

11-18 CR

PRELIMINARY DESIGN OF A UNIVERSAL MARTIAN LANDER

204266

147 P

West Virginia University
Department of Mechanical and Aerospace Engineering
Morgantown, West Virginia

NASW-4430

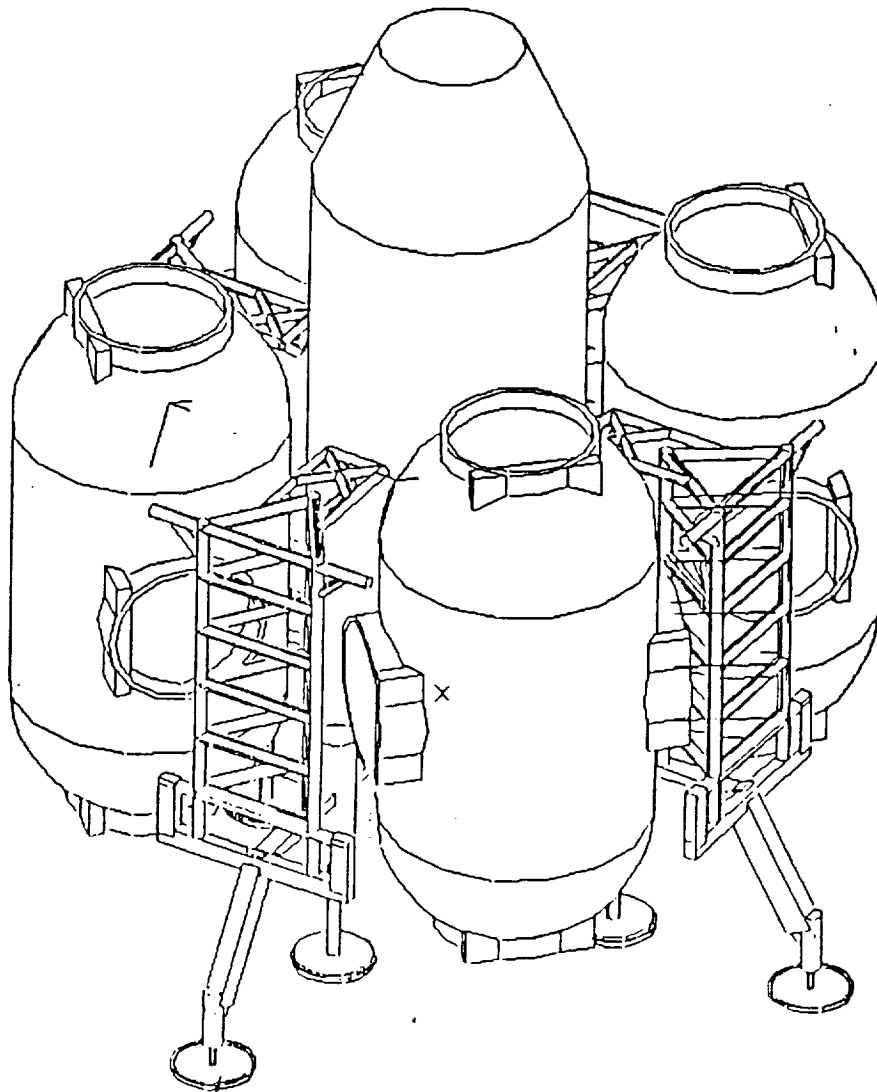
Dr. Timothy L. Norman, Asst. Prof

Mr. David E. Gaskin, NASA/USRA Grad TA

Mr. Sean Adkins
Ms. Mary Gunawan
Ms. Raquel Johnson
Mr. David MacDonnell
Mr. Andrew Parlock
Mr. John Sarick

Mr. Charles Bodwell
Mr. Kouichi Hashimoto
Mr. Michael Kessinger
Mr. Loran Miller
Mr. Matthew Phares
Mr. James Simpson

Mr. Jeff Galusky
Mr. Jonathan Hicks
Mr. Michael Kincheloe
Mr. Steve Nine
Mr. Enoch Ross
Mr. Scott Wenger



N94-25007

Unclas

G3/18 0204266

(NASA-CR-195529) PRELIMINARY
DESIGN OF A UNIVERSAL MARTIAN
LANDER (West Virginia Univ.)
147 P

Mrs. Marilyn Host
Ms. Rebecca Howell

Administrative Secretary
Research Writer

B. Discovery Lab
Mr. Scott Warner

IV. University of Minnesota
Michael C. Wright
Chris Thyen

1992-93 NASA/USRA Grad TA
Past NASA/USRA Grad TA

V. Rockwell International Corporation
A. Rocketdyne Division - Canoga Park, CA
Henry M. Minami, Jr.

Marketing Manager, Propulsion Systems

VI. NASA - Lyndon B. Johnson Space Center
A. Exploration Programs Office
Mr. Jim Young
B. Training Office, Data Management Systems
Mr. Thadd Bowers

VII. Redstone Scientific Information Center

VIII. United States Army Missile Command at Redstone Military Arsenal

IX. Remtech Inc.

Mr. Ronald P. Schmitz

Director of Business Development

Abstract

**PRELIMINARY DESIGN OF A
UNIVERSAL MARTIAN LANDER**

**West Virginia University
Department of Mechanical and Aerospace Engineering
Morgantown, West Virginia**

**Dr. Timothy L. Norman, Asst. Prof.
David E. Gaskin, NASA/USRA Grad. TA**

In the next 25 years, mankind will be undertaking yet another giant leap forward in the exploration of the solar system: a manned mission to Mars. This journey will provide important information on the composition and history of both Mars and the Solar System. A manned mission will also provide the opportunity to study how humans can adapt to long term space flight conditions and the Martian environment.

As part of the NASA/USRA program, nineteen West Virginia University students conducted a preliminary design of a manned Universal Martian Lander (UML). The UML's design will provide a "universal" platform, consisting of four modules for living and laboratory experiments and a liquid-fuel propelled Manned Ascent Return Vehicle (MARV). The distinguishing feature of the UML is the "universal" design of the modules which can be connected to form a network of laboratories and living quarters for future missions thereby reducing development and production costs.

The WVU design considers descent to Mars from polar orbit, a six month surface stay, and ascent for rendezvous. The design begins with an unmanned UML landing at Elysium Mons followed by the manned UML landing nearby. During the six month surface stay, the eight modules will be assembled to form a Martian base where scientific experiments will be performed. The mission will also incorporate hydroponic plant growth into a Controlled Ecological Life Support System (CELSS) for water recycling, food production, and to counteract psychological effects of living on Mars. In situ fuel production for the MARV will be produced from gases in the Martian atmosphere. Following surface operations, the eight member crew will use the MARV to return to the Martian Transfer Vehicle (MTV) for the journey home to Earth.

Forward

America needs a systematic approach towards the exploration of our solar system. It began with the first suborbital flight of an United States astronaut in late 1959. In the years that followed, the critical steps needed to land humans on the moon were mastered. More than a decade of intense preparation and sacrifice culminated in the momentous day when Neil Armstrong became the first human to set foot on the moon. Like a young child learning to walk, Americans became more and more confident living in the strange new environment. New adventures began with Skylab and later with the reusable Space Shuttle. Space Station Freedom will be the next stepping stone to even grander adventures. A permanent base on the moon and then a first manned mission to Mars should be the next space endeavors undertaken by the United States. America will build on its past successes and failures to venture beyond the horizon. ⁵¹¹

The Space Exploration Initiative (SEI) was enacted by President Bush to continue the systematic approach to explore our solar system. His announcement of the SEI came as a bold challenge to do what must be done if America is to remain a leader in space sciences.

The SEI has six goals. The first is to increase our knowledge of the solar system and beyond. Second, the advancement in science and engineering would be a rewarding byproduct of such an endeavor. Third, refocusing the United States position in world leadership is the underlying challenge facing America today. Fourth, development of new technologies for planet Earth will enhance the quality of life for many people. Fifth, facilitating further space exploration and commercialization is critical to America's space program. And finally strengthening the United States economy will help to justify such a broad space policy. ⁵¹²

The design of a vehicle to transport humans and equipment to Mars should be versatile while being low cost. The Universal Martian Lander (UML) will be able to provide America that opportunity to journey to Mars. The UML combines new technology, new thinking, and an ambitious approach to the challenge of the Space Exploration Initiative.

TABLE OF CONTENTS

Acknowledgements	i
Abstract	iii
Forward	iv
Acronyms	viii
System Integration	
Mission Statement	SI1
Design Objectives	SI1
Project Assumptions	SI1
Executive Summary	SI1
Fact Sheet	SI2
Section A: Structures, Materials, and Thermal Control	
Section Design Philosophy	A1
A1: Deployment of Universal Habitat Modules	
A1.1: UML Deployment	A1
A2: Structural Truss Design and Properties	
A2.1: Material Properties	A1
A2.2: Truss Design and Physical Properties	A3
A2.3: Landing Gear	A4
A2.4: Habitat and Laboratory Modules	A13
A2.4.1: Primary Structure of UHM	A13
A2.4.2: Secondary Structures	A15
A2.5: Manned Ascent and Rendezvous Vehicle	A16
A2.6: Center of Gravity Location	A18
A2.7: Mass Moments of Inertia	A18
A3: Radiation Control	
A3.1: Radiation Protection	A19
A4: Thermal Control	
A4.1: Thermal Shielding	A20
Section B: Guidance, Navigation, and Control	
Section Design Philosophy	B1
B1: Landing Site Selection	B1
B2: Martian Transfer Vehicle Orbital Parameters	B2
B3: Trajectory Control	
B3.1: Descent Attitude Control	B3
B3.1.1: Orbital Maneuvers	B3
B3.1.2: Descent Through Atmosphere	B4
B3.1.3: Landing Phase	B5

B3.2: Ascent Attitude Control	B5
B3.3: Rendezvous Analysis	B6
B4: Telecommunications	
B4.1: Satellite Communication and Navigation System	
B4.1.1: SCAN parameter	B7
B4.1.2: SCAN Performance and Capabilities	B8
B4.2: Telemetry and Command	B9
Section C: Aerodynamics and Propulsion	
Section Design Philosophy	C1
C1: Atmospheric Entry	
C1.1: Descent Aerodynamics	C1
C1.1.1: Aeroshield Specifications	C3
C1.1.2: Parachute Specifications	C5
C1.1.3: Thermal heating Analysis	C5
C1.2: Ascent Aerodynamics	C8
C2: Propulsion	
C2.1: Primary Systems	
C2.1.1: Engine Specifications	C9
C2.1.2: Fuel Determination	C10
C3.2: Secondary Systems	
C3.2.1: Engine Specifications and Fuel Determination	C10
C3.3: Attitude Control Systems	
C3.3.1: Engine Specifications and Fuel Determination	C11
Section D: Life Support and Power Systems	
Section Design Philosophy	D1
D1: Environmental Control, Life Support Systems	
D1.1: Environmental Control	
D1.1.1: Atmosphere Revitalization Systems	D1
D1.1.2: Food Production and Storage	D4
D1.2: Extra Vehicular Activity	
D1.2.1: Airlocks	D4
D1.2.2: Decompression	D6
D1.2.3: EVA Suits	D6
D1.3: Thermal and Safety Management	D7
D2: Physiological and Psychological Management	
D2.1: Physiology	D7

D2.2: Psychology	D8
D3: Power Systems	
D3.1: Universal Martian Lander	
D3.1.1: Power Requirements	D9
D3.1.2: Power Source Selection	D9
D3.1.3: Power System Mass Estimation	D11
D3.2: Habitat Modules	
D3.2.1: Power Requirements	D11
D3.2.2: Power Source Selection	D12
D3.2.3: Power System Mass Estimation	D13
D3.3: Manned Ascent and Rendezvous Vehicle	
D3.3.1: Power Requirements	D15
D3.3.2: Power Source Selection	D15
D3.3.3: Power System Mass Estimation	D15
 Section E: Economic Management	
Section Design Philosophy	E1
E1: Economic Methodology	
E1.1: Assumptions	E3
E1.2: Analysis Process	E3
E1.3: Estimates, Risks, and Uncertainty	E4
E2: Engineering and Design Analysis	
E2.1: Manned Landing Vehicle	
E2.1.1: Descent Structure	E4
E2.1.2: Habitat and Laboratory Modules	E5
E2.1.3: Manned Ascent and Rendezvous Vehicle	E5
E2.1.4: SCANS Network	E5
E2.1.5: Earth-based Integration and Checkout	E5
Conclusions	F1
Recommendations	F2
References	
Structures, Materials, and Thermal Control	F2
Guidance, Navigation, and Control	F3
Aerodynamics and Propulsion	F4
Life Support and Power Systems	F5
Economic Management	F6
Appendices	
Structures, Materials, and Thermal Control	G1
Aerodynamics and Propulsion	G19
Life Support and Power Systems	G49
Economic Management	G54

ACRONYMS

Acronym	Definition
ARM	Air Revitalization Model
ARS	Air Revitalization System
BLSS	Biological Life Support System
CAD	Computer Aided Drafting
C&H	Communications & Data Handling
CDMS	Communication & Data Management System
CELSS	Controlled Ecological Life Support System
CFC	Chlorofluorocarbon
COSPAR	Committee on Space Research
ECLSS	Ecological Controlled Life Support System
EHLV	Extended Heavy Lift Launch Vehicle
FPS	Fire Suppression System
GN&CCS	Guidance, Navigation, and Control Computer System
GPS	Global Positioning System
HMF	Health Maintenance Facility
HUD	Heads-up-display
IDEAS	Integrated Design Engineering Analysis Software
IMU	Inertial Measurement Unit
LDEF	Long Duration Exposure Facility
LEO	Low Earth Orbit
MARV	Manned Ascent and Rendezvous Vehicle
MLV	Manned Landing Vehicle
MTV	Martian Transfer Vehicle
NASA	National Space and Aeronautics Administration
PARS	Primary Air Revitalization System
PMAD	Power Management and Distribution
RCS	Reaction Control System
RFC	Regenerative Fuel Cell
RLS	Regenerative Life Support
SARS	Secondary Air Revitalization System
SCANS	Satellite Communication and Navigation System
SCOT	Self Contained Oxygen Unit
SEI	Space Exploration Initiative
SMTC	Structures, Materials, & Thermal Control
TABI	Tailorable Advanced Blanket Insulation
TCU	Thermal Control Unit
TCS	Thermal Control System
TPS	Thermal Protection System
ULV	Unmanned Landing Vehicle
UML	Universal Martian Lander
WVU	West Virginia University

System Integration

Mission Statement. The mission of the Universal Martian Lander is to provide a repetitive, low cost method of delivering humans and equipment to the surface of Mars in order to achieve a permanent presence and a scientifically rewarding mission.

Design Objectives. The design objectives were to perform a preliminary design for the UML project. Those objectives include:

1. Keeping costs low by making design components universal.
2. Ensure safety whenever reasonably possible.
3. Utilizing recycling of vital resources.
4. Minimize weight of the UML.
5. Utilize new technology whenever possible.
6. Provide adequate space to accommodate 8 people for a 6 month surface stay on Mars.

Project Assumptions. The assumptions for the UML design were used to identify areas of the design that could be left to future study. The assumptions for the UML design include:

1. A engine which can burn CO and O₂ can be built and flight-certified.
2. The manned UML will be able to land in very close proximity to the previously landed unmanned UML.
3. The UML g loadings will not exceed 3.0 Earth g's.
4. The UML is already in a 250 km polar Martian orbit.
5. A nuclear powered Martian Transfer Vehicle (MTV) will be used to transport the UMLs

between Earth and Mars. This will minimize the travel time required and maximize surface stay duration.

6. A Extended Heavy Lift Launch Vehicle (EHLLV) will be used to propel the UMLs into Low Earth Orbit (LEO).

Executive Summary. The UML design began with the unmanned and manned UML attached to the Martian Transfer Vehicle (MTV). The unmanned UML will detach from the MTV and descend to the surface of Mars. The unmanned UML will be remotely piloted from the MTV. After success landing of the unmanned UML, the manned UML will detach from the UML. It will then begin its descent into the Martian atmosphere. The aeroshield will provide the required protection from the extreme heating. At 15 to 20 km above the surface, the aeroshield will be blown in 8 pieces. At the same instant the aeroshield is blown free of the UML, the parachutes will be deployed to aide in decelerating the UML to safe landing velocities. Finally, the 4 main engines will be fired to softly land the 8 crew members in the manned UML next to the unmanned UML.

Once on the surface of Mars, the modules will be connected together to form a Martian base. During the 3 day period a few of the modules will be temporarily connected. The crew will have to live in the Manned Ascent and Rendezvous Vehicle (MARV). The first priority will be to setup the Air Revitalization System (ARS) operating in the modules. Next the power generation equipment will need to be set up and checked out to produce electricity. A large soil mover will also be grading the area to prepare a solid foundation for the permanent module sites. In the months that will follow, the in situ propellant production will be operated for the return to the MTV.

Once the sixth month surface stay is over the eight modules will be safed for the next crew to visit Mars. Then the eight astronauts will be in the MARV. After launching the MARV from the truss structure platform of the UML, the MARV will begin its roll program to get into a gravity-turn trajectory. Once reaching the upper edge of the Martian atmosphere the first stage of the MARV will be jettisoned. The apogee kick motor will then propel the MARV to the chase orbit for the MTV. Finally the design ends with the rendezvous of the MARV with the MTV.

Universal Martian Lander Fact Sheet

Aerodynamics/Propulsion Division:

Descent	
Mass of UML during descent	61,101 kg
scale height	3216 m
Entry velocity	3.543 km/s
Entry angle	-3°
Parachutes	
C_D	0.58
Trailing Length	37.03 m
Max diameter 5 km altitude	21.78 m Ascent $C_D = 1.0$
Drag Coefficient	1.0
Frontal Area	14.19 m ²
Mass of UML during ascent	total = 34,196 kg
Aeroshield	
Geometry	Spherical Cone, 20° angle cone, 4.06 m diameter center cone, and 7.5 cm thick
C_D	1.58 at 0° angle of attack 1.525 at 10° angle of attack
C_L	0 at 0° angle of attack 0.226 at 10° angle of attack
Frontal surface area	363.05 m ² of 21.5 m diameter (circle)
Weight	7298 kg
Material	Aluminum
Insulation	TABI
angle of attack	+10°
Max impingement angle	23°

	Flexible/Deployable outer Ring	will allow 13.51 m to expand to 21.5 m aeroshield. This will allow aeroshield to fit in EHLLV.
Fuel	Fuel Process	CO and O ₂ fuel production through thermal decomposition of CO ₂ at 1000K. No additional reagents required.
	Fuel characteristics	I _{sp} of CO + O ₂ engine = 305 s
	Fuel weight	CO = 11,239 kg O ₂ = 6,548 kg Total = 18,000 kg
Engine	Thrust for engines	1340 KN for 4 engines
	Specific fuel consumption	1181 kg/s
	Weight of engines	2,567.33 kg for 4 engines
	Height	3 m
	Chamber length	1 m
	Max diameter of nozzle	1.2 m
Fuel Tanks	Shape	Spherical
	Diameter CO	1.5 m
	O₂	1.035 m
 <u>Guidance, Navigation, and Control Division:</u>		
	Characteristics of MTV orbit	Polar orbit (90° inclination), 250 km circular orbit, Period of 1.84 hours, Orbital velocity of 3.45 km/s
	Characteristics of SCAN orbit	Geosynchronous Orbit (0° inclination), 20,468 km circular orbit, 2 satellites 163° apart from each other for comm. to MTV, Earth, Moon, Space Station <i>Freedom</i> , or Space Station <i>Mir</i> .
	SCAN mass	250 kg per satellite including apogee kick motors
	SCAN capabilities	navigational and communication aids for UML.
	SCAN deployment	just before reaching Martian polar orbit.
	SCAN power equipment	solar panels

Landing Site	26° N, 216°W at Elysium Mons
Descent Phase	
▲ V to initiate UML descent phase	51.7963 m/s to place into elliptical transfer orbit for aerobraking
Aeroshield	maneuver +- 10°
Minimum L/D	.11
number of maneuvers	12 recommended
Thrust needed to maneuver	60000 N with 5.53 m moment arm
orbital maneuver thrust requir.	4 90 sec. burns of 957.4 N
Ascent Phase	
Parking orbit and position	240 km, 10 km below and 50 km behind MTV
Burn-out altitude	60 km
Velocity at burn-out	3.542 km/s
Trajectory	Gravity turn
▲ V to circularize 240 km parking orbit into 240 km parking orbit	.0893 km/s
Coast time to reach 240 km orbit	52.8 min.
▲ V for rendezvous with MTV from 240km parking orbit	0.0433 km/s
Coast time to reach MTV from 240km parking orbit	54.95 min.
Number of degrees MTV to be from MARV in 240 km orbit	179.63 °
Rendezvous	
Thrust needed to rendezvous with no drag	22.52 N
Drag	23.54 N
Overall thrust	46.92 N
Assumed mass	15600 kg (this shows all fuel was burned during ascent)
CCS	
CD&H Equipment	Receivers, transmitters, satellite dish, antennas, processors, controls, and video displays, and satellite tracking equipment

CD&H mass	200 kg
GN&CCS Equipment	CD&H system plus star trackers, cameras, radar, docking lights, RCS jets, control electronics, IMUs, and sensors
GN&CCS mass	245 kg

Power Systems/Life Support/Robotics Division:

Number of crew members	8
Day/night length	typical Earth day/night length
Oxygen generation	50 metric tons per 6 month stay
ARS	2 will be used. Sabatier CO ₂ technology
Dimensions	.762 m X 1.016 m X 1.168 m
Volume	.9046 m ³
mass	224.1 kg (8 man)
Power requirement	3.056 kW
Control Panel	
mass	70 kg
dimensions	.533 m X .813 m X 1.29 m
volume	.561 m ³
Power requirements	1.2 kW
WRS	
water production	2.07 kg /hr
Power requirements	.35 kW
mass	200 kg
volume	.5 m ³
storage of waste	compressed into tank (solid, liquid , and toxic gases)
non-toxic gases disposal	vented to atmosphere
pretreatment	Hypochlorite, multifiltration

Cabin pressure	10.2 psi with O ₂ content between 21% to 40% of atmosphere
EVA equipment characteristics	
EVA suit pressure	5 psi
EVA suit characteristics	Heads-up display in helmet
Decompression time	30 min
Dimensions	2m x 1m x 1m
Number of suits	12 (8 nominal usage, 2 spares for parts, 2 to replace suit failures)
Living quarters	1 bed and 1 locker per crew member, 2 bathrooms with showers, storage areas, kitchen.
Physiological and Psychological Requirements	
sleep chambers	shield sleep chambers from sound and light (sleep chambers volume = 2.38 m ³), hygiene facilities away from sleep chambers,
Hygiene facilities	dental/oral hygiene, bathing, clothing laundering, and shower.
Mini-gym	see Exercise Equipment
Library	
Wardroom/galley	Dining/meetings
mass of galley	1296 kg
Commander's office	
Training Room	
Airlock	
Airlock dimensions	Right cylinder with diameter of 1.6 m, length of 2.1 m. Will accommodate 2 suits.
Airlock	Airlock needs air shower to blow off martian dust from suit
Health Maintenance Facility (HMF)	Will be able to handle problems commonly encountered on Earth, at least 2 medically trained crew members to handle emergencies.
Exercise equipment mass and volume	under 362.88 kg, 3.11 m ³
Exercise equipment	treadmill, ergometer, "penquin" suit, all to provide 1g capacities during exercising if needed

Power Requirements	
Surface stay	
day	2.0 kW per crew member, 2.2 kW comm. & Data Management Systems, 5.0 kW for science, .5 kW for Rover 23.4 TOTAL / per day
night	same as day but 1.0 kw for science, 19.7 TOTAL / per day
Descent/Ascent stage	17.8 kw/stage
Sources of Power	
Descent Phase	Batteries (silver-zinc) for 2 hrs until surface and fuel cells for 3 day surface stay.
Ascent	Batteries for 5 hrs until MTV rendezvous
Power used during ascent/descent	17.8 kW
Weight of 12 batteries/fuel cells during ascent	726.5 kg
Weight of 5 batteries during descent + 3 day surface by fuel cell	1329.3 kg, CLEFT GaAs/CuInSe ₂ Tandem
Surface stay	CLEFT GaAs/CuInSe ₂ Tandem Solar cells during day, regenerative fuel cells with cryogenic storage tanks during night
area of solar cells	1641.1 m ²
mass of solar cells	642.7 kg
Array capacity	189.37 kW at beginning of life
Array mast materials	3 coilable S-glass/epoxy, continuous longerons stabilized by glass/epoxy battens and steel diagonals.
mass of fuel cells w/cryogenic storage tanks & reactants	4658 kg
EVA	Batteries
PMAD	Used to integrate power generation equipment from solar arrays to RFCs.
PMAD mass	1322 kg

Life systems requirements

water

3888 kg for drinking,
4968 kg for sanitation,
36288 kg for domestic use (washing
clothes, washing dishes, showering)

atmosphere

1944 kg O₂

5000 kg N₂

BLSS characteristics

Utilizes horticulture to provide positive
psychological effects, food, O₂ replenishment,
reclamation of solid and liquid wastes, and water
reclamation

**Mass of water required for mission
using BLSS**

7258 kg

Total weight

1280 kg including water storage

Total weight of stored food

3409 kg

Power required

3.192 kW

total storage area

22.5 - 25.5 m²

Volume

12.9 m³

Structures/Materials/Thermal/CAD:

Gross Weight

150 metric tons (150,000 kg)

Mars EHLLV clearance specifications

30.48 m in length, 14.02 m in inner diameter

Ascent Vehicle

Material

Aluminum 2090

Landing Gear

Type

air/oil pistons with pivotable feet.

NOTE: EHLLV will support UML not landing
gear.

Total length

92 cm

stroke

37 cm

diameter

15 cm

number of landing gears

8

minimum height

3.77 m

Cylinder design

Aramid Kevlar epoxy, monocoque structure, 1.477

	cm skin thickness required for radiation shielding.
Material	Aluminum 2090
Hatch placement	One hatch on all 4 sides
Window placement	One in each hatch
Max force on connections to truss	82771 N
Number of bulkheads and stringers	6 and 12 respect.
Material of bulkheads and stringers	Boron aluminum
Floor material	Aramid Paper Honeycomb
Window shape	.5 m x .5 m
Window pane material	silicon glass
3 Window pane thicknesses	.62 in, 1.3 in., .62 in.
Hatch dimensions	1.5 m x 2.0 m
Airlock Area	8.2301 m ² (similar to Shuttle)
Cylinder center area	20.7995 m ²
Diameter of module	4.25 m
length of module	9.144 m
method of release from UML	explosive bolts
Truss characteristics	
height	4.9 m
ascent/descent support	base beam ring design structure will hold up ascent/decent vehicle.
Material	Boron Aluminum
<u>Miscellaneous:</u>	
Design for max G-loading	3 Earth g's
Max. Surface Air Temperature	290 K, 17 °C, 62.6 °F, 522.6 °R
Min. Surface Air Temperature	130 K, -143 °C, -225.4 °F, 234.6 °R
Martian Atmosphere Composition	95.32% Carbon Dioxide, 2.7 % Nitrogen,

	1.6% Argon, .13% Oxygen, .07% Carbon Monoxide, .03% traces of Water Vapor
Day/night length	typical Earth day/night length
Surface stay duration	6 months
Mars Micro-gravity effects on body	25% Major muscle lost during 9-day mission, 0.3-0.4% calcium loss/month, heart shrinkage, blood pressure fluctuation, body fluid shifts, motion sickness
Mars radiation effects	atmosphere can provide adequate shielding (spot shielding used for sensitive equipment)
Surface activity	Soil analysis- geological history of Mars, signs of life, history of solar system, to understand Earth's problems, why Mars is in Ice Age, how superoxidized material is converted to CO ₂ , check hydrogen levels for underground water. Astronomy - thinner atmosphere for better than Earth observations Geological experiments - volcanic activity, seismic activity, tectonic plate movements Weather data collection - better understand regional weather Upper Atmosphere experiments - search for a magnetic field, determine strength of gravitational field Mineralogical research - identify and catalog, determine properties of new materials
Weather	Frequent dust storms, evidence of seasonal changes, all surface water is vaporized.
Gravitational Acceleration	3.72 m/s ²
Terrain	Soil is light gray to black, with reddish tint from volcanic ash. Magnetite is found on surface (magnetic material). Overall terrain is comparable to Southwest United States with large and small rocks. Water erosion is evident on surface. Polar regions covered in ice. Mountain ranges can be found along equator.

SECTION A: STRUCTURES / MATERIALS / THERMAL / CAD

Sean Adkins
Charles Bodwell
Mary Gunawan
Steve Nine
Matthew Phares
Enoch Ross
John Sarick

Section Design Philosophy

The SMTC group sectioned off into subgroups to work on different aspects of the UML design. The first subgroup was the Truss Design Group which designed the overall truss structure, investigated materials, performed load estimations, and stress calculations. The second subgroup investigated the universal cylinder design, material types, hatch sizes, connections, and size and number of windows. The third subgroup designed the size, type, and number of landing gear for the lander. The fourth subgroup designed the ascent/descent vehicle's exterior structure, and placement of different components needed during both aspects of the mission. During the entire design project, CAD drawings and I-DEAS analysis were performed for each subgroup's design.

A1. Deployment of Universal Habitat Modules

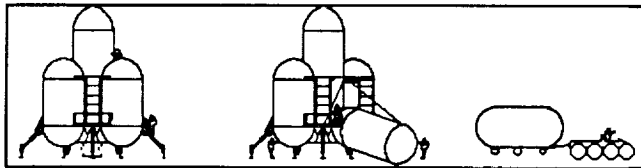


Figure A1.1: Sequence of steps in Habitat Module Deployment.

A1.1 UHM Deployment. The deployment of the Universal Habitat Modules (UHM) upon a safe landing on the surface of Mars relies upon the direct cooperation of the eight astronauts. Once on the surface all systems checked and functional, the astronauts will exit the ascent/descent module and rendezvous with the unmanned martian lander. The detachment of the MUME will take place first. The detachment of equipment, UHMs, and the MUME will all follow the same procedure. Four electrically driven winches will be located on the truss near the connection

of the interior landing gear. Each winch will have a cable preconnected to the top of each UHM. Once the winches are proven operational the explosive bolts holding the cylinder to the Main Truss will be activated.

This will disconnect the top of the UHM completely and leave the UHM supported by the two rotational shafts on the lower connection and the cable from above. The UHM will then be lowered to the surface.

Once the top of the cylinder is lowered to the ground, the cable will be manually removed and reattached to the bottom end of the UHM. The rotational shafts will be unattached from the Main Truss and the UHM will be lowered to the ground.

Once the UHM is down and unattached from the Main Truss, the MUM will transport the cylinder into position to be connected to the next UHM lowered. This will be repeated for each UHM removal.

A2. Structural Truss Design and Properties

A2.1 Material Properties. The Main Truss Design centered around three main topics; simplicity, strength, and weight. Simple in order to make the design easy and relatively inexpensive to manufacture. Strong in order to transport the astronauts to the martian surface and back safely. Light weight in order to leave most of the EHLLV's payload for life support equipment and experimental equipment.

In order to minimize the overall weight of the Truss Design, composites were the first materials to be researched. Through research, the main concern for the material selection became the effects of radiation on the structure.

In the post flight examinations of the sample composites placed on the LDEF satellite significant loss of epoxy matrix mass was found due to the radiation and Atomic Oxygen exposure. Graphite/Epoxy composites positioned on the leading edge showed a 20-30% strength loss due to the Atomic Oxygen exposure ^{A1}. Kevlar/Epoxy composites in structural load bearing members had a 60% reduction in strength in another experiment ^{A2}. For this reason the use of metal matrix composites was chosen for the Main Truss material.

From the LDEF experiment, aluminum was shown to withstand the long duration exposure to space, this led to the selection aluminum for the matrix in our metal matrix composite. In comparing different fiber types, information from the LDEF experiment showed a deterioration of Kevlar and Graphite fibers but no deterioration of the Boron fibers.

Table A2.1: Properties of Boron

Material	Density, ρ g/cm ³	Young's modulus of elasticity				Shear modulus, G_{L1} GPa 10 ⁹ pd	Poisson's ratio, ν_{12}	Ultimate tensile strength				Ultimate compressive strength				Strength, σ_{cr} MPa ksi		
		GPa	10 ⁹ pd	GPa	10 ⁹ pd			MPa	ksi	MPa	ksi	MPa	ksi	MPa	ksi			
Unidirectional composites ($V_f = 0.6$)																		
E-glass	1.94	45	6.5	12	1.7	4.4	0.64	0.25	1000	150	34	5	550	80	130	20	40	6
Kevlar 49	1.30	76	11.0	5.5	0.8	2.1	0.3	0.34	1300	200	28	4	200	40	140	20	55	8
T-300	1.47	132	19.2	10.3	1.5	6.5	0.25	0.25	1200	180	45	6.5	800	120	140	20	62	9
VSI0-32	1.63	229	33.2	6.9	1.0	5.5	0.8	0.25	1170	170	41	6	690	100	140	20	680	11
Boron	1.86	274	39.8	15	2.2	52	7.5	0.25	1310	190	34	5	2400	360	310	45	100	15
GY-70	1.61	320	46.4	5.5	0.8	4.1	0.6	0.25	690	100	41	6	620	90	140	20	96	14
Metals																		
2024-T3	2.77	72.3	10.5	72.3	10.5	27.6	4.0	0.31	462	67	455	66	345	50	345	50	276	40
7075-T6	2.80	71.0	10.3	71.0	10.3	27.6	4.0	0.31	544	79	530	77	475	69	475	69	324	47
4130	7.84	207	30.0	207	30.0	82.7	12.0	0.25	655	95	655	95	1100	160	1100	160	380	55

Material	Percent boron	Ultimate tensile strength		Elastic modulus		Strain to failure, %
		MPa	ksi	GPa	10 ⁹ pd	
2024F as fabricated	45	1287.5	186.7	202.1	29.3	0.775
	47	1420.7	206.0	222.1	32.2	0.795
	52	1721.0	249.6
	54	1798.6	260.8
	64	1527.6	221.5	275.9	40.0	0.72
	66	1739.2	251.6
	70	1927.6	279.5
2024-T6	46	1458.7	211.5	220.7	32.0	0.810
	64	1924.1	279.0	275.9	40.0	0.755
6061F	48	1489.7	216.0
6061-T6	50	1343.4	194.8	217.2	31.5	0.695
	51	1417.2	205.5	231.7	33.6	0.715

Table A2.2.1: I-DEAS output for UML truss.

Volume 2.377D+00m³
 Mass 6.419D+03kg
 Center of gravity -1.021D-02, 1.819D+00, -1.351D-09m

Moments of Inertia about c.g.
 Ixx, Iyy, Izz 9.561D+04, 1.122D+05, 9.593D+04m⁴
 Ixy, Iyz, Izx -2.740D+02, -2.290D-05, 7.575D-06m⁴

Moments about origin:
 Ixx, Iyy, Izz 1.169D+05, 1.122D+05, 1.172D+05
 Ixy, Iyz, Izx -3.933D+02, -3.868D-05, 7.663D-06m⁴

Principle axes:
 1 -1.000 0.017 0.000
 2 -0.000 -0.000 1.000
 3 0.017 1.000 0.000

Principal axes
 1 -1.000 0.017 0.000
 2 -0.000 -0.000 1.000
 3 0.017 1.000 0.000

Principal Moments of Inertia
 I11 I22 I33 9.560D+04 9.593D+04 1.122D+05m⁴

The material properties of Boron were better than any other possible fiber, Table A2.1. From the success of the use of Boron-Aluminum structural members in the Space Shuttle a Boron-Aluminum Metal Matrix composite was chosen for the main material for the truss structure.

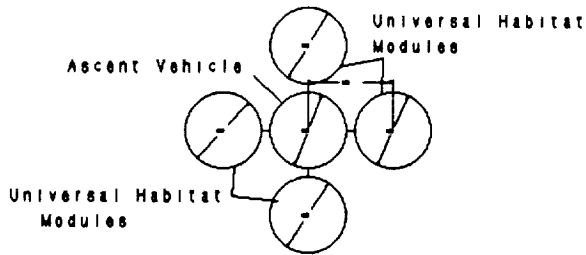


Figure A2.2.1: Planform layout of Universal Martian Lander.

A2.2 Truss Design and Physical Property Calculations. The main design criteria for the truss design were an easily manufactured, simple, and light weight truss system. Configuration and dimensions of the ascent vehicle and cylinders were specified by the preliminary conceptual design. This design can be seen in figure A2.2.1.

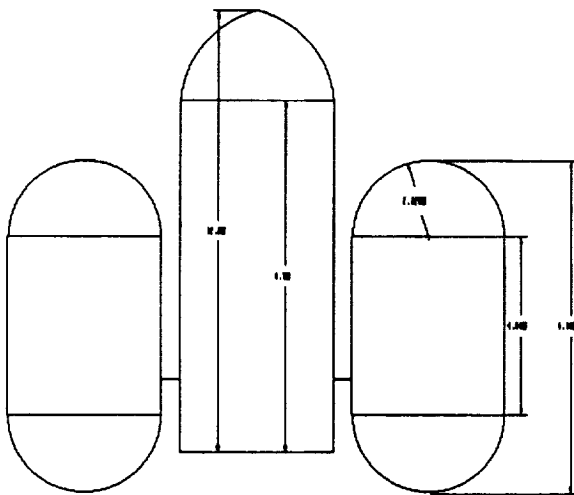


Figure A2.2.2: Conceptual Design of the Universal Martian Lander

Working with this design configuration, the limited planform geometry for use by the truss was calculated according to the space available. Once the planform

area was found, the truss was designed to fall within the area and height constraints of the vehicle design. After deciding on two upper and two lower connections for each of the universal cylinders, and four upper connections for the ascent vehicle the loading

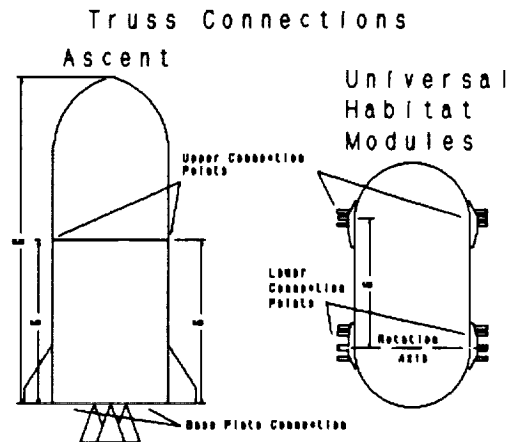


Figure A2.2.3: Connection points for the Ascent and the UHM.

calculations were performed.

The limiting g loading for the astronauts established by NASA is 3. The design of the main truss used a g loading of 3.5 with a safety factor of 2.

With the total mass each UML set by the limitations of the EHLLV at 75000 kg, the mass of the separate components were estimated. The UHM's mass was estimated at 15% of the total mass or 11250 kg. The ascent vehicle was estimated at 30% of the total mass or 22500 kg. The truss mass was estimated at 10% of the total mass or 7500 kg. The truss mass estimation included the mass of the landing gear.

With the safety factor, the G loading, and the estimated masses of each component known, the loads per connection were calculated.

Total G loading:

$$(9.806 \frac{m}{s^2})(2)(3.5) = 68.642 \frac{m}{s^2} \quad (A1)$$

Universal Module's Loading Per Connection:

$$(11250kg)(68.642 \frac{m}{s^2}) \frac{1}{4} = 193055.6N \quad (A2)$$

Ascent Loading on Upper Truss:

$$(68.642 \frac{m}{s^2}) \times (\frac{22500kg}{12}) (30\%) = 38611.13N \quad (A3)$$

Ascent Loading on Base Ring:

$$(68.642 \frac{m}{s^2}) (22500kg) (70\%) \frac{1}{36} = 30027.7N \quad (A4)$$

In equation A3 the loading is divided by 12 to include the three connection points on each of the four main upper ascent vehicle connections shown in Figure A2.2.1. Equation A4 includes a factor of 1/36 due to the use of I-DEAS Finite Element Analysis Beam Modeling Process. The main base ring was modeled from 36 elements connected in a circle shown in Figure A2.2.2. The load was uniformly spread over the nodes to model the surface pressure load.

The available area made it necessary to have a truss system between each cylinder at least 4.9 meters tall. Once the decision to have similar structures between the cylinders, a way of connection the four structures together was approached. A beam structure that could connect the trusses and support the ascent vehicle for take-off was designed.

The main concern was to have the base ring carry most of the load of the ascent vehicle and have the trusses transfer the load to the base ring through members coming out to support the truss. Now the truss system as a whole could be analyzed and optimized. Utilizing the abilities of Integrated Design Engineering Analysis Software, I-DEAS, the design of the truss geometrical layout of the truss was almost entirely in I-DEAS. The imputed structure was modeled using beam elements in I-DEAS finite element family. Different configurations were drawn up entered into I-DEAS, stressed, and compared to other designs until a simple, lightweight truss design was found.

Through I-DEAS the center of gravity and moments of inertia of the Truss were computed. The output of I-DEAS are presented in Table A2.2.1. The values directly from I-DEAS are computed using a center reference point located in the bottom of the base ring. Values below were shifted to a global reference frame referenced to the top of the ascent vehicle.

$$\begin{aligned} X_{cg} &= -1.021e-2 \text{ meters} \\ Y_{cg} &= -1.351e-9 \text{ meters} \\ Z_{cg} &= 10.381 \text{ meters} \end{aligned}$$

A total mass of the truss structure was calculated by I-DEAS to be 6419 kg.

The analysis on I-DEAS showed a strong inward moment when the habitat modules were removed and the ascent vehicle was disconnected for take off. For this reason, the number of landing gear was doubled to eight accommodate the high bending stress.

For ease of production of the truss structure a circular tube cross section was chosen for the aluminum-boron members. Technology for the production of the circular tube members is already known from the boron-aluminum members used in the main structure of the Shuttle. The size of the cross section of the members was varied to find a size that produced stress levels below the yield strength of Boron-Aluminum. Main stress points occurred at the connections of the landing gear and the vertical truss structure to the base ring.

Resulting stress and deflection outputs from I-DEAS can be seen in Figures A6-A8. The deflection results are not to scale, the results were scaled up in to see the high deflection points. The maximum deflection in the Truss was at the universal module connections = 2.84cm. The highest Von Mises stress was found to be $6.20E+07N/M^2$. The maximum principle stress was $3.83E+07N/M^2$ and the maximum shear stress was $3.31E+07N/M^2$. I-DEAS results can be found in the Appendix A1.

A2.3 Landing Gear. The landing gear of the UML had to be both simple and highly efficient. The first step in determining the configuration of the landing gear was to pick a basic design. The design chosen modeled one of the most simple landing gear concepts, small shock absorbing vertical legs.

The problem of how to damp the shock of the initial impact was the first thing addressed. This was solved using the design of a simple shock absorber, similar to that of an aircraft landing gear. The shock absorber decided upon was an Oleo-Pneumatic shock. This design combines and maximizes the best features of the oleo (air) and pneumatic (oil) shocks. This particular shock absorber design was chosen due to its reliability and high efficiency ratings, eighty to ninety percent ⁴⁴.

The next step in the construction of the shock was to determine the stroke length of the shock upon impact with the martian surface. This was done using Newton's

$$\frac{MV^2}{2g} = NSn_M \quad (A5)$$

second law.

Database: none
View : No Stored View
Task: Mesh Creation
Model: 6-BREF2

Units : SI
Display : No stored Option
Model Bin: 1-MRIN
Associated Workset: 77-WORKING_SET77

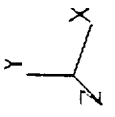
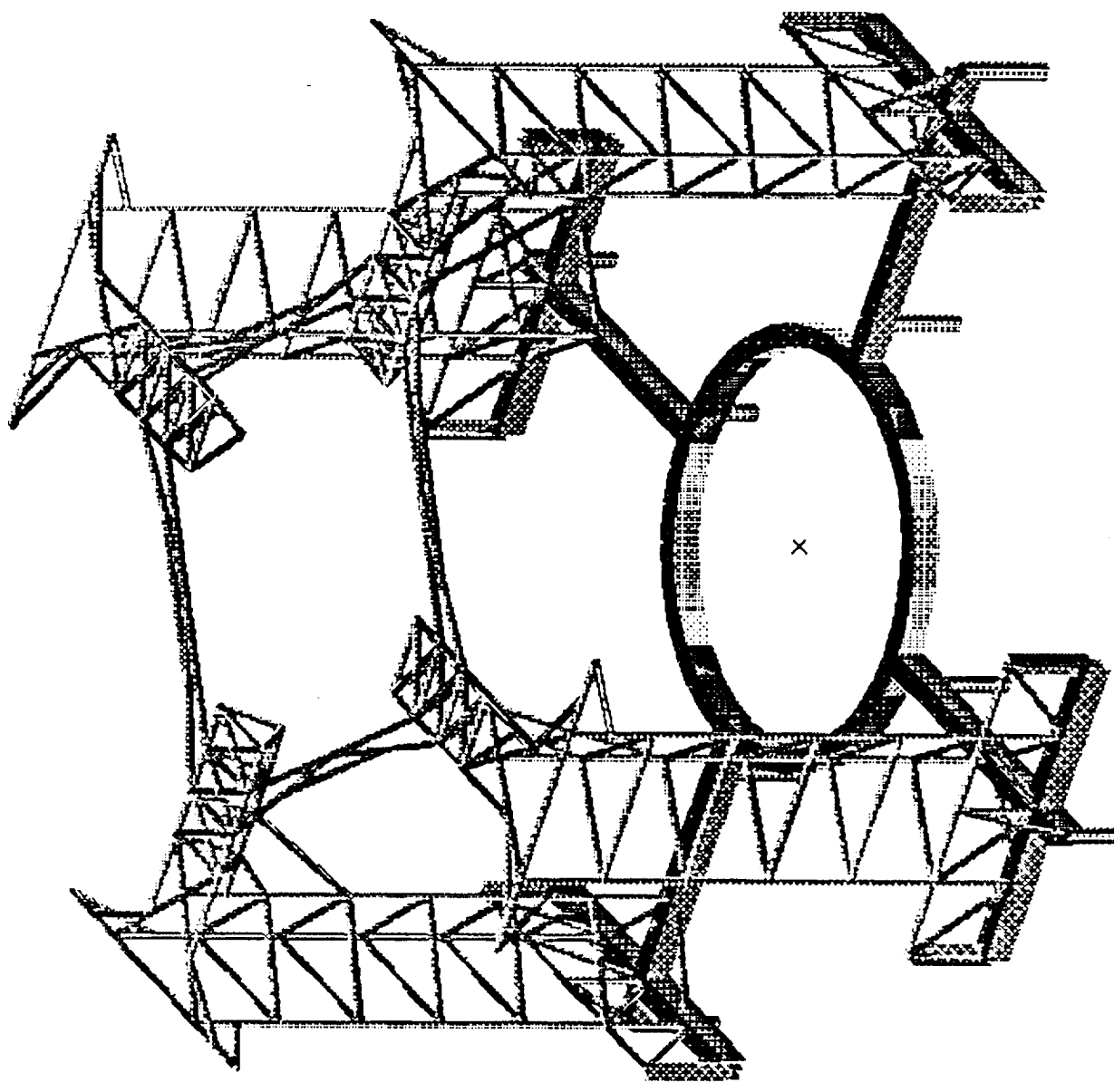


Figure A2.2.4: I-DEAS Beam Element Input of Universal Martian Lander.

Database: none
View : No stored View
Task: Boundary Conditions
Model: 6-BASE2

Units : SI
Display : Plotted Optim
Model Bin: 1-HIGH
Associated Workset: 77-WORKING_SET77

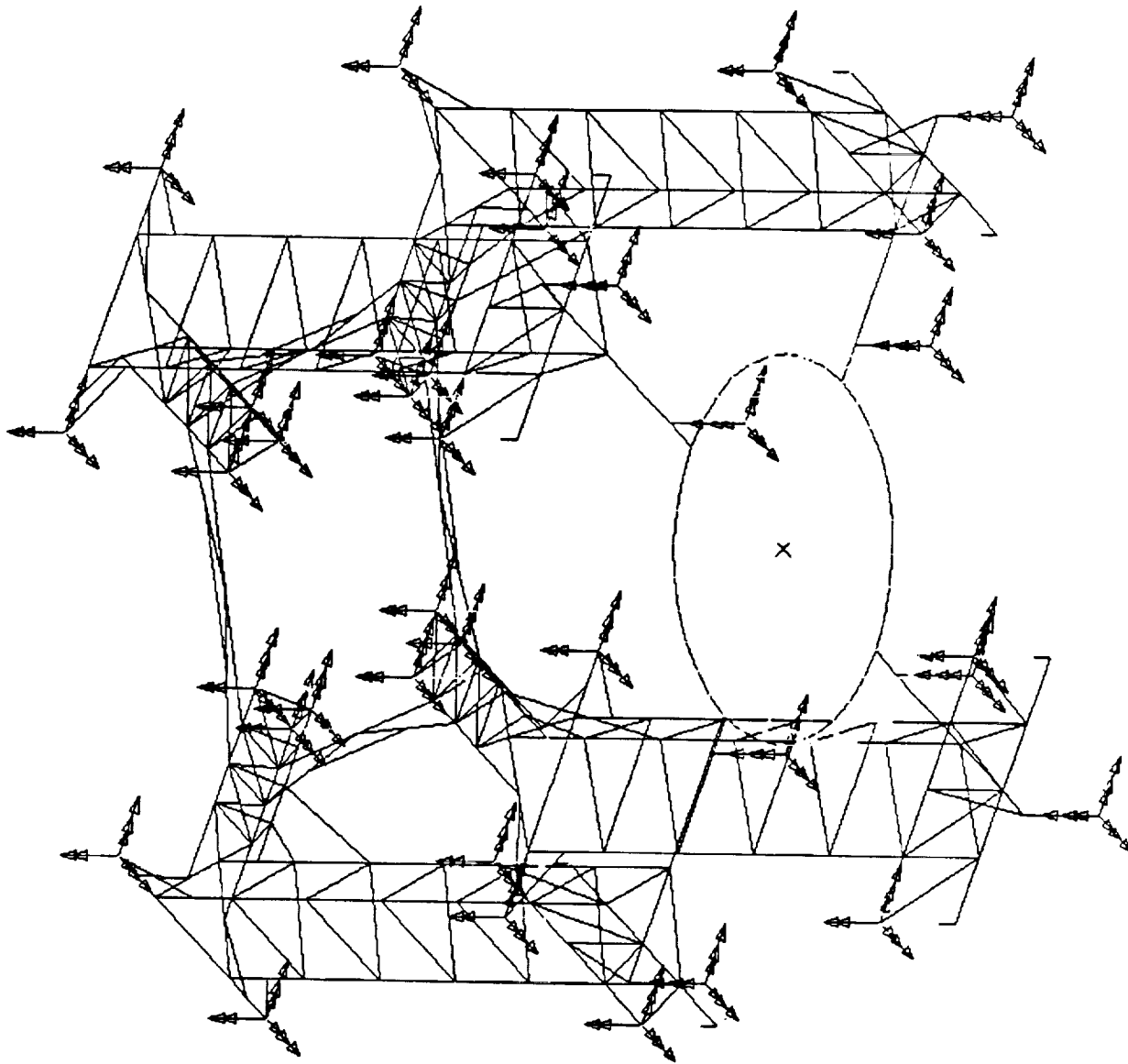


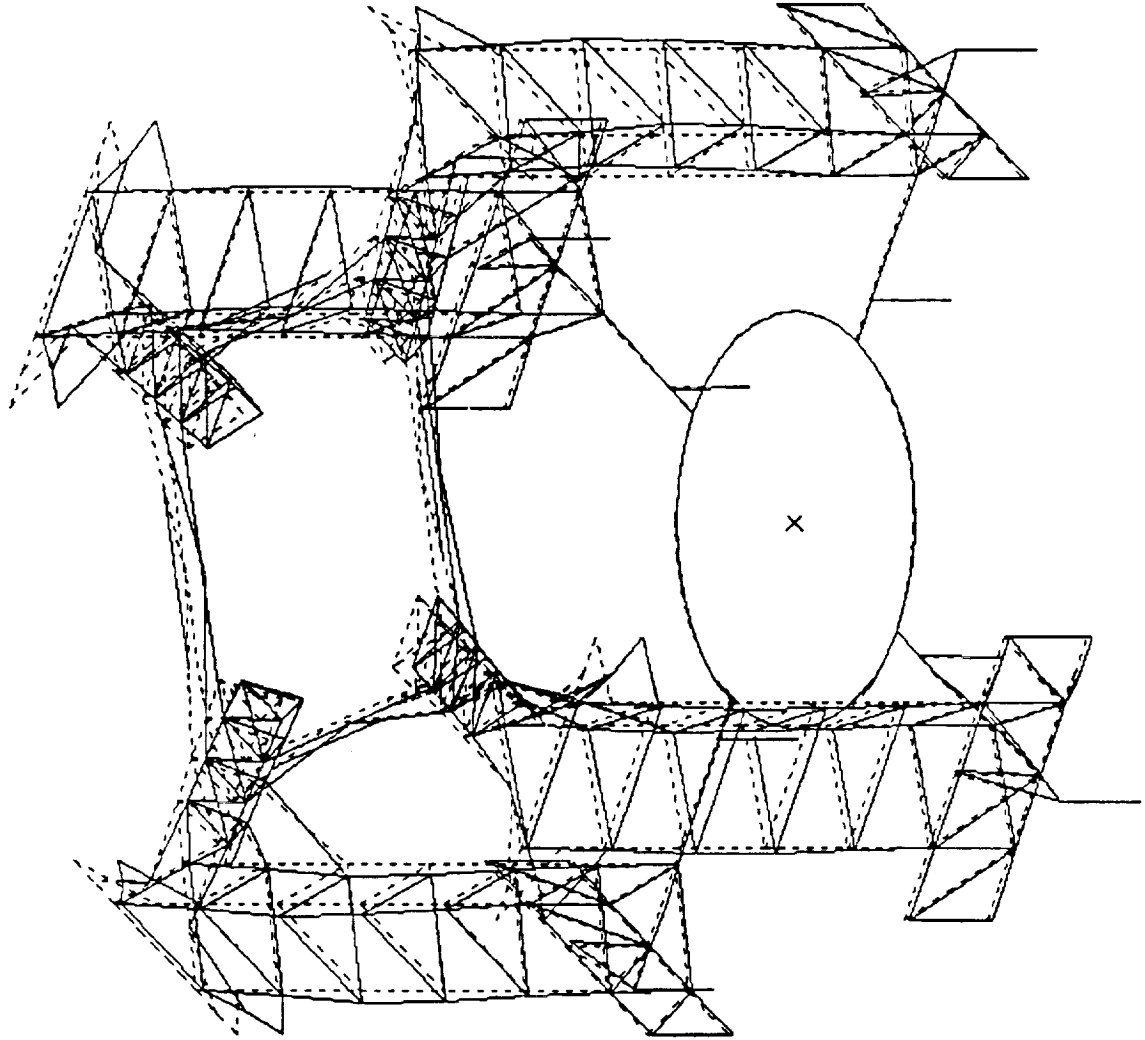
Figure A2.2.5: Loads and Restraints for the Universal Martian Lander.

Database: none
View : No stored View
Task: Post Processing
Model: 6-BASE2

Units : SI
Circular : No stored Option
Model Pin: 1-10111
Associated Material set: 77-MPPY-ING_SET177

none

LOAD SET: 1 - LOAD SET 1
DISPLACEMENT - MIN/MAX: 0.00 MAX: 0.028403



A7

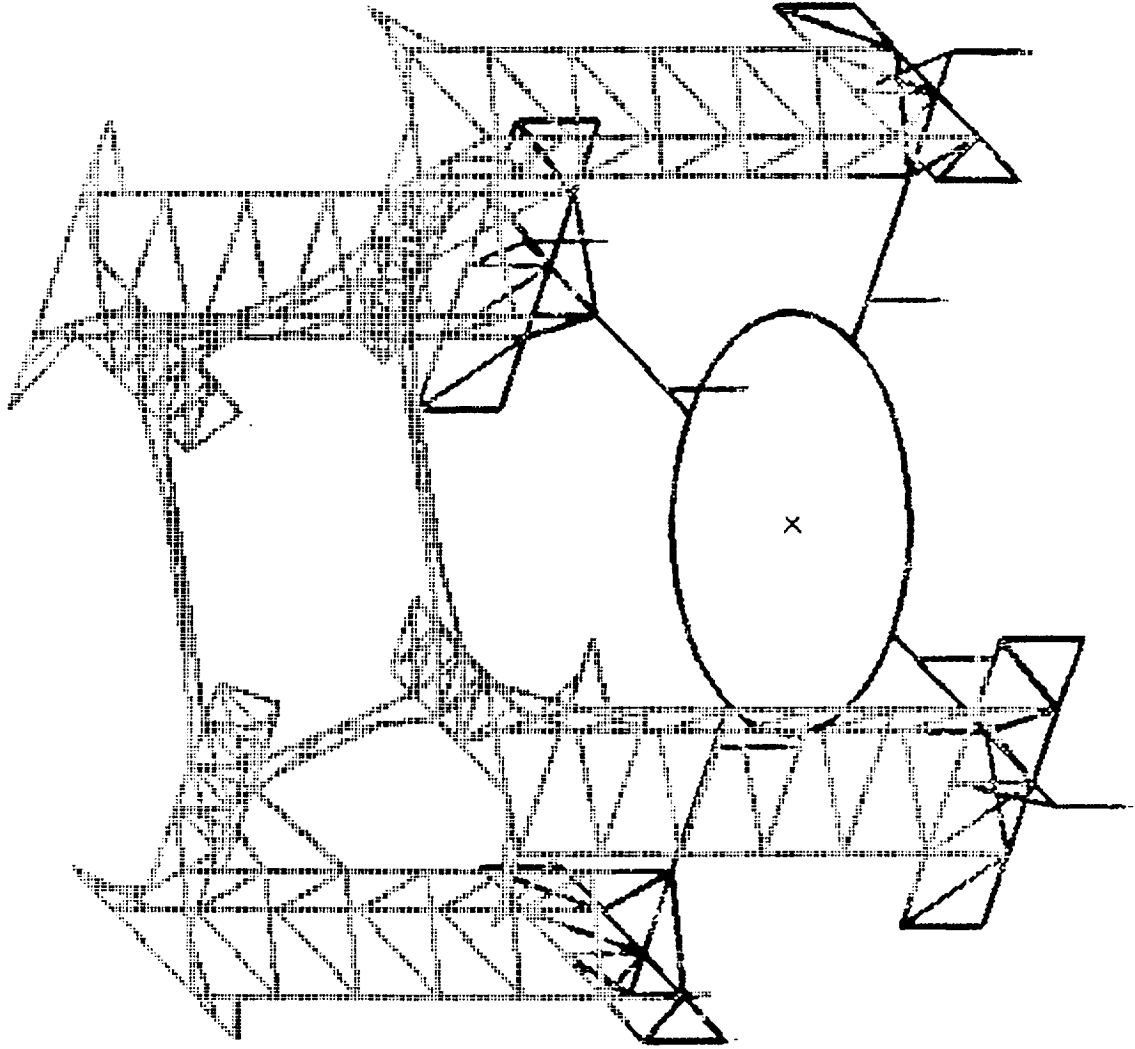
Figure A2.2.6: Resulting Deflections for the Universal Martian Lander.

Database: none
View : No stored View
Task: Post Processing
Model: 6-BRSE2

Units : SI
Display : No stored Options
Model Bin: 1 MBIN
Associated Modelset: 77-WORKING_SET177

LOAD SET: 1 - LOAD SET 1
FRAME OF REF: GLOBAL
STRESS - VON MISES MIN: 0.00 MAX: 6.20E+07

none



6.20E+07
5.31E+07
4.43E+07
3.54E+07
2.66E+07
1.77E+07
8.85E+06

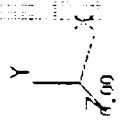


Figure A2.2.7: Resulting Von Mises Stress for the Universal Martian Lander.

Database: none
View : No stored View
Task: Post Processing
Model: 6: RISE2

LOAD SET: 1 - LOAD SET 1
FRAME OF REF: GLOBAL
STRESS - MAX PRIN MIN:-4.40E+05 MAX: 3.83E+07

none

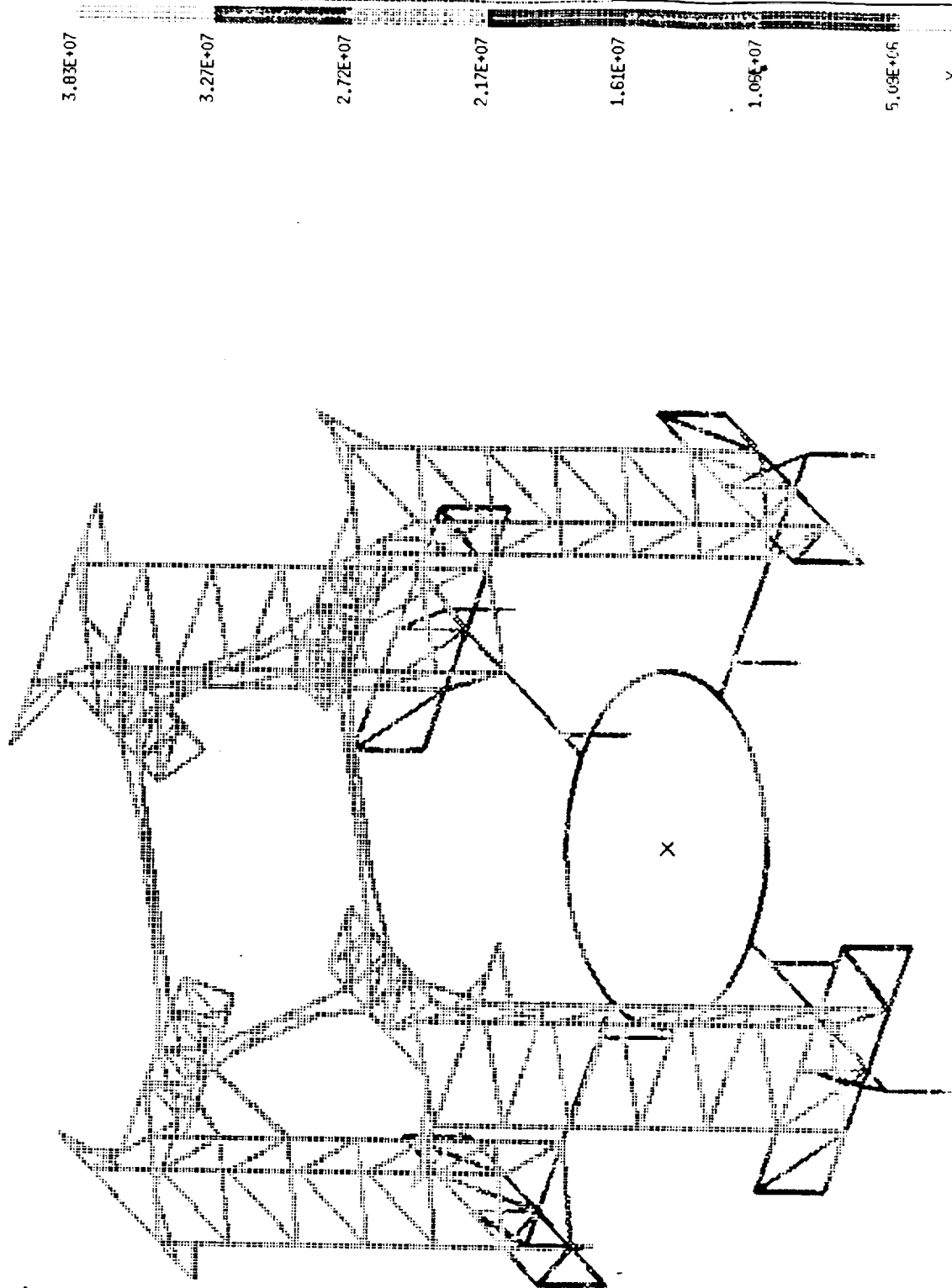


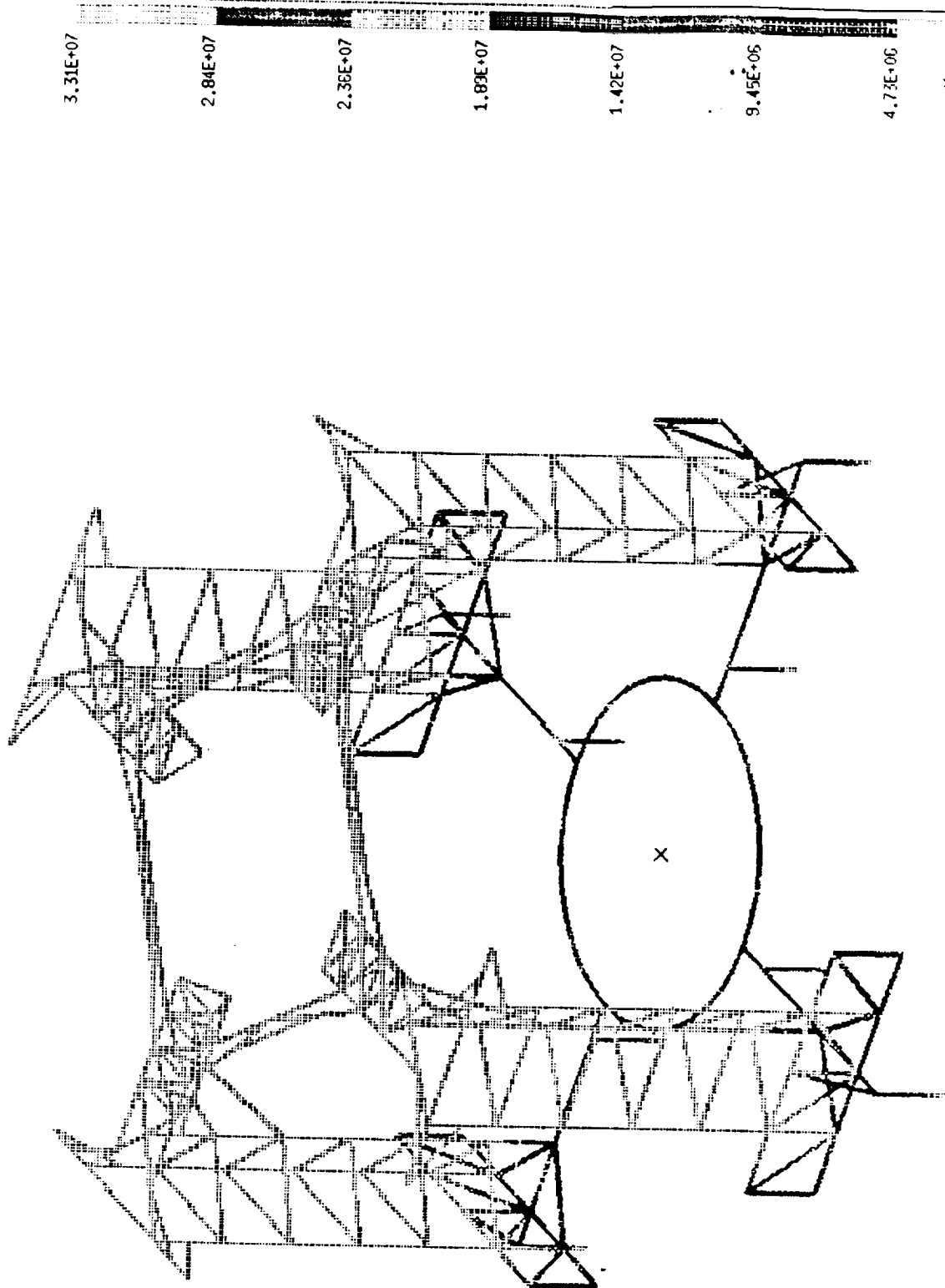
Figure A2.2.8: Resulting Principal Stress for the Universal Martian Lander.

09-11-93 09:07:01
Units : SI
Display : No stored Option
Model Bldg: 1-NMIM
Associated Workset: 77 UNFOLDING_SE177

Database: none
View : No stored View
Task: Post Processing
Model: 6-BHSE2

LOAD SET: 1 - LOAD SET 1
FRAME OF REF: GLOBAL
STRESS - MAX SHEAR MIN: 0.00 MAX: 3.31E+07

none



A10

Figure A2.2.2: Resulting Maximum Shear Stress for the Universal Martian Lander.

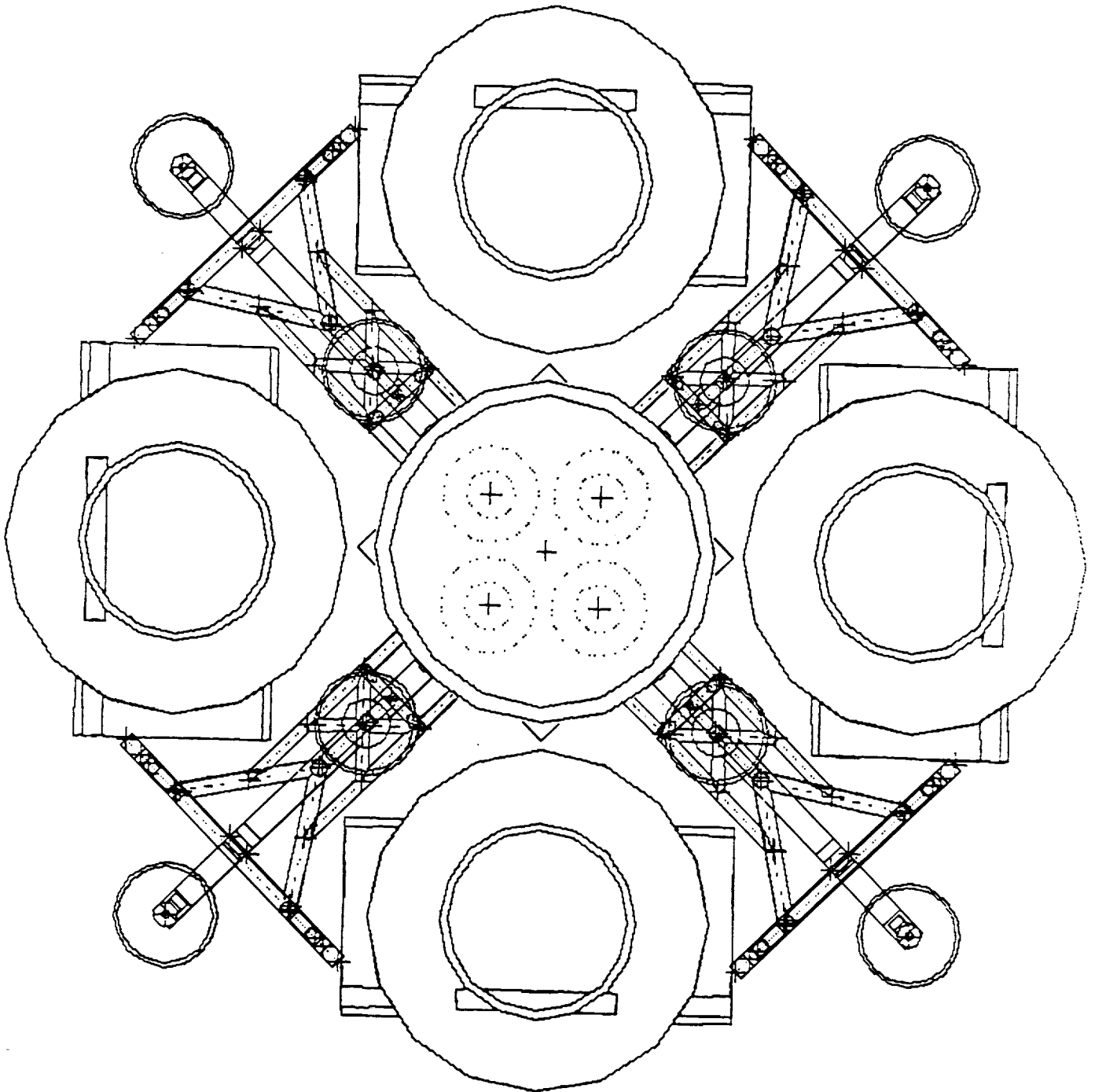


Figure A2.2.10: Planform view for the Universal Martian Lander.

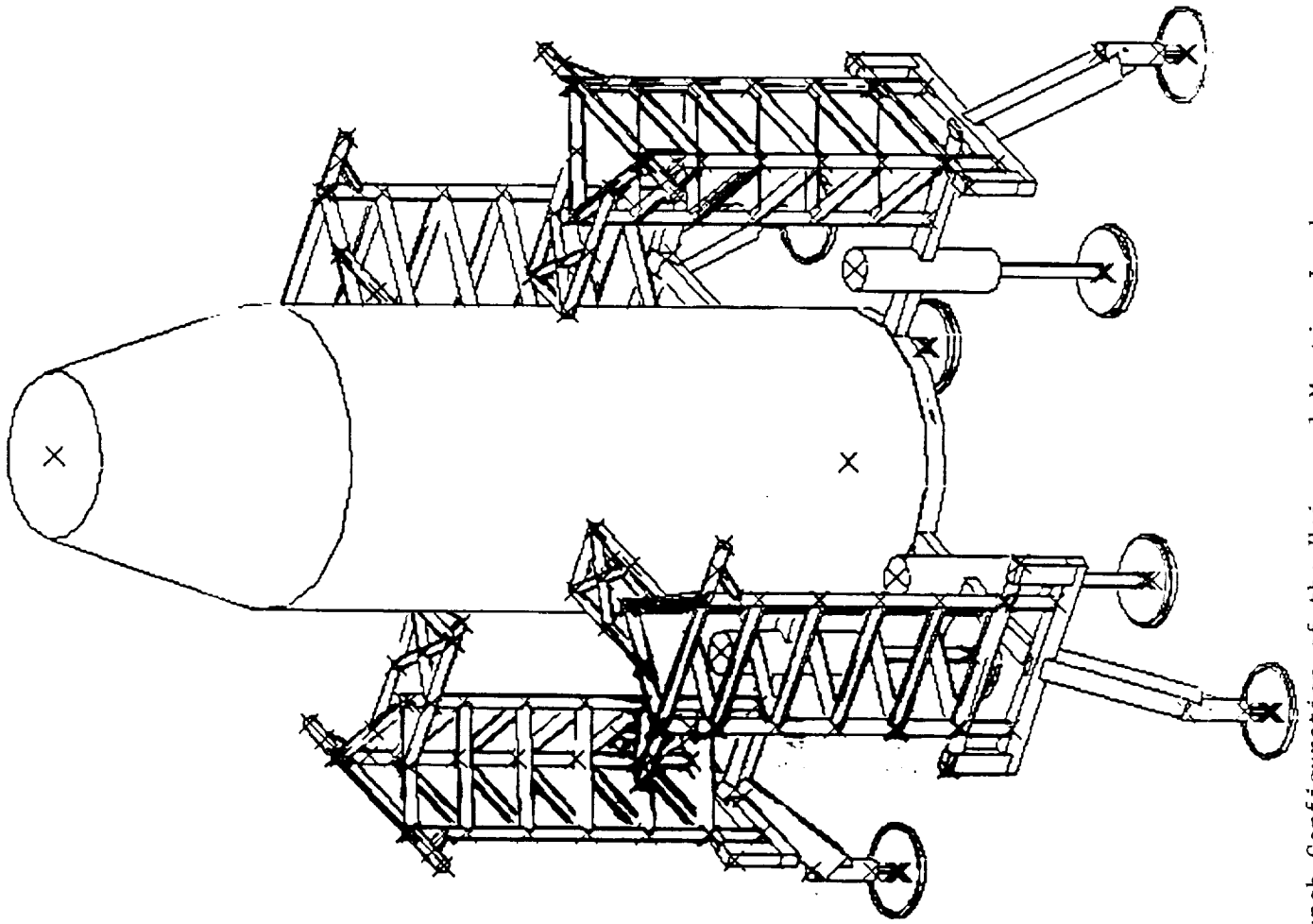


Figure A2.2.11: Launch Configuration of the Universal Martian Lander.

The final equation used was:

- $N=3$ g landing gear load factor
- S = strut stroke
- $n_s=85\%$ =efficiency of shock
- $V=5.18$ m/s=impact velocity
- $g=3.72$ m/s²=martian gravity
- $M=62000$ kg=mass of UML

The implementation of the above equation divided by the number of landing gear, four, allowed the group to calculate a stroke of 37 cm for a 62,000kg UML. By calculating the stroke of the shock the total minimum length and diameter of the shock was calculated to be 92 cm and 15 cm.

The final problem associated with the development of a feasible landing gear was the construction of the foot and how to stow the landing gear when not in use. The foot of the shock was to be made of a thick solid metal plate that would pivot on a joint at the base of the leg. This pivotable foot gave the UML the ability to fold up and be stowed in the heat shield when not in use as shown in Figure A2.3.1. It also gave the UML some flexibility when landing on uneven surfaces.

Redundancy, to some extent, was achieved by placing four additional landing gear within the inner structural ring. These four landing gear had the same design specifications as the first four. This allowed the lander the ability to land on seven or even six legs, if need be. The placement of the four additional gear within the inner structural ring also allowed for a decrease in the moments associated with the truss structure while the ascent vehicle sat upon it.

To provide the UML's engines with enough ground clearance the minimum height of the vertical leg was determined to be 3.77 m. This gave the engines the room needed to ignite and shut down.

A2.4 Habitat and Laboratory Module. Habitat module design is divided into two parts, primary or load bearing structures, and secondary structures. The habitat module must withstand all loads encountered during prelaunch handling, launch, ascent, descent, deployment, attitude changes, docking, and other activities in orbit.

In most cases, maximum loads occur during launch. For the preliminary design of the habitat module, the loads occurring during launch are considered greater than other loads.

The selection of the shell configuration is based on achieving a minimum practical weight that will resist launch loads and fluctuating pressures while providing the necessary meteoroid and radiation protection.

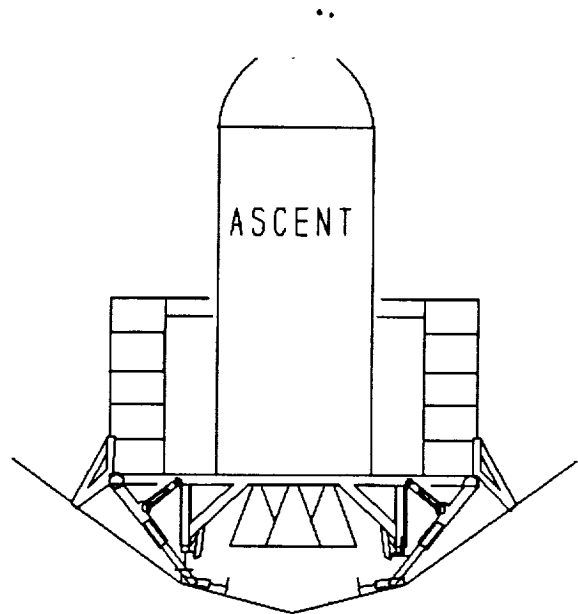


Figure A2.3.1: Stowed landing gear on UML with aeroshield.

A2.4.1 Primary structure of UHM. Four habitat modules are used in the design. Each of them is composed of a cylinder with hemispherical ends. Each contains a habitable volume of 109.62 cubic meters. The dimensions of the habitat modules are as follow:

- Length = 9.094 m
- Diameter = 4.25 m

Figure A2.4.1.1 shows the cut out of the habitat module and its dimensions.

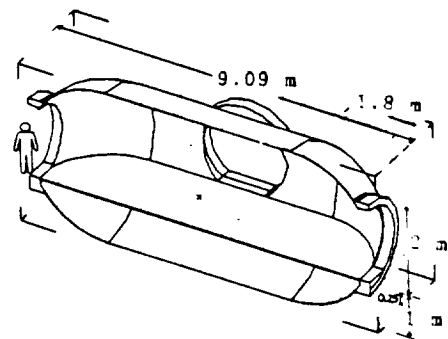


Figure A2.4.1.1. Universal Habitat Module cutout with dimensions.

The primary mode of failure for the shell under launch is instability or buckling from compressive loading. Skin thickness was also based on its ability to withstand radiation. Using these factor a skin thickness of 6 mm was obtained.

The habitat module is designed as a semimonocoque structure to withstand the loading. Semimonocoque construction is the most efficient when compared to pure monocoque and sandwich construction based on the longitudinal loading and structure weights. The loads applied to the habitat module are carried by the skin and closed cross sectional truss members. Since buckling is the primary mode of failure during launch, the truss members were sized based on the critical stress for buckling. The equation used is

$$\sigma = \frac{c\pi^2 EI}{L^2 A} \quad (A6)$$

The design satisfies the buckling requirement. The stress critical of buckling is bigger than the stress applied at the member. The ultimate stress was used for the applied stress for the reason of safety. Figure A2.4.1.2 shows the truss member cross sectional area.

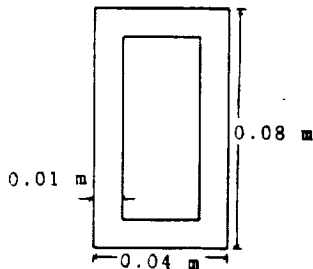


Figure A2.4.1.2: Truss member cross sectional area.

Aluminum Alloy 2219 is used for the cylinder skin. Two different materials were compared to be used for the cylinder skin: Alloy 2219 and Alloy 2090 (Table A2.4.1.1)

Table A2.4.1.1 Comparison of Aluminum Alloy 2219 and 2090

Properties	Alloy 2219	Alloy 2090
Composition	6.3Cu-1Mn-18Zr-1V-06Ti	2.7Cu-22Li-12Zr
Elastic Modulus ($\times 10^4$ GPa)	73.8	76
Poisson's Ratio (20°C)	0.33	0.34
Coeff. of thermal expansion from 20 to 100 ($m/m^\circ C$)	23.4	23.6
Melting Range ($^\circ C$)	543	560-650
Density at 20°C (g/cm^3)	2.84	2.59

From Table A2.3.1.1it can be seen that Alloy 2090 has advantages over Alloy 2219 with its higher elastic modulus, higher melting point, and lower density. Also from the research that has been done, it shows that Alloy 2090 has higher strength than 2219 at temperature between -300 to 250°C. As the temperature decreases the strength and toughness of both Alloys increase. Changes in strength and toughness are more pronounced in 2090 than in 2219.

Although Alloy 2090 has some better properties than 2219, Alloy 2219 is used for the skin of the cylinder. The research has been shown that the compressive strength of aluminum 2090 is much lower than that of aluminum 2219. However, there is no real values for the compressive strength of Aluminum 2290. The properties of 2219 are as follows:

- Poisson's ratio = .33
- Elastic modulus in tension = 73.8 GPa.
- Elastic modulus in compression = 75.2 GPa.
- Coefficient of thermal expansion = 23.4 $m/m^\circ C$.
- Density = 2.84 g/cm^3

Aluminum Boron is also used in the primary structure of the habitat module to carry the loads. A metal matrix composite is used for these members because metal can be used to produce very efficient unidirectional stiffening members and can be spot welded to the aluminum cylinder skin Boron aluminum is selected as the truss members' material for the same reasons it was selected for the truss structure ⁵.

A2.4.2 Secondary Structures

A2.4.2.1 Floor. The area of the floor is 25.443 m². The floor is placed 1 meter from the habitat module structure (See Figure A2.4.1.1) The floor material is Aramid Paper Honeycomb. This material is:

- Tough and damage resistant product.
- The mechanical properties of the material as a structure core are somewhat lower than aluminum, especially in modulus. However it processes a unique ability to survive overloads in areas without permanent damage.
- The base material is relatively incombustible which is very important for the safety reasons.

A2.4.2.2 Hatches. In four habitat modules, each has one airlock hatch, one EVA hatch and three other hatches which can be used to connect the modules to each other. In the other four habitat modules, each will have four hatches. The hatch's function is connecting one module to the other (Figure A2.4.2.2.1).

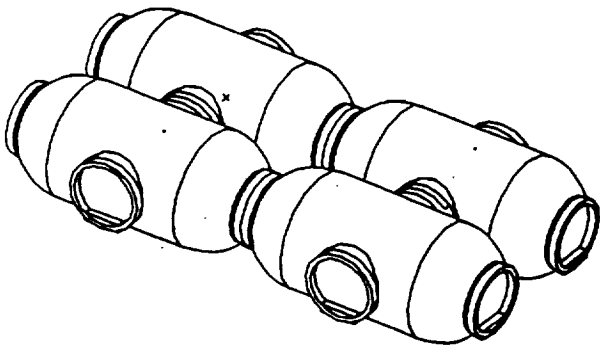


Figure A.2.4.2.2.1: Habitat Module layout on the surface.

The hatches are similar to the one that were used in Space Shuttle are used in this design. All of the hatch design meet the hatch design requirement. They are capable of operation from either side of the hatch, are closed in the direction of positive pressure differential, and capable for equalization of pressure across the hatch. Also all of hatches are provided with hinge linkages to control hatch motion.

A2.4.2.3 Airlock Hatches. Space shuttle airlock hatch design is used for the habitat module hatches. The construction and the dimension of the airlock hatches can be seen in Figure A2.4.2.3.1. The airlock hatches consist of the following hardware:

- actuator
- hatch crank that will rotate 45° clockwise and counter clockwise.
- 6 latches that pull the hatch flush with the bulkhead and the o-rings form an airtight seal as the hatch crank rotates.
- o-ring
- purge ports that equalize the pressure between the two sides of the hatch to allows the hatch to be easily opened.

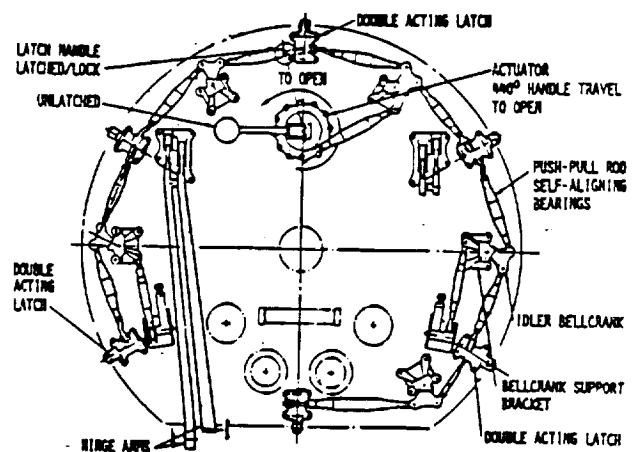


Figure A2.4.2.3.1: Airlock Hatch

A2.4.2.4 EVA Hatches. The EVA hatches are based on the design implemented by the space shuttle and are also called Ingress/Egress hatches (Figure A2.3.2.4.1). The hatches consist of the following hardware:

- bellcrank support
- bellcrank
- drive link
- link
- jam nut
- clevis
- hatch crank
- actuator
- hinge
- window

These hatches provide access from the inside of the airlocks to the outside of the modules.

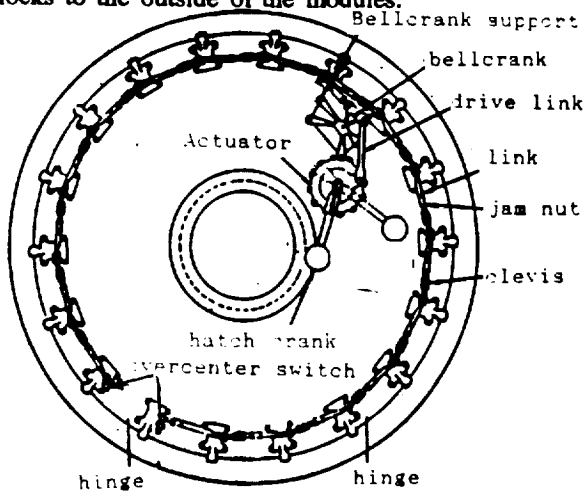


Figure A2.4.2.4.1: EVA Hatches

A2.4.2.5 Windows. Each habitat module has a total of eight windows. The window panes have the same design as the windows used in the Space Shuttle. Each window consists of three panes.

- an outer pane of .62 in. silica glass.
- a middle pane of 1.3 in. fused silica glass.
- an inside pane of .62 in. aluminosilicate glass.

There is one window in each hatch and one on each side of the side doors. The windows in the hatches are constructed as circles with a diameter of .3 m. The windows in the sides of the hatch are also constructed as circles with diameters of .5 m. The windows must maintain structural integrity and leak-tightness after exposure to shock, vibration, acoustics, pressure changes, temperature extremes, ionizing radiation, and impacts by micrometeoroids or crewmen. In addition, the windows provide radiation protection to the crew and equipment which is provided by the material and the structure of the window.

A2.5 Manned Ascent and Rendezvous Vehicle. The Manned Ascent and Rendezvous Vehicle (MARV) is a very important component of the UML. It provides the means for the astronauts to safely land on and take off the surface of Mars. It is designed as a single stage rocket, comprising four different sections. It will weigh approximately 30000 kg, and will be 12 m tall and 4.25 m in diameter.

The first section is the Main Engine Section (MES), which houses the four engines and their accessories. It will be 1.9 m tall. The dry weight of the MES will be approximately 3000 kg.

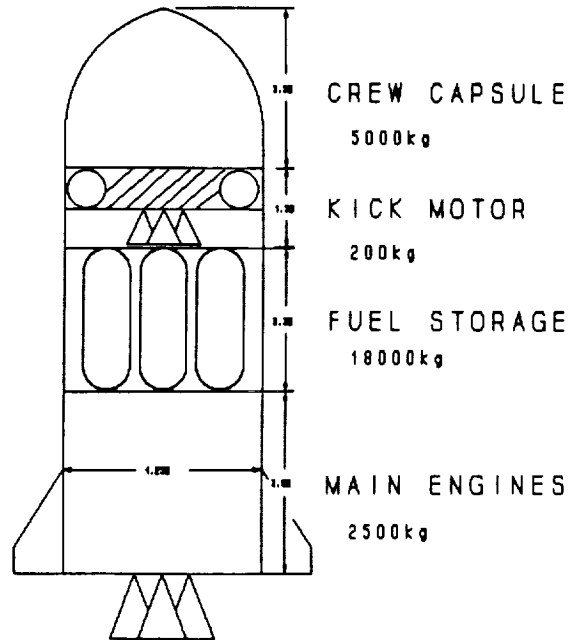


Figure A2.5.1: Layout of the interior of the Manned Ascent and Rendezvous Vehicle.

The second section is the Fuel and Plumbing Section (FPS), which will house the fuel tanks and most of the associated plumbing. This section will be 5.3 m tall. The FPS will weigh approximately 20000 kg when fully loaded, and 2000 kg dry. The next section is the Secondary Motor Interstage Section (SMIS), which houses the secondary motor for the command module and provides the necessary coupling between the first stage and the command module. The SMIS will be 1.5 m tall. The Command Module Section (CMS) attaches to the SMIS and is where the crew functions during takeoff and landing. It is also the component of the MARV which actually rendezvous with the MTV after the mission is complete. The secondary motor is actually attached to the CMS, but for ease of inertial calculations, it is considered a part of the SMIS. The CMS will be 3.5m tall. A schematic of the rocket layout can be seen in Figure A2.5.1.

The winds on the surface of Mars can sometimes approach 90 m/s, so a prelaunch moment diagram was calculated, after finding the force acting on the rocket at a maximum wind gust.¹

The equation used to find the maximum gust force was:

$$Force = \frac{1}{2} \rho_{Marv} V^2 C_D A^2 \quad (A7)$$

V = wind velocity
 C_D = cylinder drag coefficient
 A = wetted area of MARV
 ρ_{Marv} = atmosphere density

The force acting on the rocket at a height of 10 m was 2255 N. This produced a maximum bending moment of 22550 N-m at ground level on the MARV.

The axial thrust load was calculated using the basic rocket equations.² The equation used to find the axial thrust load for the MES was:

$$T_{MES} = \frac{T_o}{W_o} \times W_o = 1340KN \quad (A8)$$

T_{MES} = MES thrust
 T_o = total available thrust
 W_o = total weight

where the thrust load equals the ratio of thrust available to total weight multiplied by the total weight. This will be different for each of the four sections because the weight carried by each section will be lower the higher the section is. The thrust load for the FAS is then:

$$T_{FAS} = \left(\frac{T_o}{W_o}\right)(W_{KMIS} + W_{CMS}) = 232KN \quad (A9)$$

T_{FAS} = FAS thrust
 W_{KMIS} = weight of KMIS
 W_{CMS} = weight of CMS

The results of the prelaunch moment and axial thrust loading can be seen in Figures A2.5.2 and 2.5.3, respectively.

The MARV will be built in a manner similar to aircraft fuselages, using longerons and ribs. The basic material will be Aluminum 2019. It was chosen because it will represent the same material that will be used to construct the habitat modules, and thus does not introduce any undue complications in manufacture. It also will not cause the price of the UML to increase due

to the difficulty of acquiring different materials.

The Guidance, Navigation and Control section has designed some of the control software for the MARV to use fins for control during ascent. These will be attached to the lower portion of the MARV in the MES. This section will have more ribs than some of the other sections and will provide a stronger attachment point for the fins.

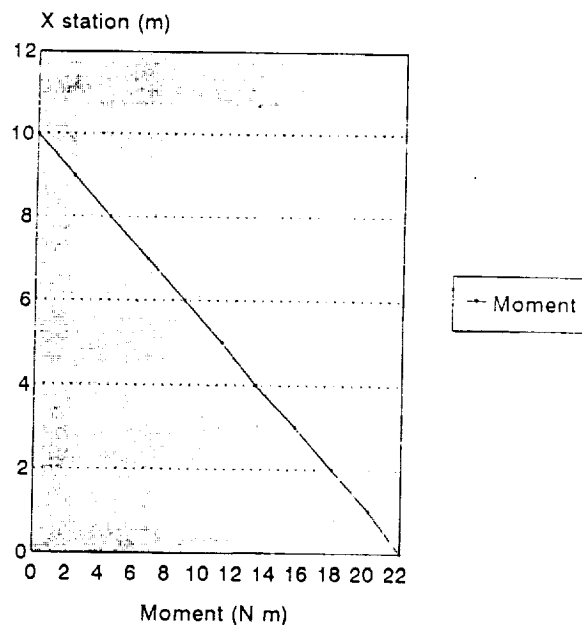


Figure 2.5.2: Prelaunch bending moment on UML MARV.

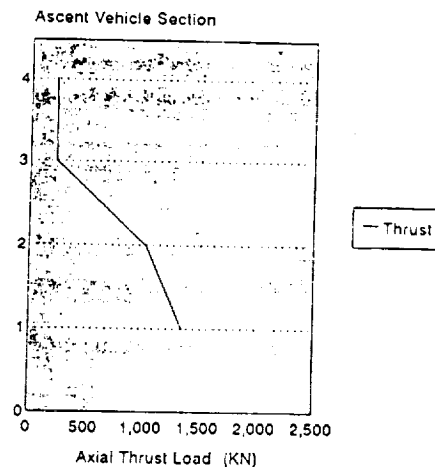


Figure 2.5.3: Axial thrust load for UML MARV.

A2.6 Center of Gravity Location. The center of gravity location for the UMLs accounted for all subsystems and respective components. The manned UML accounted for items such as:

1. Aerodynamic fins for MARV
2. Flight seats
3. Life support equipment
4. Airlock in MARV
5. Air Revitalization for MAR
6. Fuel CO
7. Fuel O₂
8. 4 main engines
9. Secondary engines
10. Aeroshield
11. 12 space suits
12. 12 batteries/fuel cells
13. 8 landing gear
14. Truss structure
15. 4 modules

The mass quantities for each component was provided by their respective division.

The center of gravity was calculated using the component mass and distances between tip of nose on the MARV and the centroid of each component. Next the total mass of the manned UML, unmanned UML, and MARV was found.

Manned UML	59,447 kg
Unmanned UML	51,142 kg
MARV	31,204 kg

The center of gravity for the UML was found using:

$$X_{cg} = \frac{\sum md}{\sum m} \quad (A10)$$

where m was the mass of each component, and d was the distance from the reference point (tip of nose cone on MARV) and centroid of each component. Center of gravity locations were as follows:

Manned UML:	
X _{cg}	-0.03 m
Y _{cg}	0.11 m
Z _{cg}	8.27 m

Unmanned UML:	
X _{cg}	0.00 m
Y _{cg}	-1.00 m
Z _{cg}	11.00 m

MARV:	
X _{cg}	0.00 m
Y _{cg}	0.01 m
Z _{cg}	8.64 m

The individual mass quantities of each subsystem and components are found in Appendices A1 and A2.

A2.7 Mass Moments of Inertia. The moments of inertia for the manned UML, unmanned UML, and the MARV were found about the center of gravity. Using the parallel axis theorem, the moments of inertia for each component were combined together to form the moments of inertia for the entire manned UML, unmanned UML, and MARV. The equations which were used to find the moments of inertia for the entire manned UML, unmanned UML, and MARV were:

$$I_{xx} = I_x + m(\bar{Y}_{cg}^2 + \bar{Z}_{cg}^2) \quad (A11)$$

$$I_{yy} = I_y + m(\bar{X}_{cg}^2 + \bar{Z}_{cg}^2) \quad (A12)$$

$$I_{zz} = I_z + m(\bar{X}_{cg}^2 + \bar{Y}_{cg}^2) \quad (A13)$$

$$I_{xy} = I'_{xy} + m(\bar{X}_{cg}\bar{Y}_{cg}) \quad (A14)$$

$$I_{yz} = I'_{yz} + m(\bar{Y}_{cg}\bar{Z}_{cg}) \quad (A15)$$

$$I_{zx} = I'_{zx} + m(\bar{Z}_{cg}\bar{X}_{cg}) \quad (A16)$$

where I_x, I_y, I_z, I'_{xy}, I'_{yz}, and I'_{zx} were the component moments of inertia, m was the mass of each component, X_{cg}, Y_{cg}, and Z_{cg} were the distances from the center of gravity of vehicle to the centroid of each component. Next all values of I_{xx} for each component was summed to find the total I_{xx} of the vehicle. Then the other moments of inertia for the vehicle were calculated by repeating the last step.

The mass moments of inertia were calculated to be:

Manned UML (Kg m²):

Ixx	15.5 X 10 ⁶
Iyy	15.5 X 10 ⁶
Izz	2.0 X 10 ⁶
Ixy	630
Iyz	2.0 X 10 ⁵
Izx	-4.6 X 10 ⁴

Unmanned UML (kg m²):

Ixx	17.5 X 10 ⁶
Iyy	17.5 X 10 ⁶
Izz	2.2 X 10 ⁵
Ixy	370
Iyz	-4.9 X 10 ⁵
Izx	-2.5 X 10 ⁵

MARV (kg m²):

Ixx	9.8 X 10 ⁶
Iyy	9.8 X 10 ⁶
Izz	4.6 X 10 ⁵
Ixy	908
Iyz	6.6 X 10 ⁴
Izx	-1.4 X 10 ⁴

A3. Radiation Protection

The problem of radiation shielding for members of the crew and for the electronics aboard the lander turned out not to impose a major design problem. The shielding of the crew was limited to martian surface because of the assumption that the crew would be staying in MTV for the trip from earth to Mars.

From models of the Martian atmosphere, it has been calculated that the atmosphere represent a radiation shield from 16 g/cm³ to 22 g/cm³, depending on the COSPAR density model is used (see table A3.1) ^{A3.1}. This is based on the radiation coming from straight overhead. If the radiation comes in at an angle the radiation protection will increase, because radiation will pass through more of the atmosphere. The current radiation limits for astronauts is 50 rem/yr for their blood forming organs (BFO). The atmosphere will reduce the effects of radiation from galactic cosmic radiation (GCR) and major solar flare events (SFE) on BFO to 19-22 rem/yr, well below the limit of 50 rem/yr. The short term dose limit for BFO is 25 rems, and this will not be exceeded by any known solar flare events as along as the astronaut is below 8 km ^{A3.1}. The atmosphere will provide the necessary radiation

shielding for the crew when they are on the surface of Mars.

Table A3.1. Martian Atmospheric Protection in the Vertical Direction
(Data from reference 3.1A)

Altitude, km	Low-density model, g CO ₂ /cm ²	High-density model, g CO ₂ /cm ²
0	16	22
4	11	16
8	7	11
12	5	8

The assumption that the crew will be staying in MTV from Earth to Mars means that lander will only need to provide radiation shielding for the electronics. Finding information on the shielding of electronics prove to be a problem. A graph in reference A3.2 was the only information found. It showed the cumulative radiation dose acquired during orbits of different altitudes around the Earth. It compared shielding thicknesses of .5 g/cm² and 3 g/cm² of aluminum. For the average skin thickness of the modules and ascent vehicle will 6mm of an aluminum alloy. This is roughly 1.5 g/cm² of aluminum shielding. From the graph, assuming the highest orbit represented (10⁴ - 10⁵ nmi) is representative of the radiation environment of the lander from Earth to Mars, cumulative radiation dose would be under 10⁵ rads, for a ten year mission ^{A3.2}. An assumption made in the design of this lander is the MTV will use a nuclear thermal rocket as a propulsion option. This will reduce the trip time from Earth to Mars to under a year ^{A3.3}. The radiation dose will be reduce by a factor of 10 of that which is accumulated for a 10 year period. The total radiation dose can then be assumed to be around 10⁴ rads. Most semiconductor devices will handle this much radiation. The more sensitive devices will be spot shielded. It should be noted that the effect shielding will not be just the skin of the module or ascent vehicle, it will be everything in

between the device and space. For instance if a device is placed behind a stringer in a module, shielding will increase to 6.7 g/cm^2 of aluminum alloy. Based on these assumption it is recommended that no extra shielding for the electronics or the crew is required, than the structure of the module and ascent vehicle.

A4. Aeroshield Structure. The overall shape of the aeroshield was defined by the Aerodynamics/Propulsion group. From that shape the Structures group pick materials and designed the structure. The basic concept for the aeroshield came from reference A4.1. It describe a flexible/deployable aeroshield that uses, for the foldable part, Tailorable Advanced Blanket Insulation (TABI) to give it the flexibility needed. Because the original report was for a smaller aeroshield, there was a number of changes in size and configuration.

The Aero/Prop group defined the shape to be a 21.5 diameter cone with a rounded nose of the radius 4.06 meters. The cone would rise at a 20 degree angle.

From the basis of reference A4.1 a deployable aeroshield was design so it would fit into a cargo hold that was 46 ft in diameter (diameter of EHLLV). This resulted in a rigid core with a diameter of 9.26m which the flexible outside ring attached to it.

The flexible ring can fold from a 21.5 diameter to a diameter 13.57 (44.5 ft). The original design had 12 ribs supporting the TABI material, but because of the larger size, 20 ribs are need to keep the maximum distance between ribs the same. The rigid core will be an aluminum shell backed by four ribs.

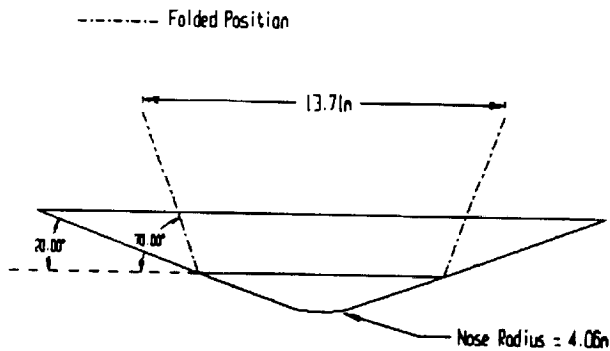


Figure A4.1. A sketch of the side view of the aeroshield in its unfolded and folded position.

The original report suggested the use of complex truss structures for the ribs, but for comparisons some simple modeling was done of the just the rib structure using I-

DEAS. The ribs tried where a channel beam which had a height of .15m, width of .15m, and a thickness of .02. The loading used was the maximum dynamic pressure on the aeroshield times a safety factor of 1.5. This produce a uniform load roughly equal to $15,000 \text{ N/m}^2$. This loading produced a deflection of over 2m on the outside edge of the shield and created stresses over the ultimate stress limit of Aluminum 2090 by almost a factor of 10. From this results it can be seen that using something other than a complex truss structure will require a massive beam to support the loading.

The thermal protection system (TPS) is made up of the TABI material and SLA-561 ablative material which is used to cover the rigid core. From analysis of the heating rate it was found that 5 mm of the SLA-561 material will need to be applied to the rigid core center. The TABI will have be at least 1.5 cm thick, but since most of the properties of TABI that are available are for a specimen that is 1.9 cm (.75 inches) thick, that it was recommended this thickness to be used for the aeroshield.

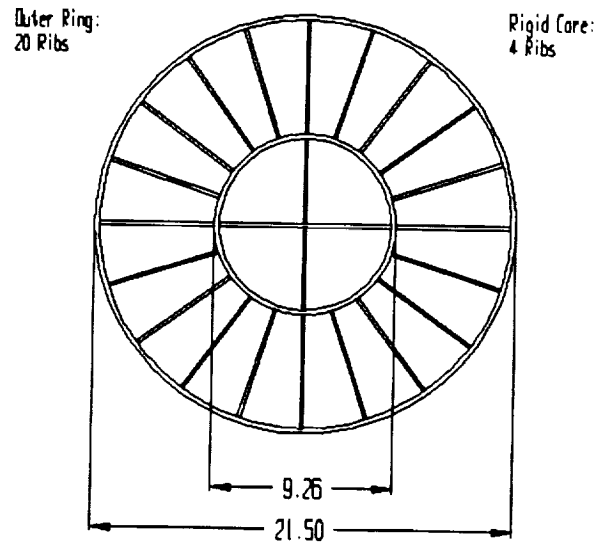


Figure A4.2. Overhead view the aeroshield, looking at inside structure.

SECTION B -- GUIDANCE, NAVIGATION, & CONTROL

Michael Kessinger
Loran Miller
Andrew Parlock

Section Design Philosophy

The Guidance, Navigation, and Control (GN&C) group of the Universal Martian Lander (UML) design team was faced with the task of designing the Mars spacecraft operations over the duration of the mission. The main functions of the GN&C group can be broken up into the following: landing site selection, Martian Transfer Vehicle (MTV) orbital parameters, trajectory control, and telecommunications.

A landing site was selected for the mission by using key characteristics that would insure a productive mission. The types of characteristics that were used in selecting the landing site were safe location, relatively mild climate, and more importantly, scientific value. The site of Elysium Mons, at 26° N and 216° W, was decided upon because it has many of these desirable characteristics.

The MTV parking orbit was selected and evaluated to insure that it would meet the criteria that would allow all mission objectives to be completed. The Manned Landing Vehicle (MLV) descent attitude control, including orbital maneuvers, descent through the atmosphere, and the landing phase, were a major design task that GN&C considered. Starting with separation from the MTV, the MLV descent phase will continue until the main engines are fired for landing. The landing phase will immediately follow the descent phase and will end at MLV touchdown.

The Manned Ascent and Rendezvous Vehicle (MARV) ascent and rendezvous phases were also considered. The MARV ascent phase will begin at lift-off from the Martian surface and end after the MARV has parked in a 240 km, circular, polar orbit. The rendezvous phase will begin a short time after the ascent phase has been completed and a systems check has been performed. The rendezvous phase will end after the MARV has completed docking procedures with the MTV.

Lastly, telecommunications were the responsibility of the GN&C group. The telecommunications design includes the Satellite Communications and Navigation System (SCAN), telemetry and command, and a GN&C computer concept. The SCAN system is a system of two satellites that will be used for uplink and downlink of information between the MTV, MARV, the base on Mars, the Russian Mir Space Station, Space Station Freedom and the Space Shuttle.

B1. Landing Site Selection

The landing site selection was the first objective for the Guidance, Navigation, and Control Division. There were three criteria used in determining the landing site; safety, climate, and experimental value. The research involved studying the information on the Viking missions and the data obtained from those missions.

First, safety was the most important factor in determining the landing location. More consideration had to be given to safety mainly because this was a manned mission. Therefore, the topographic features and climate could not be too hostile for a safe landing and stay on the surface. A location at low altitude was desired.¹ This would allow for the Universal Martian Lander to be exposed to the atmosphere longer, creating more drag and longer time frame for the parachute and aeroshield to be more effective for deceleration.

The climate was chosen that would be less malevolent. The Mars surface has high winds which could create large dust storms. However, the southern hemisphere was noted to be worse than the northern hemisphere. The wind can reach high speeds, but the low air density greatly reduces the force of the wind. Next, the northern hemisphere has the longer summer.² The warmest temperatures are near the equator and the maximum temperature is 62.6°F. The longer summer will be to the advantage of the crew members because the temperature can drop to -225.4°F.

The experimental value of the landing site was dependent on the topographic features near and around the location of a Mars base. For safety reasons, the lander could not touch down too close to any particular feature. However, as many different features should be within range of a rover or other experimental vehicle. The surface is scattered with craters, volcanoes or mountains, plains, and channels. The southern hemisphere is very heavily cratered. It does not have many smooth plain areas like the northern hemisphere which is less cratered. Volcanoes, volcanic overflow channels, and other channels could be found in either hemisphere. Some of the channels lead to the speculation that water once existed on the surface. The data collected from Viking missions suggest that there is better chance of finding water in the northern hemisphere due to a higher humidity. In addition, the channels may reveal stratification in the soil which is a valuable tool for determining the history of events. Archaeologists often use stratification of soil to determine the age of artifacts. Site selection was heavily dependent on the proximity to various topographical features.

From the information gathered, it appeared that the best landing site would be in the northern hemisphere. To meet all the criteria for safety, climate, and experimental value, there was some compromise. The area known as Elysium Mons was chosen for the landing site. This is a plain area of medium altitude approximately 26° N and 216° W. The main feature is a large volcano with a 14 km diameter caldera and several channels believed to be caused by lava overflow when it was active over a billion years ago.³ There are several small craters in the area. To the northwest there are large channels or canyons possibly formed by water. There are also tectonic troughs near.

The climate at Elysium Mons is slightly cooler than the equator. However, if the site was chosen further north, a rapid temperature drop would be experienced. The indication of channels and water suggest the any life forms would be most likely found in that area. Since the mission is for the purpose of researching and investigating Mars, a landing at Elysium Mons would be safe and valuable experimentally.

B2. Martian Transfer Vehicle Orbital Parameters

The MTV will be parked in a 250 km, circular orbit about Mars, with an inclination of 90°. Various parameters were investigated to justify the selection of this orbit. The first, the orbital period, was calculated using the following equation⁴

$$T = 2\pi\sqrt{\frac{a^3}{\mu}} \quad (B1)$$

where $a=250$ km and $\mu=43104.9314$ km³/s². Using equation (B1), the time period was calculated to be 1.84 hours. The orbital velocity of the MTV was also of concern and was calculated using the following equation⁵

$$V = \sqrt{\frac{\mu}{a}} \quad (B2)$$

The orbital velocity was calculated using equation (B2) to be 3.45 km/s.

The time of decay for the MTV in it's 250 km orbit was calculated to insure that the orbit would not substantially decay over the duration of the mission. Assuming a worst case scenario, a frontal area of 30.48 frontal area for the MTV would only occur if it were travelling 90° to it's intended configuration. Using a C_d of 1.2 and a total mass of 150,000kg, the decay lifetime was

calculated using the following equation⁶

$$t_d = \frac{h}{\sqrt{\mu R_{Mars}} B^* \rho_o} (e^{\frac{H_o}{h}} - 1) \quad (B3)$$

where $R_{Mars} = 3376.1$ km (polar radius of Mars), $H_o = 250$ km, $h = 7$ km, and where the density at 250 km was calculated using the Ideal Gas Law:

$$\rho_o = \frac{P}{R T_{avg}} \quad (B4)$$

The term B^* was calculated using the following equation

$$B^* = \frac{C_d Area_{MTV}}{Mass_{MTV}} \quad (B5)$$

Using equation (B5), the decay lifetime of the MTV in a 250 km orbit was calculated to be 1.89×10^8 years. This value is relatively large, therefore the significance of orbital decay at this altitude is not substantial enough to require orbital maneuvers of the MTV at any time during the mission.

An analysis was also performed to investigate the Mars oblateness effects on the MTV orbit. Mars, like Earth, bulges at it's equator. This bulge leads to 2 major effects on a low orbit. The first effect, called the regression of the nodes, causes the orbit plane to precess like the motion of a top. In other words, the extra mass around the equator causes a torque on the orbit. This torque will lead to gyroscopic effects in the motion of the MTV orbit. This motion causes the orbit plane to precess at the rate

$$\Omega = -\frac{3nJ_2R_{Mars}^2}{2a^2(1-e^2)^2} \cos i \quad (B6)$$

where $J_2 = 0.001082$ and is a constant, dimensionless number that characterizes the departure of Earth from a sphere and was used for an approximation for Mars' deviation from a sphere. The usual orbital elements, e and i , are the eccentricity and inclination, respectively.

The value of $e = 0$ and the value of $i = 90^\circ$.

The angular frequency, n , of the MTV orbit was calculated using the following equation⁹

$$n = \sqrt{\frac{\mu}{a^3}} \quad (\text{B7})$$

Using equation (B7), the regression of nodes was plotted versus the number of orbits. Figure B1 shows this plot and it can be seen that the node has only regressed a maximum of -0.00885 radians after approximately 9 months. Therefore, the regression of the nodes will not affect the MTV parking orbit of 250 km to a degree of concern.

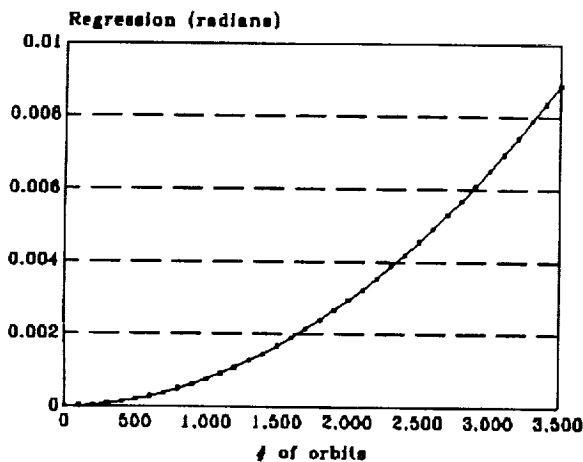


Figure B2.1: Regression of nodes for MTV in 250 km orbit up to 3500 orbits.

The second effect of the equatorial bulge is that an elliptic orbit will rotate in its own plane.¹⁰ This effect is called the advance of the perigee. The MTV orbit will be circular, therefore the advance of the perigee can be ignored.

After completing the decay lifetime analysis and the regression of nodes analysis, it was determined that the MTV's 250 km, polar, circular orbit will more than satisfy the mission requirements that were set forth by the design team.

B.3 TRAJECTORY CONTROL

B3.1 Descent Attitude Control. Once the MTV has achieved its final orbit the each of the Martian landing vehicles, the ULV, Unmanned Landing Vehicle, and the MLV, Manned Landing Vehicle, will start preparing for the descent through the atmosphere of Mars. The ULV

will be the first vehicle released from the MTV. Once the ULV has landed safely on the surface of Mars and begun transmitting back to the MTV and the MLV, MLV will begin descending from orbit. Upon release from the MTV each of the vehicles will go through a series of orbital maneuvers so to correctly orientate themselves for safe atmospheric penetration. Once descending through the atmosphere each vehicle will have the ability to continually make attitude adjustments for the purpose of maintaining stability and to correct their trajectories. Finally the MLV will land within 50 m of the ULV on the surface of Mars and began operation. For the purposes of this section and report only the MLV's descent will be analyzed in detail (It can be assumed that the ULV will go through the same phases of descent at an early time)

B.3.1.1 Orbital Maneuvers. The MLV will be released from the MTV with the use of explosive bolts and a short separation burn. However, in the configuration the MLV is released from the MTV, this being the X-axis being normal to the orbital path, Y-axis being Mars pointing, and the Z-axis being tangential to the orbit path, the MLV will be in a unstable gravity gradient configuration. Therefore upon being released, the MLV will first achieve of safe distance from the MTV, this being approximately 10 km and with this distance achieved the MLV will perform a gravity gradient stabilization maneuver, in order to orientate itself in a gravity gradient stable configuration. This configuration being the X-axis being tangent to the orbit path, the Y-axis being Mars pointing, and the Z-axis being normal to the orbit path.[see Figure B.3.1.1.1] This maneuver will consist of one 90° 957.4N burn about the Y-axis to begin the MLV rotating and the another identical burn to stop the rotation once the MLV is in its stable configuration, these burns will be performed using the Reaction Control System, RCS, on the MLV. Two more identical burns will take place just prior to atmosphere penetration so to rotate the MLV into the correct atmospheric penetration orientation. The values for this maneuvers were calculated using the following equation,¹¹

$$H = I_{yy} \omega = 2 r \times F \Delta t \quad (\text{B.8})$$

Upon achieving a gravity gradient stable configuration the MLV will orbit Mars until it is exactly 180° opposite the point of atmospheric penetration. At this point the RCS on the MLV will perform a deorbit burn. This deorbit burn will have a Δv of 0.035 km/s. This value

was found from the following equation, ¹¹

$$\Delta V = \sqrt{2 \frac{\mu}{a_1} - 2 \frac{\mu}{a_1 + a_2}} - \sqrt{\frac{\mu}{a_1}} \quad (B9)$$

where $a_1 = 250 + 3376 = 3626$ km and $a_2 = 60 + 3376 = 3436$. This deorbit burn will place the MLV into an elliptical orbit which will allow the MLV to begin atmospheric penetration at 60 km above the Martian surface, with an atmospheric penetration velocity of 3543 m/s. All of these orbital maneuvers will be performed using the MLV's RCS.

Through all orbital maneuvers the MLV will be in constant contact with the MTV, the Satellite Communication And Navigation System, or SCANS which is the satellite network which will be installed around Mars, and the ULV's navigation beacons which will be already located on the surface. This will allow the MLV to know its exact position, attitude, and velocity at all times during its orbital maneuvers and descent through the atmosphere. Enabling the MLV to adjust its trajectory continually to account for any misalignments or unforeseen problems through the orbital maneuvers phase of descent.

B.3.1.2 Descent Through Atmosphere. Upon completion of all orbital maneuvers the MLV will have an initial entrance angle into the atmosphere of -2.25° . This value presents the angle between the velocity vector of the MLV and the tangent plane of the atmosphere. This shallow entrance angle allows for an overall lower g-loading on the vehicle, although this low atmospheric angle also places the vehicle in the critical heating zone of the atmosphere for a period about twice as long as that for an entrance angle of -5° or greater. However, it has been calculated that, due to the very low density of the atmosphere, the reentry heating experienced by MLV in the Martian atmosphere will not represent a design obstacle or limitation. Whereas the additional g-loading experienced by the astronauts and MLV at higher negative entrance angles will present a definite problem both structurally and physically on the astronauts.

While descending through the Martian atmosphere the MLV will position itself with a positive 10 degree angle of attack. This value represents the angle between the velocity vector and the Z axis of the MLV, which is the forward pointing axis through descent. [see Figure B3.1.1.1]

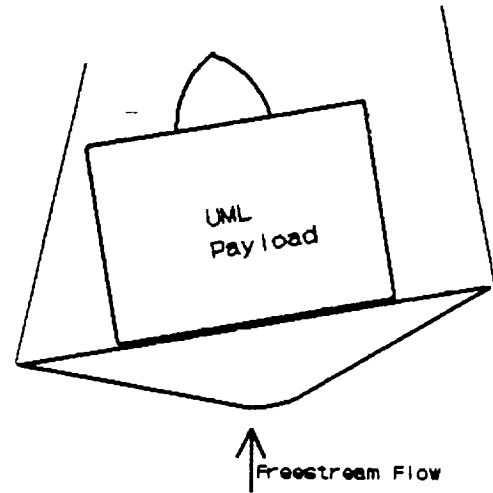


Figure B3.1.1.1: MLV in Atmospheric Descent Configuration

This angle of attack gives the MLV a lift over drag ratio, L/D, of 0.15 ¹². Figure B.3.1.1.2 graphic shows the L/D ratio versus angle of attack. Although the MLV's

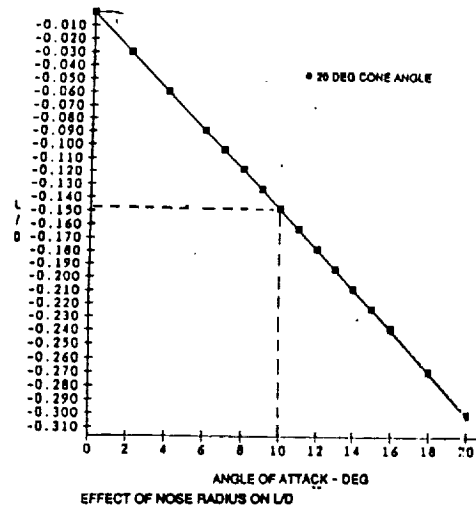


Figure B.3.1.2.1: L/D versus Angle of Attack for MLV Through Martian Atmosphere

low center of gravity enables it to be a relatively stable vehicle while descending through the atmosphere the relatively high L/D ratio is necessary for the vehicle to maintain complete control over itself. With the use of RCS and the main engines the MLV is able to maintain adequate control over itself while descending through atmosphere.

The RCS is the system that regulates most of the vehicles motion, attitude, and control while descending. To maintain sufficient controllability over the MLV through the descent phase, it is recommended that the RCS be able to perform a maximum of eight 10 deg/s maneuvers while descending through the atmosphere. [see Figure 3.1.2.2]. This number has been scaled down from the information presented in Reference B11 due to the shorter distances traversed when comparing the distance from low Earth orbit to the surface, for which the data in the reference is presented, and the distance from low Mars orbit to the surface, for which the MLV is actually traveling. The smaller amount of atmospheric variation found in the Martian atmosphere, when compared to that of Earth, also played an important role in enabling this number to be lower to an acceptable level. These maneuvers exercise control over the vehicle through the adjustment the direction of the lift vector and thus adjusting the attitude and trajectory of the vehicle. The RCS controls the direction of the lift vector by rotating the vehicle about the Z-axis, which is the axis perpendicular to the lift vector, while maintaining the a consistent 10° angle of attack. Locating the thrusters of the RCS near the edges of the aeroshield will give the RCS a moment arm of 10.25meters. With this large moment arm the thrust for each of the 8 maneuvers will be approximately 50000 N. This can be seen in Figure B.3.1.2.3, where the MLV has a moment of inertia about the Z-axis of $3.2 \times 10^6 \text{ kg m}^2$ and a moment arm of 10.25 m.

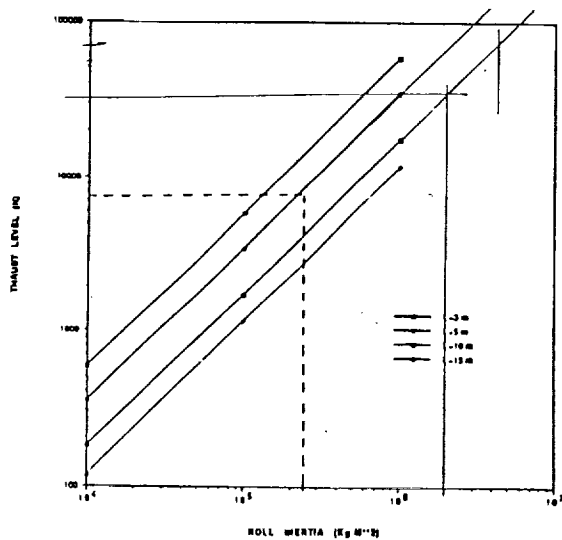


Figure 3.1.1.1: Required roll thrust for roll maneuvers at 10 deg/sec.

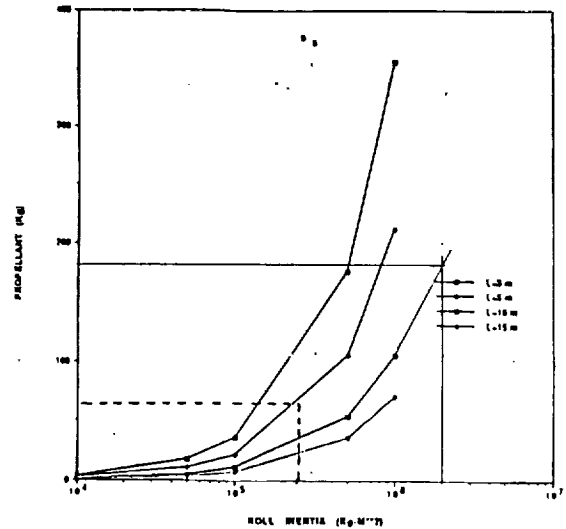


Figure 3.1.1.2: Propellant required for 12 maneuvers at 10 deg/sec max rate per maneuver.

B.3.1.3 Landing Phase. The landing phase begins at 6000m above the Martian surface. This phase begins with the release of the parachute which occurs directly after the critical heating phase of the descent is over.

Once the parachute is released the main engines will begin a 3 second warm-up phase. Once completed the main engines will be fired and the aeroshield will be released simultaneously at 5000 m above the surface of Mars. The thrust from the main engines along with the added drag from the parachutes will pull MLV away from the aeroshield. Once the aeroshield is a safe distance from the MLV explosive bolts will blow the aeroshield into 8 pieces and away from the MLV, insuring that no part of the aeroshield will negatively effect the MLV.

At approximately 2000 the parachute will be released and the main engines will take over complete control of the deceleration of the MLV, bringing it down to a soft landing within 50 m of the ULV which had previously landed.

B3.2 Ascent Attitude Control. The next step in the design process was to investigate the trajectory control for the MLV and MARV throughout the mission. The ascent analysis was started by selecting the proper pre-rendezvous orbit for which to park the MARV. The "below and behind" geometry will be used in order to allow proper phasing of the rendezvous procedure. ¹⁴ This method will allow the MARV to park in a lower orbit than the MTV in order to properly check all systems before the rendezvous procedure begins. A perigee raising maneuver will be used to bring the MARV up to the MTV's 250 km orbit.

The MARV will have to park in a lower orbit before parking in a pre-rendezvous orbit of 240 km due to the fuel and thrust constraints that the Aerodynamics/Propulsion (Aero/Prop) group obtained from their primary fuel analysis. A simple B.A.S.I.C. program was written that uses Aero/Prop's data for the MARV ascent to calculate the velocity, height, and displacement versus time of the MARV in a gravity-turn trajectory. A gravity-turn trajectory¹⁵ was chosen because it will allow the MARV to be rotated from its vertical launch angle to a horizontal position at burn-out at 60 km. This translation away from the vertical launch position will be accomplished automatically by the dynamics of a gravity-turn trajectory.

After the MARV has been in flight for 50 seconds, it will be rotated at a constant rate of 2.4 rad/s for the remainder of the burn time, or 37.5 seconds. The MARV will then circularize in a 60 km orbit. The total ΔV that will be needed to circularize in the 60 km orbit was calculated using¹⁶

$$V_{circular} = \sqrt{\frac{\mu}{R_{Mars}}} \quad (B10)$$

The velocity at burn-out was calculated using equation (B10) to be 3.54 km/s. The ΔV of the MARV after the gravity turn trajectory was calculated in the program to be 2.584 km/s. Therefore, a small ΔV of .956 km/s will be needed using the Reaction Control System at the instant of main engine burn-out in order to successfully park the MARV in a 60 km orbit. Equation B3.2.2 shows how the ΔV 's from the RCS and the gravity turn trajectory will give the desired value of ΔV .

$$\Delta V_{60km} = \Delta V_{gravityturn} + \Delta V_{RCS} \quad (B11)$$

In order to use the gravity-turn trajectory, the assumption that the thrust vector is always aligned with the velocity vector was made. This assumption is valid because the main engines of the MARV will have thrust vector control (TVC).

After the MARV is parked in a 60 km orbit, it will need to perform another series of maneuvers to reach a 240 km orbit before rendezvous can begin. The total ΔV that will be needed to circularize in a 240 km orbit,

starting in a 60 km orbit, was calculated using the following equations¹⁷

$$\Delta V_1 = \sqrt{\frac{2\mu}{a_1} - \frac{2\mu}{a_1+a_2}} - \sqrt{\frac{\mu}{a_1}} \quad (B12)$$

$$\Delta V_2 = \sqrt{\frac{\mu}{a_2}} - \sqrt{\frac{2\mu}{a_2} - \frac{2\mu}{a_1+a_2}} \quad (B13)$$

where $a_1 = 3436$ km and $a_2 = 3616$ km. The total ΔV that will be needed was calculated to be 0.0893 km/s by adding the two maneuvers:

$$\Delta V_{60km-240km} = \Delta V_1 + \Delta V_2 \quad (B14)$$

The associated coast time for the MARV, in other words the total time from 60 km to 240 km, was calculated to be 52.8 minutes using¹⁸

$$\Delta t = \pi \sqrt{\frac{a^3}{\mu}} \quad (B15)$$

The value of a is simply the average orbital height of the maneuver.

B3.3 Rendezvous Analysis. Orbital rendezvous and docking operations are essential to the execution of space missions, and the UML project is no exception. It is crucial that a safe rendezvous and docking procedure be carried out in order for the mission to be a success.

The MARV rendezvous and docking procedure will be accomplished by an autonomous system with astronaut-override capabilities. It is a valid assumption to assume that autonomous docking systems will be developed and flight rated before the UML mission is underway due to the vast research that has been completed in this area.

The astronaut override capability will allow the astronauts to control the MARV during rendezvous and docking procedures if the need arises. This system will constantly monitor the MTV's rotational and translational accelerations and guide the MARV to a soft docking. The ascent and rendezvous maneuvers were examined to determine the basic parameters involved for both the MARV and the MTV. After the MARV is parked in a

240 km orbit, a complete systems check will be made in order to insure proper safety for the rendezvous and docking stages. The rendezvous analysis began by calculating the ΔV 's that will be needed for the MARV to rendezvous with the MTV using the following equations ¹⁹

$$\Delta V_3 = \sqrt{\frac{2\mu}{a_3} - \frac{2\mu}{a_3+a_4}} - \sqrt{\frac{\mu}{a_3}} \quad (B16)$$

$$\Delta V_4 = \sqrt{\frac{\mu}{a_4}} - \sqrt{\frac{2\mu}{a_4} - \frac{2\mu}{a_3+a_4}} \quad (B17)$$

where $a_3 = 3616.1$ km and $a_4 = 3626.1$ km. The overall ΔV that will be needed for rendezvous maneuvers was calculated to be 0.00476 km/s using:

$$\Delta V_{240km-250km} = \Delta V_3 + \Delta V_4 \quad (B18)$$

The associated coast time for the MARV during rendezvous maneuvers was calculated using:

$$\Delta t = \pi \sqrt{\frac{a^3}{\mu}} \quad (B19)$$

The value of a is simply the average orbital height for the maneuver and was found using ²⁰

$$a = \frac{a_3 + a_4}{2} \quad (B20)$$

The coast time for the MARV was found to be 54.95 minutes.

Also of interest was the position that the MTV will need to be in with respect to the MARV at the start of MARV rendezvous procedures. This calculation was performed using the following equations

$$\frac{dt}{d\theta}_{250km} = \frac{T_{cir}}{360^\circ} = \frac{2\pi a^{\frac{3}{2}}}{\sqrt{\mu}} \quad (B21)$$

$$MTV_{position} = \frac{d\theta}{dt}_{250km} * MTV_{\Delta t} \quad (B22)$$

Using the above two equations, when the MTV is "behind" the MARV at 179.63°, the MARV rendezvous maneuvers should begin.

The thrust needed for rendezvous maneuvers was estimated using Newton's second law

$$(T-D)_{MARV} = (Mass * ACC)_{MARV} \quad (B23)$$

By assuming that all of the 18000 kg of fuel will be consumed in reaching the 60 km orbit, the mass of the MARV was estimated to be 15,600 kg. The velocity was estimated by calculating an average velocity between the 240 km and 250 km orbits. COSPAR data was used to obtain a density of 250000 kg/km³ at a 245 km orbit. A C_d of 1.0 and a frontal area of 14.19 m² for the MARV were obtained from the Aero/Prop group. Using the

above data, the following equation yielded a thrust of 46.92 N for rendezvous maneuvers:

(B24)

$$T = (M \cdot a) + \left(\frac{1}{2} \rho V^2 C_d A \right)$$

B4. Telecommunications

B4.1 Satellite Communication and Navigation System

B4.1.1 SCAN Parameters. The Guidance, Navigation, and Control Division assumed the task of developing a communication and navigation network for the Mars mission. The development of satellites and their orbits to deliver the optimum surface coverage and a continuous link with the Mars Transfer Vehicle and Earth was the primary objective. The Satellite Communication and Navigation System was named SCAN for short. Its purpose was to provide navigation, communication, and data transfer for the crew during the length of the mission.

Two satellites were designed to be carried on the MTV and launched into orbit after entering the polar orbit and previous to launch of descent vehicle to surface. The satellites would then enter a geosynchronous orbit around Mars.²¹ This orbit was calculated by

$$a = \left(\frac{\mu P^2}{4\pi^2} \right)^{\frac{1}{3}} \quad (25)$$

Where a was the orbit radius, μ was a constant equal to $43104.9314 \text{ km}^3/\text{s}^2$ for Mars, and P was the period of revolution. A geosynchronous orbit implies that the satellite will orbit with the same period of revolution as the planet. Therefore, the satellite would always remain in the same position relative to an observer from the planet. The period of revolution for a typical Mars day was 88,620 seconds. The orbit radius was determined to be 20,468.07 km. A geostationary orbit for Earth was 35,786 km.

Once in a geostationary orbit, one satellite would be positioned over the landing site. Note, for a satellite to be in geosynchronous orbit, it must be placed in an equatorial orbit. Otherwise, it would tend to decay into an equatorial orbit. The position of the satellite would be moved longitudinally, but not laterally from directly overhead. This meant it would be positioned 26° south of the landing site and would not pose a problem for transferring

information from the surface base. The second satellite would be 163° from the first one. This configuration, in Figure B4.1.1.1, would allow for a continuous link between surface and the MTV.²² Only two satellites were necessary. Information could be relayed from one to the other without interference from the planet. In addition, there was no "blind spot" or position of the MTV that could not receive information from one of the SCAN satellites. Therefore, there would be a contest communication link independent of the transfer vehicle's location.

B4.1.2 SCAN Performance and Capabilities. The surface base, MTV, and two satellites make up the SCAN system. The system would provide navigational aid during descent and ascent stages and provide communication transfer during the stay on the surface. Both satellites would contain equipment to perform this dual role in the mission.

Section 4.1.2.1: Navigation. The SCAN satellites would serve as positioning beacons during descent, ascent, and rendezvous stages of the mission. Using the MTV with the satellites provides three points of reference for the command module's flight computer to determine its position in space. When on the ground, the satellites could be used by a rover for navigation purposes. Using the land base with the satellites provides three points of reference for positioning for crew members on an expedition. Especially in the case of a sudden, blinding dust storm the satellites could relay navigation information to a rover so it could pin-point its location and determine the direction to the base.

Section 4.1.2.2: Communication. Communication is vital to any space mission. Through the use of the SCAN system crew members would always be informed in the case of an emergency. Communication to Earth could be obtained through the Deep Space Network (DSN) currently in operation. Contact between the crew members and the MTV or DSN satellites could be obtained through a network concept designed by the GN&C Division.

The main feature of the network involves leaving the largest portion of computers and data storage equipment on the MTV in a polar orbit. Then workstations on the surface could transfer information to the transfer vehicle via the satellites. However, if the transfer vehicle was flying in close proximity to the surface base, then the satellites could be bypassed and information sent directly to the MTV. Each experiment would contain its own

workstation. Figure B4.1.2.2. demonstrates this concept. The advantages would be less weight transfer between MTV and surface and less chance of failure due to a mishap involving crew members. A satellite dish, transmitting, receiving, and relay equipment would be needed on the surface as well as the MTV.

Section B4.1.2.3: Satellite Characteristics. The SCAN satellites were modeled and scaled after the Tracking Data and Relay System (TDRS) currently used by NASA as a switchboard in the sky. The SCAN satellites were much smaller and more mission specific. The SCAN system would only be concerned with one base and one orbiting vehicle. This would greatly reduce size and mass. Current systems are highly reliable over several years. By assuming the SCAN satellites would only be used for the duration of the mission, the need for complete duplicates was neglected. Instead, each satellite would contain several redundant systems and hardware which would most likely fail.²³ A list of some components and redundancies with mass estimates was listed in Table B1.

The total weight was estimated to be 300 kg per satellite. The satellites would be powered by solar ray during exposure to the sun and any excess energy would be stored in fuel cells for night operation along with the batteries. The receivers and transmitters on the satellites would have a field of view of 24°. This requirement was necessary for one satellite to communicate with the other and the surface or MTV. Figure B4.1.2.2. explains the field of view perception. The large orbit radius allows for the entire planet and MTV orbit to be within this field of view.

The design of the SCAN system and network was believed to reduce cost by limiting the size of the satellites. A reduction in weight was expected to allow for more experiments and supplies. Without taking up space for the major computer systems and data storage systems, the already limited living space would be increased. Also, these systems would require large amounts of power and cooling that could be better provided by the nuclear powered MTV.

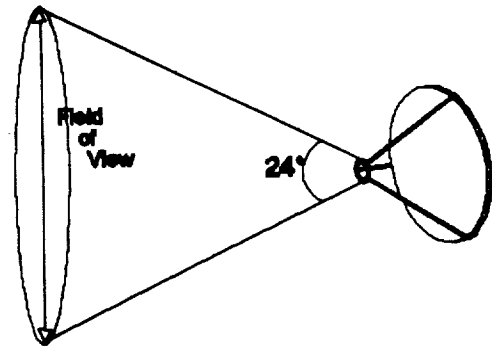


Figure B4.1.2.3.1: Visual explanation of field of view.

B4.2 Telemetry and Command

The GN&C Division designed a conceptual computer system to handle vehicle dynamics and mechanics. (see Table 4.2.1) The design was for a system which would gather information from sensors, evaluate vehicle parameters, and make any adjustments. The design involved integrating a fully autonomous system into the command structure. The pilot monitors the system, but allows the computer to make all decisions. However, at any time the pilot could override the computer. Fully autonomous implies that the descent, ascent, and rendezvous stages of the mission could be carried out by the autopilot.

Some of the characteristics of the computer which improve its efficiency and reliability were determined for processors and displays. From research, a system containing several computer processors had many advantages. With multiprocessors comes shorter buses. The bus is the architecture and distance that information has to travel within the processor. A shorter bus and many processors increase computation and software execution. The system was also less vulnerable to complete failure. If one would fail, information could be re-routed and the entire system could remain operational [B25]. With interchangeable processors, the crew could quickly fix or replace the bad parts. LCD displays were chosen because they require less space and power than other displays. In addition, LCD displays have high resolution and reliability.

A block diagram was developed to demonstrate the operation of the GN&C computer system. The diagram was shown in Figure B.4.2.1. During the descent of the manned lander navigation beacons on the unmanned lander sent to the surface would act as a homing device to ensure a landing in close proximity. The SCAN satellites could be used for positioning. Information from the satellites and beacons would be transferred into the flight computer. At the same time information would be gathered from the inertial measurement unit (IMU). The IMU and sensors would measure velocity or acceleration, angular velocity or acceleration, altitude, attitude, wind loads, and other forces or vehicle dynamics. Then the flight computer would determine trajectory, stability, and flight path and correct for any errors or activate the necessary control devices.²⁴ The control devices include main engines and the reaction control system (RCS).

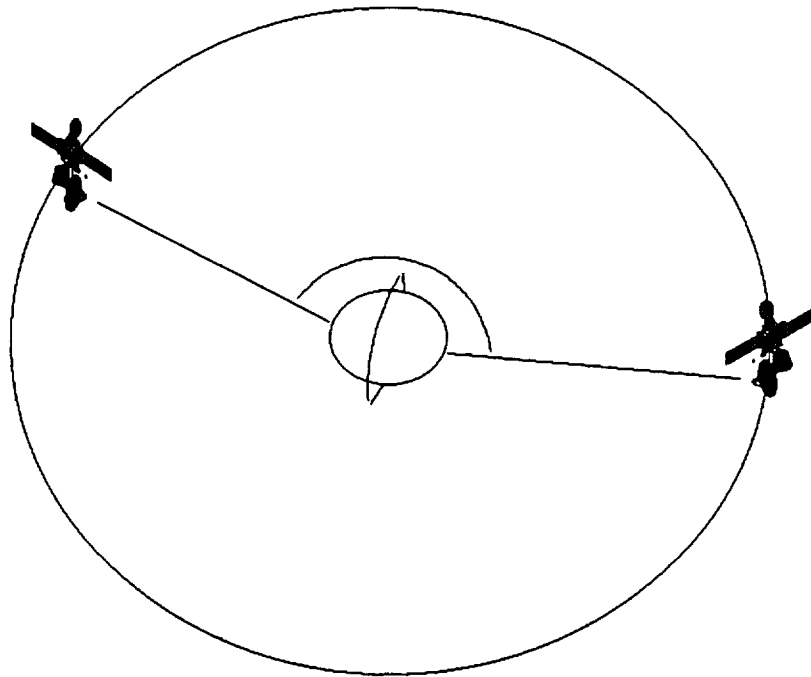


Figure B4.1.1.1: Schematic of SCAN satellite parameters.

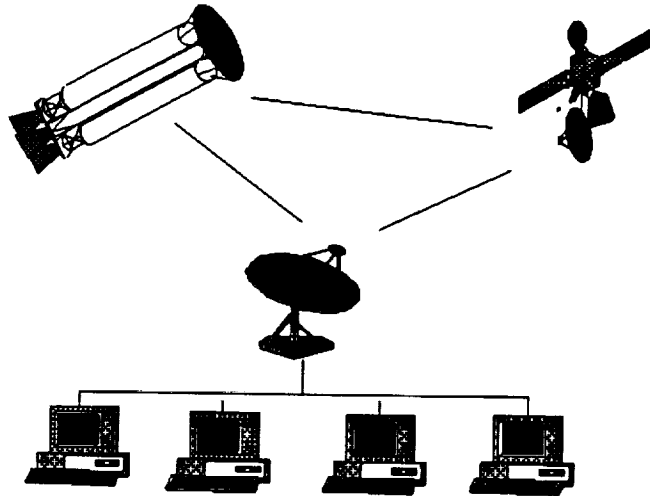


Figure B4.1.2.2.1: Schematic of network concept for communication.

Table B4.2.1 : List of some components in SCAN satellites.

EQUIPMENT	QNTY	WEIGHT kg, EA	REDUN- DANCY
Low Noise Amp.	11	0.14	10
Power Amplifier	3	0.27	3
Power Amplifier	3	0.14	3
Down Converter	10	2.27	9
DC/DC Converter	1	1.8	2
Up Converter	5	2.27	5
Demodulator	9	1.36	8
RF Demultiplexer	1	0.68	--
Power Combiner	1	0.3	--
Modulator and Source	1	2.27	1
Feed Assembly	1	1.59	--
Antenna	1	31.8	--
Antenna Controller	6	0.23	12
Antenna Control Microprocessor	1	0.23	2
Acquisition and Tracking Receiver	1	0.55	1
Electric Circuitry	1	10	1
Hardware/Structure	1	18.4	--
Solar panels and Batteries		40	
Fuel and Motors		52	

determine trajectory, stability, and flight path and correct for any errors or activate the necessary control devices ²⁴. The control devices include main engines and the reaction control system (RCS). The computer system continuously measures, evaluates, and adjusts during each maneuvering stage. The RCS could be used for small adjustments for

attitude or for rendezvous. During the landing and rendezvous phases, the RADAR equipment would aide the computer in positioning and altitude determination.

The system would include a hazard detection and avoidance subsystem. The computer must also be designed to recognize bad data from faulty sensors or equipment. This would include error propagation, fault detection, fault management, and system reconfiguration.

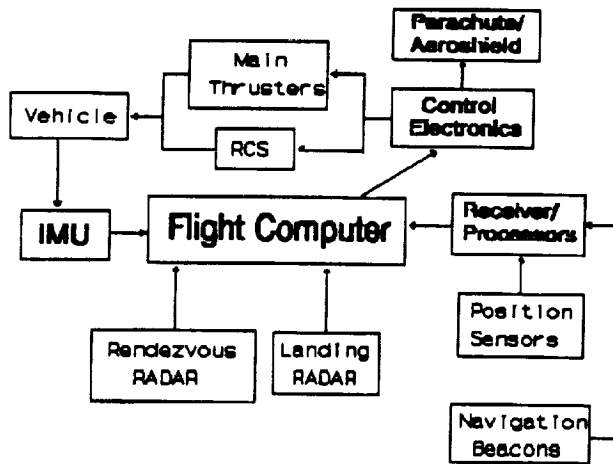


Figure B10: GN&C computer block diagram.

The GN&C computer would be located in the command module. The total weight of the system was estimated to be 245 kg plus another 200 kg for the communication equipment. This equipment for communication would include receivers, transmitters, antennas, processors, controls, and displays. The GN&C computer equipment added to the lander includes cameras, RADAR, docking lights, and sensors. For an unmanned lander, the computer system could be placed on a pallet on the lander.

SECTION C: AERODYNAMICS AND PROPULSION

Raquel Johnson
Michael Kincheloe
David MacDonnell

C2

Section Design Philosophy

The purpose of the Aerodynamics and Propulsion division was to determine the aerodynamic characteristics for the descent and ascent phases of the mission. The Descent Aerodynamics portion of the design consisted of the following:

1. The determination of acceleration and velocity profiles as the UML descends through the Martian atmosphere.
2. Design of an aeroshield to protect the UML through the critical heating phase of atmospheric entry and to provide drag for deceleration.
3. Determination of parachute characteristics such as size and height of deployment.
4. Aerodynamic heating calculations.

For the Ascent phase of the mission the Aerodynamics and Propulsion division determined the velocity and acceleration profiles of the MARV as it ascends from the Martian surface to rendezvous with the MTV. This also required determination of the MARV displacement and burnout time of the main engines. Another primary responsibility of the Aerodynamics and Propulsion division was selection of the propulsion systems:

1. Attitude control system.
2. Secondary propulsion system for orbital adjustment.
3. Primary propulsion system for ascent and descent.
4. Fuel selection and production for ascent stage.

C1. Atmospheric Entry

C1.1 Descent Aerodynamics. A first-order approximation of ballistic entry for Mars descent was used for preliminary design purposes. The first-order approximation assumed that the ULV and the MLV will follow a ballistic trajectory during descent through the Martian atmosphere.

Velocity was given as a function of height from the first order approximation using Equation C1

$$V = V_{Entry} e^{\left(\frac{H_s}{2}\right) \left(\frac{\rho_o}{\sin \gamma_s}\right) \left(\frac{SC_D}{m}\right) e^{-\left(\frac{H}{H_s}\right)}} \quad (C1)$$

and acceleration as a function of height using Equation

$$a = \rho_o (V_{Entry})^2 e^{\left(\frac{2\rho_o g H_s}{2\beta \sin \gamma_s}\right) e^{-\left(\frac{H}{H_s}\right)} \left(-\frac{H}{2H_s}\right)} \quad (C2)$$

where V_{Entry} is the atmospheric entry velocity, ρ is the sea level density, H_s is the scale height of the Martian atmosphere, γ is the entry angle, C_D is the drag coefficient of the lander, H is the altitude, and β is the ballistic coefficient.^{1,2}

The ballistic assumption used in the model remained valid until an altitude of 6000 meters. At this height, the parachute was ejected providing additional drag. From this point, the descent of the UML was modeled using Newton's second law where the sum of the aerodynamic forces would be equal to the UML's deceleration.

The Martian atmosphere was modeled as an exponentially decaying atmosphere using Equation C3

$$\rho = 0.0182 e^{-\left(\frac{H}{3216}\right)} \quad (C3)$$

This equation was based on data from the revised COSPAR atmospheric model.³

Although other atmospheric models were available such as Viking summer, Viking winter, and original COSPAR, as reported by Tartabini and Suit, variations in the atmospheric models have little effect on the Mars entry characteristics.³ The revised COSPAR model provided a suitable mean atmospheric model sufficient for Mars atmospheric entry simulation. Figure C1.1.1 shows Equation 3 plotted against the revised COSPAR model.

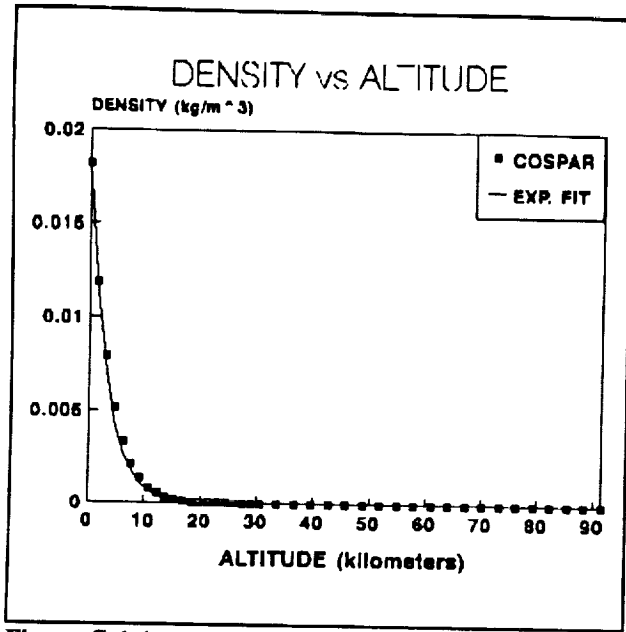


Figure C.1.1.1: Plot of density versus altitude for Martian atmosphere showing comparison of experimental fit (Equation C3) and revised COSPAR data.

A B.A.S.I.C. computer program, entitled ADP.BAS, was written based on Equations (C1-C3) to allow investigation of the effects of variations in the UML parameters on descent velocity and acceleration (see Appendix C1.1 for listing of the computer code). The program allowed input and variation of the applicable UML parameters (see Table C1.1.1). The program provided output of the UML deceleration and velocity as functions of altitude (see Figures C1.1.2 and C1.1.3). Using the program the Aerodynamics and Propulsion division was able to investigate the sensitivity of the UML's deceleration and velocity profiles to changes in selected UML parameters.

As a design constraint, it was desired to have a maximum deceleration of 3 Earth g's (29.43 m/s²) for the MLV. For the final MLV configuration (Table C1.1.1) with a Mars entry angle and velocity of -2.25° and 3500 m/s respectively, the 3 Earth g limit was maintained.

Table C.1.1.1: ADP.BAS inputs for the final configuration of the MLV.

UML PARAMETERS	VALUES
Aeroshield C _D	1.58
UML Mass	59,447 kg
Frontal Area	363 m ²
Entry Angle	-2.25°
Entry Altitude	60,000 m
Entry Velocity	3540 m/s
Parachute Diameter	15 m
Parachute Deployment Height	6,000 m

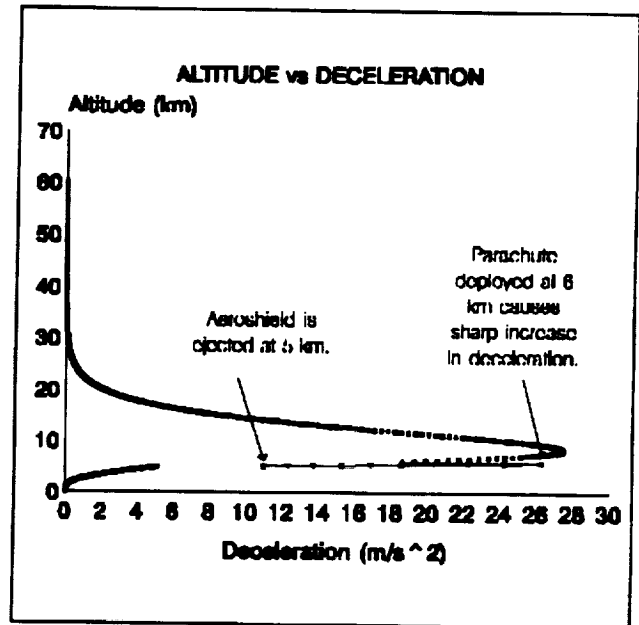


Figure C1.1.2: UML deceleration as a function of altitude.

It is important to note that Figure C1.1.2 shows that the 3 Earth g acceleration limitation was satisfied. Figure

C1.1.2 also demonstrates the deployment of the parachute and the ejection of the aeroshield.

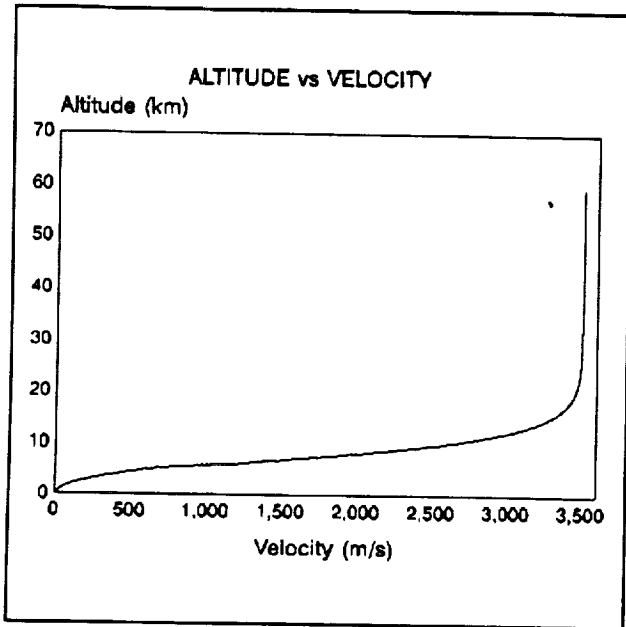


Figure C1.1.3: Velocity versus altitude for the final UML configuration.

C1.1.1 Aeroshield Specifications. An acceptable aeroshield was designed to protect the UML's modules and truss from impinging flow and aerodynamic heating during atmospheric entry. The aeroshield was selected from a NASA Marshall Space Flight Center design of a Lunar Transfer Vehicle Aerobrake⁴. The aeroshield is a spherical/cone as seen in Figure C1.1.1.1.

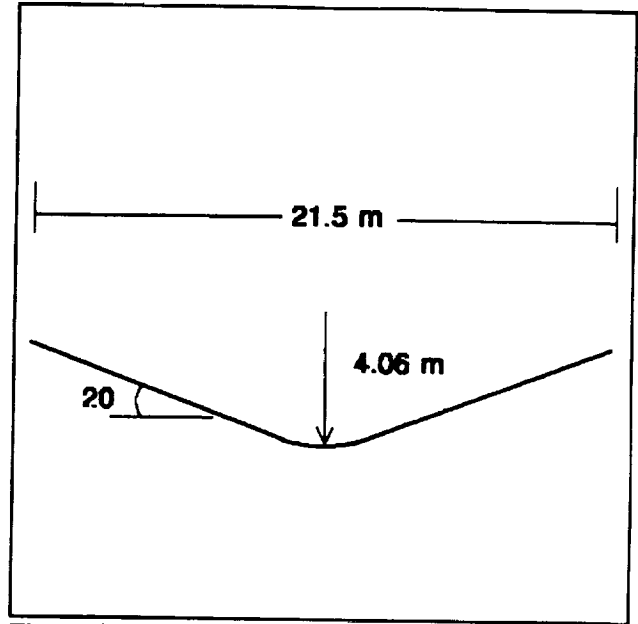


Figure C1.1.1.1: Aeroshield configuration.

This geometry was selected due to its simple deployment and maneuverability. The symmetry of the aeroshield allows the UML to perform a ballistic entry, however when an angle of attack is introduced a lift is created allowing maneuverability. This maneuverability was needed by the Guidance, Navigation, and Control division so the MLV can be guided to within range of the ULV.

As seen in Figure C1.1.1.1, the radius of the sphere is 4.06 meters and the cone angle is 20°. The total diameter of the aeroshield was calculated by determining the impingement flow around the edge of the aeroshield.

To find the impingement angle the envelope of maneuverability was selected to be an angle of attack of ± 10 degrees. The impinging flow was calculated at this angle of attack. According to Figure C1.1.1.2, which was derived from data by Hair and Engel of REMTECH, INC., the total impingement angle would be 22° toward the payload as seen in Figure C1.1.1.3.

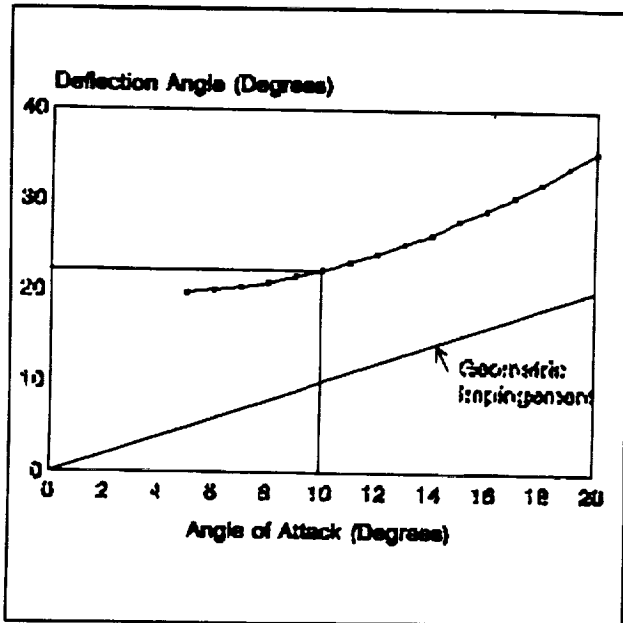


Figure C1.1.2: REMTECH, INC., impingement angle data.

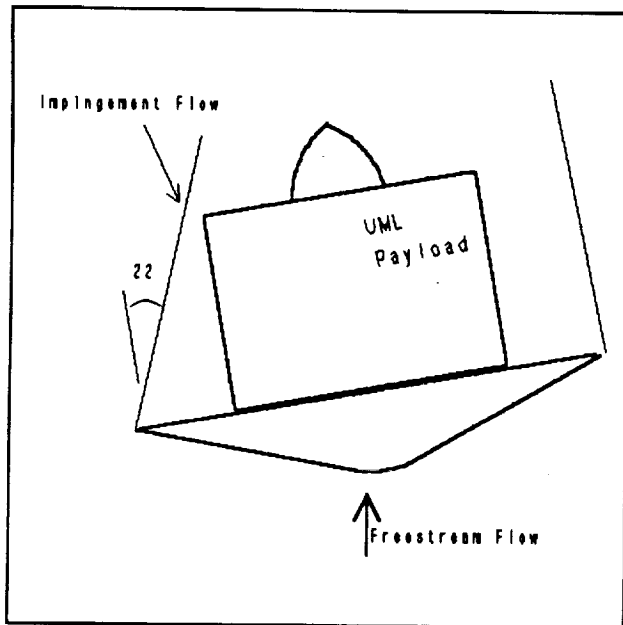


Figure C1.1.1.3: Impinging flow at a 10° angle of attack.

Figure C1.1.1.3 shows the flow over the aeroshield at an angle of attack of 10° with a diameter of 21.5 meters. This diameter is sufficient to protect the payload from the impinging flow.

The radius of the sphere was 4.06 meters. This was obtained from a ratio of the radius to the diameter from

an aerobrake in a NASA report on Lunar Transfer Vehicle Aerobraking (see Appendix C1.1.1 for sample calculations).⁴ The aerodynamic characteristics of the aeroshield are dependent on the angle of attack. For an angle of attack of zero the lift over drag is zero and the drag coefficient is 1.58. For an angle of attack of -10 degrees the lift over drag is 0.15 while the drag coefficient is 1.525. Figure C1.1.1.4 shows the plot of lift over drag versus angle of attack.

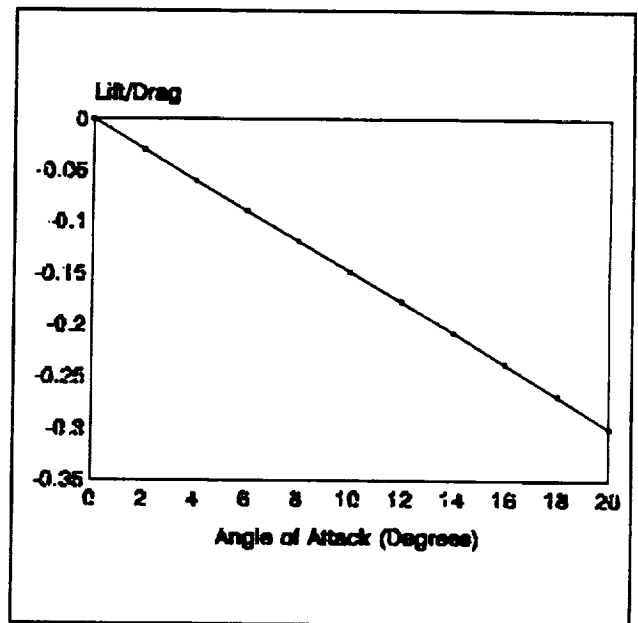


Figure C1.1.1.4: Lift/Drag versus angle of attack.

Figure C1.1.1.5 shows the relation of angle of attack to the drag coefficient.

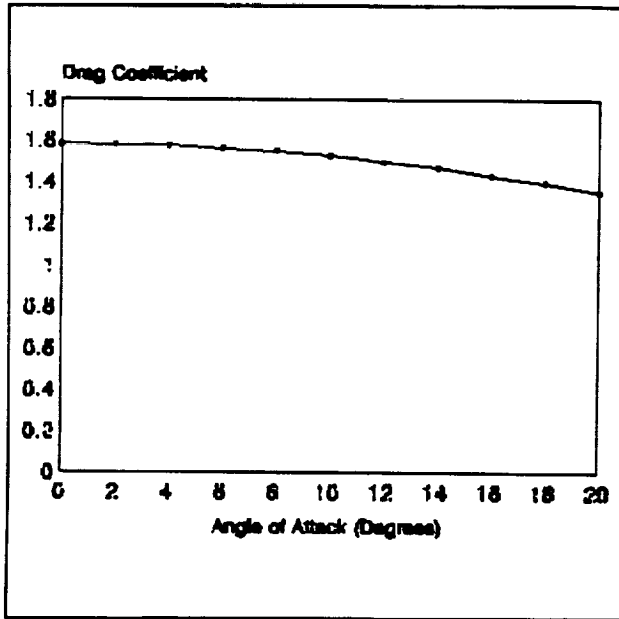


Figure C1.1.1.5: Drag Coefficient versus angle of attack.

C1.1.2 Parachute Specifications. The parachute diameter and deployment height were determined by using Newton's second law

$$F_p + F_A = ma \quad (C4)$$

where F_p and F_A are the forces due to drag from the parachute and the aeroshield, respectively, m is the mass of the UML, and a is the acceleration due to gravity which was limited to 3 Earth g 's. To find the drag forces due to the parachute and aeroshield Equations C5 and C6 were used

$$F_p = \frac{1}{2} \rho V^2 C_{D_p} S_p \quad (C5)$$

$$F_A = \frac{1}{2} \rho V^2 C_{D_A} S_A \quad (C6)$$

where ρ is the density, V is the velocity, C_D is the coefficient of drag for the parachute and aeroshield, and S is the frontal surface area. S_p was evaluated by

$$S_p = \frac{\pi D^2}{4} \quad (C7)$$

where D is the diameter of the parachute. C_D for the parachute was estimated by data from Fluid Mechanics for drag of two-dimensional bodies at $Re \geq 10^4$.⁵ An estimation of C_{DP} for a 2-D half-cylinder was 1.7 while C_{DA} was 1.58 for $\alpha = 0^\circ$. For altitudes where the UML has ejected the aeroshield, the C_D for the UML was estimated to be 1.0. Equations (C4-C7) were included in the ADP.BAS computer code since many iterations were needed in order to determine the optimum parachute specifications. While the deceleration was limited to 3 Earth g 's, the parachute diameter and deployment height were varied. At 6,000 m, the 15 m diameter parachute made of dacron will be ejected from the UML by a mortar.⁶ The parachute's weight was estimated to be 40.87 kg by performing a ratio calculation from the Viking mission supersonic parachute tests (see Appendix C1.1.2 for sample calculations).⁶ The suspension line length was determined to be 25.5 m.⁶

C1.1.3 Thermal Heating Analysis. During atmospheric entry the UML's aeroshield will experience massive heat flux from both radiation and convective heat transfer. Calculations for the heat fluxes, Q_r and Q_c were included in the ADP.BAS computer code. Table C1.1.3.1 shows temperature with respect to altitude in the Martian atmosphere.⁷

Figure C1.1.3.1: Temperature as a function of altitude in the Martian atmosphere.

ALTITUDE (km)	TEMPERATURE (°K)
0	214
5	212
10	206
15	194
20	186
25	181
30	176
35	168
40	164
45	158
50	154
55	148
60	145
65	142
70	140
75	140
80	140
85	140
90	140
95	140
100	140

These values were curve fitted and included in the ADP.BAS computer program. Revised COSPAR data was used to create Table C1.1.3.2 which shows density as a function of altitude.⁷

Table C1.1.3.2: Density from revised COSPAR data as a function of altitude in the Martian atmosphere.

HEIGHT (m)	DENSITY (kg/m ³)	HEIGHT (m)	DENSITY (kg/m ³)
0	1.82e-2	33528	1.06e-7
1524	1.19e-2	36576	3.31e-8
3048	7.89e-3	39624	1.13e-8
4572	5.18e-3	42672	4.18e-9
6096	3.35e-3	45720	1.25e-9
7620	2.11e-3	48768	5.65e-10
9144	1.31e-3	51816	3.29e-10
10668	8.04e-4	54864	2.14e-10
12192	4.87e-4	57912	1.49e-10
13716	2.86e-4	60960	1.07e-10
15240	1.66e-4	64008	7.89e-11
16764	9.40e-5	67056	5.89e-11
18288	5.26e-5	70104	4.43e-11
19812	2.89e-5	73152	3.36e-11
21336	1.57e-5	76200	2.57e-11
22860	8.41e-6	79248	1.97e-11
24384	4.50e-6	82296	1.52e-11
25908	2.41e-6	85344	1.19e-11
27432	1.29e-6	88392	9.32e-12
28956	6.95e-7	91440	7.37e-12
30480	3.75e-7		

The convective and radiation heat transfer coefficients were dependent on values for the Tailorable Advanced Blanket Insulation (TABI) since the thickness of both the aluminum and the SLA-561 ablative material on the aeroshield were considered negligible when compared to the thickness of TABI. The values of both thermal conductivity ($k = .01 \text{ W/mK}$) and emissivity ($\epsilon = .8$) of TABI were assumed to be constant at temperatures of 311°K and 294°K , respectively.⁸

The convective heat flux was calculated using Equation C8

$$Q_c = \frac{h_c(T_s - T_a)}{A} \quad (C8)$$

where h_c is the convective heat transfer rate, T_s is the surface temperature of the aeroshield, T_a is the ambient temperature, and A is the frontal surface area of the UML. The value of h_c was determined by

$$h_c = Ck \left(\frac{V}{\nu R} \right)^{1/2} Pr^A \quad (C9)$$

where C is a constant, .933 for a 2-D sphere approximation, k is the thermal conductivity, V is the velocity, ν is the kinematic viscosity, R is the center cone radius of the aeroshield and Pr is the Prandtl number.⁹ Velocity was obtained from the ADP.BAS computer program and R is 4.06 m. Equation C10 was used to calculate kinematic viscosity

$$\nu = \frac{\mu}{\rho} \quad (C10)$$

where μ is dynamic viscosity and ρ is density. The dynamic viscosity was determined by using Sutherland's Law

$$\mu = \mu_o \left(\frac{T_s}{T_o} \right)^{3/2} \left(\frac{T_o + 110}{T_s + 110} \right) \quad (C11)$$

where the subscript o refers to a reference value. From Heat Transfer, the value of μ_o for CO_2 at $200^\circ K$ is $1.02e-5$ kg/ms.¹⁰ The Prandtl number was determined by equation C12

$$Pr = \frac{Cp\mu}{k} \quad (C12)$$

where Cp was assumed constant at 759 J/kgK. T_s was found by relating the free stream flow conditions to the local conditions

$$T_s = T_a \left(1 + \frac{\gamma - 1}{2} M^2 \right) \quad (C13)$$

where γ is 1.32 for Mars and Mach number was given by

$$M = \frac{V}{a} \quad (C14)$$

where V is velocity and a the local speed of sound. To obtain a , equation C15 was used

$$a = \sqrt{\gamma RT_s} \quad (C15)$$

where the natural gas constant, R is 191.46 for Mars. Although the UML will be at Mach numbers in the hypersonic range, it was assumed that equations for a calorically perfect gas still applied when calculating T_s behind the shock wave. Figure C1.1.3.1 justifies that assumption by showing that since the UML's velocity will be 3.5 km or less it will follow the curve of a calorically perfect gas close enough to assume the equations are still applicable.¹¹

Table C1.1.3.3: Radiation and convective heat fluxes during atmospheric entry on Mars.

HEIGHT (km)	Q _r (kW/m ²)	Q _c (kW/m ²)
20	232.737	39.272
32.2	256.511	6.147
40	255.681	1.837
50	254.162	.39

As Table C1.1.3.3 indicates, the radiation heat flux far exceeds that due to convection. The maximum heating range will occur at 32.2 km which is a radiation heat flux of 256.511 kW/m².

C1.2 Ascent Aerodynamics. The ascent phase was modeled by using Milne's Method of integration which is considerably more accurate than Euler's method.¹² A program was constructed to cover the iterations used in this method. The program modeled a rocket which accelerated straight up and did not take a gravity turn into consideration.

The acceleration of the ascent module was limited to 3 Earth g's or 29.43 m/s². The module must be throttled down from full thrust of 1,340,000 Newtons to meet this criteria. The fuel was estimated to be over half the MARV weight at 18,000 kg. The total ascent module weight, including fuel, was found to be 31,204 kg. The nose cone geometry was selected to be an ogive configuration with a radius of curvature of 3.25 meters. This nose cone gives the module more space for equipment and crew. Figure C1.2.1 illustrates the dimensions of the MARV.

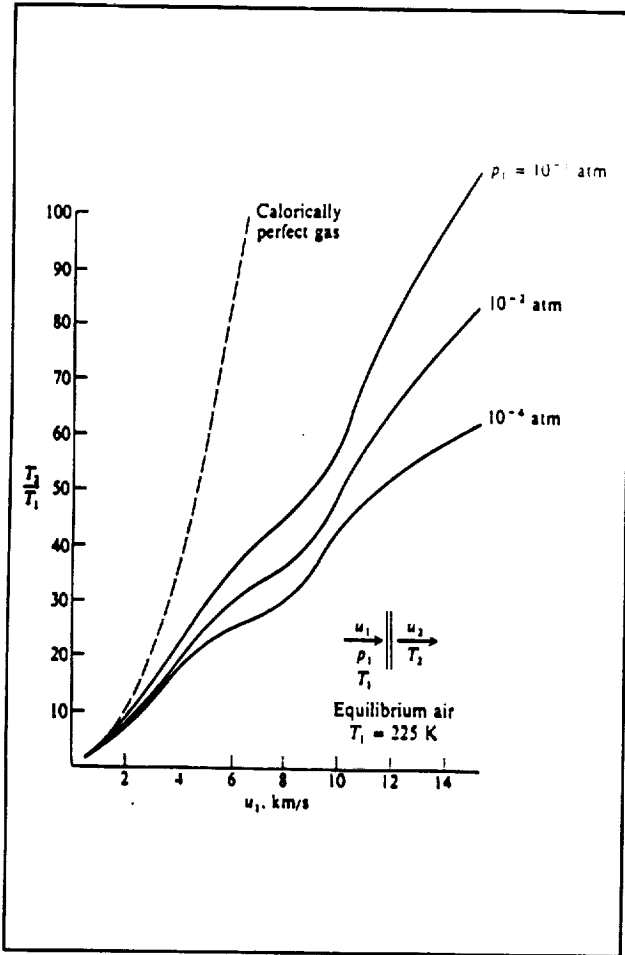


Figure C1.1.3.1: Velocity as a function of T₂/T₁ for a calorically perfect gas.¹¹

Radiation heat flux was determined by evaluating Equation C16

$$Q_r = \frac{h_r(T_s - T_e)}{A} \quad (C16)$$

where h_r is the radiation heat transfer coefficient given by

$$h_r = 4\epsilon(5.67e-8)T_m^3 \quad (C17)$$

where T_m is the average between T_s and T_e.¹⁰ Table C4 lists values of expected convection and radiation heat flux from the aeroshield of the UML during atmospheric entry.

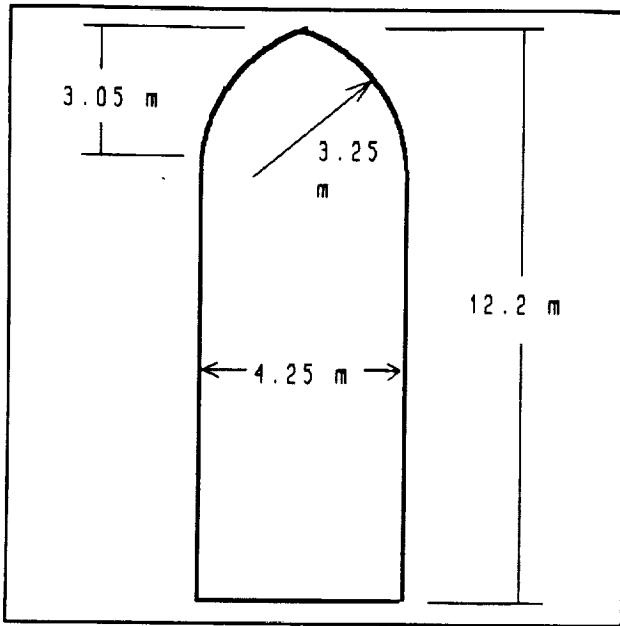


Figure 1.2.1: MARV dimensions.

The drag coefficient of the ascent module was calculated within the iterations of the program. The skin friction drag coefficient was estimated to be 0.04. While the wave drag coefficient was found using the following equation.¹³

$$C_{Dw} = \left(0.083 + \frac{0.096}{M^2} \right) \left(\frac{2\pi n^{-1} \frac{1}{d}}{10} \right)^{1.8} \left(1 - \frac{2 \left(1.9 \left(\frac{n}{d} \right)^2 - 16 \right)}{28(M + 18) \left(\frac{n}{d} \right)^2} \right) \quad (C18)$$

where M is the Mach number, n is the length of the nose cone, and the d is the diameter of the MARV.

The ascent engines are fueled by carbon monoxide and oxygen which create a specific impulse of 305 seconds. The specific fuel consumption was calculated using Equation C19.

$$\dot{m} = \frac{\text{Thrust}}{I_{sp} g} \quad (C19)$$

where g is the acceleration of gravity for Earth.

According to the ADP.BAS program the time to burnout for the ascent module is 87.5 seconds. If the module were to accelerate straight upward, it would reach an altitude of 105 kilometers. The velocity would peak at 2.5 km/s. The module will not reach 100%

thrust due to the 3 Earth g limitation. The ascent module will have to be throttled down even more due to acceleration from a gravity turn. This gravity turn places the ascent module in a circular orbit around Mars.

C2. Propulsion

C2.1.1 Engine Specifications. The final descent sequence will begin with ignition of the four primary system's landing rockets. Each engine was designed to use in situ propellant. Currently, these rocket engines are theoretical. Consequently, all specifications in this report were based on estimates that have been established by researchers at NASA's Lewis Research Center.¹⁴ It was determined that four engines each with a thrust of 334 kN (1340 kN total), a specific impulse (Isp) of 305 sec, and a weight of 641.8 kg (2567.33 kg total) are to be used for the descent phase of the mission. Liquid carbon monoxide and oxygen were chosen as the propellants.

Each engine is 3.0 m tall with a nozzle diameter of 1.2 m and a chamber length of 1.0 m. Figure C2.1.1.1 shows the dimensions of the engine.

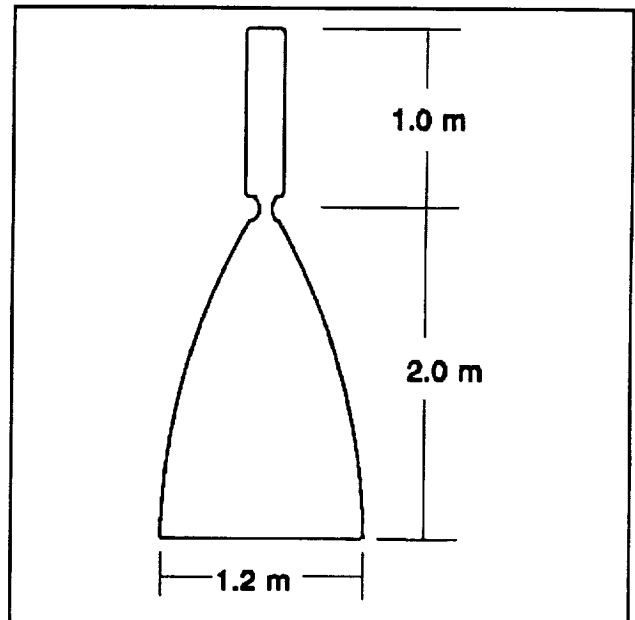
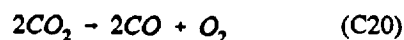


Figure C2.1.1.1: Dimensions of the Primary system engine.

These dimensions were based on information in a NASA technical memorandum entitled, "Propulsion Systems Using In-Situ Propellants for a Mars Ascent Vehicle."¹⁵

C2.1.2 Fuel Determination. A major component in the design process for the UML was selection of a propellant for the ascent/descent engine. For simplicity and weight/space savings it was desired that the same propulsion system be used for both the descent phase and the ascent phase. In order to insure that the MLV stays under the 3 Earth g acceleration limit, the engine must be throttleable. For this reason, liquid propellants were chiefly investigated.

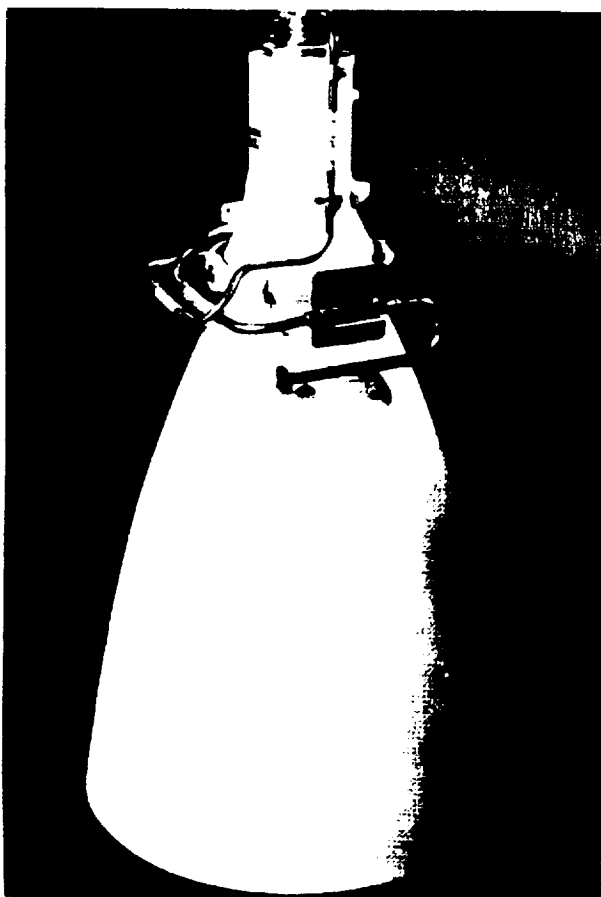
There is a significant amount of current research directed toward in situ (on site) propellant for return from the Martian surface.^{16,17} Due to the size limitations on the MLV and the ULV it would be advantageous to be able to manufacture the return propellant on the surface of Mars. Several options of indigenous propellants were investigated: Methane and Oxygen, Acetylene and Oxygen, Acetylene as a monopropellant, and Carbon Monoxide and Oxygen. The first three options require that a quantity of hydrogen be transported to Mars from Earth.¹⁸ Carbon Monoxide and Oxygen, however, can be manufactured directly from the Martian atmospheric CO₂, thus gaining additional weight savings. Carbon Monoxide and Oxygen are produced from the thermal decomposition of atmospheric CO₂:



CO and O₂ separate from CO₂ at temperatures in excess of 1000° K over a catalyst of zirconia. In order to produce the necessary 18,000 kg of fuel for liftoff from the Martian surface (11,239 kg of CO and 6,548 kg of O₂), a production rate of 150 kg per day was assumed. At this rate it will take 120 days to produce the required propellant. The Mars surface mission is designed for 180 days leaving a 60 day window when fuel production must be completed. A 50,840 kg production plant could produce 784 kg of carbon monoxide and oxygen per day.¹⁹ This 50,840 kg plant was to have been set up by several unmanned missions prior to the arrival of a manned mission. Due to the lower production rate required for the UML mission and the desire to limit the number and complexity of precursor missions, the fuel production system will be transported to Mars in the ULV. A fuel production system must be designed that is capable of producing the required 150 kg per day of propellant with a working mass of 8,000 kg.

C3.2 Secondary Systems

C3.2.1 Engine Specifications and Fuel Determination. For the ascent stage, the four landing rockets will be used again to propel the MARV to 60 km. Once the ascent vehicle reaches this altitude, the four primary system engines will be ejected along with their now empty fuel tanks. In order to rendezvous with the MTV, it will be necessary to use a secondary system engine to boost the ascent vehicle to an altitude of 240 km. An engine similar to Rockwell's RS-41 will be used to provide a thrust of 12 kN.²⁰ Figure C3.2.1 shows a picture of the sample engine.



RS-41

Type: Storable Liquid-Propellant,
Upper Stage
Thrust: 2700 lb
Propellants: Nitrogen Tetroxide/
Monomethylhydrazine
Specific Impulse: 320 sec
Mixture Ratio (O/F): 1.63:1
Chamber Pressure: 191.3 psia
Area Ratio: 150:1 TO 260:1
Weight: 152 lb
Dimensions: 74 in. long/38.5 in. wide

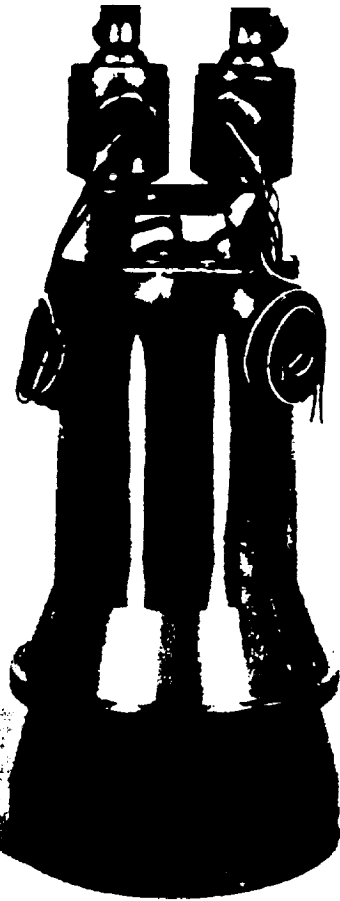
Figure C.3.2.1.1: Rockwell's RS-41 spacecraft engine.²⁰

A storable liquid propellant called nitrogen tetroxide/monomethylhydrazine will be used to fuel the secondary engine with an Isp of 320 seconds. With a length of 1.88 m and a nozzle diameter of 0.98 m, the secondary upperstage engine weighs 68.95 kg.

C3.3 Attitude Control Systems

C3.3.1 Engine Specifications and Fuel

Determination. At 240 km, the attitude control system engines will ignite and maneuver the ascent vehicle into its final position for rendezvous with the MTV. After ignition, the secondary engine and fuel tanks will be ejected from the ascent vehicle. Engines similar to Rockwell's Apollo Attitude Control System will be used for the maneuvering of the ascent vehicle into rendezvous position.²⁰ Each of engine weighs 3.69 kg and provides 413.68N of thrust with a specific impulse of 273 seconds. For additional weight savings, nitrogen tetroxide/monomethylhydrazine propellant will be used for both the secondary and attitude control system engines. The dimensions of each engine are .28 m long with a nozzle diameter of .13 m. Figure 13 provides a picture of the Apollo attitude Control System.



APOLLO ATTITUDE CONTROL SYSTEM

Type: Hypergolic Liquid-Propellant, Ablative

Thrust: 93 lb

Propellants: Nitrogen Tetroxide/
Monomethylhydrazine

Specific Impulse: 273 sec

Mixture Ratio (O/F): 2.1:1

Chamber Pressure: 140 psia

Area Ratio: 9:1

Weight: 8.13 lb

Dimensions: 12 in. tall/5 in. wide

Successful Firings in Space: 100

Figure 3.2.1.2: Rockwell's Apollo Attitude Control System spacecraft engine.

SECTION D: ENVIRONMENTAL CONTROL, LIFE SUPPORT SYSTEMS

Jonathan Hicks
Jeff Galusky
Kouichi Hashimoto
Scott Wenger

Section Design Philosophy

Most people on Earth take for granted the fact that they have air to breathe and water to drink. When planning a trip to another world, one must consider how you will breathe or what will you eat and drink. Providing this type of environmental control is not easy. Environmental controls and life support systems are the most important aspect of any manned mission outside Earth's atmosphere. Without these systems, space exploration would not be possible.

In order to design an effective Controlled Ecological Life Support System (CELSS) for crew atmosphere revitalization, the crew requirements must first be determined. Figure D1.1.1 shows the primary metabolic requirements of oxygen, water and food needed by a crew member under a moderate workload. These requirements can be met through storage, resupply, recovery, or any combination thereof. During extended missions in space storage of food, water and oxygen becomes increasingly weighty and bulky with longer mission durations. On a comparatively long distance mission, such as one to Mars, resupply becomes too costly an alternative. This leaves recovery, or revitalization as the preferred choice for most manned Mars mission life support systems. The primary components of a CELSS are integrated as shown in Figure D1.1.2^{D2}. The following sections will discuss the necessary equipment to effectively support life on Mars.

D1.1 Life Support Systems

D1.1.1 Atmosphere Revitalization System.

Considering the mission duration and distance, neither storage or resupply of the necessary crew atmospheric components is feasible for the UML program. Thus, a functional air revitalization system (ARS) must be capable of providing these metabolic requirements, as well as be able to maintain nitrogen levels, control humidity, handle excess water, and have instrumentation to control and monitor the operation of the integrated systems. All of these functions must be maintained for the entire 6 month stay, with the capability of continued operation in case of a mission duration change.

The ARS designed for this mission is divided into two subsystems, the primary ARS (PARS), and the secondary

ARS (SARS). These 2 systems were designed around the following criteria:

- 1) Overall system reliability
 - Maintainability
 - Allow for down time
- 2) Mass & volume requirements
- 3) Power requirements
- 4) Available technology
- 5) Psychological effects on crew

With the maximization of these criteria, an optimum ARS was designed ^{D3}.

The primary ARS will consist of 2 parts: a Biological Life Support System (BLSS) and a mechanical ARS-8 system. The BLSS uses higher plants, namely lettuce, tomatoes and carrots to provide air & water revitalization, as well as providing a constant supply of fresh food supplements. The characteristics of the BLSS used for the UML mission are as follows:

- Total plant growth area = 12.2 m²
- Total mass = 2180 kg
(including H₂O storage)
- Power required:
 - Illumination = 1.8 kW
 - Equipment = 1.39 kW
 - Total power required = 3.19 kW
- Total storage area = 25.5 m²
- Total ARS contribution = 6% of crew requirement

This system's primary contribution to the CELSS is in food production, water reclamation and crew psychological well-being. The entire BLSS will only contribute 6% of the total air revitalization for the mission. It will thus be necessary to supplement the BLSS with a second system. The ARS-8 system (Figure D1.1.3) is a proposed version of a mechanical ARS assembly designed and tested at the Ames Research Center to provide total air revitalization for extended missions in space. The prototypes of this system have been under development since the early 1970's, and are well tested and reliable. The mechanical component of the ARS-8 is composed of

HUMAN METABOLIC REQUIREMENTS (POUND/MAN-DAY)

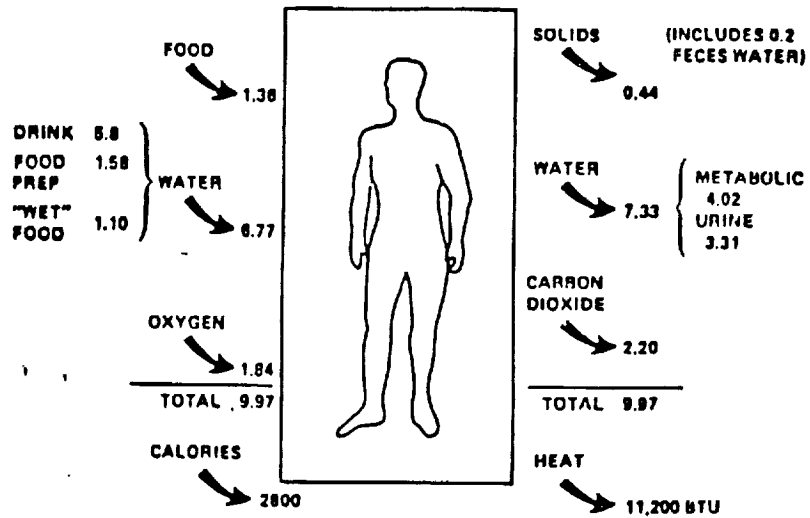


Figure D1.1.1: The primary input/output masses of a person under a moderate workload, requiring approximately 2800 cal/day. The oxidation of food results in the production of water as well as energy.

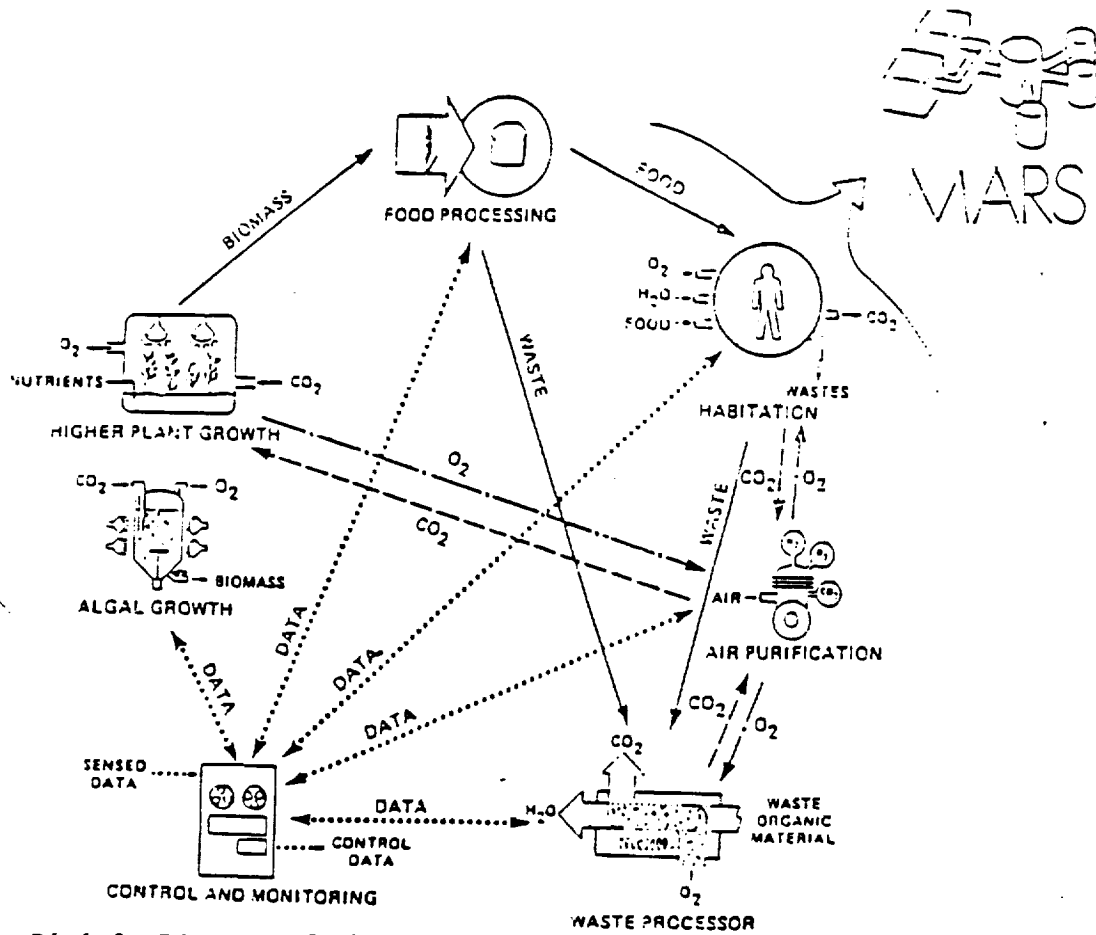


Figure D1.1.2: Diagram of the major components of a CELSS.

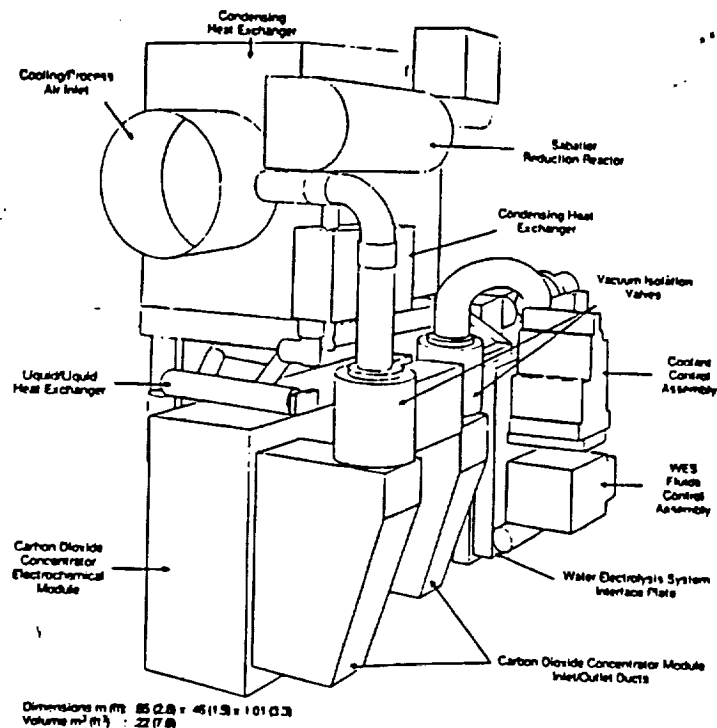


Figure D1.1.3: Air Revitalization System (ARS-8)

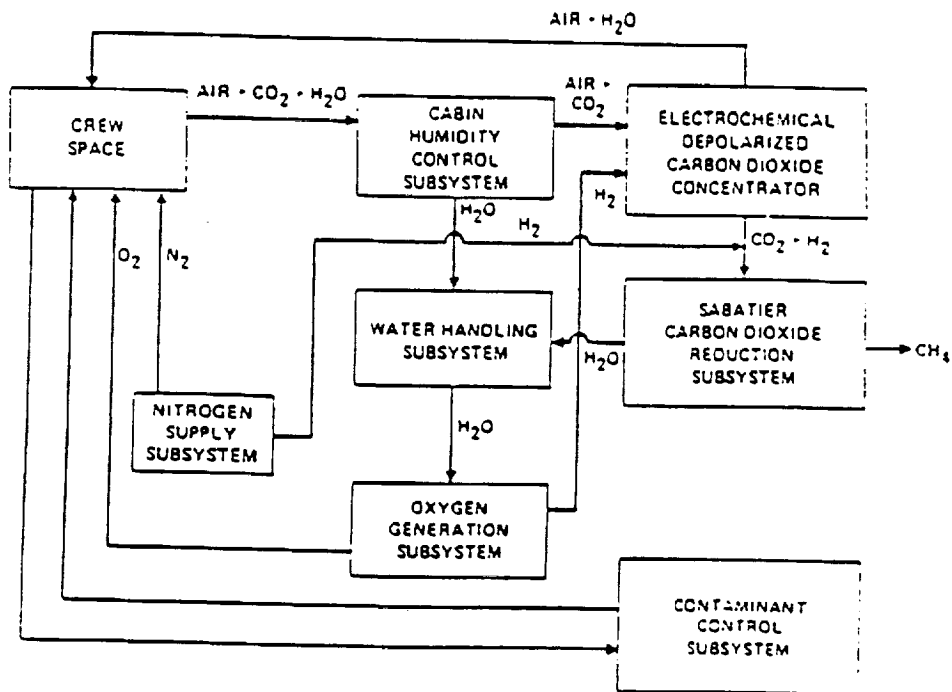


Figure D1.1.4: Air Revitalization System Block Diagram

the integration of the following sub-systems:

- CO₂ concentration
- CO₂ reduction
- Utilizes the Sabatier process
- Oxygen generation
- Nitrogen generation
- Tract containment control
- Cabin humidity control
- Water handling

Figure D1.1.4 illustrates how these sub-systems are integrated to provide the total ARS^{D4}. Control of the ARS is provided by the Control / Monitor Instrumentation or C/MI (see Figure D1.1.5)^{D4}. This system is specifically designed around the ARS-8. The characteristics are as follows:

ARS-8 apparatus

- Total mass = 224.1 kg
- Dimensions = 0.76m X 1.02m X 1.17m
- Total power required = 3.06 kW

C/MI apparatus

- Total mass = 70 kg
- Dimensions = 0.53m X 0.81m X 1.29m
- Total power required = 1.2 kW

Both the BLSS and ARS-8 will be placed in the life support module (LSM). All primary life support will be conducted in this module. The module to be used for this will be the evacuated EVA rover module (4U). The primary LSS will be stored beneath the floor boards of module 4U for the trip to Mars and during the descent phase. The PARS will be set up and put on-line immediately upon removal of the EVA rover, shortly after touch-down. A second system will then be required to supply ARS for the crew before the primary ARS is put on-line.

The SARS has 2 primary functions: as a second system to be activated before the PARS is put on-line, and as a backup system in case of a PARS failure. The SARS is comprised of 2 individual ARS-8 and C/MI systems. One ARS-8 is to be placed in the 4M module, and 2M module. Each system is individually capable of providing the crew with total air revitalization, or can be used as spare parts to maintain the PARS.

D1.1.2 Food Production and Storage. The BLSS will provide 3% of the total crew diet in the form of supplemental salad plants. Lettuce, tomatoes, and carrots will be grown and harvested in the LSM to provide the

crew with a daily supplementary salad. The bio-mass harvest rate of the BLSS is as follows:

- Lettuce = 26 g/person/day
- Tomatoes = 138.9 g/person/day
- Carrots = 23.9 g/person/day

^{D1}. This system may seem unnecessary when considering the somewhat minimal contributions it provides in ARS and food production. However, the water recovery benefit of providing 19% of the total water reclamation is a significant contribution. But most importantly, there is the added benefit of the crew's psychological well being. After being confined to such a small area for a long period of time, and in a foreign world, it will be a much needed boost to moral to preoccupy the crew with the harvest and preparation of the days salad crop. The fresh salad plants will also provide a break from the monotony of dehydrated rations. The remaining crew food supply must be provided by on-board food storage systems. Thus, the total food storage for 6 month stay, with 3 months of emergency reserves comes to 3409 kg, and will occupy an area of 12.9 m³.

D1.2 Extravehicular Activity. Extravehicular Activity (EVA) is an essential part of any manned space mission. EVA's enable crew members on any mission to conduct experiments, gather samples, assemble, maintain, repair, inspect, and service their habitat and user systems. Three major components of a manned mission to Mars EVA are: Airlocks, Decompression, and EVA suits (better known as "Space Suits"). Without any one of these components EVA's, would be impossible.

D1.2.1 Airlocks. First, an airlock allows an astronaut to safely transfer from the habitat to outer habitat environments and vice versa. The airlock may also be utilized for pressure sensitive experiments since the airlocks pressure may be varied.

There will be 2 identical airlocks installed for the UML. The area of the airlock was chosen as 8.23 m². This size airlock was chosen because it will be able to accommodate 2 EVA suits with recharging units (125 kg), 2 crew members, and a CO₂ shower. The CO₂ will be used to blow off debris gathered on the suit during an EVA. This enables the crew to maintain a zero contamination of the habitat modules. The airlock will also serve as a hyperbaric chamber. This feature is for the treatment of decompression sickness (to be discussed shortly).

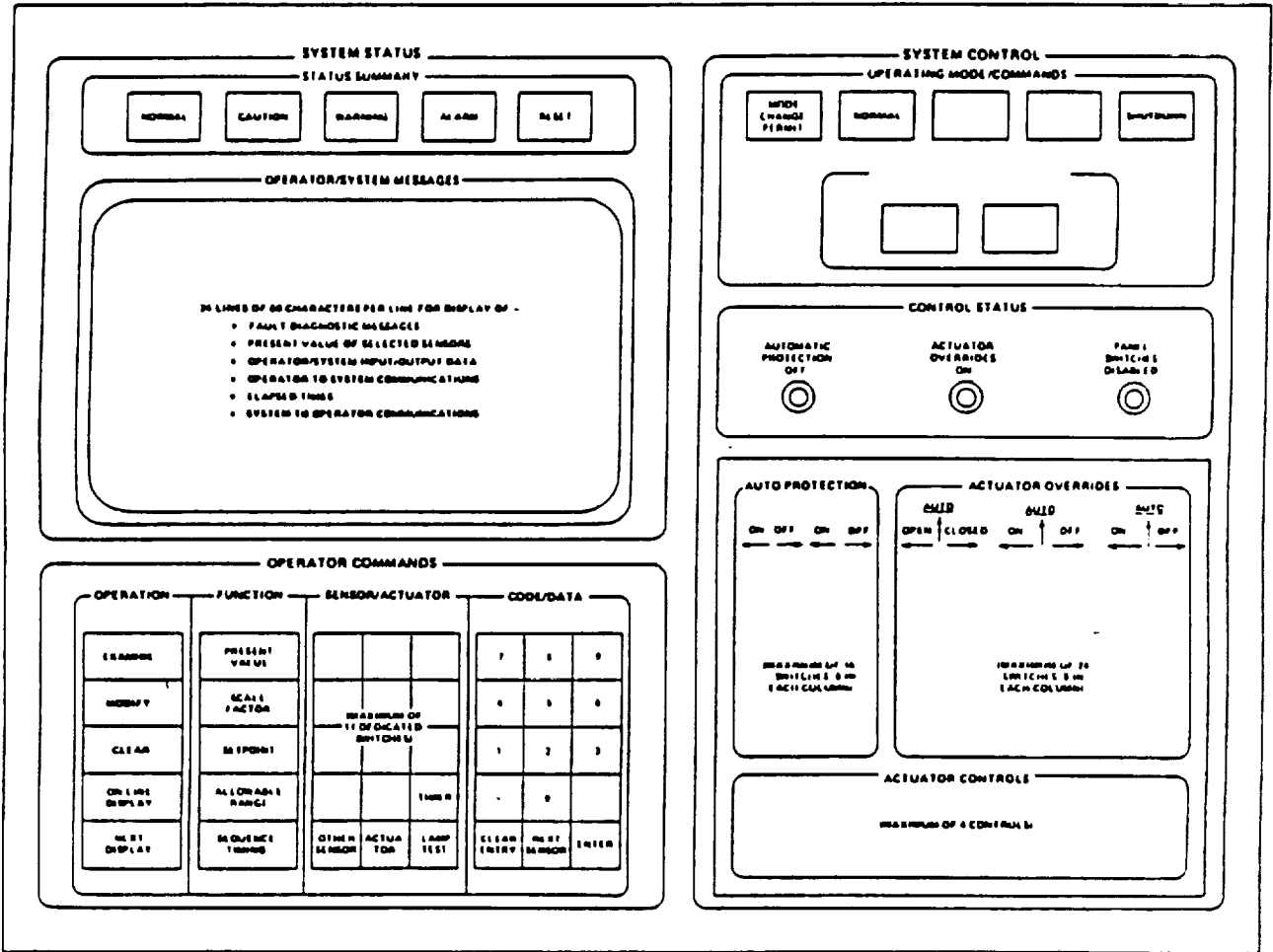


Figure D1.1.5:C/MI Control Panel

One of the airlocks will be installed on the manned lander while the second airlock will be located on the unmanned lander. This configuration was chosen because after landing on Mars, the crew members must be able to link up with the unmanned lander. This means the crew members must travel outside. The second airlock was also chosen to be in a separate module in order to distribute the weights more evenly.

D1.2.2 Decompression. Decompression sickness is an extremely dangerous injury. If a pressure change on the body occurs too rapidly, nitrogen in the blood a bubble. This bubble can cause extreme pain and even death. The airlock provides a safe means of controlling the pressure change and therefore, preventing decompression sickness.

Decompression becomes a major concern for the crew of the UML. Considering the length and cost of this mission, many EVA's will be conducted in order to maximize the information that can be gathered. Unlike the Space Shuttles maximum of three EVA's per flight, the members on the Mars mission will be required to perform as many as one EVA per day if not more. If these EVA's were conducted using Shuttle protocol, this would mean long, boring decompression times and a greater chance for error and accidents. This, therefore, called for improved safety in performing EVA's.

Decompression times are primarily based upon how long a person has been subject to a particular pressure, what is the partial pressure of oxygen at that pressure, and what is the required pressure change. After evaluating the Space Shuttle's and Apollo's procedures on EVA's, it was decided to decrease the pressure difference between the habitat modules and suit environments. The Shuttle requires approximately a 13 hour decompression time. This is because the Shuttle maintains a 1 atm environment at 21% O₂ and the EVA suit are at 0.27 atm at 98% O₂. Table D1.2.1 shows the minimum time required for decompression for different cabin pressures, suit pressures, and oxygen percentages. Note that all of the times in table D1.2.1 are based on living in that particular environment for at least 24 hours and the suits O₂ percentage is about 98%. Although, when the cabin pressure is decreased, the crew may feel as though they are living high in the mountains (ie. the oxygen partial pressure drops). This is corrected by increasing the percentage of oxygen in the cabin environment from a typical 21.0% to 40.0%. In doing so, decompression times decrease further. The final time needed for decompression will be approximately 30 minutes. This is roughly the time it takes to don and perform a safety check on an EVA suit.

TABLE D1.2.1: Various decompression times for different cabin and suit pressures

Cabin press (atm)	% O ₂	Suit press (atm)	MIN. Decomp Time
1	21	0.65	none
0.87	24	0.54	none
0.75	25	0.46	1 hr
0.69	28	0.41	1 hr
0.69	40	0.34	30 min

D1.2.3 EVA Suits. The EVA suit is the piece of equipment that allows a person to survive in a hostile space environment. The suit must also allow the crew member to perform any outside activity. In doing so, several considerations must be investigated.

TABLE D1.2.2: This table shows the considerations for Mars EVA life support

- 1) **Thermal Control:** The EVA suit must have an active temperature control system to provide protection from the bitter cold on Mars.
- 2) **Water Management:** Some of the EVA's performed on Mars will take up to 7 hours, therefore, the suit must provide a drinking and urination collection system.
- 3) **Nutrition:** Again, the hours spent in an EVA suit, some kind of food system must be provided.
- 4) **Maintenance:** Suits must be kept operational. They need to be durable, easy to clean, and easily repaired.
- 5) **Monitoring:** EVA time is limited to the amount of O₂ one can carry. So, monitoring of O₂ levels is essential as well as CO₂ levels and heart rate.
- 6) **Mechanical Mobility:** The person in the suit must be able to perform useful work without injury. The suit must fit comfortably, have good low-effort movement, and must be easy to walk in on Mars.

Due to the great expense of the mission to Mars, EVA is one place to save money. Today's suits are extremely reliable and have an expected life of 15 years. In order to save money, these suits can be used for the UML. This saves on the research and development costs associated with designing a new suit. However, slight modifications will be made in order to meet the criteria shown in table D1.2.2.

One such modification will be to remove the display panel from the chest and incorporate the readouts in a heads-up-display (HUD). This allows the person in the suit to read the information faster and more accurately, instead of relying on a mirror mounted on their wrist. The best feature in the HUD will be its compass navigation system which is fixed to the UML's ascent vehicle. This will keep the EVA member oriented to "home" via directional and distance indicators.

Temperature will be controlled by a thermal control unit (TCU). A liquid filled garment will be donned by the crew member and connected to the TCU. The TCU captures heat generated from the body as well as heat from mechanical parts in the EVA pack. The TCU manages the temperature and provides heating and cooling when needed.

Water management and nutrition are provided by a small drink bag located on the right breast of the EVA suit. This drink bag provides approximately 12 oz. of high calorie liquid diet. A urine collection system (one for males and one for females) will provide up to 1000 ml of urine collection and is connected to the liquid filled garment.

Maintenance starts with a CO₂ shower after every EVA. This shower blows off any Martian debris which the suit may have collected while outside. Regularly scheduled inspections and maintenance will properly maintain the suits in working condition. Twelve suits will provide each of the 8 crew members with one suit each. The remaining 4 suits will be used in case of an emergency and/or spare parts. Each suit is about 125 kg and is approximately 2m X 1m X 1m.

D1.3 Thermal & Safety Management. The thermal controls and safety management systems are two important sub-systems of the mission to Mars. The thermal control system (TCS) is responsible for heat collection & rejection or distribution. The main components of the safety management sub-system are fire detection and suppression.

The TCS consists of a 2 phase fluid loop. Pressurized loops gather excess heat from the modules. The heat is then transferred into a pipe loop. The fluid in the pipe is ammonia. The reason ammonia was chosen over water,

CO₂, or CFC's was because of the good thermal characteristics of ammonia and, because a hitch hiker experiment using this type of TCS was performed on a 1992 Shuttle flight. The outcome of this experiment showed promising results. The ammonia pipes are connected into a TCS regulator. This regulator determines whether to reject or distribute the heat or distribute air conditioning in order to maintain habitat temperatures within $\pm 8^\circ$ C. If the heat collected is rejected, it is done so through a radiator mounted outside of the modules. The fire detection and suppression sub-systems consists of controlling a situation before it gets out of hand. The detection of fire is performed in 3 ways: visual detection, smoke detection, and heat detection.

Once a fire has been detected, an alarm will sound and that area will be sealed. Fire extinguishers are strategically placed and are readily available. Self contained oxygen tanks (SCOT) will also be easily accessible. Once the compartment is sealed CO₂ from the Martian atmosphere is pumped into the compartment and extinguishes the fire. CO₂ was chosen because of its effectiveness against fires and it requires no storage.

D2. Physiological/Psychological

Because of the nature of a mission to Mars, the astronauts will be in a micro-gravity environment, as well as a near zero gravity environment during flight, for a long time. A mission to Mars is a quantum leap for the space program. Micro-gravity effects on human beings is a highly unresearched area. Long duration space flight is something that is very new. Because of this, little is known about the effects on humans and the adjustments they will have to make on the trip back to earth. Therefore, the physiological and psychological needs of the astronauts place an extremely important role in planning a Mars mission.

D2.1 Physiological. In terms of micro-gravity effects on humans, a number of problems were identified. These effects include muscle wasting, bone brittleness, body fluid shifts, and cardio-pulmonary problems. The cardio-pulmonary problems include heart shrinkage, blood pressure fluctuation, and blood cell changes. A minor problem is motion sickness which affects only some astronauts^{D11}. It has not been shown that after a long duration of being in a microgravity environment that the individual cannot readjust to earth conditions without serious physiological consequences. The major results of research

that has been done indicates that the way to help counteract the effects of microgravity is to have a rigid exercise regime. Table D2.1.1 shows durations of exercises for an exercise regime to simulate 1g earth conditions^{D12}. Throughout the duration of the mission the crew members will adhere regularly to a exercise program. Additionally, astronauts chosen for this Mars mission will be in top physical condition.

To counteract the effects of fluids pooling in upper body, a Lower Body Negative Pressure (LBNP) device can be used. The LBNP is worn around the lower body. The negative pressure created by the device causes fluid to go to the lower body, which is normal for earth.

A dedicated mini-gym will be part of the number 1 manned module. The gym will include a treadmill, a rowing ergometer, and weight machines. Additionally, a "Penguin suit", as used by Russian cosmonauts can be

Table D2.1.1: Exercise Regime.

Physical Parameter	Walk (min/day)	Jog (min/day)
leg strength	15-20	5
leg endurance	25-30	15-20
cardio-respir. endurance	30	15-20
bone strength	1-2 hrs	30-90
coordination	10	5

used. This suit is worn on the entire body and causes a load to be exerted in the opposite direction that the astronaut moves a body part. This works the muscle by creating a load on the muscle. Exercise not only serves as physiological need, but as a form of recreation.

Another aspect that relates to the physical need of the astronauts is routine and emergency medical treatment. A Health Maintenance Facility (HMF) was incorporated into the UML and is designed to handle emergencies that can be treated by a "Earth" emergency room, as well as possible Mars emergencies^{D13}. Medication and equipment to treat routine sickness will be included as well. Equipment that will be standard in the HMF to handle

emergencies can be found in Table D2.2^{D14}.

In addition to the equipment listed, miscellaneous materials were also included: bandages, tape, antibiotics & medication, thermometers, gloves, stethoscope, Q-tips, splints, and surgical utensils. The list included is not intended to be an all encompassing list but only an overview of the nature of the equipment that will be included.

D2.2 Psychological. Psychological needs are extremely important for a mission that is the length of the Mars mission. Windows have been included in the modules^{D16}. The combined size of the modules is only about the size of a mobile home, not very large. The windows give the astronauts the chance to reflect on home life and also reduce "cabin fever"^{D13}. Exercise will not only serve as a physiological need but also as a psychological need as well, a way for the astronauts to pass the time. In addition, books, CD's and television will be included in the habitat modules for the astronauts.

**Table D2.1.2
HMF Equipment**

- Aspirator/Suction Machine
- Defibrillator and monitor
- Electrocardiograph
- Surgical/examination lamp
- ECG monitor
- Autoclave
- Stethoscope, ultrasonic
- Operating table
- Telemetry receiver, radio
- tourniquet, automatic
- ventilator, pressure
- blood warmer
- heart-lung bypass unit
- dialysis unit
- hypothermia unit
- anesthesia unit
- anesthesia ventilator
- IV equipment
- BP cuff

The goal in designing the habitat modules was to make life in space similar to earth as much as possible. This reduces the anxiety of space. A number of recommendations have been incorporated into the

design of the hab modules to do this. Each astronaut has a private sleeping chamber. The library is an additional area where an astronaut can obtain privacy if desired. A wardroom/galley was designed to accommodate all crew members for meetings and meals. Hygiene facilities for oral hygiene and bathing have been included. Also, a shower, bathroom and a laundering area were incorporated into the design (Figure D.2.1),^{D15 D22}

D3. POWER SYSTEMS

The availability of electrical power has been partially responsible for the mission success to Mars. The power systems are designed to supply the sufficient amount of electrical power for all of the system equipments during the entire mission. These include power requirements for communications, data gathering and handling, vehicle attitude control, guidance and navigation control, life support equipments, science equipments, and many other areas.

The objectives of the design are to estimate the power requirements for each stage of the mission, to select the appropriate power sources, and to determine the mass and the size of the system. All the designs of power system are based on the current available technology. The power supply before the release of the Universal Martian Lander from the Martian Transfer Vehicle (MTV) and after the rendezvous will be provided by the MTV power system, which is not included in the design.

D.3.1 Universal Martian Lander

D.3.1.1 Power Requirements. Electrical power requirements during the descent stage are driven by the crew size and the operation time of the descent stage power system. The crew size function was estimated to be 2.0 kW per astronaut for the life support system. This estimation was based on the "rule of thumb" for the primary design of the power system. The electrical power for the life support system is allocated to all of the components of the life support systems, such as illumination inside a module, oxygen/water supply and revitalization system, and the thermal controlling system. During the descent, the attitude control, flight control, and the thermal control systems are to be operated and these systems require some amount of electrical power. It was estimated that 1.8 kW of electrical power was required for the avionics. This was based on the estimation of the lunar mission^{D23}.

The operation time of the descent stage power system consists of two phases. The research from the Guidance, Navigation and Control Division (GNC) showed that it

takes approximately 2 hours from the lander release at the 250 km Martian orbit to the touch-down on the Martian surface. The GNC research also showed that three days of Martian surface stay was required before the full operation of the Martian-base power system as the safety reasons. Therefore, the descent stage power system must be designed to fulfill the power requirements for both of the operation time phases. Table D.3.1.1. shows the power requirements for the 8-person crew systems during the descent stage.

Table D.3.1.1.1: Power Requirements for Descent Stage

Component	Power Requirement
Life Support	2.0 kW / Astronaut
Avionics	1.8 kW
Total	17.8 kW

D.3.1.2 Power Source Selection. The selection of the power source generally depends on the amount of power requirement and the mission duration. Fig.D.3.1.2. shows the map of the selected weight-optimized power system applications dating from the early days of the space program through the present space station^{D24}.

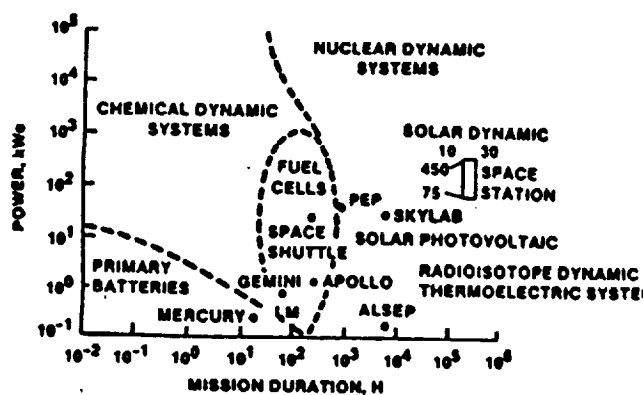


Figure D3.1.2.1: Selected weight-optimized power system configuration.^{D24}

The power sources for the descent stage were selected as Ag-Zn (Silver-Zinc) batteries and H₂-O₂ (Hydrogen-Oxygen) fuel cells. Ag-Zn batteries were

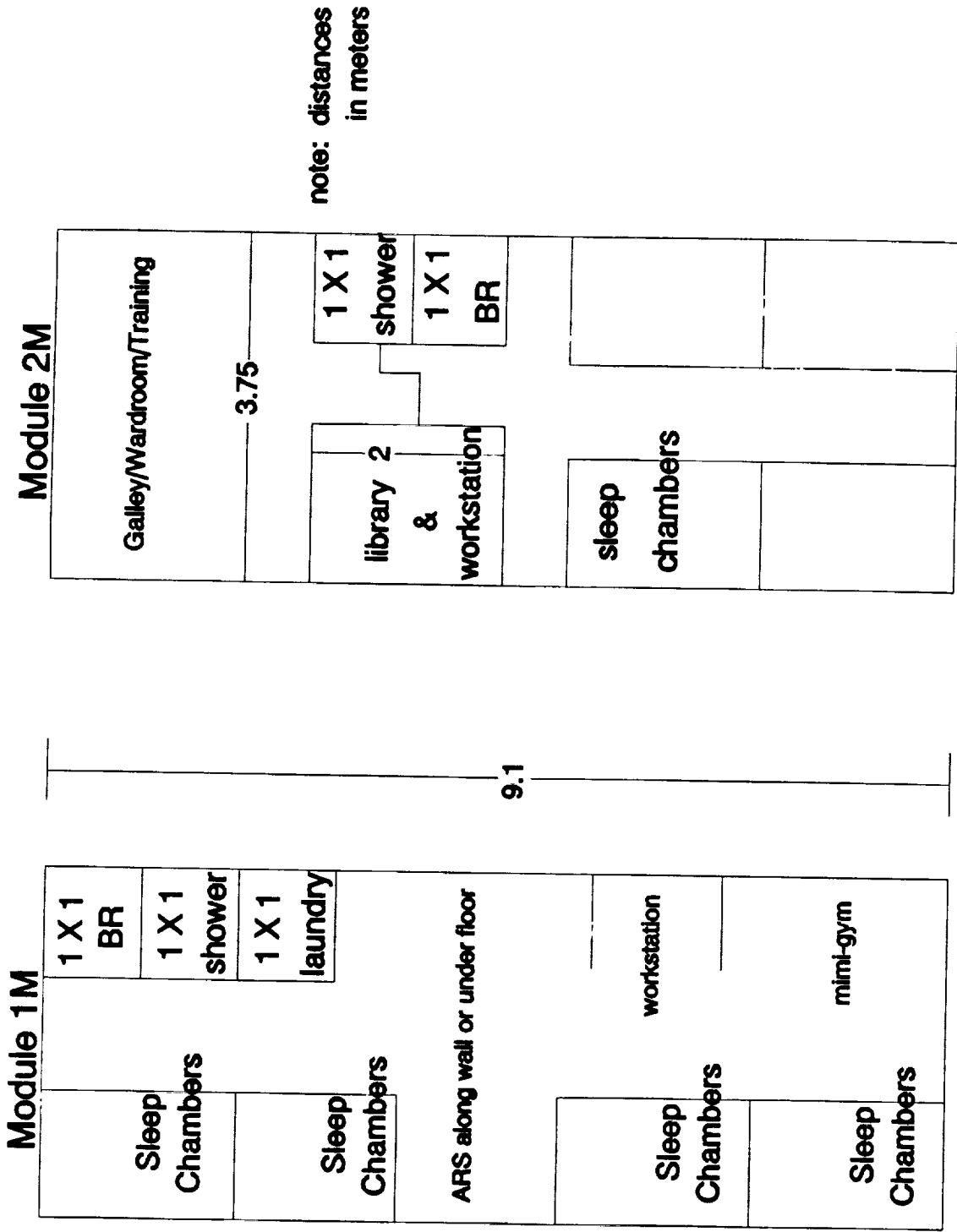


Figure D2.2.1: Layout of Manned Modules, 1M and 2M.

designed as the power source for the 2 hours flight from the 250 km orbit to the Martian surface, while H₂-O₂ fuel cells were designed as the power source for the three-day Martian surface stay.

Ag-Zn batteries are known as the reliable, quick reaction, and low maintenance required power source for short periods. Ag-Zn batteries have a relatively high energy density (175 W-hr/kg) as compared to other batteries, and Ag-Zn batteries are known as the primary power source of the APOLLO Lunar Lander ^{D25} ^{D26}.

For the three-day Martian surface stay, the fuel cell was selected as the power source because the battery system was not appropriate to use for more than a day long (refer to Fig.D.3.1.2.1) Hydrogen-oxygen fuel cells were selected for the power source during the three-day surface stay. H₂-O₂ fuel cells are a reliable and environmentally clean power source. Environmentally cleanness indicates that the inputs of H₂-O₂ fuel cell are either gaseous or liquid hydrogen and oxygen and the outputs are electricity, heat, and pure water. H₂-O₂ fuel cell power source has been used as the primary power system in Space Shuttle Orbiter. The cryogenic reactant storage tanks are used for Space Shuttle power system, and they were selected for the design of the Martian lander reactant storage system. The reason for the selection is that the cryogenic storage system reduce the overall system mass as compared to the pressurized gas storage system^{D27}. Thus, Ag-Zn batteries are used for 2 hours descent and the H₂-O₂ fuel cell with cryogenic reactant storage system are used for the three-day Martian surface stay.

D.3.1.3 Power System Mass Estimation. The total mass of the descent stage power system is the sum of the Ag-Zn batteries and the H₂-O₂ fuel cells with cryogenic reactant storage systems. The characteristics of the Ag-Zn batteries are based on the Ag-Zn batteries which are used in the APOLLO missions^{D25}, while the fuel cell characteristics are based on the fuel cell system used in the current Space Shuttle Orbiter. The calculations of system mass are performed in Appendix D.3-(i) where the method of calculations are referred as Pg 405 of reference D26.

Results of calculation show that 5 Ag-Zn batteries are required to provide 17.8 kW of power requirement for 2 hours descent period and the battery system weighs 290.6 kg. Three fuel cells with three sets of cryogenic reactant storage tanks are required to provide 17.8 kW of power requirement for three-day surface stay and the system weighs 1038.7 kg. The total weight of the descent stage power system is 1329.3 kg. The descent stage power system was illustrated in Fig.D.3.1.3.1.

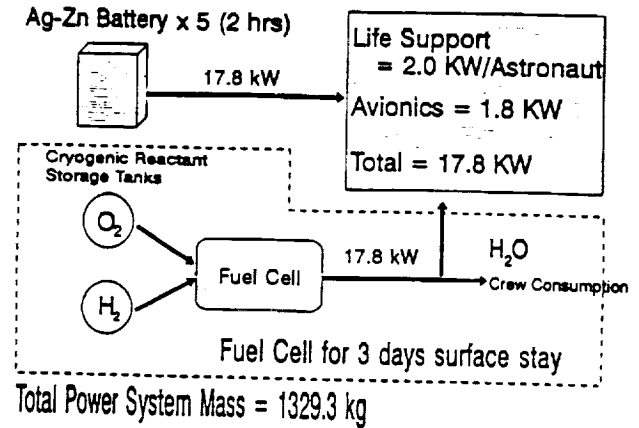


Figure 3.1.3.1: Power distribution schematic for descent stage.

D.3.2 Habitat Modules

D.3.2.1 Power Requirements. Electrical power requirements for habitat modules during Martian surface stay are driven by the length of the mission, the crew size, type and size of the communication and data handling system, the total number of the scientific experiments, and the requirements for the fuel production. The length of the time period for the Martian surface stay is 6 months. Thus, the power system should be designed to supply a continuous amount of electrical power for the 6 month duration.

The crew size function is estimated to be 2.0 kW per astronaut for the life support system. The power requirement for the life support system in the habitat modules is considered to be unchanged from the power requirement for the life support system during the descent stage. This amount of electrical power is required to keep operating the life support systems for both the daytime and night time period.

A continuous 2.2kW of electrical power is required to operate Communicate and Data Management System (CDMS). The power requirement is based on the estimation of the Lunar mission^{D29}. CDMS deals with the acquisition, processing and storing of system controlling data and the experimental data. CDMS also deals with the subsystem controlling and the

communication system controlling. This amount of electrical power is required to operate CDMS for both daytime and night time.

For the Martian daytime, the 5.0kW of electrical power is allocated for the scientific experiments. During the Martian night, the 1.0kW of electrical power is allocated for the scientific experiments. Increasing the power allocation for daytime-experiments enables astronauts to perform a variety scientific experiments during the Martian surface stay. In addition to the scientific experiments' power supply, a continuous 0.5kW of electrical power for both daytime and night time is allocated for a rover. This information is based on the estimation of the Lunar mission^{D23}.

For the Martian daytime period, the fuel production is taken place. The 31.2kW of electrical power is required for the conversion of the carbon-dioxide to the carbon-monoxide and the oxygen during the fuel production process^{D31}. The calculation of the power requirement for the propellant production is performed in Appendix D.3-(iii).

The power requirements for the habitat module during the Martian surface stay are estimated as 54.9 kW continuously during daytime, and 19.7 kW continuously during night. Table D.3.2.1.1 shows the power requirements for the habitat module of the eight person crew systems.

Table D.3.2.1.1: Power Requirements for Habitat Module

Component	Power Requirement
Life Support	2.0 kW / Astronaut
CDMS	2.2 kW
Science	5.0 kW
Rover	0.5 kW
Fuel Production	31.2 kW
Total	54.9 kW

Night Time:

Component	Power Requirement
Life Support	2.0 kW / Astronaut
CDMS	2.2 kW
Science	1.0 kW
Rover	0.5 kW
Total	19.7 kW

D.3.2.2 Power Source Selection. As outlined by the NASA Office of Exploration, several Mars mission scenarios are under consideration which will require light weight, compact power sources. For one of these scenarios, which outlined an early manned expedition to Mars, a photovoltaic array with energy storage was identified as the candidate power system. The planar array concept is selected for the Mars mission rather than the concentrator type of array. Reasons are due to the estimates of difficulty and risk in deployment of concentrator type of array, manufacturing cost, the cost of launching the higher mass of a concentrator array, and the operating costs (including replacement costs) appeared to substantially outweigh the saving in solar cell cost^{D24}. Notice that the solar array system can be operated only during the time period when the sun is available. For this reason, an energy storage subsystem is required to provide sufficient amount of electrical power to the habitat modules during the Martian night.

The selection of a solar cell for the solar array involves the consideration of cell material, physical size, contact configuration, cover glass type, and the array assembly method. Due to its light weight and high efficiency, CLEFT GaAs/CuInSe₂ Tandem solar array concept was studied for the daytime power production^{D28}. Cleavage of Lateral Epitaxial Film for Transfer (CLEFT) Gallium Arsenide (GaAs) with Copper Indium Selenide (CuInSe₂) array shows the highest efficiency (23.1%) ever-reported for thin-film photovoltaic cell. The 4 cm² cell with a 1-mil thick cover glass weighed 258 mg even without optimal substrate trimming. This provides a cell-cover glass specific power of 442 W/kg and offers a potential for specific power up to 750 W/kg. Technical research shows that it is more resistant to damage caused by the space radiation^{D28}. The schematic of the GaAs/CuInSe₂ Tandem cells is shown in Fig.D.3.2.2.1.

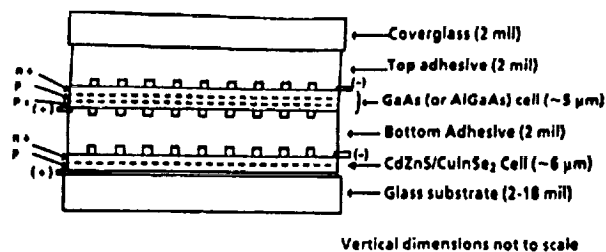


Figure 3.2.2.1: Schematic of Tandem Cell Structure.^{D28}

The solar array can be either erected or mechanically deployed. It is considered that the added cost of deployment mechanics may be small compared to the value of astronaut EVA time. Also, it is the fact that deployable type solar arrays are available with current technology, such as OAST-1 solar array experiment aboard a space shuttle flight [Ref. D.3-2]. For these reasons, deployable type solar array is selected rather than an erectable (rigid) array configurations for the design of the Martian daytime power source.

In the interest of lowering the total array deployment costs, the same type of mast as used for the OAST-1 wing was chosen. Although the OAST-1 wing was designed for the zero-gravity environment, the application for the Mars mission (3 Earth gravity environment) is still considered to be possible. The mast consists of three coilable, S-glass/epoxy, continuous longerons stabilized by glass/epoxy battens and steel diagonals. The mast is stowed in and deployed from a cylindrical aluminum canister in which the longerons are elastically coiled^{D24}. The mast and its canister are illustrated in Fig.D.3.2.2.2.

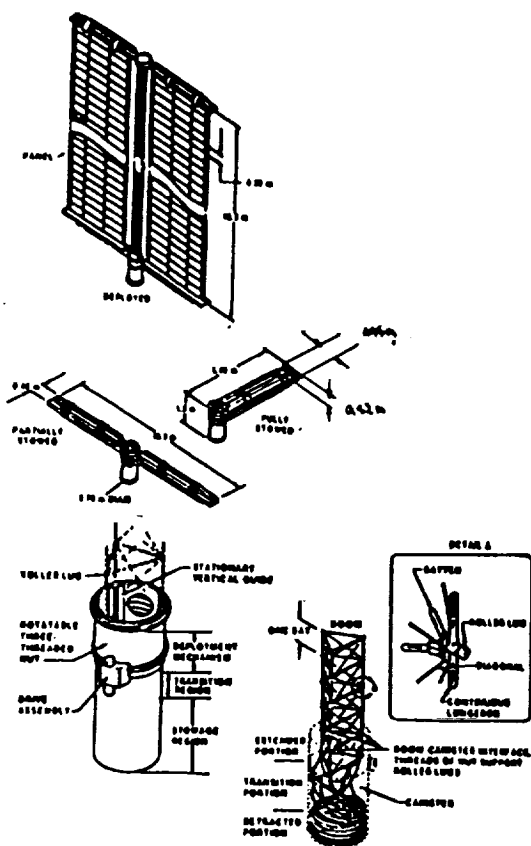


Figure D3.2.2.2: Space Station Freedom solar array wings and solar array deployment assembly. ^{D24}

A primary candidate for the energy storage subsystem is the Hydrogen-Oxygen Regenerative Fuel Cell (RFC) which exhibits the highest energy density of all the non-nuclear systems for storage periods exceeding 2 hours. For this reason, the H₂-O₂ RFC subsystem is selected for the energy storage system, and it is utilized as the power source during Martian night. The primary components of the conventional RFC subsystem include a fuel cell unit, an electrolysis unit, reactants and reactant storage tankage. The cryogenic reactant storage tank are selected for the Mars application in order to reduce the overall system mass as compared to the pressurized gas storage systems^{D27}. The RFC system with cryogenic storage system is illustrated in Fig.D.3.2.2.3.

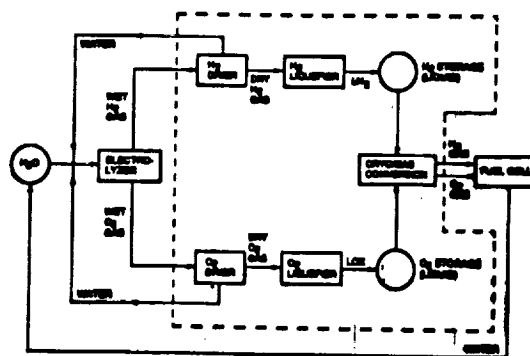


Figure D3.2.2.3: Regenerative fuel cell system (reactants stored as cryogenic fluids). ^{D27}

The CLEFT GaAs/CuInSe₂ Tandem with deployable solar array system was selected for the power source during Martian daytime. The hydrogen-oxygen regenerative fuel cell with cryogenic reactant storage system was selected for the power source for the nighttime power production.

D.3.2.3 Power System Mass Estimation. The total mass of the power systems for habitat modules on the Martian surface is the sum of the solar array system, the H₂-O₂ RFC system, and the power management subsystems.

The power produced by the solar array needs to fulfil both the daytime power requirements and the excess power required for storage for use at night. Using a code for the fuel cell / electrolyzer subsystem which was developed at the NASA Lewis Research

Center, the power input requirement for the RFC system is estimated as 47.4 kW^{D27}. This results in the 102.3 kW of electrical power requirement from the solar array to fulfill both 54.9 kW of daytime power requirement and the 47.4 kW for RFC systems. The calculations of the array size and the array weight are performed in Appendix D.3-(ii) where the methods of calculations are referred as Pg 419-420 of reference D26. The characteristics of GaAs/CuInSe₂ Tandem cells are provided by reference D28, and the photovoltaic design values at Mars are provided by reference D29 and reference D30.

Results show that 2361.2 m² of CLEFTGaAs/CuInSe₂ Tandem solar array is required to produce sufficient amount of electrical power on the Martian surface. This array surface area can be obtained by using 19 wings of the OAST-1 deployable solar array. The solar array weight is calculated as 924.7 kg. The mass of the H₂-O₂ RFC subsystem is provided by the code from NASA Lewis Research Center. The results for the 20 kW system show that the RFC system mass is determined as 4658 kg including the fuel cell unit, electrolyzer unit, cryogenic storage system with reactants. The mass of each component is shown in Table D.3.2.3.1.

Table D3.2.3.1: RFC Subsystem with cryogenic Reactant Storage system Mass Breakdown for 20 kW system.

Component	Mass (kg)
Fuel cell stack	252
Fuel cell stack Ancillaries	23
Electrolyzer stack	323
Electrolyzer stack Ancillaries	112
FC radiator system	106
H ₂ tank	449
O ₂ tank	258
H ₂ O tank	47
Reactants: H ₂ O	2439
H ₂ residual	14
O ₂ residual	108
Gaseous drying equipment	9
Drying radiator	17
H ₂ liquefaction unit	141
H ₂ liquefaction radiator system	241
O ₂ liquefaction unit	45
O ₂ liquefaction radiator system	74
Total for 20 kW system	4658

The mass of Power Management And Distribution (PMAD) subsystem is estimated as 1322 kg^{D30}. The mass of each component is shown in Table D.3.2.3.2.

Table D.3.2.3.2: PMAD Subsystems Mass Breakdown

Components	Mass (kg)
Ribbon cable edge connectors for array modules	80
Inverter modules for array modules	560
3 Inverter units for fuel cell stack	60
3 Down converter/rectifier units for electrolyzer stacks	45
3 Pallets for RFC units	177
Main bus (transmission line)	300
User interface	100
Total PMAD mass	1322

For the 6 months Martian surface stay, CLEFT GaAs/CuInSe₂ Tandem solar photovoltaic array was selected as the power source during Martian daytime. During Martian night, the hydrogen-oxygen regenerative fuel cells with cryogenic reactant storage system were selected as the power source. The total power system mass is 6904.7 kg. The power system for the habitat modules during the Martian surface stay was illustrated in Fig.D.3.2.3.1.

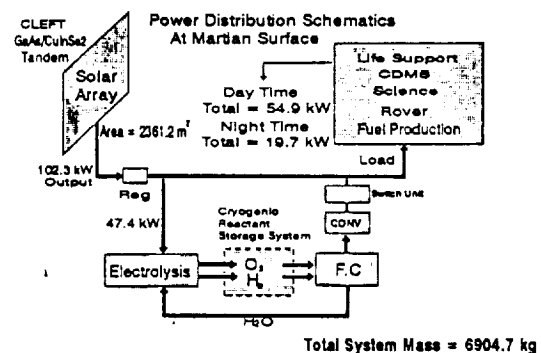


Figure D3.2.3.1: Power distributions schematic at Martian surface.

D.3.3 Manned Ascent Vehicle

D.3.3.1 Power Requirements. Electrical power requirements during the ascent stage are considered to be the same as the power requirements during the descent stage. 2.0 kW/astronaut of electrical power is estimated for the life support system and 1.8 kW of electrical power is required for avionics. Again, the estimation of power requirements is based on the Lunar mission^{D23}. The difference from the descent stage is the operation time of the power system. The research from the Guidance, Navigation, and Control Division shows that the total travel time from the Martian surface to the rendezvous at 250 km Martian orbit is estimated as 5 hours. This includes the safety factor of "3 extra turn orbit" in the case of the unsuccessful rendezvous. Thus, for the 8-person crew system, the power requirement during the ascent stage is estimated as the continuous 17.8 kW of electrical power for 5 hours. Table D.3.3.1.1 shows the power requirements during the ascent stage.

Table D.3.3.1.1: Power Requirement for Ascent Stage.

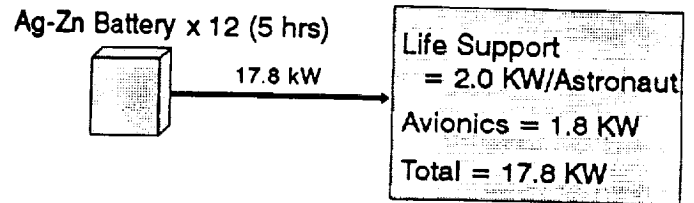
Component	Power Requirement
Life Support	2.0 kW / Astronaut
Avionics	1.8 kW
Total	17.8 kW

D.3.3.2 Power Source Selection. The power source for the ascent stage is selected as Ag-Zn batteries due to the reliability, quick reaction, and high energy density. The battery system is considered to be still appropriate for 5 hours use from the map of the selected weight-optimized power system applications shown in Fig.D.3.1.2.1.

D.3.3.3 Power System Mass Estimation. The total mass of the ascent stage power system depends on the total number of the Ag-Zn battery used. The characteristics of the Ag-Zn batteries for ascent are assumed to be the same as those of the descent batteries. The calculations of system mass are performed in Appendix D.3-(i).

Results show that 12 Ag-Zn batteries are required to provide 17.8kW of power requirement for 5 hours ascent stage. The total weight of the ascent stage power system is determined as 726.5 kg. The ascent stage power system was illustrated in Fig.D3.3.3.1.

Power Distribution Schematics Ascent Stage



Total Power System Mass = 726.5 kg

Figure 3.3.3.1: Power distribution schematic for Ascent stage.

SECTION E: ECONOMIC MANAGEMENT

James Simpson

Section Design Philosophy

The task of the economic management group was to analyze the cost incurred by developing and producing 2 flight ready UMLs. The cost of each subsystem in the UML was calculated in order to obtain a more accurate description of the overall cost for the UML project. There were four different design phases used to consider the cost of each subsystem of the UML. The first design phase was the research undertaken in order to supply the appropriate technology for a given hardware requirement. The second design phase was the development of that technology into flight hardware to fulfill a mission requirement. Third, the testing of that hardware would be needed verify that the equipment functioned properly under flight conditions. Finally the last phase would produce the flight-rated hardware that will be used during the mission.

The economic management of the UML is as important aspect of the entire design. The importance of economic issues is most prevalent in past NASA programs. The Apollo program had several flights deleted from its manifest due to the budget constraints. The Space Station Freedom program is even today under severe scrutiny from the federal government to maintain cost control. Thus it is obvious that the cost considerations can play a major role in the overall operation of a NASA program. Cost considerations can help in decision making tasks. Is the project feasible when overall cost is considered? The overall cost of a program may justify its cancellation from U.S. Space Policy. When can this project be completed? Cost is a major factor when trying to develop, build, test, and then operate hardware and software. The project may be completed sooner if more money were allocated to fund more employees and more equipment.

The UML hardware development and production cost will be \$8.70 billion in 1993 constant value dollars. This estimate is based on existing flight hardware cost estimates or from developing new technologies to produce new flight hardware. This analysis uses primarily cost estimates from a Lunar Lander cost analysis. This Lunar Lander analysis used previous cost data from the Apollo and Space Station Freedom programs. This cost analysis also looks at the manned UML only. The unmanned UML will differ from the manned lander UML because the unmanned UML will be remotely piloted from the orbiting Martian Transfer

Vehicle.^{E1}

The communication network (SCANS) will be made up of two advanced communication satellites in geosynchronous orbit around Mars. They will provide constant communication with the MTV, Earth, Space Station Freedom, or Space Station Mir. These satellites will be deployed shortly before reaching the Martian polar orbit.

The Earth-based integration and testing will provide a method of verifying proper functioning of the UML. The integration will consist of all subsystems in the UML testing together for an operational readiness test. This testing will provide the engineers with the means of finding errors in the assembling of subsystems. This test will help to ensure mission success.

A summary of the cost for the entire manned UML is presented below.

Table 1: Summary of manned and unmanned UML cost analysis with SCANS and Earth-based integration and testing.

Item	Cost in 1993 constant value dollars (Millions)
Truss Structure	283
Aeroshield	311
Landing Gear	119
Manned Module 1	335
Manned Module 2	362
Manned Module 3	414
Manned Module 4	343
Unmanned Module 1	345
Unmanned Module 2	421
Unmanned Module 3	421
Unmanned Module 4	356
GN&C hardware	164
GN&C software	42
CO fuel tank	150
O ₂ fuel tank	150
Main engines	1,105
Secondary engine	221
Secondary engine fuel tank	112
Unmanned UML (duplicate of manned UML)	2,660
SCANS	252
Earth-based integration and checkout	107
TOTAL COST	8,702

E1. Economic Methodology

E1.1 Assumptions. The major assumptions used when calculated the cost of the UML project can be found below.

1. No UML systems are presumed to be government or contractor provided. This also includes the purchase of new equipment without modification.
2. All costs are calculated in constant 1993 dollars.
3. All designs will use pre-existing technology or enhancements will be made to the hardware to incorporate state of the art technology.
4. The engineers are experienced with an average of 4 years experience designing similar hardware.
5. No spares or consumables are included in this cost analysis.
6. The cost of new facilities is not included in this cost analysis.
7. The tolerances used in manufacturing parts and assemblies will be 0.002 inches.
8. The operational costs of the UML is not included in this cost analysis.
9. The assembly tolerances will be the same as part tolerances.
10. A new method of integrating the subsystems will have to be developed.
11. The cost analysis allows for 7 prototypes to be built as well as 2 production landers. 5 prototypes will be used for testing and 2 for integration testing. The 2 production landers will be used to fly the missions. There will be only 3 prototypes of each module built and used for testing and integration. Eight production modules will then be used for the actual Mars mission.^{E1}

E1.2 Analysis Process. The UML's development and production costs were calculated using existing reports. Reports from Lunar Lander projects as well as Congressional Budget Offices were used to determine the cost of the UML project. The data from these reports was reviewed for an understanding of the data's origin. Next, the particular relevance of the reports'

operational scenario was studied. This was done so that adjustments to the UML cost calculations could be done more accurately. For example, if a piece of hardware that was designed for a lunar lander was similar in use to the UML hardware and yet was using older technology, the cost of the UML hardware component was scaled to incorporate increased research and development costs. Higher production costs might occur due to the fact that a new production process might be needed for the UML.^{E1}

This analysis estimates the cost to design, test, build, and integrate the various UML subsystems. Each phase has a development and production phase. The analysis is broken down into these 8 different phases. The cost analysis allows for any portion of the project cost to be defined carefully.^{E1}

The eight design phases were used to describe the different activities of the UML project. The drafting development covers the support personnel and support equipment needed to accurately assist in developing the UML subsystems. The support personnel equipment and equipment include draftsmen, computer aided drafting, and computer aided engineering. The computer aided engineering equipment could be used to perform structural and thermal analysis of the subsystem.^{E1}

Development and production design of technologies and producing plans to build a prototype component. Engineers will be working on new technologies to produce prototypes. By modifying existing hardware for new technologies will allow hardware to be improved. These upgrades can provide a tremendous cost savings over developing new technologies. Experimenting with new technologies can also provide better, faster, and cheaper methods of performing mission critical tasks.^{E1}

Development and production project management will supervise the development of the technology into prototype hardware. The management of contracts and the support personnel (contractors) within the contract can be a very large task. The management of civil servants working directly for NASA will need to be paid and managed.^{E1}

The manufacturing of the development and production prototype hardware pertains to the fabrication of the UML subsystems. The cost of materials will be a large factor in this phase of the cost analysis. The cost modifying the pre-existing facilities used to fabricate these prototypes will be taken into account in this phase. The machinery used in this process will be included in the cost analysis. The cost of the personnel needed to fabricate the prototype will be included in this phase.^{E1}

Testing of development prototypes and production components will provide the confidence in this new technology to use the hardware for a mission to Mars. The independent testing of the hardware will need to be accomplished in this phase. This will prove the hardware will perform properly in not only nominal but extreme flight conditions.^{E1}

There are several things that must be noted in this cost analysis. The operational cost of the UML is not considered. The cost analysis merely describes the cost of having the system ready to be given to the operators for operational integration. The cost analysis does not cover software development to operate the UML, except for the GN&C software. The cost of developing the testing and integration software is not included in the analysis.^{E1}

The cost analysis was developed to reflect constant 1993 dollars. Using financial economic equations, the cost data was transformed to the constant 1993 dollars.

$$F=P(1+i)^n \quad (E1)$$

where F was the 1993 constant dollar value, P was the dollar value for any given year, i was the constant yearly inflation rate, and n was the number of years from current year to 1993.^{E2}

E1.3 Estimates, Risks, and Uncertainty. Estimates that were made on hardware that contained new technology was based on similar existing hardware. The engines used on the UML have not been developed beyond the research phase. Therefore, preexisting engines of similar thrust production was used as a basis for accurately estimating the cost of the UML engines. The fuel production equipment is another example that had similar preexisting equipment used to generate a cost estimate. Cost estimates for similar technology hardware and GN&C software were based on historical data from Apollo, Space Shuttle, and Space Station programs. UML cost estimate data originated from cost estimates from the previous NASA programs. This data was then transformed into constant 1993 dollars.^{E1}

Risks and uncertainty exist with the development of key subsystems of the UML design. The engines used on the UML are still conceptual designs because of their fuel. This subsystem is critical to mission success. It is imperative that this subsystem function perfectly. The fuel production equipment is another vital part of the UML design. This is another important part of the

UML design. Without the fuel produced by this equipment, the astronaut crew could not return from Mars. This is just two major examples of the key subsystems that pose grave risk to the overall mission success of a manned mission to Mars. The uncertainty of the successful development of production components for these two examples is an issue that must be further studied. The question of whether or not these critical subsystems can be developed and fabricated for a projected cost need to be addressed. An uncertainty exists that these subsystems will increase the overall cost of the UML project. Cost control is a prime concern to Congress, who must vote on the allocation of federal funds when a new budget is being reviewed.

E2. ENGINEERING AND DESIGN ANALYSIS

E2.1 Universal Martian Lander

E2.1.1 Descent Phase. The cost of the UML during the descent phase of the mission will contain the cost of the truss structure, the landing gear, aeroshield, the MARV, and the modules. For a discussion of the cost for the modules see Section E2.1.2. That section will discuss the cost of the modules in more detail. The MARV will be discussed in more detail in Section E2.1.3. This section will focus on the hardware that is unique to only the descent phase of the mission. This is done in order to minimize the reader's effort to extract the necessary information. The truss structure, the aeroshield, and the landing gear cost will be analyzed in this section.

The cost of the truss structure was determined from extrapolated Lunar Lander cost estimates. The Lunar Lander design is different with respect to size of the vehicle and weight. However, the approach taken for the design process of the structure is similar for both the UML and Lunar Lander. The cost of the UML truss structure is slightly higher than the Lunar Lander because the UML truss has more truss members.^{E1}

The aeroshield was priced as though it was another structure. The cost analysis of the aeroshield was determined from the truss structure cost data. An aeroshield can be considered another structure that must undergo similar design steps as a truss structure. Due to the fact that the aeroshield has fewer structural members, the cost of the entire shield will be less than the truss structure. However, the thermal analysis applied to the aeroshield will be costly to the project. The extreme flight conditions (temperature) will drive the cost of testing the aeroshield up beyond most structural testing.

Landing gear cost was estimated to be 1/4 that of the truss structure. Landing gear designs have been prevalent in the aerospace industry for many years. The expertise built on the past landing gear designs will drive down the cost of the landing gear. Considerations that the UML landing gear size will have to be increased in order to withstand the heavier landing loads.

E2.1.2. Habitat and Laboratory Modules. The cost of the modules was calculated in similar manner as the truss structure with much more complicated internal systems. The module cost analysis did not only include the external shell with the hatches. The cost of the life support equipment accessories integrated into the module was included. These accessories include such items as ventilation duct work, fans, fire detection and suppression equipment, lighting, flooring, electrical wiring, sensor wiring, and computer wiring. The windows that were installed in the modules need to be included in the cost analysis. All these subsystem components had to be integrated into one subsystem.

The cost of the laboratory equipment increased the cost of the laboratory modules. Tighter tolerances for the laboratory equipment was needed in order to ensure accurate data collection. Sensitive laboratory equipment is generally more expensive than other equipment. Flight-rating that equipment will also increase the cost of that equipment. Launch and landing loads as well as the off gasing effects from the equipment are major concerns in the manned spaceflight industry. The laboratory equipment will also need to be isolated from vibrations. This will require more design and increase the cost of the laboratory module.

E2.1.3 Manned Ascent Return Vehicle. The cost considerations of the GN&C MARV contributions were found from the Lunar Lander cost estimates. The cost of the GN&C hardware was determined by using the Lunar Lander estimates as a starting point. The GN&C software estimates were done only for the development manufacturing and production manufacturing. These estimates were provided as a way of having a more complete analysis and yet should not be considered to be the final cost for the software. The GN&C software would be part of an active flight control system. The fuel tanks needed by the 4 main engines was analyzed as a structural component. Hence, the cost of these tanks was determined by extrapolating from the truss structure data. The fabrication costs for these tanks is the highest contributor to tank cost because of the composite material that will be used to create the tank.^{E1}

The main engine cost analysis was performed using existing price models but with a greatly increased design development and manufacturing development cost. The engines are still in the research phase and will require a large amount of money to develop into actual flight units. However, these engines are critical to the UML design because of the fuel used and the fact that the fuel will be produced on Mars.^{E1}

Section E2.1.4 Satellite Communication And Navigation System (SCANS). The cost of SCANS was determined from data gathered on other GN&C subsystems. SCANS is a key component of this system and will be used to communicate with the orbiting Martian Transfer Vehicle and Earth. The satellite system will consist of 2 satellites in geosynchronous orbits around Mars. These satellites will be deployed shortly before the Martian Transfer Vehicle enters the Martian polar orbit. The cost of these satellites was found by extrapolating from the GN&C cost estimates. The cost is greatest in the production of prototypes and of the flight hardware because of the new technologies being used in the new satellites.

E2.1.5 Earth-based Integration and Checkout. The concept of Earth-based Integration and Checkout will be to test an entire flight ready UML before it is flown on the mission. This process of integration and testing of the individual UML subsystems and finally all subsystems operating together as one flight unit will greatly enhance the chance of mission success. The cost of the integration testing was assessed at the cost levels reported for integration testing of the lunar lander. The Lunar Lander system is similar in the amount of systems needed to fulfill the mission. For this reason, the Lunar Lander was a good reference point for estimating the cost of the Lunar Lander.^{E1}

CONCLUSIONS

The UML design can provide a truly "universal" method of delivering payloads to the Martian surface. This design will be able to adapt to any mission requirement. Requirements for more science equipment or more supplies for a preexisting Martian base scenario can be met using an UML.

The UML will have a standard pressurized or unpressurized vessel which can be used for a variety of needs. Each module is the same as other modules, with the exception of a large swinging door to allow the deployment of small transport vehicles. These swinging doors will be resealable. Each module can be attached from all 4 sides, thus increasing the flexibility of arranging the modules to form a base. Each module hatch can be pressurized or unpressurized for changing base needs.

The same vehicle is used to provide final deceleration and complete transport to Martian orbit during ascent. During ascent, the engines will be used to slow the UML to landing velocity. Once the fuel tanks are refueled with in situ propellant, the same motors will be used to rendezvous with the orbiting MTV. Having only one set of engines will greatly reduce mass and cost of the UML.

The recycling of vital resources helps reduce the total mass of the UML. By recycling all water, the amount of mass saved is several thousand kilograms. This saved mass will allow more scientific payloads to be taken to Mars. The same is true of replenishing air. Without recycling resources such as water and air, the mission would be greatly hampered.

The UML design has attempted to minimize psychological effects on a closed environment. There are 8 windows per module. This will give the crew members the feeling of an open environment. The UML design calls for individual sleep chambers for privacy and an overall easier atmosphere with which to live and work in. Through the use of BLSS, the hydroponically grown plants (wheat, lettuce) will provide the crew members a change of scenery that is reminiscent of Earth. The UML design uses a variety of means to counteract the closed environment effects.

RECOMMENDATIONS

A further study of the key subsystems within the UML design is crucial to mission success. The in situ propellant production equipment will need to be flight rated and tested on actual martian probe. The main engines of the MARV will need extensive research and development before production units can be built. These systems must function if the UML is to be flown as proposed in this design.

Of course, more detailed analysis will need to be performed on all subsystems. In some cases, linear approximations were used when the problem required nonlinear methods for finding solutions. There is a lack of component testing to verify that all technologies will work as needed. Once the individual components have been tested, the integration of these components into subsystems will require preparation time and testing.

REFERENCES

- SI1 Aldrin, Buzz, McConnell, Malcolm, Men From Earth, Bantam, New York, 1989.
- SI2 Stafford, Thomas P., America at the Threshold: America's Space Exploration Initiative, Synthesis Group, Arlington, VA, 1991.
- A1. Stein, Bland, A. and Philip R. Young, LDEF Materials Workshop 91, NASA Conference Publication 3162 Part 2, Langely Research Center, Hampton Va, November 19-22, 1991
- A2 French, James R. and Micheal D.Griffin, Space Vehicle Design, AIAA Educational Series, Washington D.C., 1991
- A3 Composite Materials Handbook
- A4 Currey, Norman, Landing Gear Design Handbook, Lockheed-Georgia Co., Marietta Georgia, 1982
- A5 Preliminary Load Assesment of Inline Launch Vehicles, Structures and Thermal analysis Branch, Marshal Space Flight Center, October 1989
- A6 Wodd, Dr. K. D., Spacecraft Design, Aerospace Vehicle Design Vol. II, Johnson Publishing Co. Boulder Co., 1964
- A7 Bradway, M. W. et. al, "Independent Orbiter Assessment: Assessment of The Mechanical Actuation Subsystem." 1988.
- A8 Lubin, George and Samuel J. Dastin. "Aerospace Applications of Composites." Handbook of Composites
- A9 Marshall A. "Sandwich Construction." Handbook of Composites
- A10 Noton. Composites Materials.
- A11 Salkind and Holister. "Applications of Composite Materials."
- A12 ASM Handbook Volume 3.
- A3.1 Simonsen, L.C. & Nealy, J.E. 1991: *Radiation Protection for Human Missions to the Moon and Mars*. NASA TP-3079
- A3.2 Griffin, M.D. & French, J.R. *Space Vehicle Design*. AIAA. 1991. p 70.
- A3.3 Borowski, S.K., Gabris, E.A., & Martinell, J. "Nuclear thermal rockets: next step to space." *Aerospace America*. June 1989. Vol 27, No. 6. pp. 16-18.

- B1 Masursky, Harold & Crabill, Norman. *Viking Site Selection and Certification*. NASA Scientific and Technical Information Branch. Washington, D.C. 1981. p.9
- B2 Spitzer, Cary R. *Viking Orbiter Views of Mars*. NASA Scientific and Technical Information Branch. Washington, D.C. 1980. pp. 161-167
- B3 Carr, Michael H. *Surface of Mars*. Yale University Press. New Haven, Connecticut. 1981. p.106
- B4 Wiesel, William E. *Spaceflight Dynamics*. McGraw-Hill, Inc. 1989. p.55
- B5 Ibid., p.73
- B6 Ibid., p.85
- B7 Ibid., p.84
- B8 Ibid., p.86
- B9 Ibid., p.56
- B10 Ibid., p.87
- B11 Ibid., p.73
- B12 George C. Marshall Space Flight Center. *LTV Aerobrake Assesment: Comparison of Rigid and Flexible/Deployable Concepts*. NASA. 1990.
- B13 Regan, Frank. *Re-entry Vehicle Dynamics*. American Institute of Aeronautics and Astronautics. New York. 1984.
- B14 Griffin, M.D. & French, J.R. *Space Vehicle Design*. AIAA. p.57
- B15 Wiesel, William E. *Spaceflight Dynamics*. McGraw-Hill, Inc. 1989. p.207
- B16 Ibid., p.73
- B17 Ibid.
- B18 Ibid.
- B19 Ibid.
- B20 Ibid.
- B21 Agrawal, Brij N. *Design of Geosynchronous Spacecraft*. Prentice-Hall, Inc. Englewood Cliffs, New Jersey. 1986
- B22 NASA 60 GHz Intersatellite Communication Link Definition Study. Ford Aerospace Communications Corporation. Palo Alto, California. 1990
- B23 Ibid., A-40

- ..
- B24 *In-House Study Design of a LEV*. Marshall Space Flight Center. December 1991.
- B25 *Definition of Avionics Concepts for a Heavy Lift Cargo Vehicle*. General Dynamics Space Systems Division. September 1989
- C1. Regan, Frank J., Re-Entry Vehicle Dynamics., AIAA, 1984, pp. 97-135.
- C2. Griffin, Michael D., and French, James R., Space Vehicle Design., AIAA, 1984, pg. 261.
- C3. Tartabini, Paul V., and Suit, W.T., "Aerobraking Characteristics for Several Potential Manned Mars Entry Vehicles", NASA Langley Research Center, Hampton, VA, N90-14272, Nov. 1989.
- C4. Dickerson, Tom, "LTV Aerobrake Assessment", NASA Marshall Space Flight Center, Huntsville, AL, July 1990.
- C5. White, Frank M., Fluid Mechanics., 2nd edition, McGraw-Hill, New York, 1986, pg. 418.
- C6. Bendura, R.J., Lundstrom, R.R., Renfro, P.G., and LeCroy, S.R., "Flight Tests of Viking Parachute System in Three Mach Number Regimes," NASA Langley Research Center, Hampton, Virginia, NASA TN-7734, Nov. 1974, pp. 4-56.
- C7. Luhman, J.G., Tatrallyay, Mariella, and Pepin, Robert O., Venus and Mars: Atmospheres, Ionospheres, and Solar Wind Interactions., American Geophysical Union, Washington D.C., 1992, pg. 94.
- C8. Clayton, Joseph, and Knox, E.C., "TABI Thermal and Structural Properties," NASA contract NAS8-37740 for Marshall Space Flight Center, REMTECH RTR 195-1-01, December 1988.
- C9. White, Frank M., Heat and Mass Transfer., Addison-Wesley, New York, 1988, pg. 342.
- C10. Mills, A.F., Heat Transfer., Irwin, Concord, 1992, pp. 16 and 825.
- C11. Anderson, John D., Jr., Hypersonic and High Temperature Gas Dynamics., McGraw-Hill, New York, 1989, pp. 507-513.
- C12. James, M.L., Smith, and Woford. Applied Numerical Methods for Digital Computation., 3rd Ed., Harpers and Row, 1985.
- C13. Chin, S.S., Missile Configuration Design., The Martin Company, Orlando, FL, McGraw-Hill, New York, 1961.
- C14. Roncace, E.A., "In-Situ Propellant Rocket Engines for Mars Mission Ascent Vehicle", NASA Lewis Research Center, Cleveland, Ohio, N91-24305, June 1991.
- C15. Roncace, E.A., and Wadel, M.F., "Propulsion Systems Using In Situ Propellants for a Mars Ascent Vehicle", NASA Lewis Research Center, Cleveland, Ohio, N92-27654, July 1992.
- C16. Linne, Dianne L., and Meyor, Michael L., "Technical Projects for Utilizing Extraterrestrial Propellants for Space Exploration", NASA TM-105263, Prepared for the 42nd International Astronautical Congress, Montreal Canada, October 1991.

- C17. Zubrin, Robert M., "Indigenous Martian Propellant", Aerospace America, August 1989, pp. 48-51.
- C18. Landis, Geoffrey A., and Linne, Dianne L., "Acetylene Fuel from Atmospheric CO₂ on Mars", Journal of Spacecraft and Rockets, Vol. 28, No. 2, 1991.
- C19. Armstrong, Elizabeth S., "Cooling of In Situ Propellant Rocket Engines for Mars Mission", NASA TM-103729, NASA Lewis Research Center, Cleveland, OH, January 1991.
- C20. Rockwell International, "a past to build a future on," Rocketdyne Division, Canoga Park, CA, 1991.
- D1 Gustan, E. and Vinopal, T., Controlled ecological Life Support System: A Transportation Analysis. NASA-CR 166420. NASA-Ames Research Center Moffett Field, CA (1982)
- D2 McElroy, R. D., CELSS and Regenerative Life Support For Manned Missions To Mars Life Science Division NASA-Ames Research Center
- D3 Skoog, A. I., BLSS: A Contribution To Future Life Support Adv. Space Rec. Vol. 4, No. 12, pp. 251-262, 1984
- D4 Quattrone, P. D., Extended Mission Life Support Systems AAS 81-237. NASA-Ames Research Center
- D5 First Lunar Outpost / Lunar Habitat Documentation. NASA, Marshall Space Flight Center, May 1992.
- D6 Lunar Lander Design for the First Lunar Outpost. NASA, Johnson Space Flight Center, June 1992.
- D7 Space Flight Vehicle. Interim Review Briefing #7. Boeing, August 1991.
- D8 Extravehicular Activity System and Airlock Comprehension. Mission Operations Directorate, Matt Lindsey, Nov. 1991.
- D9 Bader, Manfred Moeller, Peter & Munder, Jost; Conceptual Design of the Columbus Free Flying Laboratory Resource Module Thermal Control System. Germany: Dornier GmbH, 1991. SAE 911447.
- D11 Buchko, Mathew & Schweickart; Development and Test of a 2 Phase Reservoir for Thermal Transport Systems Used in Micro-Gravity. Greenbelt, MD: 1991. SAE 911591.
- D12 "The Unbearable Lightness of Space Travel," Popular Science.
- D13 Nicogossian, Arnauld E. & Parker, James F., Jr. Space Physiology & Medicine. Washington, D.C.: NASA, 1992. TL521.A333 no.447.
- D14 Design of a Fast Crew Transfer Vehicle to Mars. Scientific and Technical Aerospace Report, 1988, N89-15968.
- D15 Medical Equipment Management in Hospitals.
- D16 Space Station Wardroom Habitability and Equipment Study. Scientific and Technical Aerospace Report, N90-17308.

- D17 Human Performance Issues Arising from Manned Space Station Missions. Scientific and Technical Aerospace Report, N88-25156
- D18 Lunc, Michal, ed. Life in Space Craft. New York: Gordon & Breach, 1966. TL787.I44 15th, 1965. V.7.
- D19 The Human factor. TL521.A333,no. 4213.
- D20 Johnston, Richard S., et al. Biomedical Results of Apollo. Washington, D.C.: NASA, 1975. TL521.A333 no.368.
- D21 Connors, Mary M., et al. Living Aloft: Human Requirements for Extended Spaceflight. Washington, D.C.: NASA, 1985. TL521.A333 no.483.
- D22 Johnston, Richard S. & Dietlein, Lawrence F., eds. Biomedical Results from Skylab. Washington, D.C.: NASA, 1977. TL521.A333 no.377.
- D23 Stuster, Jack. Space Station Habitability Recommendations Based on a Systematic Comparative Analysis of Analogous Conditions. Scientific and Technical Aerospace Report, 1986. N88-25372.
- D24 George C. Marshall Space Flight Center, First Lunar Outpost, Lunar Habitat Documentation, May 1992. NASA
- D25 William E. Simon, Donald L. Nored, Manned Spacecraft Electrical Power Systems, Proceedings of the IEEE, vol.75, No.3, March 1987
- D26 The National Aeronautics and Space Administration Manned Spacecraft center, APOLLO SPACECRAFT NEWS REFERENCE, Grumman Aircraft engineering corporation, Pg EP-1
- D27 Michael D. Griffin, James R. French, Space Vehicle Design, AIAA
- D28 Lisa L. Kohout, Cryogenic Reactant Storage For Lunar Base, Regenerative Fuel Cells, NASA TM-101980
- D29 N. P. Kim, R. M. Burgess, R. A. Mickelsen, and B. J. Stanbery, Progress in GaAs/CuInSe₂ Tandem Junction Solar Cells, N91- 19190
- D30 Robert J. Giudici, Electrical Power Systems For Mars, Marshall Space Flight Center, N87-17795
- D31 Barbara I. McKissock, Lisa L. Kohout, A Solar Power System For An Early Mars Expedition, NASA TM-103219
- D32 Robert C. Smoot, Jack Price, Richard G. Smith, Chemistry - A Modern Course, Merrill Publishing Co. Columbus, Ohio.
- E1 Eagle Engineering, Inc. "Lunar Base Scenario Cost Estimates," Eagle Engineering, Inc., Houston, Texas, N89-15286, Oct.1988.
- E2 "Engineer-In-Training Reference Manual," Engineering Economic Analysis, Professional Publications, Inc., Belmont, 1992, p13-5.

Appendix A1: Manned UML Mass Moment of Inertia calculations.

Item	Mass (kg)	X bar (m)	Y bar (m)	Z bar (m)	Xcg bar (m)	Ycg bar (m)	Zcg bar (m)
Fin #1	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
Fin #2	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
Fin #3	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
Fin #4	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
CD&H	200	0.00	0.00	0.50	-0.03	0.11	8.77
Crew Flight Seats	100	0.00	0.00	1.50	-0.03	0.11	9.77
Life Support equip (food, oven, clothes, water, toilet)	250	0.00	0.00	2.00	-0.03	0.11	10.27
Airlock	300	0.00	0.00	3.50	-0.03	0.11	11.77
ARS (8 person)	224	0.00	0.00	3.00	-0.03	0.11	11.27
Fuel CO	11,239	0.00	0.00	6.33	-0.03	0.11	14.59
Fuel O2	6,548	0.00	0.00	8.98	-0.03	0.11	17.24
Main Engine 1 (chamber)	542	1.00	1.00	12.20	0.97	1.11	20.47
Main Engine 1 (nozzle)	100	1.00	1.00	12.50	0.97	1.11	20.77
Main Engine 2 (chamber)	542	0.00	1.00	12.20	-0.03	1.11	20.47
Main Engine 2 (nozzle)	100	1.00	0.00	12.50	0.97	0.11	20.77
Main Engine 3 (chamber)	542	0.00	-1.00	12.20	-0.03	-0.89	20.47
Main Engine 3 (nozzle)	100	0.00	-1.00	12.50	-0.03	-0.89	20.77

Appendix A1: Manned UML Mass Moment of Inertia calculations.

Main Engine 4 (chamber)	542	-1.00	0.00	12.20	-1.03	0.11	20.47
Main Engine 4 (nozzle)	100	-1.00	0.00	12.50	-1.03	0.11	20.77
Secondary Motor (chamber)	25	0.00	0.00	4.25	-0.03	0.11	12.52
Secondary Motor (nozzle)	50	0.00	0.00	4.25	-0.03	0.11	12.52
Secondary Motor fuel tank	125	0.00	0.00	4.00	-0.03	0.11	12.27
Aeroshield	7,298	0.00	0.00	13.50	-0.03	0.11	21.77
Suit #1	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #2	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #3	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #4	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #5	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #6	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #7	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #8	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #9	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #10	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #11	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #12	125	0.00	0.00	3.00	-0.03	0.11	11.27
Battery/Fuel Cell #1	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #2	60	0.00	0.00	2.00	-0.03	0.11	10.27

Appendix A1: Manned UML Mass Moment of Inertia calculations.

Battery/Fuel Cell #3	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #4	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #5	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #6	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #7	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #8	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #9	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #10	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #11	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #12	60	0.00	0.00	2.00	-0.03	0.11	10.27
Landing Gear # 1	100	5.04	0.00	13.00	5.01	0.11	21.27
Landing Gear # 2	100	3.09	0.00	13.00	3.06	0.11	21.27
Landing Gear # 3	100	0.00	-5.04	13.00	-0.03	-4.93	21.27
Landing Gear # 4	100	0.00	-3.09	13.00	-0.03	-2.98	21.27
Landing Gear # 5	100	-5.04	0.00	13.00	-5.06	0.11	21.27
Landing Gear # 6	100	-3.09	0.00	13.00	-3.12	0.11	21.27
Landing Gear # 7	100	0.00	3.09	13.00	-0.03	3.20	21.27
Landing Gear # 8	100	0.00	5.04	13.00	-0.03	5.15	21.27
Truss Structure	6,419	0.00	0.00	10.00	-0.03	0.11	18.27
1 M Module	4,495	0.00	-4.50	7.00	-0.03	-4.39	15.27
2 M Module	5,172	4.50	0.00	7.00	4.47	0.11	15.27
3 M Module	5,855	0.00	4.50	7.00	-0.03	4.61	15.27
4 M Module	5,502	-4.50	0.00	7.00	-4.53	0.11	15.27

Appendix A1: Manned UML Mass Moment of Inertia calculations.

Item	Ixx (kg m ²)	Iyy (kg m ²)	Izz (kg m ²)	Ixy (kg m ²)	Iyz (kg m ²)	Izx (kg m ²)
Fin #1	21,385	16,426	1,950	931	-3,775	-3,775
Fin #2	21,385	16,426	1,950	931	-3,775	-3,775
Fin #3	21,385	16,426	1,950	931	-3,775	-3,775
Fin #4	21,385	16,426	1,950	931	-3,775	-3,775
CD&H	15,395	28	-1	0	0	0
Crew Flight Seats	9,876	339	0	0	0	0
Life Support equip (food, oven, clothes, water, toilet)	28,349	2,003	-1	0	0	0
Airlock	54,396	12,866	-1	0	0	0
ARS (8 person)	34,496	6,054	-1	0	0	0
Fuel CO	5,246,835	2,854,124	-32	0	0	0
Fuel O2	6,683,088	4,736,696	-19	0	0	0
Main Engine 1 (chamber)	1,217,973	991,685	13,806	6,610	80,642	80,642
Main Engine 1 (nozzle)	240,027	196,791	2,608	1,250	15,625	15,625
Main Engine 2 (chamber)	1,217,459	984,562	6,594	0	80,642	0
Main Engine 2 (nozzle)	238,777	196,669	1,261	0	0	15,625
Main Engine 3 (chamber)	1,217,459	984,324	6,622	0	-80,642	0
Main Engine 3 (nozzle)	239,932	195,403	1,252	0	-15,625	0

Appendix A1: Manned UML Mass Moment of Inertia calculations.

Main Engine 4 (chamber)	1,211,419	991,081	6,549	0	0	-80,642
Main Engine 4 (nozzle)	238,787	196,680	1,239	0	0	-15,625
Secondary Motor (chamber)	5,836	1,920	0	0	0	0
Secondary Motor (nozzle)	11,687	3,840	0	0	0	0
Secondary Motor fuel tank	26,820	8,014	0	0	0	0
Aeroshield	21,852,500	18,395,006	-21	0	0	0
Suit #1	19,336	3,471	0	0	0	0
Suit #2	19,336	3,471	0	0	0	0
Suit #3	19,336	3,471	0	0	0	0
Suit #4	19,336	3,471	0	0	0	0
Suit #5	19,336	3,471	0	0	0	0
Suit #6	19,336	3,471	0	0	0	0
Suit #7	19,336	3,471	0	0	0	0
Suit #8	19,336	3,471	0	0	0	0
Suit #9	19,336	3,471	0	0	0	0
Suit #10	19,336	3,471	0	0	0	0
Suit #11	19,336	3,471	0	0	0	0
Suit #12	19,336	3,471	0	0	0	0
Battery/Fuel Cell #1	6,783	479	0	0	0	0
Battery/Fuel Cell #2	6,783	479	0	0	0	0

Appendix A1: Manned UML Mass Moment of Inertia calculations.

..

Battery/Fuel Cell #3	6,783	479	0	0	0	0
Battery/Fuel Cell #4	6,783	479	0	0	0	0
Battery/Fuel Cell #5	6,783	479	0	0	0	0
Battery/Fuel Cell #6	6,783	479	0	0	0	0
Battery/Fuel Cell #7	6,783	479	0	0	0	0
Battery/Fuel Cell #8	6,783	479	0	0	0	0
Battery/Fuel Cell #9	6,783	479	0	0	0	0
Battery/Fuel Cell #10	6,783	479	0	0	0	0
Battery/Fuel Cell #11	6,783	479	0	0	0	0
Battery/Fuel Cell #12	6,783	479	0	0	0	0
Landing Gear # 1	267,438	255,209	33,051	0	0	85,142
Landing Gear # 2	265,864	233,053	12,446	0	0	52,221
Landing Gear # 3	297,921	222,129	33,009	0	-85,142	0
Landing Gear # 4	277,338	220,588	12,420	0	-52,221	0
Landing Gear # 5	267,490	255,261	32,940	0	0	-85,142
Landing Gear # 6	265,896	233,085	12,378	0	0	-52,221
Landing Gear # 7	277,338	220,724	12,404	0	52,221	0
Landing Gear # 8	297,921	222,350	32,983	0	85,142	0
Truss Structure	205,700	272,500	9,759,740	105	105	115
1 M Module	3,226,413	1,628,360	637,647	0	-991,103	0
2 M Module	3,082,915	2,610,724	735,667	0	0	1,140,426
3 M Module	4,202,778	2,132,677	829,250	0	1,291,028	0
4 M Module	3,282,176	2,779,857	777,180	0	0	-1,213,191

Appendix A2: Unmanned UML Mass Moment of Inertia

Item	Mass (kg)	X bar (m)	Y bar (m)	Z bar (m)	Xcg bar (m)	Ycg bar (m)	Zcg bar (m)
Fin #1	15	-2.48	-2.48	10.07	-3	-2	18
Fin #2	15	-2.48	-2.48	10.07	-3	-2	18
Fin #3	15	-2.48	-2.48	10.07	-3	-2	18
Fin #4	15	-2.48	-2.48	10.07	-3	-2	18
CD&H	200	0.00	0.00	0.50	0	0	9
Crew Flight Seats	100	0.00	0.00	1.50	0	0	10
Life Support equip (food, oven, clothes, water, toilet)	250	0.00	0.00	2.00	0	0	10
Airlock	300	0.00	0.00	3.50	0	0	12
ARS (8 person)	224	0.00	0.00	3.00	0	0	11
Fuel CO	11,239	0.00	0.00	6.33	0	0	15
Fuel O2	6,548	0.00	0.00	8.98	0	0	17
Main Engine 1 (chamber)	542	1.00	1.00	12.20	1	1	20
Main Engine 1 (nozzle)	100	1.00	1.00	12.50	1	1	21
Main Engine 2 (chamber)	542	0.00	1.00	12.20	0	1	20
Main Engine 2 (nozzle)	100	1.00	0.00	12.50	1	0	21
Main Engine 3 (chamber)	542	0.00	-1.00	12.20	0	-1	20

Appendix A2: Unmanned UML Mass Moment of Inertia

Main Engine 3 (nozzle)	100	0.00	-1.00	12.50	0	-1	21
Main Engine 4 (chamber)	542	-1.00	0.00	12.20	-1	0	20
Main Engine 4 (nozzle)	100	-1.00	0.00	12.50	-1	0	21
Secondary Motor (chamber)	25	0.00	0.00	4.25	0	0	13
Secondary Motor (nozzle)	50	0.00	0.00	4.25	0	0	13
Secondary Motor fuel tank	125	0.00	0.00	4.00	0	0	12
Aeroshield	7,298	0.00	0.00	13.50	0	0	22
Suit #1	125	0.00	0.00	3.00	0	0	11
Suit #2	125	0.00	0.00	3.00	0	0	11
Suit #3	125	0.00	0.00	3.00	0	0	11
Suit #4	125	0.00	0.00	3.00	0	0	11
Suit #5	125	0.00	0.00	3.00	0	0	11
Suit #6	125	0.00	0.00	3.00	0	0	11
Suit #7	125	0.00	0.00	3.00	0	0	11
Suit #8	125	0.00	0.00	3.00	0	0	11
Suit #9	125	0.00	0.00	3.00	0	0	11
Suit #10	125	0.00	0.00	3.00	0	0	11
Suit #11	125	0.00	0.00	3.00	0	0	11
Suit #12	125	0.00	0.00	3.00	0	0	11
Battery/Fuel Cell #1	60	0.00	0.00	2.00	0	0	10

Appendix A2: Unmanned UML Mass Moment of Inertia

Battery/Fuel Cell #2	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #3	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #4	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #5	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #6	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #7	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #8	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #9	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #10	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #11	60	0.00	0.00	2.00	0	0	10
Battery/Fuel Cell #12	60	0.00	0.00	2.00	0	0	10
Landing Gear # 1	100	5.04	0.00	13.00	5	0	21
Landing Gear # 2	100	3.09	0.00	13.00	3	0	21
Landing Gear # 3	100	0.00	-5.04	13.00	0	-5	21
Landing Gear # 4	100	0.00	-3.09	13.00	0	-3	21
Landing Gear # 5	100	-5.04	0.00	13.00	-5	0	21
Landing Gear # 6	100	-3.09	0.00	13.00	-3	0	21
Landing Gear # 7	100	0.00	3.09	13.00	0	3	21
Landing Gear # 8	100	0.00	5.04	13.00	0	5	21
Truss Structure	6,419	0.00	0.00	10.00	0	0	18
1 U Module	13,173	0.00	-4.50	7.00	-0.03	-4.39	15.27
2 U Module	4,313	4.50	0.00	7.00	4.47	0.11	15.27
3 U Module	4,313	0.00	4.50	7.00	-0.03	4.61	15.27
4 U Module	7,530	-4.50	0.00	7.00	-4.53	0.11	15.27

Appendix A2: Unmanned UML Mass Moment of Inertia

Item	Ixx (kg m ²)	Iyy (kg m ²)	Izz (kg m ²)	Ixy (kg m ²)	Iyz (kg m ²)	Izx (kg m ²)
Fin #1	5,128	5,138	179	89	-653	-690
Fin #2	5,128	5,138	179	89	-653	-690
Fin #3	5,128	5,138	179	89	-653	-690
Fin #4	5,128	5,138	179	89	-653	-690
CD&H	15,372	15,370	3	-1	192	-45
Crew Flight Seats	9,539	9,538	1	0	107	-25
Life Support equip (food, oven, clothes, water, toilet)	26,352	26,349	3	-1	281	-66
Airlock	41,537	41,534	4	-1	387	-91
ARS (8 person)	28,448	28,445	3	-1	277	-65
Fuel CO	2,403,096	2,402,969	10,258	-32	17,969	-4,232
Fuel O2	1,949,339	1,949,265	2,872	-19	12,370	-2,913
Main Engine 1 (chamber)	227,689	227,536	1,249	586	12,304	10,803
Main Engine 1 (nozzle)	43,492	43,464	229	108	2,304	2,023
Main Engine 2 (chamber)	227,689	227,022	735	-16	12,304	-286
Main Engine 2 (nozzle)	43,370	43,464	107	11	228	2,023
Main Engine 3 (chamber)	227,452	227,022	498	12	-9,874	-286

Appendix A2: Unmanned UML Mass Moment of Inertia

Main Engine 3 (nozzle)	43,449	43,369	90	2	-1,849	-54
Main Engine 4 (chamber)	227,029	227,592	644	-61	1,215	-11,375
Main Engine 4 (nozzle)	43,370	43,474	117	-11	228	-2,130
Secondary Motor (chamber)	3,917	3,917	1	0	34	-8
Secondary Motor (nozzle)	7,849	7,848	2	0	69	-16
Secondary Motor fuel tank	18,822	18,820	14	0	168	-40
Aeroshield	3,896,766	3,896,683	439,189	-21	17,406	-4,099
Suit #1	15,962	15,961	96	0	154	-36
Suit #2	15,962	15,961	96	0	154	-36
Suit #3	15,962	15,961	96	0	154	-36
Suit #4	15,962	15,961	96	0	154	-36
Suit #5	15,962	15,961	96	0	154	-36
Suit #6	15,962	15,961	96	0	154	-36
Suit #7	15,962	15,961	96	0	154	-36
Suit #8	15,962	15,961	96	0	154	-36
Suit #9	15,962	15,961	96	0	154	-36
Suit #10	15,962	15,961	96	0	154	-36
Suit #11	15,962	15,961	96	0	154	-36
Suit #12	15,962	15,961	96	0	154	-36
Battery/Fuel Cell #1	6,306	6,305	1	0	67	-16

Appendix A2: Unmanned UML Mass Moment of Inertia ..

Battery/Fuel Cell #2	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #3	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #4	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #5	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #6	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #7	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #8	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #9	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #10	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #11	6,306	6,305	1	0	67	-16
Battery/Fuel Cell #12	6,306	6,305	1	0	67	-16
Landing Gear # 1	45,227	47,738	2,513	55	233	10,659
Landing Gear # 2	45,227	46,164	940	34	233	6,516
Landing Gear # 3	47,654	45,226	2,429	13	-10,481	-55
Landing Gear # 4	46,114	45,226	888	8	-6,338	-55
Landing Gear # 5	45,227	47,790	2,565	-55	233	-10,769
Landing Gear # 6	45,227	46,196	972	-34	233	-6,626
Landing Gear # 7	46,249	45,226	1,024	-8	6,804	-55
Landing Gear # 8	47,875	45,226	2,650	-13	10,947	-55
Truss Structure	205,700	272,500	1,161,104	105	105	115
1 U Module	3,324,013	3,070,101	253,930	1,492	-882,927	-5,189
2 U Module	1,005,237	1,091,525	86,391	2,114	7,215	294,597
3 U Module	1,096,829	1,005,188	91,646	-513	303,510	-1,699
4 U Module	1,755,028	1,909,174	154,327	-3,734	12,596	-520,264

Appendix A3: MARV Mass Moment of Inertia calculations.

Item	Mass (kg)	X bar (m)	Y bar (m)	Z bar (m)	Xcg bar (m)	Ycg bar (m)	Zcg bar (m)
Fin #1	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
Fin #2	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
Fin #3	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
Fin #4	15	-2.48	-2.48	10.07	-2.51	-2.37	18.34
CD&H	200	0.00	0.00	0.50	-0.03	0.11	8.77
Crew Flight Seats	100	0.00	0.00	1.50	-0.03	0.11	9.77
Life Support equip (food, oven, clothes, water, toilet)	250	0.00	0.00	2.00	-0.03	0.11	10.27
Airlock	300	0.00	0.00	3.50	-0.03	0.11	11.77
ARS (8 person)	224	0.00	0.00	3.00	-0.03	0.11	11.27
Fuel CO	11,239	0.00	0.00	6.33	-0.03	0.11	14.59
Fuel O2	6,548	0.00	0.00	8.98	-0.03	0.11	17.24
Main Engine 1 (chamber)	542	1.00	1.00	12.20	0.97	1.11	20.47
Main Engine 1 (nozzle)	100	1.00	1.00	12.50	0.97	1.11	20.77
Main Engine 2 (chamber)	542	0.00	1.00	12.20	-0.03	1.11	20.47
Main Engine 2 (nozzle)	100	1.00	0.00	12.50	0.97	0.11	20.77
Main Engine 3 (chamber)	542	0.00	-1.00	12.20	-0.03	-0.89	20.47
Main Engine 3 (nozzle)	100	0.00	-1.00	12.50	-0.03	-0.89	20.77

Appendix A3: MARV Mass Moment of Inertia calculations.

Main Engine 4 (chamber)	542	-1.00	0.00	12.20	-1.03	0.11	20.47
Main Engine 4 (nozzle)	100	-1.00	0.00	12.50	-1.03	0.11	20.77
Secondary Motor (chamber)	25	0.00	0.00	4.25	-0.03	0.11	12.52
Secondary Motor (nozzle)	50	0.00	0.00	4.25	-0.03	0.11	12.52
Secondary Motor fuel tank	125	0.00	0.00	4.00	-0.03	0.11	12.27
Aeroshield	7,298	0.00	0.00	13.50	-0.03	0.11	21.77
Suit #1	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #2	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #3	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #4	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #5	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #6	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #7	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #8	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #9	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #10	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #11	125	0.00	0.00	3.00	-0.03	0.11	11.27
Suit #12	125	0.00	0.00	3.00	-0.03	0.11	11.27
Battery/Fuel Cell #1	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #2	60	0.00	0.00	2.00	-0.03	0.11	10.27

Appendix A3: MARV Mass Moment of Inertia calculations.

Battery/Fuel Cell #3	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #4	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #5	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #6	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #7	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #8	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #9	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #10	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #11	60	0.00	0.00	2.00	-0.03	0.11	10.27
Battery/Fuel Cell #12	60	0.00	0.00	2.00	-0.03	0.11	10.27

Appendix A3: MARV Mass Moment of Inertia calculations.

Item	I_{xx} (kg m^2)	I_{yy} (kg m^2)	I_{zz} (kg m^2)	I_{xy} (kg m^2)	I_{yz} (kg m^2)	I_{zx} (kg m^2)
Fin #1	21,385	16,426	1,950	931	-3,775	-3,775
Fin #2	21,385	16,426	1,950	931	-3,775	-3,775
Fin #3	21,385	16,426	1,950	931	-3,775	-3,775
Fin #4	21,385	16,426	1,950	931	-3,775	-3,775
CD&H	15,395	28	-1	0	0	0
Crew Flight Seats	9,876	339	0	0	0	0
Life Support equip (food, oven, clothes, water, toilet)	28,349	2,003	-1	0	0	0
Airlock	54,396	12,866	-1	0	0	0
ARS (8 person)	34,496	6,054	-1	0	0	0
Fuel CO	5,246,835	2,854,124	-32	0	0	0
Fuel O2	6,683,088	4,736,696	-19	0	0	0
Main Engine 1 (chamber)	1,217,973	991,685	13,806	6,610	80,642	80,642
Main Engine 1 (nozzle)	240,027	196,791	2,608	1,250	15,625	15,625
Main Engine 2 (chamber)	1,217,459	984,562	6,594	0	80,642	0
Main Engine 2 (nozzle)	238,777	196,669	1,261	0	0	15,625
Main Engine 3 (chamber)	1,217,459	984,324	6,622	0	-80,642	0
Main Engine 3 (nozzle)	239,932	195,403	1,252	0	-15,625	0

Appendix A3: MARV Mass Moment of Inertia calculations.,.

Main Engine 4 (chamber)	1,211,419	991,081	6,549	0	0	-80,642
Main Engine 4 (nozzle)	238,787	196,680	1,239	0	0	-15,625
Secondary Motor (chamber)	5,836	1,920	0	0	0	0
Secondary Motor (nozzle)	11,687	3,840	0	0	0	0
Secondary Motor fuel tank	26,820	8,014	0	0	0	0
Aeroshield	21,852,500	18,395,006	-21	0	0	0
Suit #1	19,336	3,471	0	0	0	0
Suit #2	19,336	3,471	0	0	0	0
Suit #3	19,336	3,471	0	0	0	0
Suit #4	19,336	3,471	0	0	0	0
Suit #5	19,336	3,471	0	0	0	0
Suit #6	19,336	3,471	0	0	0	0
Suit #7	19,336	3,471	0	0	0	0
Suit #8	19,336	3,471	0	0	0	0
Suit #9	19,336	3,471	0	0	0	0
Suit #10	19,336	3,471	0	0	0	0
Suit #11	19,336	3,471	0	0	0	0
Suit #12	19,336	3,471	0	0	0	0
Battery/Fuel Cell #1	6,783	479	0	0	0	0
Battery/Fuel Cell #2	6,783	479	0	0	0	0

Appendix A3: MARV Mass Moment of Inertia calculations. ..

Battery/Fuel Cell #3	6,783	479	0	0	0	0
Battery/Fuel Cell #4	6,783	479	0	0	0	0
Battery/Fuel Cell #5	6,783	479	0	0	0	0
Battery/Fuel Cell #6	6,783	479	0	0	0	0
Battery/Fuel Cell #7	6,783	479	0	0	0	0
Battery/Fuel Cell #8	6,783	479	0	0	0	0
Battery/Fuel Cell #9	6,783	479	0	0	0	0
Battery/Fuel Cell #10	6,783	479	0	0	0	0
Battery/Fuel Cell #11	6,783	479	0	0	0	0
Battery/Fuel Cell #12	6,783	479	0	0	0	0

APPENDICES

Appendix C1.1

ADP.BAS Computer Program

```
DECLARE FUNCTION TEMP! (H!)
DECLARE FUNCTION VPARA! (V!, RHOSL!, G0!, HS!, H!, BETA!, GAMAE2!, S!, MASS!, VH, PD,
CL, CD)
DECLARE SUB HEATRAN (CD!, VENTRY!, RHOSL!, G0!, HS!, H!, BETA!, GAMAE2!, MASS!, S!,
HENTRY!, RADIUS!, MU!, CL!, HPD, PD)
DECLARE FUNCTION VBAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)
DECLARE FUNCTION ABAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS, V, PD,
HPD)
DECLARE SUB DECBAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY,
RADIUS, MU, CL, CD, PD, HPD)
DECLARE SUB INPT (GAMAE, HS, CD, S, MASS, VENTRY, HENTRY, H, RHOSL, G0)
DECLARE SUB BALIST (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY,
RADIUS, MU, CL, CD, PD, HPD)
DECLARE SUB ASCENT ()
DECLARE FUNCTION GAMMA (HS, S, CL, CD, MASS, RHOSL, GAMMAE2, H)
```

```
CLS
```

```
SCREEN 9
```

```
LINE (100, 50)-(550, 50), 12: LINE (100, 50)-(100, 250), 12: LINE (100, 250)-(550, 250), 12: LINE (550,
250)-(550, 50), 12
```

```
LOCATE 7, 15: PRINT "          UNIVERSAL MARTIAN LANDER PROJECT"
LOCATE 9, 15: PRINT "    PROGRAM TO COMPUTE ASCENT AND DESENT PARAMETERS"
LOCATE 10, 15: PRINT "    BASED ON USER ENTERED U.M.L. PARAMETERS."
LOCATE 12, 15: PRINT "          Aerodynamics and Propulsion Division"
LOCATE 14, 15: PRINT "          PRESS ANY KEY TO BEGIN"
```

```
DO: LOOP WHILE INKEY$ = ""
```

```
SCREEN 0
```

```
PD = 42.76
```

```
GAMAE = -3
```

```
HS = 3216
```

```
S = 363
```

```
MASS = 59447
```

```
VENTRY = 3500
```

```
RHOSL = .0182
```

```
HENTRY = 60000
```

```
CD = 1.58
```

```
CF = .8
```

```
G0 = 3.72
```

```
MU = 4.30489E+13
```

```
GAMAE2 = GAMAE * 3.1415927# / 180
```

```
RADIUS = 3382000
```

```
BETA = (MASS * G0) / (CD * S)
```

```
HPD = 8000
```

```
100 CLS
```

```
MASS = 59447
```

```
PRINT "These are the selected atmospheric entry parameters:"
```

```
PRINT
```

```

PRINT "1. Aeroshield Drag Coefficient: "; CD
PRINT "2. Mass of Landing Vehicle: "; MASS; "kg"
PRINT "3. Frontal Area of Aeroshield: "; S; " m^2"
PRINT "4. Skin Friction Coefficient: "; CF
PRINT
PRINT "5. Entry Angle: "; GAMAE; " degrees"
PRINT "6. Entry Altitude: "; HENTRY; "meters"
PRINT "7. Entry Velocity: "; VENTRY; "m/s"
PRINT
PRINT "8. Scale Height of Atmosphere: "; HS; " meters"
PRINT "9. Sea Level Density: "; RHOSL; " kg/m^3"
PRINT "10. Sea Level Gravity: "; G0; " m/s^2"
PRINT "11. Parachute Diameter "; PD; " m"
PRINT "12. Height of Parachute Deployment: "; HPD; " meters"
PRINT
GAMAE2 = GAMAE * 3.1415927# / 180
BETA = (MASS * G0) / (CD * S)

```

```

110 INPUT "Enter the number of the parameter you want to change (0 for NONE, 9 for ALL):"; PAR

```

```

IF PAR = 1 THEN
  CLS
  INPUT "ENTER NEW CD: "; CD
  GOTO 100
ELSEIF PAR = 2 THEN
  CLS
  INPUT "ENTER NEW LANDER MASS (in kg): "; MASS
  GOTO 100
ELSEIF PAR = 3 THEN
  CLS
  INPUT "ENTER NEW FRONTAL AREA (in m^2):"; S
  GOTO 100
ELSEIF PAR = 4 THEN
  CLS
  INPUT "ENTER NEW SKIN FRICTION COEFFICIENT"; CF
  GOTO 100
ELSEIF PAR = 5 THEN
  CLS
  INPUT "ENTER NEW ENTRY ANGLE (in degrees):"; GAMAE
  GOTO 100
ELSEIF PAR = 6 THEN
  CLS
  INPUT "ENTER NEW ENTRY ALTITUDE (in meters): "; HENTRY
  GOTO 100
ELSEIF PAR = 7 THEN
  CLS
  INPUT "ENTER NEW ENTRY VELOCITY (in m/s): "; VENTRY
  GOTO 100
ELSEIF PAR = 8 THEN
  CLS
  INPUT "ENTER NEW SCALE HEIGHT (in meters): "; HS
  GOTO 100
ELSEIF PAR = 9 THEN
  CLS

```

```

INPUT "ENTER NEW SEA LEVEL DENSITY (in kg/m^3): "; RHOSL
GOTO 100
ELSEIF PAR = 10 THEN
  CLS
  INPUT "ENTER NEW GRAVITY AT SEA LEVEL"; G0
  GOTO 100
ELSEIF PAR = 0 THEN
  CLS
  GOTO 150
ELSEIF PAR = 11 THEN
  CLS
  INPUT "ENTER NEW PARACHUTE DIAMETER"; PD
  GOTO 100
ELSEIF PAR = 12 THEN
  CLS
  INPUT "ENTER NEW PARACHUTE DEPLOYMENT HEIGHT: "; HPD
  GOTO 100
ELSEIF PAR = 13 THEN
  CLS
  CALL INPT(GAMAE2, HS, CD, S, MASS, VENTRY, HENTRY, H, RHOSL, G0)
  GOTO 100
ELSE
  PRINT "SELECTION NOT VALID, ENTER NEW CHOICE"
  GOTO 110
END IF

```

```

150
CLS
PRINT "          MODELING MENU"
PRINT "          _____"
PRINT
PRINT
PRINT
PRINT "          1. RE-ENTRY"
PRINT "          2. ASCENT"
PRINT "          "
PRINT
INPUT "          Enter Selection: "; MODEL
175   IF MODEL = 1 THEN
200 CLS
PRINT TAB(15); "BALLISTIC RE-ENTRY MODEL"
PRINT TAB(15); "          _____"
PRINT
PRINT TAB(15); "ENTER SELECTION --"
PRINT TAB(15);
PRINT TAB(15); "1. PLOT ALTITUDE vs VELOCITY."
PRINT TAB(15); "2. PLOT ALTITUDE vs DECELERATION."
PRINT
PRINT TAB(15); "3. AERODYNAMIC HEATING CALCULATIONS."
PRINT
PRINT TAB(15); "9. CHANGE PARAMETERS."

```

```

PRINT TAB(15); "10. RETURN TO MODELING MENU"
PRINT
210 LOCATE 15, 15: INPUT "ENTER CHOICE:(0 TO END) "; CH

  IF CH = 1 THEN
    CALL BALIST(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS,
MU, CL, CD, PD, HPD)
  ELSEIF CH = 2 THEN
    CALL DECBAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY,
RADIUS, MU, CL, CD, PD, HPD)
    GOTO 200
  ELSEIF CH = 3 THEN
    CALL HEATRAN(CD, VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY,
RADIUS, MU, CL, HPD, PD)
    GOTO 200

  ELSEIF CH = 9 THEN
    GOTO 100
  ELSEIF CH = 0 THEN
    END
  ELSEIF CH = 10 THEN
    GOTO 150

  ELSE
    PRINT "INVALID SELECTION, ENTER NEW CHOICE"
    GOTO 210
  END IF

GOTO 200

```

```

  ELSEIF MODEL = 2 THEN
    CALL ASCENT
    GOTO 150
  ELSE
    INPUT "INVALID SELECTION -- ENTER NEW CHOICE: "; MODEL
    GOTO 175
  END IF

```

```

FUNCTION ABAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS, V, PD, HPD)
SS = 3.1416 / 4 * PD ^ 2
IF H > HPD THEN SS = 0
IF H > 5000 THEN : CDD = 1.58: AREA = S
IF H <= 5000 THEN : CDD = 1!: AREA = 70.93
FP = RHOSL * EXP(-H / HS) * .5 * V ^ 2 * SS * 1.7
FA = RHOSL * EXP(-H / HS) * .5 * V ^ 2 * CDD * AREA
ABAL = (FA + FP) / MASS

END FUNCTION

```

```

SUB ASCENT
1999  DTPR = 2
      FUEL = 0
      H = .1

```



```

EPSI = .001
MAV = 31758
FMASS = 18000
2000  CLS
      G = 3.72
      TH = 1340000
      FR = TH / (305 * 9.81) * 3.72
      PRINT "CURRENT ASCENT PARAMETERS:"
      PRINT
      PRINT "1. Mass of Ascent Vehicle and Fuel: "; MAV; " kg"
      PRINT "2. Initial Fuel Mass: "; FMASS; " kg"
      PRINT "3. Nominal Engine Thrust: "; TH / 1000; " KN"
      PRINT "4. Specific Fuel Consumption: "; FR; " kg/s"
      PRINT
      PRINT "Numerical Integration Parameters: "
      PRINT
      PRINT "5. Time Step: "; DTPR; " s"
      PRINT "6. Convergence Test: "; EPSI
      PRINT
      PRINT "9. RETURN TO MODELING MENU."
      PRINT

2010  INPUT "Enter the number of parameter to be changed (0 to accept): "; PAR3
      IF PAR3 = 1 THEN
        CLS
        INPUT "ENTER NEW MASS OF ASCENT VEHICLE AND FUEL: "; MAV
        GOTO 2000
      ELSEIF PAR3 = 2 THEN
        CLS
        INPUT "ENTER NEW FUEL MASS: "; FMASS
        GOTO 2000
      ELSEIF PAR3 = 3 THEN
        CLS
        INPUT "ENTER NEW THRUST "; F
        GOTO 2000
      ELSEIF PAR3 = 4 THEN
        CLS
        INPUT "ENTER NEW SPECIFIC FUEL CONSUMPTION: "; FR
        GOTO 2000
      ELSEIF PAR3 = 5 THEN
        CLS
        INPUT "ENTER NEW TIME STEP"; DTPR
        GOTO 2000
      ELSEIF PAR3 = 6 THEN
        CLS
        INPUT "ENTER NEW CONVERGENCE TEST: "; EPSI
        GOTO 2000
      ELSEIF PAR3 = 0 THEN
        GOTO 2050
      ELSE
        PRINT "INVALID SELECTION"

```



```

#####
IF ANSS = "Y" OR ANSS = "y" THEN
  PRINT #101, USING FRMS; W1; WF; TH; FR; G; DK; H; DTPR; EPSI
  PRINT #101,
  PRINT #101, " TIME      DISPL      VEL      ACC"
END IF
IF SEL$ = "TABLE" THEN
  PRINT "TIME      DISPL      VEL      ACC"
ELSEIF SEL$ = "GRAPH" THEN
  DTPR = .5
  SCREEN 9
  WINDOW (-10, 70)-(70, -15)
  LINE (0, 0)-(70, 0): LINE (0, 0)-(0, 100)
  LINE (0, -5)-(15, -5), 9: LINE (25, -5)-(34, -5), 12: LINE (50, -5)-(59, -5), 14
  LOCATE 23, 1: LOCATE 23, 21: PRINT "DISP. x E-3": LOCATE 23, 46: PRINT "VEL. x
E-2": LOCATE 23, 71: PRINT "ACCEL."
  LOCATE 22, 1: PRINT "      0      10      20      30      40      50      60      "
  LOCATE 1, 1: PRINT "      ASCENT POSITION, VELOCITY, AND
ACCELERATION vs TIME"
  LOCATE 1, 15
  FOR LABELA = 60 TO 0 STEP -10
    PRINT
    PRINT
    PRINT LABELA

    NEXT LABELA
  ELSE
    PRINT "ERROR"
    GOTO 1850
  END IF

  FMT2$ = "##.##   ###.##   ###.##   ###.##"
  T = 4 * H
  TMAX = 200
  TPR = DTPR
2015  FR = TH / (305 * 9.81) * 3.72
  W = W1 - FR * H
  W1 = W
  FUEL = FUEL + FR * H
  YD4P = YD + 4 / 3 * (2 * YDD1 - YDD2 + 2 * YDD3) * H
  RHOAS = .0182 * EXP(-Y4 / 3216)
  MACH = YD4C / SQR(191.46 * 1.32 * TEMP(Y4))
  IF MACH >= 1 THEN
    CDW = (.083 + .096 / MACH ^ 2) * 1.2703 * (1 + 30 / (14.4205 * (MACH + 18)))
  END IF
  CDS = .04
  DK = .5 * RHOAS * 4.25 ^ 2 * 3.1416 / 4 * CDW
  DKK = .5 * RHOAS * (3.1416 * 4.25 * 9.15 + 4.25 * 3.25 * 3.05) * CDS
  YDD4 = G * TH / W - G - DK * G / W * (YD4P ^ 2) - DKK * G / W * (YD4P ^ 2)
  IF YDD4 > 29.43 THEN
    YDD4 = 29.43
    TH = W / G * YDD4
    YD4C = YD3 + YDD4 * H

```

```

        GOTO 2018
    ELSE
        TH = 1340000
    END IF
2016  MACH = YD4C / SQR(191.46 * 1.32 * TEMP(Y4))
    IF MACH >= 1 THEN
        CDW = (.083 + .096 / MACH ^ 2) * 1.2703 * (1 + 30 / (14.4205 * (MACH + 18)))
    END IF
    CDS = .04
    DK = .5 * RHOAS * 4.25 ^ 2 * 3.1416 / 4 * CDW
    DKK = .5 * RHOAS * (3.1416 * 4.25 * 9.15 + 4.25 * 3.25 * 3.05) * CDS
    YD4C = YD2 + H / 3 * (YDD2 + 4 * YDD3 + YDD4)
    YDD4 = G * TH / W - G - DK * G / W * (YD4C ^ 2)
    IF YDD4 > 29.43 THEN
        YDD4 = 29.43
        TH = W / G * YDD4
        YD4C = YD3 + YDD4 * H
        GOTO 2018
    ELSE
        TH = 1340000
    END IF
    IF (ABS(YD4C - YD4P) - EPSI) <= 0 THEN
        GOTO 2018
    END IF
2017  YD4P = YD4C
    GOTO 2016
2018  Y4 = Y2 + H / 3 * (YD2 + 4 * YD3 + YD4C)
    IF (ABS(T - TPR) - .001) > 0 THEN GOTO 2021

2019  FRM2$ = "###.# #####.# #####.## ##.##  #####~  ##  #####"
2020
    MACH = YD4C / SQR(191.46 * 1.32 * TEMP(Y4))
    IF MACH >= 1 THEN
        CDW = (.083 + .096 / MACH ^ 2) * 1.2703 * (1 + 30 / (14.4205 * (MACH + 18)))
    END IF
    UI = .0000102
    TI = 200
    U = UI * ((TEMP(Y4) / TI) ^ (3 / 2)) * ((TI + 110) / (TEMP(Y4) + 110))
    REAS = RHOAS * YD4C * 12.2 / U
    IF FUEL >= WF THEN GOTO 2024
    IF ANS$ = "Y" OR ANS$ = "y" THEN PRINT #101, USING FRM2$, T; Y4; YD4C; YDD4; REAS;
MACH; CDW
    IF SEL$ = "TABLE" THEN
        PRINT USING FRM2$, T; Y4; YD4C; YDD4; REAS; MACH; CDW
    ELSEIF SEL$ = "GRAPH" THEN
        PSET (T, Y4 / 1000), 9
        PSET (T, YD4C / 100), 12
        PSET (T, YDD4), 14
    END IF

    TPR = TPR + DTPR
2021  IF (ABS(T - TMAX) - .01) > 0 THEN GOTO 2023
    GOTO 2024
2023  T = T + H

```

```

YDD1 = YDD2
YDD2 = YDD3
YDD3 = YDD4
YD = YD1
YD1 = YD2
YD2 = YD3
YD3 = YD4C
Y2 = Y3
Y3 = Y4
GOTO 2015
2024   CLOSE #101
DO: LOOP WHILE INKEY$ = ""
SCREEN 0
GOTO 1999
2050
END SUB

```

```

SUB BALIST (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS, MU,
CL, CD, PD, HPD)

```

```

CLS

```

```

'Calculate the VELOCITY as a function of altitude and plot to screen.

```

```

CLS
1800  PRINT "DO YOU WANT THIS WRITTEN TO A DATA FILE (Y/N)? "
      INPUT AN$$
      IF AN$$ = "Y" OR AN$$ = "y" THEN
        INPUT "ENTER THE FILENAME AND EXTENSION: "; FAMES$
        OPEN FAMES$ FOR OUTPUT AS #1
      ELSEIF AN$$ = "N" OR AN$$ = "n" THEN
        GOTO 1900
      ELSE
        PRINT "INVALID SELECTION -- TRY AGAIN"
        GOTO 1800
      END IF

```

```

1900

```

```

'Initialize the variables

```

```

V = VENTRY
H = HENTRY

```

```

SCREEN 9
WINDOW (-.5, 65)-(7, -15)
'VIEW SCREEN (-.5, 65)-(7, -15)
LINE (0, 0)-(7, 0): LINE (0, 0)-(0, 65)
LOCATE 1, 1: PRINT " ALTITUDE           LANDER VELOCITY AS A FUNCTION OF ALTITUDE"
LOCATE 22, 1: PRINT "   0.5  1.0  1.5  2.0  2.5  3.0  3.5  4.0  4.5  5.0  5.5  6.0  6.5"
LOCATE 23, 30: PRINT " V E L O C I T Y "

```

```

LOCATE 3
PCOUNT = 10
LABEL = HENTRY / 1000
FOR LABEL1 = 1 TO 7
  PRINT LABEL
  LABEL = LABEL - 10
  PRINT
  PRINT

```

```

NEXT LABEL1

```

'Calculations for ballistic re-entry. Assuming constant gravity

```

DO WHILE H > 0!
  IF H > HPD THEN
    V = VBAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)
  ELSEIF H <= HPD THEN
    IF H = 5000 THEN MASS = MASS - 7298
    V = VPARA(V, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS, VH, PD, CL, CD)
  ELSE
    V = V - 24.525 * 100 / VH
  END IF

  CL = 0
  GAMA3 = GAMMA(HS, S, CL, CD, MASS, RHOSL, GAMAE2, H)
  VH = V * SIN(GAMA3)
  IF ANS$ = "Y" OR ANS$ = "y" THEN
    PRINT #1, H, V
  END IF

  HP = H / 1000
  VP = V / 1000
  VHP = VH / 1000
  PSET (VP, HP), 10
  PSET (VHP, HP), 13
  H = H - 100
LOOP

```

```

DO: LOOP WHILE INKEY$ = ""
SCREEN 0
CLOSE #1

```

```

END SUB

```

```

SUB DECBAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS, MU,
CL, CD, PD, HPD)
500
CLS

```

'Calculate the VELOCITY as a function of altitude and plot to screen.

```

IF CH = 1 THEN
  CLS
505 PRINT "DO YOU WANT THIS WRITTEN TO A DATA FILE (Y/N)? "

```

```

INPUT ANSS
IF ANSS = "Y" OR ANSS = "y" THEN
  INPUT "ENTER THE FILENAME AND EXTENSION:"; FAMES
  OPEN FAMES FOR OUTPUT AS #1
ELSEIF ANSS = "N" OR ANSS = "n" THEN
  GOTO 510
ELSE
  PRINT "INVALID SELECTION -- TRY AGAIN"
  GOTO 505
END IF

510 PRINT "VELOCITY      HEIGHT"
PRINT "          "
FMT1$ = "###.##      ###.##"
ELSE

END IF

515 INPUT "DO YOU WANT TO WRITE OUTPUT TO A DATA FILE"; ANSS
IF ANSS = "Y" OR ANSS = "y" THEN
  PRINT
  INPUT "ENTER THE FILENAME AND EXTENSION"; FANAMES
  OPEN FANAMES FOR OUTPUT AS #1

  ELSEIF ANSS = "N" OR ANSS = "n" THEN
    CLS
    GOTO 520
  ELSE
    GOTO 515
  END IF

520
'Initialize the variables

V = VENTRY
H = HENTRY

SCREEN 9
WINDOW (-5, 30)-(7, -7)
LINE (0, 0)-(7, 0): LINE (0, 0)-(0, 35)
LOCATE 1, 1: PRINT " ALTITUDE      LANDER DECELERATION AS A FUNCTION OF
ALTITUDE"
LOCATE 22, 1: PRINT "    0.5  1.0  1.5  2.0  2.5  3.0  3.5  4.0  4.5  5.0  5.5  6.0  6.5"
LOCATE 23, 15: PRINT "D E C E L E R A T I O N   G ' S (EARTH)"
LOCATE 3
PCOUNT = 10
LABEL = 30
FOR LABEL1 = 1 TO 7
  PRINT LABEL
  LABEL = LABEL - 5
  PRINT
  PRINT

```

NEXT LABEL1

'Calculations for ballistic re-entry. Assuming constant gravity

DO WHILE H > 0!

IF H > HPD THEN

V = VBAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)

ELSEIF H <= HPD THEN

IF H = 5000 THEN MASS = MASS - 7298

V = VPARA(V, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS, VH, PD, CL, CD)

ELSE

V = V - 24.525 * 100 / VH

END IF

CL = 0

GAMA3 = GAMMA(HS, S, CL, CD, MASS, RHOSL, GAMAE2, H)

VH = V * SIN(GAMA3)

a = ABAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS, V, PD, HPD)

IF ANS\$ = "Y" OR ANS\$ = "y" THEN

PRINT #1, H, a

END IF

HP = H / 1000

AP = a / 9.81

PSET (AP, HP), 10

H = H - 100

LOOP

DO: LOOP WHILE INKEY\$ = ""

SCREEN 0

CLOSE #1

END SUB

FUNCTION GAMMA (HS, S, CL, CD, MASS, RHOSL, GAMMAE2, H)

a = 1 / (2 * HS)

B = S * CD / MASS

GAMMA = ATN(SQR(1 - (COS(GAMMAE2) + a * B * CL / CD * RHOSL * EXP(-H / HS)) ^ 2) /
(COS(GAMMAE2) + a * B * CL / CD * RHOSL * EXP(-H / HS))))

END FUNCTION

SUB HEATRAN (CD, VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS,
MU, CL, HPD, PD)

CLS

11800 PRINT "DO YOU WANT THIS WRITTEN TO A DATA FILE (Y/N)? "

INPUT ANS\$

IF ANS\$ = "Y" OR ANS\$ = "y" THEN

INPUT "ENTER THE FILENAME AND EXTENSION: "; FAMES\$

OPEN FAMES\$ FOR OUTPUT AS #1

ELSEIF ANS\$ = "N" OR ANS\$ = "n" THEN

GOTO 11900

ELSE


```

PRINT "INVALID SELECTION -- TRY AGAIN"
GOTO 11800
END IF

```

```

11900
H = HENTRY
V = VENTRY

```

```

SCREEN 9

```

```

WINDOW (-5, 300)-(350, -15)

```

```

LINE (0, 0)-(350, 0): LINE (0, 0)-(0, 300)

```

```

'LOCATE 1, 1: PRINT " ALTITUDE          LANDER HEATING AS A FUNCTION OF ALTITUDE"

```

```

'LOCATE 22, 1: PRINT "   5  10  15  20  25  30  35  40  45  50  55  60  65"

```

```

LOCATE 23, 30: PRINT "   QR   QC   TS   QR+QC"

```

```

LINE (10, -8)-(20, -8), 15: LINE (50, -8)-(60, -8), 14: LINE (90, -8)-(100, -8), 10: LINE (130, -8)-(140, -8), 13

```

```

LOCATE 3

```

```

PCOUNT = 10

```

```

'LABEL = HENTRY / 1000

```

```

'FOR LABEL1 = 1 TO 7

```

```

'   PRINT LABEL

```

```

'   LABEL = LABEL - 10

```

```

'   PRINT

```

```

'   PRINT

```

```

'NEXT LABEL1

```

```

'Calculations for ballistic re-entry HEATING. Assuming constant gravity

```

```

DO WHILE H > 0!

```

```

IF H > HPD THEN

```

```

V = VBAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)

```

```

ELSEIF H <= HPD THEN

```

```

IF H = 5000 THEN MASS = MASS - 7298

```

```

V = VPARA(V, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS, VH, PD, CL, CD)

```

```

END IF

```

```

CL = 0

```

```

GAMA3 = GAMMA(HS, S, CL, CD, MASS, RHOSL, GAMAE2, H)

```

```

VH = V * SIN(GAMA3)

```

```

IF H > 5000 THEN AREA = 363

```

```

IF H <= 5000 THEN AREA = 70.93

```

```

Cp = 759

```

```

a = SQR(1.32 * 191.46 * TEMP(H))

```

```

M = V / a

```

```

TS = TEMP(H) * (1 + (1.32 - 1) / 2 * M ^ 2)

```

```

TI = 200

```

```

UI = .0000102

```

```

U = UI * ((TEMP(H) / TI) ^ (3 / 2)) * ((TI + 110) / (TEMP(H) + 110))

```

```

' *u = *, u

```

```

D = 21.5
Re = RO * V * D / U
' "Re = ", Re

K = .01
Pr = Cp * U / K
' "Pr = ", Pr

C = .933
NU = U / (RHOSL * EXP(-H / 3216))
RADIUS = 4.06
HC = C * K * (V / NU * RADIUS) ^ .5 * Pr ^ .4
QC = HC * (TS - TEMP(H)) / AREA

EM = .8
TM = (TS + TEMP(H)) / 2
Hr = 4 * EM * 5.67E-08 * (TM) ^ 3
' "Hr = ", Hr
QR = Hr * (TS - TEMP(H)) / AREA

"
' "qR = ", QR
' "qC = ", QC

FORMATS = "#####  #####.##  ###.#  #.#####~  #.#####~"

IF ANSS = "Y" OR ANSS = "y" THEN
  'PRINT USING FORMATS; H; TS; TEMP(H); QC; QR
  PRINT #1, H, QC, QR

END IF

HP = H / 1000
VP = V / 1000
VHP = VH / 1000
QRP = QR / 1000!
QCP = QC / 1!
TSP = TS / 100

'PSET (VP, HP), 10
'PSET (VHP, HP), 13
PSET (QRP, HP), 15
PSET (QCP, HP), 14
PSET (TSP, HP), 10
PSET ((QR + QC) / 1000, HP), 13

H = H - 100

LOOP
DO: LOOP WHILE INKEY$ = ""

```

END SUB

```
SUB INPT (GAMAE, HS, CD, S, MASS, VENTRY, HENTRY, H, RHOSL, G0)
INPUT "ENTER AEROSHELL DRAG COEFFICIENT: "; CD
INPUT "ENTER FRONTAL AREA OF AEROSHELL: "; S
INPUT "ENTER LANDER MASS (at entry): "; MASS
INPUT "ENTER ENTRY ANGLE: "; GAMAE
INPUT "ENTER SCALE HEIGHT OF MARTIAN ATMOSPHERE: "; HS
INPUT "ENTER ATMOSPHERIC ENTRY HEIGHT: "; HENTRY
INPUT "SEA - LEVEL DENSITY: "; RHOSL
INPUT "ENTER ENTRY VELOCITY: "; VENTRY
INPUT "ENTER NEW SEA LEVEL DENSITY: "; G0
```

END SUB

```
FUNCTION TEMP (H)
IF H >= 0 AND H <= 8600 THEN
  TEMP = 212
ELSEIF H > 8630 AND H <= 71000 THEN
  TEMP = -SQR((H - 8000) / 9.844) + 220
ELSE
  TEMP = 140
END IF
```

END FUNCTION

```
FUNCTION VBAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)
```

```
  VBAL = VENTRY * EXP((RHOSL * G0 * HS * EXP(-H / HS)) / (2 * BETA * SIN(GAMAE2)))
```

END FUNCTION

```
FUNCTION VPARA (V, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS, VH, PD, CL, CD)
```

```
  SS = 3.1416 / 4 * PD ^ 2
IF H > 5000 THEN : CDD = 1.58: AREA = S
IF H <= 5000 THEN : CDD = 1!: AREA = 70.93
FP = -.5 * 1.7 * RHOSL * EXP(-H / HS) * V ^ 2 * SS
FA = -.5 * CDD * RHOSL * EXP(-H / HS) * V ^ 2 * AREA
FG = MASS * G0 * SIN(GAMMA(HS, S, CL, CD, MASS, RHOSL, GAMMAE2, H))
  ACCEL = (FP + FA + FG) / MASS
  VPARA = V + ACCEL * 100 / VH
```

END FUNCTION

```
CLS
INPUT "ENTER NEW SKIN FRICTION COEFFICIENT"; CF
GOTO 100
ELSEIF PAR = 5 THEN
CLS
INPUT "ENTER NEW ENTRY ANGLE (in degrees)"; GAMAE
```



```

INPUT "          Enter Selection: "; MODEL
175   IF MODEL = 1 THEN
200 CLS
PRINT TAB(15); "BALLISTIC RE-ENTRY MODEL"
PRINT TAB(15); " _____ "
PRINT

PRINT TAB(15); "ENTER SELECTION --"
PRINT TAB(15);
PRINT TAB(15); "1. PLOT ALTITUDE vs VELOCITY."
PRINT TAB(15); "2. PLOT ALTITUDE vs DECELERATION."
PRINT
PRINT TAB(15); "3. AERODYNAMIC HEATING CALCULATIONS."
PRINT
PRINT TAB(15); "9. CHANGE PARAMETERS."
PRINT TAB(15); "10. RETURN TO MODELING MENU"
PRINT
210 LOCATE 15, 15: INPUT "ENTER CHOICE:(0 TO END) "; CH

    IF CH = 1 THEN
        CALL BALIST(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS,
MU, CL, CD)
    ELSEIF CH = 2 THEN
        CALL DECBAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY,
RADIUS, MU, CL, CD)
        GOTO 200
    ELSEIF CH = 3 THEN
        CALL HEATRAN(CD, VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY,
RADIUS, MU, CL)
        CLS

        QMAX = BHEAT(CF, MASS, S, CD, HS, VENTRY, GAMAE2)
        RHOCRIT = RHEAT(MASS, S, CD, HS, GAMAE2)
        VCRIT = VENTRY / EXP(1 / 3)
        PRINT
        PRINT TAB(15); "AERODYNAMIC HEATING"
        PRINT TAB(15); "-----"
        PRINT
        PRINT
        PRINT
        PRINT
        PRINT TAB(5); " THE SKIN FRICTION COEFFICIENT (CF) : "; CF
        PRINT TAB(5); " MAXIMUM HEATING RATE IS :"; QMAX / 1000000!; " megawatts"
        PRINT TAB(5); " CRITICAL DENSITY IS :"; RHOCRIT; " kg/m^3"
        PRINT TAB(5); " OCCURS AT VELOCITY: "; VCRIT; " m/s"
        PRINT
        PRINT
        PRINT TAB(5); "PRESS ANY KEY TO CONTINUE"
        DO: LOOP WHILE INKEY$ = ""
        GOTO 200

    ELSEIF CH = 9 THEN
        GOTO 100
    ELSEIF CH = 0 THEN

```

```
END
ELSEIF CH = 10 THEN
  GOTO 150
```

```
ELSE
  PRINT "INVALID SELECTION, ENTER NEW CHOICE"
  GOTO 210
END IF
```

```
GOTO 200
```

```
ELSEIF MODEL = 2 THEN
  CALL ASCENT
  GOTO 150
```

```
ELSE
  INPUT "INVALID SELECTION -- ENTER NEW CHOICE: "; MODEL
  GOTO 175
END IF
```

```
FUNCTION ABAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2)
ABAL = (RHOSL * VENTRY ^ 2 * EXP((2 * (RHOSL * G0 * HS) / (2 * BETA * SIN(GAMAE2))) *
EXP(-H / HS)) * EXP(-H / HS)) / (2 * BETA)
END FUNCTION
```

```
SUB ASCENT
```

```
G = 3.72
TH = 1340000
FR = 1181
DK = .5 * RHO * 4.25 ^ 2 * 3.1416 / 4 * 1
H = .1
DTPR = 2
EPSI = .001
MAV = 28767
FMASS = 18000
```

```
2000 CLS
```

```
PRINT "CURRENT ASCENT PARAMETERS:"
PRINT
PRINT "1. Mass of Ascent Vehicle and Fuel: "; MAV; " kg"
PRINT "2. Initial Fuel Mass: "; FMASS; " kg"
PRINT "3. Nominal Engine Thrust: "; TH / 1000; " KN"
PRINT "4. Specific Fuel Consumption: "; FR; " kg/s"
PRINT
PRINT "Numerical Integration Parameters: "
PRINT
PRINT "5. Time Step: "; DTPR; " s"
PRINT "6. Convergence Test: "; EPSI
PRINT
PRINT "9. RETURN TO MODELING MENU."
PRINT
```

```
2010 INPUT "Enter the number of parameter to be changed (0 to accept): "; PAR3
```

```

IF PAR3 = 1 THEN
  CLS
  INPUT "ENTER NEW MASS OF ASCENT VEHICLE AND FUEL: "; MAV
  GOTO 2000
ELSEIF PAR3 = 2 THEN
  CLS
  INPUT "ENTER NEW FUEL MASS: "; FMASS
  GOTO 2000
ELSEIF PAR3 = 3 THEN
  CLS
  INPUT "ENTER NEW THRUST "; F
  GOTO 2000
ELSEIF PAR3 = 4 THEN
  CLS
  INPUT "ENTER NEW SPECIFIC FUEL CONSUMPTION: "; FR
  GOTO 2000
ELSEIF PAR3 = 5 THEN
  CLS
  INPUT "ENTER NEW TIME STEP"; DTPR
  GOTO 2000
ELSEIF PAR3 = 6 THEN
  CLS
  INPUT "ENTER NEW CONVERGENCE TEST: "; EPSI
  GOTO 2000
ELSEIF PAR3 = 0 THEN
  CLS
  PRINT "INVALID SELECTION"
  GOTO 2010
ELSE
  CLS
  PRINT "INVALID SELECTION"
  GOTO 2010
END IF

W1 = MAV * G
WF = FMASS * G
CLS
ANSS = "N"
CLS
1850 PRINT "DO YOU WANT THIS WRITTEN TO A DATA FILE (Y/N)? "
INPUT ANSS
IF ANSS = "Y" OR ANSS = "y" THEN
  INPUT "ENTER THE FILENAME AND EXTENSION: "; FAMES
  OPEN FAMES FOR OUTPUT AS #101
ELSEIF ANSS = "N" OR ANSS = "n" THEN
  GOTO 1950
ELSE
  PRINT "INVALID SELECTION -- TRY AGAIN"
  GOTO 1850
END IF
1950 CLS
PRINT
PRINT TAB(10); "DISPLAY RESULTS AS:"
PRINT TAB(10); "1. TABLE (on screen)"

```

```

PRINT TAB(10); "2. GRAPH"
PRINT
PRINT
1955 LOCATE 6, 10: INPUT "ENTER SELECTION "; PARA
    IF PARA = 1 THEN
        SEL$ = "TABLE"
        GOTO 1970
    ELSEIF PARA = 2 THEN
        SEL$ = "GRAPH"
        GOTO 1970
    ELSE
        PRINT "INVALID SELECTION"
        GOTO 1955
    END IF
END IF

1970
' OPEN "A:rocket.dat" FOR INPUT AS #100
' OPEN "A:RESULTS.DAT" FOR OUTPUT AS #101
' INPUT #100, W1, WF, TH, FR, DK, G, H, DTPR, EPSI, Y, Y1, Y2, Y3, YD, YD1, YD2, YD3,
YDD1, YDD2, YDD3
IF ANS$ = "Y" OR FAMES$ = "Y" THEN PRINT #101, " W    F    TH    R    G    K
H DTP EPSI"
Y = 0
Y1 = .2146 * .3048 * 1.5
Y2 = .8591 * .3048 * 2
Y3 = 1.935 * .3048 * 3
YD = 0
YD1 = 4.294 * .3048 * 1.5
YD2 = 8.598 * .3048 * 2
YD3 = 12.911 * .3048 * 3
YDD1 = 42.99 * .3048 * 1.5
YDD2 = 43.08 * .3048 * 2
YDD3 = 43.18 * .3048 * 3

FRMS$ = "#####.## ##### ##### ##### ##.### #.#### #.## #.##
#####
IF ANS$ = "Y" OR ANS$ = "y" THEN
PRINT #101, USING FRMS$, W1, WF, TH, FR, G, DK, H, DTPR, EPSI
PRINT #101,
PRINT #101, " TIME    DISPL    VEL    ACC"
END IF
IF SEL$ = "TABLE" THEN
PRINT "TIME    DISPL    VEL    ACC"
ELSEIF SEL$ = "GRAPH" THEN
DTPR = .5
SCREEN 9
WINDOW (-10, 70)- (70, -15)
LINE (0, 0)- (70, 0): LINE (0, 0)- (0, 100)
LINE (0, -5)- (15, -5), 9: LINE (25, -5)- (34, -5), 12: LINE (50, -5)- (59, -5), 14
LOCATE 23, 1: LOCATE 23, 21: PRINT "DISP. x E-3": LOCATE 23, 46: PRINT "VEL. x
E-2": LOCATE 23, 71: PRINT "ACCEL."
LOCATE 22, 1: PRINT "    0    10    20    30    40    50    60"
LOCATE 1, 1: PRINT "    ASCENT POSITION, VELOCITY, AND
ACCELERATION vs TIME"

```



```

LOCATE 1, 15
FOR LABELA = 60 TO 0 STEP -10
  PRINT
  PRINT
  PRINT LABELA

  NEXT LABELA
ELSE
  PRINT "ERROR"
  GOTO 1850
END IF

FRM2$ = "###.##   ###.##   ###.##   ###.##"
T = 4 * H
TMAX = WF / FR
TPR = DTPR
2015  W = W1 - FR * T
      YD4P = YD + 4 / 3 * (2 * YDD1 - YDD2 + 2 * YDD3) * H
      RHOAS = .0182 * EXP(-Y4 / 3216)
      DK = .5 * RHOAS * 4.25 ^ 2 * 3.1416 / 4 * .25
      YDD4 = G * TH / W - G - DK * G / W * (YD4P ^ 2)
      IF YDD4 > 29.43 THEN YDD4 = 29.43
2016  YD4C = YD2 + H / 3 * (YDD2 + 4 * YDD3 + YDD4)
      YDD4 = G * TH / W - G - DK * G / W * (YD4C ^ 2)
      IF YDD4 > 29.43 THEN YDD4 = 29.43
      IF (ABS(YD4C - YD4P) - EPSI) <= 0 THEN
        GOTO 2018
      END IF
2017  YD4P = YD4C
      GOTO 2016
2018  Y4 = Y2 + H / 3 * (YD2 + 4 * YD3 + YD4C)
      IF (ABS(T - TPR) - .01) > 0 THEN GOTO 2021

2019  FRM2$ = "#####.##   #####.###   #####.###   #####.###"
2020  IF ANSS$ = "Y" OR ANSS$ = "y" THEN PRINT #101, USING FRM2$, T; Y4; YD4C; YDD4
      IF SEL$ = "TABLE" THEN

        PRINT USING FRM2$, T; Y4; YD4C; YDD4
      ELSEIF SEL$ = "GRAPH" THEN
        PSET (T, Y4 / 1000), 9
        PSET (T, YD4C / 100), 12
        PSET (T, YDD4), 14
      END IF

      TPR = TPR + DTPR
2021  IF (ABS(T - TMAX) - .01) > 0 THEN GOTO 2023
      GOTO 2024
2023  T = T + H
      YDD1 = YDD2
      YDD2 = YDD3

```

```

YDD3 = YDD4
YD = YD1
YD1 = YD2
YD2 = YD3
YD3 = YD4C
Y2 = Y3
Y3 = Y4
GOTO 2015
2024 CLOSE #101
' CLOSE #100

```

```

'PRINT "PRESS ANY KEY TO CONTINUE"
DO: LOOP WHILE INKEY$ = ""
SCREEN 0
GOTO 2000
2050
END SUB

```

```

SUB BALIST (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS, MU,
CL, CD)

```

```

CLS

```

```

'Calculate the VELOCITY as a function of altitude and plot to screen.

```

```

CLS
1800 PRINT "DO YOU WANT THIS WRITTEN TO A DATA FILE (Y/N)? "
INPUT ANS$
IF ANS$ = "Y" OR ANS$ = "y" THEN
    INPUT "ENTER THE FILENAME AND EXTENSION: "; FAMES$
    OPEN FAMES$ FOR OUTPUT AS #1
ELSEIF ANS$ = "N" OR ANS$ = "n" THEN
    GOTO 1900
ELSE
    PRINT "INVALID SELECTION -- TRY AGAIN"
    GOTO 1800
END IF

```

```

1900

```

```

'Initialize the variables

```

```

V = VENTRY
H = HENTRY

```

```

SCREEN 9
WINDOW (-.5, 65)-(.7, -15)
LINE (0, 0)-(.7, 0): LINE (0, 0)-(0, 65)
LOCATE 1, 1: PRINT " ALTITUDE          LANDER VELOCITY AS A FUNCTION OF ALTITUDE"
LOCATE 22, 1: PRINT "    0.5  1.0  1.5  2.0  2.5  3.0  3.5  4.0  4.5  5.0  5.5  6.0  6.5"
LOCATE 23, 30: PRINT "V E L O C I T Y"

```

```

LOCATE 3
PCOUNT = 10
LABEL = HENTRY / 1000
FOR LABEL1 = 1 TO 7
  PRINT LABEL
  LABEL = LABEL - 10
  PRINT
  PRINT

```

```

NEXT LABEL1

```

```

'Calculations for ballistic re-entry. Assuming constant gravity

```

```

  CL = .1
  DO WHILE H > 0!

```

```

    V = VBAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)
    GAM = VSKIP(GAMAE2, H, HS, CL, CD, RHOSL, BETA, G0, VENTRY)
    VS = VENTRY * EXP(-(1 / (CL / CD)) * (GAMAE2 + GAM))
    IF ANSS = "Y" OR ANSS = "y" THEN
      PRINT #1, H, V
    END IF

```

```

    HP = H / 1000
    VP = V / 1000
    VPS = VS / 1000

```

```

    PSET (VP, HP), 10
    PSET (VPS, HP), 11
    H = H - 100

```

```

  LOOP

```

```

DO: LOOP WHILE INKEY$ = ""
SCREEN 0
CLOSE #1

```

```

END SUB

```

```

FUNCTION BHEAT (CF, MASS, S, CD, HS, VENTRY, GAMAE2)
BHEAT = (CF / 6 * EXP(1)) * (MASS / (S * CD)) * HS * VENTRY ^ 3 * SIN(GAMAE2)

```

```

END FUNCTION

```

```

SUB DECBAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS, MU,
CL, CD)

```

```

500
CLS

```

```

'Calculate the VELOCITY as a function of altitude and plot to screen.

```

```

  IF CH = 1 THEN
    CLS

```

```

505 PRINT "DO YOU WANT THIS WRITTEN TO A DATA FILE (Y/N)? "
    INPUT ANSS

```

```

IF ANSS = "Y" OR ANSS = "y" THEN
    INPUT "ENTER THE FILENAME AND EXTENSION: "; FAMES
    OPEN FAMES FOR OUTPUT AS #1
ELSEIF ANSS = "N" OR ANSS = "n" THEN
    GOTO 510
ELSE
    PRINT "INVALID SELECTION -- TRY AGAIN"
    GOTO 505
END IF

510 PRINT "VELOCITY      HEIGHT"
    PRINT "          "
    FMT1$ = "###.##      ###.##"
ELSE

    END IF

515 INPUT "DO YOU WANT TO WRITE OUTPUT TO A DATA FILE"; ANSS
    IF ANSS = "Y" OR ANSS = "y" THEN
        PRINT
        INPUT "ENTER THE FILENAME AND EXTENSION"; FANAMES$
        OPEN FANAMES$ FOR OUTPUT AS #1

        ELSEIF ANSS = "N" OR ANSS = "n" THEN
            CLS
            GOTO 520
        ELSE
            GOTO 515
        END IF

520
'Initialize the variables

V = VENTRY
H = HENTRY

SCREEN 9
WINDOW (-.5, 30)-(7, -7)
LINE (0, 0)-(7, 0): LINE (0, 0)-(0, 35)
LOCATE 1, 1: PRINT " ALTITUDE      LANDER DECELERATION AS A FUNCTION OF
ALTTITUDE"
LOCATE 22, 1: PRINT "   0.5  1.0  1.5  2.0  2.5  3.0  3.5  4.0  4.5  5.0  5.5  6.0  6.5"
LOCATE 23, 15: PRINT "D E C E L E R A T I O N   G ' S (EARTH)"
LOCATE 3
PCOUNT = 10
LABEL = 30
FOR LABEL1 = 1 TO 7
    PRINT LABEL
    LABEL = LABEL - 5
    PRINT
    PRINT
NEXT LABEL1

```

'Calculations for ballistic re-entry. Assuming constant gravity

```
DO WHILE H > 0!
```

```
  A = ABAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2)
```

```
  IF ANS$ = "Y" OR ANS$ = "y" THEN
```

```
    PRINT #1, H, A
```

```
  END IF
```

```
  HP = H / 1000
```

```
  AP = A / 9.81
```

```
  PSET (AP, HP), 10
```

```
  H = H - 100
```

```
LOOP
```

```
DO: LOOP WHILE INKEY$ = ""
```

```
SCREEN 0
```

```
CLOSE #1
```

```
END SUB
```

```
FUNCTION GRAVITY (H, MASS, MU, RADIUS)
```

```
GRAVITY = (MU * MASS) / (H + RADIUS) ^ 2
```

```
END FUNCTION
```

```
SUB HEATRAN (CD, VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, MASS, S, HENTRY, RADIUS,  
MU, CL)
```

```
CLS
```

```
11800 PRINT "DO YOU WANT THIS WRITTEN TO A DATA FILE (Y/N)? "
```

```
INPUT ANS$
```

```
IF ANS$ = "Y" OR ANS$ = "y" THEN
```

```
  INPUT "ENTER THE FILENAME AND EXTENSION: "; FAMES$
```

```
  OPEN FAMES$ FOR OUTPUT AS #1
```

```
ELSEIF ANS$ = "N" OR ANS$ = "n" THEN
```

```
  GOTO 11900
```

```
ELSE
```

```
  PRINT "INVALID SELECTION -- TRY AGAIN"
```

```
  GOTO 11800
```

```
END IF
```

```
11900
```

```
H = HENTRY
```

```
V = VENTRY
```

```
SCREEN 9
```

```
WINDOW (-5, 65)-(70, -15)
```

```

LINE (0, 0)-(70, 0): LINE (0, 0)-(0, 65)
LOCATE 1, 1: PRINT " ALTITUDE          LANDER HEATING AS A FUNCTION OF ALTITUDE"
LOCATE 22, 1: PRINT "   5  10  15  20  25  30  35  40  45  50  55  60  65"
LOCATE 23, 30: PRINT "H E A T  F L U X"
LOCATE 3
PCOUNT = 10
LABEL = HENTRY / 1000
FOR LABEL1 = 1 TO 7
    PRINT LABEL
    LABEL = LABEL - 10
    PRINT
    PRINT
NEXT LABEL1

```

'Calculations for ballistic re-entry HEATING. Assuming constant gravity

```

    VB = VENTRY
    DO WHILE H > 0!
        KP = .5
        RHO2 = RHOSL * EXP(-H / HS)
        V = VBAL(VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)
        QSP0 = KP / SQR(4.06) * SQR(RHO2) * V ^ 3
        PR = 783.14 * 1.28E+07 / .01435
        J = 1
        ALPHA = 1 - 1 / 3 * PR ^ (-.6)
        BET = 1 / HS
        CP2 = 783.14
        MCP = 760
        A = (2 * J * G0 * CP2) / (ALPHA * MCP)
        LP = 1.95E+07
        LAMBDA = A * LP
        KI = 1
        KII = KI

        DMDV = -2 * A * KI * KP / SQR(RHO2) * V / ((LAMBDA + V ^ 2) * (CD * S /
MASS) * SQR(RHO2))
        DMDT = DMDV * (-.5 * CD * S / MASS * RHO2 * V ^ 2 + G0 * SIN(GAMAE2))

        QPA = KII * KP / (BET * SQR(4.06)) * V ^ 2 * RHO2 ^ .5 * 2 / SIN(GAMAE2)
        QTOT = KII * KP / SQR(4.06) * MASS / (CD * S) * .5 * (VB - V) ^ 2 / SQR(RHO2)

        NEWQ = .25 * CF * RHOSL * V ^ 3 * (EXP((-3 * RHOSL * G0 * HS) / (2 * BETA *
SIN(GAMAE2))) * EXP(-H / HS)) * EXP(-H / HS)

CF = .8

        NEWQ = .25 * RHO2 * V ^ 3 * CF
        IF ANSS$ = "Y" OR ANSS$ = "y" THEN
            FORMS$ = "#####  #####.##  #####.###"
            PRINT #1, USING FORMS$, H; V; NEWQ
        END IF

        HP = H / 1000

```

```

QTOTP = QTOT / 1000000
NEWQP = NEWQ / 1000000
PSET (QTOTP, HP), 10
PSET (NEWQP, HP), 13
H = H - 100
VB = V

```

```

LOOP

```

```

SLEEP

```

```

END SUB

```

```

FUNCTION HTAMAX (HS, S, CD, MASS, RHOSL, GAMAE2)
HTAMAX = HS * LOG(-HS * (S * CD / MASS) * (RHOSL / SIN(GAMAE2)))
END FUNCTION

```

```

SUB INPT (GAMAE, HS, CD, S, MASS, VENTRY, HENTRY, H, RHOSL, G0)
INPUT "ENTER AEROSHELL DRAG COEFFICIENT: "; CD
INPUT "ENTER FRONTAL AREA OF AEROSHELL: "; S
INPUT "ENTER LANDER MASS (at entry): "; MASS
INPUT "ENTER ENTRY ANGLE: "; GAMAE
INPUT "ENTER SCALE HEIGHT OF MARTIAN ATMOSPHERE: "; HS
INPUT "ENTER ATMOSPHERIC ENTRY HEIGHT: "; HENTRY
INPUT "SEA - LEVEL DENSITY: "; RHOSL
INPUT "ENTER ENTRY VELOCITY: "; VENTRY
INPUT "ENTER NEW SEA LEVEL DENSITY: "; G0

```

```

END SUB

```

```

FUNCTION RHEAT (MASS, S, CD, HS, GAMAE2)
RHEAT = -(2 / 3) * (MASS / (S * CD)) * HS * SIN(GAMAE2)

```

```

END FUNCTION

```

```

FUNCTION RHO (HS, H, RHOSL)
RHO = RHOSL * EXP(-H / HS)

```

```

END FUNCTION

```

```

FUNCTION VBAL (VENTRY, RHOSL, G0, HS, H, BETA, GAMAE2, S, MASS)

```

```

    VBAL = VENTRY * EXP((RHOSL * G0 * HS * EXP(-H / HS)) / (2 * BETA * SIN(GAMAE2)))
    'IF H >= 10000 THEN VPREV = VBAL
    'IF H <= 10000 THEN
    ' S = 3.1416 / 4 * 20 ^ 2
    ' FNET2 = -.5 * 1.5 * RHOSL * EXP(-H / HS) * VPREV ^ 2 * S + MASS * G0
    ' ACCEL = FNET2 / MASS
    ' VBAL = VPREV + ACCEL
    ' VPREV = VBAL
    'END IF

```

END FUNCTION

FUNCTION VSKIP (GAMAE2, H, HS, CL, CD, RHOSL, BETA, G0, VENTRY)
Y = ((RHOSL * GO * H) / (2 * BETA) * (CL / CD) * EXP(-H / HS) + COS(GAMAE2))
VSKIP = ATN(((1 - Y ^ 2) ^ .5) / (Y))

END FUNCTION

Appendix C1.1.1

Aeroshield Sphere Radius Calculations

$$\frac{D_{A_{LTV}}}{R_{A_{LTV}}} = \frac{D_{A_{TDRG, Mission}}}{R_{A_{TDRG, Mission}}}$$

$$\frac{45.0 \text{ (ft)}}{10 \text{ (ft)}} = \frac{18.3 \text{ (m)}}{R \text{ (m)}}$$

$$R_A = 4.06 \text{ m}$$

$$\frac{D_{P, \text{Viking Mission}}}{W_{P, \text{Viking Mission}}} = \frac{D_{P, \text{LSC Mission}}}{W_{P, \text{LSC Mission}}}$$

$$\frac{16.15 \text{ (m)}}{44 \text{ (kg)}} = \frac{15 \text{ (m)}}{W \text{ (kg)}}$$

$$W_p = 40.87 \text{ kg}$$

APPENDIX D.3-(I) : Calculation of Power System Mass for Descent/Ascent Stages

1. Ag-Zn battery characteristics [Ref.D.3-3]

Nominal voltage	30.0 Volt dc
Minimum voltage	28.0 Volt dc
Maximum voltage	32.5 Volt dc
Weight	61.2 kg/battery
Energy density	175 W-hr/kg for 100% discharge
Maximum DOD	70 % (common value)

2. Power system mass calculations [Ref.D.3-4]

Define: Depth of Discharge (DOD)

$$\text{DOD} = (a * b) / (c * d)$$

where

a = Load power

b = Discharge time

c = Capacity

d = Battery average discharge voltage

From above,

$$\text{Total capacity} = c = (a * b) / (\text{Max DOD} * d)$$

$$\text{Battery capacity} = \text{Total capacity} * d$$

$$\text{Battery Weight} = \text{Battery capacity} / \text{Energy density}$$

a) Descent Stage (2 hrs + 3 days at Martian surface)

Ag-Zn batteries are used for 2 hrs. and Fuel cells are used for 3 days surface stay.

$$\text{Total capacity} = \frac{(17.8 \text{ kW}) * (2 \text{ hrs})}{(0.7) * (30 \text{ Volt})}$$

$$= 1695.2 \text{ A-h}$$

$$\text{Battery capacity} = (1695.2 \text{ A-h}) * (30 \text{ Volt}) = 50856 \text{ W-h}$$

$$\text{Battery weight} = \frac{(50856 \text{ W-h})}{(175 \text{ W-h/kg})}$$

$$= 290.6 \text{ kg}$$

$$\text{Number of batteries} = (290.6 \text{ kg}) / (61.2 \text{ kg/battery})$$

$$= 4.75 \quad 5$$

Fuel Cell characteristics: used in Space Shuttle Orbiter [Ref.D.3-2]

Power output = 7 kW average / cell
Weight = 115.7 kg/cell

To supply 17.8 kW power requirement, 3 Fuel cells with 3 sets of cryogenic reactant storage tanks are used.

Tank weights:
O₂ tank 95.3 kg/tank
H₂ tank 101.9 kg/tank

Radiator weight: 100 kg

Total weight for the Fuel cell system

$$= (115.7 \text{ kg/cell}) * 3 + (95.3 \text{ kg} + 101.9 \text{ kg}) * 3 + 100 \text{ kg} \\ = 1038.7 \text{ kg}$$

$$\text{Total weight for the Descent stage power system} = 290.6 \text{ kg} + 1038.7 \text{ kg} = 1329.3 \text{ kg}$$

b) Ascent Stage (5 hrs for ascent)

Ag-Zn batteries are used for 5 hrs.

$$\text{Total capacity} = \frac{(17.8 \text{ kW}) * (5 \text{ hrs})}{(0.7) * (30 \text{ Volt})} \\ = 4238.1 \text{ A-h}$$

$$\text{Battery capacity} = (4238.1 \text{ A-h}) * (30 \text{ Volt}) = 127143 \text{ W-h}$$

$$\text{Battery weight} = \frac{(127143 \text{ W-h})}{(175 \text{ W-h/kg})} \\ = 726.5 \text{ kg}$$

$$\text{Number of batteries} = (726.5 \text{ kg}) / (61.2 \text{ kg/battery}) \\ = 11.87 \quad 12$$

Total weight for the Ascent stage power system

$$= 726.5 \text{ kg}$$

APPENDIX D.3-(II): Size & Weight calculation for GaAs/CuInSe₂ Solar Array

1. Power-output requirements from solar array [Ref.D.3-5]

The power produced by the solar array needs to fulfill both the daytime power requirements and the excess power required to recharge the electrolyzer.

Power-Output requirement from Solar array = 54.9 kW + 47.4 kW = 102.3 kW

2. CLEFT GaAs/CuInSe₂ Tandem characteristics [Ref.D.3-6]

At low earth orbit, the efficiency is achieved 22% at operation temperature of 28°C.

CLEFT GaAs/CuInSe₂ Tandem : 442 W/kg

3. Photovoltaic Design Values at Mars [Ref.D.3-7, Ref.D.3-8]

Solar intensity on Martian surface	582.8 W/m ²
Operating Temperature at surface	17°C (during day time)
Degradation (6 month)	0.95
Obscuration (Dust/Spectral response)	0.8
Array geometry (for flat panel)	0.53
Inter connection efficiency	0.95
Power conditioning efficiency	0.93
Temperature coefficient	-0.5% per °C
Packing factor	0.9
Structure factor (Simple Deployment)	1.5

4. Solar array Size and Weight calculations [Ref.D.3-4]

Total power (EOL) = 102.3 kW including energy storage

Temperature factor = (17-28) * 0.005 = -0.055

Set

a = Degradation

b = Obscuration

c = Geometry

d = Temperature factor

e = Inter connection eff.

f = Power conditioning eff.

$$\text{Array Capacity (BOL)} = \frac{\text{Total Power (EOL)}}{a * b * c * d * e * f}$$

$$= \frac{102.3 \text{ kW}}{0.95 * 0.8 * 0.53 * (1 - (-0.055)) * 0.95 * 0.93}$$

Array Capacity (BOL) = 272.47 kW

Total cell Area = $\frac{\text{Array Capacity (BOL)}}{\text{Solar intensity} * \text{Array efficiency}}$

$$= \frac{272.47 \text{ kW}}{(582.8 \text{ W/m}^2) * 22\%}$$

$$= 2125.1 \text{ m}^2$$

Array Size = Total cell Area / Packing factor

$$= \frac{2125.1 \text{ m}^2}{0.9}$$

$$= 2361.2 \text{ m}^2$$

Array Weight = $\frac{\text{Array Capacity} * \text{Structure factor}}{\text{Array performance}}$

$$= \frac{272.47 \text{ kW} * 1.5}{442 \text{ W/kg}}$$

$$= 924.7 \text{ kg}$$

APPENDIX D.3-(III): Calculation of Power Requirement of the Propellant Production

Conversion: $2\text{CO}_2 \rightarrow 2\text{CO} + \text{O}_2$

$$G_f^\circ = 394.4 \text{ kJ/mol [Ref.1]}$$

CO_2 : 44 g/mol

Need to produce 18000 kg of propellant

$$\text{Energy requirement} = \frac{(394.4 \text{ kJ/mol}) * (18 * 10^6 \text{ g})}{(44 \text{ g/mol})}$$

$$= 161.3 * 10^6 \text{ kJ}$$

Notice that the sun is available for 12 hr/day.
System operation is during daytime only.

For 4 months propellant production period:

Power requirement =

$$\frac{161.3 * 10^6 \text{ kJ}}{(4 \text{ month}) * (30 \text{ days/month}) * (12 \text{ hr/day}) * (3600 \text{ sec/hr})}$$

$$= 31.2 \text{ kW}$$

Appendix E1: Developmental and constant 1993 total cost analysis of UML in thousands of dollars.

<i>Item</i>	<i>Drafting Development</i>	<i>Design Development</i>	<i>Project Management Development</i>	<i>Manufacturing Prototype Development</i>	<i>Testing Prototype Development</i>	<i>1993 Subtotal (Development and Production)</i>
Truss Structure	\$12,000	\$40,000	\$7,500	\$60,000	\$6,000	\$283,302
GN&C/DMS (hardware)	\$7,500	\$35,000	\$12,000	\$28,000	\$5,000	\$164,133
GN&C/DMS (software)	\$0	\$0	\$0	\$19,000	\$0	\$42,823
CO Fuel Tank	\$5,000	\$24,000	\$5,000	\$36,000	\$2,700	\$150,483
O2 Fuel Tank	\$5,000	\$24,000	\$5,000	\$36,000	\$2,700	\$150,483
Main Engine	\$40,000	\$250,000	\$90,000	\$225,000	\$45,000	\$1,105,374
Secondary Motor	\$3,000	\$50,000	\$4,000	\$50,000	\$8,000	\$221,209
Secondary Motor fuel tank	\$3,000	\$18,000	\$3,500	\$25,000	\$2,500	\$112,076
Aeroshield	\$7,000	\$30,000	\$6,000	\$50,000	\$20,000	\$311,271
Landing Gear	\$3,000	\$10,000	\$3,500	\$25,000	\$5,000	\$119,102
(duplicate of Manned Lander)						\$2,660,259
1 M Module	\$15,000	\$65,000	\$12,000	\$70,000	\$6,000	\$355,968
2 M Module	\$15,000	\$70,000	\$12,000	\$70,000	\$6,000	\$362,659

Appendix E1: Developmental and constant 1993 total cost analysis of UML in thousands of dollars.

3 M Module	\$20,000	\$65,000	\$13,000	\$75,000	\$8,000	\$414,850
4 M Module	\$13,000	\$60,000	\$10,000	\$70,000	\$6,000	\$343,924
1 U Module	\$13,000	\$60,000	\$10,000	\$70,000	\$7,000	\$345,262
2 U Module	\$20,000	\$70,000	\$13,000	\$75,000	\$8,000	\$421,541
3 U Module	\$20,000	\$70,000	\$13,000	\$75,000	\$8,000	\$421,541
4 U Module	\$20,000	\$60,000	\$12,000	\$70,000	\$6,000	\$355,968
SCANS communication network	\$7,500	\$15,000	\$5,000	\$50,000	\$15,000	\$252,925
Earth-based Integration and checkout	\$2,000	\$15,000	\$3,500	\$10,000	\$8,000	\$107,058
					TOTAL	\$8,702,213

Appendix E2: Production cost analysis of UML.

<i>Item</i>	<i>Production Drafting</i>	<i>Production Design</i>	<i>Production Project Management</i>	<i>Testing Production Working Component</i>	<i>Production of Working Component</i>	<i>Quantity of Prototypes</i>	<i>Quantity of Working Component</i>
Truss Structure	\$700	\$3,500	\$7,000	\$15,000	\$60,000	7	2
GN&C/DMS (hardware)	\$750	\$3,400	\$2,500	\$4,500	\$24,000	7	2
GN&C/DMS (software)	\$0	\$0	\$0	\$0	\$13,000	1	1
CO Fuel Tank	\$350	\$1,400	\$3,100	\$4,900	\$30,000	7	2
O2 Fuel Tank	\$350	\$1,400	\$3,100	\$4,900	\$30,000	7	2
Main Engine	\$1,000	\$4,000	\$11,000	\$20,000	\$140,000	18	8
Secondary Engine	\$300	\$1,500	\$3,500	\$5,000	\$40,000	18	8
Secondary engine fuel tank	\$250	\$1,000	\$2,000	\$3,500	\$25,000	7	2
Aeroshield	\$600	\$3,000	\$6,000	\$35,000	\$75,000	4	2
Landing Gear (duplicate of Manned Lander)	\$500	\$2,000	\$5,000	\$10,000	\$25,000	7	19
1 M Module	\$2,000	\$7,000	\$7,000	\$12,000	\$70,000	3	1
2 M Module	\$2,000	\$7,000	\$7,000	\$12,000	\$70,000	3	1

Appendix E2: Production cost analysis of UML.

3 M Module	\$2,000	\$9,000	\$8,000	\$15,000	\$95,000	3	1
4 M Module	\$2,000	\$7,000	\$7,000	\$12,000	\$70,000	3	1
1 U Module	\$2,000	\$7,000	\$7,000	\$12,000	\$70,000	3	1
2 U Module	\$2,000	\$9,000	\$8,000	\$15,000	\$95,000	3	1
3 U Module	\$2,000	\$9,000	\$8,000	\$15,000	\$95,000	3	1
4 U Module	\$2,000	\$7,000	\$7,000	\$12,000	\$70,000	3	1
SCANS communication network	\$2,000	\$3,000	\$4,000	\$7,500	\$80,000	3	4
Earth-based Integration and checkout	\$1,000	\$2,000	\$3,500	\$10,000	\$25,000	1	1