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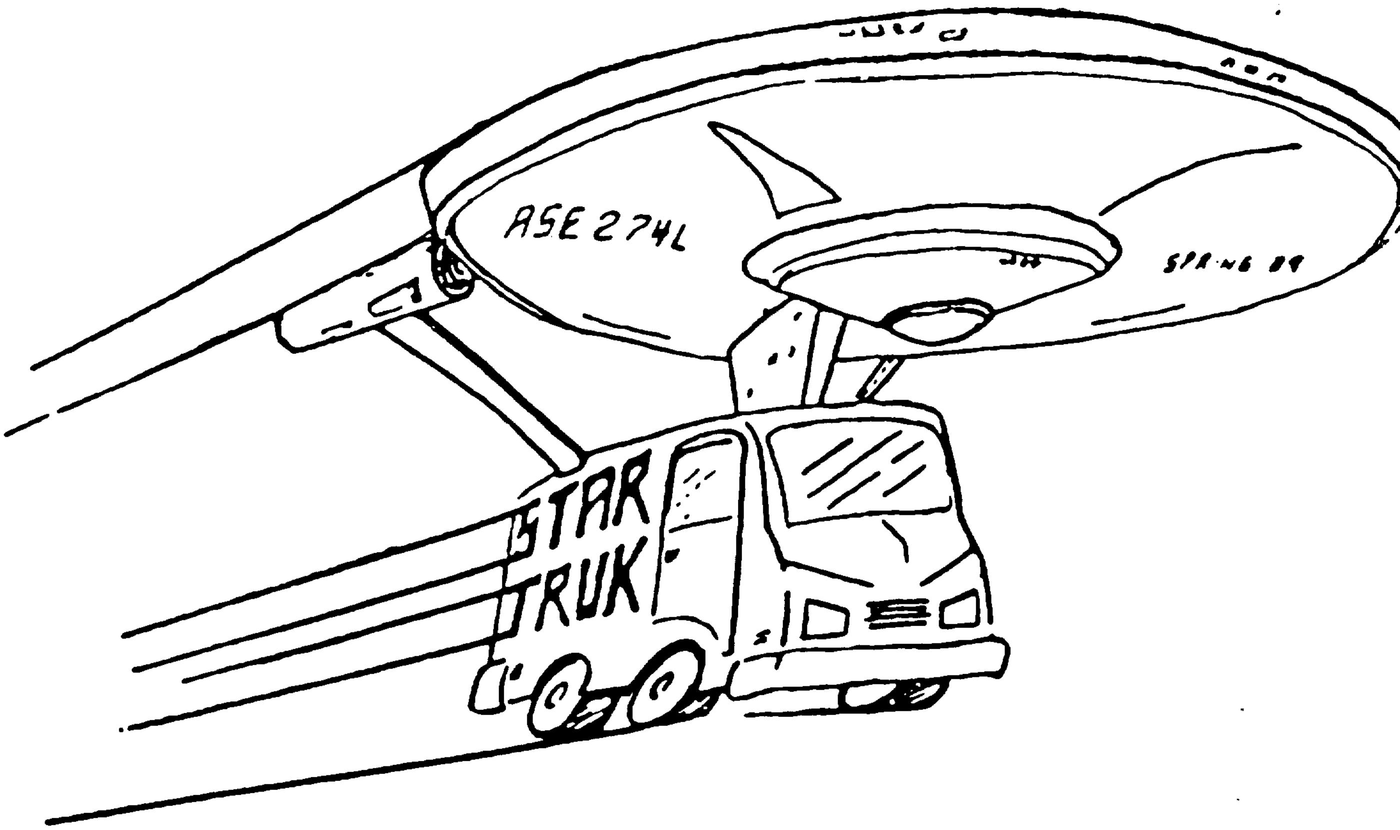
Final Design Review Report for a Mars/Phobos Transportation System



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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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.

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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

OWL	One Way Lander
PDR	Preliminary Design Review
RFP	Request For Proposal
TOF	Time of Flight

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 refered 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 1. Orbital Vehicle Mission Scenarios Summary

Component	Upside	Downside
ITV	Insert at Phobos	Insert at Phobos
OTV	OWL: HMO -> LMO	(Lander engines used
	OWL: de-orbit burn	for orbit mnvrs and
	ADV: HMO -> LMO	de-orbit)
OMV	Phobos delivery	Phobos delivery
	Prox-Ops	Prox-Ops
LMO Node	Fuel depot	Fuel stored at bases
	Service facility	Facilities at Phobos
	Prox-Ops w/ OMV	N/A
	Safe Haven for Man	N/A

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 ΔV 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

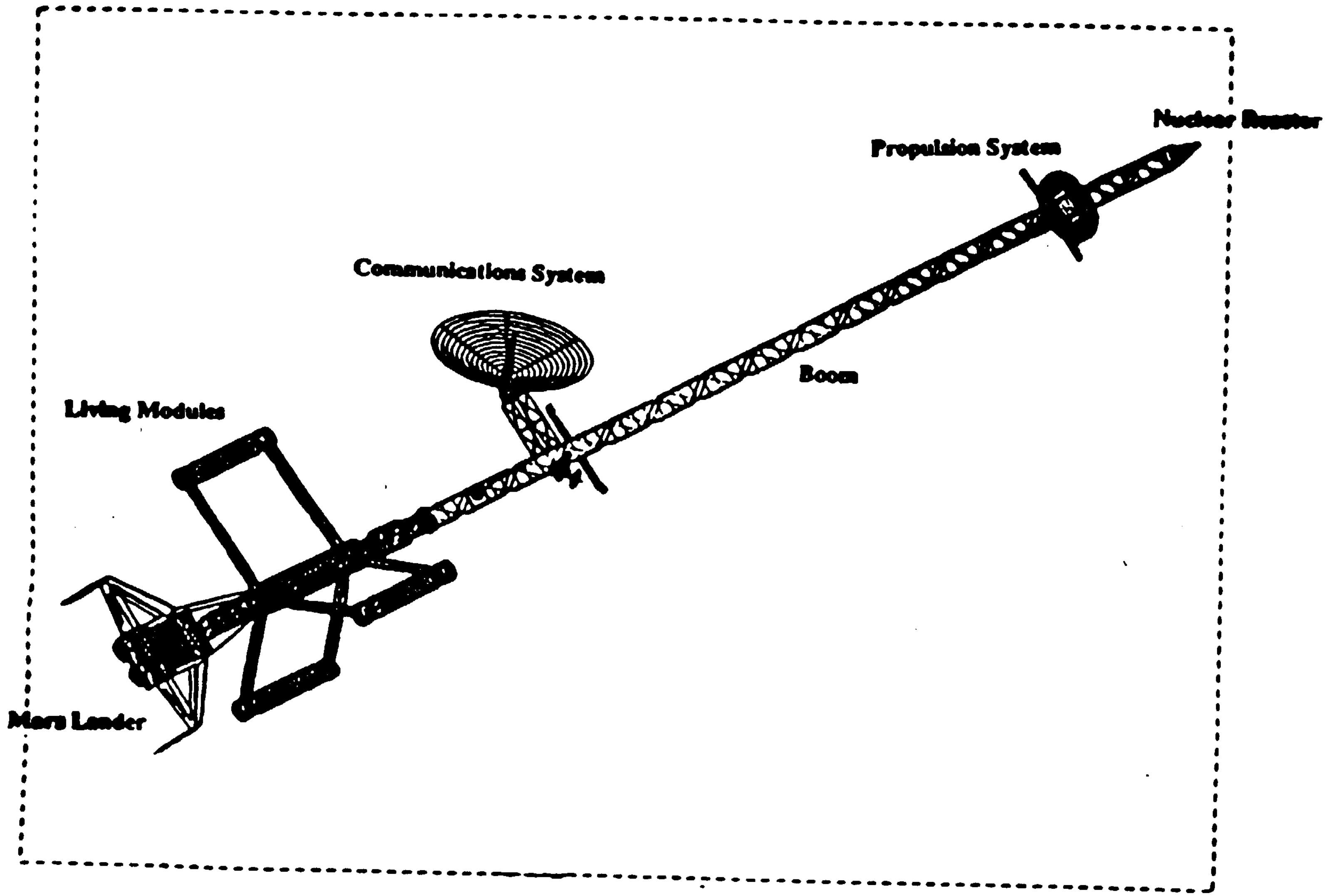

Figure 1. Kepler Interplanetary Transfer Vehicle

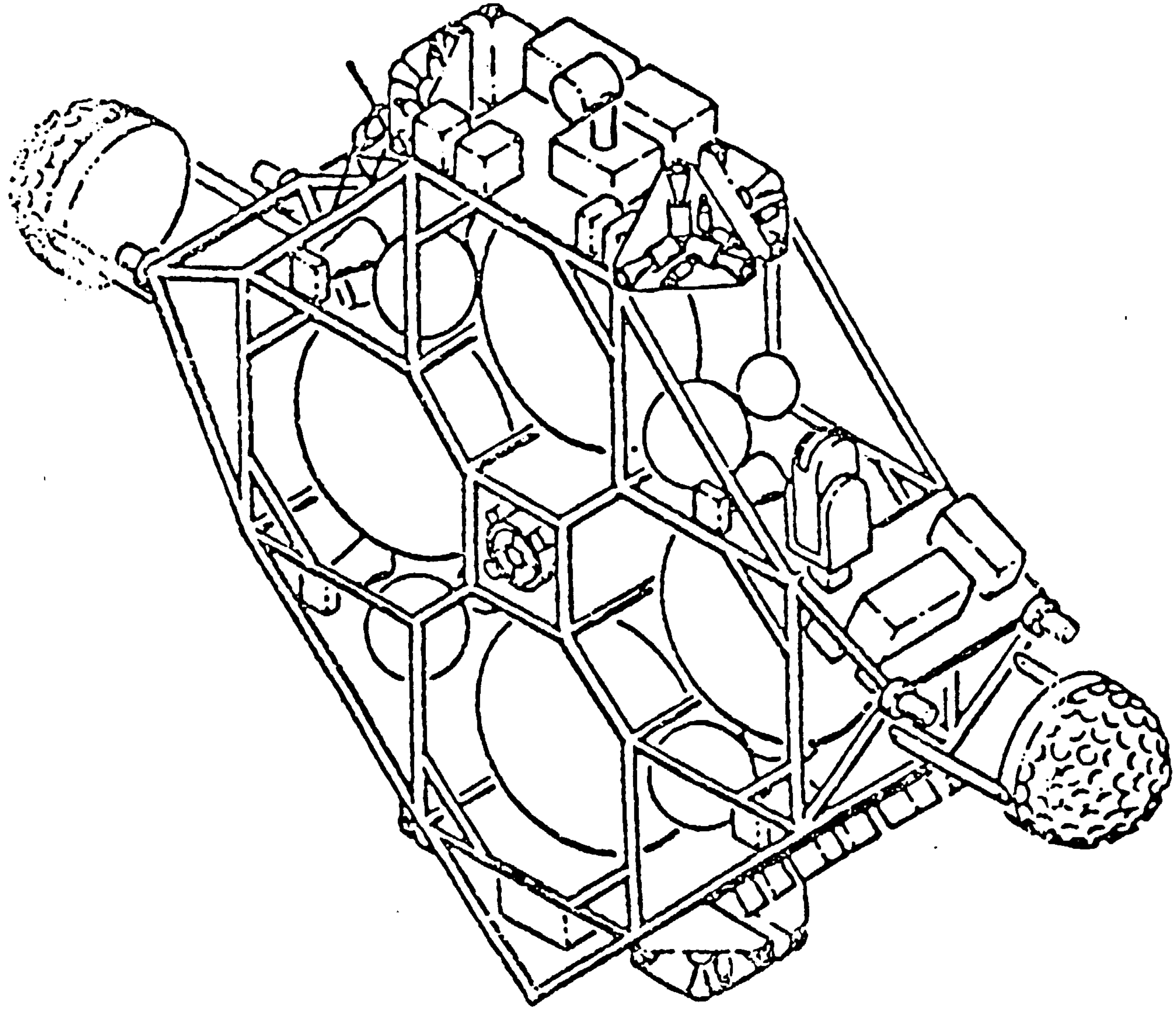

Figure 2. Orbital Maneuvering Vehicle

Table 2. Orbital Vehicle Sizing Results

Vehicle	Payload	Mass (MT)	Required Prop (MT)	LO2 Tank Diameter (m)	LH2 Tank Diameter (m)
OTV $\Delta i=50$	OWL-200 - wet	1550	1252	11.7	16.9
	OWL-200 prop.	850	724	9.8	14.1
	OWL-200 - dry	700	611	9.3	13.4
	ADV-50 - wet	1060	883	10.5	15.1
	ADV-50 prop.	905	766	10.0	14.4
	ADV-50 - dry	155	199	6.4	9.2
	ADV-10 - wet	245	268	7.1	10.2
	ADV-10 prop.	210	241	6.8	9.8
	ADV-10 - dry	35	109	5.3	7.8
OMV	Standard Payload Module	120	.649	1.0	1.4

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.

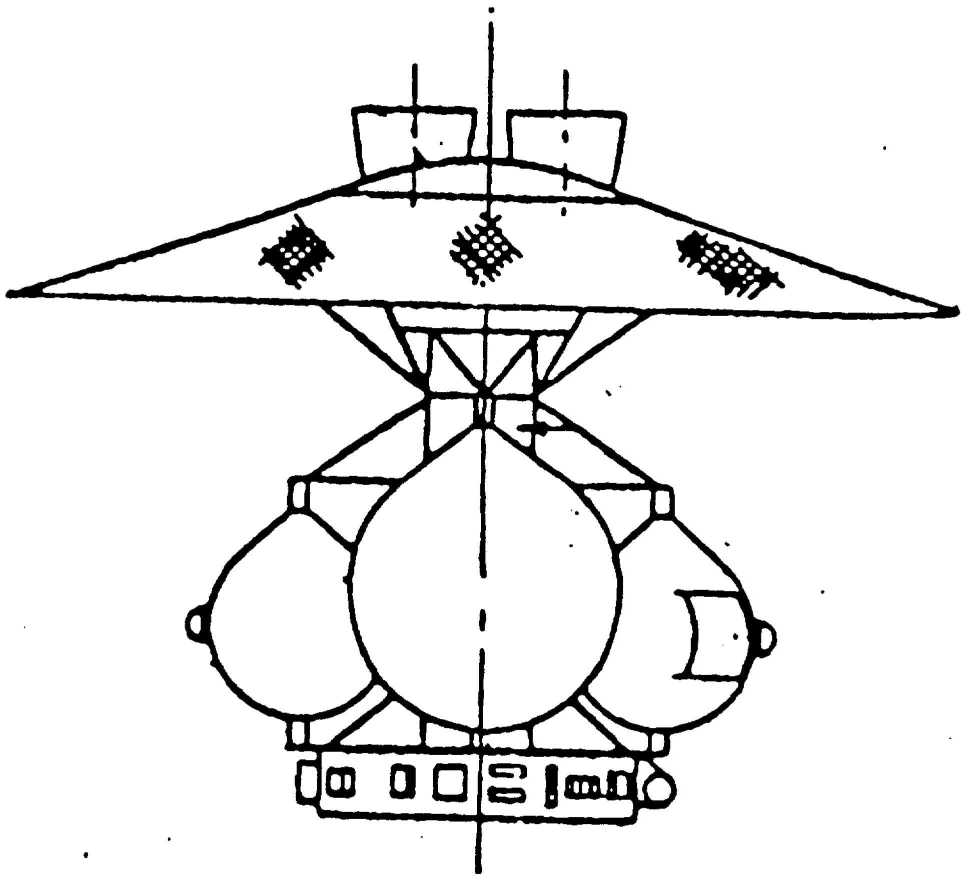

Figure 3. Orbital Transfer Vehicle

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.

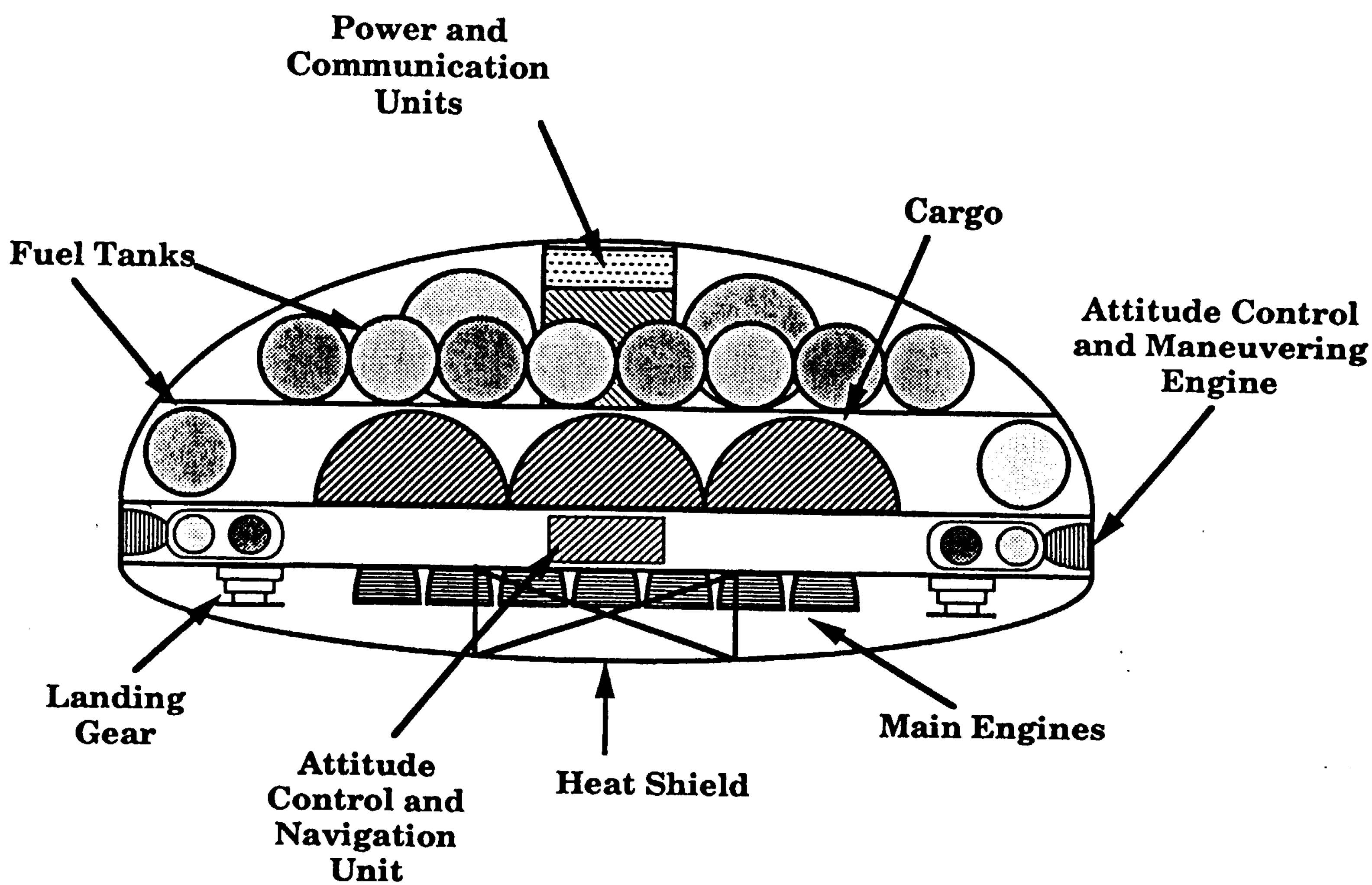

Figure 4. OWL-200

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 dicussion 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.

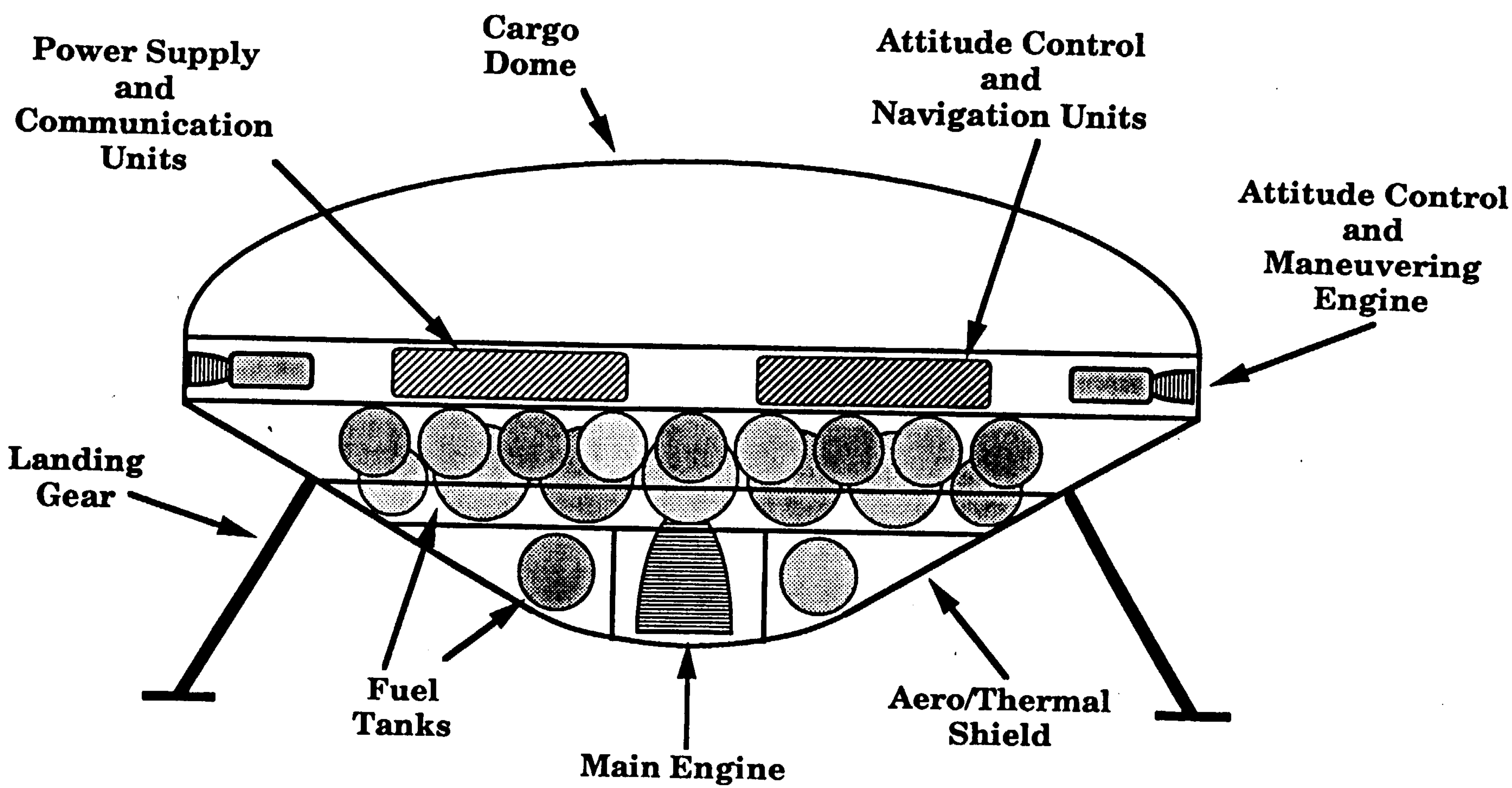

Figure 5. ADV-50

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 believe 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 relevants 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.

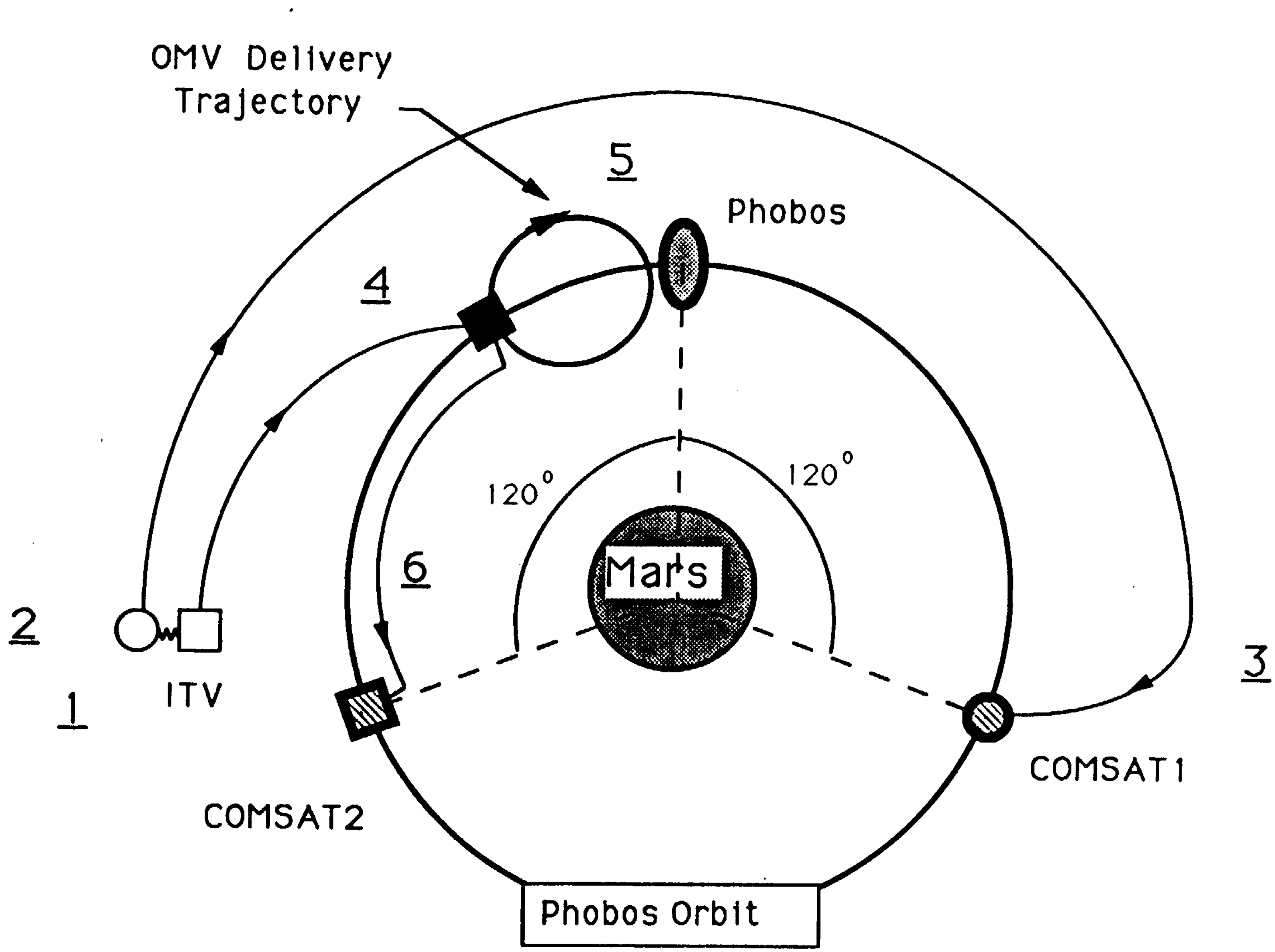

Figure 6. Mission Scenario for Phobos Base Delivery

5 While the ITV is in this chaser orbit, the cargo will be removed as described in section 2.4.3.

6 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.

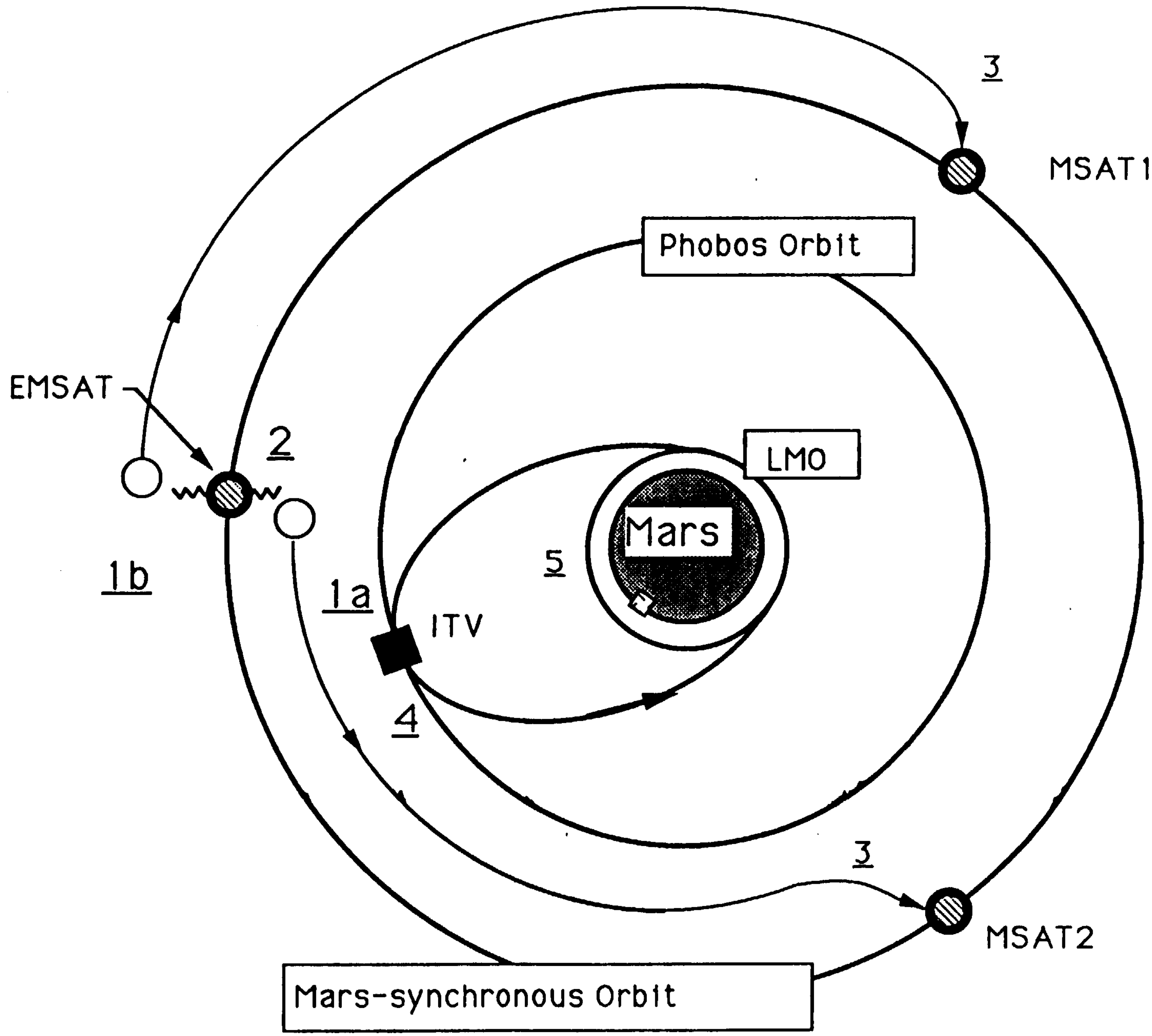

Figure 7. Mission Scenario for Mars Base Delivery

1b The satellite cluster will insert into a Mars synchronous orbit at an orbital radius of approximately six Mars radii.

2 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.

3 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

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

5 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

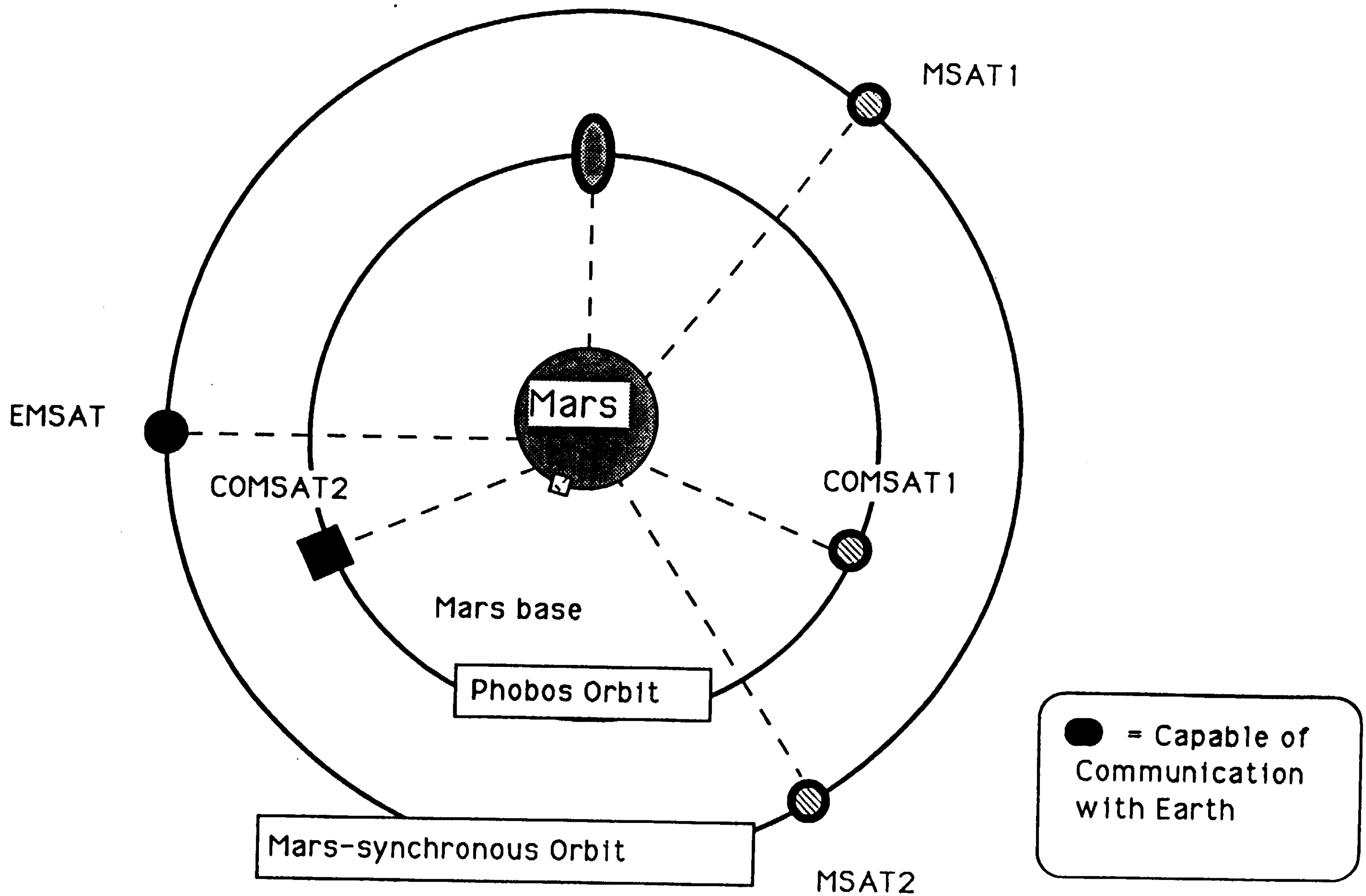

Figure 8. Final Configuration of Communication Satellites.

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

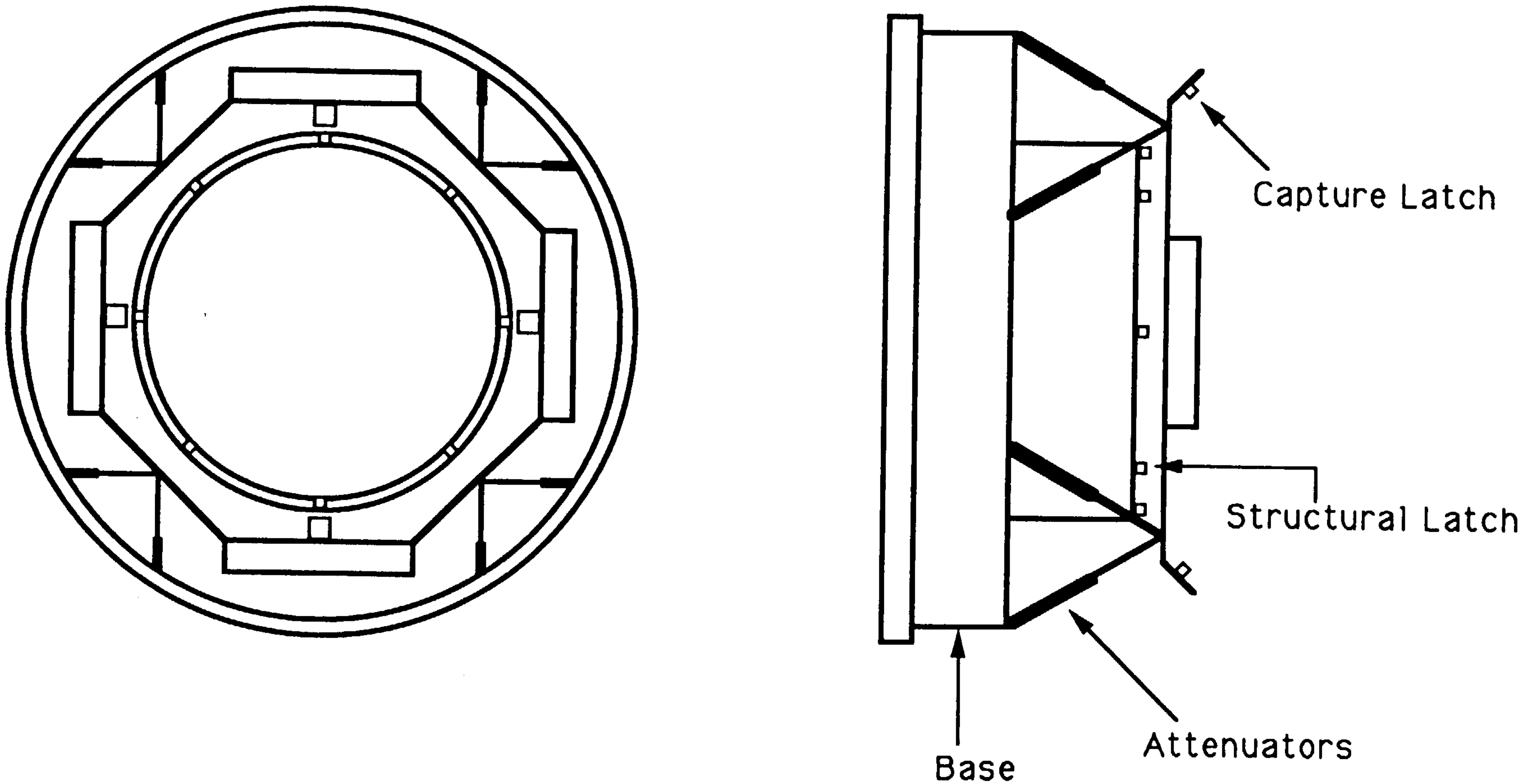

Figure 9. Docking Mechanism

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:

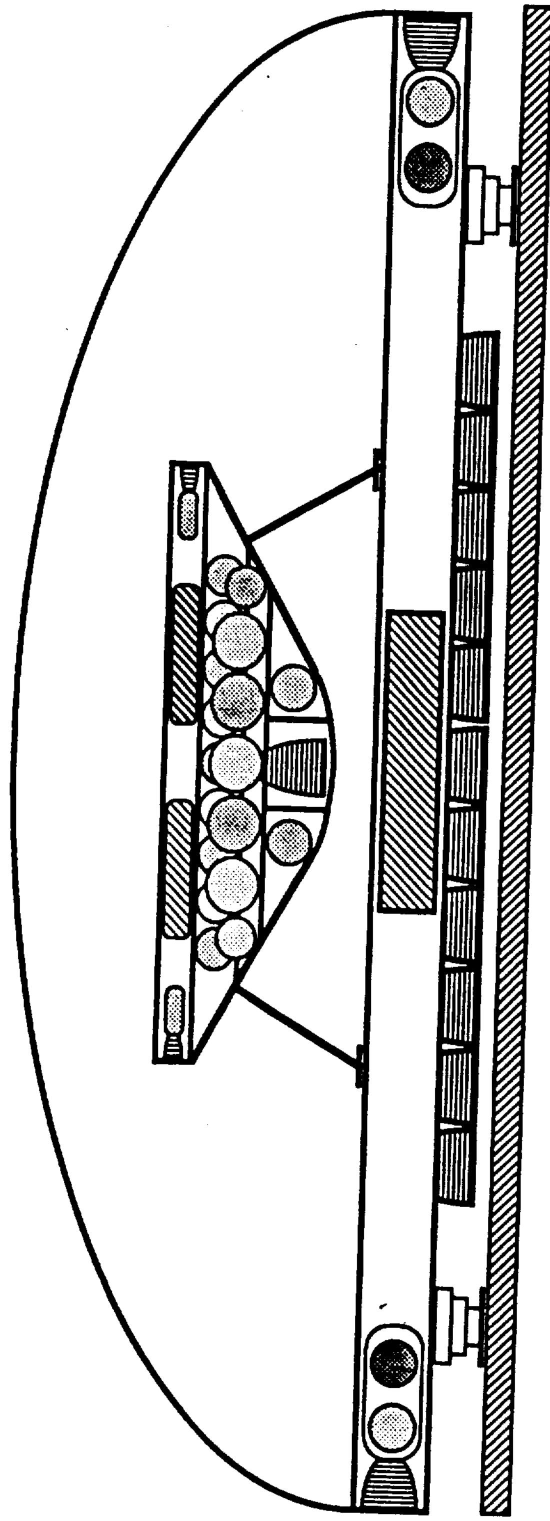

Figure 10. ADV Servicing Facility

• **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 determined 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

StarTruk Company

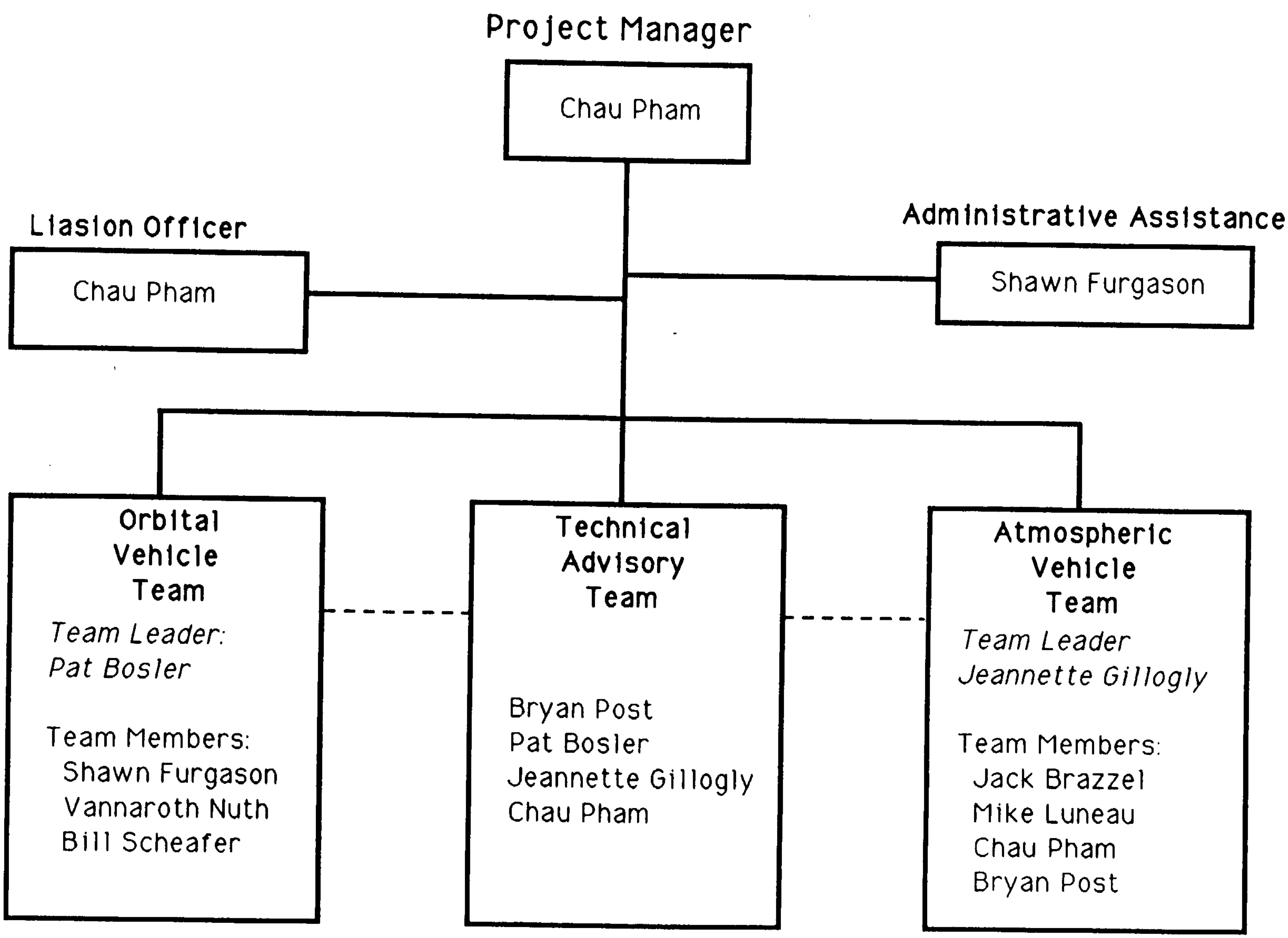

Figure 11. Organization Structure

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.

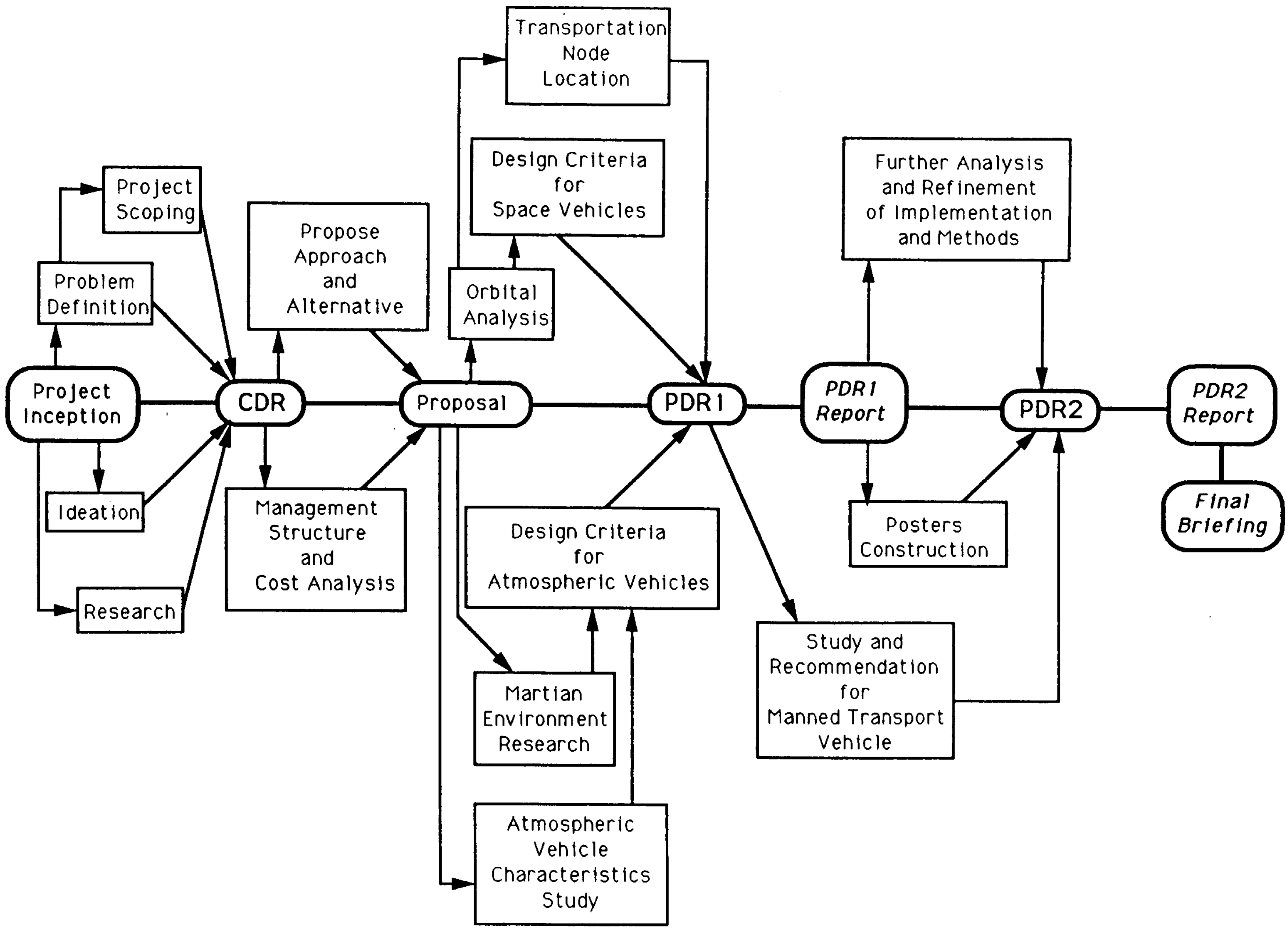

Figure 12. Critical Path Chart

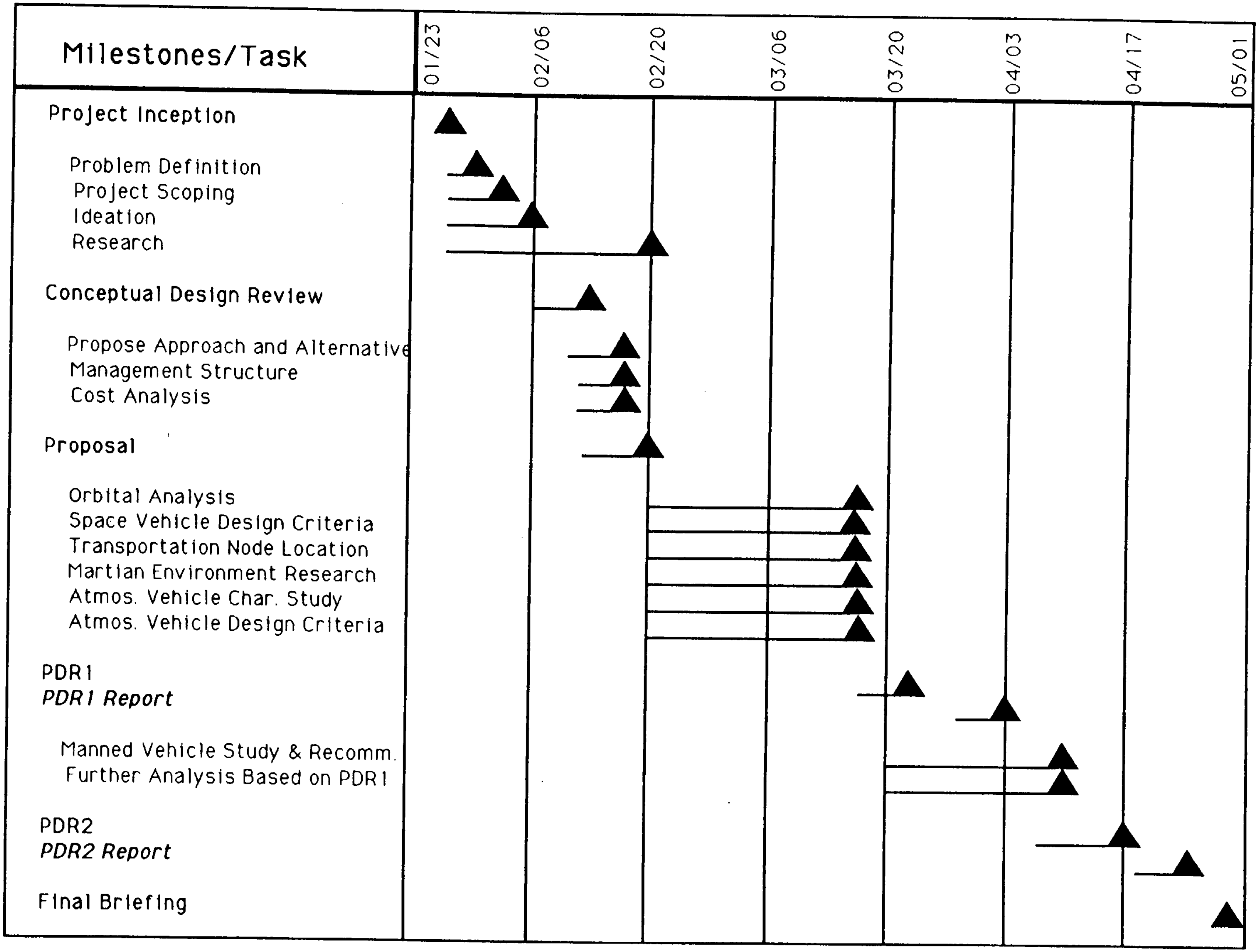

Figure 13. Program Schedule

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

	Actual	Projected
Labor	1547.5 Hrs $31,445	1843 Hrs $37,801
Material	$2400	$2400

5. Bibliography

Briggs, G.A. and F.W. Taylor, Photographic Atlas of the Planets. Cambridge: Cambridge University Press, 1982.

Burns, Joseph A., Planetary Satellites, Tucson: University of Arizona Press, 1977.

Cruz, Manuel I., Generic Planetary Aerocapture: Research and Technology Development -- Final Report, Jet Propulsion Laboratory, Pasadena, California, May 1983.

Gateway: An Earth Orbiting Transportation Node. University of Texas Spacecraft Design Lab, May 1988

Johnson, C. C. "Engineering Principal to Assure Comparable Docking Between Future Spacecraft of USA and USSR". The Journal Of Astronautics, Vol. xxiii, No. 1, pp.1-17, Jan.-Mar., 1975.

Kerry, Mark Joels, The Mars One Crew Manual.: Ballantine Books, 1985.

Kondratyev, K.Y. and G.E. Hunt, Weather and Climate on Planets. New York: Pergamon Press, 1982.

Maegley, W.J. and D.P. Diederich, "Martian Sandstorms and Their Effects on the 1975 Viking Lander System," in Journal of Testing and Evaluation, JTEVA, vol. 3, no. 5, Sept. 1975.

A Manned Mission to Mars, Preliminary Design Review 1. University of Texas Spacecraft Design Lab, April 1986.

A Manned Mission to Mars, Preliminary Design Review 2. University of Texas Spacecraft Design Lab, Spring 1986

Oberg, James, Mission to Mars. Harrisburg: Stackpole Books, 1982.

A Phobos Industrial Production and Supply Base. University of Texas Spacecraft Design Lab, Dec. 1986.

Project Kepler: Manned Mars Mission. University of Michigan, Winter 1986

A Robotically Constructed Production and Supply Base on Phobos: Final Report, University of Texas Spacecraft Design Lab, Spring 1989.

Robotic Construction of a Permanently Manned Mars Base: Final Report, University of Texas Spacecraft Design Lab, Spring 1989.

Thomas, P. and J. Veverka, "Grooves on Asteroids: A Prediction,"in ICARUS, vol. 40, no.3, Dec. 1979

Thomas, P., "Surface Features of Phobos and Deimos," in ICARUS, vol. 40, no. 2, Nov. 1979

ed. Weast, Robert C., CRC Handbook of Chemistry and Physics 60th edition. Boca Raton: CRC Press, Inc., 1980.

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
- biconic 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.

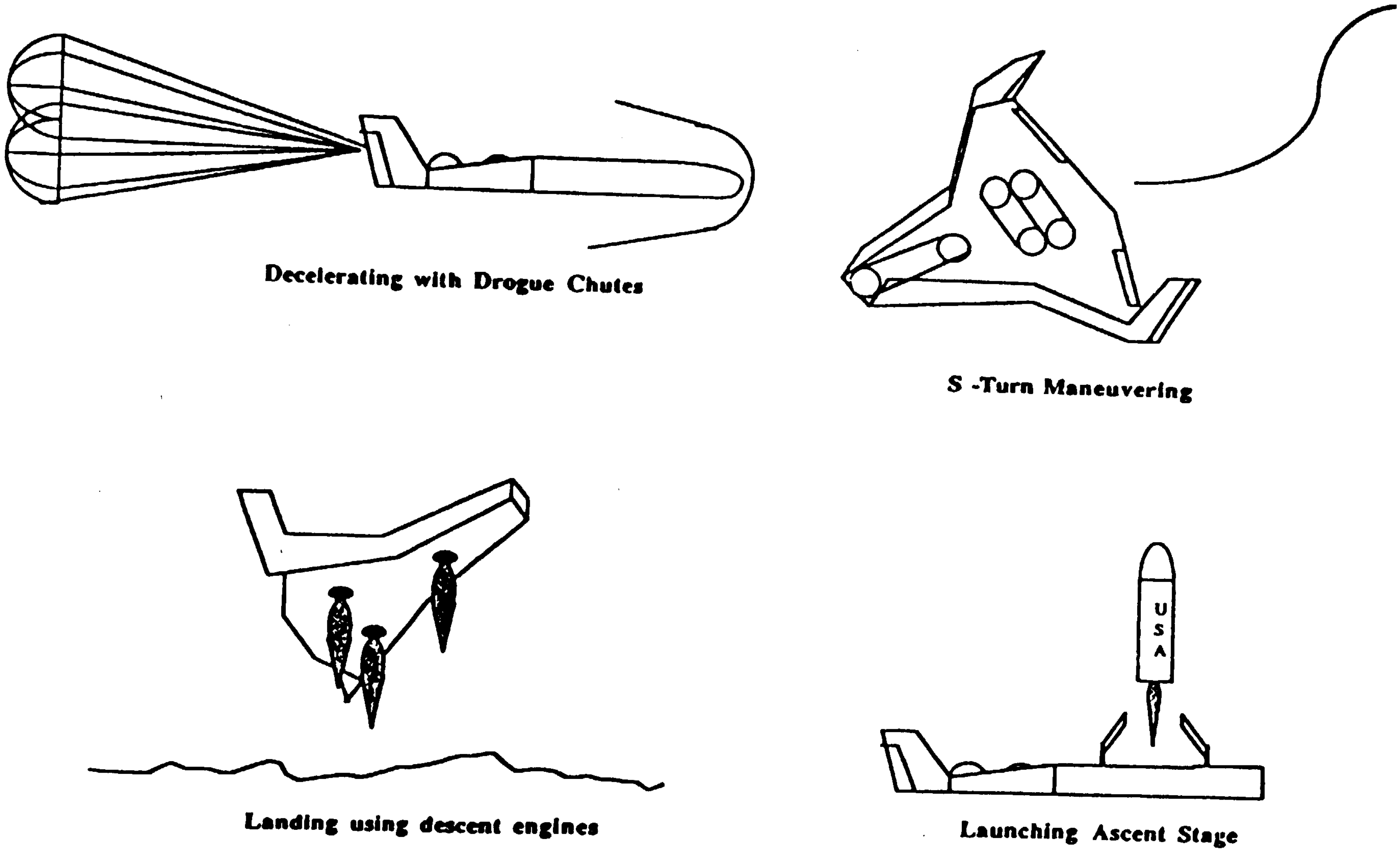

Figure A1. University of Michigan Proposed Mars Lander

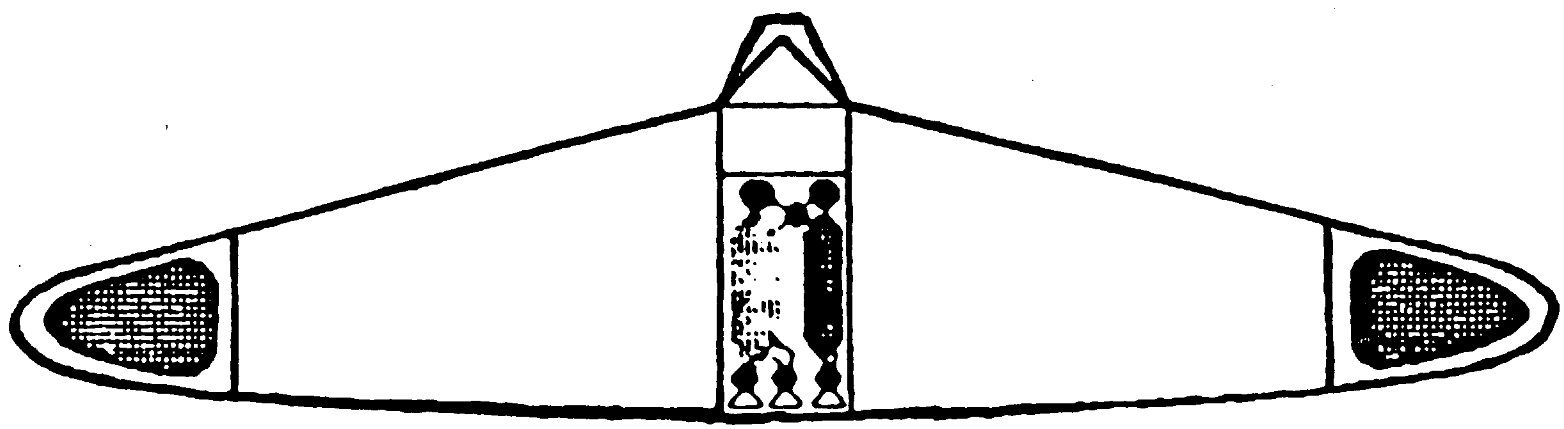

Figure A2. Flattened Apollo Type Lander

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

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[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.

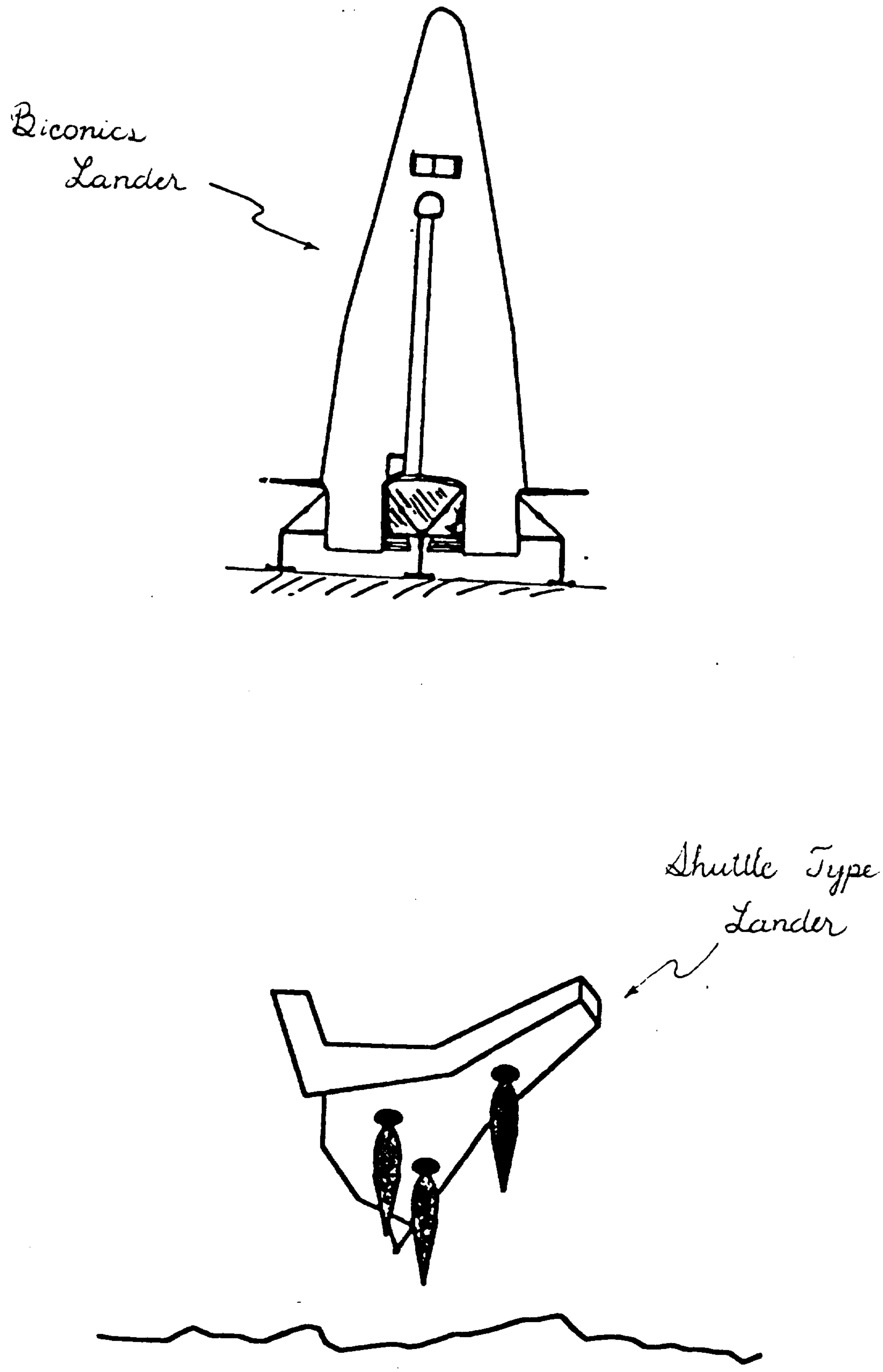


Figure A3. Biconic and Shuttle Type Vehicles

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 TK! Solver model used to compute the minimum insertion ΔV 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 ΔV 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 ΔV 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.

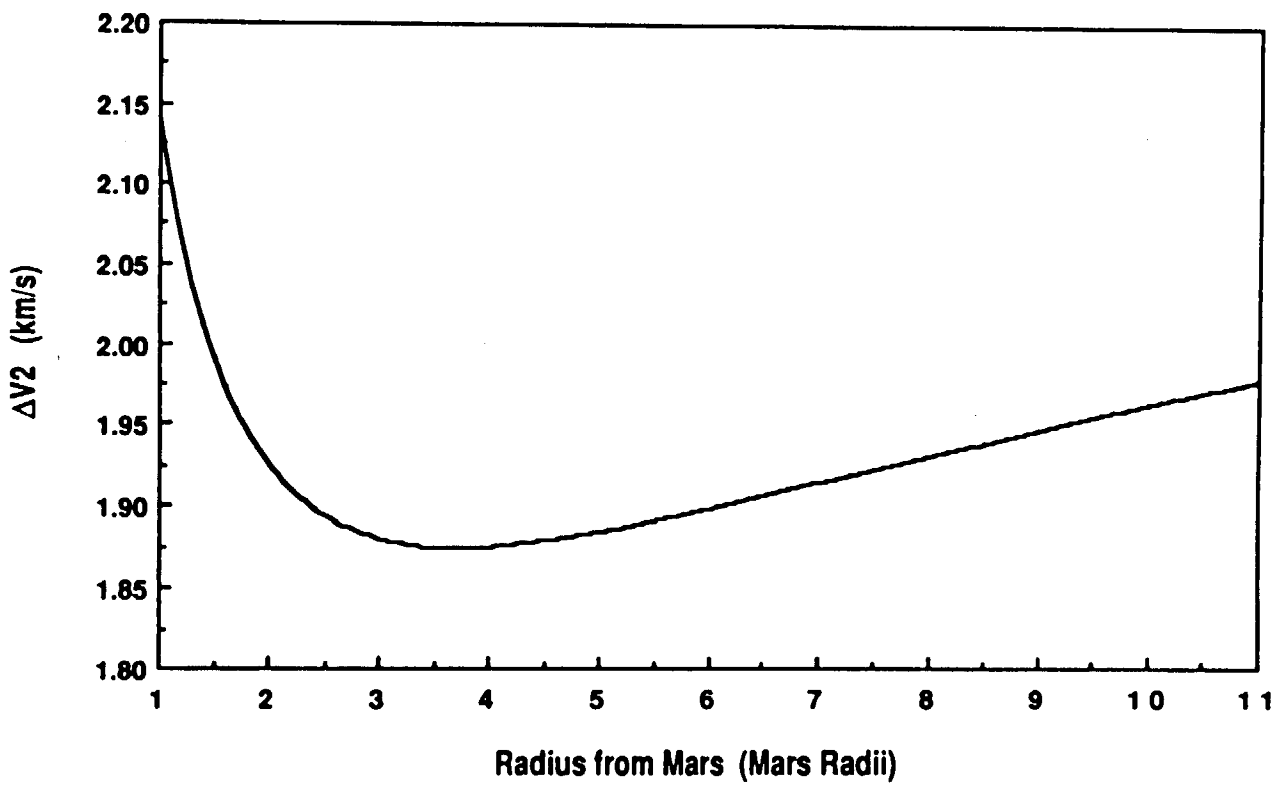


Figure B1. Insertion Orbit ΔV vs. Radius of Orbit

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 Δi 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 (R1) was chosen to reduce the ΔV associated with the plane change. When using a Hohmann transfer between LMO and Phobos, the ΔV associated with the plane change decreases as the LMO radius decreases. Thus, R1 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 R1. Then, based on a gravity model of Mars incorporating J2 effects, the period of repetition of LMO groundtracks was adjusted so that a favorable

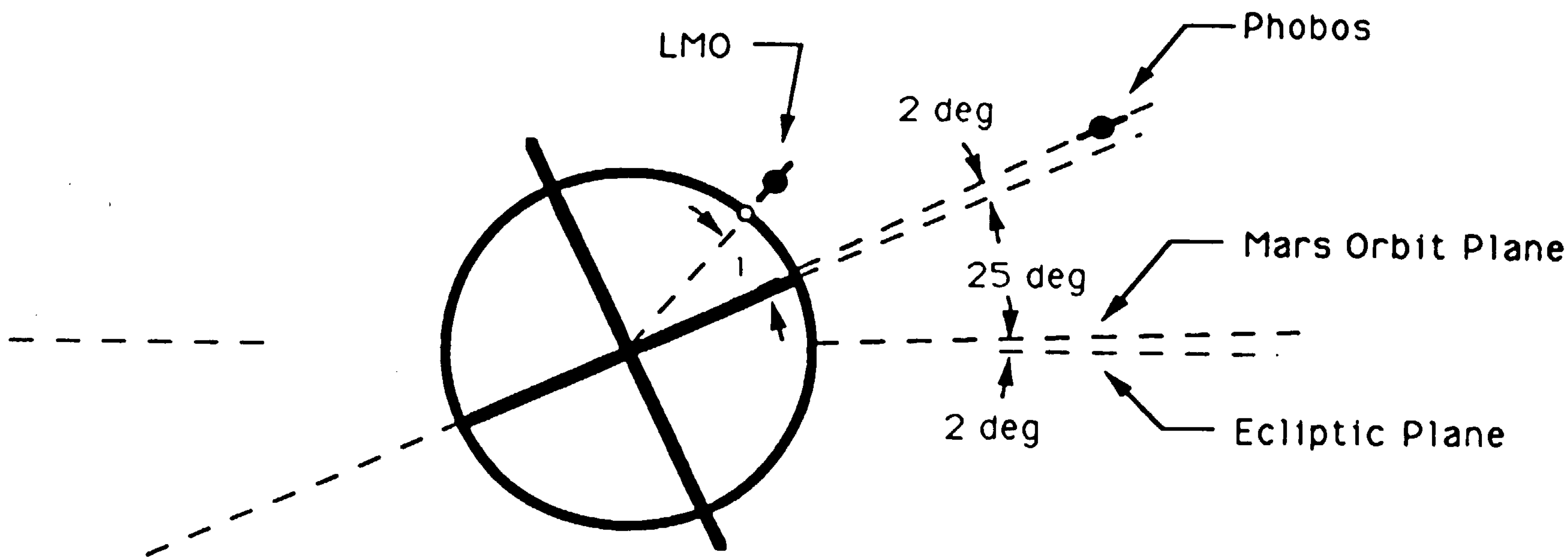


Figure B2. Martian System Orbital Schematic launch window occurred. An orbit radius was calculated for each latitude under consideration for the Mars base.

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.

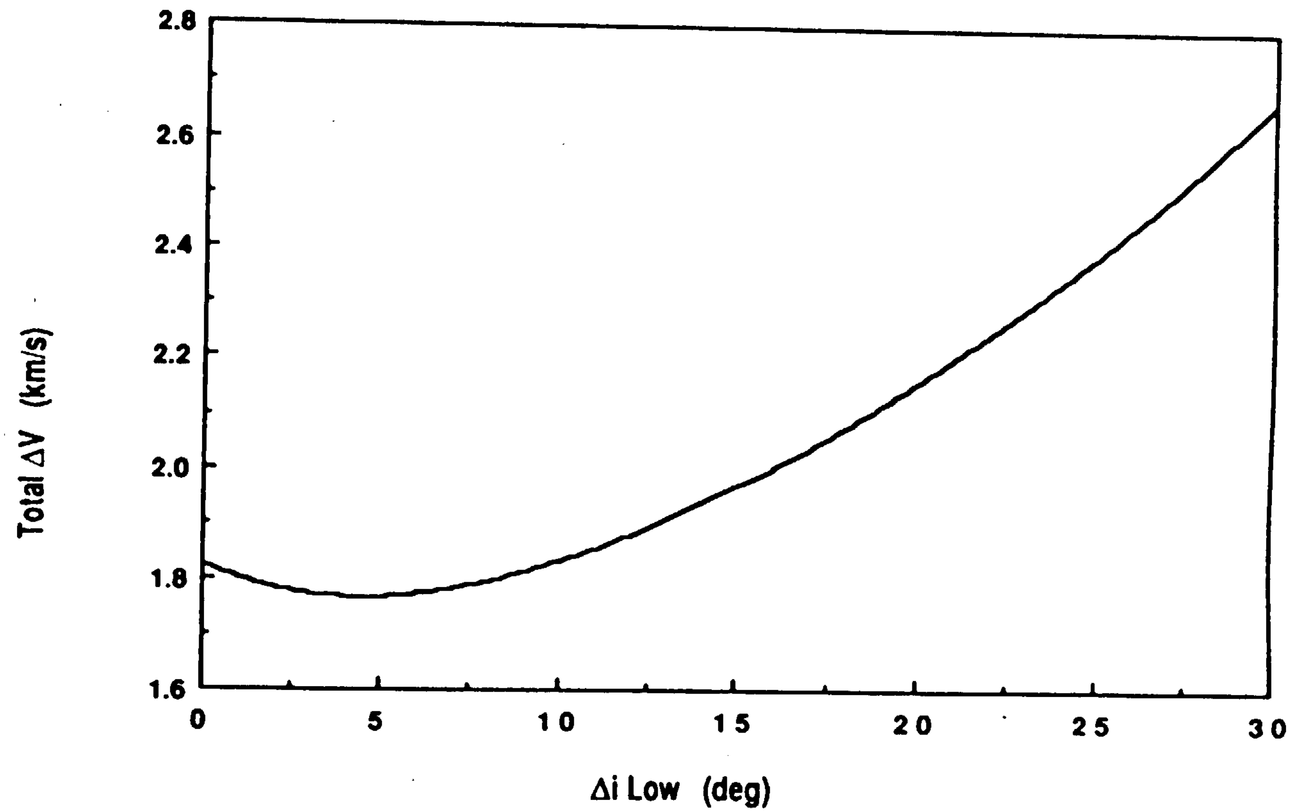


Figure B3a. ΔV vs. Δi Low from LMO to Phobos

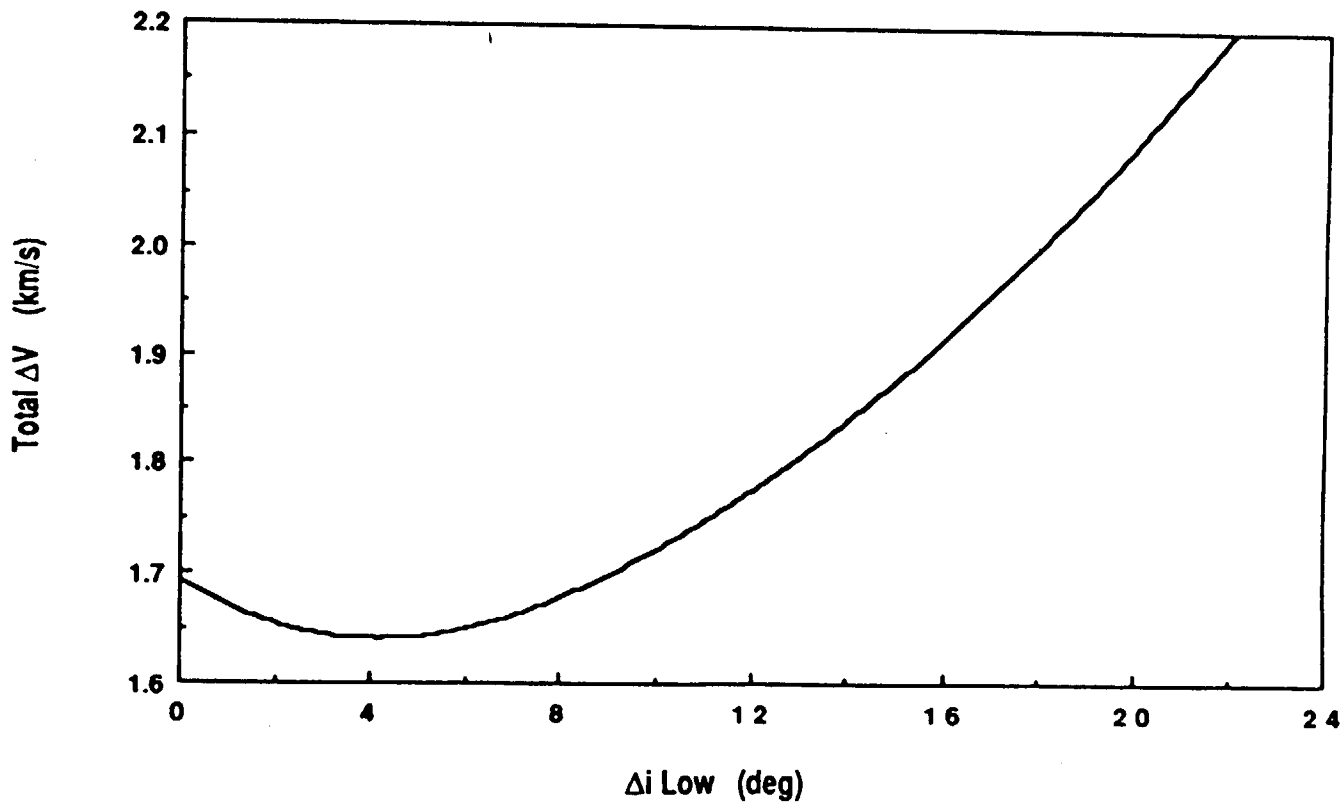


Figure B3b. ΔV vs. Δi Low from LMO to Phobos

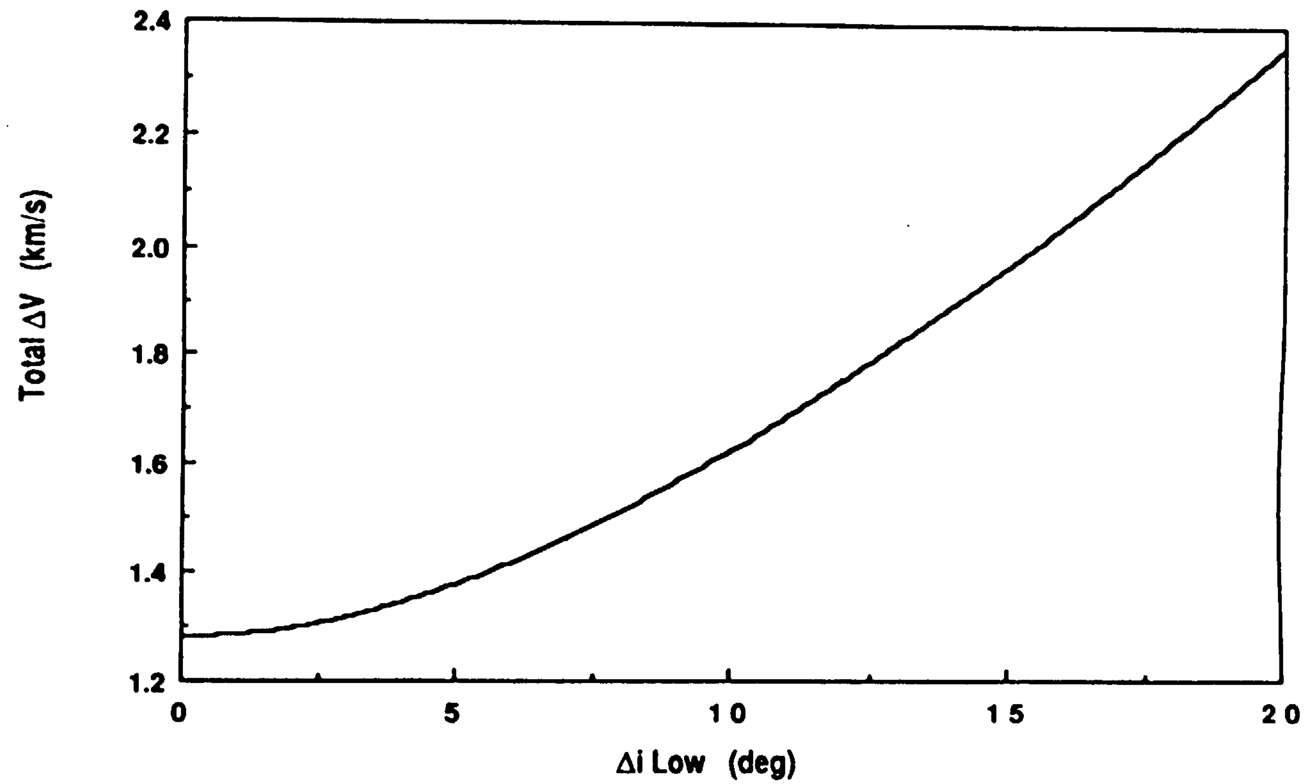


Figure B3c. ΔV vs. Δi Low from LMO to Phobos

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 TK! 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 (I_{sp}) 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

$$\frac{M_{final}}{M_{initial}} = e^{(\Delta V/c)} \qquad \text{where } C = I_{sp} * g$$

Also,

$$M_{final} = e^{(\Delta V/c)} * M_{initial}$$

therefore M_{prop} for the return is

$$M_{prop} = (1 - e^{(\Delta V/c)}) * M_{initial}$$

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_{prop} for the payload delivery. The two M_{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., ΔV'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.

$$M_{prop} = M_{LO2} + M_{LH2}$$

$$M_{LO2} = 5.5\ M_{LH2}$$

Therefore, $\quad M_{prop} = 6.5\ M_{LH2}.$

Solving for the mass of the fuel and oxidizer yields,

$$M_{LH2} = \frac{M_{prop}}{6.5}$$

and, $\quad M_{LO2} = (5.5/6.5)M_{prop}.$

Table C1. Payload Mass vs. Total Propellant for One Trip

Payload (MT)	Propellant (MT)	Oxid tank dia	Fuel tank dia
1550	1252.0980212701	11.740663137152	16.934982042601
850	724.15830466399	9.7997960850166	14.135434155823
700	611.02836539126	9.2655719210713	13.364857867461
1060	882.54021964581	10.460787162775	15.088861734946
905	765.63928239733	9.981595523578	14.397665529931
155	199.98958603368	6.4092087170655	9.244778873463
245	267.86754959732	7.0580643470557	10.180702024801
210	241.47056376701	6.8205149236928	9.8380556877175
35	109.4856346155	5.2536342554137	7.5779537096411

Using mass density values of 1185 kg/m^3 and 71 kg/m^3 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 ΔV requirements are 850 times less. The CW equations were used to calculate the ΔV 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:

- ΔV required for landing,
- ΔV 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 of determine analytically the ΔV 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 ΔV 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 ΔV In the sizing process for each of the vehicles, it was assumed that the ΔV 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 ΔV The escape velocity for Mars was used as the ΔV 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:

LH2: 1010 kg/m^3,
LOX: 62.2 kg/m^3.

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 ΔV 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 ΔV to help boost the vehicle into orbit. Assuming a non-rotating planet neglects this potential ΔV gain therefore increases the propellant requirement to launch the vehicle.

Landing ΔV Estimating or determining this ΔV is necessary to use the ideal rocket equation for determining propellant requirements. By neglecting the drag of the Martian atmosphere, the ΔV 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 ΔV The ΔV 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 ΔV 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: 15 0 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:

Mass of vehicle/payload combination	700 MT
Mass of vehicle/payload/fuel combination	1550 MT
Mass of propellant to land	850 MT
Volume of propellant to land	1000 m^3

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:	8 0 meters,
Protective shell radius:	8 0 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 m^3

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.

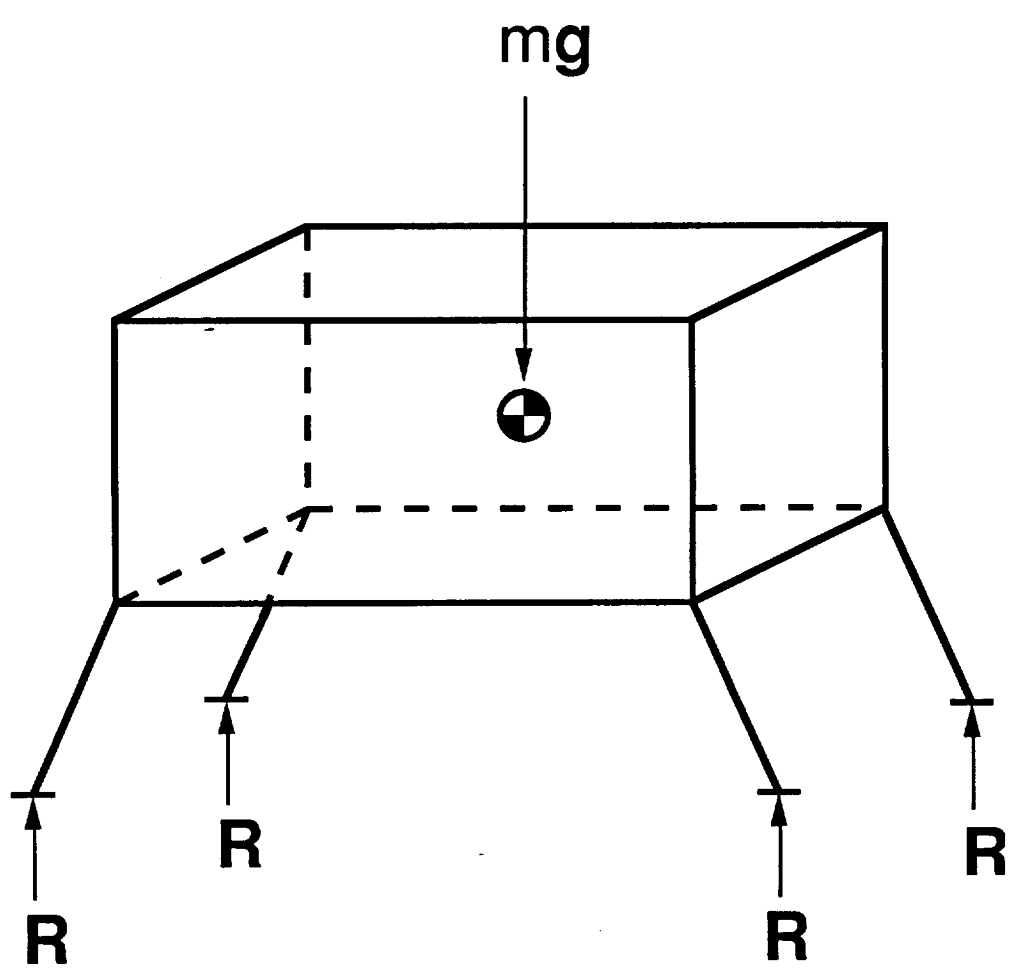
Vehicle mass prior to de-orbit:	240 MT
Vehicle mass when landed:	110 MT
Propellant mass to land:	130 MT
Vehicle mass at launch:	110 MT
Vehicle mass on orbit:	35 MT
Propellant mass to achieve orbit:	75 MT

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.



A vector analysis,

$$\sum \overline{\mathbf{M}} = \overline{\mathbf{r}} \times \overline{\mathbf{F}} = \overline{0} \qquad \text{(Eq. 1)}$$

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

$$\overline{\mathbf{R}} = \frac{\mathrm{mg'}}{\text{number of legs}} \qquad \text{(Eq. 2)}$$

where

$\overline{\mathbf{r}}$ = vector from any point to point of force reaction,

$\overline{\mathbf{F}}$ = any reaction force $\overline{\mathbf{R}}$,

$\mathbf{m}$ = vehicle mass, and

$\mathbf{g}'$ = gravitational acceleration.

The axial stress in each leg will be:

$$\sigma = \frac{\mathbf{F}}{\mathbf{A}} \qquad \text{(Eq. 3)}$$

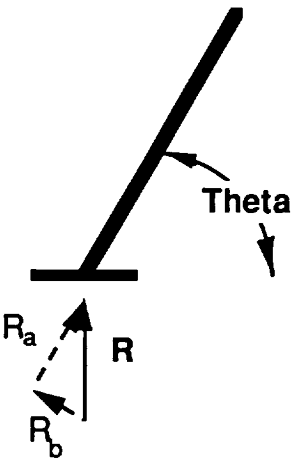
which leads to:

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

where

d = diameter of a leg, and

θ = the angle between the leg and the foot plate (see illustration).



Therefore,

$$\sigma_{axial} = \frac{.38\ mg}{\#\ of\ legs \times \pi(d/2)^2} \sin\ \theta \qquad \text{(Eq. 5)}$$

The bending stress in the leg will be:

$$\sigma_{bending} = \frac{\mathbf{Mc}}{\mathbf{I}} \qquad \text{(Eq. 6)}$$

where

M = moment induced by $\mathbf{R_b}$,

C = distance from neutral axis of member to point of stress,

and

I = moment of inertia of the leg.

Since maximum stress occurs when

$\mathbf{M = R_b l}$ (where **l** = length of leg)

C = radius of leg (Eqs. 7)

For a cylindrical leg,

$$I = \pi \frac{d^4}{64} \qquad \text{(Eq. 8)}$$

Thus

$$\sigma_{bending} = \frac{R_b l \frac{d}{2}}{\pi \frac{d^4}{64}} \qquad \text{(Eq. 9)}$$

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

$$\sigma_{total} = \sigma_{axial} + \sigma_{bending} \qquad \text{(Eq. 10)}$$

A FORTRAN program was written which calculates σ_{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:

$$\sigma_{total} = \frac{0.38 \ mg \ \sin\theta(4)}{\pi d^2} + \frac{32m(0.38g) \ \cos\theta(l)}{\pi d^3} \qquad \text{(Eq. 11)}$$

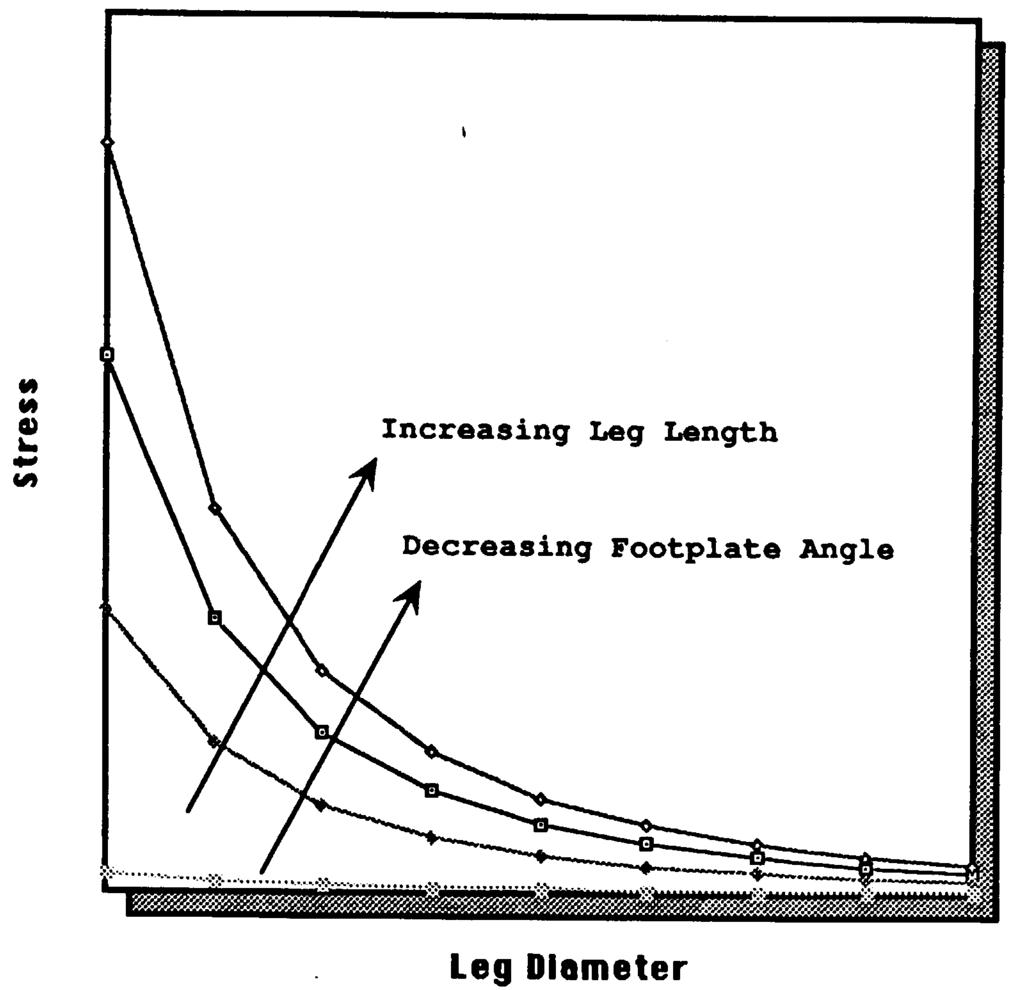


Figure H1. Trend in Leg Stresses

Appendix G: INSERT TK! Solver Model

Purpose: This TK! Solver model will be used to determine the minimum ΔV 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
- $\Delta V2$ (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.

Theory: A patch conic - Hohmann Transfer is used, assuming Earth and Mars are coplannar. See: **Fundamental of Astrodynamics**, by R. R. Bate, D. D. Mueller, J. E. White, Dover Publications, Inc., New York, 1971.

Listing: See below

Sample run: See below

Verification: N/A

VARIABLE SHEET

St	Input	Name	Output	Unit	Comment
	1.3272E11	mus		km^3/s^2	Solar Gravitational Parameter
	1	rp1		AU	Radius of Planet 1's Orbit
		Vp1	29.804744	km/s	Velocity of Planet 1
		Vt1	32.751467	km/s	Velocity in Hohmann at Planet 1
		Vp2	24.145542	km/s	Velocity of Planet 2
		Vt2	21.494822	km/s	Velocity in Hohmann at Planet 2
		at	1.2618455	AU	Semi-major Axis of Transfer Orbit
		TOF	258.35127	day	Time of Flight
		Phase	44.344806	deg	Phase Angle at Burn 1
	398603.22	mu1		km^3/s^2	Gravity Parameter of Planet 1
	6672.756	rper1		km	Periapse Radius at Planet 1
		Vorb1	25357.289	ft/s	Circ. Speed at Periapse - Planet 1
		Vper1	11.320557	km/s	Periapse Approach Speed - Planet 1
		Vinf1	9667.726	ft/s	Hyperbolic Excess Speed - Planet 1
L		Dv1	3.5916556	km/s	Burn Magnitude an Planet 1
	42905.686	mu2		km^3/s^2	Gravity Parameter of Planet 2
	3309.524	rPlanet		km	Radius of Planet 2
	1.523691	rp2		AU	Radius of Planet 2's Orbit
L		Dv2	2.0984904	km/s	Burn Magnitude at Planet 2
L		rper2	3640.4764	km	Periapse Radius at Planet 2
		Vorb2	11263.24	ft/s	Circ. Speed at Periapse - Planet 2
		Vper2	5.531526	km/s	Periapse Approach Speed - Planet 2
		Vinf2	2.6507193	km/s	Hyperbolic Excess Speed - Planet 2
L		DV	5.690146	km/s	Total Mission Delta V
L	1.1	Factor			

```
───────────── RULE SHEET ─────────────
S Rule──────────────────────────────────
  "Patched Conic - Hohmann to Mars

* Vp1 = sqrt(mus/rp1)
* Vt1 = sqrt(mus*(2/rp1 - 1/at))
* at = (rp1 + rp2)/2
* TOF = pi()*sqrt(at^3/mus)
* Phase = pi() - sqrt(mus/(rp2^3))*TOF

* Vt2 = sqrt(mus*(2/rp2 - 1/at))
* Vp2 = sqrt(mus/rp2)
* Vinf1 = abs(Vt1 - Vp1)
* Vinf2 = abs(Vt2 - Vp2)
* Vinf1^2/2 = Vper1^2/2 - mu1/rper1
* Vinf2^2/2 = Vper2^2/2 - mu2/rper2
* Vorb1 = sqrt(mu1/rper1)
* rper2 = Factor*rPlanet
* Vorb2 = sqrt(mu2/rper2)
* Dv1 = abs(Vper1 - Vorb1)
* Dv2 = abs(Vper2 - Vorb2)
* DV = Dv1 + Dv2
```

Appendix H: DELV TK! Solver Model

Purpose: This TK! Solver model will be used to determine the minimum ΔV required to transfer between Phobos orbit and the Mars parking orbits whose inclination are the same as base latitudes. The ΔV 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:

ΔV_{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

VARIABLE SHEET

St	Input	Name	Output	Unit	Comment
	42828.235	mu		km^3/s^2	Mars' Gravitational constant
	3513.481	r1		km	Radius of Low Mars Orbit
	9408	r2		km	Radius of PHOBOS orbit
		Vc1		km/s	Circular Velocity at LMO
		Vc2		km/s	Circular Velocity at PHOBOS
		Vt1		km/s	Velocity at perigee of transfer orbit
		Vt2		km/s	Velocity of apogee of transfer orbit
		at		km	radius constans
L	0	dil		deg	Plane change at first burn
		dih		deg	Plane change at second burn
	0	ditotal		deg	Total plane change
		dV1		km/s	Change of velocity at first burn
		dV2		km/s	Change of velocity at second burn
L		dVtotal		km/s	Total velocity change for the transfer

RULE SHEET

```
S Rule
  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*Vc1-2*Vt1*Vc1*cos(dil))
  dV2 = sqrt(Vt2*Vt2+Vc2*Vc2-2*Vt2*Vc2*cos(ditotal-dil))
  ditotal = dil+dih
  dVtotal = dV1+dV2
```

dVtotal vs. dil for 25 deg

dil	dVtotal
0	1.69276934
1	1.67051331
2	1.65468931
3	1.64510352
4	1.64144281
5	1.64331157
6	1.65027071
7	1.66187215
8	1.67768554
9	1.69731599
10	1.72041415
11	1.74668018
12	1.7758636
13	1.80776052
14	1.8422095
15	1.87908655
16	1.91829997
17	1.95978507
18	2.00349888
19	2.049415
20	2.09751845
21	2.14780074
22	2.20025508
23	2.25487202
24	2.31163558
25	2.37052011

dVtotal vs. dil for 30 deg

dil	dVtotal
0	1.82326679
1	1.79985255
2	1.78273643
3	1.77171253
4	1.76645447
5	1.76655208
6	1.77155027
7	1.78098357
8	1.7944027
9	1.81139243
10	1.83158162
11	1.85464742
12	1.88031538
13	1.90835706
14	1.93858651
15	1.97085604
16	2.00505211
17	2.04109128
18	2.07891645
19	2.11849339
20	2.15980738
21	2.20286003
22	2.24766605
23	2.29424993
24	2.34264259
25	2.39287777
26	2.44498844
27	2.49900319
28	2.55494282
29	2.61281727
30	2.67262316

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

St	Input	Name	Output	Unit	Comment
					MASS SIZING PROCEDURE
	2305	delV		m/s	Change of velocity for the transfer
		c	4414.5	m/s	Exit velocity
		ratio	1.4919159		Mass ratio
		Mi2	59.676638	MT	Initial mass for the return trip
	40	Mf2		MT	Final mass for the return trip
		Mp2	19.676638	MT	Mass of fuel for the return trip
		Mpaux2	21.644302	MT	Additional fuel mass
		Mf1	181.6443	MT	Final mass for the outgoing
L	1550	Mpl		MT	Mass of the payload
		Mi1	270.99803	MT	Initial mass for the outgoing
		Mpaux1	98.289101	MT	Additional fuel mass
		Mp1	89.353728	MT	Mass of fuel for the outgoing
L		Mptot	119.9334	MT	Total fuel mass
					TANK SIZING PROCEDURE
		Mfl	18.451293	MT	Mass of fuel
	5.5	OFR			Ox/Fuel ratio
		Mox	101.48211	MT	Mass of Ox
		Vox	171.27782	m^3	Volume of Ox
	.5925	Denox		MT/m^3	Density of Ox
		Vfl	519.75472	m^3	Volume of fuel
	.0355	Denfl		MT/m^3	Density of fuel
L		TDox	5.4140496	m	Ox tank diameter
L		TDfl	7.8093402	m	Fuel tank diameter
	450	Isp			

RULE SHEET

```
S Rule
  ratio   = exp(delV/c)
  Mi2     = ratio * Mf2
  Mp2     = Mi2 - Mf2
  Mpaux2  = 1.1 * Mp2
  Mf1     = Mpaux2 + Mf2 + Mpl
  Mi1     = ratio * Mf1
  Mpaux1  = 1.1 * Mp1
  Mp1     = Mi1 - Mf1
  Mptot   = Mpaux1 + Mpaux2
  Mfl     = Mptot / (OFR + 1.0)
  Mox     = Mptot - Mfl
  Vox     = Mox / Denox
  Vfl     = Mfl / Denfl
  TDox    = 2 * ((3/(4 * pi()) * (Vox/2))^(.33))
  TDfl    = 2 * ((3/(4 * pi()) * (Vfl/2))^(.33))
  c       = Isp * 9.81
```

TABLE: Table1

Screen or Printer:	Screen
Title:	Payload Mass versus Total Propellant Mass for One
Vertical or Horizontal:	Vertical
Row Separator:	
Column Separator:	:
First Element:	1
Last Element:	

List	Numeric Format	Width	Heading
Mpl		15	Payload (MT)
Mptot		15	Propellant (MT)
TDox		15	Oxid tank dia
TDfl		15	Fuel tank dia

LIST SHEET

Name	Elements	Unit	Comment
delV	3	m/s	delta V for various incl changes: 0, 25, 30
Mpl	9	MT	Payload Mass
Mptot	9	MT	Total Propellant Mass
Mf2	8	MT	Final mass of OTV
TDox	9	m	Oxidizer tank diameter
TDfl	9	m	Fuel tank diameter

Appendix J: Computation of ΔV's for the OMV

J1 ΔV 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 ΔV'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_o = 0, z_o = 0, x = -11000, z = 6000), and the resulting velocities are displayed. A TOF of 40 minutes or less produced excessive ΔV'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 m
z = 6000 m
a = 9408 km

TIME min	DVtot m/s	DVo m/s	DV m/s	Vxo m/s	Vzo m/s
0					
9.56541667	43.8560753	21.8624009	21.9936744	-20.433787	7.77334719
19.1308333	22.2500571	10.995655	11.2544021	-10.723759	2.4301083
28.69625	15.2200171	7.42087804	7.79913906	-7.3967008	.598538232
38.2616667	11.8350728	5.67431276	6.16076005	-5.6644391	-.33459632
47.8270833	9.89811159	4.65823572	5.23987587	-4.5718556	-.89291446
57.3925	8.67060911	4.00331294	4.66729618	-3.8023064	-1.2525895
66.9579167	7.83386817	3.54948221	4.28438597	-3.2211528	-1.4909723
76.5233333	7.22887852	3.21623509	4.01264343	-2.761624	-1.6485148
86.06875	6.76885888	2.95916284	3.80969604	-2.386802	-1.7492344
95.6541667	6.4036367	2.75229729	3.65133941	-2.0744699	-1.8087883
105.219583	6.10304502	2.5798613	3.52318372	-1.8102838	-1.8380851
114.785	5.84840679	2.43200617	3.41640062	-1.5844102	-1.8450741
124.350417	5.62786687	2.30244843	3.32541846	-1.389756	-1.8357143
133.915833	5.43371521	2.18709676	3.24661845	-1.2209846	-1.8145492
143.48125	5.26080438	2.08322929	3.17757509	-1.0739433	-1.7850742
153.046667	5.10559577	1.98899109	3.11660468	-.94531509	-1.74999
162.612083	4.96557663	1.90308332	3.06249331	-.83239734	-1.7113856
172.1775	4.83890106	1.82456968	3.01433138	-.73295555	-1.6708772
181.742917	4.7241677	1.75275544	2.97141226	-.64512185	-1.6297145
191.308333	4.62028172	1.68711184	2.93316988	-.56732201	-1.588865
200.87375	4.52636925	1.62722909	2.89914016	-.49822106	-1.5490805
210.439167	4.44172471	1.57278762	2.86893709	-.43668133	-1.5109501
220.004583	4.36577909	1.52354112	2.84223797	-.3817295	-1.4749441
229.57	4.29808191	1.47930743	2.81877449	-.33252995	-1.4414487

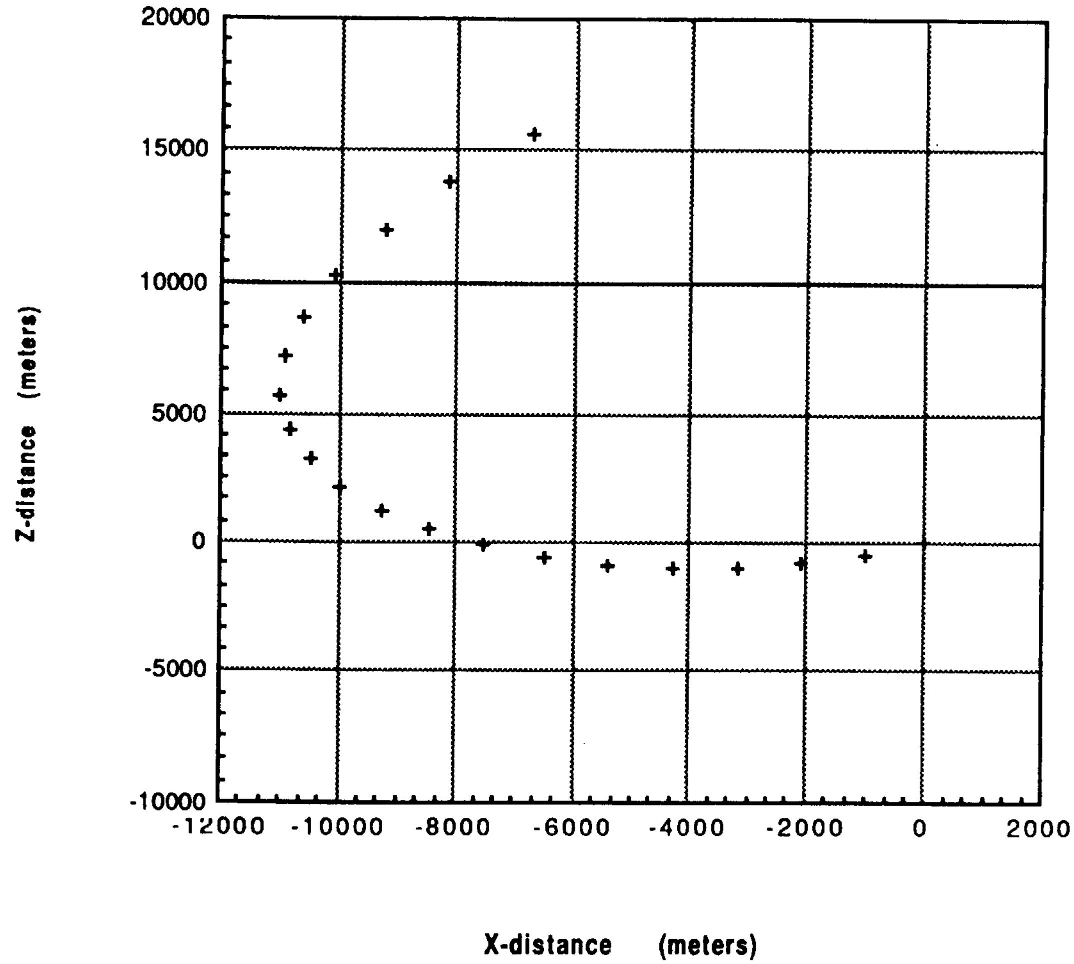


Figure J1. Flight Path for 76.52 min TOF

Table J2. Trajectory for TOF = 76.52 min

v_{xo} = -2.761 m/sec

v_{zo} = -1.6485 m/sec

TIME min	X meters	Z meters	Vx m/s	Vz m/s	DVtot m/s
0	0	0	-2.761624	-1.6485148	6.43247018
5.78947368	-1001.0907	-495.13225	-2.990259	-1.2007184	6.43855935
11.5789474	-2069.0482	-833.28934	-3.1464083	-.74520289	6.44968717
17.3684211	-3178.5053	-1012.2973	-3.229068	-.28489657	6.45784673
23.1578947	-4303.828	-1031.0054	-3.2377067	.177241301	6.45878952
28.9473684	-5419.2803	-889.29322	-3.1722689	.638239718	6.45207194
34.7368421	-6499.1897	-588.07192	-3.0331753	1.09513501	6.44105652
40.5263158	-7518.1121	-129.27796	-2.8213201	1.54498988	6.43288566
46.3157895	-8450.9956	484.139153	-2.5380652	1.9849123	6.4382922
52.1052632	-9273.3414	1248.23588	-2.1852317	2.41207409	6.47097612
57.8947368	-9761.3611	2158.09999	-1.7650879	2.82372911	6.5462473
63.6842105	-10492.13	3207.88212	-1.2803347	3.2172309	6.67886871
69.4736842	-10843.735	4390.83344	-.73408861	3.59004971	6.880567
75.2631579	-10995.414	5699.34895	-.12986129	3.93978875	7.15816347
81.0526316	-10927.689	7125.01647	.528462785	4.26419964	7.51305618
86.8421053	-10622.496	8658.67064	1.23665137	4.56119677	7.94210248
92.6315789	-10063.294	10290.4519	1.99015165	4.82887082	8.4391363
98.4210526	-9235.1776	12009.8698	2.78411951	5.06550097	8.99642726
104.210526	-8124.9683	13805.8705	3.61345069	5.26956594	9.60570699
110	-6721.3022	15666.9079	4.47281356	5.43975387	10.2587477

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

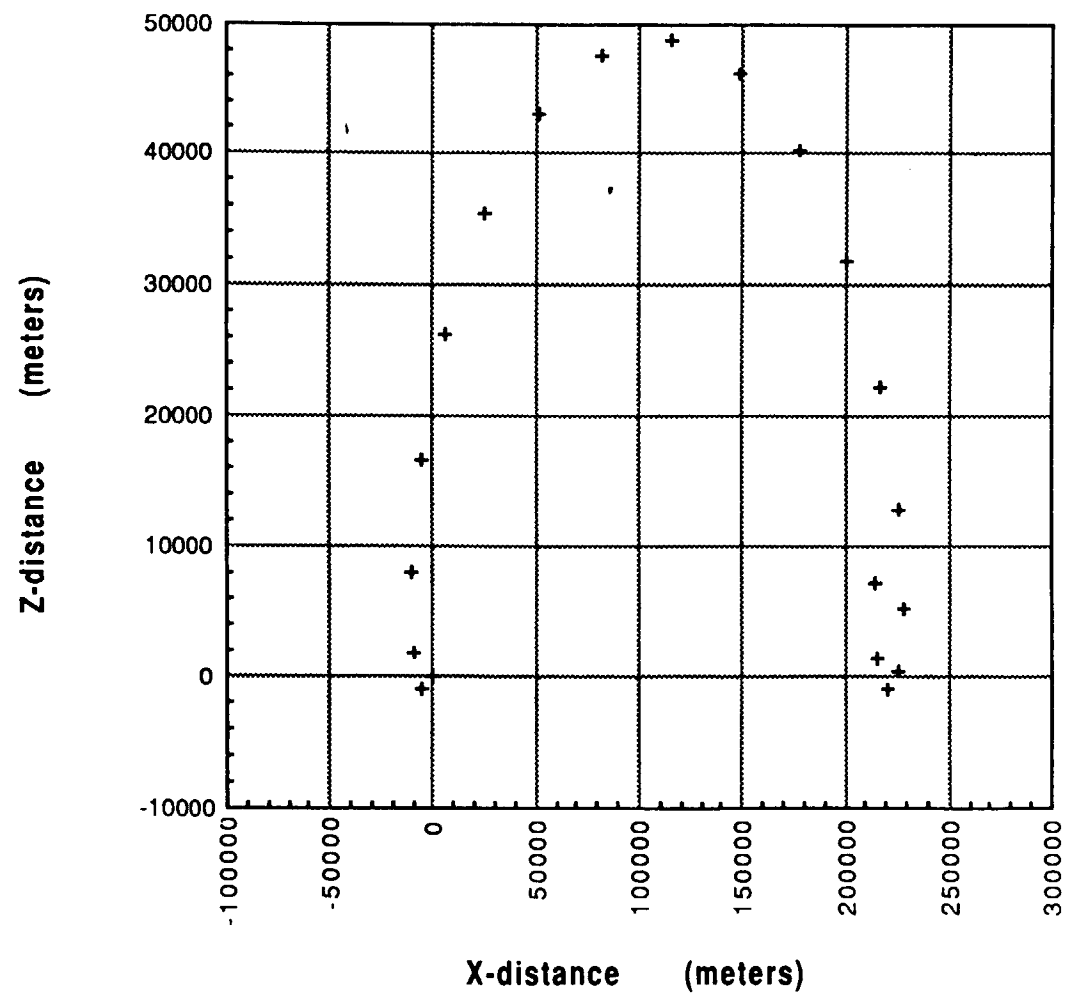


Figure J2. Abort Trajectory

Table J3. Contingency TOF = 535 min

TIME min	X meters	Z meters	Vx m/s	Vz m/s	DV m/s
0	0	0	-2.761624	-1.6485148	3.21623509
28.1578947	-5268.6951	-918.0412	-3.1855438	.575551993	3.2371205
56.3157895	-9788.3671	1895.81628	-1.8862014	2.71315052	3.30438219
84.4736842	-10777.146	8018.83157	.941195758	4.44313789	4.54173136
112.631579	-5983.627	16531.1112	4.87187282	5.50560884	7.35165787
140.789474	5974.88935	26153.8098	9.31530306	5.74094265	10.9422709
168.947368	25404.6663	35441.2578	13.6039261	5.11378386	14.5333269
197.105263	51489.5196	42998.152	17.0934391	3.71835391	17.4931935
225.263158	82413.4374	47689.1798	19.2595939	1.76429571	19.3402352
253.421053	115633.409	48809.5828	19.7769574	-.45482209	19.7821866
281.578947	148261.475	46191.0369	18.5673032	-2.6056095	18.7497338
309.736842	177498.602	40226.9403	15.8137892	-4.3649423	16.4051409
337.894737	201055.195	31813.3112	11.9286655	-5.4665064	13.1224092
366.052632	217495.075	22214.1742	7.49611526	-5.7505074	9.44775524
394.210526	226451.248	12871.6592	3.18206414	-5.1685789	6.06957492
422.368421	228681.033	5189.34225	-.36536479	-3.810147	3.82762483
450.526316	225952.295	321.379009	-2.6132223	-1.8792964	3.21880187
478.684211	**220777.842**	**-1000.8903**	-3.2238006	.333891052	**3.24104513**
506.842105	216037.914	1421.18598	-2.1053693	2.49691624	3.2660635
535	214547.471	7223.72629	.574043574	4.2848163	4.32309805

Table J4. Varied Return TOF from

x = 11000 m
z = 6000 m
a = 9402 km

TIME min	DVtot m/s	DVo m/s	DV m/s	Vxo m/s	Vzo m/s
0					
16.3976186	25.8224372	12.7995295	13.0229077	12.359025	-3.3290325
32.7952371	13.5112473	6.54216565	6.96908169	6.54087128	-.13013163
49.1928557	9.69183778	4.54834023	5.14349755	4.44653046	.956956517
65.5904743	7.93936978	3.60642129	4.33294849	3.29532531	1.4653005
81.9880929	**6.95508613**	3.0628704	3.89221573	**2.53817484**	**1.71430557**
98.3857114	6.31516016	2.70088793	3.61427223	1.99383102	1.82193124
114.78333	5.85133524	2.4327743	3.41856095	1.58355936	1.84681632
131.180949	5.48963938	2.21945111	3.27018828	1.26592098	1.82302147
147.578567	5.19498137	2.04232333	3.15265804	1.01581829	1.77177814
163.976186	4.94931498	1.8915336	3.05738138	.81651618	1.70666753
180.373804	4.74244077	1.76307698	2.97936379	.656156687	1.63642868
196.771423	4.56800821	1.65263833	2.91536989	.526024114	1.56668625
213.169041	4.42175722	1.55363053	2.86312719	.419568231	1.50109681
229.56666	4.30075622	1.47977882	2.8209774	.331794757	1.4421018

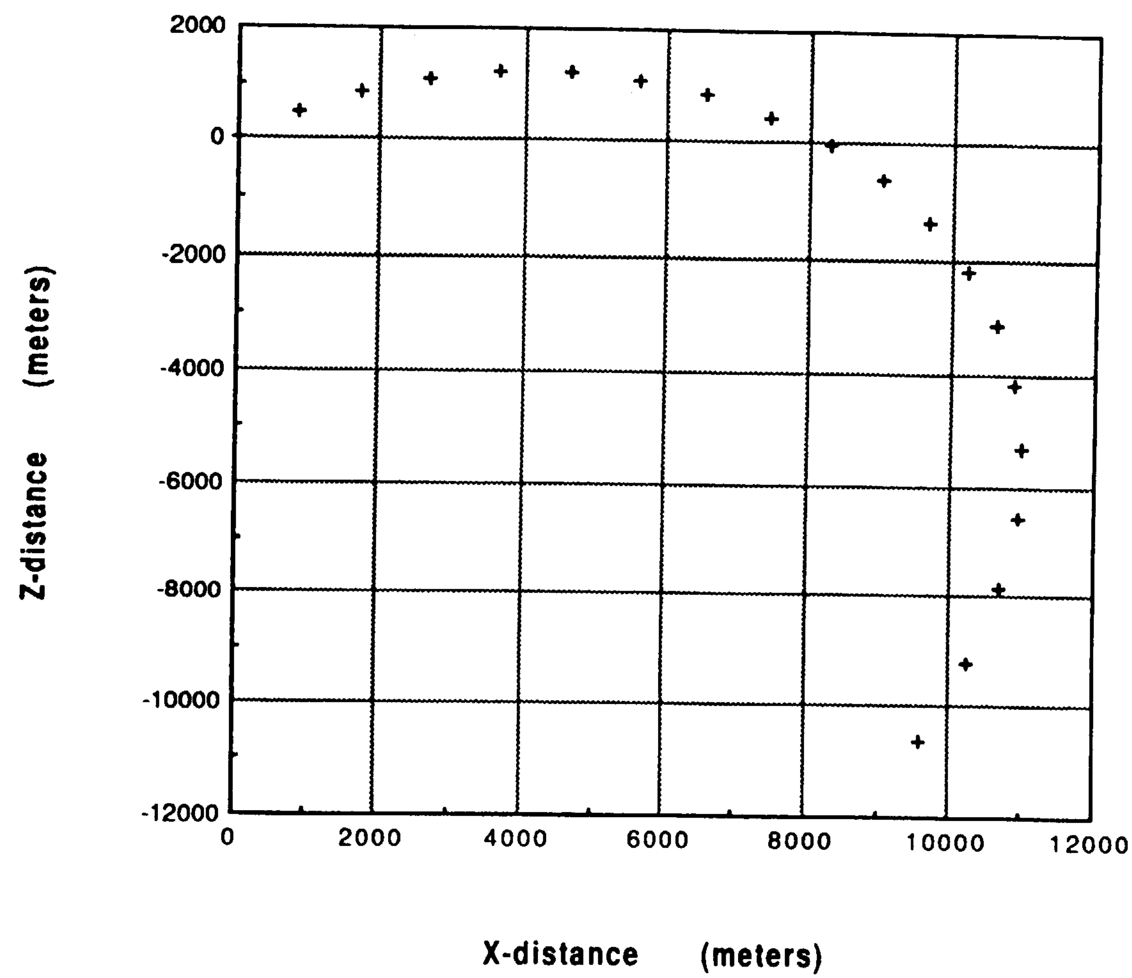


Figure J3. Flight Path for 81.99 min TOF

Table J5. Return Trajectory for TOF = 81.99 min

$v_{xo} = 2.538$ m/sec
$v_{zo} = 1.714$ m/sec

Time min	X meters	Z meters	Vx m/s	Vz m/s	DVtot m/s
0	0	0	2.53817484	1.71430557	6.12574081
5.26315789	838.174614	482.409487	2.76114817	1.33959835	6.13182185
10.5263158	1738.06756	845.311277	2.92888413	.957759451	6.14437417
15.7894737	2682.08666	1086.77338	3.04048975	.570821696	6.15647919
21.0526316	3652.40482	1205.5103	3.09537086	.180845036	6.16351964
26.3157895	4631.05491	1200.88993	3.09323529	-.2100944	6.16323235
31.5789474	5600.02548	1072.93686	3.03409441	-.59991534	6.15570523
36.8421053	6541.35658	822.33228	2.91826307	-.98654249	6.14337773
42.1052632	7437.23544	450.410342	2.74635792	-1.3679175	6.13104254
47.3684211	8270.09123	-40.848938	2.51929415	-1.7420102	6.12578446
52.6315789	9022.68865	-648.83022	2.23828059	-2.1068288	6.1367324
57.8947368	9678.2197	-1370.2968	1.90481327	-2.4604312	6.17446731
63.1578947	10220.3931	-2201.4077	1.52066748	-2.8009349	6.24997976
68.4210526	10633.5211	-3137.7384	1.09788833	-3.1265272	6.37325911
73.6842105	10902.6028	-4174.3041	.608779808	-3.4354747	6.55186725
78.9473684	11013.4045	-5305.5863	.085892568	-3.7261326	6.78999281
84.2105263	10952.5347	-6525.5625	-.47798968	-3.9969535	7.08830348
89.4736842	10707.5163	-7827.7377	-1.079865	-4.2464957	7.44451783
94.7368421	10266.8521	-9205.1796	-1.7165291	-4.4734307	7.85432678
100	9620.08686	-10650.555	-2.3845926	-4.6765503	8.31228983

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.

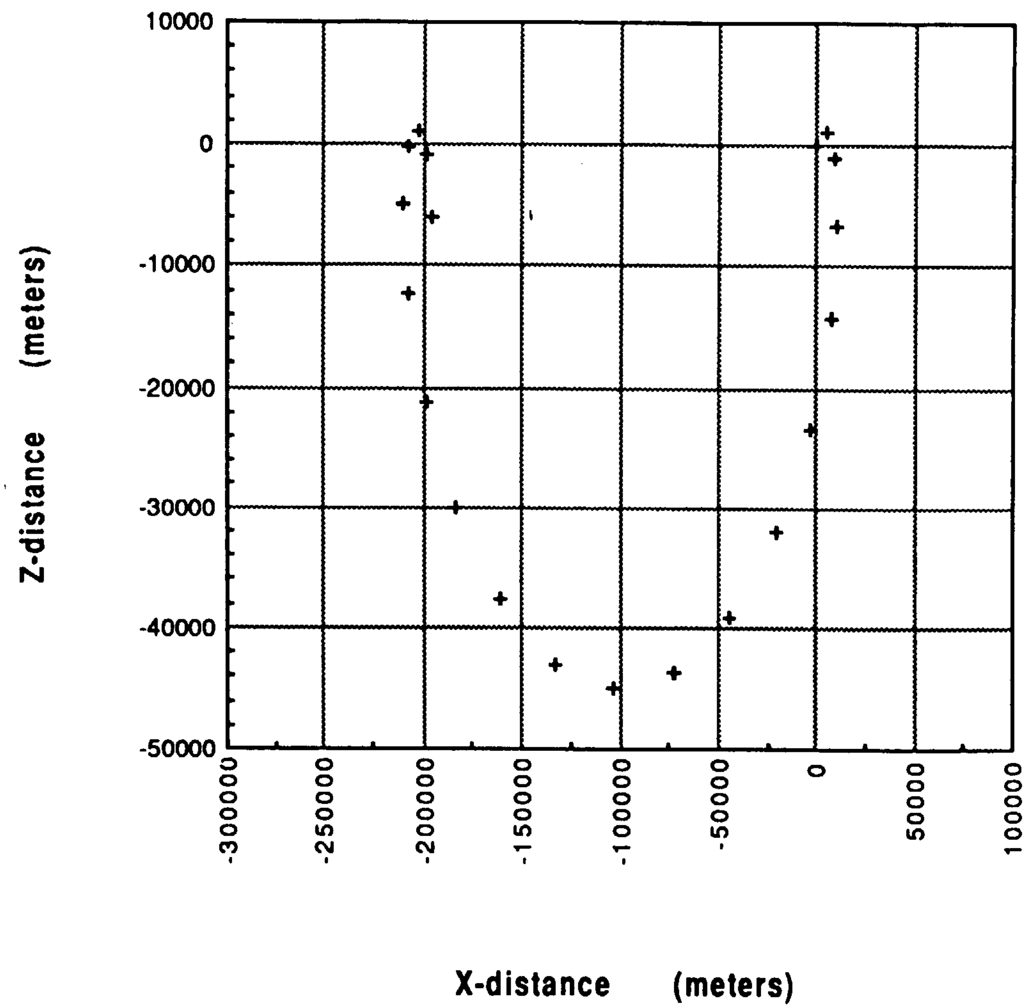


Figure J4. Contingency Flight Trajectory

Table J6. Return Contingency TOF = 535 min

Time min	X Meters	Z meters	Vx m/s	Vz m/s	DV m/s
0	0	0	2.53817484	1.71430557	3.0628704
28.1578947	4972.21702	1170.11443	3.07901061	-.34676542	3.09847585
56.3157895	9492.7326	-1142.1579	2.01026078	-2.3556414	3.09680401
84.4736842	10944.757	-6588.7749	-.50720695	-4.0099475	4.0418978
112.631579	7173.36819	-14349.915	-4.0944646	-5.0606782	6.50961632
140.789474	-3190.1302	-23257.374	-8.2115599	-5.3496782	9.8004476
168.947368	-20522.194	-31970.407	-12.23879	-4.8334473	13.1586545
197.105263	-44150.372	-39177.531	-15.569978	-3.5896882	15.9784249
225.263158	-72454.529	-43793.935	-17.703714	-1.8056111	17.7955538
253.421053	-103110.7	-45124.759	-18.318831	.250245675	18.3205407
281.578947	-133440.9	-42969.689	-17.322742	2.26843562	17.4706375
309.736842	-160816.2	-37653.104	-14.865376	3.94518177	15.3799822
337.894737	-183052.45	-29975.254	-11.316615	5.02810112	12.383359
366.052632	-198739.03	-21091.806	-7.2106179	5.35419318	8.98111328
394.210526	-207451.16	-12339.891	-3.1654165	4.87437474	5.81200401
422.368421	-209813.87	-5036.8437	.210107443	3.66086784	3.66689221
450.526316	-207407.87	-281.91486	2.40787165	1.89632893	3.0649485
478.684211	**-202531.7**	**1209.18595**	3.09706977	-.15364447	**3.10087855**
506.842105	-197855.66	-787.98103	2.17396403	-2.1804914	3.07906841
535	-196019.96	-5972.8032	-.22250015	-3.8791318	3.88550768

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 ΔV required to de-orbit. The first two parameters are inputs to the Mars Descent program while the ΔV 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

ΔV 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

Theory: Standard orbital mechanics theory using the two-body and impulsive burn assumptions is used, but a spherical, rotating Mars is incorporated. A good reference is: **Fundamentals of Astrodynamics**, by R. R. Bate, D. D. Mueller, J. E. White, Dover Publications, Inc., New York, 1971.

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

St	Input	Name	Output	Unit	Comment
	42828	mu		km^3/s^2	Gravitational parameter of Mars
	6720	a		km	Major axis of parking orbit
		e	.4203869		Eccentricity of parking orbit
		Ra	9545	km	Apoapsis of parking orbit
	3895	Rp		km	Periapsis of parking orbit
L	180	f		deg	True anomaly of parking orbit
		Vi	1.6126682	km/s	Velocity before burn
		r	9545	km	Radius at burn
		gammaI	5.091E-15	deg	Initial flight path angle at burn
	6500	at		km	Major axis of transfer orbit
	3395	Rpt		km	Periapsis of transfer orbit
		Rat	9605	km	Apoapsis of transfer orbit
		et	.47769231		Eccentricity of transfer orbit
		ft	173.27854	deg	True anomaly of transfer orbit
		Vt	1.5443413	km/s	Velocity after burn
		gammaT	-6.072074	deg	Flight path angle after burn
		ftc	186.72146	deg	Corrected true anomaly after burn
L		DeltaV	180.59345	m/s	Delta velocity of burn
L		Dgamma	-6.072074	deg	Change in flight path angle at deorbit
	3495	re		km	Radius at atmospheric entry
L		Ve	4233.1113	m/s	Atmospheric entry velocity
		fe	24.287816	deg	True anomaly at atmospheric entry
L		gammaE	7.7944227	deg	Flight path angle at entry

RULE SHEET

```
S Rule
* r = a * (1 - e^2) / (1 + e * cos(f))
* Vi = sqrt(mu * (2/r - 1/a))
* tan(gammaI) = e * sin(f) / (1 + e * cos(f))
* Rat = 2 * at - Rpt
* et = (Rat - Rpt) / (2 * at)
* r = at * (1 - et^2) / (1 + et * cos(ft))
* Vt = sqrt(mu * (2/r - 1/at))
* tan(gammaT) = et * sin(ftc) / (1 + et * cos(ftc))
* ftc = 2 * PI() - ft
* DeltaV = sqrt(Vi^2 + Vt^2 - 2 * Vi * Vt * cos(gammaT - gammaI))
* Ra = 2 * a - Rp
* e = (Ra - Rp) / (2 * a)
* Ve = sqrt(mu * (2/re - 1/at))
* re = at * (1 - et^2) / (1 + et * cos(fe))
* Dgamma = gammaT - gammaI
* tan(gammaE) = et * sin(fe) / (1 + et * cos(fe))
```

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:

1) Original source: "Mars Cargo Descent Vehicle Sizing Analysis." Report written by Preston Carter for the Universities Space Research Association at NASA Johnson Space Center, August 7, 1985.

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
*
```

```
*     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,*)'PULLOUT ALTITIUDE (KM)?'
      READ (5,*)HEQUIL
      HEQUIL = HEQUIL * 1000.0
      WRITE(6,*)'EMMISIVITY OF ABLATIVE SURFACE ?'
      READ(5,*)E
      WRITE (6,*)'OUTPUT TAPE NUMBER?'
      WRITE(6,*)'USE TAPE 6 FOR TTY OUTPUT'
      WRITE(6,*)'USE TAPE 9 FOR OUTPUT DIRECTED TO FILE MARS.DAT'
      WRITE(6,*)'OUTPUT TAPE NUMBER ?'
      READ (5,*)IUNIT
*
*      PROPULSIVE INFORMATION
*
      WRITE(6,*)'INITIAL PROPULSIVE DECELERATION ALTITUDE (KM)?'
      READ(5,*)PALT
      PALT = PALT*1000.0
      IF(PALT .EQ. 0.0) GOTO 333
      WRITE(6,*)'FUEL MASS FLOW RATE (KG/S)?'
      READ(5,*)FFR
      WRITE(6,*)'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,*)'UPPER TRAJECTORY OUTPUT INTERVAL (S) ?'
      READ(5,*)NSTEPS
      WRITE(6,*)'SIMULATION IN PROGRESS...'
      TIME = 0.0
      OPEN (9,FILE='MARS.DAT')
      WRITE(IUNIT,*)'    >>>  MARS DESCENT SIMULATION  <<<'
      WRITE(IUNIT,*)' '
      WRITE(IUNIT,*)'              DESCENT PROFILE'
      WRITE(IUNIT,*)' '
      WRITE (IUNIT,*)'M/(CL*S) (KG/M**2) = ',BALLCL
      WRITE(IUNIT,*)'INITIAL MASS (KG) = ',AMO
      WRITE(IUNIT,*)'COEFFICIENT OF LIFT = ',CL
      WRITE(IUNIT,*)'RADIUS OF VEHICLE (M) = ',RB
      WRITE (IUNIT,*)'L/D = ',LD
      WRITE (IUNIT,*)'EMMISIVITY OF ABLATIVE SURFACE = ',E
      WRITE (IUNIT,*)'H (M) =',H
      WRITE (IUNIT,*)'V (M/SEC) =',X(4)
      WRITE (IUNIT,*)'GAMA (DEG) =',ANGLE
      WRITE(IUNIT,*)'THRUST (N) = ',THRUST
```

```
      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
```

```
      END

      SUBROUTINE OUTPUT(TIME,X,DX,TW,IUNIT)
*
*     THIS IS AN OUTPUT ROUTINE FOR PRINTING AN EPHEMEROUS 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
1       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)
2       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)
3       U(I) = X(I) + F(I)
      CALL DERIV (U,F)
      DO 4 I=1,N
4       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
*
*     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     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*TWO*THREE/(SBK*E)
      TW4 = ABS(TW4)
      TW = SQRT(SQRT(TW4))
C
C     TW IS CONVERTED FROM RANKINE TO KELVIN
C
      TW = TW/1.8
C
 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,*)' '
      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
ABOVE
C      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

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

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

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

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

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

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

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) = 797.8786

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) = -1.788962
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) = -41.21057
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

TIME (SEC) = 1261.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 196968.5
X DOWNRANGE (KM) = 3200.875
Y CROSSRANGE (KM) = 16.83976
H ALTITUDE (KM) = 1.825250
V VELOCITY (M/SEC) = 156.4251
GAMA FLT. PATH ANGLE (DEG) = -45.84177
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.650902
STAGNATION TEMP (K) = 353.9706

TIME (SEC) = 1262.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 196581.5
X DOWNRANGE (KM) = 3200.980
Y CROSSRANGE (KM) = 16.84134
H ALTITUDE (KM) = 1.714750
V VELOCITY (M/SEC) = 149.2789
GAMA FLT. PATH ANGLE (DEG) = -46.73426
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.642900
STAGNATION TEMP (K) = 347.6528

TIME (SEC) = 1263.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 196194.5
X DOWNRANGE (KM) = 3201.080
Y CROSSRANGE (KM) = 16.84282
H ALTITUDE (KM) = 1.607750
V VELOCITY (M/SEC) = 142.1418
GAMA FLT. PATH ANGLE (DEG) = -47.66507
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.637941
STAGNATION TEMP (K) = 341.2604

TIME (SEC) = 1264.000 ROLL (DEG) = 0.0000000E+00
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 195807.5
X DOWNRANGE (KM) = 3201.172
Y CROSSRANGE (KM) = 16.84421
H ALTITUDE (KM) = 1.504500
V VELOCITY (M/SEC) = 135.0077
GAMA FLT. PATH ANGLE (DEG) = -48.63600
AZE (DEG) = 0.8574303
ACCELERATION (EARTH G) = -3.636083
STAGNATION TEMP (K) = 334.7930

TIME (SEC) = 1265.000 ROLL (DEG) = 0.0000000E+00
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
PROPULSION SYSTEMS ARE ON
MASS OF VEHICLE (KG) = 193872.5
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
PROPULSION SYSTEMS ARE ON
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
PROPULSION SYSTEMS ARE ON
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
PROPULSION SYSTEMS ARE ON
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 be 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: **Mechanics of Materials**, by J. M Gere and S. P. Timoshenko, Belmont, California; Brooks/Cole Publisher, 1984

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
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*SIN(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

DIAMETER	STRESS
0.20	0.40946E+09
0.25	0.21050E+09
0.30	0.12231E+09
0.35	0.77339E+08
0.40	0.52021E+08
0.45	0.36683E+08
0.50	0.26850E+08
0.55	0.20253E+08
0.60	0.15662E+08

THETA = 50.DEGREES

DIAMETER	STRESS
0.20	0.37338E+09
0.25	0.19210E+09
0.30	0.11171E+09
0.35	0.70687E+08
0.40	0.47582E+08
0.45	0.33578E+08
0.50	0.24595E+08
0.55	0.18566E+08
0.60	0.14368E+08

THETA = 55.DEGREES

DIAMETER	STRESS
0.20	0.33447E+09
0.25	0.17224E+09
0.30	0.10025E+09
0.35	0.63496E+08
0.40	0.42780E+08
0.45	0.30217E+08
0.50	0.22152E+08
0.55	0.16737E+08
0.60	0.12964E+08

THETA = 60.DEGREES

DIAMETER	STRESS
0.20	0.29300E+09
0.25	0.15107E+09
0.30	0.88034E+08
0.35	0.55822E+08
0.40	0.37653E+08
0.45	0.26625E+08
0.50	0.19541E+08
0.55	0.14781E+08

0.60	0.11461E+08

THETA = 65.DEGREES

DIAMETER	STRESS
0.20	0.24931E+09
0.25	0.12875E+09
0.30	0.75145E+08
0.35	0.47723E+08
0.40	0.32239E+08
0.45	0.22832E+08
0.50	0.16782E+08
0.55	0.12712E+08
0.60	0.98711E+07

THETA = 70.DEGREES

DIAMETER	STRESS
0.20	0.20372E+09
0.25	0.10545E+09
0.30	0.61684E+08
0.35	0.39261E+08
0.40	0.26580E+08
0.45	0.18864E+08
0.50	0.13895E+08
0.55	0.10546E+08
0.60	0.82061E+07

THETA = 75.DEGREES

DIAMETER	STRESS
0.20	0.15658E+09
0.25	0.81345E+08
0.30	0.47754E+08
0.35	0.30500E+08
0.40	0.20719E+08
0.45	0.14753E+08
0.50	0.10902E+08
0.55	0.83008E+07
0.60	0.64786E+07

THETA = 80.DEGREES

DIAMETER	STRESS
0.20	0.10825E+09
0.25	0.56622E+08
0.30	0.33460E+08
0.35	0.21507E+08
0.40	0.14700E+08
0.45	0.10530E+08
0.50	0.78256E+07
0.55	0.59919E+07
0.60	0.47018E+07

THETA = 85.DEGREES

DIAMETER	STRESS
0.20	0.59098E+08
0.25	0.31468E+08
0.30	0.18911E+08
0.35	0.12350E+08
0.40	0.85693E+07
0.45	0.62260E+07
0.50	0.46901E+07
0.55	0.36374E+07
0.60	0.28893E+07

THETA = 90.DEGREES

DIAMETER	STRESS
0.20	0.94927E+07
0.25	0.60754E+07
0.30	0.42190E+07
0.35	0.30997E+07
0.40	0.23732E+07
0.45	0.18751E+07
0.50	0.15188E+07
0.55	0.12552E+07
0.60	0.10548E+07

LEG LENGTH = 2.0

THETA = 45.DEGREES

DIAMETER	STRESS
0.20	0.54370E+09
0.25	0.27924E+09
0.30	0.16209E+09
0.35	0.10239E+09
0.40	0.68802E+08
0.45	0.48469E+08
0.50	0.35441E+08
0.55	0.26708E+08
0.60	0.20634E+08

THETA = 50.DEGREES

DIAMETER	STRESS
0.20	0.49542E+09
0.25	0.25459E+09
0.30	0.14787E+09
0.35	0.93457E+08
0.40	0.62836E+08
0.45	0.44292E+08
0.50	0.32405E+08
0.55	0.24434E+08
0.60	0.18887E+08

THETA = 55.DEGREES

DIAMETER	STRESS
0.20	0.44336E+09
0.25	0.22800E+09
0.30	0.13252E+09
0.35	0.83815E+08
0.40	0.56392E+08
0.45	0.39777E+08
0.50	0.29122E+08
0.55	0.21973E+08
0.60	0.16997E+08

THETA = 60.DEGREES

DIAMETER	STRESS
0.20	0.38793E+09
0.25	0.19967E+09
0.30	0.11616E+09
0.35	0.73534E+08
0.40	0.49519E+08
0.45	0.34959E+08
0.50	0.25617E+08
0.55	0.19345E+08
0.60	0.14977E+08

THETA = 65.DEGREES

DIAMETER	STRESS
0.20	0.32955E+09
0.25	0.16983E+09
0.30	0.98919E+08
0.35	0.62694E+08
0.40	0.42269E+08
0.45	0.29876E+08
0.50	0.21917E+08
0.55	0.16570E+08
0.60	0.12843E+08

THETA = 70.DEGREES

DIAMETER	STRESS
0.20	0.26866E+09
0.25	0.13869E+09
0.30	0.80924E+08
0.35	0.51377E+08
0.40	0.34697E+08
0.45	0.24565E+08
0.50	0.18050E+08
0.55	0.13669E+08
0.60	0.10611E+08

THETA = 75.DEGREES

DIAMETER	STRESS
0.20	0.20572E+09
0.25	0.10650E+09
0.30	0.62313E+08
0.35	0.39669E+08
0.40	0.26861E+08
0.45	0.19067E+08
0.50	0.14046E+08
0.55	0.10664E+08
0.60	0.82985E+07

THETA = 80.DEGREES

DIAMETER	STRESS
0.20	0.14122E+09
0.25	0.73502E+08
0.30	0.43228E+08
0.35	0.27658E+08
0.40	0.18821E+08
0.45	0.13424E+08
0.50	0.99356E+07
0.55	0.75771E+07
0.60	0.59229E+07

THETA = 85.DEGREES

DIAMETER	STRESS
0.20	0.75645E+08
0.25	0.39940E+08
0.30	0.23814E+08
0.35	0.15438E+08
0.40	0.10638E+08
0.45	0.76787E+07
0.50	0.57491E+07
0.55	0.44331E+07
0.60	0.35021E+07

THETA = 90.DEGREES

DIAMETER	STRESS
0.20	0.94927E+07
0.25	0.60754E+07
0.30	0.42190E+07
0.35	0.30997E+07
0.40	0.23732E+07
0.45	0.18751E+07
0.50	0.15188E+07
0.55	0.12552E+07
0.60	0.10548E+07

LEG LENGTH = 2.5

THETA = 45.DEGREES

DIAMETER	STRESS
0.20	0.67795E+09
0.25	0.34797E+09
0.30	0.20187E+09
0.35	0.12744E+09
0.40	0.85583E+08
0.45	0.60255E+08
0.50	0.44033E+08
0.55	0.33164E+08
0.60	0.25607E+08

THETA = 50.DEGREES

DIAMETER	STRESS
0.20	0.61745E+09
0.25	0.31707E+09
0.30	0.18403E+09
0.35	0.11623E+09
0.40	0.78091E+08
0.45	0.55005E+08
0.50	0.40215E+08
0.55	0.30302E+08
0.60	0.23407E+08

THETA = 55.DEGREES

DIAMETER	STRESS
0.20	0.55226E+09
0.25	0.28375E+09
0.30	0.16478E+09
0.35	0.10413E+09
0.40	0.70004E+08
0.45	0.49337E+08
0.50	0.36091E+08
0.55	0.27209E+08
0.60	0.21030E+08

THETA = 60.DEGREES

DIAMETER	STRESS
0.20	0.48286E+09
0.25	0.24828E+09
0.30	0.14429E+09
0.35	0.91247E+08
0.40	0.61385E+08
0.45	0.43293E+08
0.50	0.31692E+08
0.55	0.23910E+08
0.60	0.18493E+08

THETA = 65.DEGREES

DIAMETER	STRESS
0.20	0.40978E+09
0.25	0.21091E+09
0.30	0.12269E+09
0.35	0.77665E+08
0.40	0.52299E+08
0.45	0.36920E+08
0.50	0.27052E+08
0.55	0.20428E+08
0.60	0.15814E+08

THETA = 70.DEGREES

DIAMETER	STRESS
0.20	0.33359E+09
0.25	0.17194E+09
0.30	0.10016E+09
0.35	0.63493E+08
0.40	0.42814E+08
0.45	0.30265E+08
0.50	0.22206E+08
0.55	0.16791E+08
0.60	0.13016E+08

THETA = 75.DEGREES

DIAMETER	STRESS
0.20	0.25486E+09
0.25	0.13166E+09
0.30	0.76873E+08
0.35	0.48837E+08
0.40	0.33004E+08
0.45	0.23381E+08
0.50	0.17191E+08
0.55	0.13026E+08
0.60	0.10118E+08

THETA = 80.DEGREES

DIAMETER	STRESS
0.20	0.17419E+09
0.25	0.90381E+08
0.30	0.52996E+08
0.35	0.33810E+08
0.40	0.22942E+08
0.45	0.16318E+08
0.50	0.12046E+08
0.55	0.91624E+07
0.60	0.71439E+07

THETA = 85.DEGREES

DIAMETER	STRESS
0.20	0.92192E+08
0.25	0.48413E+08
0.30	0.28717E+08
0.35	0.18525E+08
0.40	0.12706E+08
0.45	0.91314E+07
0.50	0.68081E+07
0.55	0.52287E+07
0.60	0.41150E+07

THETA = 90.DEGREES

DIAMETER	STRESS
0.20	0.94927E+07
0.25	0.60753E+07
0.30	0.42190E+07
0.35	0.30997E+07
0.40	0.23732E+07
0.45	0.18751E+07
0.50	0.15188E+07
0.55	0.12552E+07
0.60	0.10547E+07

LEG LENGTH = 3.0

THETA = 45.DEGREES

DIAMETER	STRESS
0.20	0.81220E+09
0.25	0.41671E+09
0.30	0.24165E+09
0.35	0.15249E+09
0.40	0.10236E+09
0.45	0.72041E+08
0.50	0.52625E+08
0.55	0.39619E+08
0.60	0.30579E+08

THETA = 50.DEGREES

DIAMETER	STRESS
0.20	0.73949E+09
0.25	0.37955E+09
0.30	0.22019E+09
0.35	0.13900E+09
0.40	0.93345E+08
0.45	0.65719E+08
0.50	0.48026E+08
0.55	0.36170E+08
0.60	0.27927E+08

THETA = 55.DEGREES

DIAMETER	STRESS
0.20	0.66116E+09
0.25	0.33951E+09
0.30	0.19705E+09
0.35	0.12445E+09
0.40	0.83616E+08
0.45	0.58897E+08
0.50	0.43060E+08
0.55	0.32445E+08
0.60	0.25063E+08

THETA = 60.DEGREES

DIAMETER	STRESS
0.20	0.57779E+09
0.25	0.29688E+09
0.30	0.17241E+09
0.35	0.10896E+09
0.40	0.73251E+08
0.45	0.51627E+08
0.50	0.37768E+08
0.55	0.28474E+08
0.60	0.22008E+08

THETA = 65.DEGREES

DIAMETER	STRESS
0.20	0.49002E+09
0.25	0.25199E+09
0.30	0.14647E+09
0.35	0.92636E+08
0.40	0.62328E+08
0.45	0.43964E+08
0.50	0.32187E+08
0.55	0.24286E+08
0.60	0.18786E+08

THETA = 70.DEGREES

DIAMETER	STRESS
0.20	0.39853E+09
0.25	0.20519E+09
0.30	0.11940E+09
0.35	0.75609E+08
0.40	0.50931E+08
0.45	0.35966E+08
0.50	0.26362E+08
0.55	0.19913E+08
0.60	0.15421E+08

THETA = 75.DEGREES

DIAMETER	STRESS
0.20	0.30400E+09
0.25	0.15682E+09
0.30	0.91432E+08
0.35	0.58006E+08
0.40	0.39146E+08
0.45	0.27695E+08
0.50	0.20336E+08
0.55	0.15389E+08
0.60	0.11938E+08

THETA = 80.DEGREES

DIAMETER	STRESS
0.20	0.20716E+09
0.25	0.10726E+09
0.30	0.62765E+08
0.35	0.39961E+08
0.40	0.27063E+08
0.45	0.19212E+08
0.50	0.14155E+08
0.55	0.10748E+08
0.60	0.83649E+07

THETA = 85.DEGREES

DIAMETER	STRESS
0.20	0.10874E+09
0.25	0.56885E+08
0.30	0.33620E+08
0.35	0.21613E+08
0.40	0.14774E+08
0.45	0.10584E+08
0.50	0.78671E+07
0.55	0.60244E+07
0.60	0.47278E+07

THETA = 90.DEGREES

DIAMETER	STRESS
0.20	0.94927E+07
0.25	0.60753E+07
0.30	0.42190E+07
0.35	0.30997E+07
0.40	0.23732E+07
0.45	0.18751E+07
0.50	0.15188E+07
0.55	0.12552E+07
0.60	0.10547E+07

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_o, z_o, x and z respectively)
- time of flight (t)

For orbit propagation:
- an initial state (x_o, z_o, v_{xo} and v_{zo})
- 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.

VARIABLE SHEET

St	Input	Name	Output	Unit	Comment
	3.8241122	ge		m/s^2	Mars' Gravitational Acceleration
	3407	re		km	Radius of Mars
	9408	ro		km	Radius of Orbit
L	229.56666	t		min	Time of Flight
	0	xo		m	Initial x-Position
	0	zo		m	Initial z-Position
	-2.761624	vxo		m/s	Initial x-Velocity
	-1.648515	vzo		m/s	Initial z-Velocity
L		x	11000	m	x-Position
L		z	-6000	m	z-Position
L		zup		m	z-Position (Positive-Up)
L		vx	-2.441451	m/s	x-Velocity
L		vz	-1.413234	m/s	z-Velocity
		w	.0002311	rad/s	Orbit Angular Rate
		c	-.9991336		Cosine
		s	-.0416189		Sine
L		delvo	1.4797788	m/s	Initial Change in Velocity
L		delv	2.8209774	m/s	Final Change in Velocity
L		delvtot	4.3007562	m/s	Total Change in Velocity
		detb	3.2585E-9		Equation Parameter
		b22	-180.0875		Equation Parameter
		T1	11000		Equation Parameter
		b12	17300.742		Equation Parameter
		T2	-6000		Equation Parameter
		b21	-17300.74		Equation Parameter
		b11	-42042.35		Equation Parameter

RULE SHEET

```
S Rule
  "Propogation of relative state equations:

  " Equations used to produce the position of item;

* x =-(2*vzo/w)*c + (4*vxo/w-6*zo)*s + (6*zo-3*vxo/w)*w*t + (xo+2*vzo/w)
* z = (2*vxo/w-3*zo)*c + (vzo/w)*s + (4*zo-2*vxo/w)
* zup=-z

* vx = (4*vxo/w-6*zo)*w*c + 2*vzo*s + (6*zo-3*vxo/w)*w
* vz = -(2*vxo/w-3*zo)*w*s + vzo*c

* w = sqrt(ge*(re^2)/(ro^3))
* s = sin(w*t)
* c = cos(w*t)

* delvo = sqrt(vxo^2 + vzo^2)
* delv  = sqrt(vx^2 + vz^2)
* delvtot = delvo + delv

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

  "vxo = detb*(b22*T1 - b12*T2)
  "vzo = detb*(-b21*T1 + b11*T2)

  "T1 = x - xc - (-6*s+6*w*t)*zo
  "T2 = z - (-3*c+4)*zo

  "b11 = ((4/w)*s-3*t)
  "b12 = ((-2/w)*c+2/w)
  "b21 = ((2/w)*c-2/w)
  "b22 = s/w

  "detb = 1/(b11*b22 - b12*b21)
```

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='ATMMOD.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 ****************************************************************
C THE ABOVE STATEMENTS ARE USED FOR INTIALIZATION 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),
     1ANS(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
C****************************************************************
C
C
C
      SUBROUTINE MODATM (Z,PP,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(III)*SQRT(T1/RAT(III))/(OMEG(T1,III,IQF)
     1)
C
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*********************************M**********************************
      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=I
      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
```

```
C
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***************************************************************
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=II
      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 PHIIJ=(1.0+SQRT(VI(TX,III,1)/VI(TX,JJJ,1))*(AM(JJJ)/ AM(III))*
     1*(1.0/4.0))**2/(2.82842712*SQRT(1.0+AM(III)/AM(JJJ)))
      SUMB1=SUMB1+PHIIJ*VOLPER(JJJ)/VOLPER(III)
 1002 CONTINUE
      XMU=XMU+VI(TX,III,1)/(1.0+SUMB1)
 1001 CONTINUE
      XMU=XMU*.1
      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)*1000.0/(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))*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
     1)**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
   II=I
   IF (PP.GT.P(1)) GO TO 16
   IF(PP-P(I)) 16,41,17
 16 CONTINUE
   HA=H(II)
 42 DO 10 I=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(II)*RE/((RE-H(II)))
   GO TO 12
 17 I=II-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)*ALOG(PP/P(I))/(CONN*1000.0)
 22 HA=HA
   Z=HA*RE/((RE-HA))
   GO TO 23
   END
C
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****************************************************************
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,F5.1,F6.1,F6.0)
      TM(1)=T(1)
      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
C  INPUT DATA DEFINING THE ATMOSPHERIC STRUCTURE
C
   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
      READ (5,976,END=2) IPO,RGM,Z(I),MM(I),T(I)
  976 FORMAT (I1,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
C*****************************************************************
C
C
      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,19HMOLECULAR 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
     1OGEN = ,5X,2PF7.3,11X,16HPERCENT 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 ,/
     8)
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,'  ')
C
C**************************************************************
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**************************************************************
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,
     +'COEFFV',/)
      RETURN
      END
C
C**************************************************************
C

      SUBROUTINE REFRAC (Z1,Z2,XLAMDA,PHI,PHIPR,PSI,SLANT,ANS,RE,CONN)
      DIMENSION ANS(35)
```

```
C
C*****************************************************************
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*****************************************************************
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,'ALTI',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*****************************************************************
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*************************************************************
C
C QUANTITIES ENDING IN S ARE FOR THE SATELLE
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, ANTHE 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
```

```
C**************************************************************
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(K1+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
C    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
      DO 4 M=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))
      IF (PHI.GT..017)PHI=PHI-.0092833
      IF (PHI/CON.GT.90.0)WRITE (6,88) PHI/CON
   88 FORMAT (///,1X,'WARNING,ZENITH ANGLE OF UNREFRACTED PATH EXCEEDS
```

```
    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
    1AS 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 MODATM(ZZZ(J+1),PSURF-DELP*FLOAT(J),'PRES',XLAMDA,
    1ANS,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 MODATM (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(I.EQ.(K1*10)) HAFDEL=0.
     CALL MODATM (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(PSI)*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,///,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,///,1X,' TOTAL REFRACTED ANGLE THRU ATMOSPHERE = ',1PE15.4
     1,' DEGREES')
      WRITE (6,21) ANS(21)/.0174532925
   21 FORMAT (1X,//,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(A.LT.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
C GIVEN THE TEMPERATURE AND PRESSURE AT EACH OF 2 POINTS AND THE ALTITUDE
OF
C    THE LOWER POINT, THIS FUNCTION CALCULATES THE ALTITUDE OF THE HIGHER
POINT
C ALTITU IS IN KM.  CONN IS A CONSTANT =  -M*G/R
C
C***************************************************************
C
      D=TMHIGH-TMLOW
      IF(D) 2,3,2
```

```
   2 D=CONN*1.0E+03/(ALOG(PHIGH/PLOW))*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
C*****************************************************************
C

     FUNCTION PRES(PLOW,D,TMLOW,TMHIGH,DH,ANS,RE,CONN)
     DIMENSION ANS(35)
C
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
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
C*****************************************************************
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)=9.49
      OM(7,4)=8.86
      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
```

```
      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
C******************************************************************
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
```

```
      CP(1,2) =6.9806
      CP(2,2) =7.7331
      CP(3,2) =8.8942
      CP(4,2) =9.8762
      CP(5,2) =10.6646
      CP(6,2)=11.3098
      CP(7,2) =11.8456
C CP( ,3) = MOLECULAR OXYGEN
      CP(1,3) =6.9567
      CP(2,3) =6.9615
      CP(3,3) =7.0237
      CP(4,3) =7.1961
      CP(5,3) =7.4315
      CP(6,3) =7.6704
      CP(7,3) =7.8837
C CP( ,4) = ARGON
      CP(1,4) =4.9681
      CP(2,4) =4.9681
      CP(3,4) =4.9681
      CP(4,4) =4.9681
      CP(5,4) =4.9681
      CP(6,4) =4.9681
      CP(7,4) =4.9681
C CP( ,5) = NEON
      CP(1,5) = 4.9681
      CP(2,5) = 4.9681
      CP(3,5) = 4.9681
      CP(4,5) = 4.9681
      CP(5,5) = 4.9681
      CP(6,5) = 4.9681
      CP(7,5) = 4.9681
C CP( ,6) = HYDROGEN
      CP(1,6)=5.3934
      CP(2,6)=6.5182
      CP(3,6)=6.8938
      CP(4,6)=6.9753
      CP(5,6)=6.9932
      CP(6,6)=7.0091
      CP(7,6)=7.0369
C CP( ,7) = HELIUM
      CP(1,7) =4.9681
      CP(2,7) =4.9681
      CP(3,7) =4.9681
      CP(4,7) =4.9681
      CP(5,7) =4.9681
      CP(6,7) =4.9681
      CP(7,7) =4.9681
C CP( ,8) = WATER
      CP(1,8)=7.9606
      CP(2,8)=7.9694
      CP(3,8)=8.0276
      CP(4,8)=8.1864
      CP(5,8)=8.4161
      CP(6,8)=8.6779
      CP(7,8)=8.9571
C CP( ,9) = CARBON MONOXIDE
      CP(1,9)=6.9564
      CP(2,9)=6.9574
```

```
      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
```