Final Design Review Report
for a Mars/Phobos Transportation System

StarTruk Company

The University of Texas at Austin
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Final Design Review Report
for a Mars/Phobos Transportation System

submitted to

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Special thanks to our parents for making us possible.
Abstract

A transportation system will be necessary to support construction and operation of bases on Phobos and Mars beginning in the year 2020 or later. The Star Truk Company presents an approach to defining a network of vehicles and to specifying the types of vehicles which may be used in the system. The network will provide a convenient, integrated means for transporting robotically constructed bases to Phobos and Mars. All the technology needed for the current plan is expected to be available for use at the projected date of cargo departure from the Earth system. The modular design of the transportation system provides easily implemented contingency plans, so that difficulties with any one vehicle will have a minimal effect on the progress of the total mission.

The transportation network proposed by the Star Truk Company consists of orbital vehicles and atmospheric entry vehicles. Initially, only orbital vehicles will participate in the robotic construction phase of the Phobos base. The Interplanetary Transfer Vehicle (ITV) will carry the base and construction equipment to Phobos where the Orbital Maneuvering Vehicles (OMV's) will unload the cargo. In addition, the OMV's will participate in the initial construction of the base. When the Mars base is ready to be sent, one or more ITV's will be used to transport the atmospheric entry vehicles from Earth. These atmospheric vehicles are the One Way Landers (OWL's) and the Ascent/Descent Vehicles (ADV's). They will be used to carry the base components and/or construction equipment.
The OMV's and the Orbital Transfer Vehicles (OTV's) will assist in carrying the atmospheric entry vehicles to low Martian orbit where the OWL's or ADV's will descent to the planet surface. The ADV's were proposed to accommodate expansion of the system. Additionally, a smaller version of the ADV class is capable of transporting personnel between Mars and Phobos.
Table of Contents

Abstract................................................................................................................................. i
Table of Contents.................................................................................................................. i ii
List of Figures and Tables.................................................................................................... vi
List of Acronyms.................................................................................................................. viii

1 General Summary ............................................................................................................. 1
  1.1 Project Background ................................................................................................. 1
  1.2 Design Objectives ................................................................................................. 3
  1.3 Assumptions .......................................................................................................... 3
  1.4 Design Philosophy ................................................................................................. 4
  1.5 Final Product ......................................................................................................... 5

2 Technical Section ............................................................................................................. 6
  2.1 Vehicle Selection Process ...................................................................................... 6
  2.2 Orbital Vehicles ...................................................................................................... 7
    2.2.1 Orbital Vehicle Scenarios ............................................................................... 7
      2.2.1.1 Upside Scenario ..................................................................................... 9
      2.2.1.2 Downside Scenario .............................................................................. 11
    2.2.2 Orbital Vehicle Requirements ........................................................................ 11
    2.2.3 Orbital Vehicle Descriptions ........................................................................ 12
      2.2.3.1 Interplanetary Transfer Vehicle (ITV) .................................................. 13
      2.2.3.2 Orbital Maneuvering Vehicle (OMV) ................................................. 13
      2.2.3.3 Orbital Transfer Vehicle (OTV) ......................................................... 17
      2.2.3.4 Transportation Node (MarsPort) ....................................................... 17
    2.2.4 Orbital Vehicle Subsystems .............................................................................. 17
      2.2.4.1 Computer Systems and Communications ........................................... 19
      2.2.4.2 Radiation Protection ............................................................................ 20
      2.2.4.3 Power System ....................................................................................... 20
      2.2.4.4 Thermal Control System ...................................................................... 20
  2.3 Atmospheric Entry Vehicles ...................................................................................... 21
    2.3.1 Atmospheric Vehicle Requirements ................................................................ 21
    2.3.2 One Way Lander (OWL-200) ....................................................................... 22
      2.3.2.1 Purpose, Requirements, and Constraints ........................................... 22
      2.3.2.2 Physical Descriptions .......................................................................... 24
      2.3.2.3 Subsystem Requirements ................................................................... 24
    2.3.3 Ascent/Descent Vehicle-50 (ADV-50) .......................................................... 28
2.3.3.1 Purpose, Requirements and Constraints ................................... 29
2.3.3.2 Physical Descriptions .................................................................... 29
2.3.3.3 Subsystems Requirements ............................................................ 31
2.3.4 Ascent/Descent Vehicle-10 (ADV-10) ............................................. 34
2.3.4.1 Purpose, Requirements and Constraints ........................................ 34
2.3.4.2 Physical Descriptions ..................................................................... 34
2.3.4.3 Subsystem Requirements .............................................................. 35

2.4 Orbital Operations .............................................................................. 36
2.4.1 ITV Orbit Insertion and Satellite Deployment ..................................... 37
  2.4.1.1 Phobos Mission ........................................................................ 37
  2.4.1.2 Mars Mission ........................................................................... 39
2.4.2 Docking and Cargo Removal ............................................................. 41
  2.4.2.1 Design Requirements ................................................................. 43
  2.4.2.2 Design Description and Docking Procedure .................................. 43
2.4.3 Payload Management ........................................................................ 45
  2.4.3.1 Phobos Base Delivery ................................................................ 45
  2.4.3.2 Mars Base Delivery ..................................................................... 47
    2.4.3.2.1 Orbital Vehicles Mission Profile ........................................ 47
    2.4.3.2.2 OWL Mission Profile ............................................................ 48
    2.4.3.2.3 ADV-50 Mission Profile ......................................................... 49
    2.4.3.2.4 ADV-10 Mission Profile ......................................................... 50
2.4.4 Service Facility Requirements .......................................................... 51
  2.4.4.1 Methods and Equipment ............................................................. 51
  2.4.4.2 Propellant Transportation and Storage ........................................ 52
  2.4.4.3 Refueling Facilities ..................................................................... 52
  2.4.4.4 Facilities at Phobos ..................................................................... 53
  2.4.4.5 Facilities at MarsPort ................................................................. 54
  2.4.4.6 Facilities at Mars Base ................................................................. 55
2.4.5 Operation Control Center ................................................................. 55

2.5 System Evolution .................................................................................. 55
2.6 Recommendations ................................................................................ 58

3. Management Report .............................................................................. 61
  3.1 Personnel and Responsibilities ........................................................... 61
    3.1.1 Individual Responsibilities ............................................................... 61
    3.1.2 Team Responsibilities ................................................................... 63
  3.2 Program Schedule ............................................................................... 64
  3.3 Management Status ........................................................................... 64
3.4 Recommendations ........................................................................... 67
4. Cost Status .................................................................................. 69
5. Bibliography ................................................................................ 70
Appendices ....................................................................................... 72
Appendix A: Vehicle Study ................................................................. 73
Appendix B: Orbital Analysis .............................................................. 82
Appendix C: Orbital Vehicle Sizing Procedure .................................. 88
Appendix D: Atmospheric Vehicle Sizing Procedure ......................... 94
Appendix E: Atmospheric Vehicles Numerical Sizing ......................... 99
Appendix F: Landing Gear Stress Analysis ......................................... 104
Appendix G: INSERT TK! Solver Model ........................................... 109
Appendix H: DELV TK! Solver Model ................................................ 113
Appendix I: OVSIZE TK! Solver Model ............................................. 117
Appendix J: Computation of ΔV's for the OMV ................................. 121
Appendix K: Description of Descent Profile ...................................... 134
Appendix L: DODV TK! Solver Model ............................................... 136
Appendix M: Mars Descent FORTRAN Program ............................... 139
Appendix N: Landing Gear Stress FORTRAN Program ...................... 167
Appendix O: CW Equation TK! Solver Model .................................... 179
Appendix P: Mars Atmospheric Model .............................................. 184
List of Figures and Tables

Figure 1. Kepler Interplanetary Transfer Vehicle ........................................ 14
Figure 2. Orbital Maneuvering Vehicle .................................................... 15
Figure 3. Orbital Transfer Vehicle ......................................................... 18
Figure 4. OWL-200 .................................................................................. 23
Figure 5. ADV-50 ....................................................................................... 30
Figure 6. Mission Scenario for Phobos Base Delivery .............................. 38
Figure 7. Mission Scenario for Mars Base Delivery .................................. 40
Figure 8. Final Configuration of Communication Satellites ...................... 42
Figure 9. Docking Mechanism ................................................................... 44
Figure 10. ADV Servicing Facility ............................................................. 56
Figure 11. Organization Structure ............................................................. 62
Figure 12. Critical Path Chart .................................................................... 65
Figure 13. Program Schedule ..................................................................... 66
Figure A1. University of Michigan Proposed Mars Lander ..................... 76
Figure A2. Flattened Apollo Type Lander .................................................. 77
Figure A3. Biconic and Shuttle Type Vehicles .......................................... 80
Figure B1 Insertion ∆V vs. Orbital Radius ................................................ 84
Figure B2 Martian System Orbital Schematic .......................................... 86
Figure B3a ∆V vs ∆i Low from LMO to Phobos (i = 30°) ....................... 87a
Figure B3b ∆V vs ∆i Low from LMO to Phobos (i = 25°) ....................... 87b
Figure B3c ∆V vs ∆i Low from LMO to Phobos (i = 0°) ....................... 87c
Figure H1. Trend in Leg Stresses ............................................................... 108
Figure J1. Flight Path for 76.52 min TOF ................................................ 123
Figure J2. Abort Trajectory ..................................................................... 126
Figure J3. Flight Path for 81.99 min TOF ................................................ 129
Figure J4. Contingency Flight Trajectory ................................................ 132

Table 1. Orbital Vehicle Mission Scenarios Summary .............................. 10
Table 2. Orbital Vehicle Sizing Results ..................................................... 16
Table 3. Cost Status Summary .................................................................. 69
Table C1. Payload Mass vs. Total Propellant for One Trip ................................................................. 91
Table J1. Varied Trajectory TOF ........................................................................................................ 122
Table J2. Trajectory for TOF = 76.52 min ......................................................................................... 124
Table J3. Contingency TOF = 535 min .............................................................................................. 127
Table J4. Varied Return TOF ........................................................................................................... 128
Table J5. Return Trajectory for TOF = 81.99 min ........................................................................... 130
Table J6. Return Contingency TOF = 535 min ................................................................................ 133
List of Acronyms

AAO  Administrative Assistance Officer
ACME  Attitude Control and Maneuvering Engine
ADV  Ascent/Descent Vehicle
CDC  Computer Data Corporation
COMSAT  Communication Satellite
DAB  De-orbit Assist Boosters
DODV  De-orbit ΔV
ECLSS  Environment Control and Life Support Systems
EMSAT  Earth-Mars Communication Satellite
GEO  Geosynchronous Earth Orbit
ITV  Interplanetary Transfer Vehicle
LEO  Low Earth Orbit
LH2  Liquid Hydrogen
LLO  Low Lunar Orbit
LO  Liaison Officer
LO2  Liquid Oxygen (also abbreviated as LOX)
LMO  Low Mars Orbit
MSAT  Mars System Communication Satellite
ME  Main Engine
NASA  National Aeronautics and Space Administration
OCC  Operation Control Center
OMV  Orbital Maneuvering Vehicle
OTV  Orbital Transfer Vehicle
<table>
<thead>
<tr>
<th>Acronym</th>
<th>Description</th>
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<tbody>
<tr>
<td>OWL</td>
<td>One Way Lander</td>
</tr>
<tr>
<td>PDR</td>
<td>Preliminary Design Review</td>
</tr>
<tr>
<td>RFP</td>
<td>Request For Proposal</td>
</tr>
<tr>
<td>TOF</td>
<td>Time of Flight</td>
</tr>
</tbody>
</table>
1. General Summary

This document was prepared in response to a Request for Proposal (RFP) for a transportation system to support the robotic construction of bases on Mars and Phobos. The Star Truk Co. presents an approach to defining a transportation system and to specifying the types of vehicles which may be used in the system. This document is composed of four main sections: general summary, technical information, management information, and cost information. The general summary provides an introduction to the project: its objectives and assumptions as seen by the members of the Star Truk Co. The technical section describes an integrated transportation fleet, presents candidate vehicle designs for use in the system, and outlines the process which was used to select and modify these designs. The management section details the organization structure, status, and program schedule of the company. Finally, the cost section presents status of expenses in the performance of the contract.

1.1 Project Background

Before specifying the objectives of the transportation system, the long term goals of a manned presence in the Martian system will be addressed. Idealistically, establishing bases on Mars and Phobos will provide a source of national pride and international cooperation, a technology catalyst, and scientific knowledge. Commercially, these bases will provide fuel and material production sites, a
proving ground for life support systems, and a refueling point enroute to the asteroid belt and outer planetary system. Eventually, a Mars base could support life independent of Earth's resources.

Early steps toward achieving these goals include the construction of the initial human outposts. Science/technology bases on Phobos and Mars will also establish first generation fuel production facilities. A modular design will allow for updating, replacing, and expanding of facilities.

The bases will be transported as prefabricated units by a low-thrust Interplanetary Transfer Vehicle (ITV) to the Martian system and assembled by robots. In this way, humans will not be exposed to the harsh radiation environment during construction. In addition, these unmanned vehicles will use a slow, low-energy transfer orbit to Mars, reducing the fuel consumption. Furthermore, using unmanned vehicles simplifies the vehicles by reducing the redundancy requirements. However, these autonomous missions will require highly sophisticated technology in the following areas that is either currently exists or being developed.

- Cryogenic Fluid Management
- Automated Proximity Operations
- Autonomous Rovers (Telerobotics)
- Aerocapture
- On-orbit Assembly and Construction
- Surface Power
- Advanced Propulsion
- Propellant Production
- Advanced Life Support Systems
- Vehicle Maintenance Facilities
1.2 Design Objectives

The main objective of the Star Truk Co. is to design a fleet of vehicles to transport base components, cargo, and personnel between a Martian parking orbit and the base locations. This fleet will include a combination of vehicles, the designs of which depend on the specific functions to be served. Possible vehicles include: Mars one way landers (OWL), Mars ascent/descent vehicles (ADV), orbital maneuvering vehicles (OMV), orbital transfer vehicles (OTV), and a transportation node in Martian orbit (MarsPort). In addition, satellites will be deployed to provide communication.

1.3 Assumptions

In order to define the scope of the transportation problem, the following assumptions have been made:

- Year of initial cargo departure from low Earth orbit (LEO) and/or low Lunar orbit (LLO) will be between 2020 and 2030.

- A space station suited for vehicle assembly will be operational in LEO and/or LLO.

- A Lunar base will be operational and producing liquid oxygen.

- Elements of the Phobos and Mars bases, along with equipment needed to assemble the bases will be delivered to the Martian system from the Earth system by a single ITV or a series of ITV's.

- Each ITV will enter Martian orbit in the vicinity of Phobos. It will carry base equipment, satellites, and elements of the transportation system. MarsPort (a truss structure, similar to the delta-
truss space station proposed for Earth orbit) could be the backbone of the vehicle or just another payload.

• OMV's and OTV's will be operational (with one generation of experience).

• An inspection team will travel to Phobos to verify base construction. After the initial inspection, the Phobos base will be crew-tended.

• Mars base construction will begin after Phobos base construction has been verified.

• Data from precursor missions of cooperating nations will be available.

• Accurate models of the Martian atmosphere will be available.

• Before the ITV carrying the Mars base arrives, the Phobos base will have produced fuel to support vehicle flights as required by mission scenarios.

1.4 Design Philosophy

Since this project is a conceptual design of mission and spacecrafts, the emphasis is on the overview of the design and not at the subsystem level. Furthermore, the design of a vehicle or system is an iterative process, and the mission and vehicle designs presented in this report represent the results from only the initial iteration.
1.5 Final Product

The Star Truk Co. will provide a conceptual design for a convenient, integrated means to transport materials and personnel to and from the bases within the Martian system. All the technology needed for the current plan is expected to be available for use at the projected date of cargo departure from the Earth system. A modular design for the transportation system will provide for easily implemented contingency plans, so that difficulties with any one vehicle will have a minimal effect on the progress of the total mission.
2. Technical Section

A transportation fleet will be required to move construction equipment and base components from a parking orbit to the surfaces of Mars and Phobos. OMV's will be used for proximity operations within an orbit including cargo transfer from the ITV to Phobos. Travel to Mars will require vehicles capable of soft landing on the Martian surface. The majority of the initial Mars base cargo will be delivered with by the OWL's; smaller ADV's will be used for delivering additional cargo and personnel. Both types of landers will be transferred from Phobos orbit to low Mars orbit using OTV's rather than their own engines to maximize the fuel efficiency of the fleet.

An iterative approach was taken to the design of the fleet and its components. Vehicles were adapted to the proposed scenario, and the scenario was modified as vehicle capabilities were better defined.

2.1 Vehicle Selection Process

A maintainable and efficient transportation system was designed to accommodate the robotic construction of bases on both Mars and Phobos. This system employs a MarsPort, Mars surface landers, OMV's, and OTV's. The specific configuration and the number of each type of vehicle were determined by a set of design criteria. Important considerations during vehicle design include:
• Fulfillment of mission objective. Accomplishing a prescribed mission is the highest priority for any vehicle design. However, mission requirements may dictate a design that is not optimal.

• Modular design with redundancy. If one vehicle fails, another vehicle should be able to complete the failed vehicle’s mission. Similarly, components from vehicles should be interchangeable (i.e. the OWL, ADV, and OTV could all use the same type of main engine.)

• A favorable propellant to total vehicle mass ratio. This ratio provides insight into the efficiency of the vehicle as an overall system.

• Flexibility and growth capability. With time, the role of the Martian outpost in interplanetary exploration will undoubtedly expand. Ability to upgrade flight software and hardware such as computers is desirable since designing entirely new systems is expensive.

The reader is referred to Appendix A for a more complete discussion of vehicle selection.

2.2 Orbital Vehicles

This section discusses the orbital vehicles which will be used in the Star Truk Company transportation system. OMV’s will transfer cargo from the ITV to Phobos and perform prox-ops, OTV’s will transfer the atmospheric entry vehicles and cargo to LMO, and MarsPort will refuel and service vehicles in LMO.

2.2.1 Orbital Vehicle Scenarios

The Star Truk Co. operations scenario covers two missions which will occur during different time frames: Phobos base
construction and, later, Mars base construction. One or more ITV's will be dedicated to each mission, and as each ITV approaches its parking orbit, the communication satellites on board the ITV will be deployed. The ITV will then proceed to a co-orbit with Phobos. After each ITV is positioned in its parking orbit near Phobos, the other vehicles, base components, cargo, and the MarsPort will be removed by an OMV and transported to their respective locations by an OMV or an OTV. Phobos cargo will proceed directly to the surface, and landers bound for the Martian surface will receive fuel from Phobos before OTV's transfer them to LMO. Details of the orbital mechanics for both the solar system and the Martian system are covered in Appendix B.

The remainder of the scenario requires a prediction of the technology and the support behind the program. Rather than choosing a technology baseline for a given year, it was decided to construct two scenarios that represent opposite ends of the spectrum regarding the assumed technology and the assumed political and economic environments supporting the program. These scenarios have been termed upside and downside scenarios.

An upside scenario assumes that the political and economic environments favor the development of a large scale program. With this support, it assumes that all advanced technologies critical to the program can be developed within the specified context and time frame. Also, an upside scenario suggests areas of technology which should be developed.

A downside scenario, on the other hand, assumes an unfavorable political and economic environment. Without the desired support,
limited technology and funding will be available to the program. Current technology or a conservative projection of future technology is used, therefore the scenario does little to identify which technologies should be developed.

This report emphasizes the upside scenario because it describes a more efficient and adaptable transportation system. The upside requires a larger initial investment, but the long-term benefits of a more sophisticated vehicle fleet will better prepare man for a permanent presence at Mars.

Table 1 lists the vehicles and their uses in the two scenarios. An overview of the two scenarios follows. Descriptions of specific operations are found in section 2.4.

2.2.1.1 Upside Scenario

All vehicles mentioned above are used in the upside scenario. The first ITV will insert into a chaser orbit with Phobos, and the OMV's will then be used to deliver the Phobos base elements.

OTV's and the MarsPort will not be used with the Phobos mission, but the MarsPort will be delivered on this cargo mission to alleviate the cargo mass requirements on the subsequent Mars base mission.

Years later, another ITV with the Mars base cargo will insert into Phobos orbit. Since fuel production and vehicle service facilities will be available at Phobos, the ADV's, the OWL's and the OTV's will be transported from Earth without fuel, thereby reducing the payload mass.
<table>
<thead>
<tr>
<th>Component</th>
<th>Upside</th>
<th>Downside</th>
</tr>
</thead>
<tbody>
<tr>
<td>ITV</td>
<td>Insert at Phobos</td>
<td>Insert at Phobos</td>
</tr>
<tr>
<td>OTV</td>
<td>OWL: HMO -&gt; LMO</td>
<td>(Lander engines used for orbit mnvrs and de-orbit)</td>
</tr>
<tr>
<td></td>
<td>OWL: de-orbit burn</td>
<td></td>
</tr>
<tr>
<td></td>
<td>ADV: HMO -&gt; LMO</td>
<td></td>
</tr>
<tr>
<td>OMV</td>
<td>Phobos delivery</td>
<td>Phobos delivery</td>
</tr>
<tr>
<td></td>
<td>Prox-Ops</td>
<td>Prox-Ops</td>
</tr>
<tr>
<td>LMO Node</td>
<td>Fuel depot</td>
<td>Fuel stored at bases</td>
</tr>
<tr>
<td></td>
<td>Service facility</td>
<td>Facilities at Phobos</td>
</tr>
<tr>
<td></td>
<td>Prox-Ops w/ OMV</td>
<td>N/A</td>
</tr>
<tr>
<td></td>
<td>Safe Haven for Man</td>
<td>N/A</td>
</tr>
</tbody>
</table>
The tanks of MarsPort will also be fueled at Phobos prior to delivery of the MarsPort to LMO by the OTV's. At that time service and refueling facilities will be available in the LMO and at Phobos. The OTV's will then begin transporting the Mars landers to LMO where the OTV's will assist the OWL's with de-orbit burns. Since this maneuver will require large amounts of fuel, the OTV will refuel at MarsPort prior to the burn. (As discussed in section 2.4.4.5, the efficiency of locating refueling facilities at MarsPort should be addressed in further studies.)

2.2.1.2 Downside Scenario

The OTV's and MarsPort will not be used in the downside scenario, so the refueling and service facilities will be located at the Phobos base only. Furthermore, the Mars landers will be required to perform their own orbit maneuvers and de-orbit burns. The delivery of the Phobos base will proceed as in the upside scenario, using the OMV's and the ITV.

2.2.2 Orbital Vehicle Requirements

All orbital vehicles must be transportable from Earth to Mars. This study assumes that this criterion has been considered and can be satisfied. In addition, the vehicles must be designed to meet the mission requirements listed below:

- OTV's must provide the necessary orbital transfer \( \Delta V \) to a given payload mass such as the OWL.
• OMV's must maneuver Phobos base components and achieve zero velocity three meters above the Phobos surface.

• For the first mission, all fuel used in the Martian system must be transported from Earth.

• Empty fuel tanks will be delivered to Phobos.

• The orbital mechanics of the transportation fleet scenario (such as placement of the MarsPort) must accommodate efficient use of vehicles and fuel.

• MarsPort should be a stable platform requiring minimal reboost and station keeping propellants.

• There must be contingency plans in case of vehicle failure.

• All operations must be performed autonomously.

• The entire transportation system must be adaptable to future expansion of the Mars and Phobos base capabilities.

2.2.3 Orbital Vehicle Descriptions

The designs for the OMV, OTV, and MarsPort were adapted from the designs of vehicles which are currently being considered for the Earth system. Although an attempt was made to account for the technological advances which will be made in these areas, the possibility of a low-thrust OTV was not explored. Considering the mass of the OWL, and the absence of time constraints on the orbit transfer maneuver, a future study should address the feasibility of using a low-thrust OTV. This type of vehicle may be more efficient
for moving large masses, although the orbit transfer times will be longer.

In the following sections, the design and purpose each vehicle is briefly presented. The method used to size the orbital vehicles is presented in Appendix C.

### 2.2.3.1 Interplanetary Transfer Vehicle (ITV)

This study is concerned only with the configuration of the cargo on the ITV and the size of the nuclear power plant which may be removed from the vehicle for use at the bases. Therefore, the selection of an ITV and its subsystems will not be addressed in this study. As a reference, the Kepler vehicle, presented in the University of Michigan Study, is shown in Figure 1. This vehicle uses nuclear powered ionic propulsion and incorporates a 10 megawatt nuclear power plant.

### 2.2.3.2 Orbital Maneuvering Vehicle (OMV)

OMV's are proposed for moving payloads in the proximity of MarsPort and for delivering payloads to Phobos. A representative OMV is shown in Figure 2. The purpose of the OMV is to move payloads short distances, to perform prox-ops, and (with a service package attached) to perform some vehicle maintenance functions.

The main propulsion system and fuel tanks were sized such that one OMV can move 120 MT (the mass of the Phobos base) from the ITV to Phobos (10 km). These results are shown in Table 2. The RCS will provide 3-axis attitude control with a high degree of control.
Figure 1. Kepler Interplanetary Transfer Vehicle
Table 2. Orbital Vehicle Sizing Results

<table>
<thead>
<tr>
<th>Vehicle</th>
<th>Payload</th>
<th>Mass (MT)</th>
<th>Required Prop (MT)</th>
<th>LO2 Tank Diameter (m)</th>
<th>LH2 Tank Diameter (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>OTV</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Δι=50</td>
<td>OWL-200 - wet</td>
<td>1550</td>
<td>1252</td>
<td>11.7</td>
<td>16.9</td>
</tr>
<tr>
<td></td>
<td>OWL-200 prop.</td>
<td>850</td>
<td>724</td>
<td>9.8</td>
<td>14.1</td>
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<tr>
<td></td>
<td>OWL-200 - dry</td>
<td>700</td>
<td>611</td>
<td>9.3</td>
<td>13.4</td>
</tr>
<tr>
<td></td>
<td>ADV-50 - wet</td>
<td>1060</td>
<td>883</td>
<td>10.5</td>
<td>15.1</td>
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<tr>
<td></td>
<td>ADV-50 prop.</td>
<td>905</td>
<td>766</td>
<td>10.0</td>
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<tr>
<td></td>
<td>ADV-50 - dry</td>
<td>155</td>
<td>199</td>
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<td>9.2</td>
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<td>ADV-10 - wet</td>
<td>245</td>
<td>268</td>
<td>7.1</td>
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<tr>
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<td>ADV-10 prop.</td>
<td>210</td>
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<td>6.8</td>
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<td></td>
<td>ADV-10 - dry</td>
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<td>109</td>
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<td>OMV</td>
<td>Standard Payload</td>
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<td></td>
<td>Module</td>
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</tbody>
</table>
and accuracy. Both propulsion systems will use LH2/LOX fuel produced on Phobos.

2.2.3.3 Orbital Transfer Vehicle (OTV)

The OTV's, shown in Figure 3, were sized without aerobraking shield to transport Martian landing vehicles and payloads between Phobos and Low Mars Orbit (LMO). OTV's are being designed for LEO to geosynchronous orbit (GEO) missions and it is proposed to adapt these vehicles to meet Star Truk Co. mission requirements.

The aerobraking shield shown in the figure would reduce the propellant required to perform the Phobos to Mars orbit transfer. Implementation of this technique is questionable at Mars due to the thin atmosphere and the high mountain peaks.

2.2.3.4 Transportation Node (MarsPort)

A delta truss structure is proposed for the MarsPort. The truss structure will provide a stable configuration with large surface area available for attachment of payloads and fuel tanks. Some of the possible functions of the MarsPort include: storage depot for supplies and fuel, servicing station for vehicles, and refueling station for vehicles. MarsPort will have attitude control devices such as control moment gyros, but reboost will be provided by its attending OMV or by an OTV.

2.2.4 Orbital Vehicle Subsystems

The following section discusses the major subsystems of the orbital vehicles.
2.2.4.1 Computer Systems and Communications

The operation of the orbital vehicles will be monitored and coordinated by an expert computer system, the foreman. Initially, this foreman will function as the main computer of the ITV. In that capacity, it will perform GNC for the ITV, monitor the payloads and other vehicles, communicate with Earth, and act as trouble shooter for the mission. The foreman will be delivered to Phobos with the robots. There, its program will be reconfigured to monitor and coordinate the robots and to continue observing and coordinating the other vehicles. Communications between the foreman and the fleet will be accomplished with the COMSAT's.

The OMV's and OTV's will operate autonomously, but the foreman will monitor them more closely during critical tasks such as docking. In addition, the foreman will conduct periodic systems checks on the vehicles.

Each of the OMV's and OTV's will have an onboard computer with simpler capacities than those of the foreman. It will be programmed to perform a particular task such as to deliver the dock to the beacon on the Phobian surface. An orbital vehicle will also monitor its own equipment status and perform GNC. If it encounters any problems, it will communicate with the foreman for instructions.

If an OMV or OTV has a thruster misfire or otherwise strays from its planned mission, it will attempt to communicate with the foreman. If communication fails, it will home in on a beacon (located on Phobos and on the ITV) and proceed to that location. In the worst case, if main guidance systems fail, the vehicle will rely
on a star tracker to locate its position relative to the satellites or to selected stars.

2.2.4.2. Radiation Protection

Since the orbital vehicles will not transport personnel within them, radiation shielding will be confined to sensitive equipment areas. In addition, shielded hangar facilities will be available to protect the OMV’s and OTV’s during long-term storage and while sensitive equipment is exposed for service.

2.2.4.3 Power System

Power for the computers, cryogenic pumps, and other equipment on each vehicle will be provided by rechargeable cells which will be charged at Phobos. Depending on the technology available, it is possible that each vehicle will be equipped with a small nuclear reactor.

2.2.4.4 Thermal Control System

The need for thermal control will be minimal on the exposed truss structures of the orbital vehicles. Small radiator devices could be provided for each heat generating component, or a central heat dissipating system could be used. During transport of the landers, a larger thermal control system could be attached to the OTV to dissipate the heat accumulated in the lander.
2.3 Atmospheric Entry Vehicles

This sections discuss the vehicles that will be used in delivering cargo to the Martian surface. Designs for three different saucer type vehicles will be presented. They are:

- a large one way lander
- a medium size ascent/descent vehicle
- a small ascent/descent vehicle

Each of these vehicle types will be discussed in the following sections. For a detailed description of the vehicle sizing, the reader is refer to Appendix D.

2.3.1 Atmospheric Vehicle Requirements

Mars landers are required to take base construction equipment and base modules from Mars orbit to the Martian surface. Also, a lander may be required to transport some Phobos base construction equipment and one of the Phobos base modules to the Martian surface. Transportation will also be required for personnel. The following requirements will be placed on the landers:

- All operations must be performed autonomously.
- Landers must be able to withstand aerodynamic heating during descent.
- Landers will be limited to 10-g loadings for cargo and 3-g for personnel.
- Landers must be stable on the Martian surface.
- Landers must be capable of carrying a variety of payload sizes and shapes.
• OWL fuel tanks will be removable and will be used for fuel storage on the Martian surface.

• Landers will employ some level of redundancy in critical systems such as communication and control.

• Landers must provide for easy cargo loading and unloading.

• ADV's must be capable of ascending to at least LMO (104 km).

• At least one ADV will be capable of transporting personnel.

2.3.2 One Way Lander (OWL-200)

The following section will discuss the purpose, requirements, constraints, physical descriptions, and subsystem requirements of the OWL-200.

2.3.2.1 Purpose, Requirements, and Constraints

The One Way Lander-200 (OWL-200) is illustrated in Figure 4. The numeric suffix indicates the payload capacity of this vehicle in metric tons. Four or five of these large landers are proposed for landing the Mars base construction robots and the Mars base modules. The first OWL-200 will carry robots and equipment needed for initial base construction. Subsequent landings of the OWL-200 will carry base habitat modules and a manufacturing facility to the surface.

The following requirements have been imposed on this vehicle:

1) The vehicle must land 200 MT of payload.
2) Once landed, the vehicle will be used as a service/storage facility for the smaller landers and therefore, must be large enough for the ADV-50 and ADV-10 to fit inside.

3). Main engines must be removable for spare parts.

2.3.2.2 Physical Descriptions

The OWL will be a saucer shaped vehicle with two main decks. The upper deck will carry fuel tanks which will surround a central core containing navigation and power supply packages. The propellant will be located above the payload in order to have the vehicle center of gravity decrease in height as it descends instead of increase as would be the case if the propellant were located below the payload. The lower deck will carry cargo and additional fuel tanks. Below the lower deck will be the main engines which will be used to provide a soft landing. A heat shield will cover the lower surface of the OWL during atmospheric entry. See Appendix E for a discussion on the numerical sizing of the OWL.

2.3.2.3 Subsystem Requirements

The following discussion includes the most important subsystems as perceived by the Star Truk Company members. Wherever applicable, the subsystems on board the OWL will be removable from the vehicle.

- Communication System

The OWL-200 will maintain communication with the operations control center (OCC) during descent. (The OCC will be located at
Phobos as discussed in section 2.4.5.) Communication will be constant provided that all COMSATS are operational and except for the unavoidable blackout time during atmospheric entry (if this phenomenon occurs on Mars). This two-way communication system will send the spacecraft's attitude data to and receive commands from the OCC for any attitude adjustments. Furthermore, special connection ports will be integrated in the system to allow the robots to communicate with the OCC. Since this communication system could be salvaged by M.I.N.G. for temporary communication during initial base construction and backup communication in an emergency, the communication package on selected OWL's may include additional features such as tele-video, keyboard, etc., to accommodate human communications. In addition, due to the crucial role of the communication system in the OWL mission, this system will have at least double redundancy.

- **Equipment Status Monitoring (ESM) System**

  During the transit from Earth to Mars, the OWL will perform periodic self-diagnostic tests which will ensure that all systems (with the possible exception of pyrotechnic and propulsion) are functioning properly. The ESM will relay the results to Earth via the communication system of either the ITV or the OWL, whichever appropriate. Upon arrival in the Martian system, all subsystems (including the ESM) will undergo an equipment status verification test which will be performed autonomously or manually during the fueling process.
• **Attitude Control and Navigation System (ACNS)**

During the descent sequence, the attitude and altitude of the OWL will be monitored by the ACNS. This system will issue commands for attitude correction and hover sequence initiation during the descent profile of the OWL. All ACNS commands can be overridden by OCC in case of system malfunction. This means that the commands received from OCC must agree with the programmed commands in at least two of the three ACNS on board to avoid erroneous commands. If total override is required, then the OCC must issue a command to shut down the Logic Unit of the ACNS, where the programmed commands are stored and executed.

• **Heat Shield**

Although the Martian atmosphere is extremely thin compared to that of Earth, aerodynamic heating cannot be ignored during atmospheric entry on Mars. Therefore, it is proposed that a heat shield be used to protect the OWL from aerodynamic heating during entry. This shield will be ejected pyrotechnically in order to minimize the total mass of the vehicle during propulsive deceleration. Unless there is a need to use the heat shield in another capacity after ejection, the shield will be of a charring ablator type, since charring ablators are relatively light, inexpensive and easily removed.

• **Landing Gear**

The landing gear will absorb the landing impact and provide clearance between the ground and main engine nozzles. However,
since the vehicle is a cargo transport, the vehicle height should be minimized to facilitate convenient cargo removal. Because of this dual requirement, it is proposed that the legs be collapsible. During transport and before heat shield ejection, the landing gear will be in the collapsed position. After shield ejection, the gear will be extended hydraulically and collapsed after landing.

- **Power System**

  The OWL will be equipped with a self-contained nuclear power supply unit. This unit could be removed from the OWL to provide extra room when the vehicle is converted to a storage unit or hangar. Because the power system could be used in the Mars base operation, either in a temporary capacity or in an emergency situation, the power requirement may be higher than that needed for the OWL operation.

- **Radiation Shielding**

  Since the OWL will not be used as a personnel carrier during the base construction phase, radiation shielding will be minimal in order to minimize vehicle mass. Radiation protection will be provided mainly by the casing of the individual subsystems.

- **Propulsion Systems**

  The propulsion systems of the OWL-200 include the Main Engines (ME), the Attitude Control and Maneuvering Engines (ACME) and the De-orbit Assist Boosters (DAB). The Main Engines will provide the propulsive power to decelerate the vehicle during entry and to hover.
the vehicle at the landing site. These engines will be used later as replacement parts for the ADV's and the OTV's. The ACME will provide 3-axis attitude correction as well as lateral thrust to guide the vehicle to the landing site. The fuel to be used in the ME and the ACME will be LH2/LOX combination since this fuel will be produced at Phobos. The ACME may use a different type of fuel if a more powerful fuel is developed (i.e., high specific impulse - mass ratio). The DAB will help de-orbit the OWL (along with the OTV's) and will separate from the vehicle before the latter enters the atmosphere. The type of fuel to be used by the DAB is determined mainly by economic factors, since DAB could either be expendable or reusable.

• Thermal Control System

The OWL may experience radiation heating during transit from Earth to Mars. To dissipate this heat, the OWL will be connected to the thermal control system of the ITV. During its trip from the ITV to MarsPort, the OWL will be connected to an external thermal control system which could be attached to the OTV or directly to the OWL. A system attached to the OWL would be removed from the vehicle before de-orbit occurs. The elapsed time of the descent profile will be relatively short, so the internal temperature of the OWL should not exceed the operating limits of the subsystems.

2.3.3 Ascent/Descent Vehicle-50 (ADV-50)

The following section will discuss the purpose, requirements, constraints, physical descriptions, and subsystem requirements of the ADV-50.
2.3.3.1 Purpose, Requirements and Constraints

The ADV-50 will be used to land individual base modules which will be added to the Martian base. This capability will be necessary during initial construction of the base, since the Phobos base medical module will be moved to the Martian base for permanent use there.

The following requirements have been placed on this vehicle:

1) The vehicle must land 50 MT of payload. This is the mass of an individual Mars base module.

2) The vehicle must be able to fit an individual base module in its payload bay area. A base module is a half-cylinder 3 meters in radius and 13 meters in length.

3) The vehicle must ascend to the low Mars parking orbit with 30 MT of payload.

2.3.3.2 Physical Descriptions

The ADV design, shown in Figure 5, includes a payload platform covered by a protective shield, all necessary vehicle systems, and a retractable heat shield. Cargo will be located on top of the payload platform in standardized attachable containers. Because the cargo will be located above the main platform, the vehicle will be capable of transporting a wide range of payload in sizes and shapes. A disadvantage of this method of cargo placement is that the location of the vehicle center of gravity will rise as it burns its propellant since the propellant tanks are located below the payload. See Appendix E for a discussion of numerical sizing of the ADV-50.
2.3.3.3 Subsystems Requirements

Although none of the ADV-50 subsystems is intended to be removed like those of the OWL, most should have easy access for repair and maintenance purposes.

- **Communication Systems**

  Similar to the OWL-200 communication requirements, the ADV-50 will maintain constant communication with the OCC during descent. This two-way communication system will send the spacecraft attitude data to and receive commands from OCC for any attitude adjustments. Due to the crucial role of the communication system in the ADV mission, this system will have at least double redundancy. Since the ADV is expected to be in service for a long period of time, the design of the communication system must allow for incorporation of new technology into the system.

- **Equipment Status Monitoring (ESM) Systems**

  The ADV-50 will be equipped with an ESM system similar to that of the OWL's. An equipment status verification test will be performed on the ADV before departure and after arrival in subsequent missions of the vehicle in addition to the initial test upon its arrival at the Martian system.

- **Attitude Control and Navigation System (ACNS)**

  The ACNS of the ADV will be similar to the OWL's. However, the accuracy of this system will be higher than that of the OWL since the ADV is required to land in a prepared landing site.
• **Heat Shield**

Due to the reusability of the ADV, its heat shield will not be ejected like that of the OWL-200. Therefore, it is proposed that an advanced carbon-carbon heat tile shield be used for aerodynamic heating protection. This type of tile is a proven technology; therefore, it is more reliable than other experimental heat tiles. Although titanium heat tile has been suggested, it is believed that the use of titanium would be too expensive to justify because of its high demand in military applications.

The ADV heat shield must also expose the main engines during propulsive deceleration; to accomplish this, the heat shield will open in a flower petal-like configuration to expose the main engine nozzle. If the heat shield cannot be opened, this region of the shield will be pyrotechnically removed. All heat tiles will be inspected before and after each ADV mission.

• **Landing Gear**

Although the landing gear system on the ADV serves the same dual requirement as that of the OWL (protects the vehicle and allows easy access to cargo), the same collapsible concept cannot be used here since it is believed that such concept would not resulted in a favorable mass versus convenience factor. Instead, a simple rod is proposed. Therefore, the ADV will have three rods or legs stored flush along the side of the heat shield and will be deployed at an angle of 30° from the vertical. This angle was chosen as a first order analysis based on the following constraints: physical,
stability, and structural. Minimum stress occurs when the leg is exactly vertical, however, this configuration does not seemed to provide the spacecraft with stability when landed. Since the leg has finite length, there exists an angle which provides the optimum combination of stability and clearance between the main engines and the ground. See Appendix F for stress analysis on landing gear.

- **Power Systems**

  The ADV will be equipped with a self-contained nuclear power supply unit. Due to a low power output requirement, the ADV can be equipped with batteries or fuel cells if nuclear power is not feasible.

- **Radiation Shielding**

  Design criteria for radiation protection on the ADV will be similar to that of the OWL.

- **Propulsion Systems**

  The propulsion systems of the ADV-50 consist of the ME and the ACME. The Main Engines will provide the propulsive power to decelerate the vehicle during descent and to hover the vehicle at the landing site. In addition, they can be replaced by the ME's from the OWL-200 if need. The ACME will provide 3-axis attitude correction as well as lateral thrust to guide the vehicle to the landing site. The fuel to be used in the ME and the ACME will be LH2/LOX combination since this fuel is produced at Phobos.
Thermal Control System

The thermal control system of the ADV will have the same requirements as those of the OWL.

2.3.4 Ascent/Descent Vehicle-10 (ADV-10)

The following section will discuss the purpose, requirements, constraints, physical descriptions, and subsystem requirements of the ADV-10.

2.3.4.1 Purpose, Requirements and Constraints

There are actually two types of vehicle which fall within the ADV-10 designation. The first type is a payload vehicle which is essentially a down-sized ADV-50. This vehicle is proposed for transporting up to 10 MT of cargo to and from the Mars base. The second vehicle type is designated as the ADV-10P and its purpose is to transport personnel to and from the Martian base. This vehicle will have a higher level of redundancy than the cargo ADV, an ECLSS system, and it will allow for manual override of the autonomous control system.

2.3.4.2 Physical Descriptions

The ADV-10 class is physically smaller than the ADV-50 class, otherwise, both classes are similar in shape. See Appendix E for a discussion of the numerical sizing for the ADV-10.
2.3.4.3 Subsystem Requirements

The subsystem requirements of the ADV-10 class are similar to those of the ADV-50 class, therefore, only major differences that exist between the ADV-10 and the ADV-10P will be noted here. It should be emphasized that since the ADV-10P is man rated, it will have higher redundancy requirements than the ADV-10.

- **Airlock**

The ADV-10P will be equipped with an airlock for docking with the Phobos base and for exiting the vehicle. This airlock will be located inside the ADV-10P at the top of the vehicle personnel carrier dome.

- **Communication Systems**

Due to human presence on the ADV-10P, tele-video and other relevant communication equipment will be included in the personnel carrier dome in addition to the usual complement of communication systems on the ADV-10.

- **ECLSS**

The ADV-10P will be equipped with an Environmental Control and Life Support System which will be located in the personnel carrier dome. Since the vehicle is not expected to make long trips, the ECLSS will provide life support for a maximum of three days with the maximum crew capacity. Waste management and system replenishment will be performed at Phobos and Mars base.
• **Attitude Control and Navigation Systems (ACNS)**

The ADV-10P will be equipped with manual override of the ACNS, otherwise, this system will function similar to the ADV-50 and ADV-10.

• **Power Systems**

Due to the higher redundancy and the additional equipment such as the ECLSS, the power output requirement of the ADV-10P will be higher than that of the ADV-10.

• **Radiation Shielding**

Since the ADV-10P is a personnel carrier, it must provide adequate radiation protection for the crew. To minimize the vehicle mass, however, only the crew deck portion of the vehicle will have the maximum radiation protection. Vehicle subsystems will be shielded only where necessary.

### 2.4 Orbital Operations

This section discusses the following operations:

- ITV Orbit Insertion and Satellite Deployment
- Docking and Cargo Removal
- Payload Management
- Service Facility Requirements
2.4.1 ITV Orbit Insertion and Satellite Deployment

A description of ITV orbit insertion and satellite deployment follows.

2.4.1.1 Phobos Mission

Phobos Mission Orbit insertion and satellite deployment are illustrated in Figure 6. In the description which follows, the steps of the procedure correspond to the numbered locations in the figure. The Phobian satellites will provide a communication link between the robots on the Phobos surface as well as a communication with Earth through COMSAT2.

1. Initially, the ITV will insert into an orbit having the same inclination and eccentricity as the Phobian orbit, but its radius of periapsis will be greater. The orbit of the ITV will then have a slower mean motion, so Phobos will eventually "catch up" with it.

2. After a status check from Earth but before reaching Phobos, the ITV will deploy the COMSAT1 with a spring loaded device. Deployment will force the satellite into an orbit at least 25 km higher than that of Phobos. This will allow the satellite to pass behind Phobos.

3. A burn will be executed to insert the satellite into the Phobian orbit 120° behind Phobos.

4. When the ITV reaches the Phobos vicinity, it will also insert into a co-orbit with Phobos.
Figure 6. Mission Scenario for Phobos Base Delivery
While the ITV is in this chaser orbit, the cargo will be removed as described in section 2.4.3.

The ITV communications system, capable of Mars-to-Earth communications, along with a small attitude control system will then be deployed as a second COMSAT. The satellite's orbit radius will be decreased with a thruster burn, moving it away from Phobos in the orbit direction. Another burn will insert COMSAT2 into a co-orbit with Phobos and COMSAT1, 120° from each.

2.4.1.2 Mars Mission

Mars Mission Orbit insertion and satellite deployment are illustrated in Figure 7. In the description which follows, the steps of the procedure correspond to the numbered locations in the figure. The Mars-synchronous satellites, in conjunction with the Phobos satellites, will provide a constant communication link between Phobos and Mars as well as communication with Earth through two of the satellites. Communication with the ADV's and OWL's will also be accomplished with the satellites.

During the transfer from Earth to Mars and prior to arrival at the Martian system, a cluster of three communication satellites will be released by a spring mechanism. This satellite cluster package consists of one Mars-to-Earth communications satellite (EMSAT) and two Mars System communications satellites (MSAT1 and MSAT2).

After satellite deployment, the following events will occur:

1a The ITV will insert into a co-orbit with Phobos to facilitate fuel and cargo transfer.
The satellite cluster will insert into a Mars synchronous orbit at an orbital radius of approximately six Mars radii.

The satellite cluster will maneuver into a position such that there is a 60° separation angle between the cluster and the future Mars base location.

Spring loaded devices, exerting equal and opposite forces on EMSAT will deploy MSAT1 and MSAT2 simultaneously. One will go into a low, fast orbit with respect to the EMSAT, while the other goes into a high, slow orbit.

Each MSAT will be inserted into Mars-synchronous orbit after achieving a separation angle of 120° from EMSAT.

The Mars landing vehicles will be fueled at Phobos. Then the OTV's will transport them to LMO. (See section 2.4.3 for details.) De-orbit burns will be executed, allowing the landers to enter the Martian atmosphere.

The final configuration of the communication satellites, after the Phobos base and Mars base deliveries, is illustrated in Figure 8.

2.4.2 Docking and Cargo Removal

The Star Truk Company docking system design presents ideas which may be used in the future to connect U.S. and Russian manned
Figure 8. Final Configuration of Communication Satellites.
spacecraft. The docking system requirements, design, and procedure are described in this section.

2.4.2.1 Design Requirements

The following requirements will be placed on the docking system:

- All vehicles in the Star Truk transportation system will be equipped with compatible docking mechanisms.

- Each vehicle will be equipped with two docking mechanisms located symmetrically on opposite sides of the vehicle. This configuration will allow simultaneous attachment to the OMV and ITV.

2.4.2.2 Design Description and Docking Procedure

The docking mechanism system is shown in Figure 9. The base will be attached directly to the vehicle while supporting the attenuators and capture frame.

The vehicle referred to as "active" is the maneuvering vehicle and the "passive" vehicle is the maneuvered one. Upon initiation of the docking procedure, the active vehicle will have its attenuators extended while the passive vehicle's attenuators will be retracted. The attenuators will act as shock absorbers to reduce the impact forces of docking and also allow for minor misalignment of vehicles upon capture. Once the capture latches have been engaged, the attenuators will retract to realign the mating vehicles. Finally, the
structural latches will be engaged and the procedure will be completed.

2.4.3 Payload Management

The OMV is well suited for retrieval and deployment of vehicles as well as maneuvering payloads. The OTV, on the other hand, can provide the thrust necessary for orbit transfer. Specializing the functions of the vehicles in the transportation fleet results in a more efficient system.

The OMV can be operated very close to the ITV and MarsPort whereas the OTV and landers do not have the capability to use their own engines within 300-1000 m of the ITV or the MarsPort. According to the 1986 Space Port Systems group study at The University of Texas, the primary concerns dictating this limitation are environmental contamination due to propulsion effluents, plume impingement on the ITV, and safety. A secondary concern is the difficulty in achieving zero-momentum, precision docking into a relatively small area, particularly with an unmanned vehicle.

2.4.3.1 Phobos Base Delivery

During the Phobos mission, three OMV's will be transported to the Martian system, each docked to the payload package it will transport. Their fuel tanks will be pre-connected to the cryogenic storage tanks, allowing them to be filled soon after arrival in the parking orbit (10 km from the surface of Phobos.) After a systems check, they will begin operations.
The OMV's will transfer the components of the Phobos base to the surface of Phobos (point 5 in Figure 6) along a trajectory determined by the CW equations. As the OMV and payload slowly travel approximately 5 m above the surface, retro-rockets will bring the vehicle to a hover approximately 3 m above the delivery location. There, the OMV will release the payload, allowing it to fall to the surface. After delivering the payload, the OMV will be available to assist the robots with base set-up.

The delivery sequence proceeds as follows:

1. The OMV will transfer the dock, empty ITV fuel tanks, and engines to an unprepared site at Stickney Crater. Next it will extend a power cable to an unprepared site for the power plant connection.

2. After the robots follow the power cable from the dock to the power plant site, they will prepare the site for plant arrival. The OMV will then transfer the power plant to the site and extend a cable to the prospective manned base site.

3. After the robots follow the power cable from the power plant to the base site, they will prepare the site for base arrival. Next, the OMV will transfer the habitat modules to its site.

4. Upon returning to the stripped ITV, COMSAT2 will be deployed and the remaining truss structure and fuel tanks will be taken to the surface/service facility.

5. After the OMV's have completed all base construction tasks they will be checked out, safed, and stored in a hangar.
2.4.3.2 Mars Base Delivery

This section describes the sequence of vehicle operations for the orbital vehicles and the atmospheric vehicles in delivering the Mars base.

2.4.3.2.1 Orbital Vehicles Mission Profile

When the ITV carrying the Mars base arrives in the Martian system, the Phobos fuel production facility will have a supply of LH2/LOX stored in tanks. The ITV will transport the OTV's, OWL's, and ADV's to the Martian system with their fuel tanks empty, and they will be filled at Phobos or at MarsPort.

Upon arrival of the ITV in co-orbit with Phobos, the three OMV's will be removed from their hangar on Phobos, fueled, and prepared for operation. The OMV's will first assist the robots in attaching some of the tanks to the truss structure (MarsPort) which will be in place from the Phobos base delivery. Due to limitations of the OMV capabilities, MarsPort must initially have a mass of 120 MT or less.

An OMV will then transport the partially assembled MarsPort to the ITV vicinity, keeping it 300-1000 m from the ITV. Meanwhile, the other two OMV's will transport filled fuel tanks to the ITV in order to fuel up the OTV's. (The larger vehicles will not be brought into the sphere of influence of Phobos in order to minimize the thrust requirements of the OMV).

After delivering fuel to the OTV's, an OMV will dock with an OTV, and remove the latter from the ITV. The OMV will then move the OTV from 300 to 1000 meters to MarsPort, and attach the OTV to
MarsPort. Finally, the OMV's will complete the transfer of fuel from Phobos to MarsPort.

One OMV will remain with MarsPort, and the OTV will transfer the MarsPort and its attached OMV to LMO. This will establish a service/refueling facility in LMO, complementing the facility at Phobos.

The OTV will then return to the Phobian orbit. There, an OMV will retrieve it, mate it to a landing vehicle, and move the coupled pair 300-1000 m from the ITV. This OMV could then retrieve a returning OTV and repeat the cycle. Meanwhile, another OMV will continue to transfer fuel from Phobos to the returning OTV's.

The OTV will move the lander to LMO where the MarsPort-based OMV will ferry them to MarsPort. There all three of the vehicles will be refueled. The OTV/lander pair will be ferried to the de-orbit position where the OTV will assist with the de-orbit burn. The OTV will then separate from the atmospheric vehicle, return to MarsPort for fuel, and proceed to Phobos.

2.4.3.2.2 OWL Mission Profile

The OWL will arrive in the Martian system attached to the ITV. From there, it will be transported to the low Mars parking orbit by one or more of the OTV's. Once a final vehicle checkout is performed and final landing site reconnaissance is made, the OWL will ignite the de-orbit assist boosters to perform its de-orbit burn in addition to the burn provided by the attached OTV. It will follow a ballistic trajectory through the Martian atmosphere and, at an optimum altitude, it will deploy parachutes to reduce its velocity. The
ejection of the OWL's ablative heat shield will coincide with parachute deployment since the deceleration caused by the parachutes will aid in heat shield ejection. Immediately after heat shield ejection, the main engines will initiate their propulsive deceleration burn and simultaneously, the landing gear will be extended. At a predetermined altitude, the OWL will commence hover sequence and approach the landing site. Upon landing, system safing will occur and a system checkout will be performed by the status monitoring equipment. The results of this checkout will be relayed to OCC and the OWL will then be ready for unloading.

Once the OWL has landed, it will be used on the surface as a temporary storage area for its cargo and then as a permanent hangar facility for the Mars base construction equipment, robots, and the ADV-50 and ADV-10 landing vehicles. It is also possible that one of the OWL's will be converted to a recreation facility for the base personnel.

2.4.3.2.3 ADV-50 Mission profile

The ADV-50's will arrive in the Martian system on the ITV. From there, they will be transported to the low Mars parking orbit by one of the OTV's. Although the ADV could perform its own de-orbit burn by exposing the ME, it is recommended that this capability should be restricted to emergency situation only because there is a possibility that mechanical failure might prevent the opened heat shield from closing. Therefore, the ADV will be assisted in its de-orbit burn by an OTV. The ADV-50 will follow a ballistic trajectory through the Martian atmosphere and at an optimum altitude, will
deploy parachutes to reduce its velocity. At a predetermined altitude, the ADV-50 will open its heat shield, deploy its landing gear and ignite its main engines in order to further reduce its velocity until a hover can be established. Upon landing, system safing will occur, and a system checkout will be performed by the status monitoring equipment. The results of this checkout will be relayed to the OCC and the ADV-50 will then be ready for unloading.

When an ascent mission is required of the ADV-50, the following sequence will be performed. The vehicle will be prepared for launch and all systems will be checked out prior to propellant loading. Once the vehicle is checked out and fueled, it will be cleared for launch and will wait until the proper launch window occurs. The vehicle will burn its main engines until orbit insertion into the parking orbit occurs. The ADV will be fueled at MarsPort or on Mars since it is not expected to travel beyond the LMO. In the LMO the cargo dome will be removed from the ADV by the OMV and OTV will transport the payload to its final destination.

2.4.3.2.4 ADV-10 Mission Profile

The mission profile of the ADV-10 is identical to that of the ADV-50. Due to the 3-g constrains imposed on the ADV-10P, the mission profile of this vehicle may require some modification to meet this requirement. Furthermore, the ADV-10P will be transported to Phobos along with the personnel dome. In addition, the personnel on the ADV-10P will have the ability to control the vehicle or let the automatic systems perform all operations.
2.4.4 Service Facility Requirements

The vehicles operating in the Martian system will need scheduled maintenance and refueling in order to ensure reliable and optimal performance. In addition, unscheduled service may be required, placing high demands on the artificial intelligence capabilities of the vehicles and the robots.

Scheduled servicing includes vehicle inspection, component testing, replenishment of depleted resources, preventive maintenance, equipment replacement, and mission-specific reconfiguration. Unscheduled servicing, on the other hand, is that which is needed to restore the vehicle to an acceptable level of operation following a malfunction or an accident.

2.4.4.1 Methods and Equipment

Like all other aspects of the mission, service will be fully automated. However, it is likely that results of inspections and tests will be relayed to Earth for analysis prior to execution of critical tasks. This will keep humans in the control loop which will be especially important when handling any unscheduled repairs. Any vehicle requiring repairs beyond the capability of the automated systems will be stored until humans are present. The modular design of the vehicle fleet allows other OMV's or OTV's to take over the responsibilities of a damaged vehicle.

OMV's will have limited servicing capabilities with the pre-flight attachment of specialized repair kits. If a vehicle is damaged
or malfunctions at a remote location, an OMV or an OMV/OTV combination will assist or retrieve it.

Aside from remote repairs, most service will be performed in hangar facilities on Phobos and Mars. The service robots provided for the robot crews at those locations will be used along with a remote manipulator system (RMS). At MarsPort, however, only the RMS will be available unless a robot is transferred there for an emergency repair.

2.4.4.2 Propellant Transportation and Storage

This study considers LH2/LOX as the primary fuels because it is assumed that these fuels will be produced on Phobos. It is possible that nuclear or other advanced propellants will be used, but the information on these topics is limited.

Until production begins on Phobos, all fuel will be transported from the Earth system. It could be stored as either LH2/LOX or as water (with later electrolysis to obtain the LH2 and LOX). As noted in the 1987 Gateway report at the University of Texas, further trade studies are needed to compare transportation costs and power requirements of the thermal control systems of each method. In addition, electrolysis power and system requirements must be estimated in order to complete the comparison.

2.4.4.3 Refueling Facilities

The servicing and refueling areas on Phobos and Mars will be separate to achieve minimum contamination and scheduling problems as well as maximum safety. Refueling at Phobos will occur at the
dock; at the Mars base, ADV's will be refueled at the launch pad. In
addition, the orbiting MarsPort refueling facilities should provide a
berthing area for vehicles being refueled. The refueling areas should
be isolated from habitation modules in case of rupture, leak, or
other hazardous events. Finally, all facilities must be able to store
the required propellants, pressurants, secondary fluids, electrolysis
equipment, and refueling equipment.

2.4.4.4 Facilities at Phobos

Phobos will have full service and refueling capabilities for OMV's
and OTV's as well as refueling capability for other vehicles such as
the OWL and ADV. Liquid oxygen and liquid hydrogen produced on
Phobos will be stored in tanks which were used on the ITV's for the
orbit insertion burns and for storage of OMV/OTV propellant. Low-g
pumping equipment there will be used to transfer fuel to the
vehicles in one of two ways. First, a vehicle secured to the dock
could have LO2/LH2 pumped directly into its storage tanks. Second,
fuel could be stored in spare tanks. The RMS would then be used to
remove the empty tanks from the docked vehicle and install the full
tanks. Similarly, full tanks could be transported to the vehicle with
the OMV/OTV, leaving the larger vehicle in orbit. With either
method, fuel will be available for all vehicles used in the Martian
system.

A service hangar will be located near the docking site. One of the
surface vehicles or an RMS will move a docked OMV or OTV to the
enclosed facility where it will assist the service robot with service
operations. In addition, the service package used with the OMV will be stored and installed at the hangar.

A hangar, provided by Phobia Co., will provide radiation, thermal, and micrometeorite protection. This will allow for safe storage and servicing of vehicles, equipment, tools and spare parts. Initially, this hangar will be large enough to store the OMV's during the years between Phobos base delivery and Mars base delivery. When the OTV's arrive with the Mars mission, the hangar will be enlarged or another will be built.

2.4.4.5 Facilities at MarsPort

MarsPort will have refueling capabilities and limited servicing capability. OTV's will transport fuel as cargo from Phobos to MarsPort. Both tank refill and tank exchange techniques will be used, so numerous attachment points for spares must be available on the truss structure. In addition, an RMS and an OMV will be available for rendezvous and prox-ops.

Service will be performed by the RMS and the OMV with service package. However, there will be no hangar and no service robot. Unscheduled repairs could require transfer of a service robot from Phobos to MarsPort, but scheduled maintenance will take place at Phobos or Mars.

MarsPort will have refueling capabilities for several reasons. First, the OTV's will operate more efficiently if they refuel in LMO rather than carry fuel for their return mission to Phobos as extra cargo. Second, the ADV's will refuel there prior to de-orbit. Third, the OMV based at MarsPort will require refueling. Finally, it is
possible that the OWL's will be transported to LMO without fuel in order to reduce the requirements on the OTV's.

2.4.4.6 Facilities at Mars Base

ADV's will be serviced inside the empty cargo bay of an OWL (see Figure 10). The lifting equipment used to build the Mars base will be used to move the vehicles, and the service robot at the base will perform the required maintenance. A hangar facility will be constructed in the empty cargo bay of one of the OWL's. With the addition of radiation protection, and a sealed liner, a shirt sleeve environment could eventually be provided.

Refueling will be performed at the launch facility prior to launch of the ADV's.

2.4.5 Operation Control Center

During the atmospheric vehicles operation there will be communication between the vehicle and OCC to ensure a successful mission. The Star Truk Company has tentatively decided that OCC should be located at Phobos. To minimize any extra burden on the crew at Phobos, the mission sequence will be sent to Phobos from Earth prior to mission initiation.

2.5 System Evolution

In order to develop a transportation system that is responsive to growth, the Star Truk Company proposes the following evolutionary phases for the system:
• **Phase I:** This phase of the mission profile was also discussed in the Orbital Vehicle Scenario. The initial phase of the transportation system can be divided as follows:

  (a) The materials for Phobos base construction will be delivered to the Martian system using an ITV and unloaded on Phobos using OMV's. Two communication satellites will be deployed to provide communication between Earth and Phobos and on the surface of Phobos.

  (b) Mars base construction cargo and three more satellites will be delivered after the completion of the Phobos base. The delivery process will require an ITV to carry OWL's and ADV's from Earth to the Martian system. These landers will be transfer to LMO by OTV's. Subsequent delivery on Mars will be accomplished by ADV's unless payload size requires OWL delivery. The three additional satellites will provide communication between Mars, Phobos and Earth as well as communication on the Martian surface.

• **Phase II:** After base construction is completed, an improved service facility will enable better vehicle maintainability. Mandatory vehicle inspection will be enforced for all vehicles at the beginning of Phase II to ensure performance reliability. Whereas most of the repairs on the vehicles during Phase I will be performed autonomously by robots, it is preferable that more direct human involvement in vehicle repair occur in Phase II. It is expected that the transportation system will primarily support Mars base expansion during this phase. As the system ages, the need for vehicle replacement will arise. The actual time scale for
replacement will ultimately be determined by the components used in vehicle construction.

• Phase III: After routine operations of the original Mars base are achieved, the human presence on Mars will expand. New vehicles or types of vehicles may be introduced into the transportation fleet at this time. It is conceivable that during this phase of the mission profile, the Star Truk Company may be expected to provide transportation for bases at additional locations on the Martian surface. In this situation, the Star Truk Co. would have a centralized transportation base and a new type of vehicle for traveling within the Martian atmosphere or on the Martian surface. Furthermore, future traffic patterns between Phobos and Mars might dictate a change in the objectives of the Star Truk Company.

2.6 Recommendations

Due to the time constraint and the lack of technical expertise, there were many issues which were not addressed in the design of the transportation system and the vehicles in the fleet. They were:

• Vehicle sizing analysis refinement

All analyses performed by the Star Truk Company engineers were intended to provide some ideas about the dimensions of the vehicles. To generate a more precise dimension will require introducing more variables in the analysis thus increasing the level of technical difficulty and the time dedicated to analysis.
• Advanced propulsion system study

This study only addresses LOX/LO2 as propellants for the propulsion system since Phobos is expected to produce them. Using a different type of fuel will complicate the study since the fuel supply must be located. However, the penalty imposed on the design by using only LOX/LO2 indicates that wherever possible, a different type of propulsion system should be used; especially, the need to study the propulsion system for the low-thrust OTV.

• Entry vehicle dynamics and characteristics study

Due to the lack of training in entry vehicle, the Star Truk engineers initially relied on a FORTRAN program written by a previous design group. Unfortunately, after spending a considerable amount of time, the results obtained from this program appeared to be questionable. Without the technical knowledge of vehicle entry dynamics, the Star Truk engineers cannot determine the validity of the results.

• Artificial intelligence study

Although the transportation system designed by the Star Truk engineers required human interactions in critical phases of vehicle operations, all routine operations were required to be autonomous. To realistically address how a certain operation can be performed autonomously, a study in artificial intelligence must be conducted.

• Vehicle construction materials

Due to the lack of time, no materials were studied and proposed for use in constructing the vehicles in the transportation system. A
research on construction materials will eliminate another unknown variable in the vehicle sizing.

- MarsPort trade-off study

Although MarsPort was proposed to provide additional flexibility in the transportation system, its existence imposed rendezvous constraint between the ADV's and the OTV's. Therefore, a study should be conducted to determine whether or not the flexibility provided by MarsPort outweighed its rendezvous constraint.

It is suggested that these topics should be considered in detail in future studies in order to optimize the design of the transportation system.
3. Management Report

3.1 Personnel and Responsibilities

Figure 11 summarizes the organizational structure of the Star Truk Company. In the conceptual stage of the project, the Star Truk Co. consisted of two Project Managers, two Team Leaders, two senior engineers, and three engineers; in addition, one senior engineer served as an Administrative Assistance Officer (AAO) and one Project Manager served as Liaison Officer (LO). During the design stage, the company employs only one Project Manager, two Team Leaders, two senior engineers, and four engineers; the AAO and LO retain their responsibilities. For sake of brevity, this report will only address the company structure in the second stage of the project; the reader is asked to refer to the Star Truk Co. Proposal for description of the management structure in the company's initial stage.

3.1.1 Individual Responsibilities

The Project Manager has final responsibility for all administrative and technical matters. He is responsible for conducting meetings of the Technical Advisory Team and the Star Truk Co. and maintaining communication with the base construction groups in his capacity as the Liaison Officer. To ensure that administrative matters will not be ignored or compromised, the Administrative Assistance Officer is accountable for personnel time.
Figure 11. Organization Structure

Project Manager
- Chau Pham

Liason Officer
- Chau Pham

Administrative Assistance
- Shawn Furgason

Atmospheric Vehicle Team
- Team Leader: Jeannette Gillogly
- Team Members: Jack Brazzel, Mike Luneau, Chau Pham, Bryan Post

Technical Advisory Team
- Bryan Post
- Pat Bosler
- Jeannette Gillogly
- Chau Pham

Orbital Vehicle Team
- Team Leader: Pat Bosler
- Team Members: Shawn Furgason, Vannaroth Nuth, Bill Scheafer
schedules, cost management, and general bookkeeping. The responsibilities of the Team Leaders are to provide technical guidance for their respective teams. Administrative duties are kept to a minimum in order to avoid unnecessary distraction from the research and engineering tasks assigned to the individual team members.

3.1.2 Team Responsibilities

In the conceptual stage of the project, the group was divided into two teams: Vehicle Technology Survey Team and Planetary Survey Team. The main purpose for this structure was to collect relevant data for the vehicle design effort. After the Conceptual Design Review, the Star Truk Co. was restructured in order to respond to its task more effectively. The new structure divides the group into three teams: Orbital Vehicle Team, Atmospheric Vehicle Team, and Technical Advisory Team. The Orbital Vehicle Team responsibilities include orbital and trajectory analysis, selection of the transportation node location, and establishment of the orbital vehicle design criteria. The duties of the Atmospheric Vehicle Team encompass Martian environment research, a study of atmospheric vehicle characteristics, and atmospheric vehicle design criteria formulation. Unlike the two teams mentioned above, the Technical Advisory Team has a more general directive: to ensure that the final design will provide the most logical and practical transportation system. To accomplish this task, the team will provide the background information that will enable the group to see the transportation system in a broad perspective.
In addition to the regular team division, the Star Truk Co. also employs an Editing Team for report preparation. This team consists of members of both the Orbital and Atmospheric Teams who serve on a voluntary basis. The primary responsibility of the Editing Team is to ensure the publication of quality reports. These reports must accurately reflect the direction, objectives, and technical results of the Star Truk Company. Furthermore, all reports must be prepared in a manner consistent in both style and content.

3.2 Program Schedule

The program is following the critical path chart, shown in Figure 12. The program schedule, shown in Figure 13 shows the estimated time-line for milestones (in boldface) and tasks of the project. (The shaded triangles indicate completed tasks).

3.3 Management Status

The Star Truk Company has not encountered any major problems in its management organization. The shift from two Project Managers to one was due to unanticipated time constraints of one of the managers. Productivity of the company has increased as the project has become more clearly defined and the tasks have become more specific. The company has not deviated greatly from the initial time schedule; deadline extensions were approved by the Project Monitor only when major milestones were closely spaced.
Figure 13. Program Schedule
3.4 Recommendations

The following recommendations are based upon some of the problems encountered by the Star Truk Company during the performance of the contract:

1. Better defined project requirement
2. Documentation of previous studies
3. Documentation of computer programs
4. More interactions inside and outside the department

At the beginning of the project, the members of the Star Truk Company spent a considerable amount of time in trying to scope the project. This effort took away valuable time that could have been spent in research or brainstorming. Therefore, to avoid this problem, project statement should be better defined.

Throughout the duration of the contract, the research effort was compromised by lack of documentation of previous studies. For example, important reference sources were not discovered until late in the semester. These references would have been discovered sooner had there been a listing or catalog of previous studies.

The problem which gave the Star Truk Company the most difficulty was the lack of documentation in computer programs by previous groups. Valuable time were spent and lost mostly on trying to figure out exactly what a program does. The dependency on programs written from previous groups were due to the lack of technical expertise in the Star Truk Company. Unfortunately, once all programs were verified, it was discovered that they were written especially for a particular design or the results obtained
were very questionable. Although some effort is made in this report to provide the user with better documentation for all programs written in the performance of the contract by the Star Truk Company, it is hoped that a more standardized format on documentation will be developed and implemented in the future.

The lack of expertise in certain technical areas that existed in the Star Truk Company indicated the necessity for better interactions within and without the Aerospace Department. Since the members of the Star Truk Company were trained in the orbital field of the aerospace engineering program, the company was inadequately prepared to perform atmospheric analysis. Therefore, a better communication with the atmospheric group in the Aerospace Department would be helpful in the future. Although there were no direct need by the Star Truk Company to interact with other departments, our observation of PHOBIA Co. and MING Co. indicated that such interaction would be beneficial.
4. Cost Status

Labor and material costs have been totaled for the fifteen weeks of work performed to date. Table 3 presents a synopsis of current labor and cost totals as compared to projected totals for this point in time. Actual man hour totals are 1547.5 hours. The projected total after fifteen weeks was 1843 hours. This represents a difference of 16% below man hour projections. Total actual labor cost was $31,445 as compared to a projected figure of $37,801. This is 17% below projected cost for labor. (The original proposal was based upon an erroneous schedule of 16 weeks. The original projected total cost was $40,477 as stated above. The cost figure of $37,801 is the correct adjusted figure). Material costs were incurred as expected.

Although actual expenditures are lower than expected, Star Truk has successfully completed each technical and program milestone. The work was completed on schedule and to the required level of technical expertise.

**TABLE 3 Cost Status Summary**

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<td>Material</td>
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5. Bibliography


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Appendices
Appendix A: Vehicle Study

A.1 Vehicle Design

Vehicle selection and modification will be performed according to mission requirements and projected environmental conditions. The selection and modification process will be repeated for each type of vehicle. First, previously studied vehicle designs will be examined. Second, the advantages and disadvantages of these possible choices will be examined. Finally, the merits of each design will be evaluated with a preliminary set of selection criteria. Based on these criteria, trade studies will be conducted for the atmospheric and orbital vehicles. The results of these studies will be used to suggest modifications to the original design. As the vehicles, the mission, and the operating environments are more clearly defined, the selection criteria may be changed or amended.

A.1.1 Orbital Vehicles

Four types of orbital vehicles will be used:

- Interplanetary Transfer Vehicle
- Orbital Transfer Vehicle
- Orbital Maneuver Vehicle
- Transportation Node

A.1.2 Atmospheric Entry Vehicles

Five types of Mars landers were considered:

- a non-aerodynamic type (similar to the lunar landers)
- an Apollo capsule type
- bicone types
• a Space Shuttle derived type
• a flattened Apollo capsule type (flying saucer)

A.1.2.1 Non-Aerodynamic Lander

A non-aerodynamic vehicle is defined as a vehicle which is designed to operate in a negligible atmosphere. The Lunar lander is an example of this type of vehicle. The non-aerodynamic lander would be capable of carrying a diversity of payloads. Some of the base modules that will be used on Mars or transported from Phobos to Mars may require this flexibility. The vehicle envisioned is essentially a propulsion system with a guidance and control package and a mount for cargo. A ballistic trajectory like that of the lunar landing would be required for this vehicle since it will not be able to "fly" in the atmosphere.

A.1.2.2 Apollo Capsule-Type Lander

Due to the geometry of the Apollo-type vehicle, the payload capabilities are limited. Also, the vehicle layout would locate the payload in the nose of the lander. In a vertical landing, the payload would be 10 - 20 meters off the surface, making it more difficult to unload. A horizontal landing of the Apollo-type lander would require a prepared landing runway.

A.1.2.3 Biconic Type Lander

This type of lander has a fairly high lift to drag ratio, but no inherent aerodynamic control surfaces. Also, like the Apollo capsule-type lander, the biconics have payload limitations due to vehicle geometry. The biconics are capable of either vertical or
horizontal landings and, like the Apollo-type, either of these landings has drawbacks.

A.1.2.4 Space Shuttle Derived Lander

A Space Shuttle type lander would have a relatively high lift to drag ratio. This would allow the lander to "fly" long distances in the atmosphere. On the other hand, the low aerodynamic drag characteristic of the vehicle increases the requirement for chemical propulsion to decelerate the vehicle.

In Figure A1, showing the University of Michigan's proposed Mars lander, it can be seen that parachutes and S-turn maneuvering are used to slow the vehicle in the Martian atmosphere. However, the capability of the drogue chute and the S-turn maneuver to reduce velocity is questionable in the low density of the Martian atmosphere. Also shown in the figure is an ascent vehicle launching from the lander. The University of Michigan group proposed this vehicle as a means to return personnel to their orbiting spacecraft.

A.1.2.5 Flattened Apollo-Type Lander

The features of the flattened Apollo or flying saucer type vehicle, shown in Figure A2, include considerable cross-range capability, high aerodynamic drag, high accessibility of payload to surface when landed, and high stability when landed.
Figure A1. University of Michigan Proposed Mars Lander
A.2 Vehicle Selection

A.2.1 Orbital Vehicle Selection

All of the vehicles considered for use in the Mars orbital system will have been developed for use in the Earth system. Relying on this proven technology, the selection process for orbital vehicles is simply a matter of choosing which vehicles will actually be used and adapting them to the mission requirements. In order to define these mission requirements, a scenario describing the use of the vehicles within the system must be described. This, in turn, requires a prediction of the technology and the level of support behind the program.

A.2.2 Atmospheric Vehicle Selection

Each of the vehicles presented in section A.1.2 was evaluated on the following design criteria:

- payload capability
  - mass
  - volume
- accessibility of payload to surface
- stability on surface
- reusability/transformability\(^1\)
- ascent vehicle integration\(^2\)
- atmospheric controllability
- crossrange capability/hover time

\(^1\) Since maximum efficiency is desired in the use of material resources, any Mars lander that cannot be reused as a transportation vehicle will be designed so that it can be transformed into a structure (or structural parts) which can be used by the Martian base or transportation system.

\(^2\) If it is not feasible to design a vehicle capable of descent and ascent, a landing vehicle shall be required to carry an ascent vehicle as cargo.
- aerodynamic features

This preliminary qualitative analysis was not intended to yield an ultimate design, but rather to quickly provide the "best" starting point for further evaluation.

The non-aerodynamic type vehicle (e.g., lunar lander) was considered as a possible Mars lander. While the simple structure of a non-aerodynamic vehicle provides a savings in mass and ease of access to cargo, the vehicle has no inherent aerodynamic stability. In addition, even though the Martian atmosphere is very thin, there is sufficient atmospheric density for aerodynamic heating to occur during entry. Therefore, some modifications would be necessary to enable the vehicle to enter the atmosphere successfully. Propulsive deceleration might be required throughout the descent trajectory to reduce the vehicle velocity, thereby lessening the amount of aerodynamic heating.

The Apollo capsule-type vehicle is not currently under consideration due to the limitations that the vehicle geometry imposes on the size and shape of payloads that could be carried and the difficulties involved with either vertical or horizontal landing.

Although the biconic and the shuttle type vehicles (see Figure A3) provide good aerodynamic stability and high lift to drag ratios, they both rely primarily on aerodynamic control surfaces for attitude control as they fly through the atmosphere. To fly in the thin atmosphere of Mars (approximately 10 mbar maximum) the control surfaces would have to be extremely large to achieve satisfactory vehicle performance. The weight and size of these control surfaces might make the vehicle impractical to operate.
Figure A3. Biconic and Shuttle Type Vehicles
Furthermore, a satisfactory horizontal landing of the biconic or shuttle derived vehicle would require a prepared landing strip, which might not be available during the Mars base construction phase. Vertical landings of these vehicles is not desirable due to the position of the payload relative to the surface when landed.

The flying saucer vehicle has been chosen as the best vehicle for initial analysis. This vehicle type has good cross-range capability, high aerodynamic drag, good accessibility of payload to the surface, and high stability when landed. Cross-range capability allows more flexibility in the atmospheric entry trajectory of the vehicle. Since the lander can maneuver through the atmosphere, greater accuracy in landing at a designated landing site can be expected. High aerodynamic drag is advantageous because it facilitates velocity reduction in the atmosphere. Any velocity that can be lost in the atmosphere reduces the amount of chemical propellant required to slow the vehicle prior to landing. Ease of access to the Martian surface and high stability once landed result from the flying saucer's relatively low profile. Easy access to the surface reduces the complexity of removing payloads from the lander and placing them on the surface. Stability of the vehicle on the surface is necessary for surface operations.
Appendix B: Orbital Analysis

In order to explain some of the orbital vehicle requirements, two orbital mechanics studies are presented. The first examines the minimum ΔV required for insertion into a Mars parking orbit from an Earth-to-Mars transfer orbit. The second shows the minimum ΔV required to transfer within the Mars system. A simplified analysis was performed using circular orbits and Hohmann transfers incorporating a plane change. It was assumed that the computed ΔV was strictly a function of inclination and orbit radius, and that launch windows were not time critical.

**ITV Parking Orbit**

Although this study is not concerned with the specifics of the ITV, the parking orbit was addressed in an attempt to minimize both the ΔV required for insertion into the Mars system and the ΔV required to transfer from the parking orbit to the base locations. First impressions might suggest that the ITV will approach Mars in the Mars orbit plane, indicating that the parking orbit should be in this plane. However, the difference in ΔV required to park in the Phobos orbit rather than the Mars orbit plane is negligible. Therefore, Phobos orbit was selected as the insertion orbit.

As the vehicle approaches Mars in its low energy transfer orbit, it has a particular velocity, V1. The insertion burn can be minimized if the selected parking orbit has an orbital velocity corresponding to V1.
A TKI Solver model used to compute the minimum insertion $\Delta V$ as a function of the parking orbit radius is listed in Appendix G. As shown in Figure B1, the optimal parking orbit radius is approximately 3.5 Mars radii, which is very near the Phobos orbit radius of 2.8 Mars radii. The difference between $\Delta V$ requirements for the two orbits is negligible, so the Phobos orbit was chosen as the parking orbit for the ITV.

Parking in a Phobos chaser orbit is favorable because it minimizes the $\Delta V$ required for Phobos cargo delivery, and it simplifies operations. Orbit transfer, hence fuel consumption and operations associated with OTV's, is eliminated in this scenario. In addition, incoming vehicles can be serviced and refueled at or near Phobos.

Two opposing factors influence the position selected for the ITV in Phobos' orbit. Because of gravitational attraction between the vehicle and Phobos, the distance between them must be large enough to reduce station keeping requirements. On the other hand, increasing the travel distance between the ITV and Phobos increases the OMV thrust requirements. As an approximation of the best location, a distance of 10 km was selected.

Further studies could optimize the placement distance. It may also be more efficient to dock the ITV to Phobos, eliminating the 10 km excursion of the OMV. In this case, the OMV would still be necessary in the vehicle fleet to move the Phobos base components across the surface and to perform prox-ops for the OTV.
Figure B1. Insertion Orbit $\Delta V$ vs. Radius of Orbit
**Low Mars Orbit**

MarsPort will be placed in LMO at an altitude of approximately 100 km if it is incorporated into the overall transportation system. This LMO may, however, be only a rendezvous point between ADV's and OTV's. Referring to Figure B2, it is apparent that a plane change is necessary to transfer between the Mars base and Phobos unless the Mars base is located at 2° latitude. It is assumed that the landing vehicles have limited cross-range capability, so they will require a groundtrack which takes them over the base. That is, the orbit inclination must be greater than or equal to the latitude of the base. Similarly, the inclination of an ascent vehicle's initial orbit can not be less than its launch latitude. Of course, low $\Delta i$ is desired within the orbit system, so the latitude of the Mars base was chosen as the inclination for LMO.

The radius of the LMO ($R_1$) was chosen to reduce the $\Delta V$ associated with the plane change. When using a Hohmann transfer between LMO and Phobos, the $\Delta V$ associated with the plane change decreases as the LMO radius decreases. Thus, $R_1$ should be small, yet must allow for favorable launch windows between LMO and Phobos. The actual time between launch windows is not critical for the robotic missions. Only a regularity was pursued.

As a conservative estimate of the lowest possible altitude for LMO, 100 km was chosen as an initial approximation of $R_1$. Then, based on a gravity model of Mars incorporating J2 effects, the period of repetition of LMO groundtracks was adjusted so that a favorable
launch window occurred. An orbit radius was calculated for each latitude under consideration for the Mars base.

With the radius and the inclination of each orbit in the system defined, the ΔVs required to transfer between orbits was analyzed. This data is critical when estimating the vehicle fuel requirements and when sizing the vehicles. Appendix H lists the TK! Solver model used to study ΔV versus plane change executed at LMO. Figures B3a through B3c show the output plots. For 30° latitude, the minimal ΔV of 1.829 km/s is accomplished when a 4° plane change is made at the LMO burn and the rest is made at the Phobos orbit burn. For 25° latitude, the minimum ΔV of 1.7371 km/s is associated with a LMO plane change of 3°. Finally, for 0° latitude (no plane change), the minimum ΔV is 1.4835 km/s.

Based on these figures, it is recommended that the Mars base is located as close to the equator as possible to reduce the required ΔV. However, the Martian topography may influence the location.
Figure B3a. $\Delta V$ vs. $\Delta i$ Low from LMO to Phobos
Figure B3b. $\Delta V$ vs. $\Delta i$ Low from LMO to Phobos

$\Delta i$ Low (deg)

Total $\Delta V$ (km/s)
Figure B3c. \( \Delta V \) vs. \( \Delta i \) Low from LMO to Phobos

\[ \text{TOTAL } \Delta V \text{ (km/s)} \]
Appendix C: Orbital Vehicle Sizing Procedure

The objective in an initial vehicle sizing is to determine the amount of propellant required by a proposed vehicle to accomplish a given mission. From the propellant requirements, propellant tank sizes can be determined. An initial sizing process has been performed to determine propellant requirements and tank sizes for the OTV and OMV.

For the initial sizing, the worst case scenario was used for the OTV. This scenario represents the most massive proposed payload undergoing the maximum ΔV foreseen. Additional iterations of the sizing process can be performed to refine the entail results obtained.

The following mission is used for sizing the OTV. First, the OTV travels from Phobos or the ITV to LMO delivering its payload. Next, the OTV returns empty to Phobos or the ITV. As part of the worst case scenario, it is assumed that no propellant is located in LMO for refueling the OTV. Therefore, the OTV must carry its return propellant during the cargo delivery trip.

A TKI Solver model entitled OVSIZE was created to aid in OTV sizing. The model and documentation are presented in Appendix I. OVSIZE calculates the total propellant required by a given vehicle for a particular mission and also determines propellant tank diameters. It uses the ideal rocket equation and Hohmann transfers with plane changes, and the following assumptions are made: no
drag, zero -G, no perturbations, no time of flight constraints and no departure or arrival constraints.

OVSIZE requires the following inputs: total ΔV for the round trip, specific impulse of the engine, oxidizer to fuel ratio, payload mass, and final OTV mass after the round trip. The required ΔV was calculated by the TK! Solver program DELV. This program was adapted from a routine provided by Elfego Pinon, a graduate student in the Aerospace Engineering Department of the University of Texas at Austin. The specific impulse (Isp) used for the OTV engine was 450 seconds. Liquid oxygen/liquid hydrogen was assumed for propellant and an oxidizer to fuel ratio (OFR) of 5.5 was chosen. The dry mass of the OTV was used as the final OTV mass after the round trip. From the ideal rocket equation, the initial to final mass ratio is

\[ \frac{M_{\text{final}}}{M_{\text{initial}}} = e^{\left(\frac{\Delta V}{c}\right)} \]

where \( C = I_{sp} \cdot g \)

Also,

\[ M_{\text{final}} = e^{\left(\frac{\Delta V}{c}\right)} \cdot M_{\text{initial}} \]

therefore \( M_{\text{prop}} \) for the return is

\[ M_{\text{prop}} = (1 - e^{\left(\frac{\Delta V}{c}\right)}) \cdot M_{\text{initial}} \]

Final mass for the payload delivery leg of the round trip is the initial mass of the return leg plus payload mass. Applying the ideal
rocket equation again gives the required $M_{\text{prop}}$ for the payload delivery. The two $M_{\text{prop}}$'s are summed for the total propellant required. Ten percent is added to the propellant requirement to account for ullage.

Table C1 presents the propellant requirements which were determined for various payloads and orbit plane inclination changes (i.e., $\Delta V$'s). The corresponding tank diameters for the oxidizer (LO2) and fuel (LH2) are also shown.

Initial propellant tank sizes were calculated for the maximum propellant quantity as calculated above. Both oxidizer and fuel supplies were divided between two tanks each.

The following equations were used to arrive at the propellant tank radii. First, using an oxidizer to fuel ratio of 5.5, the propellant mass can be determined.

\[
M_{\text{prop}} = M_{\text{LO2}} + M_{\text{LH2}}
\]

\[
M_{\text{LO2}} = 5.5 \ M_{\text{LH2}}
\]

Therefore, \[M_{\text{prop}} = 6.5 \ M_{\text{LH2}}.\]

Solving for the mass of the fuel and oxidizer yields,

\[
M_{\text{LH2}} = \frac{M_{\text{prop}}}{6.5}
\]

and, \[M_{\text{LO2}} = (5.5/6.5)M_{\text{prop}}.\]
Table C1. Payload Mass vs. Total Propellant for One Trip

<table>
<thead>
<tr>
<th>Payload (MT)</th>
<th>Propellant (MT)</th>
<th>Oxid tank dia</th>
<th>Fuel tank dia</th>
</tr>
</thead>
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<tr>
<td>1550</td>
<td>1252.0980212701</td>
<td>11.740663137152</td>
<td>16.934982042601</td>
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<td>35</td>
<td>109.4856346155</td>
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</tr>
</tbody>
</table>
Using mass density values of 1185 kg/m³ and 71 kg/m³ for the LO2 and LH2, respectively, the fuel and oxidizer masses were then divided by the appropriate density to obtain the volume. Assuming that all tanks will be spherical to maximize strength and minimize tank size for the given propellant volume, the spherical radii were then calculated using the volumes and the relationship

\[ r = \frac{3}{4\pi} \text{ volume} \]

These calculations are "first order" and many factors have not been addressed. These factors include, but are not limited to, tank pressure and temperature ranges, internal tank hardware, oxidizer and fuel coefficients of expansion, and the burning rate constant. Tank construction will make use of composite material(s) to the greatest extent possible. This is to maximize the strength to weight (mass) ratio of the tanks. Further definition of tank construction must address baffles for propellant slosh control and propellant feed systems (i.e., pumps or gas pressure such as He or N2).

The tanks must also have an efficient thermal control system to maintain proper temperature of the cryogenic propellant. The use of cryogenic propellant also requires additional measures in tank construction to minimize boil-off. Valves used in the propulsion system must be highly reliable and resistant to freezing. A high degree of redundancy in the entire propellant feed system is also
necessary to eliminate single point critical failure modes. Pop-off relief valves will be incorporated to prevent catastrophic failure of a tank.

Another important consideration in tank construction is the ability to replace tanks. As the transportation system matures, propellant tanks could be changed out to smaller sizes when less propellant is required. This would increase the OTV efficiency by tailoring the system and increasing the payload to OTV mass ratio. The OTV could also be enhanced by oversizing the structural volume where the tanks are installed, thereby permitting installation of larger tanks. The OTV design will also include the capability to add or "strap on" additional tanks for contingencies, missions beyond Mars, or for general growth beyond present demands.

**Orbital Maneuvering Vehicle**

Sizing of the OMV follows the same methods described for the OTV since both vehicles perform in the same environment with the same type of propulsion system. However the OMV payload mass requirements are at least three times less than the OTV, and the round trip $\Delta V$ requirements are 850 times less. The CW equations were used to calculate the $\Delta V$ required for a payload transfer (see Appendix J), and a "standard" payload mass was defined as 120 MT. With this information, the ideal rocket equation, as modeled in the OVSIZE program, was used to determine the mass of propellant required. The dry mass of the OMV was estimated as 5 MT. This estimate was based upon the mass of three Shuttle orbital maneuvering system engines.
Appendix D: Atmospheric Vehicle Sizing Procedure

Vehicle Sizing Procedure

A necessary step in the vehicle design process is determining the size of the proposed vehicle. Determining the propellant required by a vehicle for landing and the propellant required for ascent is one of the major steps in vehicle sizing. During the early stages of this project, several computer programs were located which were believed to be useful for determining propellant requirements. Appendix K discusses the descent profile, and Appendices L and M present the computer programs which analyze the de-orbit and entry phases of the profile. After a great deal of time and effort was expended implementing these programs and performing analysis with them, the validity of the results was brought into question. The questionable characteristics of these programs is discussed in the appendix which addresses each of the programs. The sizing process presented here is based on the ideal rocket equation.

Assumptions Made in the Sizing Process

The assumptions made in the vehicle sizing process are presented in this section. The significance of these assumptions will be discussed in the following section.

Assumptions were made in each of the following areas:

- $\Delta V$ required for landing,
- $\Delta V$ required for ascent,
- vehicle to payload mass ration,
• propellant mix ratio,
• non-rotating planet, and
• negligible atmosphere.

All of these assumptions were made in order to simplify the vehicle sizing process during the preliminary design stage. Each of the assumptions and results are discussed in detail below.

**Negligible atmosphere** This assumption eliminates the requirement to either estimate or determine analytically the $\Delta V$ which is lost due to atmospheric drag.

**Non-rotating planet** This assumption was made since the latitude of the Mars base has not been decided upon. Determining the $\Delta V$ which can be provided to a launch vehicle by the angular velocity of a planet depends on the latitude of the launch site and the azimuth angle at which the vehicle is launched.

**Landing $\Delta V$** In the sizing process for each of the vehicles, it was assumed that the $\Delta V$ required for descent was equal to the vehicle orbital velocity in a circular orbit 104 km above the Martian surface (104 km is the altitude of the proposed low Mars parking orbit). The velocity in this orbit is approximately 3500 m/s.

**Ascent $\Delta V$** The escape velocity for Mars was used as the $\Delta V$ required to launch the ADV's into the 104 km parking orbit. Escape velocity for Mars is approximately 5000 m/s.
Vehicle to payload mass ratio  This ratio was used to estimate the dry mass of the vehicle (i.e., the vehicle without its payload or propellant). A vehicle to payload mass ratio of 1:2.5 was used in the sizing process. Current space vehicles do not achieve this ratio, but it is expected that the ratio used here will be reasonable for the time frame which is being addressed in this study.

Propellant mix ratio/density  The propellant mix ratio and densities were used to determine the volume of tankage required to store the propellant in each of the vehicles. A mix ratio of 6:1 (LH2:LOX) was used for the sizing process. The following densities for the LOX and LH2 propellants were used:

\[
\begin{align*}
\text{LH2:} & \quad 1010 \text{ kg/m}^3, \\
\text{LOX:} & \quad 62.2 \text{ kg/m}^3.
\end{align*}
\]

Ideal rocket equation  The ideal rocket equation was used for determining the propellant requirements of the different vehicles.

Results of Assumptions Made in the Sizing Process  The significance of the assumptions made in the vehicle sizing process are presented in this section.

Negligible or non-existent atmosphere  During a descent trajectory, a planet atmosphere helps to reduce the vehicle velocity.
On ascent, the atmosphere increases the \( \Delta V \) required for the vehicle to achieve orbit. Therefore, by assuming a negligible atmosphere, the propellant required for descent is overestimated and that required for ascent may be underestimated.

**Non-rotating planet** When launching a vehicle, the angular velocity of the planet can provide additional \( \Delta V \) to help boost the vehicle into orbit. Assuming a non-rotating planet neglects this potential \( \Delta V \) gain therefore increases the propellant requirement to launch the vehicle.

**Landing \( \Delta V \)** Estimating or determining this \( \Delta V \) is necessary to use the ideal rocket equation for determining propellant requirements. By neglecting the drag of the Martian atmosphere, the \( \Delta V \) for landing is equal to the velocity of the landing vehicle when it enters the planet atmosphere. This velocity depends on the type of orbit which the vehicle is in prior to entry and it is possible that the actual velocity may be higher or lower than the value used here.

**Ascent \( \Delta V \)** The \( \Delta V \) for ascent is affected by the atmospheric drag, the angular velocity of the launch site, and the altitude of the target orbit. The escape velocity for Mars was used as the \( \Delta V \) for ascent with the belief that it is a conservative value to use and will lead to overestimating the propellant requirements rather than underestimating them.
Vehicle to payload mass ratio  This factor affects the sizing process significantly. Increasing the ratio of the vehicle to payload mass (for example, 4:1 instead of 2.5:1) increases the mass of propellant required for the vehicle to perform its mission.

Propellant mix ratio  The propellant mix ratio has a direct effect on the volume of tankage which a vehicle will require. Increasing the ratio of LH2 to LOX (for example, 8:1 instead of 6:1) reduces the volume of tankage required.

Ideal rocket equation  The ideal rocket equation is very useful in preliminary design due to its simplicity. A more accurate method would numerically integrate the vehicle equations of motion through a desired trajectory to determine propellant requirements.
Appendix E: Atmospheric Vehicles Numerical Sizing

OWL-200 Sizing Procedure

Based on the requirements listed in section 2.3.2.1, the following dimensions for the payload deck were determined for the OWL-200A:

radius: 150 meters,  
height: 13.5 meters.

The OWL-200B is slightly larger in order to accommodate the Mars base manufacturing facility which has a dimension of 25 x 25 x 4 meters. Since the need for this type of vehicle has just surfaced, there was not enough time to perform sizing analysis.

The next step in the sizing process was to determine the mass and volume of propellant required to perform the landing mission using the ideal rocket equation. Beginning with the payload mass of 200 MT, a vehicle to payload mass ratio of 2.5 was used to determine the mass of the overall vehicle (500 MT) without payload or fuel. The sum of the vehicle and payload mass (700 MT) was used as the final mass of the lander on the surface of Mars. With the final mass of the vehicle and the ΔV required to land (3500 m/s) known, the following data was calculated:

<table>
<thead>
<tr>
<th>Description</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass of vehicle/payload combination</td>
<td>700 MT</td>
</tr>
<tr>
<td>Mass of vehicle/payload/fuel combination</td>
<td>1550 MT</td>
</tr>
<tr>
<td>Mass of propellant to land</td>
<td>850 MT</td>
</tr>
<tr>
<td>Volume of propellant to land</td>
<td>1000 m³</td>
</tr>
</tbody>
</table>
Knowing the volume of propellant that the vehicle would require and the maximum volume of the proposed payload, the payload deck was configured as shown in Figure 4. The propellant tanks and the deck on which they are mounted will be removable so that the entire interior of the payload deck can be used as a vehicle hangar. The propellant tanks will be used by the Mars base for propellant storage.

**ADV-50 Sizing Procedure**

The dimensions of a base module determined the following dimensions for the payload deck and standard payload protective shell:

- Payload deck radius: 80 meters,
- Protective shell radius: 80 meters,
- Protective shell height: 3.5 meters.

To continue the sizing process for this vehicle it was necessary to determine the volume of propellant required for it to perform its mission. The first calculations in this process were performed using the ideal rocket equation method and were based on the following assumptions:

1) After landing a base module, the ADV-50 would ascend to low Mars orbit with 30 MT of payload.
2) The ΔV for descent and ascent are 3.5 km/s and 5.0 km/s, respectively.

Using a vehicle to payload mass ratio of 2.5, and the assumptions above, the following data was calculated:

Vehicle/payload mass after ascent: 55 MT
Propellant mass for ascent: 325 MT
Vehicle/payload/propellant mass on surface: 480 MT
Propellant mass for descent: 580 MT
Vehicle/payload/propellant mass on orbit: 1060 MT
Fuel volume for descent/ascent mission: 420 m³

It was determined that this volume of propellant could not be accommodated with the vehicle dimensions as stated above. Therefore, the radius of the proposed vehicle was increased, and the following dimensions were calculated for a vehicle which can accommodate the propellant required along with the necessary vehicle systems.

Payload deck radius: 10 meters
Height of cylindrical mid-section: 2 meters
Height of lower section: 5 meters

An ADV of these dimensions will fit inside the OWL and also allow for larger payloads to be transported (within the 50 MT limit).

At this point in the sizing process it was realized that a large amount of propellant is required to land the propellant required for ascent. For the case mentioned above, 395 MT of propellant is required just to land the 325 MT of propellant required for ascent. For this reason, it is strongly recommended that propellant
production on the Martian surface be investigated. Also, more powerful and less massive propellants should be developed.

A second iteration of the sizing process was made using the assumption that propellant was available on the Martian surface and that the ADV would not carry its ascent fuel during descent. With this assumption, the following vehicle dimensions were calculated:

- Payload deck radius: 8 meters
- Height of cylindrical mid-section: 1 meter
- Height of lower section: 5 meters

This vehicle would easily fit inside of the OWL and allow room for maintenance and repairs equipment.

**ADV-10 Sizing Procedure**

The physical dimensions of the basic ADV-10 have been estimated at:

- Vehicle radius: 6 meters
- Height of upper section: 3 meters
- Height of cylindrical mid-section: 1 meter
- Height of lower section: 3 meters

The proposed payload capacity of the ADV-10 is 10 metric tons (MT). By applying a vehicle to payload mass ratio of 2.5, an overall vehicle mass of 25 MT is arrived at; this is the vehicle mass without propellant. Using the ideal rocket equation method and a ΔV of 5000
m/s to launch the vehicle into orbit, the following masses are determined.

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<td>Vehicle mass prior to de-orbit:</td>
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<tr>
<td>Vehicle mass when landed:</td>
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</tr>
<tr>
<td>Propellant mass to land:</td>
<td>130 MT</td>
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<tr>
<td>Vehicle mass at launch:</td>
<td>110 MT</td>
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<tr>
<td>Vehicle mass on orbit:</td>
<td>35 MT</td>
</tr>
<tr>
<td>Propellant mass to achieve orbit:</td>
<td>75 MT</td>
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</table>

The ADV-10P is expected to be more massive than the basic vehicle due to the requirement for multiple redundancies in man-rated systems as well as life support system. It is also possible that this vehicle will be physically larger than the basic ADV-10.
Appendix F: Landing Gear Stress Analysis

A first order approximation analysis was performed in order to determine the lengths and diameters of the landing gear for the ADV's (Ascent/Descent Vehicles). The following assumptions were made in the analysis:

- Spacecraft lands equally on all legs.
- Gravity is the only external force acting in the vertical direction through the vehicle centroid.

\[ mg \]

\[ R \]

\[ R \]

\[ R \]
A vector analysis,

$$\sum \overrightarrow{M} = \overrightarrow{r} \times \overrightarrow{F} = \overrightarrow{0}$$ \hspace{1cm} (Eq. 1)

shows that the only reaction forces $\overrightarrow{R}$ are in the vertical direction and

$$\overrightarrow{R} = \frac{mg'}{number \ of \ legs}$$ \hspace{1cm} (Eq. 2)

where

- $\overrightarrow{r}$ = vector from any point to point of force reaction,
- $\overrightarrow{F}$ = any reaction force $\overrightarrow{R}$,
- $m$ = vehicle mass, and
- $g'$ = gravitational acceleration.

The axial stress in each leg will be:

$$\sigma = \frac{F}{A}$$ \hspace{1cm} (Eq. 3)

which leads to:

$$\sigma_{axial} = \frac{R}{A} = \frac{R}{\pi(d/2)^2} \sin \theta$$ \hspace{1cm} (Eq. 4)

where

- $d$ = diameter of a leg, and
- $\theta$ = the angle between the leg and the foot plate (see illustration).
Therefore,

\[ \sigma_{axial} = \frac{0.38 \text{mg}}{\# \text{ of legs} \times \pi (d/2)^2} \sin \theta \]  \hspace{1cm} \text{(Eq. 5)}

The bending stress in the leg will be:

\[ \sigma_{bending} = \frac{M \cdot c}{I} \]  \hspace{1cm} \text{(Eq. 6)}

where

- \( M \) = moment induced by \( R_b \),
- \( C \) = distance from neutral axis of member to point of stress,

and

- \( I \) = moment of inertia of the leg.

Since maximum stress occurs when

\[ M = R_b \cdot l \]  \hspace{1cm} \text{(where} \ l \text{ is length of leg)}

\[ C = \text{radius of leg} \]  \hspace{1cm} \text{(Eqs. 7)}

For a cylindrical leg,
\[ I = \pi \frac{d^4}{64} \]  
(Eq. 8)

Thus

\[ \sigma_{\text{bending}} = \frac{R_{\text{bl}}d}{2} \frac{d^4}{\pi^4} \]  
(Eq. 9)

The total stress in a leg will be the sum of the bending and axial stress.

\[ \sigma_{\text{total}} = \sigma_{\text{axial}} + \sigma_{\text{bending}} \]  
(Eq. 10)

A FORTRAN program was written which calculates \( \sigma_{\text{total}} \) for a vehicle with a given mass and number of legs (see Appendix N). The program will give the results for total stress for leg lengths of 1.5, 2.0, 2.5, and 3.0 meters with foot plate angles from 45° to 90° (in 5° increments).

Generally, the stresses increase as the leg length increases. Also, as the foot plate angle approaches 90°, the stress in the legs approaches a minimum and the bending stress becomes negligible (see Figure H1). Here one would have to look at buckling analysis as the leg length increases.

In the worst possible situation when the spacecraft landed on only one leg, the total stress becomes:

\[ \sigma_{\text{total}} = \frac{0.38mg \sin \theta(4)}{\pi d^2} + \frac{32m(0.38g) \cos \theta(1)}{\pi d^3} \]  
(Eq. 11)
Figure H1. Trend in Leg Stresses
Appendix G: INSERT TK! Solver Model

**Purpose:** This TK! Solver model will be used to determine the minimum $\Delta V$ required for insertion into the Mars system.

**Note on inputs and outputs:** The variables in TK! Solver model can be defined as inputs or outputs depending on the objective of the analysis. The variables shown below are the ones which the current design team will use in their analysis.

**Inputs:**
- Solar Gravitational Parameter
- Gravitational Parameter of Earth
- Gravitational Parameter for Mars
- Periapse radius at Earth
- Radius of Mars
- Radius of Mars Orbit
- Factor (ratio of insertion orbit to Mars orbit)

**Outputs:**
- Periapse radius at Mars (the insertion radius) and $\Delta V_2$ (the required fuel burn for insertion)

**Source:** Dr. Wallace T. Fowler, graduate mission design class instructor at The University of Texas at Austin.

**Modifications made:** Not applicable.

Listing: See below
Sample run: See below
Verification: N/A
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<td>1.1</td>
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RULE SHEET

S Rule
"Fetched Conic - Hohmann to Mars"

\[ Vp1 = \sqrt{\mu_1 / r_1} \]
\[ Vt1 = \sqrt{\mu_1 (2/r_1 - 1/a_1)} \]
\[ a_1 = (r_1 + r_2) / 2 \]
\[ TOF = \pi^2 / \sqrt{\mu_1 / (r_2^3 / \mu_1)} \]
\[ \text{Phase} = \pi / 2 - \sqrt{\mu_1 / (r_2^3 / \mu_1)} \times TOF \]

\[ Vt2 = \sqrt{\mu_1 (2/r_2 - 1/a_2)} \]
\[ Vp2 = \sqrt{\mu_1 / r_2} \]
\[ Vinf1 = \text{abs}(Vt1 - Vp1) \]
\[ Vinf2 = \text{abs}(Vt2 - Vp2) \]
\[ Vinf1^2 / 2 = Vper1^2 / 2 - \mu_1 / rper1 \]
\[ Vinf2^2 / 2 = Vper2^2 / 2 - \mu_2 / rper2 \]
\[ Vorb1 = \sqrt{\mu_1 / rper1} \]
\[ rper2 = \text{Factor} \times \text{Planet} \]
\[ Vorb2 = \sqrt{\mu_2 / rper2} \]
\[ Dv1 = \text{abs}(Vper1 - Vorb1) \]
\[ Dv2 = \text{abs}(Vper2 - Vorb2) \]
\[ DV = Dv1 + Dv2 \]
Appendix H: DELV TK! Solver Model

**Purpose:** This TK! Solver model will be used to determine the minimum $\Delta V$ required to transfer between Phobos orbit and the Mars parking orbits whose inclination are the same as base latitudes. The $\Delta V$ for the transfer will be used to size the Orbital Transfer Vehicle (OTV).

**Note on inputs and outputs:** The variables in TK! Solver model can be defined as inputs or outputs depending on the objective of the analysis. The variables shown below are the ones which the current design team will use in their analysis.

**Inputs:**
- Gravitational Parameter for Mars
- Radius of Low Mars Orbit (the parking orbit)
- Radius of Phobos Orbit
- Plane Change at first burn
- Total Plane Change (the same as base latitude)

**Outputs:**
- $\Delta V_{total}$ (the required fuel burn for the transfer)

**Source:** Written by the Orbital Analysis Team of the Star Truk Co., Spring 1989, The University of Texas at Austin.

**Modifications made:** N/A

**Theory:** A Hohmann transfer incorporating a plane change for the circular orbit is used. Refer to *Fundamental of Astrodynamics*, by R. R. Bate, DD. Mueller, J. E. White, Dover Publications, Inc., New York, 1971.
**Listing:** See below.

**Sample run:** See below.

**Verification:** N/A
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<td>Vt1</td>
<td>Vt2</td>
<td>km/s</td>
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<td>km/s</td>
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<td>d1h</td>
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<td>deg</td>
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<td>Total velocity change for the transfer</td>
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### RULE SHEET

- Vc1 = sqrt(mu/r1)
- Vc2 = sqrt(mu/r2)
- Vt1 = sqrt(mu*((2/r1)-(1/at))
- Vt2 = sqrt(mu*((2/r2)-(1/at))
- at = 0.5*(r1+r2)
- dV1 = sqrt(Vt1*Vt1+Vc1*Vc1+Vc1*cos(d1))
- dV2 = sqrt(Vt2*Vt2+Vc2*Vc2-Vt1*Vc1*cos(d1)-d1))
- d1total = d1+d1h
- dvtotal = dV1+dV2
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Appendix I: OVSIZE TK! Solver Model

Purpose: This TK! Solver model was used to determine the propellant mass required by the OTV to transfer a given payload between orbits using a Hohmann transfer.

Note on inputs and outputs: The variables in a TK! Solver model can be defined as inputs or outputs depending on the objective of the analysis. The variables shown below were used in this particular analysis.

Inputs:
- Total ΔV for the round trip transfer
- Specific impulse
- Oxidizer / fuel ratio
- Payload mass
- Final mass of the OTV after a round trip
- Oxidizer and fuel densities

Outputs:
- Total propellant quantity in metric tons
- Diameter of spherical tank containing half of each propellant component

Source: This model was created by the Orbital Vehicle Team of Star Truk Company in April 1989.

Modifications made: N/A

Theory: Standard propulsion theory was used in applying the ideal rocket equation. This theory assumes there are neither perturbations nor atmospheric drag. The rocket equation solves for
an initial to final mass ratio. With the ratio and the input final mass of the OTV the initial mass prior to one leg of the transfer can be calculated. The required propellant is then the difference between the initial and final masses. The initial mass of the OTV for the return leg plus the payload mass becomes the final mass for the transfer. The procedure just described is repeated to determine the masses and the propellant quantity for the payload transfer. The total quantity for the round trip is the sum of the calculated quantities. The oxidizer/fuel ratio is applied to find the mass of each of the propellant component quantities. The component masses are divided by the specified volumes to obtain the propellant component volumes. The equation for the volume of a sphere is then used to determine the tank diameter.

**Listing:** See below

**Sample run:** See below

**Verification:** Hand calculations were performed to verify the program. A number of test points were calculated and substantiated the results of the model.
VARIABLE SHEET

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<td></td>
<td>c</td>
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<td>ratio</td>
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<td>m</td>
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</table>

Comment

MASS SIZING PROCEDURE

Change of velocity for the transfer
Exit velocity
Mass ratio
Initial mass for the return trip
Final mass for the return trip
Mass of fuel for the return trip
Additional fuel mass
Final mass for the outgoing
Mass of the payload
Initial mass for the outgoing
Additional fuel mass
Mass of fuel for the outgoing
Total fuel mass

TANK SIZING PROCEDURE

Mass of fuel
Ox/Fuel ratio
Mass of Ox
Volume of Ox
Density of Ox
Volume of fuel
Density of fuel
Ox tank diameter
Fuel tank diameter

S Rule

\[
\begin{align*}
\text{ratio} &= \exp(\text{delV}/c) \\
\text{Mf1} &= \text{ratio} \times \text{Mf2} \\
\text{Mf2} &= \text{Mf1} - \text{Mf2} \\
\text{Mpaux2} &= 1.1 \times \text{Mp2} \\
\text{Mf1} &= \text{Mpaux2} + \text{Mf2} + \text{Mp1} \\
\text{M1} &= \text{ratio} \times \text{Mf1} \\
\text{Mpaux1} &= 1.1 \times \text{Mp1} \\
\text{Mp1} &= \text{M1} - \text{Mf1} \\
\text{Mptot} &= \text{Mpaux1} + \text{Mpaux2} \\
\text{Mf1} &= \text{Mptot} / (\text{OFR} + 1.0) \\
\text{Mox} &= \text{Mptot} - \text{Mf1} \\
\text{Vox} &= \text{Mox} / \text{Denox} \\
\text{Vf1} &= \text{Mf1} / \text{Denfl} \\
\text{TDox} &= 2 \times ((3/(4 \times \text{isp}(0.33))) \\
\text{Tdf1} &= 2 \times ((3/(4 \times \text{isp}(0.33))) \\
\text{c} &= \text{isp} \times 9.81
\end{align*}
\]
### Table 1: Payload Mass versus Total Propellant Mass for One Vertical

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<td>m/s</td>
<td>delta $\Delta V$ for various incl changes: 0, 25, 30</td>
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<td>Payload Mass</td>
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<td>MT</td>
<td>Total Propellant Mass</td>
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<td>Final mass of OTV</td>
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<td>$T\text{Dox}$</td>
<td>9</td>
<td>m</td>
<td>Oxidizer tank diameter</td>
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<tr>
<td>$T\text{Df}$</td>
<td>9</td>
<td>m</td>
<td>Fuel tank diameter</td>
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Appendix J: Computation of $\Delta V$'s for the OMV

J1 $\Delta V$ Determination

A targeting was made from an initial point to a desired final location for various times of flight. The output was then used for the initial trajectory states and propagated. An examination of the resulting flight paths then determined a range of acceptable $\Delta V$'s. A TK! Solver! model was created for the problem. A listing of the model and documentation are in Appendix O.

J2 Results

Through interaction with the PHOBIA Co. an approximate landing site was located at $x$-distance $= -11$ km and $z$-distance $= 6$ km from the ITV. This site is near the trenches radiating from Stickney Crater located on the Phobos, facing Mars. The sites considered by PHOBIA are near the equator of Phobos, therefore the $y$ distances and velocities were not calculated.

In Table J1 the TOF is varied for targeting from the ITV to Phobos ($x_0 = 0, z_0 = 0, x = -11000, z = 6000$), and the resulting velocities are displayed. A TOF of 40 minutes or less produced excessive $\Delta V$'s as the flight path was nearly linear. TOF's greater than 95 minutes produced a path which is greater the -11000 meters in the $x$-direction and, therefore, would pass through Phobos in its effort to rendezvous with the target. Figure J1 shows the resulting path for a 76.52 minute TOF, the state values for this trajectory are given in Table J2. This is a representative flight path for TOF's less
Table J1. Varied Trajectory TOF for
\[ x = -11000 \text{ m} \]
\[ z = 6000 \text{ m} \]
\[ a = 9408 \text{ km} \]

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Figure J1. Flight Path for 76.52 min TOF
Table J2. Trajectory for TOF = 76.52 min
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\[ v_{z_0} = -1.6485 \text{ m/sec} \]

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than 95 minutes and greater than 40 minutes. In this flight the vehicle will pass along the Phobos surface at approximately 3 to 5 meters per second. It will be able to commit to landing by executing a reversed thrust, reducing its velocity to zero. The total ΔV's for the TOF's between 95 and 40 minutes are seen to lie between 6 and 10 meters per second, respectively.

Once brought to near zero relative velocity, the OMV-Payload unit will use small thruster engines to maneuver to approximately 100 to 50 meters above the proposed landing site. The payload will be allowed to "drop" to a soft landing on Phobos. The velocity changes, and fuel usages, for this hover maneuver are assumed to be small relative to the transfer trajectory.

Should an abort occur before a commit to land, the trajectory will be that of Figure J2. Table J3 presents the state values on this trajectory. It is seen that the vehicle passes through the Phobos-ITV orbit path and exceeds the radial distance by one kilometer. At this point, a burn of 3.24 m/sec will place the vehicle and payload in a slightly higher orbit, so it will move at a slower orbital rate. The vehicle will then move toward the ITV where it may refuel for a new landing attempt.

Table J4 gives the velocities for the return of the OMV to the ITV as the time of flight is varied. Most trajectory total ΔV's are less than 10 m/sec. TOF's greater than 90 minutes result in trajectories which do rendezvous with the ITV; however, if the OMV continues on its course it will not make it around Phobos. A representative flight, with a TOF of 81.99 minutes, is shown in Figure J3 with values given in Table J5. A contingency flight is
Table J3. Contingency TOF = 535 min

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Table J4. Varied Return TOF from
x = 11000 m
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a = 9402 km

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Figure J3. Flight Path for 81.99 min TOF
Table J5. Return Trajectory for TOF = 81.99 min

\( v_{x0} = 2.538 \text{ m/sec} \)

\( v_{z0} = 1.714 \text{ m/sec} \)

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shown in Figure J4, with values in Table J6. A burn at the point where the z-velocity is zero will place the OMV in an orbit which is lower than that of Phobos and allow the OMV to catch up to the Phobos-ITV vicinity.

J3 Recommendations

The ΔV's from this study are needed for OMV sizing. It is recommended that 10 m/sec should be used for each portion of the ITV-Phobos and Phobos-ITV trajectories.
Figure J4. Contingency Flight Trajectory
Table J6. Return Contingency TOF = 535 min

<table>
<thead>
<tr>
<th>Time (min)</th>
<th>X meters</th>
<th>Y meters</th>
<th>Vx (m/s)</th>
<th>Vy (m/s)</th>
<th>DV (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>2.53817484</td>
<td>-1.71430557</td>
<td>3.0628704</td>
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<tr>
<td>56.3157895</td>
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<tr>
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<td>10944.757</td>
<td>-6588.7749</td>
<td>-5.0720695</td>
<td>-4.0099475</td>
<td>4.0418978</td>
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<tr>
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<td>15.3799822</td>
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<td>-2.22250015</td>
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<td>3.80550768</td>
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</table>
Appendix K: Description of Descent Profile

The Mars atmospheric entry can be divided into three logical segments. Separating the descent profile into segments allows for easier analysis and for clearer presentation of results. The first segment (de-orbit) goes from the parking orbit about Mars (LMO) to the edge of Mars atmosphere. Reaching from the edge of Mars atmosphere to a designated hover initiation altitude is the second segment (descent). During the descent segment, drag devices such as parachutes or rotofoils may be deployed to reduce the vehicle's velocity. Propulsive deceleration would also occur during this segment. Finally, the third segment (hover) covers the descent from hover initiation altitude to touchdown on Mars' surface. During the hover segment, the vehicle can travel to its final landing site and it is in this segment of the descent profile where the vehicle gets most of its crossrange distance.

During each of the three segments there are certain parameters of interest which are discussed below.

De-orbit segment: Of special interest during this segment is the ΔV required for de-orbit, and the velocity and flight path angle at atmospheric entry. The ΔV requirement is necessary for vehicle sizing, while entry velocity and flight path angle are inputs to the next segment of the descent.
Descent segment: During the descent segment, the first parameters of interest are the velocity and flight path angle at atmospheric entry as mentioned above. The aerodynamic qualities of the vehicle such as coefficient of drag, lift to drag ratio, and vehicle planform area (size) are also important. Related to the aerodynamic factors is the flight path angle of the vehicle as a function of altitude since the flight path angle will affect the drag and lift of the vehicle. Additional parameters of interest are the altitudes for deployment of any drag devices. The parameters mentioned above determine the velocity of the vehicle at the designated altitude for propulsive deceleration initiation. From this point, using the thrust output and fuel mass flow rate of the vehicle's engines, the velocity at the designated hover initiation point can be determined. Also, the amount of propellant used for propulsive deceleration can be determined; this is necessary for vehicle sizing. Two additional parameters of interest are the maximum stagnation temperature and maximum g-loading experienced by the vehicle during descent.

Hover segment: Of interest during the hover segment of the descent profile are the hover initiation altitude, the vehicle's mass at hover initiation, and the engine's thrust and fuel mass flow rate. From these parameters, the propellant used during hover can be determined.
Appendix L: DODV TK! Solver Model

**Purpose:** This TK! Solver model will be used to determine the flight path angle and velocity of landing vehicles at atmospheric entry as well as the $\Delta V$ required to de-orbit. The first two parameters are inputs to the Mars Descent program while the $\Delta V$ for de-orbit will be used to size the de-orbit engines and propellant tanks.

**Note on inputs and outputs:** The variables in a TK! Solver model can be defined as inputs or outputs depending on the objective of the analysis. The variables shown below are the ones which the current design team will use in their analysis.

**Inputs:**
- Orbital elements of parking orbit
- Orbital elements of transfer orbit
- True anomaly of vehicle at de-orbit burn point

**Outputs:**
- Flight path angle at atmospheric entry
- Vehicle velocity at atmospheric entry
- True anomaly of vehicle at atmospheric entry
- $\Delta V$ for de-orbit burn

**Source:** "A Manned Mission to Mars: Preliminary Design Review 2" Report written May 12, 1986 by Texas Space Services (TSS, a student design team) for an undergraduate mission design course at the University of Texas at Austin.

**Modifications made:** The model has been modified to incorporate a circular parking orbit

**Listing:** See below.

**Sample run:** See below.

**Verification:** Verification of this program was necessary since it was typed in to the TK! Solver program from the TSS report. This program has been verified by running data obtained from the TSS report as inputs and comparing the output obtained against the output published in the TSS report.
### VARIABLE SHEET

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<th><strong>St</strong></th>
<th><strong>Input</strong></th>
<th><strong>Name</strong></th>
<th><strong>Output</strong></th>
<th><strong>Unit</strong></th>
<th><strong>Comment</strong></th>
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<tr>
<td>42828</td>
<td>mu</td>
<td>6720</td>
<td>a</td>
<td>km</td>
<td>Major axis of parking orbit</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>e</td>
<td>km</td>
<td>Eccentricity of parking orbit</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Ra</td>
<td>km</td>
<td>Apoapsis of parking orbit</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Rp</td>
<td>km</td>
<td>Periapsis of parking orbit</td>
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<tr>
<td></td>
<td></td>
<td></td>
<td>f</td>
<td>deg</td>
<td>True anomaly of parking orbit</td>
</tr>
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<td></td>
<td></td>
<td>L 180</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Vi</td>
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<td></td>
<td></td>
<td>gammaI</td>
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<td>Initial flight path angle at burn</td>
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<td>km</td>
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<td></td>
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<td>deg</td>
<td>True anomaly of transfer orbit</td>
</tr>
<tr>
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<td></td>
<td></td>
<td>Vt</td>
<td>km/s</td>
<td>Velocity after burn</td>
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<td>186.72146 deg</td>
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<td>DeltaV</td>
<td>180.59345 m/s</td>
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<td>Dgamma</td>
<td>-6.072074 deg</td>
<td>Change in flight path angle at deorbit</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3495</td>
<td>re</td>
<td>km</td>
<td>Radius at atmospheric entry</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Ve</td>
<td>4233.1113 m/s</td>
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<tr>
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<td></td>
<td></td>
<td>fe</td>
<td>24.287816 deg</td>
<td>True anomaly at atmospheric entry</td>
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<tr>
<td></td>
<td></td>
<td>L</td>
<td>gammaE</td>
<td>7.7944227 deg</td>
<td>Flight path angle at entry</td>
</tr>
</tbody>
</table>

### RULE SHEET

\[
\begin{align*}
    r &= a \cdot \frac{1 - e^2}{1 + e \cdot \cos(f)} \\
    V_i &= \sqrt{\mu \cdot \frac{2}{r} - \frac{1}{a}} \\
    \tan(\text{gammaI}) &= e \cdot \sin(f) / (1 + e \cdot \cos(f)) \\
    \text{Rat} &= 2 \cdot at - Rpt \\
    \text{et} &= (\text{Rat} - \text{Rpt}) / (2 \cdot at) \\
    r &= at \cdot \frac{1 - e^2}{1 + e \cdot \cos(ft)} \\
    V_t &= \sqrt{\mu \cdot \frac{2}{r} - \frac{1}{at}} \\
    \tan(\text{gammaT}) &= et \cdot \sin(ftc) / (1 + et \cdot \cos(ftc)) \\
    ftc &= 2 \cdot \pi() - ft \\
    \Delta V &= \sqrt{V_i^2 + V_t^2 - 2 \cdot V_i \cdot V_t \cdot \cos(\text{gammaT} - \text{gammaI})} \\
    \text{Ra} &= 2 \cdot \frac{a - Rp}{(2 \cdot a)} \\
    e &= (\text{Ra} - \text{Rp}) / (2 \cdot a) \\
    Ve &= \sqrt{\mu \cdot \frac{2}{re} - \frac{1}{at}} \\
    re &= at \cdot \frac{1 - et^2}{1 + et \cdot \cos(fe)} \\
    \text{Dgamma} &= \text{gammaT} - \text{gammaI} \\
    \tan(\text{gammaE}) &= et \cdot \sin(fe) / (1 + et \cdot \cos(fe))
\end{align*}
\]
Appendix M: Mars Descent FORTRAN Program

**Purpose:** The main purpose of this program is to determine the propellant used to perform the atmospheric descent. This information will be used to size the landing vehicles. The secondary purpose of the program is to determine the maximum temperature experienced by the vehicle during the descent. This information will be used to design a thermal protection system for the landing vehicles (e.g.; thermal shield). The program will also determine the maximum g-loading experienced by the vehicle; this information will influence the structural design of the vehicle.

**Inputs:**
- Vehicle planform area (vehicle radius)
- Initial vehicle mass
- Vehicle coefficient of lift
- Vehicle lift to drag ratio
- Vehicle velocity at atmospheric entry
- Vehicle flight path angle at atmospheric entry
- Altitude of propulsive deceleration initiation
- Emissivity of vehicle heat shield
- Fuel mass flow rate
- Thrust of engine(s)
- Interval for propagating equations of motion

**Outputs:**
- Ballistic drag coefficient
- Time of descent
Maximum g-loading encountered
Maximum stagnation temperature encountered
Crossrange distance
Downrange distance
Altitude when vehicle achieves zero velocity
Propellant used during propulsive descent

Sources:


2) Current source: "A Manned Mission to Mars: Preliminary Design Review-2." Report written May 12, 1986 by Texas Space Services (TSS, a student design team) for an undergraduate mission design course at the University of Texas at Austin.

Modifications made:

1) The program was modified to read a data file of Mars atmospheric temperatures (see Appendix P) created by the current design team. This was necessary since a listing of the original data file was not provided with the program listing.

2) The program has been modified to create a data file for plotting a profile of the descent trajectory. The data file is formatted for use in the "PLOTTER" program on the CDC Dual Cybers computer at the University of Texas at Austin.

3) The program will be modified to incorporate parachute deployment during the descent trajectory. A previous design group
wrote a subroutine to perform this function, and it will be added to the current program.

4) Documentation is being added to the program. The original program was not documented and most variables were not defined.

**Theory:** The program uses a Runge-Kutta integrator to propagate the vehicle equations of motion through the descent trajectory.

**Listing:** See below

**Sample run:** See below

**Verification:** Verification of this program was necessary since it was typed in to a FORTRAN file from the TSS report. This program was verified by using data obtained from the TSS report as input and comparing the output obtained against the output published in the TSS report. The emissivity value that TSS used was not provided in their report so a value of 0.8 was chosen. The output obtained from this run was very close to the TSS output.
PROGRAM MARSDES

* (TTY,OUTPUT,DATA,TP,PLOT,TAPE3=OUTPUT,TAPE5=TTY,TAPE6=TTY,
  /   TAPE7=TP,TAPE9=DATA,TAPE8=PLOT)

* THIS IS MAIN ROUTINE OF THE MARS DESCENT VEHICLE SIMULATION WRITTEN
  * BY PRESTON CARTER. MODIFICATIONS AND ADDITIONS TO THE SIMULATION
  * WERE WRITTEN BY KYLE FIELDS AND MICHAEL ERGER. THESE ADDITIONS AND
  * MODIFICATIONS INCLUDE THE PROPULSIVE DECELERATION PHASE, AEROHEATING
  * ROUTINE, AND G-LOADING INFORMATION. THE CALCULATION UNITS ARE METERS,
  * KILOGRAMS, AND SECONDS.

* MODIFIED AND DOCUMENTED SPRING 1989 BY BRYAN POST. MODIFIED SUBROUTINE
  * TPDATA AND CHANGED PROGRAM TO RUN ON MICRO COMPUTER.

DEFINE VARIABLES

* HEQUIL = PULLOUT ALTITUDE (M)
* AMO -- INITIAL VEHICLE MASS (KG)
* RO = MARS MEAN RADIUS
* RHOO = MARS DENSITY AT GROUND LEVEL
* HMAX = MAX ALTITUDE
* E = EMISSIVITY OF ABLATIVE HEAT SHIELD
* BALLCL = BALLISTIC COEFFICIENT OF LIFT
* BALLCD = BALLISTIC COEFFICIENT OF DRAG

* X(1) = DOWNRANGE DISTANCE, X (M)
* X(2) = CROSSRANGE DISTANCE, Y (M)
* X(3) = DISTANCE FROM CENTER OF MARS TO VEHICLE (M)
* X(4) = VELOCITY (M/S)
* X(5) = FLIGHT PATH ANGLE
* X(6) = AZIMUTH (RAD)
* X(7) = MASS OF VEHICLE (KG)

* THE ARRAY DX(7) CONTAINS THE CHANGE IN EACH OF THE ELEMENTS OF
  * THE X(7) ARRAY

COMMON/RO/RO
COMMON/RHOO/RHOO
COMMON/HMAX/HMAX
COMMON/E/E
COMMON/BALLCL/BALLCL
COMMON/BALLCD/BALLCD
COMMON/G/G
COMMON/AMO/AMO
COMMON/HEQUIL/HEQUIL
COMMON/RB
COMMON/PROP/FFR,DM,PALT,THRUST
COMMON/TEMP/TEMP

* NECESSARY CHANGE

COMMON/ROLL/ROLL

DIMENSION X(7),DX(7)
DIMENSION TEMP(101)
REAL LD

142
* WRITE HEADING STATEMENTS TO DATA FILE FOR PLOTTING PROFILE
  * OF DESCENT TRAJECTORY
  *
  OPEN(8,FILE='PLOT',STATUS='NEW')
  WRITE (8,'*') TITLE # DESCENT TRAJECTORY #
  WRITE (8,'*') 'XLABEL # DOWNRANGE DISTANCE (KM) #'
  WRITE (8,'*') 'YLABEL # ALTITUDE (KM) #'
  WRITE (8,'*') 'LINE NOMARKERS'
* READ DATA FILE TO CREATE ARRAY OF ATMOSPHERIC TEMPS
  *
  CALL TPDATA(TEMP)
  *
  WRITE(6,'*') 'VERIFY TEMPS READ CORRECTLY'
  WRITE(6,'*') 'TEMP @ GROUND LEVEL = ',TEMP(0)
  WRITE(6,'*') 'TEMP @ 50 KM = ',TEMP(50)
  WRITE(6,'*') 'TEMP @ 100 KM = ',TEMP(100)
*
* NECESSARY CHANGE
*
 ROLL = 0.0
 DX(4) = 0.0
*
 GMAX = 0.0
 TWMAX = 0.0
*
 G = MARS GRAVITATIONAL ACCELERATION (M/S)
*
 G = 3.730
 RO = 3397500.0
 RHOO = 1.56E-2
*
 HMAX'S UNIT IS KILOMETERS INSTEAD OF METERS
*
 HMAX = 100.0
 DT = 1.0
*
* PRINT OPENING STATEMENTS
*
 CALL HOLA
 WRITE(6,'*') 'RADIUS OF VEHICLE (M)?'
 READ(5,*)RB
 WRITE(6,'*') 'MASS OF VEHICLE (KG)?'
 READ(5,*)AMO
 WRITE(6,'*') 'LIFT COEFFICIENT ?'
 READ(5,*)CL
 BALLCL = AMO/(CL*3.141592654*RB*RB)
 WRITE (6,'*') 'L/D?'
 READ (5,*)LD
 BALLCD = BALLCL*LD
 WRITE (6,'*') 'INITIAL HEIGHT IS 100 KM'
 WRITE (6,'*') 'H = 100000.0
 X(3) = H + RO
 WRITE (6,'*') 'INITIAL V (M/SEC)?'
 READ (5,*)X(4)
 WRITE (6,'*') 'INITIAL FLIGHT PATH ANGLE (DEG)?'
 READ (5,*)ANGLE

X(5) = 0.017453 * ANGLE
WRITE (6,'(A)') 'PULLOUT ALTITUDE (KM)?'
READ (5, *) HEQUIL
HEQUIL = HEQUIL * 1000.0
WRITE(6,*'(A)') 'EMMISIVITY OF ABLATIVE SURFACE ?'
READ(5, *) E
WRITE (6,*'(A)') 'OUTPUT TAPE NUMBER ?'
WRITE(6,*'(A)') 'USE TAPE 6 FOR TTY OUTPUT'
WRITE(6,*'(A)') 'USE TAPE 9 FOR OUTPUT DIRECTED TO FILE MARS.DAT'
WRITE(6,*'(A)') 'OUTPUT TAPE NUMBER ?'
READ (5, *) IUNIT

PROPULSIVE INFORMATION

WRITE(6,*'(A)') 'INITIAL PROPULSIVE DECELERATION ALTITUDE (KM)?'
READ(5, *) PALT
PALT = PALT * 1000.0
IF (PALT .EQ. 0.0) GOTO 333
WRITE(6,*'(A)') 'FUEL MASS FLOW RATE (KG/S) ?'
READ (5, *) FFR
WRITE (6,*'(A)') 'ENGINE THRUST (N) ?'
READ (5, *) THRUST

333 CONTINUE

DX(7) = 0.0
X(1) = 0.0
DX(4) = 0.0
X(2) = 0.0
X(6) = 0.0
X(7) = AMO

MAX TIME (S) FOR DESCENT

TMAX = 5000.0

TERMINATION ALTITUDE FOR DESCENT

TERMH = 0.0

WRITE(6,*'(A)') 'UPPER TRAJECTORY OUTPUT INTERVAL (S) ?'
READ (5, *) NSTEPS
WRITE(6,*'(A)') 'SIMULATION IN PROGRESS...'
TIME = 0.0
OPEN (9, FILE = 'MARS.DAT')
WRITE(IUNIT,'(A)') ' >>> MARS DESCENT SIMULATION <<<'
WRITE(IUNIT,'(A)') ' DESCENT PROFILE'
WRITE(IUNIT,'(A)') ' M/(CL*S) (KG/M**2) = ', BALLCL
WRITE(IUNIT,'(A)') 'INITIAL MASS (KG) = ', AMO
WRITE(IUNIT,'(A)') 'COEFFICIENT OF LIFT = ', CL
WRITE(IUNIT,'(A)') 'RADIUS OF VEHICLE (M) = ', RB
WRITE (IUNIT,'(A)') 'L/D = ', LD
WRITE (IUNIT,'(A)') 'EMMISIVITY OF ABLATIVE SURFACE = ', E
WRITE (IUNIT,'(A)') 'H (M) = ', H
WRITE (IUNIT,'(A)') 'V (M/SEC) = ', X(4)
WRITE (IUNIT,'(A)') 'GAMA (DEG) = ', ANGLE
WRITE (IUNIT,'(A)') 'THRUST (N) = ', THRUST

144
WRITE(IUNIT,*) 'INITIAL PROP DEC ALT (KM) = ', PALT/1000.0
WRITE(IUNIT,*) 'FUEL FLOW RATE (KG/S) = ', FFR
WRITE(IUNIT,*)
CALL OUTPUT(TIME, X, DX, TW, IUNIT)

LOOP FOR DESCENT BEGINS

H = X(3) - RO

200 IF ((TIME.LT.TMAX).AND.(H.GT.TERMH).AND.(X(4).GT.0.0)) THEN

SHORTEN PROPOGATION INCREMENT AT END OF TRAJECTORY

IF (X(3) - RO .LE. PALT + 7000.0) NSTEPS = 5
IF (X(3) - RO .LE. PALT + 2000.0) NSTEPS = 1
IF (X(3) - RO .LE. 100.0) NSTEPS = 1
IF (X(3) - RO .LE. 100.0) DT = .25
DO 300 I = 1, NSTEPS
   CALL RK(X, DX, DT, 7)
   TIME = TIME + DT
IF (DX(4) .LT. GMAX) GMAX = DX(4)
300 CONTINUE

CALL OUTPUT(TIME, X, DX, TW, IUNIT)
IF (TW.GT.TWMAX) TWMAX = TW
H = X(3) - RO
ELSE
   GMAX = GMAX/9.81
   WRITE(6,*)
   WRITE(6,*) 'TERMINATION TIME = ', TIME
   WRITE(6,*) 'TERMINATION ALTITUDE = ', H
   WRITE(6,*) 'MAXIMUM G-EARTH ACCELERATION = ', GMAX
   WRITE(6,*) 'MAXIMUM STAGNATION TEMPERATURE (K) = ', TWMAX
   WRITE(6,*) 'MASS OF FUEL USED (KG) = ', AMO-X(7)
   WRITE(6,*)
   IF (IUNIT.EQ.6) GOTO 444
   WRITE(IUNIT,*)
   WRITE(IUNIT,*) '---FINAL AND MAXIMUM VALUES---'
   WRITE(IUNIT,*) 'TERMINATION TIME = ', TIME
   WRITE(IUNIT,*) 'TERMINATION ALTITUDE = ', H
   WRITE(IUNIT,*) 'MAXIMUM G-EARTH ACCELERATION = ', GMAX
   WRITE(IUNIT,*) 'MAXIMUM STAGNATION TEMP (K) = ', TWMAX
   WRITE(IUNIT,*) 'MASS OF FUEL USED (KG) = ', AMO-X(7)
   WRITE(IUNIT,*)
   CHANGE TO PRINT FINAL VALUES TO SCREEN

444    CALL OUTPUT(TIME, X, DX, TW, 6)
*    GO TO 999
ENDIF
*  GO TO 200
*  LOOP FOR DESCENT ENDS
*  999 CONTINUE
*  STOP
SUBROUTINE OUTPUT(TIME, X, DX, TW, IUNIT)

* THIS IS AN OUTPUT ROUTINE FOR PRINTING AN EPHEMEROS OF THE DESCENT TRAJECTORY.

COMMON/RO/RO
COMMON/ROLL/ROLL
COMMON/PROP/FFR, DM, PALT, THRUST
COMMON/RHOO/RHOO

DIMENSION X(7), DX(7)

* CONVERT FROM RADIANS TO DEGREES

RADDEG = 57.29578

THETA = ROLL * RADDEG
DRG = X(1) / 1000.0
CRG = X(2) / 1000.0
H = (X(3) - RO) / 1000.0
V = X(4)
GFORCE = DX(4) / 9.81
GAMA = X(5) * RADDEG
AZE = X(6) * RADDEG

* THE AEROHEATING SUBROUTINE FRY IS CALLED FOR THE CURRENT VALUE OF STAGNATION TEMPERATURE IN DEGREES K

RHOF = DENS(X(3))
IF(H .LE. 1.0 .OR. V .LE. 1.0) RHOF = -9999.0
CALL FRY(X, RHOF, TW)
WRITE (IUNIT,*) ' ' WRITE (IUNIT,*) '_ TIME (SEC) = ', TIME,' ROLL (DEG) = ', THETA
IF (DX(7) .NE. 0.0) THEN
   WRITE (IUNIT,*) ' PROPULSION SYSTEMS ARE ON'
   WRITE (IUNIT,*) ' MASS OF VEHICLE (KG) = ', X(7)
ELSE
   WRITE (IUNIT,*) ' PROPULSION SYSTEMS ARE OFF'
ENDIF
WRITE (IUNIT,*) ' X DOWNRANGE (KM) = ', DRG
WRITE (IUNIT,*) ' Y CROSSRANGE (KM) = ', CRG
WRITE (IUNIT,*) ' H ALTITUDE (KM) = ', H
WRITE (IUNIT,*) ' V VELOCITY (M/SEC) = ', V
WRITE (IUNIT,*) ' GAMA FLT. PATH ANGLE (DEG) = ', GAMA
WRITE (IUNIT,*) ' AZE (DEG) = ', AZE
WRITE (IUNIT,*) ' ACCELERATION (EARTH G) = ', GFORCE
WRITE (IUNIT,*) ' STAGNATION TEMP (K) = ', TW

WRITE DATA TO PLOT FILE FOR PROFILE OF DESCENT TRAJECTORY

WRITE (8,*) DRG, H

RETURN
END
SUBROUTINE RK (X, DX, DT, N)

* THIS IS A RUNGE-KUTTA 4TH ORDER INTEGRATOR. THIS ROUTINE EXPECTS THE
* SUBROUTINE 'DERIV' TO BE SUPPLIED BY THE USER.

REAL X(7), U(7), F(7), D(7), DX(7)

CALL DERIV(X, D)
DO 1 I = 1, N
   D(I) = D(I) * DT
   U(I) = X(I) + 0.5 * D(I)
   CALL DERIV(U, F)
DO 2 I = 1, N
   F(I) = F(I) * DT
   D(I) = D(I) + 2.0 * F(I)
   U(I) = X(I) + 0.5 * F(I)
   CALL DERIV(U, F)
DO 3 I = 1, N
   F(I) = F(I) * DT
   D(I) = D(I) + 2.0 * F(I)
   U(I) = X(I) + F(I)
   CALL DERIV(U, F)
DO 4 I = 1, N
   X(I) = X(I) + (D(I) + F(I) * DT) / 6.0
   DX(4) = D(4)
   DX(7) = D(7)
RETURN
END

SUBROUTINE DERIV(X, DX)

* THIS SUBROUTINE CONTAINS THE EQUATIONS OF MOTION.

COMMON/BALLCL/BALLCL
COMMON/BALLCD/BALLCD
COMMON/G/G
COMMON/RO/RO
COMMON/AMO/AMO
COMMON/PROP/FFR, DM, PALT, THRUST
COMMON/ROLL/ROLL

DIMENSION X(7), DX(7)

* Q = (1/2) * DENSITY * (V^2)
* PALT = INITIAL PROPULSIVE DECELERATION ALTITUDE (KM)
* Q = 0.5 * DENS(X(3)) * X(4)^2
* HDOT = X(4) * SIN(X(5))
* DX(7) IS THE MASS FUEL FLOW RATE
* X(7) IS THE MASS OF THE VEHICLE
* THE FOLLOWING IF STATEMENT SAYS: IF CURRENT ALTITUDE IS LESS THEN
* PALT, AND VELOCITY IS GREATER THAN 30 M/S, AND CURRENT ALTITUDE IS
GREATER THAN 40 M, THEN.....

IF (X(3)-RO.LE.PALT.AND.X(4).GT.10.0.AND.X(3)-RO.GT.40.0) THEN
  DX(7) = -FFR
  PTHRUST = THRUST/X(7)
ELSE
  DX(7) = 0.0
  PTHRUST = 0.0
ENDIF

CALL CMROLL(X(7),X(3),X(4),X(6),HDOT,Q,ROLL)

DX(1) = X(4)*COS(X(6))*COS(X(5))
DX(2) = X(4)*SIN(X(6))*COS(X(5))
DX(3) = HDOT
DX(4) = -Q/(BALLCD*X(7)/AMO) + G*SIN(X(5)) - PTHRUST
DX(5) = Q/(BALLCL*X(7)/AMO)/X(4)*COS(ROLL)-G*X(4)*COS(X(5))
  +X(4)/X(3)*COS(X(5))
DX(6) = Q/(BALLCL*X(7)/AMO)/X(4)/COS(X(5))*SIN(ROLL)
RETURN
END

FUNCTION DENS(R)

THIS SUBROUTINE CONTAINS AN ANALYTICAL MODEL OF THE MARTIAN
ATMOSPHERE. THIS MODEL WAS DEVELOPED AT JPL FROM A BEST FIT OF THE
VIKING I AND II FLIGHT DATA.

COMMON/RHOO/RHOO
COMMON/HMAX/HMAX
COMMON/RO/RO

H = (R - RO)/1000.0
IF (H.EQ.0.0) THEN
  DENS = RHOO
ELSE IF ((H.GT.0.0).AND.(H.LE.50.0)) THEN
  DENS = RHOO*EXP(-(-0.5314+0.1083*H+2.188/H))
ELSE IF ((H.GT.50.0).AND.(H.LE.HMAX)) THEN
  DENS = RHOO*EXP(-(-2.881+0.1396*H+42.55/H))
ELSE IF (H.GT.HMAX) THEN
  DENS = 0.0
ENDIF
RETURN
END

SUBROUTINE CMROLL(W,R,V,AZE,HDOT,Q,ROLL)

THIS SUBROUTINE CONTROLS THE ROLL OF THE VEHICLE DURING
DESCENT. FOR THIS SIMULATION, THE VEHICLE'S LIFT IS MODULATED
BY THE VEHICLE'S BANK ANGLE. THIS SIMULATION HAS ASSUMED
CONSTANT L/D AND ANGLE OF ATTACK. THIS SUBROUTINE
IMPLEMENTS ALL OF DESCENT TRAJECTORY PROFILE REQUIREMENTS.
SPECIFICALLY, THIS ROUTINE CONTROLS THE VEHICLE’S RATE OF
DESCENT AND FLIGHT AZIMUTH ACCORDING TO OUR SPECIFICATIONS.

COMMON/BALLCL/BALLCL
COMMON/G/G
COMMON/RO/RO
COMMON/HEQUIL/HEQUIL
COMMON/AMO/AMO

Q = 1/2 * DENSITY * (V**2)
HEQUIL = PULLOUT ALTITUDE (M) - AT PULLOUT ALTITUDE, VEHICLE ROTATED
     TO ACHIEVE LEVEL TRAJECTORY (EQUILIBRIUM GLIDE)
G = MARS GRAVITATIONAL ACCELERATION (M/S)
W = CURRENT MASS OF VEHICLE (KG)
AMO = INITIAL MASS OF VEHICLE (KG)
V = VELOCITY (M/S)
H = ALTITUDE ABOVE SURFACE

FIRST TIME THROUGH, ROLL = 0.0

H = R - RO

ESTABLISH POS OR NEG SCALAR OF MAGNITUDE ONE

IF (ROLL.EQ.0.0) THEN
   SGN = 1.0
ELSE
   SGN = ROLL/ABS(ROLL)
ENDIF

IF (Q.EQ.0.0) THEN
   IF VELOCITY EQUAL ZERO, ROLL = 0.0
   ROLL = 0.0
ELSEIF ((H.LT.HEQUIL).AND.(HDOT.LT.0.0)) THEN
   IF BELOW PULLOUT ALTITUDE AND DESCENDING, ROLL = 0.0
   ROLL = 0.0
ELSEIF (H.GT.HEQUIL) THEN
   IF ABOVE PULLOUT ALTITUDE, ROLL -- 90 DEG
   ROLL = ACOS(0.0)
ELSE
   IF EQUAL TO OR BELOW PULLOUT ALTITUDE AND NOT DESCENDING
   COSEQG = ABS(G*(BALLCL*W/AMO)/Q*(1.0 - V**2/(G*R)))
   IF (COSEQG.GT.1.0) THEN
      ROLL = ACOS(0.0)
   ELSE
      ROLL = ACOS(COSEQG)
   ENDIF
ENDIF
ELSE
   IF AZE GT 90 DEG
IF (AZE.GT.1.57079) THEN
   ROLL = -1.0*ROLL*SGN
ENDIF

RETURN
END

SUBROUTINE FRY(X,RHOF,TW)

C
C    THIS IS THE AEROHEATING SUBROUTINE WHICH USES AN EQUATION FOR
C    CONVECTIVE HEATING TO CALCULATE THE STAGNATION TEMPERATURE.
C    THE EQUATION (PG 6:33, EQ 6:11) IS FROM CORNING'S AEROSPACE VEHICLE
C    DESIGN. IT IS ASSUMED THAT RADIATIVE HEATING EFFECTS WILL BE NEGLIGIBLE
C    SINCE THE VEHICLE WILL BE FLYING MUCH SLOWER THAN 10,000 M/S.
C
COMMON RB
COMMON/RO/RO
COMMON/E/E
COMMON/RHOO/RHOO
COMMON/TEMP/TEMP

DIMENSION TEMP(101)
DIMENSION X(7)

C
C    SEVERAL CONVERSIONS ARE NECESSARY SINCE EQUATION USES THE
C    ARCHAIC SYSTEM OF UNITS.
C
IF(RHOF .EQ. -9999.0)THEN
   TW = 0.0
   GOTO 555
ENDIF

* RBF = RADIUS OF VEHICLE (FT)
* E = EMISSIVITY OF ABLATIVE HEAT SHIELD
* RBF = RB*3.2808
* VC = 10831.5
* SBK = .48E-12
* ONE = 17600.0/SQRT(RBF)
* TWO = SQRT(RHOF/RHOO)
* THREE = (X(4)/VC)**3.25
* ONE * TWO * THREE = CONVECTIVE HEATING RATE
* TW4 = ONE*THREE*SBK/E
* TW4 = ABS(TW4)
* TW = SQRT(SQRT(TW4))

C
C    TW IS CONVERTED FROM RANKINE TO KELVIN
C
   TW = TW/1.8

555 CONTINUE

* DETERMINE ALTITUDE AS AN INTEGER (KM)
IH = INT((X(3)-RO)/1000.0)
IF(IH .LT. 1)IH=1

TT = TEMP OF ATMOSPHERE AT THE DETERMINED ALTITUDE (KELVIN)

TT = TEMP(IH)

TOTAL STAGNATION TEMP EQUAL SUM OF TEMP DUE TO HEATING AND
ATMOSPHERIC TEMP

TW = TW + TT
RETURN
END

SUBROUTINE HOLA
WRITE(6,*)
WRITE(6,*)
WRITE(6,*) 'MARS DESCENT SIMULATION <<<
WRITE(6,*)
WRITE(6,*) THIS PROGRAM INCORPORATES AERODYNAMIC DRAG'
WRITE(6,*) 'FOR BRAKING IN THE UPPER ATMOSPHERE AND'
WRITE(6,*) 'PROPULSIVE DECELERATION FOR THE FINAL LANDING'
WRITE(6,*) 'PHASE. IT IS SUGGESTED THAT THE PROPULSIVE'
WRITE(6,*) 'PHASE OF THE PROGRAM BE TURNED OFF FOR'
WRITE(6,*) 'INITIAL ANALYSES TO OBTAIN AERODYNAMIC'
WRITE(6,*) 'ENTRY DATA. THIS IS PERFORMED BY SETTING THE'
WRITE(6,*) 'INITIAL PROPULSIVE ALTITUDE EQUAL TO ZERO.'
WRITE(6,*) 'NOTE THAT THE PROPULSION SYSTEM WILL'
WRITE(6,*) 'AUTOMATICALLY SHUT OFF AT AN ALTITUDE OF'
WRITE(6,*) '40 METERS AND A VELOCITY OF 30 METERS/SECOND.'
WRITE(6,*) 'ALSO NOTE THAT THE OUTPUT INTERVAL WILL'
WRITE(6,*) 'BE RESET AS THE VEHICLE ENTERS THE PROPULSIVE'
WRITE(6,*) 'PHASE OF THE DESCENT.'
WRITE(6,*)
RETURN
END

SUBROUTINE TPDATA(TEMP)

C
C > READS TEMPERATURE FROM GROUND LEVEL TO 100 KM INTO THE ARRAY TEMP
C > EACH TEMP(I) CORRESPONDS TO ATMOSPHERIC TEMP AT A GIVEN ALTITUDE
C > ABOVE SURFACE IN KM. [TEMP(0) = TEMP @ GROUND, TEMP(100) = TEMP @ 100 KM]
C > THE DATA FILE CONTAINING THE TEMPS SHOULD BE TITLED TP.DAT AND SHOULD
C > BE IN THE CURRENT DIRECTORY WHEN RUNNING PROGRAM. EACH TEMP SHOULD
C > BE ON ITS OWN LINE.
C > TEMPERATURE IS IN DEGREES K.
C
DIMENSION TEMP(101)
OPEN(7,FILE = 'TP.DAT')

DO 69 I = 0,100
   READ(7,*) TEMP(I)
69 CONTINUE

RETURN
END
>>> MARS DESCENT SIMULATION <<<

DESCENT PROFILE

\[
\frac{M}{(CL*S)} (KG/M^{*2}) = 392.9752 \\
\text{INITIAL MASS (KG)} = 200000.0 \\
\text{COEFFICIENT OF LIFT} = 0.5000000 \\
\text{RADIUS OF VEHICLE (M) = 18.00000} \\
\text{L/D} = 0.5000000 \\
\text{EMMISIVITY OF ABLATIVE SURFACE = 0.8000000} \\
\text{H (M)} = 100000.0 \\
\text{V (M/SEC) = 3500.000} \\
\text{GAMA (DEG) = -0.5000000} \\
\text{THRUST (N) = 800000.0} \\
\text{INITIAL PROP. DEC. ALT. (KM) = 2.800000} \\
\text{FUEL FLOW RATE (KG/S) = 387.0000}
\]

\[
\text{TIME (SEC)} = 0.0000000E+00 \text{ ROLL (DEG)} = 0.0000000E+00 \\
\text{PROPULSION SYSTEMS ARE OFF} \\
\text{X DOWNRANGE (KM) = 0.0000000E+00} \\
\text{Y CROSSRANGE (KM) = 0.0000000E+00} \\
\text{H ALTITUDE (KM) = 100.0000} \\
\text{V VELOCITY (M/SEC) = 3500.000} \\
\text{GAMA FLT. PATH ANGLE (DEG) = -0.4999916} \\
\text{AZE (DEG) = 0.0000000E+00} \\
\text{ACCELERATION (EARTH G) = 0.0000000E+00} \\
\text{STAGNATION TEMP (K) = 601.8546}
\]

\[
\text{TIME (SEC)} = 50.00000 \text{ ROLL (DEG)} = 90.00000 \\
\text{PROPULSION SYSTEMS ARE OFF} \\
\text{X DOWNRANGE (KM) = 174.9382} \\
\text{Y CROSSRANGE (KM) = 3.3079055E-03} \\
\text{H ALTITUDE (KM) = 98.18800} \\
\text{V VELOCITY (M/SEC) = 3497.792} \\
\text{GAMA FLT. PATH ANGLE (DEG) = -0.6872597} \\
\text{AZE (DEG) = 2.2592656E-03} \\
\text{ACCELERATION (EARTH G) = -2.5907286E-02} \\
\text{STAGNATION TEMP (K) = 616.0103}
\]

\[
\text{TIME (SEC)} = 100.0000 \text{ ROLL (DEG)} = 90.00000 \\
\text{PROPULSION SYSTEMS ARE OFF} \\
\text{X DOWNRANGE (KM) = 349.7416} \\
\text{Y CROSSRANGE (KM) = 1.4536121E-02} \\
\text{H ALTITUDE (KM) = 95.80200} \\
\text{V VELOCITY (M/SEC) = 3494.881} \\
\text{GAMA FLT. PATH ANGLE (DEG) = -0.8771243} \\
\text{AZE (DEG) = 5.2628322E-03} \\
\text{ACCELERATION (EARTH G) = -3.3402905E-02} \\
\text{STAGNATION TEMP (K) = 635.2839}
\]

153
TIME (SEC) = 150.0000  ROLL (DEG) = 90.00000
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 524.3716
Y CROSSRANGE (KM) = 3.6715209E-02
H ALTITUDE (KM) = 92.83500
V VELOCITY (M/SEC) = 3491.187
GAMA FLT. PATH ANGLE (DEG) = -1.070368
AZE (DEG) = 9.5788063E-03
ACCELERATION (EARTH G) = -4.1945171E-02
STAGNATION TEMP (K) = 660.2806

TIME (SEC) = 200.0000  ROLL (DEG) = 90.00000
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 698.7833
Y CROSSRANGE (KM) = 7.5269416E-02
H ALTITUDE (KM) = 89.27725
V VELOCITY (M/SEC) = 3486.566
GAMA FLT. PATH ANGLE (DEG) = -1.267950
AZE (DEG) = 1.6286368E-02
ACCELERATION (EARTH G) = -5.2453484E-02
STAGNATION TEMP (K) = 691.8087

TIME (SEC) = 250.0000  ROLL (DEG) = 90.00000
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 872.9216
Y CROSSRANGE (KM) = 0.1403229
H ALTITUDE (KM) = 85.11600
V VELOCITY (M/SEC) = 3480.738
GAMA FLT. PATH ANGLE (DEG) = -1.471177
AZE (DEG) = 2.7567940E-02
ACCELERATION (EARTH G) = -6.6925675E-02
STAGNATION TEMP (K) = 730.9097

TIME (SEC) = 300.0000  ROLL (DEG) = 90.00000
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 1046.712
Y CROSSRANGE (KM) = 0.2518087
H ALTITUDE (KM) = 80.33475
V VELOCITY (M/SEC) = 3473.115
GAMA FLT. PATH ANGLE (DEG) = -1.682054
AZE (DEG) = 4.8118457E-02
ACCELERATION (EARTH G) = -8.9969561E-02
STAGNATION TEMP (K) = 778.8838

TIME (SEC) = 350.0000  ROLL (DEG) = 90.00000
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 1220.031
Y CROSSRANGE (KM) = 0.4514129
H ALTITUDE (KM) = 74.91275
V VELOCITY (M/SEC) = 3462.376
GAMA FLT. PATH ANGLE (DEG) = -1.904130
AZE (DEG) = 8.8682912E-02
ACCELERATION (EARTH G) = -0.1328440
STAGNATION TEMP (K) = 837.2604

154
TIME (SEC) = 400.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 1392.653
Y CROSSRANGE (KM) = 0.8262587
H ALTITUDE (KM) = 68.82325
V VELOCITY (M/SEC) = 3445.344
GAMA FLT. PATH ANGLE (DEG) = -2.123071
AZE (DEG) = 0.1539382
ACCELERATION (EARTH G) = -0.2239716
STAGNATION TEMP (K) = 908.5958

TIME (SEC) = 450.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 1564.111
Y CROSSRANGE (KM) = 1.286918
H ALTITUDE (KM) = 62.32200
V VELOCITY (M/SEC) = 3414.629
GAMA FLT. PATH ANGLE (DEG) = -2.200480
AZE (DEG) = 0.1539382
ACCELERATION (EARTH G) = -0.4235397
STAGNATION TEMP (K) = 990.8253

TIME (SEC) = 500.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 1733.387
Y CROSSRANGE (KM) = 1.741717
H ALTITUDE (KM) = 55.90625
V VELOCITY (M/SEC) = 3354.497
GAMA FLT. PATH ANGLE (DEG) = -2.097143
AZE (DEG) = 0.1539382
ACCELERATION (EARTH G) = -0.8365697
STAGNATION TEMP (K) = 1073.561

TIME (SEC) = 550.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 1898.452
Y CROSSRANGE (KM) = 2.185205
H ALTITUDE (KM) = 50.39375
V VELOCITY (M/SEC) = 3241.172
GAMA FLT. PATH ANGLE (DEG) = -1.659117
AZE (DEG) = 0.1539382
ACCELERATION (EARTH G) = -1.481478
STAGNATION TEMP (K) = 1132.896

TIME (SEC) = 600.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2056.301
Y CROSSRANGE (KM) = 2.609302
H ALTITUDE (KM) = 46.85225
V VELOCITY (M/SEC) = 3066.996
GAMA FLT. PATH ANGLE (DEG) = -0.8737836
AZE (DEG) = 0.1539382
ACCELERATION (EARTH G) = -1.959821
STAGNATION TEMP (K) = 1142.830
TIME (SEC) = 650.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2204.756
Y CROSSRANGE (KM) = 3.008162
H ALTITUDE (KM) = 45.59425
V VELOCITY (M/SEC) = 2871.847
GAMA FLT. PATH ANGLE (DEG) = -0.1352376
AZE (DEG) = 0.1539382
ACCELERATION (EARTH G) = -1.947137
STAGNATION TEMP (K) = 1108.416

TIME (SEC) = 700.0000  ROLL (DEG) = 14.90563
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2343.767
Y CROSSRANGE (KM) = 4.022000
H ALTITUDE (KM) = 45.55125
V VELOCITY (M/SEC) = 2692.589
GAMA FLT. PATH ANGLE (DEG) = 2.1998362E-04
AZE (DEG) = 0.8024611
ACCELERATION (EARTH G) = -1.715977
STAGNATION TEMP (K) = 1060.395

TIME (SEC) = 750.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2474.336
Y CROSSRANGE (KM) = 5.966340
H ALTITUDE (KM) = 45.39675
V VELOCITY (M/SEC) = 2533.491
GAMA FLT. PATH ANGLE (DEG) = -0.1985256
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.550063
STAGNATION TEMP (K) = 1018.561

TIME (SEC) = 800.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2597.240
Y CROSSRANGE (KM) = 7.805716
H ALTITUDE (KM) = 44.42075
V VELOCITY (M/SEC) = 2383.488
GAMA FLT. PATH ANGLE (DEG) = -0.7694979
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.540059
STAGNATION TEMP (K) = 988.6731

TIME (SEC) = 850.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2712.478
Y CROSSRANGE (KM) = 9.530395
H ALTITUDE (KM) = 42.11150
V VELOCITY (M/SEC) = 2224.271
GAMA FLT. PATH ANGLE (DEG) = -1.552287
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.738116
STAGNATION TEMP (K) = 969.9622

156
TIME (SEC) = 900.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2819.046
Y CROSSRANGE (KM) = 11.12530
H ALTITUDE (KM) = 38.49950
V VELOCITY (M/SEC) = 2034.570
GAMA FLT. PATH ANGLE (DEG) = -2.318061
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -2.152049
STAGNATION TEMP (K) = 955.4421

TIME (SEC) = 950.0000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2914.944
Y CROSSRANGE (KM) = 12.56052
H ALTITUDE (KM) = 34.13525
V VELOCITY (M/SEC) = 1797.109
GAMA FLT. PATH ANGLE (DEG) = -2.855952
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -2.674899
STAGNATION TEMP (K) = 927.6208

TIME (SEC) = 1000.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 2997.752
Y CROSSRANGE (KM) = 13.79983
H ALTITUDE (KM) = 29.76225
V VELOCITY (M/SEC) = 1514.467
GAMA FLT. PATH ANGLE (DEG) = -3.190088
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.025215
STAGNATION TEMP (K) = 874.9666

TIME (SEC) = 1050.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3065.858
Y CROSSRANGE (KM) = 14.81910
H ALTITUDE (KM) = 25.66925
V VELOCITY (M/SEC) = 1215.117
GAMA FLT. PATH ANGLE (DEG) = -3.826387
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.024169
STAGNATION TEMP (K) = 797.6786

TIME (SEC) = 1100.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3119.174
Y CROSSRANGE (KM) = 15.61703
H ALTITUDE (KM) = 21.42500
V VELOCITY (M/SEC) = 928.0188
GAMA FLT. PATH ANGLE (DEG) = -5.632490
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -2.819056
STAGNATION TEMP (K) = 710.1951
TIME (SEC) = 1150.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3158.529
Y CROSSRANGE (KM) = 16.20601
H ALTITUDE (KM) = 16.37550
V VELOCITY (M/SEC) = 663.5558
GAMA FLT. PATH ANGLE (DEG) = -9.721332
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -2.575090
STAGNATION TEMP (K) = 619.7341

TIME (SEC) = 1200.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3184.886
Y CROSSRANGE (KM) = 16.60049
H ALTITUDE (KM) = 10.20000
V VELOCITY (M/SEC) = 425.3934
GAMA FLT. PATH ANGLE (DEG) = -18.37585
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -2.269154
STAGNATION TEMP (K) = 523.8298

TIME (SEC) = 1250.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3199.272
Y CROSSRANGE (KM) = 16.81578
H ALTITUDE (KM) = 3.241000
V VELOCITY (M/SEC) = 224.3853
GAMA FLT. PATH ANGLE (DEG) = -38.09363
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.801588
STAGNATION TEMP (K) = 410.2795

TIME (SEC) = 1251.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3199.447
Y CROSSRANGE (KM) = 16.81839
H ALTITUDE (KM) = 3.102750
V VELOCITY (M/SEC) = 220.8779
GAMA FLT. PATH ANGLE (DEG) = -38.68984
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.788969
STAGNATION TEMP (K) = 407.4116

TIME (SEC) = 1252.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3199.617
Y CROSSRANGE (KM) = 16.82094
H ALTITUDE (KM) = 2.964750
V VELOCITY (M/SEC) = 217.3958
GAMA FLT. PATH ANGLE (DEG) = -39.29659
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.776069
STAGNATION TEMP (K) = 404.6983
TIME (SEC) = 1253.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3199.783
Y CROSSRANGE (KM) = 16.82342
H ALTITUDE (KM) = 2.827250
V VELOCITY (M/SEC) = 213.9397
GAMA FLT. PATH ANGLE (DEG) = -39.91404
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.762880
STAGNATION TEMP (K) = 401.7347

TIME (SEC) = 1254.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 199677.5
X DOWNRANGE (KM) = 3199.945
Y CROSSRANGE (KM) = 16.82583
H ALTITUDE (KM) = 2.691000
V VELOCITY (M/SEC) = 207.1860
GAMA FLT. PATH ANGLE (DEG) = -40.54808
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.377459
STAGNATION TEMP (K) = 396.3417

TIME (SEC) = 1255.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 199290.5
X DOWNRANGE (KM) = 3200.098
Y CROSSRANGE (KM) = 16.82814
H ALTITUDE (KM) = 2.557750
V VELOCITY (M/SEC) = 199.8178
GAMA FLT. PATH ANGLE (DEG) = -41.90250
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.757976
STAGNATION TEMP (K) = 390.4316

TIME (SEC) = 1256.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 198903.5
X DOWNRANGE (KM) = 3200.245
Y CROSSRANGE (KM) = 16.83033
H ALTITUDE (KM) = 2.427750
V VELOCITY (M/SEC) = 192.4969
GAMA FLT. PATH ANGLE (DEG) = -41.90250
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.733562
STAGNATION TEMP (K) = 384.4771
TIME (SEC) = 1257.000    ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 198516.5
X DOWNRANGE (KM) = 3200.385
Y CROSSRANGE (KM) = 16.83243
H ALTITUDE (KM) = 2.300750
V VELOCITY (M/SEC) = 185.2186
GAMA FLT. PATH ANGLE (DEG) = -42.62490
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.711651
STAGNATION TEMP (K) = 378.4723

TIME (SEC) = 1258.000    ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 198129.5
X DOWNRANGE (KM) = 3200.518
Y CROSSRANGE (KM) = 16.83441
H ALTITUDE (KM) = 2.177000
V VELOCITY (M/SEC) = 177.9776
GAMA FLT. PATH ANGLE (DEG) = -43.37884
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.692316
STAGNATION TEMP (K) = 372.4140

TIME (SEC) = 1259.000    ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 197742.5
X DOWNRANGE (KM) = 3200.644
Y CROSSRANGE (KM) = 16.83630
H ALTITUDE (KM) = 2.056500
V VELOCITY (M/SEC) = 170.7688
GAMA FLT. PATH ANGLE (DEG) = -44.16545
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.675702
STAGNATION TEMP (K) = 366.2976

TIME (SEC) = 1260.000    ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 197355.5
X DOWNRANGE (KM) = 3200.762
Y CROSSRANGE (KM) = 16.83808
H ALTITUDE (KM) = 1.939250
V VELOCITY (M/SEC) = 163.5865
GAMA FLT. PATH ANGLE (DEG) = -44.98599
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.661853
STAGNATION TEMP (K) = 360.2181
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<th>Y CROSSRANGE (KM)</th>
<th>H ALTITUDE (KM)</th>
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<th>GAMA FLT. PATH ANGLE (DEG)</th>
<th>AZE (DEG)</th>
<th>ACCELERATION (EARTH G)</th>
<th>STAGNATION TEMP (K)</th>
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PROPUlSION SYSTEMS ARE ON  
MASS OF VEHICLE (KG) = 195420.5  
X DOWNRANGE (KM) = 3201.258  
Y CROSSRANGE (KM) = 16.84549  
H ALTITUDE (KM) = 1.405000  
V VELOCITY (M/SEC) = 127.8704  
GAMA FLT. PATH ANGLE (DEG) = -49.64908  
AZE (DEG) = 0.8574303  
ACCELERATION (EARTH G) = -3.637360  
STAGNATION TEMP (K) = 328.2483  

TIME (SEC) = 1266.000  ROLL (DEG) = 0.0000000E+00  
PROPUlSION SYSTEMS ARE ON  
MASS OF VEHICLE (KG) = 195033.5  
X DOWNRANGE (KM) = 3201.337  
Y CROSSRANGE (KM) = 16.84669  
H ALTITUDE (KM) = 1.309500  
V VELOCITY (M/SEC) = 120.7239  
GAMA FLT. PATH ANGLE (DEG) = -50.70667  
AZE (DEG) = 0.8574303  
ACCELERATION (EARTH G) = -3.641807  
STAGNATION TEMP (K) = 321.6288  

TIME (SEC) = 1267.000  ROLL (DEG) = 0.0000000E+00  
PROPUlSION SYSTEMS ARE ON  
MASS OF VEHICLE (KG) = 194646.5  
X DOWNRANGE (KM) = 3201.411  
Y CROSSRANGE (KM) = 16.84778  
H ALTITUDE (KM) = 1.218250  
V VELOCITY (M/SEC) = 113.5618  
GAMA FLT. PATH ANGLE (DEG) = -51.81149  
AZE (DEG) = 0.8574303  
ACCELERATION (EARTH G) = -3.649398  
STAGNATION TEMP (K) = 314.9400  

TIME (SEC) = 1268.000  ROLL (DEG) = 0.0000000E+00  
PROPUlSION SYSTEMS ARE ON  
MASS OF VEHICLE (KG) = 194259.5  
X DOWNRANGE (KM) = 3201.478  
Y CROSSRANGE (KM) = 16.84879  
H ALTITUDE (KM) = 1.131250  
V VELOCITY (M/SEC) = 106.3781  
GAMA FLT. PATH ANGLE (DEG) = -52.96680  
AZE (DEG) = 0.8574303  
ACCELERATION (EARTH G) = -3.660098  
STAGNATION TEMP (K) = 308.1862
| TIME (SEC) | 1269.000 | ROLL (DEG) | 0.0000000E+00 |
| MASS OF VEHICLE (KG) | 193372.5 |
| X DOWNRANGE (KM) | 3201.539 |
| Y CROSSRANGE (KM) | 16.84970 |
| H ALTITUDE (KM) | 1.048500 |
| V VELOCITY (M/SEC) | 99.16702 |
| GAMA FLT. PATH ANGLE (DEG) | -54.17653 |
| AZE (DEG) | 0.8574303 |
| ACCELERATION (EARTH G) | -3.673796 |
| STAGNATION TEMP (K) | 301.3739 |

| TIME (SEC) | 1270.000 | ROLL (DEG) | 0.0000000E+00 |
| MASS OF VEHICLE (KG) | 193485.5 |
| X DOWNRANGE (KM) | 3201.594 |
| Y CROSSRANGE (KM) | 16.85052 |
| H ALTITUDE (KM) | 0.9705000 |
| V VELOCITY (M/SEC) | 91.92293 |
| GAMA FLT. PATH ANGLE (DEG) | -55.44547 |
| AZE (DEG) | 0.8574303 |
| ACCELERATION (EARTH G) | -3.690337 |
| STAGNATION TEMP (K) | 213.9000 |

| TIME (SEC) | 1271.000 | ROLL (DEG) | 0.0000000E+00 |
| MASS OF VEHICLE (KG) | 193098.5 |
| X DOWNRANGE (KM) | 3201.643 |
| Y CROSSRANGE (KM) | 16.85126 |
| H ALTITUDE (KM) | 0.8972500 |
| V VELOCITY (M/SEC) | 84.64061 |
| GAMA FLT. PATH ANGLE (DEG) | -56.77961 |
| AZE (DEG) | 0.8574303 |
| ACCELERATION (EARTH G) | -3.709568 |
| STAGNATION TEMP (K) | 213.9000 |

| TIME (SEC) | 1272.000 | ROLL (DEG) | 0.0000000E+00 |
| MASS OF VEHICLE (KG) | 192711.5 |
| X DOWNRANGE (KM) | 3201.687 |
| Y CROSSRANGE (KM) | 16.85191 |
| H ALTITUDE (KM) | 0.8290000 |
| V VELOCITY (M/SEC) | 77.31531 |
| GAMA FLT. PATH ANGLE (DEG) | -58.18653 |
| AZE (DEG) | 0.8574303 |
| ACCELERATION (EARTH G) | -3.731250 |
| STAGNATION TEMP (K) | 213.9000 |
TIME (SEC) = 1273.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 192324.5
X DOWNRANGE (KM) = 3201.725
Y CROSSRANGE (KM) = 16.85248
H ALTITUDE (KM) = 0.7660000
V VELOCITY (M/SEC) = 69.94274
GAMA FLT. PATH ANGLE (DEG) = -59.67597
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.755144
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1274.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 191937.5
X DOWNRANGE (KM) = 3201.757
Y CROSSRANGE (KM) = 16.85297
H ALTITUDE (KM) = 0.7085000
V VELOCITY (M/SEC) = 62.51918
GAMA FLT. PATH ANGLE (DEG) = -61.26079
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.780960
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1275.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 191550.5
X DOWNRANGE (KM) = 3201.785
Y CROSSRANGE (KM) = 16.85338
H ALTITUDE (KM) = 0.6565000
V VELOCITY (M/SEC) = 55.04144
GAMA FLT. PATH ANGLE (DEG) = -62.95829
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.808433
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1276.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 191163.5
X DOWNRANGE (KM) = 3201.807
Y CROSSRANGE (KM) = 16.85372
H ALTITUDE (KM) = 0.6105000
V VELOCITY (M/SEC) = 47.50683
GAMA FLT. PATH ANGLE (DEG) = -64.79256
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.837300
STAGNATION TEMP (K) = 213.9000
TIME (SEC) = 1277.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 190776.5
X DOWNRANGE (KM) = 3201.825
Y CROSSRANGE (KM) = 16.85399
H ALTITUDE (KM) = 0.5707500
V VELOCITY (M/SEC) = 39.91306
GAMA FLT. PATH ANGLE (DEG) = -66.79855
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.867347
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1278.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 190389.5
X DOWNRANGE (KM) = 3201.839
Y CROSSRANGE (KM) = 16.85419
H ALTITUDE (KM) = 0.5372500
V VELOCITY (M/SEC) = 32.25818
GAMA FLT. PATH ANGLE (DEG) = -69.03004
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.898411
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1279.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 190002.5
X DOWNRANGE (KM) = 3201.848
Y CROSSRANGE (KM) = 16.85434
H ALTITUDE (KM) = 0.5105000
V VELOCITY (M/SEC) = 24.54037
GAMA FLT. PATH ANGLE (DEG) = -71.57729
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.930399
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1280.000  ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 189615.5
X DOWNRANGE (KM) = 3201.854
Y CROSSRANGE (KM) = 16.85443
H ALTITUDE (KM) = 0.4907500
V VELOCITY (M/SEC) = 16.75782
GAMA FLT. PATH ANGLE (DEG) = -74.61418
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.963319
STAGNATION TEMP (K) = 213.9000
TIME (SEC) = 1281.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 189293.0
X DOWNRANGE (KM) = 3201.857
Y CROSSRANGE (KM) = 16.85447
H ALTITUDE (KM) = 0.4782500
V VELOCITY (M/SEC) = 9.613024
GAMA FLT. PATH ANGLE (DEG) = -78.57708
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.997340
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1282.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3201.858
Y CROSSRANGE (KM) = 16.85449
H ALTITUDE (KM) = 0.4705000
V VELOCITY (M/SEC) = 5.932095
GAMA FLT. PATH ANGLE (DEG) = -82.97979
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.873953
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1283.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3201.859
Y CROSSRANGE (KM) = 16.85450
H ALTITUDE (KM) = 0.4665000
V VELOCITY (M/SEC) = 2.216182
GAMA FLT. PATH ANGLE (DEG) = -87.38210
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -1.892900
STAGNATION TEMP (K) = 213.9000

TIME (SEC) = 1284.000 ROLL (DEG) = 90.00000
PROPULSION SYSTEMS ARE OFF
X DOWNRANGE (KM) = 3201.859
Y CROSSRANGE (KM) = 16.85450
H ALTITUDE (KM) = 0.4662500
V VELOCITY (M/SEC) = -1.512772
GAMA FLT. PATH ANGLE (DEG) = -91.84517
AZE (DEG) = 0.8575678
ACCELERATION (EARTH G) = -1.900684
STAGNATION TEMP (K) = 213.9000

---------- FINAL AND MAXIMUM VALUES ----------
TERMINATION TIME = 1284.000
TERMINATION ALTITUDE = 466.2500
MAXIMUM G-EARTH ACCELERATION = -3.997340
MAXIMUM STAGNATION TEMP (K) = 1142.830
MASS OF FUEL USED (KG) = 10707.00
Appendix N: Landing Gear Stress FORTRAN Program

**Purpose:** This FORTRAN program was used to determine the stress experienced by the legs of the landing gear with a given vehicle mass.

**Note on inputs and outputs:**

**Inputs:**
- Number of legs in landing gear system
- Gravitational acceleration
- Total vehicle mass

**Outputs:**
- Stress experience by leg for a given range of theta

**Source:** Written by the Atmospheric Vehicle Team of the Star Truk Company in Spring 1989.

**Modifications made:** N/A

**Theory:** Standard structural mechanics theory. A good reference is: 

**Listing:** See below

**Sample run:** See below

**Verification:** N/A
PROGRAM STRESS
C ******************************************************************************
C  * written by Michael J. Luneau
C  * spring 89 ASE-274L
C  * STAR TRUK Enterprises
C ******************************************************************************
C
REAL LENGTH, PI, LOAD
C
OPEN (8, FILE='LEGS.DAT', STATUS='NEW')
C
PRINT*, 'TYPE IN #OF LEGS'
READ(5,*) NUMBER
C
WRITE(8,7) NUMBER
7 FORMAT( 3X, I2, ' LEGS')
C
PI=ACOS(-1.)
LOAD=400000.0
GRAV=.38*9.81
C
DO 77 LENGTH=1.5, 3.0, .5
C
WRITE(8,3) LENGTH
3 FORMAT( 3X, 'LEG LENGTH = ', F3.1 )
C
DO 88 DEG=45, 90, 5
WRITE(8,4) DEG
4 FORMAT( 3X, 'THETA = ', F3.0, ' DEGREES' )
C
THETA=PI*DEG/180.
C
WRITE(8,2)
2 FORMAT( 5X, 'DIAMETER', 3X, 'STRESS')
C
DO 99 D=.20, .60, .05
C
AXSTRESS=(LOAD*GRAV/NUMBER*SIGN(THETA))/(PI*(D/2)**2)
C
BESTRESS=(LOAD*GRAV/NUMBER*COS(THETA)*LENGTH*D/2)/(PI*D**4/64.)
C
TOSTRESS=AXSTRESS+BESTRESS
C
WRITE(8,1) D, TOSTRESS
1 FORMAT( 5X, F4.2, 6X, E13.5 )
99 CONTINUE
C
88 CONTINUE
C
77 CONTINUE
C
END
5 LEGS

LEG LENGTH = 1.5

THETA = 45.DEGREES

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178
Appendix O: CW Equation TK! Solver Model

**Purpose:** This TK! Solver model was used to determine the ΔV required by the OMV to transfer a given payload between the ITV and the Phobos surface.

**Note on inputs and outputs:** The variables in a TK! Solver model can be defined as inputs or outputs depending on the objective of the analysis. The variables shown below are the ones which the current design team used in their analysis.

**Inputs:**

For both trajectory targeting and propagation:
- gravitational acceleration \( g_e \)
- planet radius \( r_e \)
- orbit radius \( r_o \)

For targeting:
- initial and final positions \( (x_0, z_0, x, z) \) respectively
- time of flight \( t \)

For orbit propagation:
- an initial state \( (x_0, z_0, v_{x0}, v_{z0}) \)
- time of flight \( t \)

**Outputs:**

- ΔV's
- propagation of state

**Source:** This model was created by the Orbital Vehicle Team of Star Truk Company in April 1989.

**Modifications made:** N/A
Theory: The CW Equations are a set of linearized equations of motion for an orbiting vehicle referenced to a local-vertical local-horizontal (LVLH) orbiting coordinate system. The x-axis is forward (tangent to the orbit path), the positive z-axis is radially downward while the y-axis completes the right-handed system.

The equations consist of an initial state vector, (position and velocity), multiplied by a propagation matrix to arrive at a new state. The components of the propagation matrix are functions of time, planet surface acceleration and radius, and origin orbit radius. The equations can be modified in two ways. One, given an initial state and time, the position and velocity for a later time can be determined. Two, assuming an initial and final position and a TOF, the initial velocity required to complete the trajectory can be determined. This is called targeting.

The accuracy of the CW equations is limited by the following assumptions used to linearize the set:

1. The distance traveled, relative to the orbiting reference origin, (i.e. ITV or Phobos), must be small—much less than the orbit radius of 9408 kilometers.
2. The transfer time of flight must be on the order of one orbit revolution or less if possible.
3. The main engine thrust time must be short compared to the transfer time.

Listing: See below

Sample run: See Appendix J
Verification: Hand calculations were performed to verify the program. A number of test points were calculated and substantiated the results of the model.
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<td>-180.0875</td>
<td>Equation Parameter</td>
<td></td>
<td></td>
</tr>
<tr>
<td>T1</td>
<td>11000</td>
<td>Equation Parameter</td>
<td></td>
<td></td>
</tr>
<tr>
<td>b12</td>
<td>17300.742</td>
<td>Equation Parameter</td>
<td></td>
<td></td>
</tr>
<tr>
<td>T2</td>
<td>-6000</td>
<td>Equation Parameter</td>
<td></td>
<td></td>
</tr>
<tr>
<td>b21</td>
<td>-17300.74</td>
<td>Equation Parameter</td>
<td></td>
<td></td>
</tr>
<tr>
<td>b11</td>
<td>-42042.35</td>
<td>Equation Parameter</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
RULE SHEET

S Rule

"Propogation of relative state equations:

" Equations used to produce the position of item;

\[
\begin{align*}
\dot{x} &= -(2*vzo/w) * c + (4*vxo/w-6*z0) * s + (6*z0-3*vxo/w) * w*t + (x0+2*vzo/w) \\
\dot{z} &= (2*vxo/w-3*z0) * c + (vzo/w) * s + (4*z0-2*vxo/w) \\
\dot{z} &= -z
\end{align*}
\]

\[
\begin{align*}
\dot{y} &= (4*vxo/w-3*z0) * w*c + 2*vzo*s + (6*z0-3*vxo/w) * w \\
\dot{s} &= -(2*vxo/w-3*z0) * w*s + vzo*c
\end{align*}
\]

* \( w = \sqrt{(g/e*(r^2 - (r-\alpha^2)))} \)
* \( s = \sin(w*t) \)
* \( c = \cos(w*t) \)

* \( delvo = \sqrt{(vxo^2 + vzo^2)} \)
* \( delv = \sqrt{(v^2 + vzo^2)} \)
* \( delvtot = delvo + delv \)

" Equations used to produce the initial change in velocity for targeting an object given a time of flight.

\[
\begin{align*}
vxo &= detb*(b22*T1 - b12*T2) \\
vzo &= detb*(-b21*T1 + b11*T2) \\
T1 &= x - c - (-6*s+6*w*t)*z0 \\
T2 &= z - (-2*c+4*z0)
\end{align*}
\]

\[
\begin{align*}
b11 &= ((4/w)*s-3*t) \\
b12 &= ((-2/w)*c+2/w) \\
b21 &= ((2/w)*c-2/w) \\
b22 &= s/w
\end{align*}
\]

\[
\begin{align*}
detb &= 1/(b11*b22 - b12*b21)
\end{align*}
\]
Appendix P: Mars Atmospheric Model

**Purpose:** The purpose of this program is to model the Martian atmosphere from the surface to approximately 150 km.

**Description:** The atmospheric model is a FORTRAN subroutine. A driver program is written to print the desired output parameters calculated by the subroutine.

**Inputs:** Not applicable; however, the driver program must be written to print desired outputs.

**Outputs:**

1) Temperature, pressure, density, and viscosity in 1 km increments

2) Data files for plotting temperature, pressure, density, and coefficient of viscosity versus altitude

3) Speed of sound, gravitational acceleration, molecular weight, molecular scale temperature, pressure scale height, density scale height, refractive index, zenith angle from ground station, columnar mass along the slant path, total path length, vertical temperature gradient, and number density.

**Source:** David Pitts at Space & Life Sciences Division, NASA Johnson Space Center

**Modifications made:** The driver program which was obtained with the subroutine was modified to print the outputs desired by the current design team.

**Theory:** Mathematical modelling based on results obtained from data collected.
Listing: See below

Sample run: See below

Verification: The subroutine was obtained from NASA in magnetic format and the only modifications made were to the driver program; therefore, verification is not required.
PROGRAM MARS1  
C EXAMPLE MAIN PROGRAM CALLING MODATM WHICH IS SET UP FOR MARS ATM  
C WRITTEN BY DAVID E. PITTS, NASA JSC, HOUSTON TX 1/25/88  
C WITH MOLECULAR WEIGHT VARIABILITY AND NEGATIVE ALTITUDE ADDED  
C  
DIMENSION ANS(35)  
CHARACTER*4 TEST  
OPEN (6, FILE='ATMOUT.DAT', STATUS='NEW')  
OPEN (11, FILE='TEMP.DAT', STATUS='NEW')  
OPEN (12, FILE='PRES.DAT', STATUS='NEW')  
OPEN (13, FILE='DENS.DAT', STATUS='NEW')  
OPEN (14, FILE='VISC.DAT', STATUS='NEW')  
OPEN (5, FILE='ATMMDAT.DAT', STATUS='OLD')  
XLAMDA=.5  
TEST='ALTI'  
ANS(1)=-1.0  
P=1.0  
CALL MODATM(1.0, P, TEST, XLAMDA, ANS, RE, CONN)  
C THE ABOVE STATEMENTS ARE USED FOR INITIALIZATION OF THE ROUTINES  
C  
DO 1 I=1,160  
Z=I-2  
CALL MODATM(Z, P, TEST, XLAMDA, ANS, RE, CONN)  
C  
WRITE (6,20) Z, ANS(2), ANS(1), ANS(3), ANS(8)  
WRITE (11,79) ANS(2), Z  
WRITE (12,79) ANS(1), Z  
WRITE (13,79) ANS(3), Z  
WRITE (14,79) ANS(8), Z  
1 CONTINUE  
79 FORMAT (2X, E10.3, 5X, F6.1)  
20 FORMAT (15X, F6.1, 5X, F7.2, 5X, E10.3, 5X, E10.3, 5X, 0P9F8.3)  
c The following is an example of how to use subroutine path  
c to find the refraction by the atmosphere and the zenith angle  
c from the surface to a satellite in orbit about the planet  
c ZS=1000.0  
c PHIS=0.0  
c THETAS=4.9  
c ZL=0.0  
c PHIL=0.0  
c THETAL=0.0  
c CALL PATH (XLAMDA, ZS, PHIS, THETAS, ZL, PHIL, THETAL, ANS, RE, CONN,  
c 1SUM, SUM1, SUM3, SUM4)  
CLOSE(5)  
CLOSE(6)  
STOP  
END  
C SUBROUTINE MODATM (Z, P, TEST, XLAMDA, ANS, RE, CONN)  
C  
DIMENSION H(100), P(100), T(100), ANS(35), TM(100)
1,VOLPER(15),AM(10),RAT(10),VIS(10)
REAL MM(100)
CHARACTER*4 TEST
DATA RO/8.31432E+07/
  VI(T1,III,IQF)=26.693*VIS(II)*SQRT(T1/RAT(III))/(OMEG(T1,III,IQF))
C
C*********************************************************************************
C
C Z IS IN KM, PP IS IN MB
C ANS( ) ARE OUTPUT VARIABLES
C XLAMDA IS THE WAVELENGTH IN MICRONS FOR WHICH YOU ARE CALCULATING
C ATMOSPHERIC REFRACTION
C IF TEST .EQ. 'PRES' THEN PRESSURE IS USED AS HEIGHT INDICATOR
C IF TEST.NE. 'PRES' THEN GEOMETRIC ALTITUDE (KM) IS HEIGHT INDICATOR
C YOU MUST SET ANS(1)=-1.0 BEFORE ENTERING THE SUBROUTINE THE FIRST TIME
C SO THAT THE INPUT DATA WILL BE READ IN.
C RO IS THE UNIVERSAL GAS CONSTANT BASED ON THE CARBON 12 ATOMIC WEIGHT
C SCALE IN ERGS/(DEG KELVIN-GM-MOLE)
C XMO IS MOLECULAR WEIGHT OF ATMOSPHERE
C RE = THE MEAN RADIUS OF THE PLANET IN KM
C G IS ACCELERATION OF GRAVITY AT 0 EQUIPOTENTIAL SURFACE LEVEL GIVEN IN
C CM/SEC**2
C CONN IS A CONSTANT GIVEN AS -M*G*100.0/RO WHERE M IS MOLECULAR WEIGHT
C AND G AND R ARE DESCRIBED ABOVE, CONN = DEG K/m.
C VOLPER(I) IS THE VOLUME PERCENTAGE OF GAS/100.0 FOR GAS I
C AM(I) IS THE MOLECULAR WEIGHT OF GAS I
C H(I) IS GEOPOTENTIAL ALTITUDE IN KM ABOVE THE MEAN EQUIPOTENTIAL SURFACE
C P(I) IS SIGNIFICANT LEVELS OF PRESSURE IN MB
C T(I) IS KINETIC TEMPERATURE SIGNIFICANT LEVELS IN DEG K
C TM(I) IS MOLECULAR SCALE TEMPERATURE SIGNIFICANT LEVELS IN DEG K
C MM(I) IS MOLECULAR WEIGHT SIGNIFICANT LEVELS IN GM/(GM-MOLE)
C*********************************************************************************

CT=288.15/273.16+1.0
IF(ANS(1).GE.0.0) GO TO 15
ANS(1)=0.0
CALL INPUT (P,T,H,TM,M,XMO,ANS,RE,CONN,G,VOLPER,AM,RAT,VIS)
15 IF (TEST.EQ.'PRES') GO TO 7
HA= RE*Z/(RE+Z)
C HA IS GEOPOTENTIAL ALTITUDE IN KM
23 DO 11 I=1,M
 II=1
 IF(H(I)-HA) 11,12,13
11 CONTINUE
 DO 4511 III=1,16
4511 ANS(III)=0.0
 ANS(17)=1.0
 ANS(18)=1.0
 RETURN
C ABOVE SETS ALL ATMOSPHERIC PARAMETERS=0 IF ALTITUDE > H(M)
9 II=M
13 I=II-1
 DH=H(I+1)-H(I)
 D=(TM(I+1)-TM(I))/DH
 DTDZ=D*DH/(RE*H(I+1)/(RE-H(I+1))-(RE*H(I)/(RE-H(I)))))
 W=(T(I+1)-T(I))/DH
 DH=H(I)-HA

187
C
C*******************************************************************************
C
C HEIGHT 'H' IS IN km
C HEIGHT 'Z' IS IN km
C
C ANS (1) IS PRESSURE
C PRESSURE IS IN MB
C ANS (2) IS TEMPERATURE
C TEMPERATURE IS IN DEG KELVIN
C
C ANS( 3) IS DENSITY
C DENSITY IS IN GM/CC
C
C ANS( 4) IS SPEED OF SOUND
C SPEED OF SOUND IS IN M/S
C
C ANS( 5) IS ACCELERATION OF GRAVITY
C ACCELERATION OF GRAVITY IS IN CM/SEC**2
C
C ANS(6) IS MOLECULAR SCALE TEMPERATURE (DEG K)
C
C ANS( 7) IS MOLECULAR WEIGHT
C
C ANS( 8) IS COEFFICIENT OF VISCOSITY
C VISCOSITY IS IN KG/(M SEC)
C
C ANS(15) IS PRESSURE SCALE HEIGHT
C PRESSURE SCALE HEIGHT IS IN KM
C
C ANS(16) IS DENSITY SCALE HEIGHT
C
C DENSITY SCALE HEIGHT IS IN KM
C
C ANS(17) IS REFRACTIVE INDEX DEVELOPED BY EDLEN IN TERMS OF WAVELENGTH ALONE
C INDEX IS FOR AIR AT 288 DEG KELVIN AND 760MM HG
C
C ANS(18) IS REFRACTIVE INDEX DEVELOPED BY PENNDORF IN TERMS OF
C WAVELENGTH, TEMPERATURE, AND PRESSURE
C
C ANS(21) IS THE ZENITH ANGLE FROM GROUNDSTATION IN RADIANS
C
C ANS(22) = THE TOTAL GM/CM**2 OR COLUMNAR MASS ALONG THE SLANT PATH.
C
C ANS(24) = TOTAL PATH LENGTH IN CM
C
C ANS(25) IS VERTICAL TEMPERATURE GRADIENT , DEG KELVIN/(M)
C
C ANS(21) THRU ANS(24) ARE CALCULATED IN SUBROUTINE PATH.
C
C ANS(26) IS MOLECULAR WEIGHT
C
C ANS(27) IS NUMBER DENSITY IN PARTICLES/CM**3
C
C*******************************************************************************

ANS(2) = T(I) - W*DH
ANS(6) = TM(I) - D*DH
ANS(1) = PRES(P(I), D, TM(I), ANS(6), DH, ANS, RE, CONN)
GO TO 14

12 I = I + 1
ANS(1) = P(I)
ANS(2) = T(I)
ANS(6) = TM(I)

14 ANS(5) = G*(RE/(RE+Z))**2
        ANS(3) = ANS(1)*XMO/(RO*ANS(6))*1000.0
        ANS(7) = XMO*ANS(2)/ANS(6)

C ********
        TX = ANS(2)
        CCPP = 0.0
        CCVV = 0.0
        DO 2100 IQV = 1, 10
        CCPP = CCPP + CPS(TX, IQV, 1)*VOLPER(IQV)
        2100 CCVV = CCVV + (CPS(TX, IQV, 1) - 1.9862)*VOLPER(IQV)
        GAMMA = CCPP/CCVV
        ANS(4) = 0.01*SQRT(GAMMA*RO*TX/ANS(7))
        XMU = 0.0
        DO 1001 III = 1, 10
        IF (VOLPER(III)) 1005, 1001, 1005
        1005 SUMB1 = 0.0
        DO 1002 JJJ = 1, 10
        IF (VOLPER(JJJ)) 1191, 1002, 1191
        1191 IF (III - JJJ) 1003, 1002, 1003
        1003 PHI1J = (1.0 + SQRT(VI(TX, III, 1)/VI(TX, JJJ, 1)))*(AM(JJJ)/AM(III))**2/(2.82842712*SQRT(1.0 + AM(III)/AM(JJJ)))
        SUMB1 = SUMB1 + PHI1J*VOLPER(JJJ)/VOLPER(III)
        1002 CONTINUE
        XMU = XMU + VI(TX, III, 1)/(1.0 + SUMB1)
        1001 CONTINUE
        XMU = XMU*1.0
        ANS(8) = XMU

C ABOVE EQUATION GIVES VISCOSITY KM/(M SEC)*E+05 FOR GAS MIXTURE
C ********
        ANS(25) = W

        ANS(15) = RO*ANS(6)/(XMO*ANS(5))*1.0E-05
        ANS(16) = ANS(15)/(1.0 + ANS(15)*DTDZ/ANS(6))
        ANS(26) = XMO*ANS(2)/ANS(6)
        ANS(27) = 6.02257E+23*ANS(1)**10.0/(8.31432E+07*ANS(2))

C ********
        ANS(19) = VOLPER(8)*ANS(1)
C ANS(19) = THE WATER VAPOR PRESSURE IN MB, USED IN REFRACTIVE INDEX
C ********

        IF (XLAMDA.GE.12500.00) GO TO 30
C THIS MEANS IF XLAMDA IS .GE. 1.25 CM USE MICROWAVE REFRACTIVITY
        ANS(17) = 1.0 + 1.0E-08*(6432.8+2949810.)/(146.1/(XLAMDA**2)) + 25540.1/(41.1/(XLAMDA**2))
        ANS(18) = 1.0 + (ANS(17) - 1.0)*(CT/(1.0 + ANS(2)/273.16))**2/ANS(1)/1013.25
        GO TO 31
        30 ANS(18) = 1.0 + 1.0E-06*(77.6*ANS(1)/(ANS(2)) + 373000.0*ANS(19)/(ANS(2)**2))
        ANS(17) = ANS(18)
        31 RETURN
7 DO 16 I=1,M
C THE FOLLOWING SECTION USES PRESSURE AS INDEPENDENT VARIABLE
C AND RETURNS ALTITUDE Z
    I=I
    IF (PP.GT.P(I)) GO TO 16
    IF(PP-P(I)) 16,41,17
16 CONTINUE
    HA=H(I)
42 DO 10 1=2,35
    IF (I.EQ.21.OR.I.EQ.22.OR.I.EQ.23.OR.I.EQ.24) GO TO 10
    ANS(I)=0.0
10 CONTINUE
    ANS(17)=1.0
    ANS(18)=1.0
    Z=HA
    RETURN
41 Z=H(I)**RE/((RE-H(I)))
    GO TO 12
17 I=I-1
    D=TM(I+1)-TM(I)
    IF(D) 20,21,20
20 D=CONN**1000.0*ALOG(P(I+1)/P(I))*ALOG(TM(I+1)/TM(I))
    ANS(6)=TM(I)**(PP/P(I))**D/(CONN**1000.0)
    HA=H(I)+(ANS(6)-TM(I))/D
    GO TO 22
21 HA=H(I)+TM(I)*(PP/P(I))*ALOG(PP/P(I))/(CONN**1000.0)
22 HA=HA
    Z=HA*RE/((RE-HA))
    GO TO 23
END

C ........................................................................
C SUBROUTINE INPUT (P,T,H,TM,MQ,XMO,ANS,RE,CONN,G0,VOLPER,AM
1,RAT,VIS)
DIMENSION P(100),T(100),H(100),TM(100),ANS(35),AM(15)
1,RAT(10),VIS(10),VOLPER(15),Z(100)
REAL MM(100)
CHARACTER*5 ID(8)
DATA RO/8.31432E+07/
C ........................................................................
C
TX=200.0
J=1
I=0
X=CPS(TX,J,I)
X=OMEG(TX,J,I)
C ABOVE INITIALIZES ARRAYS IN FUNCTIONS CPS AND OMEG
AM(1)=28.016
AM(2)=44.011
AM(3)=32.0
AM(4)=39.944
AM(5)=20.183
AM(6)=2.016
AM(7)=4.003
AM(8)=18.016
AM(9)=28.011
AM(10)=64.066
RAT(1)=91.5
RAT(2)=200.0
RAT(3)=100.0
RAT(4)=119.5
RAT(5)=27.5
RAT(6)=38.0
RAT(7)=10.22
RAT(8)=498.2
RAT(9)=94.5
RAT(10)=252.0
VIS(1)=3.737
VIS(2)=6.01
VIS(3)=4.620
VIS(4)=5.90
VIS(5)=2.884
VIS(6)=1.030
VIS(7)=0.964
VIS(8)=10.597
VIS(9)=3.803
VIS(10)=6.91
H(1)=0.0
READ (5,9,END=6177) P(1),T(1),G0,RE
9 FORMAT (6X,E11.5,1=5.1 ,F6.1,F6.0)
9 READ (5,3,END=6177) (VOLPER(I),I=1,10)
3 FORMAT (12X,10F5.2)
MM(1)=0.0
DO 2924 I=1,10
MM(1)=MM(1)+AM(I)*VOLPER(I)
2924 CONTINUE
C
C INPUT DATA DEFINING THE ATMOSPHERIC STRUCTURE

11 MQ=0
DO 12 I=1,100
C THIS IS THE FORMAT FOR READING SIGNIFICANT LEVELS IN NON-CODED FORM
C WHERE H(I) = METERS, P(I) = MB, T(I) = DEG K, MM(I)=MOLECULAR WTG
976 FORMAT (I 1 ,A1,7X,3(F10.2,10X))
IF (I .EQ. 1) CONN=-MM(1)*G0*100.0/RO
IF (I .EQ. 1) XMO=-MM(1)
IF (I .GT. 1) GO TO 3826
WRITE (6,5)
5 FORMAT (2X,23HCONSTRUCTION PARAMETERS,27X,16HSCIENTIFIC UNITS,35X,
1/) WRITE (6,22) P(1),T(1),MM(1),G0
22 FORMAT (2X,19HSURFACE PRESSURE = ,F9.2,3H MB,10X,22HSURFACE TEMPER
1ATURE = ,F7.2,2H K, 10X,19HMOMOLECULAR WEIGHT =
3,5X,0PF6.3,2X,18HSURFACE GRAVITY = ,0PF8.3,11H CM/SEC/SEC)
WRITE (6,556) RE,(VOLPER(K),K=1,10)
556 FORMAT (2X,9HRADIUS OF ' MARS = ',F8.2,4H(KM),12X,18HPERCENT NITR
10GEN = ,5X,2PF7.3,11X,18HPERCENT CO2 = ,2PF7.3,2X,22HPERCENT O
2XYGEN = ,2PF7.3,12X,23HPERCENT ARGON = ,2PF7.3,11X, 316HPERCENT NEON = 2PF7.3/2X,22HPERCENT HYDROGEN = ,2PF7.3, 412X,23HPERCENT HELIUM = ,2PF7.3,11X,16HPERCENT WATER = , 52PF7.3/2X,22HPERCENT CO = ,2PF7.3,12X, 623HPERCENT SO2 = , 72PF7.3//,10X,46HTEMPERATURE AND MOLECULAR WEIGHT DISTRIBUTION /

C

WRITE(6,25)
25 FORMAT (44X'MODEL ATMOSPHERE'// 1.,42X//.29X, 2'THE SIGNIFICANT LEVELS FOR THE MODEL ATMOSPHERE ARE-' 3//,27X,'ALT',10X,'PRES',10X,'TEMP',9X,'TM',11X 4,'MOL WTG',/27X,'(M)',10X,'(MB)',10X,'(K)' 510X,'(K)',10X,'(K)' )

C*************************************************************************

C

3826 IF (IPO) 630,631,630
C CONVERT Z(I) FROM GEOM TO GEOPOTENTIAL ALTITUDE
631 H(I)=Z(I)*RE/(RE+Z(I))
GOTO 782
630 H(I)=Z(I)
C CONVERT H(I) FROM GEOP TO GEOMETRIC ALTITUDE
Z(I)=RE*Z(I)/(RE-Z(I))
782 TM(I)=T(I)*XMO/MM(I)
IF (I.GT. 1) P(I)=PRES(P(I-1),(TM(I)-TM(I-1))/(H(I)-H(I-1)), 1TM(I-1),TM(I),H(I-1)-H(I),ANS,RE,CONN)
MQ=I
12 CONTINUE
C*************************************************************************

C

2 DO 26 I=MQ,100
H(I)=H(MQ)
26 P(I)=P(MQ)
DO 6 I=1,MQ
C
WRITE (6,32) H(I)*RE/(RE-H(I)),P(I),T(I), 1TM(I),MM(I)
32 FORMAT (20X,1P2E13.3,0P3F13.3)
C
35 CONTINUE
6 CONTINUE
6177 WRITE (6,86)
86 FORMAT (//)
WRITE(6,9901)
9901 FORMAT( 3X,'ALT',7X,'TEMP',7X,'PRESSURE',7X,'DENSITY',7X, 1'COEFFV',/) RETURN
END

C*************************************************************************

C

SUBROUTINE REFRAC (Z1,Z2,XLAMDA,PHI,PHIPR,PSI,SLANT,ANS,RE,CONN)
DIMENSION ANS(35)
C
C IN ORDER TO CALCULATE A CONTINUOUS PATH YOU MUST EXTERNALLY SET
PHI=PHIPR
C Z1, Z2, PHI, AND XLAMDA ARE INPUT VARIABLES
C Z1 AND Z2 ARE IN KM AND XLAMDA IS IN MICRONS
C PHIPR, PSI, AND SLANT ARE OUTPUT VARIABLES
C PHI, PHIPR, AND PSI ARE IN RADIANS AND SLANT IS IN CM
C IF YOU WANT AMOUNT OF GM/CM**2 (COLUMNAR MASS) OF ATMOSPHERE FROM Z1 TO
Z2
C USE ANS(3)*SLANT.
C
S1=RE+Z1
S2=RE+Z2
DELT=(Z2-Z1)/2.0
CALL MODATM(Z2+DELT,PP,'ALTI',XLAMDA,ANS,RE,CONN)
D2=ANS(3)
XN2=ANS(18)
CALL MODATM (Z1+DELT,PP,'ALT',XLAMDA,ANS,RE,CONN)
D1=ANS(3)
XN1=ANS(18)
PSI=SININV(S1*SIN(PHI)/S2)
PHIPR=SININV(S1*SIN(PHI)*XN1/(S2*XN2))
SLANT=S1*SIN(PHI-PSI)/SIN(PSI)*1.0E+05
RETURN
END

C
C SUBROUTINE PATH (XLAMDA,ZS,PHIS,THETAS,ZL,PHIL,THETAL,ANS
1,RE,CONN,SUM,SUM1,SUM3,SUM4)
DIMENSION ANS(35),A(3,3),B(3),C(3),PATHM(30),ZZZ(31)
C
C QUANTITIES ENDING IN S ARE FOR THE SATELLITE
C QUANTITIES ENDING IN L ARE FOR THE GROUND LOCAL
C Q1- AND -Q2- ARE DUMMY VARIABLES
C XS, YS, AND HS- ARE THE RECTANGULAR COORDINATES OF THE SPACECRAFT
C XL, YL, AN THE RECTANGULAR COORDINATES OF THE GROUND LOCAL
C THE ANGLE ABD IS THE ANGLE BETWEEN THE SUBSATELLITE POINT AND TARGET.
C ANGLE ABD IS FOUND BY USING THE DOT PRODUCT AND TAKING THE INVERSE COS
C .0092833 RADIANS IS THE TOTAL REFRACTION ON A PASS THRU U.S. STANDARD
C 'SUM' IS THE TOTAL ANGLE CHANGE DURING REFRACTION
C 'SUM1' IS THE SUM OF ALL DELTA XI CALCULATED BY LAW OF SINES
C 'SUM3' IS THE TOTAL COLUMNAR MASS IN THE SLANT PATH
C 'SUM4' IS THE TOTAL SLANT PATH IN CM
C PHI IS IN RADIANS
C
C ANS(21) IS THE ZENITH ANGLE FROM GROUNDSTATION IN RADIANS
C
C ANS(22) = THE TOTAL GM/CM**2 OR COLUMNAR MASS ALONG THE SLANT PATH.
C
C ANS(24) = TOTAL PATH LENGTH IN CM
C
193
C K1=30
PI=3.14159265
CON=.0174532925
PHIS=PHIS*CON
THETAS=THETAS*(-CON)
PHIL=PHIL*CON
THETAL=THETAL*(-CON)
CALL MODATM(ZS,PBAR,'ALTI',XLAMDA,ANS,RE,CONN)
DELP=ANS(1)
PSAT=ANS(1)
TSAT=ANS(2)
CALL MODATM(ZL,PBAR,'ALTI',XLAMDA,ANS,RE,CONN)
DELP=ABS(ANS(1)-DELP)/FLOAT(K1)
PSURF=ANS(1)
TSURF=ANS(2)
ZZZ(KI+1)=ZS
ZZZ(1)=ZL
Q1=RE+ZS
Q2=COS(PHIS)
XS=Q1*COS(THETAS)*Q2
YS=Q1*SIN(THETAS)*Q2
HS=Q1*SIN(PHIS)
Q2=COS(PHIL)
Q1=RE+ZL
XL=Q1*COS(THETAL)*Q2
YL=Q1*SIN(THETAL)*Q2
HL=Q1*SIN(PHIL)
ABD=COSINV(((XS*XL)+(YS*YL)+(HS*HL))/(SQRT(XS**2+YS**2+HS**2)
1*SQRT(XL**2+YL**2+HL**2)))
DO 3 I=1,3
3 C(I)=0.0
C FROM HERE TO STATEMENT 4 FINDS THE VECTOR (C) FROM THE TARGET TO THE
SATELLITE
A(1,1)=SIN(PHIL)*COS(THETAL)
A(2,1)=-SIN(THETAL)
A(3,1)=COS(PHIL)*COS(THETAL)
A(1,2)=SIN(PHIL)*SIN(THETAL)
A(2,2)=COS(THETAL)
A(3,2)=COS(PHIL)*SIN(THETAL)
A(1,3)=-COS(PHIL)
A(2,3)=0.0
A(3,3)=SIN(PHIL)
B(1 )=XS-XL
B(2 )=YS-YL
B(3 )=HS-HL
DO 4 I=1,3
4 C(I)=A(I,M)*B(M)+C(I)
PHIL=PHIL/CON
THETAL=THETAL/(-CON)
PHIS=PHIS/CON
THETAS=THETAS/(-CON)
PHI=ATAN2(SQRT(C(1)**2+C(2)**2+C(3)**2))
IF (PHI.GT..017)PHI=PHI-.0092833
IF (PHI/CON.GT.90.0)WRITE (6,88)
88 FORMAT (///,1X,'WARNING,ZENITH ANGLE OF UNREFRACTED PATH EXCEEDS

194
190.0 DEG', 1X, 'IT IS HIGHLY PROBABLE THAT THE AIRCRAFT OR SPACE
2CRAFT CANNOT SEE THE TARGET, ZENITH ANGLE (DEG) = ', F10.5, '/')
WRITE (6,105)
105 FORMAT ('/1X,' FOR SLANT PATH CALCULATION THE LEVELS ARE CHOSEN
1 AS FOLLOWS,'/16X,' ALTITUDE  PRESSURE TEMPERATURE',
2 '/16X,' KM MB DEG K)
WRITE (6,104) ZZZ(1), PSURF, TSURF
K111=K1-1
DO 1410 J=1,K111
CALL MOATM(ZZZ(J+1), PSURF, DELP*FLOAT(J), 'PRES', XLAMDA,
1 ANS, RE, CONN)
WRITE (6,104) ZZZ(J+1), ANS(1), ANS(2)
104 FORMAT (1X, 9X, 1P3E14.4)
1410 CONTINUE
WRITE (6,104) ZZZ(K1+1), PSAT, TSAT
IF (PHI/CON .LT. 1.0) GO TO 2223
C IF THE UNREFRACTED ZENITH ANGLE IS LESS THAN 1 DEG THEN EXIT
89 DELT=(ZZZ(2)-ZZZ(1))/10.0
 CALL MOATM(ZL+DELT*5, PP, 'ALTI', XLAMDA, ANS, RE, CONN)
 PHIINT=PHI
Z1=ZL
D1=ANS(3)
XN1=ANS(18)
SUM=0.0
SUM1=0.0
SUM3=0.0
SUM4=0.0
SUM4P=0.
DO 2 J=1,K1
JT=((10*(J-1))+1)
JTP=10*J
DELT=(ZZZ(J+1)-ZZZ(J))/10.0
DO 1 I=JT, JTP
Z2=Z1+DELT
S1=RE+Z1
S2=RE+Z2
HAFDEL=DELT*5
IF((LEQ.(K1*10)) HAFDEL=0.
 CALL MOATM(Z2+HAFDEL, PP, 'ALTI', XLAMDA, ANS, RE, CONN)
D2=ANS(3)
XN2=ANS(18)
PSI=SININV(S1*SIN(PHI)/S2)
PHIPR=SININV(S1*SIN(PHI-XN1)/(S2*XN2))
DUM=D1*S1*SIN(PHI-PSI)/SIN(PHI)*1.0E+05
SUM1=SUM1+PHI-PSI
SUM3=SUM3+DUM
IF (D1.GT. 1.0E-10) SUM4=SUM4+DUM/D1
SUM=SUM+ABS(PHIPR-PSI)
PHI=PHIPR
Z1=Z2
D1=D2
1 XN1=XN2
PATHM(J)=SUM4-SUM4P
2 SUM4P=SUM4
82 CONTINUE
Q=SUM1-ABD
PHI=PHIINT-Q/2.0
IF (ABS(Q).GE.0001) GO TO 89
ANS(21) = PHI
ANS(22) = SUM3
ANS(24) = SUM4
IF (PHI/CON.LE.90.0) GO TO 83
WRITE (6,87)
87 FORMAT (1X, ///, ' THE ANGLE FROM ZENITH IS GREATER THAN 90.0')
ANS(22) = 0.0
ANS(24) = 0.0
GO TO 83
2223 DO 2224 I = 1, K1
2224 PATHM(I) = (ZZZ(I+1) - ZZZ(I)) * 100000.0
ANS(21) = 0.0
83 WRITE (6,2114) SUM/CON
2114 FORMAT (1X, ///, ' TOTAL REFRACTED ANGLE THRU ATMOSPHERE = ', 1PE15.4
     1, ' DEGREES')
WRITE (6,21) ANS(21)/.0174532925
21 FORMAT (1X, ///, ' ZENITH ANGLE (DEGREES) = ', 1PE15.5, ///)
RETURN
END
C
C*************************************************************************
C
FUNCTION COSINV(A)
C THIS FUNCTION CALCULATES THE INVERSE COSINE OF 'A'.
COSINV = ATAN2(SQRT(1.0 - A**2), A)
RETURN
END
C
C*************************************************************************
C
FUNCTION SININV(A)
C THIS FUNCTION CALCULATES THE INVERSE SINE OF 'A'.
IF(ALT.1.) GO TO 1
101 FORMAT (' ERROR IN SININV - A='F12.9)
A = 1.
1 SININV = ATAN2(A, (SQRT(1.0 - A**2)))
RETURN
END
C
C*************************************************************************
C
FUNCTION ALTITU(TMHIGH, TMLOW, PHIGH, PLOW, HLOW, ANS, RE, CONN)
DIMENSION ANS(35)
C
C*************************************************************************
C
D = TMHIGH - TMLOW
IF(D) 2, 3, 2
2 \( D = \text{CONN} \cdot 1.0E+03 / (\text{ALOG(PHIGH/PLow)}) \cdot \text{ALOG(TMHIGH/TMLOW)} \)

ALTITU = HLOW + (TMHIGH - TMLOW) / D

GO TO 6

3 ALTITU = HLOW + TMLOW * ALOG(PHIGH/PLow) / (CONN * 1000.0)

6 RETURN

END

C

FUNCTION PRES(PLOW, D, TMLOW, TMHIGH, DH, ANS, RE, CONN)
DIMENSION ANS(35)

C

C

C THIS PROGRAM CALCULATES PRESSURE -PRES- AT SOME POINT -DH- ABOVE A
C POINT IN THE ATMOSPHERE HAVING PRESSURE -PLow- WHERE -D- IS THE
C TEMPERATURE GRADIENT AND -TMHIGH- AND -TMLOW- ARE CORRESPONDING
C TEMPERATURES. -CONN- IS CONSTANT = -M*G/R

C

IF(D) 2,3,2
2 PRES = PLOW * (TMHIGH / TMLOW)**(CONN * 1.0E+03 / D)
GO TO 4
3 PRES = PLOW * EXP(-CONN * DH * 1.0E+03 / TMLOW)
4 RETURN

END

C

FUNCTION OMEG(T,J,IQF)
C REDUCED COLLISIONAL INTEGRAL FOR JTH CONSTITUENT AS F(TEMP)
C T = TEMP IN DEG K
C SET IQF = 0 FIRST TIME TO SET UP ARRAYS
DIMENSION QT(8),OM(8,10)

IF(IQF) 9,8,9
8 CONTINUE
QT(1) = 100.0
QT(2) = 200.0
QT(3) = 300.0
QT(4) = 400.0
QT(5) = 500.0
QT(6) = 600.0
QT(7) = 700.0
QT(8) = 800.0

C OM(,1) IS NITROGEN (MOLECULAR I.E. N2)
OM(1,1) = 13.36
OM(2,1) = 11.47
OM(3,1) = 9.60
OM(4,1) = 9.43
OM(5,1) = 9.137
OM(6,1) = 8.68
OM(7,1) = 8.509
OM(8,1) = 8.23

C OM(,2) IS CARBON DIOXIDE
OM(1,2) = 23.604
OM(2,2) = 15.708
OM(3,2) = 13.26
OM(4,2) = 11.62
OM(5,2) = 10.73
OM(6,2) = 10.28
OM(7,2) = 9.83
OM(8,2) = 9.59

C OM(3) IS OXYGEN (MOLECULAR I.E. O₂)
OM(1,3) = 16.12
OM(2,3) = 11.7
OM(3,3) = 10.22
OM(4,3) = 9.45
OM(5,3) = 9.10
OM(6,3) = 8.81
OM(7,3) = 8.59
OM(8,3) = 8.40

C OM(4) IS ARGON
OM(1,4) = 17.15
OM(2,4) = 12.58
OM(3,4) = 10.75
OM(4,4) = 10.00
OM(5,4) = 9.65
OM(6,4) = 9.10
OM(7,4) = 8.96
OM(8,4) = 8.75

C OM(5) IS NEON
OM(1,5) = 9.72
OM(2,5) = 8.65
OM(3,5) = 8.239
OM(4,5) = 7.903
OM(5,5) = 7.470
OM(6,5) = 7.422
OM(7,5) = 7.203
OM(8,5) = 6.93

C OM(6) IS HYDROGEN (MOLECULAR I.E. H₂)
OM(1,6) = 11.04
OM(2,6) = 9.28
OM(3,6) = 8.53
OM(4,6) = 8.18
OM(5,6) = 8.11
OM(6,6) = 7.80
OM(7,6) = 7.52
OM(8,6) = 7.37

C OM(7) IS HELIUM
OM(1,7) = 8.13
OM(2,7) = 7.2
OM(3,7) = 6.97
OM(4,7) = 6.62
OM(5,7) = 6.43
OM(6,7) = 6.28
OM(7,7) = 6.05
OM(8,7) = 5.99

C OM(8) IS WATER (H₂O)
OM(1,8) = 37.27
OM(2,8) = 23.89
OM(3,8) = 20.14
OM(4,8) = 17.48
OM(5,8) = 15.92
OM(6,8) = 14.57
OM(7,8) = 13.52

198
OM(8,8)=12.80
C OM(,9)=CARBON MONOXIDE
   OM(1,9)=15.58
   OM(2,9)=11.27
   OM(3,9)=10.28
   OM(4,9)=9.49
   OM(5,9)=9.05
   OM(6,9)=8.76
   OM(7,9)=8.50
   OM(8,9)=8.25
C OM(,10) IS SULFUR DIOXIDE
   OM(1,10)=24.21
   OM(2,10)=17.96
   OM(3,10)=14.69
   OM(4,10)=12.91
   OM(5,10)=11.92
   OM(6,10)=11.03
   OM(7,10)=10.60
   OM(8,10)=10.33
9 DO 1 I=1,8
   IF (QT(I)-T) 1,2,3
1 CONTINUE
   GO TO 2
3 IF (I-1) 6,2,6
6 I=I-1
   OMEG=(OM(I+1,J)-OM(I,J))/(QT(I+1)-QT(I))*(T-QT(I))+OM(I,J)
   GO TO 4
2 OMEG=OM(I,J)
4 RETURN
END
C
FUNCTION CPS (T,J,IQF)
C SPECIFIC HEAT AT CONSTANT PRESSURE, CAL/(MOLE DEG K)
C T = TEMP IN DEG K
C SET IQF = 0 FIRST TIME TO SET UP ARRAYS
C J = CONSTITUENT
   DIMENSION QT(7),CP(7,10)
   IF (IQF) 9,8,9
8 CONTINUE
   QT(1)=100.0
   QT(2)=200.0
   QT(3)=300.0
   QT(4)=400.0
   QT(5)=500.0
   QT(6)=600.0
   QT(7)=700.0
C CP(,1) = NITROGEN
   CP(1,1) = 6.9562
   CP(2,1) = 6.9571
   CP(3,1) = 6.9613
   CP(4,1) = 6.9910
   CP(5,1) = 7.0703
   CP(6,1) = 7.1968
   CP(7,1) = 7.3509
C CP(,2) = CARBON DIOXIDE
<table>
<thead>
<tr>
<th></th>
<th>CP(1,2)</th>
<th>CP(2,2)</th>
<th>CP(3,2)</th>
<th>CP(4,2)</th>
<th>CP(5,2)</th>
<th>CP(6,2)</th>
<th>CP(7,2)</th>
</tr>
</thead>
</table>

C CP(,3) = MOLECULAR OXYGEN
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<tr>
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<th>CP(2,3)</th>
<th>CP(3,3)</th>
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<th>CP(5,3)</th>
<th>CP(6,3)</th>
<th>CP(7,3)</th>
</tr>
</thead>
</table>

C CP(,4) = ARGON
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<th>CP(2,4)</th>
<th>CP(3,4)</th>
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<th>CP(5,4)</th>
<th>CP(6,4)</th>
<th>CP(7,4)</th>
</tr>
</thead>
</table>

C CP(,5) = NEON
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<tr>
<th></th>
<th>CP(1,5)</th>
<th>CP(2,5)</th>
<th>CP(3,5)</th>
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<th>CP(5,5)</th>
<th>CP(6,5)</th>
<th>CP(7,5)</th>
</tr>
</thead>
</table>

C CP(,6) = HYDROGEN
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<th>CP(1,6)</th>
<th>CP(2,6)</th>
<th>CP(3,6)</th>
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<th>CP(5,6)</th>
<th>CP(6,6)</th>
<th>CP(7,6)</th>
</tr>
</thead>
</table>

C CP(,7) = HELIUM
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<th></th>
<th>CP(1,7)</th>
<th>CP(2,7)</th>
<th>CP(3,7)</th>
<th>CP(4,7)</th>
<th>CP(5,7)</th>
<th>CP(6,7)</th>
<th>CP(7,7)</th>
</tr>
</thead>
</table>

C CP(,8) = WATER
<table>
<thead>
<tr>
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<th>CP(1,8)</th>
<th>CP(2,8)</th>
<th>CP(3,8)</th>
<th>CP(4,8)</th>
<th>CP(5,8)</th>
<th>CP(6,8)</th>
<th>CP(7,8)</th>
</tr>
</thead>
<tbody>
<tr>
<td>CP</td>
<td>7.9606</td>
<td>7.9694</td>
<td>8.0276</td>
<td>8.1864</td>
<td>8.4161</td>
<td>8.6779</td>
<td>8.9571</td>
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</tbody>
</table>

C CP(,9) = CARBON MONOXIDE
<table>
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<tr>
<th></th>
<th>CP(1,9)</th>
<th>CP(2,9)</th>
</tr>
</thead>
<tbody>
<tr>
<td>CP</td>
<td>6.9564</td>
<td>6.9574</td>
</tr>
</tbody>
</table>

200
CP(3,9)=6.9656
CP(4,9)=7.0129
CP(5,9)=7.1211
CP(6,9)=7.2760
CP(7,9)=7.4507

C CP(,10) = SULFUR DIOXIDE
CP(1,10)=8.0134
CP(2,10)=8.6948
CP(3,10)=9.5451
CP(4,10)=10.3919
CP(5,10)=11.1292
CP(6,10)=11.7189
CP(7,10)=12.1755

9 DO 1 I=1,7
IF (QT(I)-T) 1,2,3
1 CONTINUE
GO TO 2
3 I=I-1
CPS=(CP(I+1,J)-CP(I,J))/(QT(I+1)-QT(I))*(T-QT(I))+CP(I,J)
GO TO 4
2 CPS=CP(I,J)
4 RETURN
END