANGLE $\beta$

CALCULATE FORCES, ACCELERATIONS, VELOCITIES, POSITIONS

1. FORCES:
   \[
   s = \mu \frac{r^2}{r^2} \\
   f = s_0 \exp \left[ - \frac{r - r_{\text{new}}}{a} \right] \\
   D = \left[ \frac{1}{2} s v^2 \right] [A, C_D] \\
   L = K D
   \]

2. ACCELERATION
   \[
   a_r = \frac{g}{g} - \left[ \frac{L \cos \beta + D \sin \beta}{m} \right] \\
   a_\Theta = \left[ \frac{L \sin \beta - D \cos \beta}{m} \right]
   \]

3. VELOCITIES
   \[
   v_r^{\text{new}} = v_r^{\text{old}} + a_r \Delta t \\
   v_\Theta^{\text{new}} = v_\Theta^{\text{old}} + a_\Theta \Delta t
   \]

4. POSITIONS
   \[
   \beta = \tan^{-1} \left( \frac{v_r^{\text{new}}}{v_\Theta^{\text{new}}} \right) \\
   r^{\text{new}} = r^{\text{old}} + \frac{1}{2} [v_r^{\text{new}} + v_r^{\text{old}}] \Delta t \\
   \Theta^{\text{new}} = \Theta^{\text{old}} + \left( \frac{v_\Theta^{\text{old}} - v_\Theta^{\text{new}}}{r^{\text{old}} + r^{\text{new}}} \right) \Delta t
   \]

RETURN TO CALCULATE FORCES FOR NEXT STEP AND REPEAT CYCLE UNTIL LANDING.
**FINAL DESCENT PROFILE**

* Separation from MAV: \( V = 3.272 \text{ km/s} \) \( \text{ALT} = 600 \text{ km} \)

* \( \Delta V \approx 8.0 \text{ m/s} \) for positioning (ALT \( \approx 20 \text{ km}, 190^\circ \text{ in 6 days, } 9.73 \text{ hours} \))

2.5\% Orbit Advance/Recede Mass Before Deorbit \( \approx 80 \text{ kg} \)

0.758 kg Fuel Req'd.

* Deorbit \( \Delta V = 167.9 \text{ m/s} \) Transfer Eccentricity \( e = 0.10 \)

<table>
<thead>
<tr>
<th>( \alpha_r )</th>
<th>0.33 g(^e)</th>
<th>( \mathbf{\beta}_{200} )</th>
<th>6.3°</th>
<th>( V )</th>
<th>3421. m/s</th>
<th>ALT. 250 km</th>
<th>Time (min)</th>
<th>29.02</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \alpha_r )</td>
<td>0.34 g(^e)</td>
<td>( \mathbf{\beta}_{100} )</td>
<td>17.6°</td>
<td>( V )</td>
<td>3562. m/s</td>
<td>ALT. 100 km</td>
<td>32.52</td>
<td></td>
</tr>
<tr>
<td>( \alpha_r )</td>
<td>2.01 g(^e)</td>
<td>( \mathbf{\beta}_{30} )</td>
<td>21.0°</td>
<td>( V )</td>
<td>1975. m/s</td>
<td>ALT. 50 km</td>
<td>33.00</td>
<td></td>
</tr>
<tr>
<td>( \alpha_r )</td>
<td>0.25</td>
<td>( \mathbf{\beta}_{10} )</td>
<td>30.5°</td>
<td>( V )</td>
<td>330. m/s</td>
<td>ALT. 10 km</td>
<td>34.35</td>
<td></td>
</tr>
</tbody>
</table>

Based on descent mass of 34.7 kg \( C_D = 1.0 \)

\[ \Delta V_{\text{deorbit}} = V_{\text{circular}} \left[ 1 - \frac{1}{1-e} \right] \]

REDUCTION OF \( \Delta V \) BY 360 m/s SAVES 10 kg

* Deploy parachute at -3.0 km ALT. / Drop heat shield 1.2 km above surface 21 seconds before impact

\[ V = 108.8 \text{ m/s} \]

\[ \alpha_r = 27.5 \text{ m/s}^2 \] \( \sim 3 \text{g}^e \) during first second

\[ \beta = 78.9^\circ \]

REDUCE \( \alpha_r \) to 25 sec

At ALT -3.5 km

\[ V = 56.5 \text{ m/s} \]

\[ \alpha_r = 3.35 \text{ m/s}^2 \] \( \sim 1/3 \text{g}^e \)

\[ \beta = 82.1^\circ \]

BASED ON MASS = 75.7 kg

* Retro fire at < 100 m above surface

THrust \( \sim 21 \text{ m/s}^2 \) RESULT \( \sim 2 \text{g}^e \)

REQUIRES 53.5 m TRAVEL TO BRING \( V \) TO ZERO.

FIRES FOR \( \sim 2.5 \) sec.

IMPACT 30.1 minutes after deorbit

* Remain on surface \( \sim 12 \) hours to perform sampling mission (12 hrs, 19 min)
Mars Atmospheric Density (kg/m^3) Based on 1982 Standard Atmosphere
Mars Atmospheric Density (kg/m^3) Based on 1982 Standard Atmosphere
<table>
<thead>
<tr>
<th>Weight (lb)</th>
<th>Terminal Velocity for Descent with Heat Shield</th>
</tr>
</thead>
<tbody>
<tr>
<td>15.95</td>
<td>1.00</td>
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<tr>
<td>Ce</td>
<td>44.70</td>
</tr>
<tr>
<td>Mass (kg)</td>
<td>R(m)</td>
</tr>
<tr>
<td>1.250</td>
<td>0.075</td>
</tr>
<tr>
<td>1.252</td>
<td>0.077</td>
</tr>
<tr>
<td>1.254</td>
<td>0.079</td>
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<tr>
<td>1.256</td>
<td>0.081</td>
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<tr>
<td>1.258</td>
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<tr>
<td>1.260</td>
<td>0.084</td>
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<tr>
<td>Alt Density</td>
<td>Speed of Sound (m/s)</td>
</tr>
<tr>
<td>3.000</td>
<td>3.003</td>
</tr>
<tr>
<td>2.000</td>
<td>2.003</td>
</tr>
<tr>
<td>1.000</td>
<td>1.003</td>
</tr>
<tr>
<td>0.000</td>
<td>0.003</td>
</tr>
</tbody>
</table>

**Heat Shield Terminal Velocity Design Chart**

| Cm | 31.55 |

**VTECH_RS.XLS Terminal Velocity Chart**

<table>
<thead>
<tr>
<th>Velocity (m/s)</th>
<th>Terminal Velocity (m/s)</th>
<th>Terminal Velocity (m/s)</th>
<th>Terminal Velocity (m/s)</th>
<th>Terminal Velocity (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.750</td>
<td>1.750</td>
<td>1.750</td>
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<td>1.752</td>
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<tr>
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<td>1.754</td>
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<tr>
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<td>1.756</td>
<td>1.756</td>
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<tr>
<td>1.758</td>
<td>1.758</td>
<td>1.758</td>
<td>1.758</td>
<td>1.758</td>
</tr>
</tbody>
</table>

**THERMAL PROTECTION SUIT**

- **Heat Shield**
  - **Material:**
  - **Thickness:**
- **Protection:**
  - **Rating:**

**THERMAL PROTECTION SUIT**

- **Heat Shield**
  - **Material:**
  - **Thickness:**
- **Protection:**
  - **Rating:**

**THERMAL PROTECTION SUIT**

- **Heat Shield**
  - **Material:**
  - **Thickness:**
- **Protection:**
  - **Rating:**

**THERMAL PROTECTION SUIT**

- **Heat Shield**
  - **Material:**
  - **Thickness:**
- **Protection:**
  - **Rating:**

**THERMAL PROTECTION SUIT**

- **Heat Shield**
  - **Material:**
  - **Thickness:**
- **Protection:**
  - **Rating:**
## Parachute Terminal Velocity Design Chart

<table>
<thead>
<tr>
<th>Density (kg/m³)</th>
<th>Speed of Sound (m/s)</th>
<th>Terminal Velocity (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>346</td>
<td>310</td>
</tr>
<tr>
<td>200</td>
<td>367</td>
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<td>400</td>
<td>409</td>
<td>388</td>
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<tr>
<td>500</td>
<td>430</td>
<td>409</td>
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<tr>
<td>600</td>
<td>451</td>
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<tr>
<td>700</td>
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<tr>
<td>1900</td>
<td>724</td>
<td>703</td>
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</table>

**Weight (kg):** 283.36

**Terminal Velocity for Descent with Parachute**

<table>
<thead>
<tr>
<th>Weight (kg)</th>
<th>Terminal Velocity (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>283.36</td>
<td>703.68</td>
</tr>
</tbody>
</table>

**Chart Notes:**

- Density (kg/m³) refers to the density of the material used in the parachute.
- Speed of Sound (m/s) is the speed at which sound travels through the material.
- Terminal Velocity (m/s) is the velocity at which the parachute will descend.

---

**FURTHER INFORMATION:**

- Detailed specifications and calculations for the parachute design can be found in the project documentation.
- Additional resources and references are available for further study.

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**CONTACT:**

For more information, please contact the project lead at info@projectteam.com.
Deorbit Velocity Change for Mars Descent

Deorbit Del V (km/sec) vs. Eccentricity

Delta V

Eccentricity

5/29/90 5:40 PM
DELTA V DEORBIT VS ECCENTRICITY

<table>
<thead>
<tr>
<th>e</th>
<th>Del V (km/sec)</th>
<th>e</th>
<th>Del V (km/sec)</th>
<th>e</th>
<th>Del V (km/sec)</th>
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**FINAL DESCENT PROFILE**

ODEB010.42 Incremental Descent in Actual Atmosphere (Eccentricity 0.10)

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**Note:**

- vt: Vegetation Tract
- vt new: Vegetation Tract New
Altitude vs. Time for Descent with Eccentricity
0.10 Atmospheric Phase

Note steep descent levels off at about 30 km altitude where the maximum dynamic pressure and deceleration occur.
DESCRIPTIVE TITLE

DESCRIPTIVE SUB-TITLE

TEXT FOR GRAPH

VELOCITY (m/s)

ALTITUDE (km)

DATA POINTS:

- [x1, y1]
- [x2, y2]
- [x3, y3]
- [x4, y4]
- [x5, y5]

NOTES:

- Description of the data represented in the graph.
- Any significant observations or interpretations.

LEGEND:

- Legend entries for any symbols or data representations.

REFERENCES:

- Relevant sources or additional information.
Descent with Eccentricity = 0.27
Intermediate similar to final case
Minimum Insertion Delta V vs. Lift-Off Mass at Optimum Launch Angle

Launch Angle Beta of 62.5 Degrees

Apogee Stage Mass at Lift-Off (kg)
**DELVOPTM.XLS**

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6/2/90 7:09 PM Page 3
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Interpolate to find time at end of burn

Use average of 82 and 83 second accelerations

0.508 100.804 53.442

0.530 4020.324

Delta V total at Mt. = 46.9 (in m/s)

Delta V for Ascent, Ignoring Atmospheric Drag Forces
Mach Number vs. Altitude

19.3 Max. During Descent

16.8 Max. During Ascent

Altitude (km)
### MACH & Q.xls Descent Mach Numbers and Dynamic Pressures

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### Descent Mach Numbers and Dynamic Pressures

- Alt: Altitude
- Density: Density
- Speed of Sound (m/s): Sound speed
- V (m/s): Velocity
- Mach: Mach number
- q (N/m^2): Dynamic pressure
- V (m/s): Velocity
- Mach: Mach number
- q (N/m^2): Dynamic pressure
Section E:
Specialized Structures

Tim Gibson
**Parachute**

During the descent through the martian atmosphere, the lander will employ a parachute to deaccelerate to a velocity at which the liquid engine can prepare the lander for touchdown. The size of the parachute, measured by its reference area, the area of a circle whose diameter is equal to the distance across the fully deployed parachute, must be large enough to create enough drag so that the required final or "terminal" velocity will be attained, without too much deacceleration during the opening "jerk". It can be estimated from Air Force research (ref. 3) that the peak deacceleration during opening jerk is slightly more than 3 times that of the deacceleration calculated from a force-balance relationship. The lander is limited to a maximum acceleration of 10 earth g's. This limits maximum force-balance calculated deacceleration to under 3.33 g's.

The force-balance relationship is:
which yields: \( D - mg = ma \)

where:
- drag, \( D = C_D \frac{1}{2} \rho v^2 S \)
- drag coefficient, \( C_D \)
- atmosphere density, \( \rho \)
- velocity, \( V \)
- reference area, \( S \)
- mass of the lander, \( m \)
- deacceleration, \( a \)

so, we have: \( C_D \frac{1}{2} \rho v^2 S - mg = ma \)

From this, by setting \( a = 0 \), we can obtain the reference area required for a desired terminal velocity:
\[
S = \frac{2 m g}{C_D \rho V^2}
\]

or, the maximum force-balance deacceleration:
\[
a = \frac{( CD \frac{1}{2} \rho v^2 S - mg )}{m}
\]

where \( V \) is the deployment velocity.

By starting at \( V = \) deployment velocity, and recalculating \( V \) every time interval \( \Delta t \) by numerically integrating the acceleration (see Appendix E.q.), the maximum descent distance and time to reach terminal velocity can be determined. This method yields a maximum because the actual acceleration rate is, as mentioned earlier, higher than the force-balance method predicts. However, the maximum is desirable for determining the minimum altitude at which the parachute can be deployed safely. Low altitude parachute deployment is desirable to minimize the effect of cross winds.

Initially, the lander was to use a 2-stage parachute. The small first stage, 3.33 m diameter, limited peak deacceleration to 3.14 g's at a deployment velocity of 280.00 m/sec at 1 km altitude (fig. E). The larger second stage, 9.9 m in diameter, limited peak deacceleration to 2.69 g's at a deployment velocity of 88.00 m/sec (fig. E) and attained a terminal velocity of less than 30.00 m/sec.
in less than 75 m. This configuration could be deployed at less than 1 km above ground level and maintain a 3 times maximum g's value under our 10g mission limit.

The parachute is an extended skirt canopy type (fig. 3) which has a higher drag than flat circular or cross-form types but has a longer opening time and correspondingly has lower peak opening forces. The canopy is fabricated of 2.60 ounce per square yard (89 g/m²) dacron cloth, per MIL-C-7350, type 1. The suspension lines are tubular braided dacron with an ultimate tensile strength of 2740N. The approximate weight of the canopy can be calculated from the surface area of 1/2 of a sphere:

\[ A_{\text{surface}} = 2 \pi r^2 \]

and a required storage volume (chute pack) based on a maximum packing density of \( \rho = 600 \text{ kg/m}^3 \).

So, the mass of the chute canopy, \( m_{\text{canopy}} = 89 A_{\text{surface}} = 89 (2 \pi r^2) \),

and the volume of the chute pack, \( V_{\text{pack}} = (89) (2 \pi r^2) / 600 \times 10^3 \).

This mass and volume dependence on the radius squared led to an optimization where the weight and size of the parachute were compared to the weight of added liquid fuel to accomplish the same deacceleration. It was determined that decreasing the parachute from 9.9 meters in diameter (\( m_{\text{canopy}} = 13.7 \text{ kg} \)) to 4 meters in diameter (\( m_{\text{canopy}} = 0.75 \text{ kg} \)), while adding \( 4 \text{ kg} \) of fuel, would decrease mass by \( \approx 8.9 \text{ kg} \). Also, the volume required for the chute pack went from \( 0.0228 \text{ m}^3 \) to \( 0.00125 \text{ m}^3 \).

Using the 4 m diameter parachute, an analysis was done (fig. 4) for the earliest possible deployment, 5 km above ground level, with the highest deployment velocity possible, \( V = 248.54 \text{ m/s} \). Maximum force-balance deacceleration was 2.47 g's, with a distance to terminal velocity of \( \approx 0.5 \text{ km} \). Next, an analysis was done (fig. 5) for a late deployment, -3 km altitude, with a corresponding terminal velocity of 187.4 \text{ m/s}. Maximum force-
balance deacceleration was 2.48 g's, with a distance to terminal velocity of ≈ 325 m.

FIG. E3

Description
EXTENDED SKIRT CANOPY
CONSTRUCTION SCHEMATIC
INFLATED PROFILE
program mars
  implicit none
  real*4 cd, rho, s, g, m, v, uterm, t, h, a, vnew, accel
  real*4 deltat
  integer*4 i
  cd = 1.0
  deltat = 0.1
  rho = 0.0127
  s = 12.5664
  g = 3.73
  m = 100.0
  v = 187.4
  uterm = sqrt((2.0*m*g)/(cd*rho*s))
  write(6,50) uterm
  format(1x,'v term = ', f7.3,'m/sec','/','f7.3','V (m/sec)',
        ' 5x','h (meters)','5x','accel (g s)','
        ' 5x','time (sec)')
  t = 0.0
  h = 0.0
  do 100 i = 1, 200
    t = t + deltat
    a = (0.5*cd*rho+(v**2)*g - m*g)/m
    h = h + v*deltat - 0.5*a*(deltat**2)
    vnew = sqrt((v**2) - 2.0*a*h)
    accel = a/9.81
    write(6,60) v, h, accel, t
  format(1x,f7.3,7x,f7.2,10x,f7.5,8x,f5.1)
  if ( v .le. (1.00001*uterm)) go to 200
  v = vnew
  enddo
  200 continue
end

CONTROL DISK IS OF POOR QUALITY.
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<th>( h ) (meters)</th>
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The structure of the heat shield serves to hold the insulation material in position, and serve as an aero-braking device. The 1.5 meter diameter heat shield is designed to withstand a maximum total aero-force of 6.9 kN which occurs at an altitude of 31 km and a corresponding free-stream velocity of 1985 m/sec. The structure in contact with the insulation encounters a maximum temperature of 700 °F for approximately 30 seconds.

Due to its high stiffness-to-weight ratio and resistance to thermal effects, a beryllium honeycomb bonded with high-temperature thermoplastic adhesive was chosen for the heat shield structure. Four beryllium tubular braces serve to brace the structure (see fig. E6).
In order to conservatively size the structural components, simplified models (see fig. E7) were chosen which would provide an adequate safety margin and facilitate analysis.

**FIG. E7**

**CASE 1**

**CASE 2**

where: unsupported length, L  
load per unit length, P  
depth of honeycomb, h  
thickness of face sheet, t  
width of beam, w

From strength of materials the stress is: stress = M\(y\)/I

now, \(y = \frac{1}{2}d + t\), but since \(t \ll \frac{1}{2}h\), \(y = \frac{1}{2}d\)

and, by the parallel axis theorem \(I = \frac{2}{3}wt^3 + 2wt(\frac{1}{2}d)^2\)

since \(\frac{2}{3}wt^3 \ll 2wt(\frac{1}{2}d)^2\) we have \(I = 2wt(\frac{1}{2}d)^2 = \frac{1}{2}wtd^2\)
so, stress = \( M^{(1/2)} \frac{h}{(1/2)w} t^2 = \frac{M}{wtd} \)

or, letting \( M = \) moment per unit width, stress = \( \frac{M}{td} \)

Also from strength of materials we know the maximum moment, \( M \), of our model beams (fig. E7) for case 1 is equal to

\[ M \text{ (max.)} = \frac{PL^2}{8}, \]

and for case 2

\[ M \text{ (max)} = \frac{PL^2}{8}. \]

Using a maximum unsupported length of \( L = 1 \) m for case 1 and \( L = 0.5 \) m for case 2, we have:

\[ M \text{ (max)} = \frac{P}{8} \]

Now stress\(_{crit}\) for beryllium = stress\(_{yield}\) = \( 3.24 \times 10^8 \) N/m and the distributed load,

\[ P = \frac{\text{Aero-force}}{\text{area}} = \frac{6.9 \text{ Kn}}{\pi (0.75 \text{ m})^2 (1 \text{ m})} = 3905 \text{ N/m} \]

Rearranging, we have: \( t d = \frac{P}{8(\text{stress}_y)} = \frac{3905}{8(3.24 \times 10^8)} \)

\[ = 1.507 \times 10^{-6} \]

Taking the minimum thickness for the beryllium face sheets to be \( t = 0.010'' = 0.000254 \) m, \( \Rightarrow d = 1.507 \times 10^{-6}/0.00254 \)

\[ = 0.00593 \text{ m} = 0.23 \text{ in} \]

so, we will specify a beryllium honeycomb with a core depth, \( d = 0.25'' = 0.00635 \) m. Checking, we have
stress_{crit} = \frac{3905}{8}(0.00635)(0.000254) = 3.03 \times 10^8 \text{N/m}^2

which is over 6% under stress_y.

Now, in order to determine the mass of the structure the density of the honeycomb core is needed. A "rule of thumb" for estimating honeycomb core density is:

\[ \rho_{core} = \frac{2\rho_{Be}V_{max}}{\text{stress}_{Be \text{ shear}}} \]

where:
- density of Be, \( \rho_{Be} = 1826 \text{ kg/m}^3 \)
- shear stress of Be, \( \text{stress}_{s} = 0.6 \text{ stress}_y \)

and the maximum shear stress, again from strength of materials, is

\[ V_{max} = \frac{1}{2}P = 1.952.5 \text{ N} \]

so,

\[ P_{core} = \frac{2(1826)(1952.5)}{0.6(3.24 \times 10^8)} = 0.0376 \text{ kg/m}^3 \]

The lightest core material available is approximately 1.5 lbs/ft or 24 kg/m. Apparently the shear force is too low in this application for the "rule of thumb" approximation to be accurate. It does indicate though that it is safe to use the lightweight core. So, the mass of the structure is:

\[
\text{Mass} = 2\rho_{Be}(\pi r^2 t) + \rho_{core}(\pi r^2 d)
\]

\[
= \pi r^2 (2\rho_{Be} t + \rho_{core} d)
\]

\[
= \pi(0.75)^2[2(1826)(0.000254) + 24(0.00635)] = 1.91 \text{ kg}
\]

The model for the tubular braces is simple compression (see fig \( \subseteq \subseteq \)). The force in each tube may be as high as

\[ F = (1.5)(\text{max. aero-force/\# of tubes}) \]
where:  
max. aero-force = 6.9 kN  
# of tubes = 4

so  
\[ F = (1.5) \frac{(6900)}{4} = 2588 \text{ N} \]

where:  
diameter, \( d \)  
radius, \( r \)  
length, \( L \)  
wall thickness, \( t \)  
force, \( F \)

For a thin wall tube under compression there are three design constraints:

1) stress limit: \( \text{stress} = \frac{F}{\pi d t} \)

2) Euler buckling: \( F = \frac{\pi^2 E I}{L^2} \)

3) crippling: \( \text{stress} = 0.3 \frac{E I}{r} \)
For a first iteration, the stress limit and Euler buckling are solved together in order to find the optimum tube dimensions, where

\[ I = \pi r^3 t \]

for a thin-wall tube.

Now, solving 1) and 2) for the force, \( F \), yields

\[ \text{(stress)}2\pi rt = \pi^2 E \pi r^3 t/\ell^2 \]

so

\[ r = \frac{2(\text{stress})L^2}{\pi^2 E} = \frac{2(0.652)^2(3.24 \times 10^8)}{\pi^2(2.896 \times 10^{11})} \]

\[ = 0.009817 \text{ m}. \]

This is a tube with a diameter of 0.0196 m or 0.773". Solving for the wall thickness, \( t = 0.0001313 \text{ m} = 0.0052 \text{ in.} \) In order to specify a tube which might be commonly available, we will use 0.75 in. (0.01875 m) diameter and 0.010 in. (0.000254 m) wall thickness beryllium tubing. Checking, we have:

1) stress limit, \( F = (\text{stress})2\pi rt = 4848 \text{ N} \)

2) Euler buckling, \( F = \pi^2 EI/L^2 = 4421 \text{ N} \)

3) crippling, \( \text{stress} = 0.3Et/r = 23.5 \times 10^8 \)

which are all considerably over any anticipated forces. Now, the mass of each tube is, \( \text{mass/tube} = \pi dt\ell \delta_{Be} \)

\[ = \pi(.01875)(.000254)(.652)(1826) = .0178 \text{ kg} \]

So, the total for four tubes is \( 4(0.0178) = .0712 \text{ kg} \)
Therefore, the mass of the entire structure is: $1.91 + .0712 = 1.98$ kg
Soil Sample Collection

The mission requirement is to bring 1 kg of Martian soil back to the orbiting MOV (Mars Orbiting Vehicle). This will be accomplished with the use of an extendable surface sampler, which consists of a collector head attached to the end of a retractable boom. The boom is constructed from two ribbons of beryllium welded together along the edges. When extended, the two layers opened to form a rigid tube. When retracted, the boom flattens onto a storage spool. A flat cable sandwiched between the boom layers transmits electrical power to the collector head.

The collector head is a scoop with a movable lid. To fill the scoop, the boom is extended along or into the surface. Once full, the lid closes. The boom is then elevated such that the scoop can be dumped into a container at the top of the lander (fig. E/F). The nose cap at the top of the lander's aero-shell (fig. E/A) is moved upward by a threaded driveshaft and, when free of the aero-shell, is then rotated out the way for dumping into the sample container.

Design considerations include tube buckling and bending stresses, and torque for pushing the scoop into the Martian soil, lifting the sample into the container, and opening and closing the container.

Data, Martian soil:

density, \( \varrho = 0.7 \) to \( 3.2 \, \text{g/cm}^3 \),
maximum penetration resistance, \( \text{PR} = 6 \, \text{N/cm}^2 \)

Scoop

Volume required for 1 kg sample, \( \varrho = 0.7 \, \text{g/cm}^3 \)

\[
\frac{1}{700} = 0.00143 \, \text{m}^3 = 1430 \, \text{cm}^3
\]
where: $s =$ scoop side, $L =$ scoop length, $t =$ material thickness.

Since the collector is being driven into the soil, titanium is used as a material for the scoop due to its toughness. The worm-gear motor should have enough torque to close the lid against maximum penetration resistance. Now, the force needed to drive the scoop into the soil,
FIG. E9

SAMPLER-ARM OPERATION

LANDER BODY

DUMP POSITION

1.5m MAX.
FIG. E10

CONTAINER COVER

AEROSHELL

DRIVESHAFT

MOTOR

CONTAINER
\[ F = PR \text{ (length)} \text{ (scoop cross sect. area)} \]

where: scoop area = 4 st, and length, \( L = 16.5 \text{ cm} \), \( s = 10.16 \text{ cm} \), and \( t = 0.010 \text{ in} = 0.0254 \text{ cm} \),

so, \[ F = (PR) \text{ (L)} [4 \text{ (s)} \text{ (t)}] = 6 \text{ (16.51)} [4 \text{ (10.16)} \text{ (0.0254)}] \]
\[ = 102.26 \text{ N} \]

A force of \( F = 200 \text{ N} \) will be used in the design of the sampler apparatus to insure an adequate safety margin.

**Mast**

The dimensions of the mast (boom) will be governed by maximum compressive stress, Euler buckling, and bending. The bending requirement will be to lift the scoop, sample, and mast into the "dump position."

**Maximum length of mast** = 1.5m

The wall thickness of the beryllium mast is taken to be 0.0508mm, which is common for the welded seam type masts.

\[ t = 0.0508 \text{ mm} = 5.08 \times 10^{-5} \text{ m} \]
\[ = 0.002 " \]

In order to find the minimum diameter, maximum compressive stress is considered:
\[
\text{stress}_y = \frac{F}{A} \quad \Rightarrow \quad F = \text{stress}_y \pi t = \text{stress}_y \pi \text{t} = 200/(3.24 \times 10^8) \pi (5.08 \times 10^{-5})
\]

\[
\Rightarrow \quad d = \frac{F}{\text{stress}_y \pi t} = \frac{200}{(3.24 \times 10^8) \pi (5.08 \times 10^{-5})} = 0.00387 \text{ m}
\]

so,

\[
d = 0.387 \text{ cm}
\]

Now consider Euler Buckling: \( F = \frac{\pi^2 EI}{L^2} = \frac{\pi^2 E \pi^3 t}{L^2} \)

\[
\Rightarrow \quad r = \left( \frac{F L^2}{\pi^3 E t} \right)^{1/3} = \left( \frac{200 (1.5)^2}{\pi^3 (2.896 \times 10^{-6}) (5.08 \times 10^{-5})} \right)^{1/3} = 0.00995 \text{ m}
\]

\[
d = 0.01991 \text{ m} = 1.991 \text{ cm} = 0.784 "
\]

and finally bending, from strength of materials: \( \text{stress} = \frac{M y}{I} \),

\[
\text{Mass of Be mast of length 1.5m.}
\]

\[
m_{\text{mast}} = \pi d L t_{\text{Be}} = \pi d (1.5)(5.08 \times 10^{-5})(1826) = 0.437 d
\]

Now, the moment, \( M = m_{\text{mast}} \left( \frac{1}{2} L \right) + m_{\text{sample}} L + m_{\text{scoop}} L \).
\[ m_{\text{scoop}} = 4(.1016)(.000254)(.1651) = 4621 \, \text{kg/m}^3 + m_{\text{worm gear motor}} \]

\[ m_{\text{worm gear motor}} = .45 \, \text{kg} \]

so

\[ m_{\text{scoop}} = 4(.1016)(.000254)(.1651)(4621) + .45 \, \text{kg} \]

\[ = .0788 \, \text{kg} + .45 \, \text{kg} \]

so

\[ M = .437d(.75) + (1.5) + .5288(1.5) = .3277d + 2.2932 \]

and

\[ l = \pi r^3 t = (\pi/8)d^3 t \]

so

\[ \text{stress}_y = My/I = (.3277d + 2.2932)^4/(\pi d^2(5.08 \times 10^{-5})) \]

\[ = (1.31d + 9.1728)/(1.596 \times 10^{-4}d^2) \]

but \[ \text{stress}_y = 3.24 \times 10^8 \], so, rearranging, we have

\[ 5.171 \times 10^4d^2 - 1.31d - 9.1728 = 0 \]

or

\[ d^2 - 2.533 \times 10^{-5}d - 1.774 \times 10^{-4} = 0 \]

solving for \( d \):

\[
\begin{array}{c|c}
  d  &  F(d) \\
  .01991  &  .0002185 \\
  .05  &  .00232 \\
  .01  &  -.000077653 \\
  .015  &  .00004722 \\
\end{array}
\]

so, for bending \( 0.010 \, \text{m} < d < .015 \, \text{m} \). The Euler buckling in compression is the limiting case, therefore the diameter is chosen to be \( d = .01991 \, \text{m} \).

Finally, an analysis of crippling failure:
\[ F_{\text{crip}} = 0.3 \pi \left(2.896 \times 10^4\right)^2 \left(5.08 \times 10^{-5}\right)^2 = 1408 \text{ N} \]

The apparatus is designed for a maximum of 200N, so there is no problem here.

**Reel Assembly**
Worst case torque required would occur while forcing scoop into soil, so
\[ T = Fr \] where \( F = 200 \text{ N} \)
\[ r = 7.25 \text{ cm} = 0.0725 \text{ m} \]
\[ T = 200(0.0725) = 14.5 \text{ Nm} \]
\[ = 128.3 \text{ in lbs} \]

A Hurst Instrument Motor #2602-001 has a mass of 0.68 kg and delivers 250 in lbs of torque at 1 RPM, which corresponds to a scoop speed of \( 1.5 \text{ cm/s} \). The maximum power requirement of each motor, \( P = 54 \text{ watts} \), so the electrical power requirements per sample collection are:

- Time for unreeling at \( 1.5 \text{ cm/s} \) = 100 sec, \( 2 \times 100 = 200 \text{ sec} \), motors running = 3, \( => 32400 \text{ watt-sec} \).
- Time to rotate through boom swing = 30 sec, \( 2 \times 30 = 60 \text{ sec} \), motors running = 3, \( => 9720 \text{ watt-sec} \).
- Time to rotate to dump position at 1 RPM = 15 sec, \( 2 \times 15 = 30 \text{ sec} \), motors running = 4, \( => 6480 \text{ watt-sec} \).
- Time to dump = 15 sec, motors running = 4, \( => 3240 \text{ watt-sec} \).

So, the total energy required per try = \( 51840 \text{ watt-sec} \).
The lander will have 300 watts available for 10 minutes (600 sec) = \( 180000 \text{ watt-sec} \). Therefore there will easily be enough energy for at least 3 complete attempts.
References

Parachute

1) Eckstrom, C. V. and J. S. Preisser. "Flight Test of a 30-Foot Nominal Diameter Disk-Gap-Band Parachute Deployed at a Mach Number of 1.56 and a Dynamic Pressure of 11.4 Pounds Per Square Foot." TM X-1451, Sept. 1967, NASA.


Heat Shield


9) Preliminary minimum sizing "rule of thumb" for honeycomb core density for shear strength at Hughes Aircraft.


Soil Sample Collection


16) N84-35248, 545pp, Mars exploration, Mars data; Mariner, Voyager, Viking spacecraft. NASA Reference Publications from UCLA Engineering Library.

CONCLUSIONS

We believe our conceptual design to represent the best version of the three being presented this quarter. This design has by far the lightest weight and shows much evidence of the optimization procedures we used to improve our designs while the competing designs were busy attempting to incorporate some of the best features of our earlier models. Although our design is by no means perfect and optimized in every possible way, it certainly represents a very solid foundation upon which a final design may eventually be built. Our team has enjoyed this project very much and we all feel we have learned a great deal while working on it and we look forward to working on similar challenging projects in the world outside of UCLA.