1998

Mars communication network design trade study

William Kok-Fai Tai
San Jose State University

Follow this and additional works at: https://scholarworks.sjsu.edu/etd_theses

Recommended Citation

Tai, William Kok-Fai, "Mars communication network design trade study" (1998). *Master's Theses*. 1777.
DOI: <https://doi.org/10.31979/etd.g3p9-mc69>
https://scholarworks.sjsu.edu/etd_theses/1777

This Thesis is brought to you for free and open access by the Master's Theses and Graduate Research at SJSU ScholarWorks. It has been accepted for inclusion in Master's Theses by an authorized administrator of SJSU ScholarWorks. For more information, please contact scholarworks@sjsu.edu.

INFORMATION TO USERS

This manuscript has been reproduced from the microfilm master. UMI films the text directly from the original or copy submitted. Thus, some thesis and dissertation copies are in typewriter face, while others may be from any type of computer printer.

The quality of this reproduction is dependent upon the quality of the copy submitted. Broken or indistinct print, colored or poor quality illustrations and photographs, print bleedthrough, substandard margins, and improper alignment can adversely affect reproduction.

In the unlikely event that the author did not send UMI a complete manuscript and there are missing pages, these will be noted. Also, if unauthorized copyright material had to be removed, a note will indicate the deletion.

Oversize materials (e.g., maps, drawings, charts) are reproduced by sectioning the original, beginning at the upper left-hand corner and continuing from left to right in equal sections with small overlaps. Each original is also photographed in one exposure and is included in reduced form at the back of the book.

Photographs included in the original manuscript have been reproduced xerographically in this copy. Higher quality 6" x 9" black and white photographic prints are available for any photographs or illustrations appearing in this copy for an additional charge. Contact UMI directly to order.

UMI

A Bell & Howell Information Company
300 North Zeeb Road, Ann Arbor MI 48106-1346 USA
313/761-4700 800/521-0600

Mars Communication Network Design Trade Study

A Thesis

Presented to

The Faculty of the Department of Mechanical and Aerospace Engineering

San José State University

In Partial Fulfillment

of the Requirements for the Degree

Masters of Science in Aerospace Engineering

by

William Kok-Fai Tai

December, 1998

UMI Number: 1392842

**Copyright 1998 by
Tai, William Kok-Fai**

All rights reserved.

**UMI Microform 1392842
Copyright 1999, by UMI Company. All rights reserved.**

**This microform edition is protected against unauthorized
copying under Title 17, United States Code.**

UMI
300 North Zeeb Road
Ann Arbor, MI 48103

© 1998

William Kok-Fai Tai

ALL RIGHTS RESERVED

APPROVED FOR THE DEPARTMENT OF MECHANICAL AND
AEROSPACE ENGINEERING

Dr. Henry J. Pernicka - Associate Professor, Mechanical and Aerospace
Engineering Dept., San José State University

Dr. Richard Desautel - Professor, Mechanical and Aerospace
Engineering Dept., San José State University

Dr. Geoffrey Briggs - Chairman, Center for Mars Exploration,
NASA/AMES Research Center

APPROVED FOR THE UNIVERSITY

ABSTRACT

Mars Communication Network Design Trade Study

By William Kok-Fai Tai

Interest in Mars research has intensified from the announcement of possible biogenic life from meteorite ALH84001 believed to be from Mars. Undoubtedly, with the potentially great demand for Mars spacecraft missions, future Mars exploration and development beckons for a communication infrastructure satisfying data transmission requirements between Mars and Earth. A relay satellite network established as a Mars-based universal communication system could sow the seeds for exploration and future human colonization of Mars.

This thesis describes a method intended as a tool for evaluating candidate satellite communications architects proposed as a possible Mars communications network. This method starts with investigating constellation design configurations based on present technology and past publications of communication network designs. Then constellation configurations were compared using a cost modeling analysis involving the summation of design parameters with an accompanying weighting factor. Varying emphases of each weighting factor were chosen at the users' discretion to determine one or more favorable communication network designs.

ACKNOWLEDGMENTS

To my wife Josephine, who patiently waited and sacrificed throughout my thesis and degree program, to Lockheed Martin (formerly Lockheed) for subsidizing my Masters education throughout this endeavor, to Professor Hank Pernicka and Geoffrey Briggs who have been as patient as Saints and in guidance with my work, to Dr. Desautel who inspired me into pursuing this Masters degree, to teammate Mark Danehy whose contributions have made this thesis possible, to Marcos Villaroman for being a friend and an invaluable resource at a moment's notice, to my parents who always believe in me, and to my nieces Marilyn and Jessica who knew I would finish this Masters thesis, though at times I wondered if I ever would. Yet despite all this work, this work remains my unfinished pursuit of human colonization on the Red Planet.

This work is dedicated to the resourcefulness and self-sufficient spirit of pioneers past, present, and for the pioneers and future civilizations to come.

The author can be reached at willie.tai@lmco.com.

TABLE OF CONTENTS

	Page
List of Tables	viii
List of Figures	xi
Appendices.....	xii
 Chapter	
1. Introduction.....	1
The Need Establish a Dedicated Mars Communication Network.....	10
2. Background in the Cost Modeling Method.....	20
3. Mars Communication Network Design Scenarios	25
The Halo Orbit Communication Network Configuration.....	25
The Aerosynchronous Network Configuration	29
The Common-Period Four Satellite Continuous Mars Coverage Configuration	32
The Multiple Low Mars Orbit Network Configuration	35
The Individually-Dedicated Communication Orbiter-Lander Configuration	37
The Configuration with Martian Landers at the Poles of the Martian Moons	39
4. The Monetary Mission Cost Parameter	44
The Monetary Mission Cost Analysis.....	46
Evaluation and Summary of the Mission Cost Driver.....	67
Total Mission Cost Summary.....	72
5. The Spacecraft Sizing Design Parameter	73
Spacecraft Sizing Driver Summary	128

TABLE OF CONTENTS (Cont.)

	Page
6. The Communications Performance Parameter.....	130
The Baseline Network Communications Design	131
The Communications Link Analysis	137
Communications Availability Analysis	147
Evaluation and Summary of Communications Performance Analysis.....	163
7. The Spacecraft Lifetime Parameter	165
Evaluation and Summary of Spacecraft Lifetime Driver	206
8. The Mission Preparation and Planning Timeline Parameter.....	208
Evaluation and Summary of the Planning Timeline Driver	220
9. Evaluation and Initial Findings from the Cost Modeling Method.....	222
Evaluation of the Individual Parameters in the Cost Model	223
The Evaluation of Multiple Parameters in the Cost Model	229
The Evaluation of the Cost Model using Current Conditions.....	230
Summary of the Cost Model	232
10. Conclusion	234
Bibliography	238
Appendices.....	243

TABLES

	Page
1. Earth and Mars Characteristics Compared.....	8
2. Orbit Parameters of the Draim Four-Satellite Constellation about Mars	33
3. Mars Moons Orbital Characteristics Compared with the Aerosynchronous Orbital Characteristics ..	40
4. Relative Mission Cost Parameters compared among Scenarios.....	46
5. Estimate of On-Board Spacecraft Software Size & Cost - Small Relay Satellite Class	54
6. Estimate of On-Board Spacecraft Software Size & Cost - Large Relay Satellite Class	55
7. Total Space Segment Costs among Scenarios for the Small Spacecraft Class	56
8. Total Space Segment Costs among Scenarios for the Large Spacecraft Class	58
9. Planetary Performance Summary of Various Launch Vehicles Based on C_3 of $13 \text{ km}^2/\text{s}^2$	60
10. Ground Segment Development Costs among Spacecraft Classes	64
11. Operations and Support Cost Per Year and Total Cost for a Five Year Program.....	66
12. Total Mission Cost Compared among Scenarios for the Small Relay Spacecraft Class	70
13. Total Mission Cost Compared among Scenarios for the Large Relay Spacecraft Class	71
14. Factors in the Spacecraft Sizing Parameter are compared among Scenarios	73
15. Relative Minimum ΔV , Minimum Total ΔV , and Minimum C_3 are compared among Scenarios	84
16. ΔV Calculations for an Earth-Mars Trajectory.....	88
17. ΔV Calculations for an Earth-to-Sun-Mars L_1 Trajectory.....	89
18. ΔV Calculations for an Earth-to-Sun-Mars L_2 Trajectory.....	90
19. Summary of the Baseline Spacecraft Mass Breakdown compared among Spacecraft Classes.....	104
20. Small Spacecraft Class Baseline Design Specifications.....	106
21. Large Spacecraft Class Baseline Design Specifications.....	108

TABLES (Cont.)

	Page
22. Generic Parabolic Antenna Mass and Sizing Relationship	111
23. Overview of the Baseline Payload Design Comparisons among Scenarios.....	112
24. Small Satellite Class Payload Sizing among Scenarios.....	113
25. Large Satellite Class Payload Sizing for the Halo Orbiter Scenario	114
26. Large Satellite Class Payload Sizing for the Aerosynchronous Scenario.....	115
27. Large Satellite Class Payload Sizing for the Low Mars Orbit and Medium Mars Orbit Scenario	115
28. Large Satellite Class Payload Sizing for the Common Period Inclined Scenario.....	116
29. Small and Large Satellite Class Structure Sizing are compared among Scenarios	117
30. Small Satellite Class Thermal Sizing are compared among Scenarios.....	118
31. Small Satellite Class Power Sizing are compared among Scenarios	119
32. Large Satellite Class Power Sizing are compared among Scenarios	119
33. Summary of Small Satellite Class Subsystem Parameters among Scenarios	123
34. Summary of Large Satellite Class Subsystem Parameters among Scenarios.....	124
35. High-Level Baseline Communications Design	132
36. The Baseline Communications Design by Service, Defined Data Rate and Bit Error Rate.....	133
37. Transmit Antenna and Receiver Antenna Characteristics	138
38. Summary among Scenarios from Communications Analysis - Small Spacecraft Class Downlink.....	142
39. Summary among Scenarios from Communications Analysis - Small Spacecraft Class Uplink.....	142
40. Summary among Scenarios from Communications Analysis - Large Spacecraft Class Downlink.....	143
41. Summary among Scenarios from Communications Analysis - Large Spacecraft Class Uplink.....	143
42. Mars Surface Coverage Capability compared among Scenarios.....	148

TABLES (Cont.)

	Page
43. Percentage of Communication Visibility Greater than 30 Degree Elevation	154
44. Communications System Configuration and Associated Normalized Rating	157
45. Mission Life Cycle Risk Subfactors compared among Scenarios	166
46. Spacecraft Fuel Requirement over a five-year Mission among the Scenarios (both Classes)	169
47. Battery Recharge Margin for the Small Relay Satellite compared among the Scenarios	173
48. Battery Recharge Margin for the Large Relay Satellite compared among the Scenarios.....	174
49. Quantitative Battery Power Loss Risk Conditions for Both Satellite Classes.....	177
50. Risk from Orbit Altitude for Both Spacecraft Classes compared among the Scenarios.....	179
51. Thermal Power Requirement compared among the Scenarios for both Spacecraft Classes.....	182
52. Fault Tolerance Risk Factors Relating to Constellation Configuration (Both Spacecraft Classes)	191
53. Spacecraft Reliability Evaluated by Spacecraft Subsystem (Both Spacecraft Class).....	194
54. Reliability of Attitude Determination and Control (Both Spacecraft Classes)	196
55. Communications Design Reliability compared among Scenarios	197
56. Reliability of Uplink and Downlink Communications Designs compared among the Scenarios.....	200
57. Electrical Power Subsystem Reliability compared among the Scenarios	203
58. Graceful Degradation Risk Factors compared among Scenarios	206
59. Time-Related Subfactors compared among Scenarios	209
60. Downlink Antenna Design Affecting Network Establishment compared among Scenarios.....	213
61. Launch Vehicle lead-times compared among Scenarios	216
62. Planning Phase Scheduling Factors compared among Scenarios	220

FIGURES

	Page
1. Lagrangian Points, L_1 through L_5 , in the Sun-Mars System	26
2. Forces acting on the Lagrangian Points L_1 through L_2 in the Sun-Mars system.....	28
3. Aerosynchronous relay network consisting of three satellites spaced 120° apart.....	29
4. Common-Period four satellite configuration achieving continuous coverage	32
5. Multiple low Mars orbit network configuration.....	35
6. Traditional dedicated communication orbiter-lander configuration	37
7. Communication landers at the poles of the Martian moons, Phobos and Deimos.....	39
8. Schematic of an interplanetary mission from a low Earth orbit to halo orbit about Sun-Mars L_1 ..	78
9. Positions of the planets relative to the Vernal Equinox (γ) using Orbit Workbench software	80
10. The ΔV and the Departure/Arrival γ of Earth's Departure Obtained Using Vector Addition	86
11. Launch Energies C_3 for Earth to Mars' L_1 , Earth to Mars' L_2 , and Earth to Mars Trajectories ...	92
12. The ΔV comparisons between the Earth to Mars' L_1 , Earth to Mars's L_2 , and the Earth to Mars trajectory at specified launch dates.....	94
13. Total ΔV comparison between the Earth to Mars' L_1 , Earth to Mars's L_2 , and various Earth to Mars trajectories at specified launch dates	96
14. "The Stack" portion of the launch vehicle system.....	100

APPENDICES

	Page
A. Descriptions of Various Restricted Forms of the Three Body Problem	243
B. Cost Estimating Relationships (CER)	245
C. Software Cost Estimation Packages Available to Date.....	250
D. Description and Flowchart of the FORTRAN Program	252
E. Derived Equations in Obtaining the ΔV Budget.....	254
F. Sample Upper Stage Mass Budget Spreadsheets	265
G. FORTRAN Source Code Used in Determining the ΔV Budget.....	268
H. Mars and Earth Positions Relative to the Solar System at Various Times	281
I. Spreadsheet of Baseline Relay Design for Small Spacecraft Class.....	282
J. Spreadsheet of Baseline Relay Design for Large Spacecraft Class.....	286
K. Cost Model Evaluation	296

CHAPTER 1

Introduction: The Search for Past Life on Mars, and the Groundwork For Potential Future Mars Human Colonization

Twenty years after the Viking 1 and 2 landings explored Mars, interest in exploring Mars has intensified from the August 6, 1996 announcement of strong circumstantial evidence of past biogenic activity from meteorite ALH84001 believed to be from Mars.¹ And like the fence that was too high for us to climb in our childhood years, Mars beckons to be explored "because it's there." But what makes Mars more attractive above other bodies in the solar system for exploration is the great potential for human colonization as well as the presence of carbon, hydrogen, nitrogen and oxygen on the Red Planet necessary in sustaining human life. The existence of these elements eliminates the need to import elements from Earth; a potential and substantial savings in fuel and overall mission cost. Interest in Mars also stems from the possibility that Mars is the only other planet

¹ McKay, D., et al, "Search for Past Life on Mars: Possible Relic Biogenic Activity in Martian Meteorite ALH84001," Science 273, no. 5277, (August 16, 1996): 924-930.

besides the Earth in the solar system that is at all likely to have had origins of life. Of course, some technological manipulation is needed to make life on the Red Planet suitable for human habitation. But as proven by the Apollo lunar missions, the technology for human habitation is very possible. And despite the ever apparent tightening of funding for space exploration, venturing into outer space gives us the drive for bigger and better things for the human race. As Walter Kronkite aptly stated in one of his film documentaries on space exploration, "Space: Can We Afford to Go?...The answer appears to be not the case. But the alternative is much worse... WE CANNOT AFFORD NOT TO GO."

Mars - the human race's next achievement and pursuit of a new frontier, and perhaps someday the human race's second home. Future human colonization of the planet Mars is needed for the expansion and prosperity of the human species, for the excelling of our technologies, and for reducing the stagnation in our society. Given Man's accomplishment in landing on the Moon, humans can adapt to life in the Martian environment. But before human colonization on Mars can be considered, the characteristics of the Red Planet will need to be explored and understood. Initial exploration of the planet achieved through Martian orbiters and landers would be the first of many milestones into opening the vast Martian frontier towards a new civilization.

Was there ever life on Mars? Can we practically live on Mars and create a new civilization? Does Mars tell us how the Earth once was or does it reveal telltale signs of a future Earth to come? Are there treasures of life hidden beneath the Martian surface? Some of these questions have been asked since the beginning of astronomical science and will be explored in future missions to the Red Planet. Ultimately these questions and many others concerning the human condition in environments outside that of the "Earth's womb" will need to be answered before a new civilization can be successfully created on Mars. But this new civilization must eventually include the general public, "not just scientists and engineers" as stated by Tom Rogers, president of the Space Transportation Association, in his belief of an American space program.² Ultimately, space travel will need to be available not merely for the elite few but for everyone, and for everyone to experience the opportunity of space travel.

All Mars missions require some form of communication design in relaying data from a point of interest about Mars to a mission control point towards its final destination on Earth. With a Mars-established communication network designed for universal usage in place, succeeding Mars missions could afford a smaller mass and volume allotment for their communication payload designs. Such a Martian communication network would

² Rogers, T., "The People Payload: Does the U. S. Space Transportation Industry Take Space Tourism Seriously?" Ad Astra, March/April 1995: 45-55.

need to satisfy virtually all communication needs including the following requirements first developed by Stevenson:³

Links

- Mars to Earth
- Mars to in-transit spacecraft
- Landers to communication network
- Communication network to orbiting satellites
- Communication within the communication network
- Rovers to communication network
- Mars to Phobos and Deimos
- Among Mars orbiters, landers, and rovers

Services

- Data
 - Science
 - Engineering
- Video

³ Stevenson, S. M., "An Evolutionary Communications Scenario for Mars Exploration," The CASE for Mars III: Strategies for Exploration-Technical, Extract AAS 87-268, vol. 75, (1990).

- Voice for human missions

Functional Requirements

- High-rate data transfer (video and data)
- Voice and lower speed data
- Navigation and precision position location
- Ubiquitous surface coverage
- Continuous time coverage
- Data storage and recording

Thus with the added convenience of an established communication network for Mars exploration, future programs could readily afford either a larger payload design with greater payload mass margin and/or a reduction in overall mission cost. And realization of such a communication network is actually quite possible. E.P. Grondine proposed in his article “Mars on the Cheap” that surplus strategic missiles leftover as a result of the Cold War could be configured to separately launch a communications network into Mars orbit and lander relays onto the surface of Mars.⁴ The traditional option of having a dedicated communication satellite for each orbiter or lander mission has become too expensive in the

⁴ Grondine, E., “Mars on the Cheap,” Space News, 18-24 November 1996: 12.

increasingly competitive aerospace environment. Thus the option of establishing a universal Martian communication network orbiting about Mars is potentially cheaper and becoming a more attractive communication design option. With such a communication network design in operation, future Mars mission designs would require a reduced communications payload on board, potentially resulting in substantial savings in overall mission cost.

Yet another far reaching objective of this communication network is to make commercial space possible and affordable; the more missions that are established, the greater the peaking of public interest, and the greater the possibility of having common people, not just scientists and engineers, experience space exploration and space travel. To include and involve the common people in future space programs is to perpetuate human curiosity, to spark interest in space exploration, and to keep the dream of exploring new worlds alive. Simply ignoring the Common Human in future space missions is a sure ingredient for the demise of future space science.

The intent of this thesis is to perpetuate the pioneering spirit of “One small step for Man, one giant leap for Mankind” in the aerospace community. In this thesis a number of scenarios were investigated using information obtained from various sources, publications, and studies. Each scenario was evaluated with advantages and disadvantages being

determined assuming use of existing technology only. The thesis has been structured to facilitate the evaluation of additional communication designs that may be proposed in the future.

Discovering Mars and Ourselves

Mars, the fourth planet in the solar system from the Sun, is also known as the Red Planet due to its reddish appearance when observed from telescopes on Earth. Table 1 compares Mars planetary characteristics to that of the Earth.⁵ Earth and Mars come into the most advantageous alignment for a Mars launch only once every 26 months. Past Martian missions such as the Mariner and Viking missions revealed a wealth of information, and the stunning success of the Mars Pathfinder has caught the imagination of the American public in the pursuit of space exploration. Yet more characteristics of the Red Planet need to be understood and fundamental technological advances need to be developed before the possibility exists for an interplanetary journey to Mars and the human survival in the Martian environment upon Mars arrival.

⁵ Pernicka, H., lecture notes from Aerospace Engineering 110 course, San José State University. (spring 1994).

Table 1. Earth and Mars Characteristics are compared.

Parameter	Earth	Mars
Sidereal Axial Rotational Period	0.99726 days	1.026 days
Equatorial Radius	6,378.1413 kilometers	3,390.74 kilometers
Gravitational Parameter	$3.98600 \times 10^5 \text{ (km}^3/\text{sec}^2)$	$4.2828 \times 10^4 \text{ (km}^3/\text{sec}^2)$
Semi-major Axis of Orbit	$1.4960 \times 10^8 \text{ km}$	$2.279441736 \times 10^8 \text{ km}$
Orbital Period	1 year	1.8808 year
Eccentricity of Orbit	0.016726	0.093368
Inclination of Orbit to the Ecliptic plane	0.0 degrees	1.84992 degrees

The basic objectives in every mission to Mars involve some form of communication network to relay information back to Earth. The traditional method of an interplanetary mission is to include a dedicated communications payload on board the spacecraft. Thus a portion of the spacecraft mass must be allotted for communications. However with a communications network established at Mars designed specifically for universal use with virtually any Mars program, either a mass savings can be achieved or a greater allotment in payload mass can be accomplished for future Mars missions.

In "The Significance of the Martian Frontier- Mars as the Final Hope for Earth," Robert M. Zubrin poignantly describes the potential benefits and rewards for future human

colonization on Mars.⁶ Zubrin describes three possible avenues for human conquest: under the oceans, lunar colonization, and Mars colonization. Present technology can be used to create a civilization on Mars according to Zubrin. He also outlines advantages of seeking the Martian frontier as opposed to the alternatives of establishing settlements beneath the Earth's oceans or having lunar settlements, although the lunar settlement possibility is now made stronger from recent observations made from the Lunar Prospector spacecraft. Lunar resources may be manipulated as a fueling station for spacecraft traveling to Mars, though lunar colonization may not be possible with the lack of resources needed in sustaining life. Unlike the Moon, Mars does have the minimum number of resources needed in sustaining life: carbon, hydrogen, oxygen, and nitrogen, although the percentages of each element need to be manipulated to suit an Earth-like environment. And on establishing a colony under the Earth's oceans, the main disadvantage of such a colony according to Zubrin is that an ocean colony would not be free from the legal, intellectual, and cultural domination of present-day society that would restrict progress in science. Ideally, Mars is a frontier far enough away from most of Earth's influence and paradigms to allow for free development of a new society, similar to that of Christopher Columbus and the beginning of America. And the Russians have shown that humans can endure a mission to Mars. Russia's Soyuz TM-20 capsule brought

⁶ Zubrin, R. M., "The Significance of the Martian Frontier- Mars as the Final Hope for Earth." Ad Astra, (September/October 1994): 29-37.

three cosmonauts back to Earth in March, 1995.⁷ Cosmonaut-physician Valeri Polyakov spent a record 439 consecutive days in space before landing in Kazakhstan, an endurance record intended to demonstrate humans could withstand the rigors of a voyage to Mars. And this is only the beginning...

The Need to Establish a Dedicated Mars Communication Network

A Mars Relay Satellite (MRS) communication network can provide several advantages to future Mars missions with respect to Mars-Earth communications. One of the advantages is that a MRS can provide a permanent relay capability for future Mars missions. With a MRS in place a reduction of mass and power requirements for individual Mars missions can be afforded due to the convenience of such a data relay network. The distance between Earth and Mars varies between 57 million kilometers (0.38 A.U.), when the Earth passes Mars at its closest distance to the Sun, to nearly 400 million kilometers (2.67 A.U.) when the Earth opposes Mars at its farthest distance from the Sun. This translates into round trip communication times varying from 7 minutes to 40 minutes. The 0.38 A.U. minimum distance between Earth and Mars occurs at 15-year intervals. Also worth noting is that the received power on a communications link decreases proportionally

⁷“Weekly Compendium,” Lockheed Martin corporate weekly publication, 24 March 1995.

to the square of distance; a possible significant loss in link margin may arise at the larger Mars-Earth distance. Even if a Mars mission could be launched to arrive at Mars during periods of relatively low Mars-Earth distance, mission operations may be restricted due to solar interference, also known as solar conjunction. Depending on the orbit design, solar conjunction may occur if the Sun-Mars-Earth angle is on the order of five degrees or less. The establishment of an MRS represents a potential cost savings by reducing the lander mass and power requirements while enhancing data relay connectivity through use of a Mars Relay Satellite (MRS) network for communications.

A MRS communication network enhances communications between Earth and Mars - providing connectivity with elements on the Martian surface that are out of line-of-sight contact with Earth. These elements include the backside of Mars, which are out of communication for a part of every day, and polar elements, which can be out of communication for up to months at a time. A relay network also contributes to telecommunications performance advantages since it can be economically designed to have high data rates with elements on the Red Planet's surface. This permits significant relaxation of the telecommunications performance requirements on surface elements, of which there will potentially be many, and avoids the need to fly and land high mass, high power, and expensive telecommunications systems. It also places a smaller load on the Earth-based tracking resources of the Deep Space Network. Moreover, a relay network

on Mars can be useful as a navigation beacon for other spacecraft approaching the planet. Thus a relay network clearly proves itself to be an invaluable resource towards future Mars missions.

Previous Contributions to Study of Mars Exploration

A myriad of past works in the realm of key trade studies towards future Mars missions have been written. These trades clearly show interest in Mars research and in future space exploration prior to the findings of biogenic life from meteorite ALH 84001 believed to have been from Mars. One such study was an internal NASA report commissioned by NASA administrator Admiral Truly after President Bush's July 20, 1989 speech.⁸ President Bush's speech established a Human Exploration Initiative (HEI) - a long-range effort to reestablish lunar exploration and to continue exploration to Mars and beyond. President Bush's strategy was to first establish Space Station Freedom, next to journey back to the Moon, and then to plan human exploration on Mars.

⁸ Truly, R. H., "Report of the 90-Day Study on Human Exploration of the Moon and Mars," internal NASA report, (November 1989).

Another study discusses the Space Exploration Initiative (SEI) that guides space exploration to the Moon and Mars for America.⁹ SEI contains six visions and are as follows: Knowledge of our Universe; Advancement in Science and Engineering; United States Leadership; Technologies for Earth; Commercialization of Space; and Strengthen the United States Economy. Then the report develops the following four architectures that vary depending on the level of human presence, exploration and science pursuit, and space resource development: Mars Exploration; Science Emphasis for the Moon and Mars; the Moon to Stay and Mars Exploration; and Space Resource Utilization. As also indicated in the NASA 90-Day Study, America's ability to return to the Moon and to begin Mars exploration depends on the fundamental technologies of the restoration of a heavy lift launch capability and the redevelopment of nuclear propulsion capability. The report concluded with the general recommendation that NASA should be authorized to tailor to the existing procurement system and to devise new procedures in effectively managing the SEI.

An ambitious mission concept was the Small Mars Sample Return Mission (SMSR) which consists of four globally distributed landers, each with a return stage along with one or two rovers, and up to four additional sets of lander/rover elements in an extended

⁹ Truly, R. H., "America at the Threshold, Report of the Synthesis Group on America's Space Exploration Initiative," NASA report, U. S. Government Printing Office, Washington, D. C., (May 3, 1991).

mission phase.¹⁰ The sample obtained from Mars is to be returned sealed and not to be exposed to the Earth's environment. Since the biogenic discovery of the Mars meteorite ALH84001, interest in such an SMSR mission has increased with a proposed advance of launching an SMSR mission to a 2001 launch from the previous 2005 launch schedule. The author anticipates human colonization on the Red Planet by the year 2040.

Another proposal for a network of landers on Mars was the Mars Environmental Survey (MESUR) program. MESUR consists a of network of up to 16 globally distributed small landers for the purpose of short term and long term observations of the Martian atmosphere and surface.¹¹ Unfortunately funding for this program of landers has halted in favor of the Mars Pathfinder mission. However the MESUR program has laid some initial groundwork for this study, with the similar goal of this study to have a MRS constellation capable of relaying up to 20 missions simultaneously.

A paper written by a group of four scientists at the Jet Propulsion Laboratory and one from Stanford Telecommunications, Incorporated, discusses orbit design considerations

¹⁰ Hubbard, S. et al., "Concepts for a Small Mars Sample Return Mission Using Microtechnology." Jet Propulsion Laboratory internal document D-8822, Pasadena, CA, (October 1991).

¹¹ Hubbard, S. et al., "Mars Environmental Survey (MESUR) Science Objectives and Mission Description." AMES Research Center study report, Moffett Field, CA (July 19, 1991).

for Mars relay satellite support of globally distributed robotic surface missions.¹² From this paper, a family of near-circular, sun-synchronous, repeat orbits provides a favorable selection for global support by a relay satellite. Consideration of other types of orbit for a mission design may be chosen but may be dependent on the MRS mass delivery into final orbit.

One paper investigates the overall maneuver requirements and describes general characteristics involved in an aerosynchronous orbiter mission.¹³ An important observation is that an aerosynchronous relay satellite will experience a greater east west drift magnitude than that of a geosynchronous satellite. However, the choice of landing sites within moderate Mars latitude from the equator of Mars is possible despite the relatively large east-west drift cycles provided the relay satellite to ground line-of-sight is maintained.

Past works on the topic of satellite sensor-to-surface coverage computation include a description of the Polygon Field of View Technique as a method of analysis of satellite

¹² Hastrup, R. et al. "Mars Relay Satellite Orbit Design Considerations for Global Support of Robotic Surface Missions," AAS Paper No. 93-564, AAS/AIAA Astrodynamics Specialist Conference (16-19 August 1993).

¹³ Palamarik T. et al. "Mission Design for an Aerosynchronous Orbiter," AAS Paper No. 89-197, AAS/AIAA Conference, (1989).

coverage of the Earth.¹⁴ This technique combines the accuracy of the Global Grid Technique with the efficient computer memory usage and computational usage of the Meridian Technique in providing useful and general coverage analysis. Use of this new algorithm allows calculation of cumulative coverage and average revisit times over selected locations on a planet surface. The approach in this study in determining MRS coverage of the Martian surface is partly derived to some degree from the reference of the Polygon Field of View Technique.

Present and Future Mars Missions

A schedule of missions to Mars as of March 6, 1996 showed the following:¹⁵

1996 Mars Global Surveyor, Mars Pathfinder, Mars '96
 1998 Mars Surveyor '98, U.S. Neolander, Planet B (Japan)
 1999 Mars Surveyor '98
 2001 Mars 2001, Mars Surveyor '01 (2 missions)
 2003 Mars Surveyor '03
 2005 Mars Sample Return Mission, Intermarsnet.

¹⁴ Middour J., "An Efficient Technique for Computation of Satellite Earth Coverage," AIAA Paper No. 89-0452, 27th Aerospace Sciences, (July 19, 1988).

¹⁵ Stokley, C., "Mars Mission Schedule", Mars Underground, March 6, 1996: 5.

Since 1996 there has been several events and significant progress. These include the successful launch of the Mars Global Surveyor spacecraft on November 7, 1996, a failed launch of the Russian Mars '96 spacecraft on November 17, 1996, the evening launch of the Mars Pathfinder mission on December 8, 1996, and the successful launch of Japan's Planet B probe. Both the Mars Global Surveyor and the Mars Pathfinder have reached their destinations successfully and are collecting data of the Red Planet. The Mars Global Surveyor is a polar-orbiting probe designed to map the Martian surface. Six of the eight Mars Observer instruments were rebuilt from the original Mars Observer blueprints.¹⁶ Both the Mars Global Surveyor and Mars Pathfinder programs have been completed under their \$150 million cost cap according to Aerospace Daily.¹⁷ And with the startling finding of Meteorite ALH84001 believed to have been from Mars, NASA will most likely plan to launch more Mars missions than the previously plan of a NASA-Mars mission every two years.

More tentative future Mars missions include the French, German, and Russian governments plan to launch the Mars 96-98 program; Japan's Institute of Space and Astronautical Science (ISAS) planned to launch their Planet-B program in 1998 to

¹⁶ Iannotta, B., "NASA Chooses Infrared Sensor", Space News, (July 31-Aug 6, 1994): 1.

¹⁷ Pearson, M., "Scientists Find NASA Well-Positioned to Pursue Mars Life", Aerospace Daily, (Aug 22, 1996): 1.

observe the interaction between the Martian atmosphere and the solar wind.¹⁸ Planning for a Sample Mars Return Mission remains in its early stages, but the present 2005 launch schedule remains tentative despite growing interest in advancing the mission to a 2003 or even 2001 launch opportunity due to the ALH84001 finding.

One more important element to include in future Mars mission design is the direct correlation between power and antenna transmission capability. This idea was addressed by Dr. Alan J. Willoughby of NASA/Lewis Research Center in his presentation of the Data Relay Solar Electric Utility Space Ship (DRSEUSS) at the Mars Forum IV on December 14, 1994.¹⁹ DRSEUSS is a concept being studied at NASA/Lewis Research Center that combines solar electric propulsion, high-rate communications, and high on-board spacecraft power. At present solar power is politically favored over nuclear power for spacecraft design, though radioisotope thermonuclear generators have been successfully used in past missions and should be used in providing power margin. Nevertheless, Mars is still close enough to the Sun for solar power to be practical for spacecraft power. Hence the objective of this concept is to provide solar powered communications relay in Mars orbit thereby enabling high data rates to Earth. With

¹⁸ PLANET-B (FY 1998), <http://www.isas.ac.jp/info/future/planetB-e.html#Tag:0> (1998) website.

¹⁹ Willoughby, A., "Data Relay Solar Electric Utility Space Ship (DRSEUSS)", Mars Forum IV. NASA/Ames Research Center, (12 - 14 December 1994).

DRSEUSS, a 2.5 meter, 300 watt, X-band antenna dish can be used to transmit data between 1.3 Mbps to 500 Mbps, about 50 times the rate over a baseline 1.0 meter, 40 watt, X-band antenna system. An added bonus to this type of communications relay according to Willoughby is its reduced need for the expensive reliance of the Deep Space Network (DSN) in downloading data from the Mars relay vehicle. However, solar electric propulsion has yet to be successfully developed for flight application. And though solar electric propulsion may be a viable option for future space missions, this option as well as options for non-nuclear propulsion (i.e. chemical) is not included in this study.

CHAPTER 2

Background in the Cost Modeling Method

The main focus of this thesis is to develop a quantitative modeling technique with the result of ranking the Martian communication networks based on pre-selected weightings of mission and system design parameters. This analysis involves investigating various options for a MRS communication network. These communication network options are each evaluated based on various performance variables. These parameters are then used as individual terms in a "cost" equation used to evaluate each communication network option. These "costs" includes factors other than monetary costs in the equation. Each of these parametric terms is accompanied by a corresponding weighting factor. Each weighting factor is arbitrarily chosen in order to emphasize the desired factors or to de-emphasize other facts at the user discretion. The "cost" equation in this study has the following form shown in Equation (2-1):

$$J = \sum_{i=1}^n (W_i * n_i) , \quad (2-1)$$

where J represents the total cost, n_i represents the normalized parametric value of each term, and W_i represents the weighting factor for each term based on various design

drivers. Further details of the individual terms and factors that comprise the “cost” equation are discussed in Chapter 9.

One limitation of this cost modeling method is the lack of accuracy in obtaining absolute values for the total cost. However, since the goal of this study is to compare scenarios, absolute accuracy is not required. Consistent evaluations are done where the relative values obtained from the cost equation between competing communication networks can be used for comparison and for determining favorable communication networks.

Hence the parameters used in this cost model include:

1. Mission monetary cost (\$)

- Includes launcher cost, spacecraft bus, mission operations and communications subsystem payload;

2. Spacecraft mission sizing (Size)

- Includes evaluating minimum launch energy (C_3) from Earth to Mars given specified epoch launch and arrival dates, evaluating total ΔV including that for Earth departure, Mars orbit insertion, and stationkeeping; iteratively estimating spacecraft component mass distribution;

3. Communications performance (Comm)

- Examines the constellation configuration of each scenario and the potential total data throughput of each satellite relay network in providing for mission needs. Total data throughput analysis utilizes the combination of the following:

Communications link capability - The highest data rates that can be passed along the link with a given error rate. This capability is dependent upon satellite parameters such as antenna size, transmitter power, and distance to the receiver. These parameters drive such items as solar panel size, spacecraft mass, and launch vehicle selection. A link analysis was performed;

Communications satellite visibility – For a given network the capability at a specified site on the Red Planet of viewing at least one satellite in the constellation at or above a specified elevation. It is another item that is affected by many parameters.

Day and night data transfer rate analysis was performed over a 74 hour period for both the Mars-bound downlink and the Earth-bound uplink;

4. Spacecraft lifetime (Life)

- Includes the analysis of spacecraft fuel, power, thermal requirements, spacecraft reliability as well as the following:

Fault tolerance - The ability of a satellite network to perform its mission objectives after a failure of one or more satellites or satellite components or satellite components during the lifetime of the communication network;

Graceful degradation - The ability of a satellite network to perform its mission objectives as its performance degrades as the satellite ages and parts fail during the spacecraft lifetime; and

5. **Planning timeline (Time)** required to plan, schedule, and establish a relay network through launch but not including the interplanetary cruise phase of the last applicable launch.

Thus the resulting cost modeling equation in this communication network design trade study yields Equation (2-2) below:

$$\mathbf{J} = w_{\$}(\$) + w_{Size}(Size) + w_{Comm}(Comm) + w_{life}(Life) + w_{Plan}(Plan) \quad (2-2)$$

Each of these parameters is discussed in detail in the ensuing chapters following the description of the various Mars communication network scenarios. Then a comprehensive numerical analysis using the above cost equation is performed upon the calculation of each parameter among each scenario in this study. Since the goal of this study was to compare

scenarios, absolute accuracy for each design parameter was not required. However, care was taken to obtain consistent evaluations such that the relative values obtained from the cost equation between competing communication networks could be used for meaningful comparison and for determining favorable communication networks.

The individual cost terms were normalized using the largest value found for each parameter. Then each parameter of this study was compiled using a range of weightings that represented several sample parameter drivers. These weightings were applied to six scenarios investigated in this study to evaluate the effects of different users' priorities on the overall Network design selection.

CHAPTER 3

Mars Communication Network Design Scenarios

This chapter begins with describing the six Mars communication network scenarios that were investigated in this study. Much of the information for each communication network design was obtained from previous work. This thesis does not attempt to cover all possible design scenarios, particularly those using future technologies. Instead, this study focuses more on present technologies readily available and practical for a potential communication network. The following is a description of each of the six design scenarios.

1. The Halo Orbit Communication Network Configuration

The first scenario involves two communications spacecraft orbiting about the libration points L_1 and L_2 of the Sun-Mars system as shown in Figure 1.¹ A single communication satellite would be placed in each of the respective halo orbits about the Sun-Mars L_1 and L_2 points. The L_1 and L_2 points are approximately 1×10^6 kilometers from Mars.

¹ Pernicka, H. et al, "Use of Halo Orbits to Provide a Communication Link between Earth and Mars." AIAA Paper No. 92-4585-CP, AIAA/AAS Astrodynamics Conference, (1992).

Figure 1. (A) Lagrangian points, L_1 , L_2 , L_3 , L_4 , and L_5 , in the Sun-Mars system viewed in the $x - y$ plane.

(B) Halo orbits about the Lagrangian points L_1 and L_2 viewed in the $x - z$ plane.

(C) Halo orbits about the Lagrangian points L_1 and L_2 viewed in the $x - y$ plane.

(Note: orbits are sketches and are not to scale.)

Each satellite would orbit about each libration point and be placed 180° out of phase compared to each other as shown in Figure 1(C), but moving in the same rotational direction about the x -axis. Using this configuration each spacecraft will each view nearly half of the Martian surface. Geometrical calculation shows that the collective network would view up to 99.81 percent of the Mars surface at all times. The remaining 0.19 percent of the planet surface not covered would consist of a thin “band” spanning the circumference of the Red Planet that would oscillate as the satellites move along their respective paths. A Mars mission orbiter or lander that enters this “band” would

experience approximately 1.5 minutes loss of communication between its relay communication satellite on the halo orbit until the mission vehicle becomes visible to the other relay satellite. Thus a favorable constellation trait of near continuous coverage could be obtained with only two satellites. The location of the two-libration points is obtained from the five particular solutions to the elliptic restricted three-body problem found by Lagrange mathematically in 1772. A review of the circular, restricted three-body problem is described in Appendix A.

Halo Orbit Characteristics. From the circular restricted three-body problem (described in Appendix A) the path of motion of the spacecraft (i.e., halo orbit) with infinitesimal mass about L_1 and L_2 in the Sun-Mars system can be determined. Figure 2 traces the orbital path of a libration point, L_1 , in the Sun-Mars system. Figure 2 also shows how the gravitational force of the Sun balances that of the combination of the gravitational force of Mars and the centrifugal force on the mass at the libration point due to the rotation about the Sun (Note: Figure 2 is not to scale). Similar logic applies to the Sun-Mars L_2 , except that with L_2 being on the opposite side of Mars with a larger orbit and with F_{Mars} directed in the opposite direction compared to L_1 .

Figure 2. Forces acting on the Lagrangian points L_1 and L_2 in the Sun-Mars system.

There have been a few missions that successfully traveled on a halo orbit. On November 20, 1978, a scientific spacecraft called the International Sun-Earth Explorer-3 (ISEE-3) was successfully placed into a halo orbit around the Sun-Earth L_1 point.² The Sun-Earth L_1 point is located between the Sun and the Earth and is about 1.5 million kilometers (about 0.1 Astronomical Units (A.U.)) from the Earth. Its mission was to perform a number of particle and field investigations around the Sun-Earth L_1 environment. One important consideration in its orbit mission design was the solar

² Farquhar R. et al, "Trajectories and Orbital Maneuvers for the First Libration-Point Satellite." Journal of Guidance, Control, and Dynamics, vol. 3, (Nov. - Dec. 1980): 549 - 555.

interference factor. Based on the design limits of its S-band antenna, the ISEE-3 mission's halo orbit was designed to pass slightly above and below the ecliptic plane. The halo orbit about the Sun-Earth L_1 has a period of approximately six months and is unstable. The maximum y-axis amplitude of the ISEE-3 halo orbit was almost 700,000 kilometers. This relatively large amplitude halo orbit was preferred since the ΔV requirement for orbit insertion was much smaller resulting in savings of about 150 meters per second. The ΔV requirement for stationkeeping was remarkably low in the range of 10 to 15 meters per second per year. SOHO and WIND missions are currently orbiting the Sun-Earth L_1 point.

2. The Aerosynchronous Network Configuration

Figure 3. Aerosynchronous network consisting of up to three satellites spaced 120° apart.

This aerosynchronous network as illustrated in Figure 3 functions in much the same fashion as geosynchronous satellites presently operating and orbiting about Earth. The only exception is that the aerosynchronous altitude is at 17,040 kilometers, lower than that of the Earth's geosynchronous altitude of 35,786 kilometers due to the smaller mass and weaker gravitational field of Mars compared to Earth. This network consists of three or four satellites in aerosynchronous orbits that are equally spaced with each other. For the three-satellite network, each satellite covers one-third of the collective network surface, or about 120 degrees of surface coverage. An aerosynchronous orbiter would provide a means for a permanent relay capability for future missions. As with the geostationary orbits on Earth, the function of the aerosynchronous orbit is to place a satellite on the equatorial plane of Mars with its mean motion matching that of the planet's rotational rate, thus keeping the satellite subpoint at the same location. An orbital period of one sidereal sol, which is approximately 24.623 hours for a Mars day, requires a semi-major axis of 20,431 km.

A distinct advantage of having satellites in aerosynchronous orbit is that the effects of nodal regression is non-existent which could otherwise reduce the amount of surface coverage as compared to satellites orbiting at an inclined plane relative to Mars' equator. Thus the relative positions between satellite positions can be maintained readily. Visibility overlapping between satellites is desirable to ensure constant line-of-sight visibility of most

of the surface of Mars. Stationkeeping requirements for aerosynchronous satellites are similar to those in geosynchronous orbits with the exception of the higher east-west stationkeeping needed for the aerosynchronous case.³ The higher stationkeeping requirement is due to the stronger triaxiality effects of Mars' gravitational potential as compared to Earth. A slightly greater stationkeeping requirement, on the order of between 10 to 50 meters per second per year, would be necessary in keeping the satellite network equally spaced in the aerosynchronous orbit.

An aerosynchronous network configuration may not be desirable since it lacks polar coverage on Mars. The polar icecaps, containing both frozen carbon dioxide and ice, are visible to Earth telescopes and the size of the polar caps can be seen varying with the changing seasons. These polar ice caps undoubtedly hold significant scientific interest and may hold the key to tapping Mars resources for use in colonization. Future Mars polar missions would need to provide their own communications payload in order to transmit data to Earth from a user in the polar region.

³ Palamarik T. et al. "Mission Design for an Aerosynchronous Orbiter." AAS Paper No. 89-197. AAS/AIAA Conference. (1989).

3. The Common-Period Four Satellite Continuous Mars Coverage Configuration

Figure 4. Common-period four satellite configuration achieving continuous coverage. Portions of each orbit path shaded in bold denote the portion of the orbit being closer to the reader. Note that satellite 1 is positioned closest to the reader in its orbit, satellite 2 being the next closest, satellite 3 is farther away from the reader, and satellite 4 is farthest away from the reader.

This scenario was devised by John E. Draim who developed geometric theorems and corollaries that led to the discovery of four-satellite constellations orbiting Earth that provide 100% continuous global coverage.⁴ This was achieved using elliptic orbits of the same period but strategically placed in different orbit planes to provide continuous global coverage. Figure 4 shows a schematic of Draim's continuous coverage, common-period

⁴ Draim, J. E., "Continuous Global N-Tuple Coverage with $(2N+2)$ Satellites," Journal of Guidance and Control 14, no. 1, (Jan.-Feb. 1991): 1.

four-satellite configuration. Draim's work was applied to Earth-based systems, but his work can be applied to Mars-based systems with some modifications.

The orbital parameters or ephemerides of this four-satellite continuous coverage network about Mars are given in Table 2 below.⁵ Note that the orbital parameters originally designed for Earth-based satellites are applied to a Mars-based network with one change accounting for the slight difference in semi-major axis of the Mars orbit of 42,560 kilometers compared to that of Earth from Draim's paper of 45,120 kilometers.^{6,7}

Table 2. Orbit Parameters of the Draim Four-Satellite Constellation about Mars.

Satellite Number	i, degrees	eccentricity	ω , degrees	Ω , degrees	M, degrees
1	31.3	0.263	270	0	0
2	31.3	0.263	90	90	270
3	31.3	0.263	270	180	180
4	31.3	0.263	90	270	90
a = semimajor axis; a = approximately 42,560 km for 74 hour orbit about Mars, i = inclination, ω = argument of periapse, Ω = right ascension of ascending node, M = mean anomaly.					

⁵ Draim, J. E., "Continuous Global N-Tuple Coverage with (2N+2) Satellites."

⁶ Draim, J. E., "A Common-Period Four-Satellite Continuous Global Coverage Constellation." Journal of Guidance, Control and Dynamics 10, no. 5, (Sept. - Oct. 1987): 492 - 499.

⁷ Danehy, M., "Martian Communications Network Design Trade Study." Masters Thesis. San José State University. (December 1997).

Draim devised four corollaries and theorems, which describe conditions for complete global coverage. In essence, a planar region is imagined containing three satellites being either tangent to the Mars surface at some point or being above the tangent point. This region is in a shape of a triangle with each of the three satellites marking the corners of the triangle. The triangle region is projected to the Mars surface, thus creating a spherical triangle on the surface of Mars. The region within the spherical triangle on the surface of Mars is made visible to at least one of the three satellites.

This same principle is then applied to four satellites creating the shape of a tetrahedron, a four-sided polyhedron, with Mars completely enclosed within the tetrahedron. The tetrahedron enclosing Mars has a satellite in each of the four corners, and as long as the sides of the imaginary tetrahedron are either tangent to the surface of Mars or above, then any point on the Mars surface is visible from at least one satellite. The tetrahedron is not equilateral in shape, and the tetrahedron changes shape or warp as the satellite constellation travels through its orbital period. Draim derived this constellation geometry from this tetrahedron description. This scenario has an advantage of establishing a constellation with total surface coverage of Mars at any time. However, in order to achieve and maintain this total surface coverage, each satellite requires periodic stationkeeping maneuvers until the propellant on board each spacecraft is expended.

4. The Multiple Low Mars Orbit Network Configuration

Figure 5. Multiple low Mars orbit network configuration.

In this scenario, a series of Mars low orbit communication satellites would be established as shown in Figure 5. The number of relay satellites in this constellation can vary. But in general, the higher the number of satellites in low Mars orbit, the greater the long-term coverage performance of the network. In this study, a six-satellite constellation with orbiting altitude in the 300-kilometer range is studied.

The level of long-term revisit coverage provided by a multi-satellite constellation in a circular, repeating ground track orbit can be evaluated analytically. Hopkins describes such a technique for calculating efficient long-term revisit coverage of Earth-orbiting

constellations.⁸ Hopkins' article showed this technique applied to two and three satellite constellations examining the maximum spacing between adjacent satellite ground tracks as a function of time. In general, over a wide range of latitudes, Hopkins' technique provides coverage results in equally spaced ascending and descending satellite ground tracks at the equator at the end of the repeat interval. At higher latitudes, the separation between adjacent ground tracks will be minimized. The time between satellite revisits of geographical locations would be reduced.

This multi-satellite network configuration is presently used and proposed in various communications networks on Earth, specifically in the realm of the wireless telephone and location determination systems. Presently, examples of multiple satellite networks include position-determination networks such as the Global Positioning System (GPS) of 24 satellites and Russia's counterpart of the Global Navigational Satellite System (GLONASS). The GPS constellation is comprised of 24 satellites, four in each of six orbital planes. In comparison, the GLONASS satellites fly at 19,100 km above the Earth at inclinations 64.8° performing a precise eight-day repeating ground-track. On of the latest three new GLONASS satellites launched on July 24, 1995 designated Cosmos 2316, 2317, and 2318, nearly complete Russia's space-based navigation system. Russia at that

⁸ Hopkins, R., "Long-Term Revisit Coverage Using Multi-Satellite Constellations," AIAA Paper No. 88-4276, AIAA/AAS Astrodynamics Conference, Minneapolis, MN, (August 15-17, 1988).

time was one launch away from completing its GLONASS constellation.⁹ The complete GLONASS constellation is comprised of 24 satellites, 21 operational and three spares, with three orbital-plane groups of eight satellites.

5. The Individually-Dedicated Communication Orbiter - Lander Configuration

Figure 6. Traditional dedicated communication orbiter-lander configuration.

Figure 6 shows the scheme of a traditional dedicated communication orbiter-lander. This configuration has been the traditional approach in planetary exploration. To date, Mars-bound spacecraft have used either dedicated on-board equipment to communicate with Earth-based communications, such as with the Mars Global Surveyor mission, or used data relayed to an orbiter dedicated to the mission, such as the case of the Viking.

⁹ Covault, C., "Russian InterBall Mission Launched," Aviation Week & Space Technology, (August 14, 1995): 64.

Such designs served well in the past. However, the cost will be a significant factor if there are to be multiple missions.

This scenario has long been the standard philosophy of mission operations. Initially in the beginning of the space program, congressional-controlled budgets were relatively tight. But when President Kennedy made his bold statement of "putting man on the Moon before [the 1960s] decade is out" in his State of the Union Address in 1961, more funding was allocated and money was less of an object. Dedicated communication spacecraft designs were favored, eliminating problems of scheduling communication times when two or more satellites are to share the same communication link. This scenario may be the scenario of choice for the Mars mission pessimist's point of view that future Mars's missions would be few and far between. However, with the ever-shrinking aerospace budget at present, this may not be a viable option particularly with the optimistic future of multiple planetary missions. Moreover, another drawback of the dedicated orbiter-lander is that if either lander or orbiter in the overall mission fails that part of the mission is lost unless a mission replacement is performed. This example is clearly demonstrated in the catastrophic loss of the infamous Mars Observer mission.

Multiple missions using the MRS network in operation instead of only using one dedicated mission would not only decrease the risk of total loss of a mission, but time and

effort would not be lost in recovering from a lost mission. The knowledge of the Mars characteristics that would have been gained from the Mars Observer in 1993 would not be achieved until the Mars Surveyor mission presently collecting environmental data of the Red Planet. With a MRS in place about Mars, future missions could be done cheaply and with greater frequency, increasing the fault tolerance of overall data collection of Mars. For the purpose of this study, this scenario has a 20-degree inclined orbit with an altitude in the 1000-kilometer range and will not have communication capabilities to more than one mission.

6. The Configuration with Martian Landers at the Poles of the Martian Moons, Phobos and Deimos

Figure 7. Communication landers at the poles of the Martian Moons, Phobos and Deimos shown.

The two moons of Mars, Phobos and Deimos, orbit Mars in a gravity-gradient stabilized orientation, such that both Martian moons keep the same face toward Mars much like the behavior of the Earth's Moon as shown in Figure 7. The Russians have

spent considerable effort in establishing a science mission to explore these moons. The orbits of both Martian moons lie very close to the equatorial plane of Mars. Table 3 shows the characteristics of the orbits of Phobos and Deimos.

Table 3. Mars Moons Orbital Characteristics Compared with Aerosynchronous Orbital Characteristics.¹⁰

	Aerosynchronous	Phobos	Deimos
Diameter		27 x 19 kilometers	15 x 11 kilometers
Orbital Period	24 hours 38 minutes	7 hours 39 minutes	30 hours 17 minutes
Orbital Radius	20,431 kilometers	9,380 kilometers	23,449 kilometers

Phobos, the inner satellite, circles Mars over three times a Martian day (sol). The outer satellite, Deimos, orbits Mars slightly less than once a sol. The two lunar orbits straddle the synchronous-orbit-distance of 20,431 kilometers for Mars, the distance at which a satellite has an orbital period exactly equal to the spin period of the Red Planet. Hence, the characteristics of the two lunar orbits would evolve differently over time.¹¹ The influence of the strong triaxiality behavior of the gravitational potential of Mars is very gradually decelerating Deimos and pushing it slowly away from Mars, while the inner

¹⁰ Hamilton, C.. "Phobos 1 and 2 to Mars." NASA National Space Science Data Center. Washington, D. C. (1994).

¹¹ Veverka, J.. "The Moons of Mars". AAS Paper 86-157. NASA Mars Conference. National Academy of Sciences. Washington, D.C.. (July 21-23, 1986): 341.

satellite Phobos is being accelerated and pulled closer to the planet. The rate of orbital decay of Phobos is detectable, and estimates reveal about 30 million years before it either spirals in to be broken up by increasing tidal forces or until it impacts the planet.

With the existence of these two moons, landers could be placed on either or both of the moons for Martian surface communication and observation, preferably at the poles. Placing the communication landers at the poles of the Martian moons eliminates the problem of losing Mars and Earth communications when the landers are on the far side of the moons if the landers are placed at latitudes other than at the poles. This configuration has the advantage of eliminating the need for spacecraft stationkeeping since the moons provide regular orbits about Mars. A similar scenario has been attempted unsuccessfully by the Russian probes PHOBOS 1 and PHOBOS 2, identical programs which were intended to be an orbiter to each send a hopper/lander to Phobos. PHOBOS 1 was accidentally turned off by a ground controller on August 29, 1988 and is presently inoperative in solar orbit. The fate of PHOBOS 2 was no better; radio contact was lost on March 27, 1989 and was unable to drop its hopper lander on Phobos on April 1989.¹²

A major difficulty of this type of mission is the extremely low escape velocities that each of the moons exhibit due to their low masses that are on the order of 10 meters per

¹² Hamilton, C.. "Phobos 1 and 2 to Mars."

second. Also due to their small sizes, very precise trajectories and fine navigation are needed to successfully place a spacecraft on the Martian moons. However, with the proximity and regularity of both moons orbiting about Mars either or both moons could make ideal space stations in studying the Red Planet, as well as studying the moons themselves.

Parameters in the Cost Equation

The process of developing a design trade study may be as detailed or as simple as the designer chooses and is an iterative process. Many of the parameters used in a trade study are interrelated to other factors used in this study. Despite the potential complexity of a design trade study, a design trade study is beneficial in formulating a mission plan and provide insights to details that otherwise could be overlooked and may prove costly in the long term.

A total of six representative scenarios in this study are discussed. With these scenarios established, chapters 4 through 8 focus on the individual parameters that comprise the Cost Equation. The values of each parameter are derived from subfactors with their own relative values that make up the parameter of interest. The relative values and derivation

of each subfactor are further detailed in chapters 4 through 8. Each chapter from Chapter 4 through Chapter 8 that focuses on the individual parameters will have the cost equation shown below the chapter title with the specific parameter of interest highlighted in bold print. Then the values of each parameter are weighted and the process is repeated with the other parameters towards evaluation of the Cost Model. The derivation of the weighting values and a summary of the preliminary findings from the cost equation and in determining favorable scenarios are discussed in Chapter 9.

CHAPTER 4

The Monetary Mission Cost Parameter

$$J = w_{\$}(\$) + w_{\text{Size}}(\text{Size}) + w_{\text{Comm}}(\text{Comm}) + w_{\text{Life}}(\text{Life}) + w_{\text{Plan}}(\text{Plan})$$

In this chapter the overall monetary mission cost parameter of the cost equation is studied and compared among the scenarios. Mission cost is perhaps one of the most if not the most important mission driver for any space program. But even though mission cost may appear to be a significant design variable at the present time, mission cost may not necessarily be the most and only significant overall cost driver (the "cost" as defined in Equation 3-1). In this study the mission cost or cost-modeling term of the "cost" equation is achieved through the parametric cost estimation process.¹ First an organizational table called the Cost Breakdown Structure (CBS) is used in organizing cost information and maintaining consistency in comparing the various scenarios. Then the cost model is developed using the parametric estimating method. Parametric estimating uses an equation called the Cost Estimating Relationship (CER) that expresses the cost as a function of design sizing and performance variables or parameters. In this study, cost versus risk data, events and milestones are omitted. Also worth noting is that absolute

¹ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 2nd ed. (Torrance, CA: Microcosm Press, and The Netherlands: Kluwer Academic Publishers, 1992): 715-736.

cost accuracy is not as important in this study as obtaining relative cost comparisons for each scenario listed.

Table 4 shows a comparison of the following four general cost parameters which combined comprises of the total mission cost: the launch vehicle selection, the space segment cost which includes spacecraft-related costs, the ground station operational costs, and the software developmental costs compared among the scenarios developed in this study. This table was developed by the author with the intent of using the table for comparison purposes rather than attempting to provide absolute values, which is beyond the scope of this study. A relative estimate of the cost parameters among the scenarios is shown in Table 4. The cost parameters include program level, space segment, launch segment, ground segment, and operations and support. Space segment costs involves system level, software development, spacecraft bus and communications payload costs, as well as program level costs involving management and systems engineering and integration costs. Launch segment costs includes launch vehicle and launch operations costs. The ground segment costs include facilities, equipment, software, logistics, and its management and systems engineering and integration costs. And operations and support costs include personnel, training, maintenance costs, and costs in obtaining spares and redundant operation. Numerical evaluation of each segment costs are discussed further in this chapter.

Table 4. Relative Mission Cost Parameters compared among Scenarios.

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Space Segment	low	medium	medium	large	low	low
Launch Vehicle	medium	large	large	large	low	medium
Ground Segment	low	medium	medium	large	low	medium
Operations and Support	low	medium	medium	large	low	medium
Total Mission	low	medium	medium	large	low	low

The Monetary Mission Cost Analysis

There are three basic techniques used to develop the mission cost term of the cost model equation. They include the following:

1. **Detailed bottom-up estimating**, which identifies and specifies a lower level of elements making up the system. Then the cost of materials and labor to develop and produce each element is estimated;
2. **Analogous estimating**, which uses the cost of a similar item and adjusts for differences in size or complexity;
3. **Parametric estimating**, which uses an equation called the Cost Estimating Relationship (CER) that expresses the cost as a function of various factors such as design

sizing, performance variables or other parameters. Complexity factors may be included into the various parameters to account for technological advances. In parametric modeling a Cost Estimate Relationship for the hardware RDT&E and theoretical first unit cost is needed. For the network design scenarios with multiple units, the total production cost would be equal to the first unit cost multiplied by a learning curve factor.

The detailed bottom-up estimating method is least appropriate for advanced system studies and would be beyond the scope of this study. The analogous estimating method is inflexible for use in this study since the luxury of having "similar items" is not always available. Of the above three options in developing the mission cost term, the parametric estimating method will be the method of choice in this study and is largely adapted from Wertz and Larson's Space Mission Analysis and Design textbook.²

The parametric estimating method described in Wertz and Larson involves a Cost Breakdown Structure (CBS) covering the following systems engineering analysis:

- Research, development, test and evaluation (RDT&E),
- Production:
 - Launch segment costs
 - Ground segment costs

² Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 715-736.

- Operations & support costs

In this study a simplified CBS is used which is comprised of the following:

- Spacecraft mass budget estimate - Two spacecraft classes serve among the scenarios as a baseline for the spacecraft mass budget estimate in this study, the large spacecraft class and small spacecraft class, discussed in detail in Chapter 5;
- Launch vehicle selection - the various launch vehicles presently available which can support each scenario and spacecraft class;
- Operational costs associated with ground segment costs; and
- Software development costs based on MIL-STD 1750A architecture.

The simplified CBS used in this study based on the CBS will be applied to the dollar-cost term of the cost modeling equation in determining relative values among scenarios and among relay spacecraft classes in this study.

Appendix B shows the CERs for both the small and large spacecraft classes. The CERs show the budget estimate of the respective spacecraft classes including launch vehicle costs, space segment cost, spacecraft software development cost, ground software cost, operational cost. The costs obtained are in fiscal year 1996 (FY96) dollars. These costs are evaluated towards total mission program cost and for comparison among

scenarios. An analysis of the various segment costs beginning with space segment costs and software development associated with the space segment is discussed below.

Space Segment Costs

A preliminary analysis of the space segment costs is developed to determine the cost budget estimates among the scenarios as well as the relay spacecraft classes considered in this study. Space segment cost parameters in this estimate include the spacecraft communications payload, the spacecraft bus, the apogee kick motor if applicable, aerospace ground equipment, and program level costs. Cost values for spacecraft subsystem mass and launch vehicle selection were obtained from Wertz, JPL, and several internet website references, and details of the spacecraft mass derivation are discussed in further detail in Chapter 5 of this study. From the information obtained in determining spacecraft sizing for the small spacecraft class and large spacecraft class individually, the cost values obtained are shown in succeeding tables later in this chapter. First, tables of launch vehicle cost are displayed based on a required Earth departure energy C_3 of $13 \text{ km}^2/\text{s}^2$, followed by itemized space segment cost data.

Generally, the total space segment cost is based on the cost of a complete scenario which includes research, development, test and evaluation costs (RDT&E), spacecraft software development costs, the complete spacecraft including the communications payload and spacecraft bus, and if applicable spacecraft spares. Spacecraft spares are not included in this study but may be considered in future studies. The elements comprising the space segment costs are in Fiscal Year 1996 millions of dollars (FY96\$M).

Software Development Costs

Typically at the beginning of a space program a Systems Requirement Review (SRR) is started where a formal review of computer resource requirements is first established.³ A general rule of thumb would be to set the amount of computer memory and throughput at four times the estimate for software size and time at SRR. Empirically, initial software size and throughput estimates double from the review to launch for at least the following two reasons: (1) early requirements are uncertain; (2) changes in software are easier than in hardware during late stages of spacecraft development. Also some reserve in the form of spare memory and throughput at launch is desired to correct anomalies or to increase performance after system calibration- establish reserve capacity when initially defining

³ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design: 715-736.

requirements. An upper bound is 100% spare (50% of machine capacity) for post-launch reserve.

Software development costs are largely spacecraft class driven rather than scenario driven. In the small satellite class, sufficient pointing accuracy on the order of two degrees peak-to-peak is assumed for acceptable relay communication. The small spacecraft is spin-stabilized using a V-slit star sensor and a V-slit sun sensor for attitude determination. Two thruster quads via a low fidelity logic loop control spacecraft attitude and hydrazine fuel slosh is used for nutation dampening. No reaction wheels are used in the small class design.

For the large satellite class design, the relay satellite is three-axis stabilized in providing high pointing accuracy on the order of several arcseconds required for relay communication. The attitude control system is to have both a high-fidelity inner logic loop for reaction wheel control and a lower-fidelity outer logic loop for thruster control.

Spacecraft Software Cost

For the software development cost, a share of the cost would be based on the use of Commercial Off-The-Shelf (COTS) software packages developed by numerous software vendors. The COTS approach in establishing mission operation and design has become a popular trend since the recession years of the early 1990s in the aerospace industry's new motto of "smaller, cheaper, faster". Using COTS software packages reduces the need to develop custom software in mission design thereby driving software costs down and reducing software development time. Software development costs will vary depending on the complexity of the mission and these costs are included in the RDT&E portion of the total space segment cost. However, in this study an assumption for the software development is that costs are driven by spacecraft class rather than by network scenario. For this reason the software development costs will be similar among the scenarios for the large spacecraft class and will also be similar among the scenarios for the small spacecraft class. The software costs for the small spacecraft class and for the large spacecraft class are studied individually.

Further assumptions are used in this study, which apply to both satellite classes. Of the two popular machine code interfaces or Instruction Set Architectures (ISA) available in satellite software design, the MIL-STD 1750A architecture will be used for convenience in

this study over the 80x86-based architecture. Also per MIL-STD guidelines, a 50% spare capacity is to be available in the target software processor at launch. The satellite software language used in estimating software development cost is based on the C-UNIX software language. Also with the selected software language, the development cost is based on memory words with 16-bit length and eight words of memory per C-UNIX source lines of code (SLOC); future studies may consider software development cost based on 32-bit memory words. With the general assumptions established for both satellite classes, some additional assumptions are made for the small satellite class. Table 5 and Table 6 shows the estimate of the on-board software size with KLOC units and cost applied to the small satellite class and large satellite class respectively. Tables 5 and 6 are derived from Wertz and Larson's Space Mission Analysis and Design Workbook.⁴

⁴ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design Workbook. (Torrance, CA: Microcosm Press, 1993), 41.

Table 5. Estimate of On-Board Software Size and Cost - Small Relay Satellite Class. The values for software size and cost are based on 16-bit words and a 1750A-class ISA with software development in C-UNIX.

Component	Sampling Rate Frequency (Hz)	Required Source Line of Memory Code (KLOC)	Cost (FY 96 \$ M) = 0.62625 x KLOC
Application Functions			
Communications			
• Command Processing	10.0	0.125	0.078
• Telemetry Processing	10.0	0.125	0.078
Sun Sensor	1.0	0.06	0.04
Star Sensor	1.0	0.25	0.16
Precession (thruster)	10.0	0.41	0.26
Control			
Simple Autonomy	1.0	0.25	0.16
Fault Detection			
• Monitors	2.0	0.5	0.31
• Fault Correction	2.0	0.25	0.16
(a) Subtotal Application		1.98	1.24
Operating System			
Local Executive		0.44	0.27
Runtime Kernal (COTS)		1.0	0.63
I/O Handlers (5)		0.25	0.16
BIT and Diagnostics		0.09	0.055
Utilities		0.15	0.094
(b) Subtotal: COTS		1.0	0.63
(c) Subtotal: Non-COTS		0.93	0.58
(d) Operating System		1.93	1.20
Subtotal: (b) + (c)			
Margin Calculations			
(e) Needed to compensate for requirements uncertainty	100% of non-COTS: (a) + (c)	2.91	1.82
(f) On-orbit spare	(a) + (d) + (e)	6.82	4.27
Estimate of spacecraft computer requirements	(a) + (e) + (f)	11.71	7.33

Table 6. Estimate of On-Board Spacecraft Software Size and Cost - Large Relay Satellite Class. The values for software size and cost are based on 16-bit words and a 1750A-class ISA with software development in C-UNIX.

Component	Sampling Rate Frequency (Hz) / Comments	Required Source Line of Memory Code (KLOC)	Cost (FY96\$ M) = 0.62625 x KLOC
Application Functions			
Communications			
• Command Processing	10.0	0.125	0.078
• Telemetry Processing	10.0	0.125	0.078
Horizon Sensor	10.0	0.1875	0.117
Sun Sensor	1.0	0.0625	0.039
Star Sensor	0.5	0.25	0.156
Kinematic Integration	10.0	0.25	0.156
Error Determination	10.0	0.125	0.078
Thruster Control	2.0	0.045	0.028
Reaction Wheel Control	2.0	0.125	0.078
Ephemeris Propagation	1.0	0.25	0.156
Orbit Propagation	1.0	1.625	1.018
Complex Autonomy	10.0	1.875	1.17
Fault Detection			
• Monitors	5.0	0.5	0.31
• Fault Correction	5.0	0.25	0.156
Power Management	1.0	0.15	0.094
Thermal Control	0.1	0.1	0.063
(a) Subtotal Application		6.047	3.78
Operating System			
Local Executive		0.44	0.27
Runtime Kernal (COTS)		1.0	0.63
I/O Handlers (5)		0.25	0.16
BIT and Diagnostics		0.09	0.055
Utilities		0.15	0.094
(b) Subtotal: COTS		1.0	0.63
(c) Subtotal: Non-COTS		0.93	0.58
(d) Operating System Subtotal: (b) + (c)		1.93	1.20
Margin Calculations			
(e) Needed to compensate for requirements uncertainty	100% of non-COTS: (a) + (c)	6.98	4.37
(f) On-orbit spare	(a) + (d) + (e)	14.95	9.36
Estimate of spacecraft computer requirements	(a) + (e) + (f)	27.98	17.52

Another option available in evaluating operational costs is the Mission Operations Cost Model (MOCM) developed by NASA's Cost Estimation Group with various areas of cost estimation available on-line via the Internet.⁵ This is a cost model that provides a rough order of magnitude cost estimate for the basic mission operations and data analysis (MODA) cost for a given spacecraft. MODA is defined as the cost of the following: maintaining and upgrading ground systems; mission control; tracking; telemetry; command functions; mission planning; data reduction and analysis; crew training and related activities. The model estimates the average annual MODA based on the type of mission, either manned, unmanned, and planetary, and the investment cost of the spacecraft. The investment cost is defined as the total development and production cost of the spacecraft, experiments and ground systems. Future study is to consider incorporating MODA developments in the cost model. Appendix C lists and describes the various software costs estimating software products that have been developed to date.

Table 7. Total Space Segment Costs among Scenarios for the Small Spacecraft Class (FY96\$M).

	Halo Orbit	Aerosynchronous	Common Period Inclined	Low Mars Orbit (6)	Dedicated	Mars Moons
RDT&E	59.4	59.4	59.4	59.4	59.4	60.5
First Unit Cost	5.41	5.41	5.41	5.41	5.41	5.41
Total Network Cost	20.2	29.4	38.4	55.6	10.6	10.3
Total Space Segment Cost	79.6	88.8	97.8	115.	70.1	71.8

⁵ Kelley, J., "Mission Operation and Data Analysis (MODA)." <http://www.jsc.nasa.gov/bu2/MODA.html> (October 20, 1997) website.

Table 7 shows the total RDT&E and production cost among scenarios for the small spacecraft class. The RDT&E costs and first unit costs are assumed to be similar among scenarios since the same design is applied to all scenarios in this spacecraft class. Due to size constraints for the small spacecraft class, there is little room for variation and customization in spacecraft design. From this analysis the dedicated scenario is the least expensive, followed by the halo orbit and Mars moons scenarios, then the aerosynchronous case, and then the Draim scenario, and finally the multiple low Mars orbiter scenario being the most costly. The CER used in determining the specific program costs for both the small and large relay spacecraft classes is shown in Appendix B. Due to limits in physical dimensions for the small spacecraft class, there is little variation in spacecraft design, and hence cost, among the scenarios. The only variation in cost stems from the differences in the scenario network configuration. Among the scenarios, the dedicated scenario has the lowest relative space segment cost, followed by the halo orbit and the Mars moons scenarios despite its added cost of landing technology development. The multiple spacecraft scenario shows to have the highest relative space segment cost.

Table 8. Total Space Segment Costs among Scenarios for the Large Spacecraft Class (FY96\$M).

	Halo Orbit	Aerosynchronous	Common Period Inclined	Multiple (6) Low Orbit	Dedicated (Medium Orbit)	Mars Moons
RDT&E	143.7	134.3	143.2	145.6	145.6	133.5
First Unit Cost	44.2	31.7	34.5	35.2	35.2	31.1
Total Network Cost	120.	140.	194.	282.	63.3	102.
Total Space Segment Cost	264.	275.	337.	427.	209.	235.

Conversely, for the large spacecraft class physical size is not a limitation, thus giving the spacecraft designer the option of customizing to the scenario of interest. Table 8 shows the total space segment costs for the large spacecraft class. The larger degree of customization of spacecraft design leads to greater variation in spacecraft cost among scenarios, though the greatest variation is still from the scenario configuration. Among the scenarios, the multiple spacecraft scenario carries the largest space segment cost, while the dedicated scenario with its medium orbit has the lowest relative space segment cost, followed by the Mars moons scenario with the added lander technology development and the halo orbit scenario. With the analysis of space segment costs discussed the launch segment costs are discussed next.

Launch Segment Costs

The selection of a launch vehicle in this study is driven by the following two parameters: the spacecraft destination the launch vehicle is to carry it to; and the spacecraft mass. The selected launch vehicle must be able to lift spacecraft into Low Earth Orbit (LEO) as a minimum for interplanetary trajectory. For a given satellite and satellite class, the launch vehicle chosen will determine whether a perigee kick motor (PKM) or apogee kick motor (AKM) would be included as part of the spacecraft or part of the launch vehicle. In general, a spacecraft launched by a Titan IIIC or Titan IV would need neither a PKM nor an AKM. A spacecraft launched by a Delta, Atlas-Centaur, or Ariane IV launch vehicle will likely need an AKM; and spacecraft launched by the Space Shuttle (a launch vehicle not considered in this study due to cost constraints) will need both a PKM and an AKM.⁶ Using a C_3 of $13 \text{ km}^2/\text{km}^2$ for the launch performance criteria for a Mars mission, Table 9 shows the planetary performance summary for the various launch vehicles and its associated cost. The information in Table 9 is consistent with the NASA classification of the launch vehicle class. The elements comprising the launch segment costs are adjusted to Fiscal Year 1996 millions of dollars (FY96\$M) for consistency for calculation in the cost model. Details of the derivation of the C_3 criteria

⁶ Sergeyevsky, A. et al. "Interplanetary Mission Design Handbook, Volume I, Part 2". JPL Publication No. 82-43, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, CA (September 15, 1983).

as well as the launch vehicle selection based on spacecraft mass are discussed further in Chapter 5 of this study.

Table 9. Planetary Performance Summary of Various Launch Vehicles Based on a C_3 of $13 \text{ km}^2/\text{s}^2$ with Cost Estimate.^{7, 8, 9, 10, 11}

Launch Vehicle Class	Launch Vehicle	Mars Payload System Mass (kg)	Launch Vehicle Cost (FY 96 \$M)
Small	Pegasus	60	13
	Taurus	210	15
	Athena	300	22
Medium	CZ - 2C	500	20
	CZ - 3	730	33
	CZ - 2E	1500	40
	Delta II (6925)	790	60
	Delta II (7925)	990	65
	Atlas I	1050	67
Intermediate	Atlas II	1300	88
	Atlas IIA	1400	85
	H II	1500	160
	Atlas IIAS	1850	95
	Ariane 4+L	2200	135
	Titan III & TOS	2450	225
	Titan III & IUS	2500	275
Large	Ariane 54L	4000	130
	Proton	5400	67
	Titan IV	7800	286
	(SRMU)/Centaur		

⁷ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design: 731.

⁸ Sergeyevsky, A., "Interplanetary Mission Design Handbook, Volume I, Part 2."

⁹ Kelley, J., "US Expendable Launch Vehicle Data for Planetary Missions," <http://www.jsc.nasa.gov/bu2/USELV.html> (September 25, 1996) website.

¹⁰ Kelley, J., "International Expendable Launch Vehicle Data for Planetary Missions," <http://www.jsc.nasa.gov/bu2/IELV.html> (September 25, 1996) website.

¹¹ "Space FAQ 13/13 - Orbital and Planetary Launch Services," <http://www.landfield.com/faqs/space/launchers.html> (February 5, 1998) website.

Then with the spacecraft destination determined, the next objective is to match the appropriate launch vehicle given a spacecraft mass. For the small spacecraft class a launch vehicle performance greater than 1050 kilograms was not considered for the purposes of practicality and cost. For the large spacecraft class a launch vehicle is needed to adequately propel a spacecraft mass larger than that of the small class to a Mars interplanetary trajectory. Thus launch vehicles with a lift performance of less than 790 kilograms for the large spacecraft class were not considered for the large spacecraft class. Also multiple launches to complete and establish a scenario network are considered in the calculations. Any given Mars launch may be shared, but launch sharing is only allowed with other missions destined to the Red Planet; a Mars launch typically cannot accommodate a geosynchronous space program.

Future work in this study is to consider future launch vehicles that are approaching operational readiness starting in 1998 if not later. The new launch vehicles are to provide payload performance at reduced operational cost, as well as to introduce the aerospace industry to commercial space the likes of which has never been experienced before.¹²

These new launch vehicles include the following:

¹² Sietzen, F., "1998: A Key Year in the Commercialization of Space." Ad Astra. (November/December 1997): 32-36.

- Boeing's Sea Launch with its transformation of a drilling platform into a mobile floating rocket base for a Zenit booster to be launched in the Pacific Ocean along the equator just Southeast of Hawaii;
- Boeing North American Division's Delta 3 with its new liquid-fueled upper stage capable of payloads in the 8000 pound range to geosynchronous transfer orbit (GTO);
- The Evolved Expendable Launch Vehicles (EELVs) to replace the current missile-derived launch vehicles at a fraction of the cost, with Lockheed Martin and Boeing competing for Air Force award on July 1, 1998; the X-33, potential replacement to the current Space Shuttle;
- The X-34, a possible enhancement to the present Pegasus, being carried aloft by an L-1011 commercial airplane; and
- Kistler Aerospace's privately developed and funded K-1 two-stage rocketship prototype to be launched from either the Department of Energy's Nuclear Test Site in Nevada, or a launch complex in Woomera, Australia.

One of the new rockets that have already contributed to new space launch capabilities includes Lockheed Martin's Atlas 2AR using Russia's RD-180 engines. Another launch vehicle is the Athena rocket which successfully launched NASA's Lunar Prospector early January, 1998 at a launch vehicle cost of a mere \$26 million according to NASA

interviews on television and radio. With the discussion of launch segment cost complete, the ground segment costs analysis is performed.

Ground Segment Costs

Ground segment costs involve costs toward establishment of a ground station system, which collects and processes data from the relay satellite network. For this model, the costs for various elements of a ground station system will be based upon typical distribution of costs between software, equipment, facilities, logistics, and systems level costs. Elements of the ground segment development costs will be based on a percentage of the software costs developed in the spacecraft software cost section of this study. Then the ground development costs are incorporated into the overall mission cost. In this study the ground development costs are approximately of the same order of magnitude among scenarios.

Table 10. Ground Segment Development Costs among Spacecraft Classes.

Ground Station Element	Development Cost as Percent of Software Cost (%)	Cost (FY 96\$M) for Small Class	Cost (FY 96 \$M) for Large Class
Software (= 0.3173 x KLOC)	100	3.72	8.88
Facilities	18	0.67	1.60
Equipment	81	3.01	7.19
Logistics	15	0.56	1.33
Systems Level			
• Management	18	0.67	1.60
• Systems Engineering	30	1.12	2.65
• Product Assurance	15	0.56	1.33
• Integration and Test	24	0.89	2.13
Total		7.51	17.94

Table 10 shows the ground segment development costs in FY96. The ground segment development cost total about \$7.5 million for the small spacecraft class and about \$18 million for the large spacecraft class. Since differences in ground segment design varies with spacecraft design, in this study the ground segment development costs will vary by the two spacecraft classes and will remain similar among the scenarios in each spacecraft class. Operations and support costs are discussed next.

Operations and Support Cost

The operations and support costs during the operational phase of the ground segment involves personnel costs over the duration of the relay mission, whether the ground system is supported by contractor or government personnel, and the maintenance cost of the equipment, software, and facilities. Maintenance costs are obtained by the following formula:¹³

$$\text{Maintenance} = 0.1 \times (\text{software} + \text{equipment} + \text{facilities})/\text{year}.$$

The values for software, equipment, and facilities costs were obtained from the Ground Software Cost section of this study.

For labor costs, there is contractor and government labor available in the aerospace industry, with government labor costing less than contractor labor in general. With the growing trend in the aerospace industry toward commercial programs and away from military programs, the contractor labor with a labor rate of \$0.14 million per staff year was used in this study. The labor rates include overhead costs and general expenses associated

¹³ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 730.

with personnel. Table 11 shows the operations and support cost per year along with the total operations and support cost for a five year program for the small spacecraft class and for the large spacecraft class.

Table 11. Operations and Support Cost Per Year and Total Cost for a Five Year Program for Both Spacecraft Classes (FY 96 \$M).

	Small Class (\$M/Yr)	Total Cost Small Class (\$M)	Large Class (\$M/Yr)	Total Cost Large Class (\$M)
Maintenance	0.74	3.72	1.77	8.85
Contractor Labor	0.14	0.72	0.14	0.72
Total Cost	0.88	4.44	1.91	9.57

A computerized version of the NASA/Air Force Cost Model (NAFCOM) for estimation of aerospace program costs is available in two versions in CD-ROM titled NAFCOM 96.¹⁴ One version is the Government Only version available only to NASA employees, and the other version is the Unrestricted Release version, which is available to anyone interested in aerospace cost analysis free of charge. The difference between the two versions is that the Unrestricted Release does not allow access to specific projects' names, costs, or weights. Therefore certain filters, features, and functions are inoperable on this version in order to protect sensitive and proprietary information. The Unrestricted

¹⁴ Kelley, J.. "Mission Operations Cost Model," website.

Release version also does not provide cost sheets of past missions for comparison and for application to this study. Future trade studies may utilize this method in evaluating mission operations and support costs.

Evaluation and Summary of the Mission Cost Driver

In the aerospace industry the monetary cost parameter appears to be the dominant-driving factor for a Mars mission. In reality many other factors related to cost also influence the selection of favorable scenarios, though the monetary cost parameter may still be the most dominant factor overall. The program cost driver is based on relative costs in FY 1996 dollars that generally includes the following four subfactors: space segment costs, launch segment costs, ground segment costs, and operations and support costs. Based on this study the total mission cost is largely affected by launch vehicle selection. The normalized values shown in Appendix K include all five-subfactor calculations, and since the total mission cost is largely dependent on launch vehicle selection, the values are shown by launch vehicle among scenarios studied. The lowest normalized values for each scenario are highlighted in bold print.

Cost overruns are seldom avoidable given the ambitious endeavor of establishing a communications network at the Red Planet. Although cost overruns have the most likely

affect in the total mission cost of any aerospace program, the amount of cost overrun is virtually impossible to estimate and has not been included in this study. However the rate of return upon successful establishment of a Mars communication network would likely be substantial. Despite the fluctuating nature of the total mission cost for a network scenario, one variable that has held steady is the rate of inflation experienced in the 1990's economy. Since mission cost is a significant mission driver, this parameter is weighted higher relative to the other design drivers in the Cost Model.

The collection of information related to mission cost in this study is shown in Appendix B, C, K-1a and K-1b. The subfactors with respective normalized values among scenarios are shown in Appendix K-1a and Appendix K-1b for the small spacecraft class and the large spacecraft class respectively. The average normalized value of the subfactors was evaluated in the halo orbiting and the Mars lunar lander scenarios. Appendix B shows the complete Cost Estimating Relationships (CER) for both spacecraft classes. Appendix C lists various software cost estimation packages that exist to date. However due to the proprietary nature associated with these software packages, these software packages were unavailable for use in this study. The total mission cost among scenarios for both spacecraft classes are discussed below.

Total mission cost - small spacecraft class. Table 12 shows the total mission cost compared among scenarios for the small relay spacecraft class prior to the values being normalized in Appendix K-1a. The values in parentheses in Table 12 indicate dual launches to achieve complete network establishment at or about Mars; all other values indicate single launches. Values in bold print indicate relative lowest cost among scenarios. The following is a list of favorable launch vehicles for each scenario:

Halo orbiter scenario - the Taurus or the Long March CZ-2C,

Aerosynchronous orbiter scenario - the Athena or the Long March CZ-2C,

Common period inclined scenario - the Long March CZ-2C,

Multiple low Mars orbiter scenario - the Long March CZ-2C or the Long March CZ-3,

Dedicated communication scenario - the Long March CZ-2C or the Taurus,

Mars lunar lander scenario - the Athena or the Long March CZ-2C or the Taurus.

Table 12. Total Mission Cost compared among Scenarios for the Small Relay Spacecraft Class. Values in Parentheses Indicate Dual Launches to Achieve Network Establishment at or about Mars; All Others are Single Launches. (FY96\$M).

	Halo Orbit	Aero-synchronous	Common Period Inclined	Low Mars Orbit (6)	Dedicated	Mars Moons
United States						
Atlas I	146.6	155.8	164.8	182.3	137.0	146.6
Delta II 6925	139.6	148.8	157.8	175.3	130.0	139.6
Delta II 7925	144.6	153.8	162.8	180.3	135.0	144.6
Athena	101.6	110.8	(141.8)	(159.3)	92.0	101.6
Pegasus						
Taurus	94.6	(118.8)	(127.8)	(160.3)	85.0	94.6
Japan						
MV	149.6	158.8	167.8	185.3	140.0	149.6
China						
CZ-2C	99.6	108.8	117.8	(155.3)	90.0	99.6
CZ3	112.6	121.8	130.8	148.3	103.0	112.6

Total mission cost - large spacecraft class. Table 13 shows the total mission cost compared among scenarios for the large relay spacecraft Class. The values in parentheses indicate the number of launches needed to achieve network establishment at or about Mars; all others are for single launches. The following is a list of favorable launch vehicles for each scenario:

Halo orbiter scenario - the Long March CZ-2E,

Aerosynchronous orbiter scenario - the Zenit, the Proton or the Long March CZ-2E,

Common period inclined scenario - the Zenit or the Long March CZ-2E,

Multiple low Mars orbiter scenario - the Zenit or the Long March CZ-2E,

Dedicated communication scenario - the Long March CZ-3 or the Long March CZ-2E,

Mars lunar scenario - the Long March CZ-2E.

Table 13. Total Mission Cost compared among Scenarios for the Large Relay Spacecraft Class. Values in Parentheses Indicate the Number of Launches Needed to Achieve Network Establishment at or about Mars; all others are for Single Launches. (FY96\$M).

	Halo Orbit	Aero-synchronous	Common Period Inclined	Low Mars Orbit(6)	Dedicated	Mars Moons
United States						
Atlas I	398 (2)	476 (3)	605 (4)	827 (6)	276	368 (2)
Atlas II	352	363	513 (2)	953 (6)	297	322
Atlas IIA	349	360	507 (2)	680 (3)	294	319
Atlas IIAS	359	370	527 (2)	710 (3)	304	329
Delta II 6925	384 (2)	455 (3)	577 (4)	605 (3)	269	354 (2)
Delta II 7925	394 (2)	470 (3)	597 (4)	815 (6)	274	364 (2)
Titan IV - Centaur	514	525	587	675	459	484
France						
Ariane 4	399	410	607 (2)	695 (2)	344	369
Japan						
H-2	424	595 (2)	657 (2)	905 (3)	369	394
MV		485 (3)	627 (2)		349	374 (2)
Russia						
Proton D1e	331	342	471 (2)	559 (2)	276	301
Zenit 2	344	355	497	585	289	314
China						
CZ-2E	304	355 (2)	417 (2)	545 (3)	249	274
CZ3	330 (2)	374 (3)	469 (4)		242	300 (2)

Total Mission Cost Summary

Preliminary findings from the total mission cost parameter yielded no unusual results. From a total program cost standpoint the small spacecraft class is more cost effective compared to the large spacecraft class. Among the scenarios studied the dedicated communication relay scenario with its single spacecraft exhibited the lowest cost, with the halo orbiting scenario and the Mars lunar lander scenarios having cost slightly more. This is expected since a minimum number of relay spacecraft and launches are needed. However, the dedicated communication relay scenario will likely have communication performance limitations as shown later in this study. The multiple spacecraft scenarios, the multiple low Mars orbiter scenario in particular with potentially multiple launches, have the least cost-effective scenario.

CHAPTER 5

The Spacecraft Sizing Design Parameter

$$J = w_{\$}(\$) + w_{\text{Size}}(\text{Size}) + w_{\text{Comm}}(\text{Comm}) + w_{\text{Life}}(\text{Life}) + w_{\text{Plan}}(\text{Plan})$$

This chapter describes the spacecraft sizing parameter of the cost equation and shows the spacecraft sizing estimation among the six scenarios. Table 14 shows the relative comparison of contributing factors that comprise the sizing parameter, ΔV_2 and C_3 , in rank of high, medium and low, among each scenario. Since ΔV_1 and C_3 are basically the same measure, only C_3 values are listed. Again the rankings are not intended for obtaining absolute values but rather to determine relative comparisons. Table 14 shows the ΔV_2 and C_3 values.

Table 14. Factors in the Spacecraft Sizing Parameter are compared among Scenarios.

	Halo Orbit Orbiters (L₁ / L₂)	Aerosynchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
ΔV_2	med / med	med-low	low	med	med-low	med-low / med-low
C_3	low / low	med	high	med	med	med

Spacecraft Sizing Approach

The determination of the spacecraft sizing parameter first involves finding the total amount of ΔV required for the interplanetary journey to Mars. The total ΔV is the summation of the following ΔV parameters; the ΔV requirement for Earth departure, the ΔV requirement for Mars arrival, and the amount of spacecraft stationkeeping where applicable. A secondary consideration in weighing the spacecraft sizing parameter among scenarios is the amount of launch energy C_3 required propelling a rocket upper stage for such a Mars mission. The combination of the ΔV and the C_3 results are used in determining the appropriate launch vehicle for each scenario studied. Next the estimate of the spacecraft mass portion of the upper stage is performed. The spacecraft mass is comprised of the spacecraft subsystems including power, propulsion, payload, payload adapter, structure, and various spacecraft hardware is performed. Consequently the spacecraft sizing can be estimated, completing a four-step process. This four-step process becomes iterative until the spacecraft sizing for the scenario of interest is determined to be practical and to satisfy mission objectives.

The ΔV requirement is the total velocity change required leaving orbit from Earth and to reach and insert into Mars orbit. This total ΔV requirement is a major factor

influencing the deployment costs of a Mars communication network. The total ΔV can be divided by the following:

- 1) ΔV_1 , the ΔV necessary to leave low Earth orbit and to begin the cruise phase of the interplanetary trajectory, and;
- 2) ΔV_2 , the ΔV needed to insert into the required orbit at arrival, and;
- 3) Stationkeeping ΔV .

In general the greater the ΔV needed to leave low Earth orbit and into the cruise phase, the greater the amount of propellant needed. And, the higher the orbit insertion ΔV requirement for an interplanetary trajectory, the greater the propellant and hence the overall mass needed to launch from Earth and to achieve orbit insertion. Another notable observation is that the more highly eccentric the final insertion orbit, the lower the ΔV requirement. However, choosing a lower ΔV elliptical orbit will result in greater variations in the data transfer rate and the signal strength due to the changes in altitude in the spacecraft's elliptical orbit. Hence a lower data rate may result, unless increases to the complexity of the communication network may be needed to compensate for variation in the signal strength and/or to maintain a minimum average data rate.

Thus the ΔV and the C_3 quantities required for an Earth to Mars interplanetary trajectory need to be investigated for a Mars Relay Satellite Network mission. The next

section describes the method in determining the amount of ΔV and the amount of C_3 needed for each of the scenarios.

Determining the ΔV and the C_3 for each Scenario

The ΔV and the C_3 for each scenario is investigated by determining the amount of fuel needed to reach the arrival destination at or about Mars. Figure 8 illustrates a representative Earth to Sun-Mars L_1 interplanetary mission; the Earth to Sun-Mars L_2 trajectory is similar. The calculation of the Earth to Mars trajectory is also similar with the exception of the use of the patch-conic method at Mars arrival. Figure 8 below shows the following interplanetary trajectory: heliocentric departure velocity V_{dep} in vector form from a 6500 kilometer circular parking orbit to achieve the hyperbolic excess velocity V_{∞}^+ , the interplanetary trajectory from Earth orbit to halo orbit about the Sun-Mars L_1 point, and the arrival ΔV (ΔV_2). A Mars mission at launch may or may not require a departure ΔV (ΔV_1) depending on whether or not the launch vehicle can propel the spacecraft directly into the interplanetary trajectory with sufficient hyperbolic escape velocity to leave Earth orbit.

Once a selected launch vehicle is able to inject the spacecraft for a direct planetary trajectory, then the total ΔV required from the mission would be equivalent to the arrival ΔV (ΔV_2) plus any stationkeeping ΔV needed for the remainder of the spacecraft mission. Otherwise the spacecraft can only be initially launched to an Earth parking orbit. Then a departure ΔV , ΔV_1 , would be required to depart from the parking orbit in addition to the arrival ΔV , ΔV_2 , plus any stationkeeping ΔV . Thus among the scenarios the total amount of ΔV needed for a Mars mission can vary. They can vary between only requiring that of the arrival ΔV ($\Delta V_{\text{total}} = \Delta V_2$), to requiring the sum of Earth departure ΔV , Mars arrival ΔV , and any applicable stationkeeping ΔV ($\Delta V_{\text{total}} = \Delta V_1 + \Delta V_2 + \Delta V_{\text{Stationkeeping}}$). However, the ΔV requirements need to be weighed against the C_3 acceleration requirements.

Note that Figure 8 shows the position of Earth at time t_1 (spacecraft launch epoch time from Earth), and the position of Mars at time t_2 (spacecraft arrival epoch time at Mars); neither the position of Earth at time t_2 nor the position of Mars at time t_1 are shown. Figure 8 can be applied to the “Earth to Sun-Mars L_2 ” mission with the arrival destination being about the L_2 point, and for the Mars orbit scenario with the arrival destination being an orbit about the Red Planet.

Figure 8. Schematic of an interplanetary mission from a low Earth parking orbit to halo orbit about the Sun-Mars L_1 point. R values are distance vectors from the Sun and V are velocity vectors. Note that vectors are not to scale.

This study found among the various scenarios that the amount of fuel required escaping the Earth's gravity sphere of influence to be similar for identical launch dates. The significant difference among the scenarios is in the amount of fuel needed for Mars arrival in which the rocket equation was used to compare the fuel mass requirements among each scenario. The values of the various component mass terms are only estimates and will vary between scenarios. The total mass of the spacecraft and of the launch vehicle upper stage at launch among each scenario along with the individual components

of the payload and rocket which are an integral part of the rocket equation are examined later in this chapter.

The C_3 for Earth departure and the ΔV required at Mars arrival, ΔV_2 , for each design configuration for a Mars planetary mission was calculated using a FORTRAN program.¹ A launch date of 25 September 1992 and arrival date of 24 August 1993, illustrated in Figure 9, correspond to the actual dates which occurred with the Mars Observer mission. Information of the Mars Observer was obtained from the Mars Observer Mission Plan produced by the Jet Propulsion Laboratory and was used as an example in testing the validity of the program.² A total ΔV of eight MRS configurations were computed using the FORTRAN program initially from a 6500 km radius low Earth orbit (LEO). These include a relay spacecraft at a halo orbit about the Sun-Mars L_1 and L_2 point, spacecraft at a 3950 km radius low Mars orbit, spacecraft in an aerosynchronous orbit, a lander spacecraft on Phobos and on Deimos, and spacecraft at a Drain 74-hour orbit.

¹ Microsoft FORTRAN, computer software, version 5.0, (Microsoft Corporation, 1982-1992).

² Rokey, M. J., and Dodd, S. R., "Mars Observer Mission Plan," Jet Propulsion Laboratory internal document D-2518, Revision F (July 1992).

Figure 9. Approximate positions of the planets relative to the Vernal Equinox (γ) shown on (a) 25 September 1992 and on (b) 24 August 1993 as obtained from the Orbit Workbench software program.

An observation worth noting is that these calculations are for worst case and will vary depending on the parking orbit altitude and Mars arrival altitude. For all the scenarios except the Mars moons and halo orbiting scenarios, a lower amount of ΔV would be needed if the orbits about Mars were higher than that described along with varying the amount of ΔV by selecting different departure and arrival dates. However, for the Mars moons and halo orbiting scenarios, the ΔV values are less flexible and can vary only by selection of departure and arrival dates.

The FORTRAN program uses the patch-conic method for each of the eight cases in calculating the C_3 used to leave the Earth's orbit and calculating the total ΔV needed at Mars arrival. The launch energy, C_3 , is the amount of energy needed for a rocket and its payload to escape the Earth gravity on its interplanetary journey to Mars. The C_3 is equal to the square of the hyperbolic excess velocity, V_∞^+ . The launch site and Earth parking orbit is assumed to be the same for each scenario. From a given launch site, the greater the amount of C_3 needed to escape the Earth's gravity, the greater the amount of thrust required by the rocket. Details in the derivation of the FORTRAN program in obtaining the ΔV budget are shown in Appendix D. Thus the objective is to find the interplanetary trajectory with the lowest C_3 possible.

While the FORTRAN program calculates the C_3 , the calculation of ΔV_1 is also found by calculating the difference in the vector velocities between the Earth's parking orbit velocity, and the velocity the spacecraft would need to be at the start of its interplanetary trajectory. Once the ΔV_1 is computed, the launch energy is found by computing the square of the hyperbolic excess velocity (V_∞) leaving Earth. The result is compared with published data from the Interplanetary Mission Design Handbook as a second source of

the FORTRAN program verification.³ The launch and departure dates were varied to determine the launch and arrival dates where the minimum launch energy and corresponding minimum total ΔV was achieved. The selection of possible launch vehicles was determined based on launch energy and corresponding maximum payload mass allowed for planetary missions. The equations associated with obtaining the ΔV budget are listed in Appendix E. A detailed flowchart of the FORTRAN program used in determining the ΔV_1 budget used in this study is shown in Appendix F. With the calculations evaluated, the Mars arrival ΔV_2 analysis is discussed next.

The Mars Arrival ΔV (ΔV_2) Analysis

For the interplanetary trajectory-to-Mars orbit portion of the MRS journey, the FORTRAN program uses the patched-conic method to determine the following: the ΔV needed to enter Mars orbit at the end of its trajectory (ΔV_2); and the total Earth to Mars ΔV budget. The heliocentric velocity vector of Mars at the instant of satellite arrival to Mars was used in the ΔV_2 calculations for all the scenarios in this study with the exception

³ Sergeyevsky, A. et al, "Interplanetary Mission Design Handbook, Volume I, Part 2." JPL publication no. 82-43. Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California (September 15, 1983).

of the halo scenarios. The ΔV_2 calculation for the halo orbits about the Sun-Mars L_1/L_2 points require the respective velocity vectors of the halo orbits about libration points L_1 and L_2 . Seeking the minimum Mars halo insertion velocity of interest, four positions with a zero velocity component normal to Mars' orbit plane in one halo orbit about Sun-Mars L_1 and Sun-Mars L_2 are chosen from computational output files obtained from the FORTRAN program. Thus the ΔV needed to change the spacecraft velocity to match the corresponding halo point velocities about the libration point of interest were calculated. These ΔV values are compared against ΔV values among the other scenarios. Appendix G shows the FORTRAN source code used in determining the ΔV budget.

The following results of minimum values of total ΔV , ΔV_1 , ΔV_2 and C_3 are shown in Table 15 below. These ΔV values are based on spacecraft leaving at 6500-kilometer radius Earth circular parking orbit prior to the cruise phase to Mars. Note also that these minimum values vary over different launch dates, so that minimum total ΔV does not necessarily equal minimum ΔV_1 and minimum ΔV_2 . The data reveals that ΔV_1 values are about the same throughout the scenarios, but have varying minimum ΔV_2 values with the halo orbit scenario showing up to a 1.7 km/s greater ΔV value compared to the lowest minimum ΔV value as observed in the Draim 74 hour case. Apparently the greater ΔV_2 value for the Sun-Mars L_2 halo scenario is due to the lack of gravity assist from Mars as

the spacecraft travels towards the Sun-Mars L_1 and L_2 halo arrival destination along with the greater difference in the heliocentric velocity vectors between Mars and the spacecraft. Note that the minimum ΔV_1 and/or minimum ΔV_2 does not necessarily have the minimum C_3 , nor is the sum of the minimum total ΔV equal to the minimum ΔV_1 and minimum ΔV_2 . Therefore the values shown in Table 15 show minimum values for that parameter only and are not related to all the other parameters listed.

Table 15. Relative Minimum ΔV (ΔV_1 and ΔV_2), Minimum Total ΔV , and Minimum C_3 are compared among Scenarios. Calculations are based on spacecraft leaving at 150-km altitude circular Earth parking orbit.

	Halo Orbit Orbiters (L_1/L_2)	Aero-synchronous Orbiters	Drain Orbiters (74 hr)	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter (Med Mars Orbit)	Martian Moon Landers (Phobos / Deimos)
C_3 (km ² /s ²)	11.279 / 11.590	11.4341	11.4341	11.4341	11.4341	11.4341 / 11.4341
Minimum ΔV_1 (km/s)	3.736 / 3.749	3.7425	3.7425	3.7425	3.7425	3.7425/ 3.7425
Minimum ΔV_2 (km/s)	1.8376 / 2.2320	0.8424	0.5609	1.5177	1.3279	1.0545/ .8178
Total ΔV	5.649 / 6.0437	5.1915	4.9099	5.6402	5.4912	5.2997 / 5.1854

The Orbital Workbench program. The Orbital Workbench program is first used to determine the positions of the planets at specified departure and arrival times from parameters stated in the Mars Observer Mission Plan.⁴ These values were later used as

⁴ Orbital Workbench, computer software, version 1.1, (Cygnus Engineering, 1990-1992).

inputs to a FORTRAN program. An important note observed from Figure 9 is that the positions of the planets are referenced with respect to the Vernal Equinox (γ). Appendix H shows the Mars-to-Sun and Earth-to-Sun distances during selected dates with their corresponding longitude and latitude positions with respect to the Vernal Equinox. Thus both the launch vector from Earth and the arrival vector to Mars are obtained. One simplification in the FORTRAN program calculation is that the latitude positions in the heliocentric frame of reference were assumed to be zero, such that the orbits of both Mars and Earth are assumed to have no inclination with respect to the ecliptic plane. An important observation to make is that the launch and arrival longitude values are with respect to the Vernal Equinox, γ , and are not with respect to the periapsis of the trajectory that is used in the FORTRAN program. However, the transfer angle values in both the FORTRAN and the Orbital Workbench programs are identical and should agree in value. Therefore both these programs can be used in verifying the correct trajectory transfer angle. Figure 10 below shows the velocities at Earth departure (a), and at the Sun-Mars L_1 arrival (b), obtained using vector addition. Though not illustrated below, the vector velocities for Sun-Mars L_2 arrival is similar to that of Sun-Mars L_1 . In the figure γ_1 is the departure flight path angle from Earth, and γ_2 is the arrival flight path angle to Mars.

Figure 10(A). The departure velocity $V_{\text{departure}}$, hyperbolic excess velocity V_{∞}^{+} , and the departure angle γ_1 of Earth obtained using vector addition (see Figure 8).

The calculation of the flight path angle γ_2 of the halo orbit about the Sun-Mars L_1 as shown in Figure 10(B) involves two possible solutions to the cosine function in the flight path angle equation. Hence care must be taken to use the correct solution of the flight path angle. Though not illustrated in this study, similar calculations are applied for the Sun-Mars L_2 case using the ΔV and the corresponding arrival angle γ_2 of the halo orbit about the Sun-Mars L_2 arrival point.

Figure 10 (B). The ΔV and the arrival angle γ_2 of the halo orbit about the Sun-Mars L_1 arrival point obtained using vector addition. The velocity components at the L_1 halo point shown above are obtained from an input file.

The Total ΔV Requirement

Tables 16, 17, and 18 below show the results of the ΔV calculations for the Earth-Mars, Earth-to-Sun-Mars L_1 , and for the Earth-to-Sun-Mars L_2 trajectories respectively from the FORTRAN program. Trajectory calculations with a time of flight of 333 days were used for consistency. The computed results for the Earth to Mars trajectory case shown in Table 16 are consistent with the data in the JPL Mars Observer Mission manual.

Table 16. Calculated ΔV for Spacecraft Leaving 150 kilometer Altitude Circular Earth Parking Orbit, Launch Energy (C_3), Mars Arrival ΔV , and the Cumulative ΔV for the Earth-Mars Trajectory Using Various Departure and Arrival Dates. The minimum launch energy value for the applicable departure-arrival date is highlighted in bold face. The cumulative ΔV value selected for the applicable departure-arrival date is highlighted in underlined bold face.

Earth to Mars-3950 km Radius Circular Trajectory					
Departure	Arrival	Departure ΔV (km/s)	Launch Energy (C_3) (km^2/s^2)	Arrival ΔV (3950km Mars circular) (km/s)	Cumulative ΔV (km/s)
7/19/92	6/17/93	5.1648	46.2422	1.4756	6.6404
8/2/92	7/1/93	4.7319	35.1781	1.4969	6.2288
8/16/92	7/15/93	4.3601	25.975	1.5445	5.9046
9/1/92	7/31/93	4.0264	17.9487	1.6433	5.6696
9/3/92	8/2/93	3.9455	16.038	1.6862	5.6318
9/30/92	8/29/93	3.7489	11.4451	1.999	<u>5.7479</u>
10/1/92	8/30/93	3.7484	11.4341	2.0167	5.7651
10/2/92	8/31/93	3.7486	11.4388	2.0347	5.7833
10/3/92	9/1/93	3.7495	11.4596	2.0531	5.8026
10/15/92	9/13/93	3.8141	12.9608	2.3028	6.117
11/1/92	9/30/93	4.0851	19.3457	2.7604	6.8455

From the cumulative ΔV information for a Mars orbiting relay mission in Table 16, normally the minimum ΔV of 5.632 kilometers per second would have been the trajectory of choice associated with the minimum ΔV . But with a correspondingly high launch energy of $16.038 \text{ km}^2/\text{s}^2$, the next least minimum ΔV of 5.7479 kilometers per second is chosen with the more appropriate corresponding launch energy of $11.445 \text{ km}^2/\text{s}^2$.

Tables 17 and 18 show the ΔV calculated for spacecraft leaving a 150 kilometer Earth altitude circular parking orbit for the Earth-Mars L_1 and L_2 trajectories respectively using

various departure and arrival dates. The minimum and hence favorable launch energy value for the applicable departure-arrival date is highlighted in bold face. The minimum cumulative ΔV value for the applicable departure-arrival date is highlighted in underlined bold face. Some effort was taken in optimizing the L_1 and L_2 trajectories by finding the maximum and the minimum libration points with respect to the heliocentric orbit plane in achieving minimum ΔV_2 . The relative maximum and minimum position information was obtained from a halo data file calculated by Danehy. This information was then included in the FORTRAN program. Details of this optimization derivation are detailed in Appendix E from equations (E-85) through (E-100).

Table 17. Calculated ΔV for Spacecraft Leaving 150 kilometer Altitude Circular Earth Orbit, Launch Energy, Mars Arrival ΔV , and Cumulative ΔV for the Earth-Mars L_1 Trajectory Using Various Departure and Arrival Dates. The minimum launch energy value for the applicable departure-arrival date is highlighted in bold face. The minimum cumulative ΔV value for the applicable departure-arrival date is highlighted in underlined bold.

Earth to L_1 trajectory					
Departure	Arrival	Departure ΔV (km/s)	Launch Energy (C_3) (km ² /s ²)	Arrival ΔV (km/s)	Cumulative ΔV (km/s)
7/19/92	6/17/93	5.1090	44.8830	2.3372	7.4461
8/2/92	7/1/93	4.6831	34.0570	2.1624	6.8454
8/16/92	7/15/93	4.3193	25.0860	1.9983	6.3170
9/1/92	7/31/93	3.9940	17.3210	1.8622	5.8566
9/20/92	8/19/93	3.7700	12.0650	1.8793	<u>5.6490</u>
9/25/92	8/24/93	3.7450	11.4890	1.9334	5.6783
9/29/92	8/28/93	3.7360	11.2960	1.9948	5.7313
9/30/92	8/29/93	3.7362	11.2790	2.0115	5.7473
10/1/92	8/30/93	3.7362	11.2840	2.0303	5.7663
10/2/92	8/31/93	3.7371	11.3040	2.0501	5.7869
10/15/92	9/13/93	3.8113	13.0270	2.3904	6.2016
11/1/92	9/30/93	4.0930	19.6660	3.0503	7.1436

From this ΔV information for a relay mission with a Sun-Mars L_1 arrival destination in Table 17, the minimum ΔV of 5.649 kilometers per second is an acceptable choice with an appropriate launch energy of $12.065 \text{ km}^2/\text{s}^2$. This launch energy is well within the maximum launch energy limit of $13 \text{ km}^2/\text{s}^2$. Thus the cruise trajectory with a 5.649 km/s ΔV is selected for evaluation towards the Cost Model in this study.

Table 18. Calculated ΔV for Spacecraft Leaving 150 kilometer Altitude Circular Earth Parking Orbit, Launch Energy (C_3), Mars Arrival ΔV , and Cumulative ΔV for the Earth-Mars L_2 Trajectory Using Various Departure and Arrival Dates. The minimum launch energy value for the applicable departure-arrival date is highlighted in bold face. The minimum cumulative ΔV value for the applicable departure-arrival date is highlighted in underlined bold face.

Earth to L_2 trajectory					
Departure	Arrival	Departure ΔV (km/s)	Launch Energy (C_3) (km^2/s^2)	Arrival Δ V (km/s)	Cumulative ΔV (km/s)
7/19/92	6/17/93	5.214	47.6230	2.7685	7.9829
8/2/92	7/1/93	4.773	36.3190	2.5940	7.3672
8/16/92	7/15/93	4.393	26.8830	2.4281	6.8207
9/1/92	7/31/93	4.048	18.5950	2.2792	6.3274
9/30/92	8/29/93	3.751	11.6280	2.3318	<u>6.0827</u>
10/1/92	8/30/93	3.75	11.6010	2.3455	6.0952
10/2/92	8/31/93	3.749	11.5900	2.3600	6.1093
10/3/92	9/1/93	3.749	11.5950	2.4091	6.1249
10/15/92	9/13/93	3.806	12.9100	2.3755	6.4372
11/1/92	9/30/93	4.067	19.0400	2.6311	7.2765

From this ΔV information for a relay mission with a Sun-Mars L_2 arrival destination in Table 18, the minimum ΔV of 6.0827 kilometers per second is comparatively higher

relative to the other scenarios in this study. This is due to the large ΔV_2 needed to reach its destination. This total ΔV corresponds to a launch energy of $11.59 \text{ km}^2/\text{s}^2$. This launch energy is well within the maximum launch energy limit of $13 \text{ km}^2/\text{s}^2$ and is acceptable for use in this study. Therefore this cruise trajectory with a 6.0827 km/s ΔV is selected as the baseline towards the Cost Model evaluation among the scenarios used in this study.

Therefore with both relay spacecraft classes considered, the scenario ranking starting with the lowest ΔV and hence the most favorable scenario observed is as follows:

1. The common period inclined scenario
2. The aerosynchronous and the Martian moon lander cases
3. The dedicated communication scenario (medium Mars orbit)
4. The multiple low Mars orbit scenario
5. The halo orbit scenario.

Figure 11. Launch energies (C_3) for L_1 , L_2 , and Earth to Mars trajectories at specified launch dates.

From Figure 11 there is an apparent minimum C_3 range of about $11.5 \text{ km}^2/\text{s}^2$ between 9/21/92 and 10/3/92 with a correspondingly low ΔV required for a Mars mission. Thus choosing a $13.0 \text{ km}^2/\text{s}^2$ C_3 value provides an adequate margin for a reasonable launch window in a Mars mission. For most scenarios the Mars to Earth C_3 curve would apply, while for the two-halo orbiter cases the Earth-to-Sun-Mars L_1/L_2 C_3 curve would apply. As was observed from Figure 11, the C_3 values among the scenarios have about the same range of values.

A note of caution worth mentioning is that a maneuver associated with ΔV_1 is used when the launch itinerary of an interplanetary mission includes a parking orbit about the Earth prior to spacecraft injection into its interplanetary trajectory. Some mission launches, such as the ill-fated Mars '96 Russian mission, may perform a direct departure from the Earth's gravity by the rocket engines thereby eliminating the need for a ΔV_1 maneuver.

Figure 12(A), (B), and Figure 13 below compare the ΔV_1 , ΔV_2 , and the total ΔV among scenarios respectively. Figure 12(A) shows the ΔV_1 comparison between the Earth-to-Mars L₁, Earth to Mars L₂, and the general Earth-to-Mars trajectory, where the non-halo scenarios are categorized in this instance, at specified launch dates. Similar ΔV_1 values are shown among the general Mars trajectories being that each of the scenarios depart Earth in the same manner and because of the relatively small differences in interplanetary distances to the respective Mars destinations. Launch dates shown in Figure 12(A), (B), and Figure 13, as well as Figure 11 have Mars's arrival dates that are cross-referenced and are shown in Table 15.

Figure 12(A). ΔV_1 comparison between the Earth to Mars' L_1 , Earth to Mars's L_2 , and the general Earth to Mars trajectory at specified launch dates.

Figure 12(B) shows the ΔV_2 comparison between the scenarios discussed in this study at specified launch dates. A greater disparity in ΔV_2 values exist among scenarios because the respective ΔV_2 values are highly dependent on the final orbit used at Mars. In the case of the non-halo orbit scenarios, ΔV_2 is dependent on the nature of the final orbit. In the case of the halo orbit scenarios, Mars is relatively distant from the Sun-Mars halo positions for Mars gravity to be used to lower the ΔV as can be done in the non-halo orbit scenarios. The lack of the use of Mars gravity did not adversely affect the minimum ΔV for the Sun-Mars L_1 case as compared to the non-halo scenarios. However the ΔV_2 curve does reach a minimum for a brief time span thereby implying a narrower window of

opportunity in achieving a low ΔV_2 . A similar ΔV_2 curve is seen in the Sun-Mars L_2 case as in Sun-Mars L_1 case but with a higher minimum ΔV_2 .

Figure 12(B). ΔV_2 comparison between the Earth-to-Mars L_1 , Earth-to-Mars L_2 , and various Earth-to-Mars trajectories at specified launch dates.

The total ΔV comparison between the scenarios at specified launch dates as illustrated in Figure 13 show a minimum total ΔV of 4.85 km/s for the Earth-to-Mars case corresponding to a September 13, 1992 launch. About a 5.65 km/s minimum ΔV was observed for the Sun-Mars L_1 case and about a 6.05 km/s minimum ΔV was observed for the Sun-Mars L_2 case, both L_1 and L_2 instances correspond to a September 20, 1992 launch. The minimum fuel requirements for the Sun-Mars L_1 example are comparable to the general Earth to Mars case, while there is a greater ΔV requirement for the Sun-Mars

L_2 case. However, this greater ΔV requirement for the Sun-Mars L_2 case can be achieved at the expense of either a smaller spacecraft mass or reduced launch window. Thus the libration point trajectories can be seen to have either a relatively small total ΔV penalty or as a smaller launch window opportunity as compared to a direct Earth-to-Mars trajectory. An important note is that the ΔV values can vary depending on the shape and size of the parking orbit about Earth prior to the hyperbolic cruise trajectory, the size of the interplanetary transfer orbit, and /or the final orbit at the Mars destination. But the relative ΔV values is of greater importance in this study. And with the individual values of ΔV estimated, these ΔV values are used as part of the inputs to determining spacecraft mass and sizing discussed in the next section.

Figure 13. Total ΔV comparison between the Earth to Mars' L_1 , Earth to Mars' L_2 , and various Earth to Mars trajectories at specified launch dates.

Initial findings of the ΔV and the C_3 among scenarios. Based on the findings for the ΔV and C_3 comparisons based on the minimum possible total ΔV observed for each scenario with a launch energy of about $13 \text{ km}^2/\text{s}^2$, the following scenarios were ranked from most favorable to least favorable:

- 1) The Draim 74 hour common-period constellation
- 2) The Mars moons and the Aerosynchronous constellations
- 3) The Medium Mars orbit scenario
- 4) The Low Mars orbit scenario
- 5) The Sun-Mars L_1 orbit scenario
- 6) The Sun-Mars L_2 orbit scenario.

The relatively high ΔV requirement for the halo orbit scenario compared to the Mars-orbiting scenarios is largely attributed to the nature of the halo orbits orbiting about libration points which are themselves heliocentric orbits which the planets in the solar system maintain. The halo orbit scenario also does not take advantage of Mars's gravity and aerobraking upon Mars arrival. However, the difference in total ΔV between the scenario extremes is not generally beyond launch vehicle performance limitations, although this may reduce the launch vehicle selection overall for the case of selection of the halo scenario.

Itemizing the Mass Components of the Launch Vehicle “Stack”

The ΔV and C_3 considerations studied above largely determine the sizing of the lower stages of the launch vehicle needed to achieve the interplanetary journey to its final destination to the Red Planet in its communication relay mission. With the ΔV and C_3 considerations in spacecraft sizing studied, the next consideration in spacecraft sizing is to investigate the sizing of the launch vehicle upper stage with the spacecraft included.

The launch vehicle upper stage components among the scenarios investigated are itemized and illustrated below in Figure 14. “The Stack” is referred to as the upper portion of the launch vehicle that continues into the interplanetary journey after the initial rocket engine stages have been expended.⁵ Components of “The Stack” include the payload, the payload adapter, the inert propellant and hardware, the propellant contained in the upper stage, the flight equipment, the structure of “The Stack”, and the launch vehicle adapter. Figure 14 also illustrates the general two-step calculation in determining the individual component masses of “The Stack.” STEP 1 mass calculations focus on the payload, which in this instance is the spacecraft mass, M_{sc} just after the ΔV that the upper

⁵ Pernicka, H., lecture notes from Aerospace Engineering 110 course, San José State University, (spring 1994).

stage imparts. Conversely, STEP 2 mass calculations focus on the mass of “The Stack” which includes the upper stage and the spacecraft mass.

In this study the term for payload mass, M_{pay} is equal to the spacecraft wet mass, $M_{\text{w/c}}$. and the terms M_{pay} and $M_{\text{w/c}}$ will be used interchangeably in this portion of this study. Another term, the wet mass, M_{wet} , is a term used in the aerospace industry to represent the total mass of “The Stack” without the mass of the launch vehicle adapter since the launch vehicle adapter does not stay attached to the rest of “The Stack” for its interplanetary trajectory. The equation for M_{wet} is shown in equation (5-1).

$$M_{\text{wet}} = M_{\text{w/c}} + M_{\text{pla}} + M_{\text{pi}} + M_{\text{prop}} + M_{\text{FE}} + M_{\text{s}} \quad (5-1)$$

$M_{\text{w/c}}$ is the spacecraft mass, and M_{pla} is the payload adapter mass, which is equal to the quantity $f_{\text{pla}} M_{\text{pay}}$. The payload coefficient factor, f_{pla} , is approximately equal to 0.03. The inert propellant mass, M_{pi} , is the sum of the initial inert propellant, M_{pio} , and the hardware mass. The hardware mass is equal to a fraction of the propellant mass described by the quantity $f_{\text{pi}} M_{\text{prop}}$. The inert propellant coefficient factor, f_{pi} , is approximately equal to 0.1. M_{prop} is the propellant mass in the upper stage, M_{FE} is the mass of the flight equipment, and M_{s} is the structural mass, which includes the fairing and the cylindrical upper stage housing. M_{s} is equal to the quantity $f_{\text{s}} M_{\text{wet}}$, where the structure coefficient, f_{s} , is approximately equal to 0.18.

Figure 14. "The Stack" portion of the launch vehicle system.

Another term used is the dry mass of "The Stack", M_{dry} . M_{dry} is equal to the wet mass of "The Stack" minus the propellant mass used by the upper stage. This dry mass is achieved at the end of a successful departure ΔV from low Earth parking orbit with all of its propellant usefully expended to place the spacecraft on its heliocentric trajectory to Mars. This dry mass is equal to the initial wet spacecraft mass. The equation for the dry

mass for “The Stack” is indicated in equation (5-2) below.

$$M_{\text{dry}} = M_{\text{wet}} - M_{\text{prop}} \quad (5-2)$$

Since the amount of propellant needed to leave low Earth orbit to Mars is important in determining the sizing and selection of the launch vehicle, the mass of propellant used in a mission is thus of particular interest and is the focus of this section. And since propellant is designed into both the payload or spacecraft portion as well as the upper stage portion of “The Stack”, a two-step process is performed before M_{prop} , the upper stage propellant mass can be readily evaluated. In the first step, the classic rocket equation is used to obtain the spacecraft mass, M_{sc} , defined in this study as comprising the relay spacecraft carrying the communications package as its primary payload along with its spacecraft propulsion, power, attitude control, thermal, and structure components. The evaluation of M_{sc} using the rocket equation towards the determination of the payload mass, M_{pay} . The payload mass evaluation in STEP 1 is then used towards the STEP 2 evaluation of the upper stage propellant mass, M_{prop} , of “The Stack”. The process in determining M_{pay} is shown next in STEP 1.

STEP 1: Determining M_{dry} , the Payload Mass of “The Stack”

The Rocket Equation. First the fundamental rocket equation is used to relate the propulsion performance capability of ΔV_2 with the ratio of the initial spacecraft mass to the final spacecraft mass. This final spacecraft mass occurs at the end of its mission. The rocket equation is shown below in equation (5-3). An important observation is that $M_{initial}$ is equal to $M_{w/c}$ in STEP 1, with $M_{initial}$ being the fully fueled spacecraft mass prior to the arrival destination of Mars.

$$R_2 = M_{initial}/M_{final} = e^{\Delta V_2 / (g I_{sp})} \quad (5-3)$$

$M_{initial}/M_{final}$ is the ratio of the initially fully fueled mass to final mass. $M_{initial}$ is the vehicle mass just before the final arrival destination around or about Mars, and M_{final} , also referred to as M_{dry} , is the final mass of the spacecraft at the end of its successful mission. Initial estimates of M_{final} are used in evaluating $M_{initial}$ which in turn is used for $M_{w/c}$, the spacecraft mass, in STEP 2 below. ΔV is the amount of ΔV needed for the Mars insertion portion of the mission, ΔV_2 , plus the amount of ΔV needed for spacecraft stationkeeping. The Earth’s gravitational constant, g , is equal to 9.81 m/s^2 . I_{sp} is the specific impulse, or the amount of momentum gained per weight of fuel consumed.

With M_{final} and M_{initial} being the only unknown variables in the rocket equation, the use of the rocket equation involves initially estimating the final spacecraft, M_{final} , in order to evaluate M_{initial} . Estimates for M_{final} are obtained through spacecraft mass specifications from past, present and proposed missions and then values for M_{initial} are refined until the value of M_{final} is within design specifications designed by the user. And as an option, a design mass margin may be included in the rocket equation to increase the safety margin of a spacecraft mission.

Baseline Spacecraft Sizing for the Large and Small Spacecraft Classes

The baseline for the small and the large spacecraft class of the spacecraft mass breakdown by subsystem is used for comparison among scenarios. The subsystems included in the mass breakdown are payload, structure, thermal, power, telemetry, tracking and commanding (TT&C), attitude determination and control subsystem (ADCS), and the propulsion subsystem. Table 19 below details the spacecraft dry mass sizing for the large and small satellite classes categorized into seven general spacecraft subsystems with each subsystem listed as a percentage of the spacecraft dry mass. The percentage values for the small spacecraft class in Table 19 were derived from Svitek's paper on the Mars Relay Satellite and the values for the large spacecraft class were

obtained from Wertz and Larson's textbook on the TDRS Satellite.^{6, 7} Since both of the derived satellites are communications based and are directly applicable to this study, no changes to the percentage values were necessary. With the baseline spacecraft mass breakdown in place, the sizing of each general subsystem is performed for both the small and large spacecraft classes.

Table 19. Summary of the Baseline Spacecraft Mass Breakdown compared among Spacecraft Classes.

Spacecraft Class	Subsystem Values as a Percentage of Spacecraft Dry Mass						
	Payload	Structure	Thermal	Power	TT&C	ADCS	Propulsion
Small	36.9	14.2	6.3	11.4	2.8	7.4	21.0
Large	24.56	28.03	2.78	26.36	4.07	6.17	6.92

There are tradeoffs between selecting the large and small spacecraft design. Small satellites have physically short thermal and load paths allowing less critical and structural design. Another advantage of small satellites is the availability of miniature technology that are widely built to exceed specifications requiring very little analytical effort to ensure structural and thermal integrity compared with conventional technology devices. And for small spacecraft there is less likelihood for its design to require special, costly thermal

⁶ Svitek, T., et al, "Mars Relay Spacecraft: A Low Cost Approach," Ninth Annual AIAA/Utah State University Conference on Small Satellites, Logan, Utah, (September 18-21, 1995).

⁷ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 2nd ed. (Torrance, CA: Microcosm Press, and The Netherlands: Kluwer Academic Publishers, 1992), 806.

structural technology devices such as high strength alloys, composites, and heat pipes. With small satellite designs there is some flexibility to sacrifice reliability for reduced cost and on-orbit mass, especially with the option available to invest in smaller launch vehicles. The possibility also exists to launch more mini-satellites in one launch and to launch more satellites than required to compensate for any possible failure risk.

Conversely, for the large spacecraft class, clearly a larger payload package can be designed. And although spacecraft designs may vary, the larger the spacecraft design is, the more efficient is payload support and less mass devoted to support structure, guidance, propulsion. With possible capability of supporting numerous missions using the large satellite design, the large satellite class provides potential cost savings in the long term particularly when multiple Mars missions are established and the amount of data rate transfers are high.

Baseline - small spacecraft class. The small spacecraft class is based on the Mars Relay Spacecraft (MRS).⁸ The small spacecraft would have a dry mass of only 42 kilograms and would be used to relay data

⁸ Svitek, T., et al, "Mars Relay Spacecraft: A Low Cost Approach."

from Mars to Earth at very low cost. The spacecraft is spin-stabilized with the spin axis pointing towards the Earth and has very few moving parts. The spacecraft body would be little more than a structure in the shape of a typical pitched home roof. The solar panel is body-mounted onto the roof-shaped structure with an optically transparent mesh high-gain antenna above the structure and the electronics and propulsion equipment tucked underneath the structure. Table 20 shows the general statistics to be used for comparison among the scenarios investigated in this study. Note that the baseline for the small spacecraft class is modest and relatively simple in design.

Table 20. Small Spacecraft Class Baseline Design Specifications shown.

Payload/Communications Design (worse case)	40 bps uplink data rate 6.3 kbps data relay downlink data rate 128 Mbit data relay buffer size
General Communications Design	X-band 37dB high gain antenna supporting between 4kbps to 80 kbps depending on the Earth-to-Mars distance. X-band 6dB low gain antenna supporting 10-40 bps low gain antenna for deep-space communications; UHF antenna for the relay spacecraft to Mars surface link at 8kbps. No antenna switching or reconfiguration. Two Small Deep Space Transponders. 65Mbit/sol from six landers in a 4-hour DSN.
Structure Mass	Propellant tank, solar panel with the attached high-gain antenna major structure components (total dry mass of 45kg).
Thermal Control	Active control only for hydrazine heaters.

Table 20. Small Spacecraft Class Baseline Design Specifications (cont.)

Power	Solar panels- single fixed GaAs/Ge from <10 W (critical) to 50 W. Li-ion battery is used as secondary power source.
Telemetry and Commanding	Integrated Spacecraft Controller (ISC) performs the following functions: <ul style="list-style-type: none"> • processes ground commands, • processes telemetry for downlink, • performs attitude control functions, • contingency and fault recovery functions, • serves as interface to all Radio Frequency (RF) units, • serves as interface to attitude sensors and propulsion hardware, • control battery charging and electric load switching.
Attitude Determination	V-slit star sensor and v-slit sun sensor.
Attitude Control	Body spinning and with spin axis being Earth pointing to 0.8 degree accuracy performed by the Integrated Sensor Control (ISC). Hydrazine fuel slosh is used for nutation damping.
Propulsion	53kg hydrazine monopropellant for axial propulsion maneuvers & attitude reorientation and spin rate control. Two large 90N thrusters are used for ΔV maneuvers Six small 1N thrusters for attitude maneuvers.

Baseline - large spacecraft class. The large spacecraft class baseline design is largely derived from the Earth-based Tracking and Data Relay Satellite (TDRS). As noted previously, the strength of this spacecraft class design is in the ability to provide large data transfer and at higher data transfer rates to and from Mars as well as provide simultaneous or multiple-access service. Thus the large spacecraft design is attractive in serving more users on a time-shared basis as well as having the user require less service time due to the higher data rates in data transmissions. Table 21 shows the general specifications for the large spacecraft class to be used for comparison among the scenarios investigated in this study. Specific details on data rate performance are discussed in Chapter 6 in this study.

Table 21. Large Spacecraft Class Baseline Specifications shown.

General Communications Design	X-band multi-phased array. With Multiple Access , the Number of Multiple Links per Spacecraft: 1 at 10 kbps (Fwd), 20 at 100kbps (Return)
Structure Mass	Propellant tank, solar panels, high-gain antenna, low-gain antenna, spacecraft bus major structure components (total dry mass of 450kg) 1565.7kg dry mass, 2150.9kg total loaded mass
Thermal Control	Passive control using insulating blankets, radiators, thermal coatings and finishes. Active control for hydrazine heaters and thermostatically controlled heaters.
Power	Solar panels- to provide up to 2250 W power, 715 W average power at 28 volts. NiH-Ion battery providing up to 50 Amp-Hr is used as secondary power source.
Telemetry and Commanding	Performs the following functions: <ul style="list-style-type: none"> • processes ground commands, • processes telemetry for downlink, • performs attitude control functions, • contingency and fault recovery functions, • serves as interface to all Radio Frequency (RF) units, • serves as interface to attitude sensors and propulsion hardware, • control battery charging and electric load switching.
Attitude Determination	Ring laser gyro (2), Star tracker (2), Sun sensor (2)
Attitude Control	Three-axis stabilized, one face towards Mars nadir, one axis in the direction of flight. Four momentum wheels. Thrusters used sparingly for momentum dumping
Propulsion	750kg hydrazine monopropellant with 10 year operational life for attitude control, unloading momentum wheels, and for ΔV and stationkeeping maneuvers. Two large 200N thrusters are used for ΔV maneuvers. Four small 50N thrusters for attitude maneuvers.

The following Tables 23 to 40 show a variety of spreadsheets of various spacecraft subsystems derived from the baseline designs. Since information of various mass elements was not readily available, use of the spreadsheet aids in determining relative values of the mass elements in obtaining reasonable values for the spacecraft dry mass, propellant mass needed for the upper stage, and the total mass of "The Stack". The information in these tables was derived from various vendors' information and the tables were used in gaining

confidence in the design class comparison. Then the spreadsheet was applied to the scenarios for overall comparison and ranking of the scenarios.

First the mass breakdown of the spacecraft dry mass of the spacecraft is investigated. In this study the spacecraft dry mass will consist of the following two spacecraft components: the communications payload and the spacecraft bus. Elements of the spacecraft bus consist of the structure, the thermal components, the electrical power subsystem, the telemetry, tracking and control components, the attitude determination and control subsystem, and the inert propellant and propellant hardware.

The sizing of the spacecraft subsystem for each scenario is far from being an exact science since the many factors involved affect each other. An increase in power margin may mean enhancing the transmitter power, which in turn may mean improving the communication link margin and enabling a higher data rate for increased science data return. But despite some subjectivity involved in determining masses, spacecraft sizing is restricted by items such as the available launch vehicle, the launch window opportunities, and the amount of resources available on Earth to support all scenarios. An example of limited Earth resources is the limited availability of the Deep Space Network (DSN) in collecting and relaying information to and from Mars due to the DSN's heavy schedule with other users such as the Galileo, Ulysses, Cassini, Mars Global Surveyor, and

Voyagers 1 and 2 missions. Thus as an example, the design of the relay satellite may include up to 24 hours of memory storage buffering satellite data needed to be downloaded in 60 minutes via DSN.

Spacecraft Payload Subsystem Sizing

The payload subsystem in this study contains mainly the communications equipment to provide universal data transmission to and from Mars to Earth. Elements in the payload subsystem include the following general items: communications antennas, transponders, filters and diplexers, and amplifiers. For the small satellite class, the antenna configuration includes an X-band high gain antenna and an UHF low gain antenna. The large satellite class antenna design can vary depending on the scenario of interest and may consist of the following antenna designs: parabolic, horn, helix, waveguides, hemispherical or omnidirectional as contingency antennas, and coaxial antennas. Both small and large spacecraft designs have at least two transponders for redundancy. The number of filters, diplexers, and amplifiers is as few as two each for the small satellite class, whereas the large satellite class may have considerably more filters, diplexers, and amplifiers as the mass and physical dimensions can accommodate.

The first step in the payload design is determining the sizing of the parabolic antenna, which would likely contain the bulk of the payload subsystem mass and dimension. Table 22 shows the relationship between the parabolic antenna mass and size. An observation made from Table 22 is that due to spacecraft moment of inertia considerations affecting antenna pointing accuracy, an antenna sizing constraint was imposed on the antenna design in this study. Hence for the small spacecraft class, antenna designs greater than one meter in diameter are not considered. And for the large spacecraft design, mass as well as moment of inertial limitations restrict the antenna diameter to five meters. This maximum antenna diameter constraint may be relaxed as future technology produces lighter and stronger antenna designs.

Table 22. Generic Parabolic Antenna Mass and Sizing Relationship.

Parabolic Antenna Mass (kg)	Diameter (m)	Area (m²)
1.02	0.5	0.20
2.30	0.75	0.44
4.09	1	0.79
9.20	1.5	1.77
13.25	1.8	2.54
36.80	3	7.07
102.22	5	19.63
408.89	10	78.54

With the antenna sizing determined for both spacecraft classes, an overview of the baseline payload design is compared for both spacecraft classes. Again note that due to dimensional constraints for the small spacecraft class design, the component quantity is

limited to a total of eight components in the payload subsystem design. For the large spacecraft class there is greater number and variation of components among scenarios in the payload design. Table 23 displays a generic overview of the payload design for both spacecraft classes, although the total quantity of components for the large spacecraft class does vary among the different scenarios.

Table 23. Overview of the Baseline Payload Design Comparisons is shown for the Large and Small Spacecraft Class among Scenarios.

Small Satellite		Large Satellite	
<u>Payload</u>	<u>Quantity</u>	<u>Payload</u>	<u>Quantity</u>
Antennas		Antennas	
X-band - Earth bound (~ 1.0 m)	1	Parabolic - Earth (5m)	1
UHF - Mars bound	1	Mars bound antenna(s)	1 - 2
		Cross-link	1
		Waveguide	2
		Omnidirectional	2
		Coax	2
	<u>Quantity</u>		<u>Quantity</u>
Transponders	2	Transponders	3
Filters, Diplexers	2	Filters, Diplexers	3
Amplifiers		Amplifiers	
Solid state 5 W	2	Relay-Earth 5 W	10
		Cross-Link 2 W	10
Payload Quantity Total	8	Payload Quantity	35 -36
		Totals	

Table 24 shows the baseline payload subsystem design for the small spacecraft class among the scenarios in this study. The payload design specifications were largely derived from the spin-stabilized Mars Relay Spacecraft design.⁹ Much of the payload mass is from

⁹ Svitek, T., et al, "Mars Relay Spacecraft: A Low Cost Approach."

the antenna and transponder, while the bulk of the power requirement is from the amplifier. With exciting new technologies allowing spacecraft miniature designs in the future, greater variation in payload design for the small spacecraft can potentially be afforded.

Table 24. Small Satellite Class Payload Sizing is compared among Scenarios.

Payload	Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)
Antennas					
X-band - Earth (~1 m)	1	4		4	
UHF - Mars bound	1	1		1	
Transponder	2	2	6.5	4	6.5
Filters, Diplexers	2	1		2	
Amplifiers					
Solid State 5 W	2	1	40	2	40
Payload Totals	8			13	46.5

In contrast to the payload design for the small spacecraft class, there is greater variation in payload design for the large spacecraft class which also allows for greater data transmission that can be collected on Earth, with multiple antennas and crosslinks in handling multiple uplink and downlink signals. This is especially true for the antenna design for the relay spacecraft to Mars link defined as the downlink in this study. For the halo scenario, a 1.8 meter antenna for is selected at a 30 GHz operating frequency, and for the aerosynchronous scenario a horn design is chosen to reduce the amount of scattered

transmitter energy at 15 GHz operating frequency. A helix antenna design is used for the low and medium Mars orbit scenario to spread the signal wide to accommodate their low altitudes at a 400 MHz operating frequency. And finally for the common-period inclined scenario a combination of a helix and horn design operating during the periapse and non-periapse portion of the relay orbit respectively at 2 GHz. Table 25 through Table 28 shows the baseline payload subsystem design for the large spacecraft class among the scenarios in this study.

Table 25. Large Satellite Class Payload Sizing - Halo Orbiter Scenario.

Halo Orbiter					
<u>Payload</u>	<u>Quantity</u>	<u>Mass (kg)</u>		<u>Total (kg)</u>	<u>Total Power (W)</u>
Antennas					
Parabolic - Earth (5m)	1	102.2		102.2	
Parabolic - Mars (1.8m)	1	13.25		13.25	
Cross-link	1	9.2		9.2	
Waveguide	2	1.4		2.8	
Omnidirectional	2	0.25		0.5	
Coax	2	0.5		1	
Transponder	3	4.45		13.35	
Receiver	3		5		15
Transmitter	3		20		60
Filters, Diplexers	3	1.2		3.6	
Amplifiers					
Relay to Earth (20W)	10	1.2	60	12	600
Crosslink (2W)	10	1.2	20	12	200
Payload Totals	35			169.9	875

Table 26. Large Satellite Class Payload Sizing - Aerosynchronous Scenario.

<u>Aerosynchronous</u>					
<u>Payload</u>	<u>Quantity</u>	<u>Mass (kg)</u>		<u>Total (kg)</u>	<u>Total Power (W)</u>
Antennas					
Parabolic (5m)	1	102.2		102.2	
Horn	1	0.5		0.5	
Waveguide	2	1.4		2.8	
Hemispherical	2	0.4		0.8	
Coax	2	0.5		1	
Transponder	2	4.45		8.9	
Receiver	2		5		10
Transmitter	2		10		20
Filters, Diplexers	2	1.2		3.6	
Amplifiers					
Relay to Earth (5W)	10	1.2	30	12	300
Crosslink (2W)	10	1.2	20	12	200
Payload Totals	32			143.8	530

Table 27. Large Satellite Class Payload Sizing is shown for the Low Mars Orbit and Medium Mars Orbit Scenario.

<u>Low Orbit & Medium Orbit</u>					
<u>Payload</u>	<u>Quantity</u>	<u>Mass (kg)</u>		<u>Total (kg)</u>	<u>Total Power (W)</u>
Antennas					
Parabolic (5m)	1	102.2		102.2	
Horn	1	0.5		0.5	
Waveguide	2	1.4		2.8	
Hemispherical	2	0.4		0.8	
Coax	2	0.5		1	
Transponder	2	4.45		8.9	
Receiver	2		5		10
Transmitter	2		10		20
Filters, Diplexers	2	1.2		3.6	
Amplifiers					
Relay to Earth (5W)	10	1.2	30	12	300
Crosslink (2W)	10	1.2	20	12	200
Payload Totals	32			143.8	530

Table 28. Large Satellite Class Payload Sizing is shown for the Common Period Inclined Scenario.

Common Period Inclined					
	Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)
Antennas					
Parabolic (5m)	1	102.2		102.2	
Horn	1	0.5		0.5	
Helix	1	0.5		0.5	
Waveguide	2	1.4		2.8	
Hemispherical	2	0.4		0.8	
Coax	2	0.5		1	
Transponder	2	4.45		8.9	
Receiver	2		5		10
Transmitter	2		10		20
Filters, Diplexers	3	1.2		3.6	
Amplifiers					
Relay to Earth (5W)	10	1.2	30	12	300
Crosslink (2W)	10	1.2	20	12	200
Payload Totals	34			144.3	530

Among the scenarios in the large spacecraft class a baseline five-meter antenna design is used for the relay spacecraft to Earth link throughout this study. Future work considerations may also include feasibility studies of the spacecraft design with variations in payload design. However, this baseline is sufficient for comparison of scenarios in this study.

Spacecraft Structure Subsystem Sizing

Table 29 shows the spacecraft structure sizing analysis. This sizing among scenarios is based on 14.2 percent of dry mass and 38.5 percent of spacecraft mass for the small

satellite class. For the large satellite class 28.0 percent of dry mass and 114.1 percent of spacecraft mass among scenarios is applied in the analysis. More exotic materials for the structure design may be needed to satisfy the very small mass for the small spacecraft design, while the large spacecraft design may work well with conventional materials for the spacecraft structure.

Table 29. Small and Large Satellite Class Structure Sizing are compared among Scenarios.

	Halo Orbiter	Aerosynchronous	Low Mars Orbit	Medium Mars Orbit	Common Period Inclined
Structure Subsystem mass (kg)					
Small Spacecraft Class	5.00	5.00	5.00	4.81	7.15
Large Spacecraft Class	193.90	164.12	164.12	164.12	164.69

Spacecraft Thermal Subsystem Sizing

Table 30 shows the thermal sizing among scenarios for the small class based on a 6.3 percent of dry mass, and the thermal sizing among scenarios for the large satellite class based on a 2.78 percent of dry mass.^{10,11} Passive thermal control most likely dominates the small spacecraft design limited to thermal blankets, although 0.1 watts is allotted to power a thermal device with very little power draw. For the large spacecraft design,

¹⁰ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 806.

¹¹ Svitek, T., et al, "Mars Relay Spacecraft: A Low Cost Approach."

active thermal control such as the use of heat pipes, thermal louvers, and other active thermal control as well as multi-layer insulation (MLI) can be incorporated in this design. Further analysis is warranted towards a final design for both spacecraft classes.

Table 30. Small Satellite Class Thermal Sizing are compared among Scenarios.

	Halo Orbiter	Aerosynchronous	Low Mars Orbit	Medium Mars Orbit	Common Period Inclined
Small Spacecraft Class					
Subsystem mass (kg)	2.20	2.20	2.20	2.12	3.15
Subsystem Pwr (W)	0.1	0.1	0.1	0.1	0.1
Large Spacecraft Class					
Subsystem mass (kg)	19.23	16.28	16.28	16.28	16.33
Subsystem Pwr (W)	38.15	37.10	37.10	37.10	37.10

Spacecraft Power Subsystem Sizing

Table 31 shows the power sizing for the small satellite class among the scenarios, and Table 32 shows the power sizing for the large satellite class among the scenarios. The following design parameters were applied to all scenarios: the battery drain efficiency is to be 90 percent; and the solar panel efficiency is to be 18 percent. A Mars solar constant of 586 W/m^2 is used in the solar panel sizing calculations.

Table 31. Small Satellite Class Power Sizing are compared among Scenarios.

	Halo Orbiter	Aerosynchronous	Low Mars Orbit	Medium Mars Orbit	Common Period Inclined
Battery Design					
Operational Power Load (W)	23.00	-	-	-	23.00
Eclipse Power Load (W)	-	23.00	23.00	23.00	-
Max duration (sec)	7200	5000	2600	2700	7200
Desired DOD	95%	95%	50%	50%	95%
Required Battery Capacity (W*Hr)	44	30	30	31	44
Desired Number of Batteries	3	3	3	3	3
Per Battery Capacity (W*Hr)	15	10	10	10.35	15
Battery Mass {NiH2} (kg)	2.904	2.019	1.992	2.07	2.904
Recharge time (Hr)	5.81	6.05	1.20	11.03	29.42
Solar Panel Design					
Required Solar Panel Area (m ²)	0.45	0.60	0.80	0.80	0.80
Minimum Solar Panel Mass (kg)	1.89	2.52	3.39	3.39	3.11
Solar Panel Mass (kg)	3.9	4.0	4.2	4.2	4.2
Power Control Unit Mass (kg)	1.26	1.26	1.70	1.70	1.55
Regulator Mass (kg)	1.58	1.58	2.12	2.12	1.94
Regulator Pwr Loss (W)	8.5	8.77	11.80	11.80	10.80
Wiring Mass (kg)	1.62	1.62	1.62	1.56	2.32
Wiring Pwr Loss (W)	2.19	2.19	2.95	2.95	2.70
Power Requirement with 15% Margin (W)	65.55	71.30	94.3	88.55	94.3
Power Capability with 1.0 m ² Solar Panel (W)	70.759	71.384	67.139	70.134	70.134

Table 32. Large Satellite Class Power Subsystem Sizing are compared among Scenarios.

	Halo Orbiter	Aero- synchronous	Low Mars Orbit	Medium Mars Orbit	Common Period Inclined	Phobos Lander	Deimos Lander
Battery Design							
Operational Power Load (W)	1125	n/a	n/a	n/a	n/a	n/a	n/a
Eclipse Power Load (W)	n/a	1097	1345	1251	1090	1217	1093
Max recharge duration available (sec)	2000	4500	2000	2000	7200	2050	5533
Desired DOD	95%	95%	50%	50%	95%	50%	95%
Required Battery Cap (W*Hr)	592	1300	1345	1251	2065	1248	1592

Table 32. Large Satellite Class Power Subsystem Sizing (cont.)

	Halo Orbiter	Aero- synchronous	Low Mars Orbit	Medium Mars Orbit	Common Period Inclined	Phobos Lander	Deimos Lander
Desired # Batteries	3	3	3	3	3	3	3
Batt Capacity (W*Hr)	197	433	448	417	688	416	531
Battery Mass {NiH2} (kg)	13.17	29	30	28	46	28	35
Recharge time (Hours)	11.85	21.86	1.28	1.08	68.84	6.93	28.43
Solar Panel Design							
Required Solar Panel Area (m ²)	20.8	13.0	26.5	21.4	12.6	14.3	24.47
Solar panel Mass (kg)	87.9	54.9	111.9	90.3	53.2	60.47	103.3
Power Control Unit Mass (kg)	43.9	27.5	55.9	45.2	26.6	30.2	27.4
Regulator mass (kg)	54.9	34.4	69.9	56.4	33.3	37.8	34.2
Regulator Power Loss (W)	305.7	191.1	389.1	314.1	185.1	191.1	191.1
Wiring mass (kg)	87.9	54.9	111.9	90.3	53.2	60.5	55.0
Wiring Pwr Loss (W)	76.4	49.8	97.3	78.5	46.3	47.8	47.8

For the halo orbiter case, the relay satellites are in constant line-of-sight with the Sun. Although the relay satellites do not experience any eclipse duration, a two-hour battery reserve is included in the power design to account for single upset events when Earth-to-spacecraft contact is lost and to accommodate for a spacecraft tumble recovery. And since neither L_1 nor L_2 spacecraft experience any amount of eclipsing, a 95 percent battery depth of discharge is placed on battery management, as well as the power load based on the operational power load. The operational power load is the amount of power needed to operate the payload subsystem since both the solar arrays and the batteries can constantly supply spacecraft power for spacecraft housekeeping. But in the event that

spacecraft attitude control is lost, payload operations would cease until attitude control is regained and stabilized. And since the operational power load used in the halo orbiter case is less than the eclipse power load used in all the other scenarios, the halo orbiter scenario has a more favorable power requirement in every power parameter studied. Also since these orbiters are in constant visibility with the Sun, a longer discharge time can be afforded and battery charge-and-discharge cycling is at a minimum for the life of the mission.

The aerosynchronous case is the next favorable type which experiences fewer depth-of-discharge cycles in its mission, ranging from 1000 cycles to 3000 cycles, thus allowing a greater depth-of-discharge of 50% in the power design. And since an aerosynchronous orbit is similar to a geosynchronous orbit with an eclipse time of about 70 minutes out of over 24 hours, longer battery recharge duration can be afforded.

The remaining scenarios have been observed to be less favorable for the small spacecraft class. The low Mars, the medium Mars orbit, and the common period inclined scenarios all require battery recharge times far longer than their respective orbit periods which greatly compromises mission battery life. This design shortcoming is observed despite a 50-watt maximum total power limitation imposed on these three scenarios in attempting to decrease the battery recharge time requirements.

Spacecraft sizing analysis summary. Tables 33 and 34 summarize the total spacecraft sizing for the small and large satellite classes respectively. Tables 33 and 34 itemize by subsystem the payload, structure, thermal, power, TT&C, ADCS, and propulsion subsystem sizing discussed by individual subsystems previously shown in Tables 23 to 32. Calculations for the subsystems show variations that are directly related to the payload subsystem characteristics. Note that the percent payload power is not applicable to the small satellite class due to physical spacecraft limitations observed in this study.

For the small spacecraft class, little variation in the subsystem mass is evident among the scenarios as shown in Table 33 below. The strict physical sizing limitation for the small spacecraft class leaves little room for variation between scenarios in its spacecraft design. However, future spacecraft design trending towards miniaturization can readily compensate for the physical size limitations and thereby provide sufficient variation in spacecraft design. Thus the small spacecraft class design may likely become a popular choice for future Mars communication relay space programs.

Table 33. Summary of Small Satellite Class Subsystem Parameters are compared among Scenarios.

	Halo Orbiter	Aerosynchronous	Low Mars Orbit	Medium Mars Orbit	Common Period Inclined
Subsystem Mass (kg)					
Payload	13.00	13.00	13.00	13.00	13.00
Structure	5.00	5.00	5.00	5.00	5.00
Thermal	2.20	2.20	2.20	2.20	2.20
Power	4.00	4.00	4.00	4.00	4.00
TT&C	1.00	1.00	1.00	1.00	1.00
ADCS	2.60	2.60	2.60	2.60	2.60
Propulsion	7.40	7.40	7.40	7.40	7.40
<i>Total s/c mass (kg)</i>	35.2	35.20	35.20	35.20	35.20
<i>Mass w/ Margin (15%)</i>	40.48	40.48	40.48	40.48	40.48
Subsystem Power (Watts)					
Payload	46.5	46.5	46.5	46.5	46.5
Thermal	0.1	0.1	0.1	0.1	0.1
Power	10.0	10.0	10.0	10.0	10.0
TT&C	0.2	0.2	0.2	0.2	0.2
ADCS	0.1	0.1	0.1	0.1	0.1
Propulsion	0.1	0.1	0.1	0.1	0.1
Batt Recharge Power Margin	5.0	5.00	30.00	30.00	5.00
Total Spacecraft Power (Watts)	62.0	62.00	87.00	87.00	62.00
Power with Margin (15%)	71.30	71.30	100.05	100.05	71.30
Battery Recharge Time (Hours)	8.72	6.05	1.00	1.04	8.72

The power design for the large satellite class has more flexibility in accommodating scenario features and shortcomings than for the small satellite class. But unlike the small spacecraft class, virtually all scenarios studied can be designed to accommodate the large spacecraft class. Conversely, the low Mars orbit, medium Mars orbit, and the common period inclined scenarios can satisfy stringent battery recharge time requirements.

Table 34. Summary of Large Satellite Class Subsystem Parameters are compared among Scenarios.

	Halo Orbiter	Aerosynchronous/ Phobos / Deimos	Low Mars Orbit	Medium Mars Orbit	Common Period Inclined
Subsystem Mass (kg)					
Payload	169.9	143.8	143.8	143.8	144.3
Structure	193.9	164.1	164.1	164.1	164.7
Thermal	19.2	16.3	16.3	16.3	16.3
Power	295.8	200.6/217.9/200.2	379.5	310.0	212.2
TT&C	28.2	23.8	23.8	23.8	23.8
ADCS	42.7	36.1	36.1	36.1	36.3
Propulsion	47.9	40.5	40.5	40.5	40.7
Total s/c mass (kg)	797.5	625.3	804.2	734.7	638.4
Total s/c mass with Margin (15%)	917.1	719.1	924.8	844.9	734.1
Subsystem Pwr (W)					
Payload	875.0	530.0	530.0	530.0	530.0
Thermal	61.3	37.1	37.1	37.1	37.1
Power	382.2	238.9	486.9	392.7	231.4
TT&C	87.5	53.0	53.0	53.0	53.0
ADCS	393.3	238.5	238.5	238.5	238.5
Propulsion	61.3	37.1	37.1	37.1	37.1
Batt Recharge Pwr Mgn	50.0	60 / 180 / 56	1050.0	675.0	30.0
Total s/c pwr (W)	1910.9	1194 / 1314 / 1190	2432.1	1963.4	1157.1
Power with Margin (15%)	2197.6	1373 / 1511 / 1369	2796.9	2557.9	1330.7

Compared to the small spacecraft class, a larger variation in spacecraft subsystem design is observed for the large spacecraft class. From Table 34 the common period inclined scenario showed the lowest power requirement, followed by the Deimos relay spacecraft, the aerosynchronous orbiter, the halo orbiter, the Phobos relay spacecraft, the medium Mars orbit relay spacecraft, and finally the low Mars orbit relay spacecraft. The medium and low Mars orbit scenarios show a substantially larger power and battery recharge power margin.

STEP 2: Determining M_{prop} the Upper Stage Propellant Mass of “The Stack”

With the spacecraft mass breakdown determined for each scenario and spacecraft class, the amount of propellant mass required for each case is determined. Using equation (5-1) with equation (5-3a), an equation for M_{prop} , the amount of propellant mass required for ΔV_1 , is obtained. Equation (5-5) below calculates the propellant mass required for the ΔV_1 required in low Earth orbit to inject onto the heliocentric trajectory

$$M_{prop} = \frac{[M_{w/c} + M_{pla} + M_{pio} + M_{FE}] (R_1 - 1) + (1 - f_s) M_{margin}}{[1 + f_{pi} - R_1(f_{pi} + f_s)]} \quad (5-5)$$

$M_{w/c}$ is the spacecraft mass after the ΔV_1 maneuver has occurred and the upper stage has been jettisoned. The payload adaptor factor, $f_{w/c}$, is measured as a fraction of spacecraft mass, typically of value 0.03. The value of $f_{w/c}$ multiplied by $M_{w/c}$ gives the payload adapter mass. M_{pio} is the dry mass of the inert propellant system that includes propellant tanks, nozzles, and propellant lines applicable to the upper stage-to-payload stack. M_{FE} is the flight equipment mass that includes the batteries, the solar panels and other flight equipment components. The spacecraft structure factor, f_s , is measured as a fraction of the total or wet mass, typically of value 0.18. The value of f_s multiplied by M_{WET} , the mass of the upper stage-to-payload stack at launch, gives the spacecraft structure mass. M_{margin}

is the mass margin, which is spacecraft mission dependent. The inert propellant factor, f_{pi} , is measured as a fraction of the mass of the upper stage, typically of value 0.10. The value of f_{pi} multiplied by M_{prop} gives the inert propellant hardware mass. R_i is the ratio of the initially fully-fueled “Stack” mass to the final fuel-dry mass ($M_{initial}/M_{final} = e^{\Delta V_i / (g_{isp})}$) similar to the description in STEP 1.

From the two steps outlined above, a spreadsheet was used to determine mass components of “The Stack.” Using Equations (5-3), (5-4), and (5-5), values from specific vendors and are used to compare the wet mass values, propellant mass values, and upper stage dry mass values from STEP 1 and STEP 2 described previously in this study.

Sizing Baseline of the Upper Stage Mass for the Spacecraft Classes

Vendor information of upper stage component mass specifications is used as benchmark values in sizing the mass of the upper stage for the small, medium and large upper stage classes. Appendix I and J show spreadsheets outlining the baseline mass characterization of “The Stack” for the small and the large satellite classes respectively. These two spreadsheets aid in the process in determining the mass components of the upper stage for each spacecraft class.

First the Rocket Equation is used for the Mars insertion portion of the mission. The Rocket Equation used in this instance involves determining the ratio of the initial and final masses of the spacecraft as well as $\Delta V_2 + \Delta V_{\text{stationkeeping}}$, the amount of ΔV needed for a Mars insertion orbit and for spacecraft stationkeeping once the final orbit is achieved. From the mass ratio information the payload mass, in this instance the spacecraft mass, can be determined upon the selection of a final dry mass value. In Table 5-3, the spacecraft dry mass is determined from spacecraft sizing which the details are discussed earlier in this chapter. The spacecraft fuel mass is then determined from the difference between the initial and final masses. Thereafter the Rocket Equation is repeated using the ratio of the wet mass, including margin, to the dry mass of “The Stack” along with ΔV_1 , the amount of ΔV needed for the spacecraft to leave Earth parking orbit and enter into its interplanetary trajectory.

Once the Rocket Equation is used for determining the respective initial and final mass ratios, the following “Stack”-related masses, the propellant mass, the dry mass of the upper stage, and the total wet mass of “The Stack.” These “Stack”-related masses are functions to varying degree of the payload adaptor mass fraction, f_{pla} , the mass of the inert propellant, M_{pio} , the mass of the flight equipment, M_{fe} , the structure mass fraction, f_s , and the inert propellant fraction f_{pi} . A constraint that can be observed from the spreadsheets is that the upper stage launch vehicle design must not exceed the wet mass of “The Stack” in

order for the launch vehicle to achieve enough launch energy to escape the Earth's gravity and to start its interplanetary journey to Mars. This constraint can be used in the grading and weighting of the scenarios towards the cost model.

With vendor information as well as information from published papers a baseline communications mission design is developed for the two general spacecraft classes. Reasonable values for the various terms in the rocket equation and equations for itemized mass were obtained. Then using the values from the terms values for upper stage propellant mass, dry mass of the upper stage and total mass of "The Stack" are calculated for each scenario. Some grading consideration is made based on variation from the baseline design. Based on the ΔV , C_3 , spacecraft mass and propellant mass evaluation information discussed above, the common-period inclined relay scenario exhibited the most favorable sizing characteristics.

Spacecraft Sizing Driver Summary

Preliminary results show that the spacecraft sizing driver is largely dependent upon the amount of ΔV needed to orbit at or about Mars. This relative value has been determined to be about the same among scenarios and with both spacecraft classes based on the

general Earth departure and Mars arrival destination points. The only exception is the halo orbiters where the arrival destination is with their respective L_1 and L_2 libration points. The L_2 halo orbiter has a ΔV compared to the other scenarios. However, the halo orbiter about L_1 has a slightly lower ΔV compared to the other scenarios. Since the derivation of this driver is more rigorously determined this driver is weighted higher relative to the drivers in the Cost Model.

CHAPTER 6

The Communications Performance Parameter

$$J = w_{\$}(\$) + w_{Size}(Size) + w_{Comm}(Comm) + w_{Life}(Life) + w_{Plan}(Plan)$$

This chapter analyzes the communications performance parameter of the cost equation largely developed by Mark Danehy.¹ An effective communications relay satellite network must be able to send Mars science data to Earth and receive commands from Earth either by radio carrier or by laser carrier that is capable of relaying and receiving data in support of design requirements. Hence the communications link capability is a measure of the maximum data rate possible that can be passed along the communications link given parameters. These parameters include relay network orbit configuration characteristics, initial bit error rate requirement, relay satellite antenna size and design, transmitter power, and the distance between the transmitter on board the relay satellite and the receiver on Earth, and the distance between the relay transmitter and the Mars receiver as applicable. These features drive solar panel and antenna sizing, spacecraft mass, and launch vehicle selection.

A baseline design was developed and used as a benchmark for relative grading among the scenarios. This communications performance analysis includes the communications

¹ Danehy, M., "Martian Communications Network Design Trade Study," Masters Thesis, San José State University, (December 1997).

link capability, and the communications satellite visibility. The communications link capability is defined as the highest data rates that can be passed along the link with a given error rate. The communications satellite visibility analyzes the capability of a specified site on Mars to view at least one satellite in the constellation at or above a specified elevation.

The Baseline Network Communications Design

The basic equation used in the baseline of a digital data link to be used in communications network design is described by equation (6-1) below:

$$E_b/N_o = P L_t G_t L_s L_a G_r / k T_s R \quad (6-1)$$

E_b/N_o is the ratio of the received energy-per-bit to noise-density, also referred to as the signal-to-noise ratio. P is the transmitter power, L_t is the transmitter-to-antenna loss, and G_t is the transmit antenna gain. L_s is the space loss, L_a is the transmission path loss, and G_r is the receive antenna gain. The value k is Boltzmann's constant, T_s is the system noise temperature, and finally R is the data rate.

A baseline network communications design was developed by Mark Danehy, in cooperation with the author of this study. The baseline communications design involves low rate telemetry, medium rate telemetry and voice, high rate telemetry and video, and

high rate user upload services. The low rate telemetry service would generally be used for monitoring relay spacecraft State of Health (SOH). The relay spacecraft State of Health telemetry information would be relayed to the Earth for analysis by personnel on Earth. The medium rate telemetry service would be used to relay experiment data or high-speed sensor samples such as accelerometer or attitude sensor data, from Mars to Earth. The medium rate service could also provide digitized voice capability for manned missions. The high rate telemetry service would support high-speed stored data dumps, video transmissions, as well as high rate payload data that can be relayed from Mars to Earth. The high rate user upload service would provide special entertainment and academic information from Earth as a special feature for manned Mars missions. Table 35 outlines a high-level baseline communications design by service, defined data rate, and defined bit error rate of 1×10^{-7} (BER). This BER number is commonly used in various NASA and scientific papers in this field.²

Table 35. High-Level Baseline Communications Design.

Service	Data Rate	Bit Error Rate
Low Rate Telemetry	2000 bps	1×10^{-7}
Medium Rate Telemetry/Voice	64,000 bps	1×10^{-7}
High Rate Telemetry/Video	1,000,000 bps	1×10^{-7}
High Rate User Uploads	1,000,000 bps	1×10^{-7}

² Danehy, M., "Martian Communications Network Design Trade Study."

Next a communication network was defined with a sample combination of data services and the required data rates calculated. Table 36 describes a detailed matrix of communication services provided by a Mars communications relay network. The numbers in parentheses following the communication service type indicates the number of channels or means of data transmission. The number of channels in each type of service has been estimated based upon the projected mission requirements.

Table 36. The Baseline Communications Design by Service, Defined Data Rate, and Defined Bit Error Rate.

Communications Services (Number of Channels)	Data Rate	Earth to Mars		Mars to Earth	
		Earth to Relay Data Rate	Relay to Mars Data Rate	Mars to Relay Data Rate	Relay to Earth Data Rate
SOH Relay Telemetry	2000 bps				2000 bps
SOH Relay Commands	2000 bps	2000 bps			
Low Rate User Telemetry (30)	2000 bps			60 kbps	
Medium Rate User Telemetry (5)	64000 bps			320 kbps	
High Rate User Telemetry (2)	1 Mbps			2 Mbps	
User Commands (40)	1000 bps	40 kbps			
High Rate User Uploads(1)	1 Mbps	1 Mbps			
Total Data Rate		1.042 Mbps	1.040 Mbps	2.38 Mbps	2.382 Mbps

Design Drivers of the Link Analysis

This section describes some definitions of the design drivers used in the link analysis developed by Danehy, followed by a synopsis and summary of the communications link

analysis adapted from Danehy. The communications link analysis is then used in the Cost Model and applied for evaluation and comparison among scenarios.

Bit Error Rate (BER). The bit error rate is the primary design driver of the communications link analysis. The BER is defined as the probability of having one bit error in a bit stream of defined size. In this study, a BER of 1×10^{-7} , or one bit error per 10,000,000 bits, is used as design criteria specifying a tolerable error rate in data transmission applied to the network scenarios in this study. This BER rating will in turn provide a minimum value for the link margin calculations of the link analysis.

Antenna gain. The antenna gain is defined as the amount of change in transmitted or received power as a result of the radiated signal by the antenna. The gain is expressed either as a ratio of the antenna output power to that of an antenna which radiates power equally in all directions, or as the ratio of the received signal power to that of the signal power output to the receiver. Gain is generally expressed in decibels (dB) defined as $10 \log_{10}(P_o/P_i)$ which simplifies calculations by adding gains and subtracting losses as opposed to multiplying and dividing parameters.

Link margin. The link margin defines the amount of strength the received signal needs to be above signal noise for successful signal collection and decoding. As gain is expressed in dB, the link margin is also expressed in dB, where for example a 3 dB link margin would require that the received signal be at least twice as strong as the received noise, or a 2 to 1 ratio. The value for the link margin varies with the type of equipment used and the method used to encode the information on the carrier, which can range from 4 to 10 dB or higher. Calculations for the link margin in this study include using 1×10^{-6} as the required BER, and the Bi-Phase Shift Keying (BPSK) with the R-1/2 Viterbi decoding for the encoding method thereby requiring a minimum link margin of 5 dB to satisfy the BER requirement.

Noise temperature. Lastly, the noise temperature parameter is used to determine the amount of noise observed by the receiver. Thus the more noise entering the receiver, the lower the signal-to-noise ratio of the incoming signal, and the more errors that are introduced to the signal. In the succeeding link analysis from Danehy, four general noise temperatures were used. These include the relay satellite looking into deep space having a noise temperature of 100K, the relay satellite looking near the Sun outside the main beam of the antenna having a 300K noise temperature. A 400K noise temperature is given for a relay satellite pointed towards the Mars surface, and a 500K noise temperature for the relay satellite facing towards the Sun within the main beam of the antenna.

The Communications Link Analysis

In the communications link analysis calculations developed by Danehy, two cases were studied: a small unmanned science lander using a 3dB hemispherical antenna, thereby simplifying the lander antenna pointing requirements; and either a staffed science station or a large autonomous science lander using a 2 meter transmit-and-receive antenna. In this study, the information from Danehy is incorporated into the cost modeling analysis.

Three transmit frequencies of 1 gigahertz, 10 gigahertz, and 30 gigahertz were compared using various sizes of parabolic transmit relay satellite antenna and Earth-based parabolic receive antenna combinations through the signal-to-noise ratio condition and link data rate observations. Table 37 characterizes the various transmit antenna and receive antenna combinations and compares the maximum link data rate possible with a given receiver noise temperature and minimum signal-to-noise ratio of 5.7 dB based on the BPSK with R-1/2 Viterbi encoding method described previously. The effects of solar interference were considered; as the Sun approaches closer to the communication path line-of-sight, the noise generated by the Sun increases, thereby degrading the received signal and forcing the data rate lower until an adequate signal-to-noise ratio is temporarily reached. This adjustment in signal-to-noise ratio continues until the Earth passes on the far side of the Sun from Mars, and significant solar interference renders the link

unavailable as the Sun passes directly between the relay satellite and the Sun which lasts about two weeks and occurs approximately every 700 days. Therefore in this two week period there would be a need for spacecraft or landers to be able to operate autonomously as any Mars mission would be out of contact with Earth. Thus the last two columns of Table 37 show maximum possible data rates with a 100K noise temperature applied to instances where the link is outside the Sun line-of-sight path, and a 300K noise temperature applied to where the communication line-of-sight path becomes affected by solar interference respectively.

Table 37. Transmit Antenna and Receiver Antenna Characteristics for Relay Satellite-to-Earth link with a 30-Watt Transmitter on the Relay Satellite.³

Transmit Antenna Size	Receiver Antenna Size	Transmit Antenna Frequency	Maximum Data Rate to Support 5dB SNR, 100K Noise Temperature	Maximum Data Rate to Support 5dB SNR, 300K Noise Temperature
5 meter	15 meter	1 GHz	0.3 kbps	90 bps
		10 GHz	30 kbps	9 kbps
		30 GHz	300 kbps	90 kbps
5 meter	30 meter	1 GHz	1 kbps	0.5 kbps
		10 GHz	100 kbps	50 kbps
		30 GHz	1 Mbps	500 kbps
10 meter	15 meter	1 GHz	1 kbps	0.5 kbps
		10 GHz	100 kbps	50 kbps
		30 GHz	1 Mbps	500 kbps
10 meter	30 meter	1 GHz	6 kbps	1.1 kbps
		10 GHz	600 kbps	110 kbps
		30 GHz	6 Mbps	1.1 Mbps

³ Danehy, M., "Martian Communications Network Design Trade Study."

From Table 37, the 10 meter transmit antenna, 30 meter receive antenna, 30 GHz frequency example exceeds the goal of a communication network being able to support at least a 2.4 Mbps data rate (value highlighted in Table 37 in bold). All other examples and particularly the cases where solar interference reduces data rate transmission, a reduction in communication services would occur, particularly with the high rate services. However, these calculations were for Earth to Mars distances at maximum opposition, and as the planets' orbital paths move away from maximum opposition, the distance between Earth and Mars will decrease, reducing the loss between the transmitter and receiver.

Next, transmitter power requirements for downlink to Mars were studied for various relay satellite antenna sizes fixing values for SNR of 5dB, the receiver antenna size of 15 meters, a data rate capability of 1 Mbps, and a noise temperature of 100K. A 30-watt requirement would be needed for a 10-meter antenna on a relay satellite, 50 watts for a 7.5-meter relay antenna, and 110 watts for a 5-meter relay antenna. If the receiver antenna size were 30 meters instead of 15 meters, the power requirement drops to 7.5 watts, 12.5 watts, and 30 watts for the 10-meter, 7.5-meter, and 5 meter antenna on the relay satellite respectively. Clearly the larger the antenna size on the relay satellite, the smaller the power requirement. However, the size of the spacecraft antenna may be restricted to the size of the fairing diameter of the launch vehicle used unless the antenna has a stowing capability within the rocket fairing and deployed later in its cruise phase of

the mission.⁴ However the stowing of an antenna would generally increase the complexity in the antenna design. This example briefly illustrates the design and decision process of antenna sizing for a communication relay satellite.

Adapted from Danehy's work, a comparison summary of the communications analysis is shown below in Table 38 through Table 41. Table 38 shows the communications analysis for the small spacecraft class, downlink; Table 39 for the small spacecraft class, uplink; Table 40 for the large spacecraft class, downlink; and Table 41 shows the communications analysis for the large spacecraft class, uplink. In this study the downlink is defined as the Earth - relay spacecraft - Mars surface communications link, while the uplink is defined as the Mars surface - relay spacecraft - Earth communications link. This analysis includes the baseline antenna design, the antenna size, the operating frequency, and the day and night data rates. The day and night data rates are calculated based on the corresponding day and night noise temperatures as discussed in Danehy's work. As implied by Danehy, the day and night data rates shown in these tables are minimum data rate values without solar interference since these values are based on worst-case calculations of a maximum Earth to Mars distance of 377 million kilometers. Solar interference adversely affects data transmission over a maximum period of 14 days and occurs once every Martian orbit. Over one Mars year of 687 days, the data rate values at

⁴ Danehy, M., "Martian Communications Network Design Trade Study."

the minimum Earth to Mars distance of about 57 million kilometers would be on the order of about 20 times greater than the minimum data rates listed in Table 38 through Table 41. Also from the combination of data rate and the percentage of communication network visibility information from Danehy an estimate of the total data throughput a relay spacecraft can provide over an extended period of time is performed. However, this total data throughput calculation is dependent also upon the amount of time each relay spends on both the Earthbound link and the Marsbound link. The data throughput calculation would be most difficult to estimate particularly with scenarios with orbiters of varying altitude. This applies to the case of the common period inclined orbiters with their highly elliptical orbits. As a result, this estimate would tend to least favor the common period inclined scenario using the data from Danehy. Future study should consider more realistic variations of data rates over the communications relay mission, where Earth to Mars distances vary between a minimum of about 57 million kilometers and 377 million kilometer maximum distance as well as more robust data throughput calculations. Despite these limitations the data rate information obtained from Danehy and the data throughput estimates are sufficient for purposes of scenario comparison in this study.

Table 38. Comparison Summary among the Scenarios from Communications Analysis for the Small Spacecraft Class Downlink.⁵

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated (Medium Mars Orbit Orbiter)	Martian Moon Landers
Antenna Design	Parabolic	Parabolic	Parabolic	Parabolic	Parabolic	Parabolic
Antenna Size	5 meters	5 meters	5 meters	5 meters	5 meters	5 meters
Operating Frequency	30 Ghz	15 Ghz	2.0 Ghz	0.4 Ghz	0.4 Ghz	15 Ghz
Day Data Rate	0.15 kbps	0.7 kbps	3.0 kbps	500 kbps	350 kbps	0.65 kbps - 0.75 kbps
Night Data Rate	0.15 kbps	0.9 kbps	6.0 kbps	950 kbps	650 kbps	0.85 kbps - 0.95 kbps

Table 39. Summary among the Scenarios from Communications Analysis compared, Small Spacecraft Class Uplink.⁶

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated (Medium Mars Orbit Orbiter)	Martian Moon Landers
Antenna Design	Parabolic (1.8m diam)	Horn	Horn & Helix	Helix	Helix	Horn
Transmitter Power Output	20 Watts	4 Watts	4 Watts	4 Watts	4 Watts	4 Watts
Transmitter Power Output	4 Watts	4 Watts	4 Watts	1 Watts	1 Watts	4 Watts
Operating Frequency	30 Ghz	15 Ghz	2 Ghz	0.4 Ghz	0.4 Ghz	15 Ghz
Day Data Rate	0.03 kbps	0.7 kbps	3.0 kbps	100 kbps	80 kbps	0.7 kbps
Night Data Rate	0.03 kbps	0.7 kbps	3.0 kbps	100 kbps	80 kbps	0.7 kbps

⁵ Danehy, M., "Martian Communications Network Design Trade Study."

⁶ Danehy, M., "Martian Communications Network Design Trade Study."

Table 40. Comparison Summary among the Scenarios of Communications Analysis for the Large Spacecraft Class Downlink.⁷

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated (Medium Mars Orbit Orbiter)	Martian Moon Landers
Antenna Design	Parabolic	Parabolic	Parabolic	Parabolic	Parabolic	Parabolic
Antenna Size	5 meters	5 meters	5 meters	5 meters	5 meters	5 meters
Transmitter Power Output	20 Watts	4 Watts	4 Watts	4 Watts	4 Watts	4 Watts
Operating Frequency	30 Ghz	15 Ghz	2 Ghz	0.4 Ghz	0.4 Ghz	15 Ghz
Day Data Rate	10000 kbps	13000 kbps	1500 kbps	8000 kbps	6000 kbps	13000 kbps
Night Data Rate	10000 kbps	25000 kbps	3000 kbps	11000 kbps	10000 kbps	25000 kbps

Table 41. Summary among the Scenarios from Communications Analysis is compared for the Large Spacecraft Class Uplink.⁸

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated (Medium Mars Orbit Orbiter)	Martian Moon Landers
Antenna Design	Parabolic (1.8m diam)	Horn	Horn & Helix	Helix	Helix	Horn
Transmitter Power Output	4 Watts	4 Watts	4 Watts	4 Watts	4 Watts	4 Watts
Operating Frequency	30 Ghz	15 Ghz	2 Ghz	0.4 Ghz	0.4 Ghz	15 Ghz
Day Data Rate	2500 kbps	15000 kbps	1500 kbps	3000 kbps	1500 kbps	15000 kbps
Night Data Rate	2500 kbps	15000 kbps	1500 kbps	3000 kbps	1500 kbps	15000 kbps

⁷ Danehy, M., "Martian Communications Network Design Trade Study."

⁸ Danehy, M., "Martian Communications Network Design Trade Study."

Preliminary findings from the link analysis. Initial findings from Danehy's work show that all scenarios studied are not capable of meeting the Mars to Earth specification of 2.3 Mbps in fulfilling a potential human mission to Mars. The requirement for the high rate user telemetry and video can be modified by reducing the number of simultaneous links from two to one, and the addition of data compression should allow accommodation of the remaining services as devised by Danehy. However, between the two spacecraft classes the data rate performance of the large spacecraft class is far more superior to that of the small spacecraft class.

The communications links for several constellation configurations have been compared to the baseline communications link design devised by Danehy. From Danehy's work, the ranking from the most favorable to the least favorable scenario for the small spacecraft class strictly from a data throughput standpoint are as follows:

1. The low Mars orbiter scenario
2. The medium Mars orbiter scenario
3. The common-period inclined scenario
4. The aerosynchronous scenario
5. The halo orbiter scenario
6. The Mars lunar lander scenario.

The data rates for the low Mars orbit and medium Mars orbit designs are consistently on the order of between 30 to over 1000 times greater than that of the remaining scenarios compared for the needs of a small station despite being transmitted at a fraction of the antenna power. Therefore the low Mars orbit and medium Mars orbit designs are by far more superior from a data throughput standpoint. From Danehy's work, the low and medium Mars orbiting satellites have the advantage in data rate throughput. This is because these relay satellites are closer to the Mars user, though the complexity of the communication switching involved in a low and medium orbiting constellation may make them less desirable choices for relay satellites. Further discussion into communication switching complexity is detailed in the reliability section of Chapter 8 towards cost model evaluation.

For the large spacecraft class the data throughput from the most favorable to the least favorable ranking are as follows:

1. The aerosynchronous scenario
2. The halo orbiter scenario
3. The low Mars orbiter scenario
4. The dedicated medium Mars orbiter scenario
5. The Mars lunar lander scenario
6. The common period inclined orbiter scenario.

Thus the common period inclined and the halo scenario are the more favorable network configurations for the large spacecraft class, with both network configurations sufficiently capable of transferring data for the needs of a large science station on the surface of Mars. Between the choice of the common period inclined scenario and the halo scenario, the common period inclined scenario provides the higher data throughput. However the halo orbiters can supply the adequate data throughput demands of a small station despite being transmitted at one fifth the power compared to the common period inclined relay orbiters. The data rate for the common period inclined scenario is consistently on the order 30 to 100 times greater than that of the halo scenario. For a large station on Mars, the halo scenario has an advantage since a parabolic antenna on the common period relay spacecraft does not perform as well at 2 Ghz as it does at 30 Ghz. This limitation could be overcome by selecting a more optimal antenna design for the link. Also further study will need to consider variations in data rates throughout a communications relay mission as well as future technological advances as technological advances become available over time.

Communications Availability Analysis

There is potentially more than 144 million square kilometers of surface coverage on the Red Planet for future exploration, along with the unknown treasures hidden beneath the surface. The communications availability analysis investigates the degree of Mars's surface coverage and visibility available by at least one satellite in the relay network on the surface of Mars in a given period of time. With the Mars relay to Earth link studied in the previous section of this Chapter, the focus in this section will be on the Mars-to-Mars relay network link. Each network configuration and different configurations within the same constellation configuration can provide various levels of service to the user on the Mars surface. Development of the Mars relay coverage and visibility was largely contributed by Mark Danehy in cooperation with the author. This section is divided into the following subfactors: the analysis of the communication link analysis, the network communications visibility and the communications configuration among scenarios.

Network Communications Visibility

There are several factors that affect the data transfer link and hence the network communications visibility in each scenario. Several criteria for Mars surface coverage

capability among scenarios are developed. Table 42 below shows the various visibility criteria applied to the Cost Model evaluation in this study. The criteria under consideration include the ability of a network to provide complete surface coverage at all times, total surface coverage upon revisit time, and the amount of coverage dropout during the observation. Once the various visibility criteria are determined, a baseline coverage design is developed and used to base the scoring among the scenarios. These values are an approximate estimate based on information reviewed in this study. The number in parentheses represent relative coverage rating among scenarios varying between 0.0 and 1.0 used as normalized values towards the Cost Model analysis; a value of 0.0 represents full coverage for consistency in evaluating this scenario comparison in this study.

Table 42. Mars Surface Coverage Capability compared among Scenarios.

	Halo Orbit Orbiters	Aerosynchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Complete Surface Coverage at Any Time?	yes (0.10)	no (0.80)	yes (0.05)	yes (0.05)	no (0.80)	no (0.50)
Total Surface Coverage Upon Revisit Time?	yes (0.01)	no (0.80)	yes (0.01)	yes (0.01)	yes (0.40)	no (0.60)

Definitions used in the communications visibility analysis. The following definitions related to the communications visibility analysis are used in this study: constellation, station-satellite vector, satellite vector, station vector, and communications availability. A constellation is the group of communications relay satellites. The satellite vector is the vector from the center of the planet to the satellite. The station vector is the vector from the center of the planet to the user on Mars. The vector from the Mars user to the relay satellite equal to the vector difference between the satellite vector and the station vector is the station-satellite vector. And finally the communications availability describes the degree of Mars's surface coverage and visibility available by at least one satellite in the relay network to a user on the surface of Mars.

The communications availability is determined by calculating the elevation of each satellite to multiple user locations on the ground, and then tallying the locations that are in view of any one of the satellites in the constellation. Then the satellites are advanced along in their orbit by a small increment defined by a time-step, and the process is repeated. Upon reaching the desired time span, or number of time-steps, the tally of the number of times a relay satellite was visible at each ground point is divided. Each ground point is divided by the number of time steps thereby giving the percentage of time that a user at that location had access to the constellation. Such an algorithm was developed by Danehy in determining communications availability using the POHOP program developed

by the Jet Propulsion Laboratory, Pasadena, California. The program requires as an initial condition a position vector for each satellite in the constellation being simulated. This vector can be specified either in Cartesian coordinates, as in the case of the Mars-orbiting satellite networks, or by coordinates in a rotating reference frame as in the case of the Sun-Mars halo orbiting scenario. For the Mars-orbiting case, the Cartesian State vectors are generated using the POHOP program. Once the initial orbit, the starting position in Keplerian elements, and the time step are specified, the POHOP program propagates the orbit, and an output file containing time, position, and velocity are generated. For the Sun-Mars halo-orbiting case, the rotating reference frame is used to allow for the simplification of the three-body problem and to facilitate obtaining solutions.

Surface coverage criteria. The amount of Mars surface coverage provided by a universal relay network scenario should be made as large as possible to accommodate a broad range of scientific missions. Hence a surface coverage criteria needs to be established in determining the usefulness of a scenario. The criteria established for complete surface coverage at all times ideally is for the scenario to have at least a mean planet-wide coverage of 85 percent at a given time in order to satisfy the surface coverage criteria. In this study coverage is defined as the Mars surface-to-relay line of sight above a 30-degree elevation from a point on the Mars surface; the 30-degree elevation is a

standard selection observed in various NASA papers. The 30-degree minimum elevation is a rather conservative constraint given that most of the Martian terrain is mostly flat with the exceptions of some canyons and mountains; future study should consider a more realistic line-of-sight constraint of above 10 degrees elevation.

In the case of the halo orbit scenario, there is at least 98 percent surface coverage at any given time provided the orbiters are given the relatively small but needed stationkeeping measures. Thus the halo orbiter scenario satisfies the coverage criteria. For the aerosynchronous orbiter scenario, it lacks complete coverage from both northern and southern polar regions, and with a mean planet-wide coverage of 51 percent coverage based on findings from Danehy at any given time it does not satisfy the coverage criteria. The common period orbiter scenario gives adequate coverage at all times provided the orbiters are kept in their respective orbits; hence this scenario satisfies the coverage criteria. The multiple low Mars orbit orbiters can provide complete surface coverage despite the potentially short lifetime of such a network with the potential of orbit decay over time. Neither the dedicated communication orbiter situation nor the Martian Moon landers achieve complete surface coverage with coverage lacking over the poles and a low mean planet-wide coverage similar to the aerosynchronous situation.

Total surface coverage upon revisit time criteria. Along with the general surface coverage criteria, the criteria of total surface coverage upon revisit time is satisfied once complete surface coverage is achieved upon multiple revisit times totaling one sol in a given scenario with the definition of coverage described in the surface criteria section. In the case of the halo orbit scenario, there is at least 98 percent surface coverage at any given time provided the orbiters are given the relatively small stationkeeping measures necessary to maintain the maximum coverage configuration. Thus this scenario again satisfies the coverage criteria. For the aerosynchronous orbiter scenario, the lack of coverage from both northern and southern polar regions along with the mean planet-wide coverage of about 50 percent in one sol period is still less than the coverage criteria with the time condition. The common period orbiter scenario gives complete coverage at all times and thereby satisfies the coverage and time criteria. The multiple low Mars orbit orbiters can provide complete surface coverage over a sol as well. For the dedicated communication orbiter situation it may not be possible to achieve total surface coverage over one sol, and again for the Martian Moon landers surface coverage over one sol is not possible with the lack of coverage over the poles.

Communications visibility analysis. In this study the communications availability is defined as the percentage of time a point on the Mars surface can achieve line-of-sight to a relay satellite in the relay network with a 30-degree minimum elevation. The Mars surface is partitioned in three general regions. One region is the polar region which includes the North Pole to 67 degrees North latitude and South Pole to 67 degrees South latitude. The second region is the temperate region that includes both 23 degrees to 67 degrees of both North and South latitudes. The equatorial region is the third region which spans from 23 degrees North latitude to 23 degrees South latitude. Table 43 shows the communications availability for Mars's polar, temperate, and equatorial regions among each scenario constellation. The analysis used for the multiple low Mars orbit case is the "Low 6" constellation case from Danehy's work. The visibility values for the dedicated communication relay orbiter and the Mars lunar lander scenarios were not calculated by Danehy are of rough order approximations derived from the multiple medium orbiter from Danehy and the aerosynchronous scenarios respectively.

Table 43. The Percentage of Mars Relay Communication Visibility Greater than 30 Degree Elevation are compared among Scenarios.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Polar	18.4	0	29.1	11.7	10	0
Temperate	42.3	39.8	65.3	2.5	2.5	30
Equatorial	59.5	84.3	77.8	1.7	2	60

Preliminary findings of communications visibility analysis among scenarios. The amount of Mars surface visibility by a relay spacecraft is found to be spacecraft class independent and spacecraft class calculations are not applicable to this subfactor. Based on the communications visibility analysis shown in Table 43 the common-period orbiters have the highest percentage and hence the most favorable surface-to-satellite visibility at the polar, temperate, and the equatorial regions. Following the common-period orbiters in visibility rank respectively are the halo scenario, the aerosynchronous Martian moon lander scenario, the multiple low Mars orbit scenario, and the dedicated communication orbiter case with the least global visibility to a relay satellite scenarios.

Though the orbits in the Mars moons lander case have orbital altitudes similar to that of the aerosynchronous scenario, the orbits for the Mars moons relay landers are dependent on the orbits of the Mars moons and are not equidistant between each relay in the Mars lunar relay network. This dependence on the lunar orbit in addition to the fewer

number of relay satellites for the Mars Moon lander case compared to the aerosynchronous case leads to a lower estimated network visibility. For the dedicated communication orbiter case with only one satellite, the visibility with its only relay satellite at 1000 kilometers circular altitude is estimated at having the lowest visibility among the scenarios compared.

This communication visibility information is to be used towards data throughput calculation for over a period of time consistent with the scenario of longest duration, a 74-hour period from the common-period scenario. From this calculation will be performed among scenarios in determining favorable scenarios from a communication performance standpoint. Refer to Chapter 9 for further details on the data throughput calculation.

Communications System Configuration

This section explores the possible communications system configuration among scenarios and among the relay spacecraft classes. The focus of this section is to examine scenarios with the most favorable communications performance from its communications system that includes the option of a relay satellite crosslink. A network crosslink has the benefit of enhanced data relay to Earth, with the transferring of continuous data to Earth and with less of a need for large on-board memory storage for each relay spacecraft. The advantages and disadvantages are then outlined among scenarios and with relay satellite classes.

Table 44 shows the communications system configuration and normalized rating compared among the scenarios in both the large and the small relay spacecraft classes. The rating value with each scenario and relay class evaluated in Table 44 is based on the advantages and disadvantages outlined in the table and each normalized value is of a rough order of magnitude. The respective small spacecraft and large spacecraft class rating values are indicated in the first column in parenthesis and favorable normalized ratings indicated by low values.

Table 44. Communications System Configuration and Associated Normalized Rating are evaluated among the Scenarios for Both Relay Spacecraft Classes.

Configuration - Ratings (Small / Large Spacecraft)	Advantages	Disadvantages
	Small Spacecraft Class - Spin-Stabilized Without Crosslink Large Spacecraft Class - Three-Axis Stabilized With Crosslink(s)	
Halo Orbit Orbiters (0.35 / 0.30)	<ul style="list-style-type: none"> Both relay spacecraft capable of relaying data independently since both spacecraft are usually visible to Earth. Provides virtually complete Mars surface coverage at any given time. 	<ul style="list-style-type: none"> Relatively high propagation delay from the Mars surface. Satellite handover required. Low data transfer rate warranting higher transmitter power to increase data transfer rate.
Aerosynchronous Orbiters (0.45 / 0.40)	<ul style="list-style-type: none"> No switching required between satellites relative to Mars surface. Little or no relay tracking from the Mars surface required. 	<ul style="list-style-type: none"> Need for stationkeeping. No polar coverage available.
Common Period Orbiters (0.40 / 0.35)	<ul style="list-style-type: none"> Provides virtually complete Mars surface coverage at any given time. 	<ul style="list-style-type: none"> Satellite handover required. Requires continuous changing of antenna pointing angles from the Mars surface. Network control complex. Need for stationkeeping.
Multiple Low Mars Orbit Orbiters (0.55 / 0.50)	<ul style="list-style-type: none"> Polar coverage with inclined orbit. Lower transmitter power to Mars due to lower altitude. 	<ul style="list-style-type: none"> Satellite handover required. Complex dynamic network control with crosslinks. Large number of satellites needed for modest link availability.
Dedicated Communication Orbiter – Lander (0.70 / 0.70)	<ul style="list-style-type: none"> No sharing of network resources. 	<ul style="list-style-type: none"> No sharing of network resources. No added value of crosslink for the large spacecraft class. Single point of failure.
Martian Moon Landers (0.60 / 0.55)	<ul style="list-style-type: none"> No stationkeeping needed. Upon proper Mars lunar landing operation, constant antenna pointing at Mars surface possible. Upon proper Mars lunar landing operation, eliminates need for attitude control. 	<ul style="list-style-type: none"> Satellite handover required. Communications performance dependent on resulting Mars lunar landing operation.

Communication configuration - small relay spacecraft class. Due to purposes of simplicity and of a limited power budget design, scenarios for the small relay spacecraft class in this study will not have the crosslink capability for its relay spacecraft. Though the satellites in this class will need a greater on-board memory storage requirement compared to the scenarios in the large relay spacecraft class, the scenarios in the small relay spacecraft class will not have the complex network control that a network with crosslink capability will have.

The communication relay performance of halo orbit scenario would not degrade from the lack of a crosslink capability, though satellite handover is required with the switching of links between satellites are needed as a satellite moves in and out of surface visibility. Also the virtual visibility of both halo spacecraft to Earth permits both satellites in this network to relay data to the Earth simultaneously thereby maintaining relatively high data relay performance. For the aerosynchronous case, the principal disadvantage is a lack of surface coverage outside the 70-degree latitudes. The degree of the lack of surface coverage can vary with the instance of the "Figure Eight" motion from an aerosynchronous orbit with some degree of inclination. Also for the aerosynchronous relay spacecraft some data memory storage is needed during periods when the relay satellite is out of Earth's visibility due to Mars.

The common period inclined scenario has the most favorable surface coverage of the scenarios in this study. However, continuous changing of the antenna pointing angles from the Mars surface is required with its highly elliptical orbit and the rotation of the Red Planet, particularly at around the periapse portion of the relay spacecraft's 74 hour orbit. Conversely, the dedicated communication orbiter scenario has the least favorable communications configuration among the scenarios in this study, with the multiple low Mars relay spacecraft being the next least favorable among scenarios in this study. The dedicated communications orbiter with its 1000 kilometer medium orbit will likely adopt a modified form of the store-and-forward system similar to that used in the Courier satellite which appeared in 1960.⁹ The dedicated relay satellite in this configuration orbits the Red Planet, receives data, and stores the data in on-board memory. Then the dedicated relay satellite transmits the data to the Earth as the satellite moves in view of the Earth. The inherent disadvantage of this scenario is that long message access times and transmission delays of up to hours are to be expected where a Mars surface location of interest waits for the relay satellite to pass into view of the surface location. The multiple low Mars relay network without crosslinks suffers from complex control of a dynamic network. The Martian Moon Landers scenario would not fare much better than the multiple low Mars relay network in that it also has a complex dynamic network of its own. And like the

⁹ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 2nd ed. (Torrance, CA: Microcosm Press, and The Netherlands: Kluwer Academic Publishers, 1992), 505.

aerosynchronous orbiter, the Martian Moon Landers would lack Mars surface coverage outside the 70-degree latitudes with its equatorial orbits.

Communication configuration - large relay spacecraft class. Based on the greater dimensional and mass margin provided by the large relay spacecraft class, the large relay spacecraft class in this study is determined to include the crosslink capability. This allows varying degrees of robustness to the communications relay capability depending on the scenario of interest. The added crosslink would add complexity to the communication design but largely offset by the benefit of a greater overall data transfer.

The communication relay performance of the halo orbit scenario, large spacecraft class would only be slightly enhanced with the added crosslink capability since both halo spacecraft are almost constantly visible to Earth. The only minor advantage of the added crosslink would be to overcome the rare solar conjunction events occurring once every two years. For the aerosynchronous case, the added crosslink feature does not alleviate the problem of a lack of surface coverage outside the 70-degree latitudes. However, the added spacecraft crosslink does provide a lower memory requirement for data storage as relay satellites outside Earth's visibility due to Mars is relayed to an adjacent relay satellite that is in Earth view and the data is relayed to Earth.

The common period inclined scenario has the most favorable surface coverage of the scenarios in this study despite the continuous changing of the antenna pointing angles from the Mars surface required with its highly elliptical orbit. And with the relay crosslink available, a higher data transfer rate to Earth is achieved. Conversely, the dedicated communication orbiter scenario has the least favorable communications system configuration among the scenarios in this study with the crosslink adding no benefit. The multiple low Mars relay spacecraft would not be favorable among scenarios in this study despite the added crosslink. The multiple low Mars relay network with crosslinks would still suffer from complex control of a dynamic network. The Martian Moon Landers scenario communication configuration suffers from a complex dynamic network. And like the aerosynchronous orbiter, the Martian Moon Landers would lack Mars surface coverage outside the 70-degree latitudes with its equatorial orbits.

Initial findings - communications system configuration. The communication system configuration subfactor is difficult to evaluate among subfactors in this design driver. This subfactor weighs the following six items: the applicable antenna pointing to Mars, satellite handover, stationkeeping, data rate performance, transmitter power performance, and the amount of network crosslinks needed among scenarios. The ranking of most favorable to

least favorable scenario based on findings from Table 44 and the communications system configuration design providing the best communication performance are as follows:

1. The halo orbit scenario
2. The common period inclined scenario
3. The aerosynchronous scenario
4. The Martian Moon lander scenario
5. The multiple low Mars orbit scenario
6. The dedicated communication orbiter scenario.

The halo orbiter scenario has been determined as having the most favorable communications configuration rating for both spacecraft classes, over the common period inclined scenario. The drawbacks for a halo orbiter are comparably minor with a higher transmitter power requirement to compensate for a lower data transfer rate performance and a longer data propagation delay between the relay and the surface Mars. Conversely the dedicated communication orbiter case is determined to have the least favorable rating with the least global visibility to a relay satellite, lack of crosslink capability for added navigation options, and a single relay spacecraft being a single point of failure.

Evaluation and Summary of Communications Performance Analysis

The communications performance analysis investigates the following four subfactors: the link analysis, the percentage of relay visibility to the Mars surface, surface coverage of a relay network over time, and the communications system configuration among the scenarios shown in Appendix K-3. These four subfactor values are normalized and weighted equally towards a single communications performance parametric calculation. The link analysis subfactor uses the evaluation of the uplink and downlink data rates for both day and night data rates. The values among scenarios were normalized based on the small spacecraft class of halo orbiter with the lowest data rates. The visibility subfactor uses the inverse of the polar, temperate, and the equatorial visibility percentage values and then takes the average value. The surface coverage subfactor uses the average values of the likelihood of complete surface coverage at any given time and complete coverage upon relay spacecraft revisit among each scenario. Finally with the values of the communications system configuration subfactor the average of the four subfactors are performed in the evaluation of the communications performance among scenarios. Row 40 in Appendix K-3 shows a sample calculation used to evaluate the value for the small spacecraft class, halo orbiter scenario.

Based on findings from Table 37 through Table 44 and evaluated in Appendix K-3 in the communications performance analysis, both spacecraft classes of the common period inclined scenario show the most favorable communications performance with the lowest normalized values followed by the large spacecraft class, halo orbit scenario. A substantial data rate penalty is observed for the small spacecraft class, halo orbit scenario that accounts for the relatively large normalized value. Varying degrees of visibility and surface coverage penalty are observed for both spacecraft classes of the aerosynchronous, low Mars orbit, dedicated communication, and Mars lunar lander scenarios accounting for their larger normalized values.

CHAPTER 7

The Spacecraft Lifetime Parameter

$$\mathbf{J} = w_{\$}(\$) + w_{\text{Size}}(\text{Size}) + w_{\text{Comm}}(\text{Comm}) + w_{\text{Life}}(\text{Life}) + w_{\text{Plan}}(\text{Plan})$$

This chapter discusses the fourth parameter in the Cost Model, the network life cycle parameter. Various considerations favoring extended lifetime network designs include reducing the possible risks involved with relay satellite operations. These considerations are used as subfactors in the spacecraft lifetime driver. The subfactors include the following: spacecraft fuel requirement, total power requirement, battery recharge margin, battery power loss risk factors, orbit altitude, thermal power requirement, network fault tolerance, spacecraft reliability, and graceful degradation. These subfactors were weighed equally in this design parameter. Table 45 below shows the findings of the network life cycle parameters among the scenarios, with the spacecraft lifetime parameter representing the collective of the nine lifetime risk subfactors described previously. These findings shown in tabular form apply to both spacecraft classes unless otherwise indicated with dual values where the first value applies for the small spacecraft class and the second value applies for the large spacecraft class. Once the various spacecraft lifetime factors are determined, these values are normalized and used for the overall cost model.

Table 45. Mission Life Cycle Risk Subfactors are compared among the Scenarios.

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Comm. Orbiter - Lander	Martian Moon Landers
Spacecraft Fuel Requirement	low	med-high	low	med	med	low
Total Power Requirement	low / med	low / med	low / med	low / high	low / high	low / med
Battery Recharge Margin	low / med	low / med	low / med	low / high	low / high	low / med
Battery Power Loss Influences	low	low	med	high	med-high	med
Spacecraft Orbit Altitude	low	low	low	high	med	low
Thermal Cycling	low	low	low	med-low	med	med
Fault Tolerance	med	med-high	med-high	low	high	med
Reliability	med / med	med / med	med / med	med / med	med / med	med / med
Graceful Degradation	med-high / med-high	med / med-high	med / med	med / med	med-low / med	med-high / med-high
Spacecraft Lifetime Risk	low / low	med / med	low / low	med / med-high	med / med-high	med / med

As indicated in Table 45, a number of subfactors influence spacecraft lifetime that also exemplifies the difficulty of establishing requirements. The actual lifetime of a relay mission is often not definite and not predictable. Also the amount of total data collection over a regular time interval depends largely to the performance of the relay network lifetime through the end of the relay mission. In practice, a relay network needs to be developed that satisfies the need for continuous data collection to be applied to an established design life and possible replenishment philosophy. The design lives of the individual spacecraft will strongly affect cost and will be determined by the level of redundancy, propellant budgets, and other key system parameters. Ideally, a graph of spacecraft cost versus design life needs to be drafted early in the design phase in setting

the spacecraft design life requirement.¹ The nine factors related to spacecraft lifetime are individually investigated and the relevance of each factor is determined towards the Cost Model.

Spacecraft Stationkeeping Fuel Requirement Subfactor

Once at Mars most mission scenarios have their useful mission lifetime limited to the amount of propulsion on board the spacecraft for stationkeeping and orbit adjustment needs. The amount of propulsion determined for a spacecraft may include an added fuel amount for safety margin. The propellant on board the relay spacecraft would be used for either or all of the following reasons: proper phasing in maintaining total or proper Mars coverage; maintaining proper stationkeeping; and maintaining proper orbit. The more propellant on board a spacecraft, the greater the potential for a longer mission life at the risk of a mass and cost penalty at launch.

For the halo scenario, a relatively small amount of stationkeeping, on the order of about 8 meters per second per year, is required as similarly demonstrated in the ISEE-3 spacecraft program, where stationkeeping on the Sun-Earth L_1 libration point is on the order of several meters/second per year. Due to its large distance from Mars, the amount

¹ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 2nd ed. (Torrance CA: Microcosm Press, and The Netherlands: Kluwer Academic Publishing, 1992), 18.

of stationkeeping for the halo scenario is least influenced by perturbative and triaxiality effects of the Red Planet.

Similarly, for the Common Period Inclined, a relatively low amount of stationkeeping is needed for the life of the mission. The estimated stationkeeping about Mars adapted from Drain's A Common-Period Four-Satellite Continuous Global Coverage Constellation is estimated to be on the order of approximately 30 meters per second per year for each satellite. This stationkeeping is to account for the higher perturbative and triaxiality effects of Mars compared to Earth.

The mission life of the Phobos and Deimos lander scenario would not be affected by stationkeeping limitations since its relay orbit is dependent on the orbit of the Martian moons the relay satellites land on. However, the propulsion would be mainly used for landing the spacecraft on a Martian moon, and the remaining propellant on board the spacecraft would be wasted once the spacecraft successfully lands on their respective Martian moons. However, the apparently favorable total fuel requirement would be more than offset by its unfavorable development cost of the Mars moon lander technology that only this scenario requires in this study.

The remaining Multiple Low Multiple Mars orbit scenario would likely require relatively more propellant for phasing and stationkeeping maneuvers during the lifetime of their respective missions requiring a greater amount of propellant. Hence the least desirable scenario in terms of total fuel requirement is the multiple low Mars orbit scenario.

Table 46. Spacecraft Fuel Requirement over a five-year Mission among the Scenarios Applicable to Both Spacecraft Classes.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Total Fuel Requirement	0.04 km/s ⁽²⁾	2.9 km/s ⁽³⁾	0.03 - 0.045 km/s ⁽⁴⁾	1.5 km/s	~1.5 km/s	0 km/s

Preliminary findings for the spacecraft fuel requirement. Table 46 shows the preliminary findings among the scenarios that apply to both spacecraft classes. From Table 46, a lifetime fuel requirement ranging from 0.0 km/s ΔV for the Mars Moons scenario to about 2.9 km/s ΔV for the aerosynchronous case are observed to account for applicable maneuvers, orbit adjustments, and proper network phasing over a five year

² Pernicka, H. et al, "Use of Halo Orbits to Provide a Communication Link between Earth and Mars," AIAA Paper no.92-4585-CP, AIAA/AAS Astrodynamics Conference (1992).

³ Palamarik T. et al, "Mission Design for an Aerosynchronous Orbiter," AAS Paper no. 89-197, AAS/AIAA Conference (1989).

⁴ Chao C., "Long-Term Orbit Perturbations of the Draim four-Satellite Constellations," Journal of Guidance, Control, and Dynamics 15, no. 6, (Nov. - Dec. 1992).

mission. Thus the spacecraft stationkeeping subfactor largely influences the aerosynchronous orbiter scenario. However in this study, the spacecraft stationkeeping parameter is considered a relatively low mission driver and is reflected in the cost model of this study.

Total Power Requirement Subfactor

For both spacecraft classes the spacecraft power requirements at the beginning of mission life play a significant role in the network lifetime mission driver. The total power requirement for the small spacecraft class is intentionally restricted to the specifications of the MRS design. This limited variation in total power among scenarios in the small spacecraft class is imposed due to physical limitations. For the general three-axis-stabilized large spacecraft class there is greater variation in power requirements with the low Mars orbit orbiter scenario having the greatest total power requirement as shown in Appendix K-5. The total power requirement influences the common period inclined scenario the least. Further details in the total power requirement are discussed in the battery recharge margin subfactor.

Battery Recharge Margin Subfactor

The spacecraft power requirement and the eclipse-sunlight ratio in the orbit of a relay communications spacecraft largely influence the battery recharge margin. A relay spacecraft with low sunlight duration available with correspondingly long eclipse duration in the relay satellite orbit would adversely affect the battery recharge margin. The battery recharge margin among spacecraft classes is discussed in detail below.

Battery Recharge Margin - Small Spacecraft Class

Data obtained from Chapter 5, Tables 23 to 25, are used in comparing the battery recharge margin for the small spacecraft class with the general spin-stabilized design with the exception of the Mars moon lander scenario that the spin-stabilized design does not apply. The physical size limitations for the small spacecraft class results in the similarity of relay design among scenarios. Thus the power requirements are similar between scenarios. The remaining factors to consider are the relay satellite orbit and the eclipse period influencing the battery recharge margin.

Table 47 shows the battery recharge margin, defined as the amount of reserve power required for contingency events during spacecraft operation, along with the total power requirement used as reference for the small relay satellite compared among the scenarios. Similar battery recharge margin values are observed for the halo orbit, aerosynchronous, common period inclined, and Martian moon lander scenarios. The multiple low Mars and the dedicated communications scenarios show a slightly greater battery recharge power due to a greater margin in the spacecraft power requirement. This greater power requirement is due to the relatively lower sunlight duration available in the relay satellite orbit unfavorably affecting the battery recharge duration. A spreadsheet displaying a comprehensive calculation of the total power requirements for the small relay satellite class is shown in Appendix I. Note that for this study the small relay satellite class portion the amount of power calculated is intentionally limited to only spacecraft housekeeping during eclipse periods and would not include communication operations at instances of penumbra in the relay spacecraft's orbit.

Table 47. Battery Recharge Margin for the Small Relay Satellite Compared among the Scenarios.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter – Lander	Martian Moon Landers
Total Power Requirement	71.3 W	71.3 W	71.3 W	100.05 W	100.05 W	~75 W (Phobos)/ ~71.0 W (Deimos)
Battery Recharge Margin	7.5 W	7.5 W	7.5 W	7.5 W	7.5 W	7.5 W (Phobos)/ 7.5 W (Deimos)

Battery Recharge Margin - Large Spacecraft Class

The battery recharge margin values are obtained from Chapter 5, Tables 26 to 28, for the large spacecraft class with three-axis stabilized design with the exception of the Mars moon lander scenario. Table 48 shows the battery recharge margin for the large relay satellite compared among the scenarios with the total power requirement values used as reference. An observation from these tables shows that both the battery recharge margin and the power requirements for this class are largely influenced by two items. These items include the amount of battery recharging needed for the spacecraft batteries to supply vehicle power during the eclipse portion of the orbit or upon a tumble recovery of variable duration in the design criteria depending on the scenario. The design of the large spacecraft class can readily accommodate larger transmitter power, which can translate to

not only communications operations during the eclipse portion of the relay orbit, but also supply larger data rates or bandwidths compared to the small spacecraft class.

Table 48. Battery Recharge Margin for Large Relay Satellite Compared among the Scenarios.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter – Lander	Martian Moon Landers
Total Power Requirement	1395.9 W	1373.8 W	1330.7 W	2976.6 W	2811.3 W	1534.8 W (Phobos) / 1362.3 W (Deimos)
Battery Recharge Margin	50.0 W	60.0 W	30.0 W	1050 W	675.0 W	180.0 W (Phobos) / 56.0 W (Deimos)

Preliminary findings - battery recharge margin. From this study, the small relay spacecraft class shows similar power design due to physical limitations, while the three-axis-stabilized large relay spacecraft class, except for the Mars moon lander scenario, shows greater variation in battery recharge margin among scenarios with less of a spacecraft size constraint. With both relay spacecraft classes considered, the scenario ranking starting with the lowest power requirement and hence most favorable scenario observed is as follows:

1. The common period inclined scenario
2. The aerosynchronous and the Martian moon lander scenarios

4. The halo orbiter scenario
5. The dedicated communication scenario (medium Mars orbit)
6. The multiple low Mars orbiter scenario.

The dedicated communication and the multiple low Mars orbit scenarios show a greater spacecraft power requirement due to the need for a greater battery recharge power margin as a result of a lower battery recharge duration available in the relay satellite orbit.

Variations in the orbit characteristic account for the differences in the amount of battery recharge margins and the total power requirement among scenarios, particularly with the large battery recharge margins of the low Mars orbiter scenario and the medium orbit dedicated communications orbiter scenario relative to the other scenarios.

Comprehensive calculations of the total power requirements and associated battery recharge margin for the large relay satellite class among scenarios is shown in Appendix J in spreadsheet form. Note that for the large relay satellite class portion of this study the amount of power calculated is not limited only to spacecraft housekeeping during eclipse periods but also includes communication or payload operation for data relay to Earth for as often as possible.

The Battery Power Loss Risk Subfactor

This investigates the ability of a scenario to sustain spacecraft power in the event of a partial or complete failure in the secondary or battery power system. The battery power loss risk subfactors include a combination of the estimated maximum eclipse duration, the estimated lifetime number of power-eclipse cycles, and the battery power loss impact. These conditions were weighted equally in this study. A partial loss in battery power would curtail communication relay operations for missions with orbit eclipses, while a complete battery power loss would mean the end of mission life for such missions. The halo orbit scenario would be the only scenario to be able to sustain spacecraft power provided that attitude control maintains solar panel pointing towards the Sun since the halo orbit receives constant sunlight. Since all the other scenarios encounter an eclipse period in their orbits, battery power loss would likely mean a partial to a complete loss of useful mission life. Hence the halo scenario would be the favored scenario in the event of spacecraft battery power loss. The findings for battery power loss apply to both the large class and the small class spacecraft.

The battery power loss impact is related to a combination of the maximum eclipse duration encountered in orbit and the number of cycles and is reflected in Table 49. Table 49 shows the battery power loss impact derived from the estimated maximum eclipse

duration in a typical orbit and the estimated five-year lifetime number of power cycles. The halo orbit with its constant visibility with the sun is most favorable among the scenarios in terms of the impact of battery power loss. The next favorable scenario is the common period inclined scenario with its short maximum eclipse duration and power cycles followed by the aerosynchronous scenario. As the Mars moon landers share similar orbital characteristics with aerosynchronous scenario, so are the values for battery power loss impact similar between the two scenarios. The multiple low Mars orbit orbiters and the medium orbit of the dedicated communication orbiter gives these scenarios the greatest battery power loss impact among the scenarios.

Table 49. Quantitative Battery Power Loss Risk Conditions for Small and Large Relay Satellite Class Compared among the Scenarios.

	Halo Orbit Orbiters	Aerosynchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter – Lander	Martian Moon Landers
Estimated Maximum Eclipse Duration (secs)	0	4160	1200	1800	1800	2500 (Phobos) / 3500 (Deimos)
Estimated Lifetime Number of Power Cycles	0	1800	600	27500	24000	2200 (Phobos) / 1500 (Deimos)
Battery Power Loss Impact	0.0	0.25	0.125	0.75	0.75	0.25 (Phobos) / 0.25 (Deimos)

Preliminary findings - battery power loss risk conditions. In this study the battery recharge margin is a significant factor towards the cost model. The battery recharge margin is dependent on the orbit characteristic of the relay spacecraft and the amount of eclipse in the relay spacecraft orbit. The battery power loss risk subfactors include a combination of the estimated maximum eclipse duration, the estimated lifetime number of power-eclipse cycles, and the battery power loss impact and the subfactors were weighted equally. The aerosynchronous scenario was most influenced by the eclipse duration, while the low Mars orbiter and the medium-orbit dedicated orbiter scenarios were most influenced by the lifetime power-eclipse cycles and the battery power loss impact subfactors. The preliminary scenario ranking starting with the lowest risk from battery power loss observed is the halo orbit scenario followed by the common period inclined scenario, the Mars lunar lander scenario, the aerosynchronous scenario, the dedicated communications scenario, and finally the low Mars orbiter scenario.

Spacecraft Orbit Altitude Subfactor

The next subfactor of interest affecting spacecraft lifetime is the spacecraft orbit altitude investigated among the scenarios. The relay orbit altitude would be an important factor particularly if the relay spacecraft potentially decays in the Martian orbit over time

and impacts the surface of the Red Planet. A scenario with a low orbit altitude also would be a limiting factor in the lifetime of a relay mission. Table 50 shows the orbit altitudes and the relative rating for both spacecraft classes among scenarios. Note that for the purposes of Cost Model evaluation, for the common period orbiter scenario an average orbit altitude value of 45000 kilometers was used.

Table 50. Risk from Orbit Altitude for Both Spacecraft Classes Compared among the Scenarios.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter	Martian Moon Landers
Relative Orbit Altitude	high	low	medium	low	Low	low / med-low
Orbit Altitude (km)	1×10^6	17040	27975.9- 50362.5 (45000)	300	1000	5989 (Phobos), 20058 (Deimos)

Preliminary findings - spacecraft orbit altitude. The halo orbiter scenario is most favorable with its virtually non-existent risk of having either relay spacecraft deorbit and impact the surface of the Red Planet. The halo orbit relay satellites orbit about the Sun-Mars L_1 and L_2 libration points respectively, and this is the only scenario in this study which the relay satellites do not orbit around the Red Planet. The aerosynchronous orbiter scenario with the highest Mars orbit altitude among the scenarios that orbit Mars also

would have no satellite impact hazard, and neither would the common period inclined orbiters pose a spacecraft impact hazard on the Red Planet. For the Martian lunar landers relay spacecraft no decay orbit exists unless the moons themselves impact the Red Planet. A remote possibility exists that Phobos will impact Mars in several million years, while a similar impact would be unlikely for the Deimos case.⁵

Conversely, the 300 kilometer altitude of the multiple low Mars orbit scenario as well as the 1000 kilometer altitude dedicated communications orbiter would likely be a disadvantage where the relay spacecraft would experience orbital decay and surface impact within five Earth years. A relay spacecraft with lower orbits will have less regular orbits as the orbit of the relay satellite decays over time, thus requiring more monitoring from Earth. There would also be a legitimate concern of having a man-made object potentially becoming undesirable debris and hence the least desirable scenario.

Thermal Power Requirement Subfactor

Despite a number of past Mars missions used to explore the various environmental aspects of the Red Planet, our knowledge of the atmospheric and environmental effects of

⁵ Veverka, J., "The Moons of Mars," AAS Paper no. 86-157, NASA Mars Conference, National Academy of Sciences, Washington, D.C., (July 21-23, 1986).

Mars on spacecraft is incomplete. But with present missions such as the Mars Global Surveyor and Mars Pathfinder missions exploring the surroundings of the Red Planet, this knowledge will be used towards spacecraft designs in future missions to Mars. This portion of this study focuses on the thermal cycling aspect in terms of thermal power of the thermal environment on the relay spacecraft.

Based on the present knowledge of the harsh Martian environment, a typical Martian mission would need some level of thermal protection. In this study the large relay spacecraft class could afford some degree of active thermal protection from better power provisions when applicable, while the small relay spacecraft class could only afford passive thermal protection. Table 51 shows the thermal power needed for active thermal equipment for the small and the large spacecraft classes. Little active thermal protection is available for the small spacecraft class for the hydrazine thrusters with the exception of the halo orbit, the common period inclined, and the Martian moon lander scenarios. The estimated lifetime number of thermal cycles for each scenario is based on data from Table 49.

Table 51. Thermal Power Requirement compared among the Scenarios for both Spacecraft Classes. The thermal power requirement is based on the amount of thermal cycling in their respective orbits.

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Comm. Orbiter - Lander	Martian Moon Landers
Estimated Amount of Thermal Cycling in Lifetime	0	1800	600	27500	24000	2200/1500
Small Spacecraft Class Thermal Power (Watts)	0.0	0.1	0.0	0.1	0.1	0.0
Large Spacecraft Class Thermal Power (Watts)	38.2	37.1	37.1	37.1	37.1	37.1

The halo orbit orbiters and the common period orbiters are the only scenarios in this study where each relay orbiter is virtually in constant view of the Sun. For the common period orbiter situation there is virtually no thermal cycling occurring in its highly elliptical orbit, as its orbit occurs mostly in the sun and the maximum eclipse experienced by a spacecraft in this orbit is on the order of less than five minutes. For both scenarios passive thermal control would be sufficient, though hydrazine heater for thrusters are used sparingly for minor orbit adjustments.

The Martian moon landers would experience some eclipse in its orbit somewhat similar to the aerosynchronous scenario. With the Mars moons in gravity-gradient orbits and with the same face constantly the Red Planet, these relay spacecraft would experience a relatively moderate amount of thermal cycling. Passive thermal control design would

only be needed, and hydrazine heaters would be used for thruster operation towards the Mars lunar landing.

For the remaining two scenarios the relay spacecraft encounters various degrees of thermal cycling as the relay spacecraft enters and exits eclipse in their respective orbits. A typical relay spacecraft for the multiple low Mars orbit scenario would experience relatively significant thermal cycling as some eclipse would occur in its orbit. The dedicated communication orbiter at 1,000-kilometer altitude would experience some eclipse in its orbit and hence would experience thermal cycling. A passive thermal control design may be sufficient for the small relay spacecraft with spin-axis stabilized. However, active thermal control for hydrazine heater function is needed for thruster operation.

Preliminary findings - thermal power requirement. A favorable scenario requires the least amount of active thermal cooling. With this criteria considered the halo orbit orbiters has the most favorable conditions among scenarios considered. Some thermal cycling occurs in the orbit of an aerosynchronous orbiter as the relay spacecraft enters and exits eclipse; passive thermal control is possible with the exception of the hydrazine heaters used for maximum thruster efficiency. The thermal power requirement affects the multiple low Mars orbiter scenario most with a large number of eclipse-power cycles as well as the thermal power most influencing the large spacecraft class. This subfactor is the least

sensitive of the subfactors in the lifetime design driver. The dedicated communication orbiter at a 1,000-kilometer altitude would experience some eclipse in its orbit and hence would experience significant thermal cycling throughout its mission.

For the common period orbiters passive thermal control would be sufficient; hydrazine heater for thrusters are not needed for any of the relay spacecraft since each spacecraft is virtually in constant view of the Sun. The Sun could be used advantageously in incorporating heating to maximize thruster efficiency into the spacecraft design. The Martian moon landers would experience some eclipse in its orbit that is somewhat similar to the aerosynchronous scenario. With the Mars moons in gravity-gradient orbits with the same face constantly the Red Planet, some spacecraft thermal cycling would occur; passive or active control; no hydrazine heaters are needed prior to thruster operation once each lander successfully lands at a pole of the Mars moon.

Network Fault Tolerance Subfactor

This section discusses a method of assessing the fault tolerance of each mission scenario. In this study fault tolerance is defined as the ability for the relay network to

continue operating after the failure of one spacecraft in the network.⁶ This study investigates fault tolerance at the constellation design level among the six scenarios, compared to reliability which is determined at the individual spacecraft level and discussed in the next section. This network fault tolerance subfactor is comprised of the following three items: coverage reduction of one spacecraft loss in the network, the approximate ΔV effort from network rephasing, and coverage reduction from network rephasing. Although fault tolerance is determined at the relay network level, the amount of fault tolerance among scenarios depends to a degree on the amount of redundancy at the spacecraft design level, as well as the amount of budgeting allotted for the redundancy in spacecraft design. Scenarios with favorable fault tolerance generally require one or more redundant satellites to function with successive satellite failures, though at reduced service. Thus spacecraft autonomy implies less need for commanding from Earth to correct possible spacecraft anomalies during its early mission. This capability of commanding from Earth has often greatly reduced the severity of many satellite failures in the past but at the expense of high operational cost. Future trends for interplanetary missions call for autonomous missions that require greater fault tolerance at the network level.

⁶ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design.

Coverage Reduction from One Spacecraft Loss in Network

At the spacecraft design level the large spacecraft class is far more fault tolerant compared to the small spacecraft class due to the luxury of redundant components in the large spacecraft design. With redundant components, the large spacecraft can tolerate several failure modes, while the small spacecraft design does not accommodate a similar situation. The lack of rigorous redundancies in the small spacecraft design makes it not fault tolerant friendly.

At the constellation design level the amount of fault tolerance also depends on the constellation configuration affecting the amount of constellation rephasing available. Thus the most fault tolerant network scenario is the low orbiting relay satellite configurations. The six satellite configurations would provide the most redundancy and fault tolerance. The aerodynamically constellation four-satellite configuration with the network orbiting on the same orbital plane would also provide a means for graceful degradation. In the event of a failed satellite the remaining three satellites could be rephased to assume a three satellite configuration. The least fault tolerant of the network configurations would be the halo configuration followed by the common period inclined scenario. Both these constellations are critically configured resulting in a loss of coverage of about 50 percent for the halo network if one satellite fails, and varies over time between 25 to 33 percent

for the common period configuration. For the single halo orbiter loss instance, an outage period of about twelve hours would occur on either the day or night side of the planet depending on the satellite outage. If feasible and applicable to the science mission, some form of data storage would be needed on the lander during the blackout period to be uplinked later at a communication visibility period in order to compensate for this loss.

Approximate ΔV Effort from Network Rephasing

For instances of a failed satellite in a scenario, network reconfigurations via network rephasing and its applicability to each scenario is considered. In the event of a failed satellite in the relay constellation, the use of rephasing the remaining satellites in the constellation may be an option. This rephasing technique would be used to maintain the best possible coverage and visibility given a failed satellite in the constellation. The effects from constellation rephasing would be similar for the large and small spacecraft classes. Once the determination of a network reconfiguration is to be accomplished, a one-time orbit adjust for each satellite in the network is performed in optimizing communication relay performance with reduced relay resources. Prior to completion of the network rephasing some network downtime would occur varying from maximum downtime from initial single spacecraft failure to minimal or absence of downtime upon completion of

network rephasing. The amount of ΔV per spacecraft needed to position each spacecraft in the network is estimated and are conservatively high estimates. Future study is to consider accurate ΔV orbit adjust calculations.

Scenarios with two or fewer relay spacecraft would not benefit from network rephasing upon a spacecraft failure applicable to where no value added is achieved through this action. Network rephasing would not be favorable to the halo orbit orbiter, and dedicated communication orbiter-to-lander, or the Martian moon lander scenarios. The halo orbit scenario would not benefit from rephasing since when one satellite is lost, network coverage is reduced from 99% to 50% with only one satellite in operation, and there is no added value to rephasing the remaining satellite. Network rephasing also does not apply to the single dedicated satellite with medium Mars's orbit in which the mission would terminate upon a communications failure occurred on the spacecraft. For the Mars moon lander network rephasing also does not apply for these landers whose orbits are fixed with the orbit of the associated moon. Network coverage is roughly halved as either of the moon lander relays spacecraft fail.

Satellite rephasing would be beneficial for the common period inclined constellation and somewhat difficult for multiple low Mars orbit constellation. For the aerosynchronous scenario the rephasing option would be a viable option since rephasing the two remaining

operational satellites would be useful in optimizing coverage performance though at a degraded state. Satellite rephasing would not be very beneficial for the halo scenario since the coverage performance would still be reduced by up to 50 percent upon failure of one satellite. Thus the rephasing option would be non-applicable for the halo scenario. Satellite rephasing would also be meaningless for the dedicated relay scenario.

The scenarios that would benefit most from a resulting network rephasing upon a spacecraft failure are the scenarios with the most satellites in the relay constellation. Moreover, for the scenario with large relay constellations, information adapted from Wertz suggested that the smaller the number of orbit planes in a relay network scenario, the greater the likelihood for the scenario to have graceful degradation.⁷ For example, in a six-satellite, two-plane constellation, if a satellite is lost, the constellation may be rephased at a small propellant cost and thus maintain a high performance level corresponding to a four-satellite constellation. In contrast, rephasing and graceful degradation may be more difficult for constellations with a larger number of orbit planes such as the low Mars orbit orbiter scenario. However despite the increased difficulty in such a rephase operation for this instance, there is benefit in network rephasing.

⁷ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 188.

Network rephasing upon a spacecraft failure for large constellations in the network would also lead to a greater fault tolerance. Scenarios such as the multiple low Mars orbiters, followed by the common period orbiter, and the aerosynchronous orbiter scenarios respectively would benefit most from a resulting network rephasing. For the case of the multiple low Mars orbit network a rephase of the remaining five satellites upon a spacecraft failure is possible with relatively low fuel impact of about 0.1 km/s ΔV per satellite. Similarly the common period inclined network can rephase its remaining three satellites though at the penalty of some coverage gaps resulting and a 0.2 km/s per relay spacecraft ΔV ; further study is needed on finding exact amount of surface coverage loss as a result of the rephase action. Rephasing for the aerosynchronous orbiter case may require moving the two remaining satellites with a 0.15 km/s per spacecraft ΔV so that the minimum spacing between the two is greater than the 120 degrees before the satellite failure, perhaps up to 180 degrees apart. Though coverage is still reduced from the three-satellite configuration.

Table 52, applicable to both small and large spacecraft classes, shows the compilation of the communications network fault tolerance risk factors compared among scenarios. General fault tolerance factors determined are coverage reduction from a spacecraft loss prior to and after the network rephase, and the approximate ΔV effort per spacecraft to perform the rephase. For the fault tolerance subfactor there is an added penalty imposed

on the halo orbiting, dedicated communication, and the Mars lunar lander scenarios. These scenarios could not readily accommodate a loss of satellite capability and hence exhibit poor fault tolerance capability.

Table 52. Network Fault Tolerance Risk Factors Relating to Constellation Configuration among Scenarios for Both Spacecraft Classes.

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Coverage Reduction from One Spacecraft Loss in Network Prior to Rephase	50%	33%	25%	17%	100%	50%
Approximate ΔV Effort per Spacecraft from Network Rephasing(km/s)	n/a	0.15 km/s	0.2 km/s	0.1 km/s	n/a	n/a
Estimated Coverage Reduction from Network Rephasing	n/a	5%	10%	0%	n/a	n/a
Fault Tolerance Penalty	0.6	n/a	n/a	n/a	0.9	0.7

Preliminary findings - network fault tolerance. The criteria in this portion of this study is to seek a favorable scenario with the most fault tolerance in its overall mission design. The amount of fault tolerance observed in this study is largely network size dependent. Scenarios with larger network constellations have an advantage in which constellation

rephasing is possible upon failure of a satellite in its network. The multiple low Mars orbit scenario exhibits the most fault tolerance among the scenarios, followed by the common period inclined scenario with its slightly lower number of spacecraft in the constellation. Conversely, the dedicated scenario had the least fault tolerance with a single satellite in the network followed by the Mars moon lander and halo orbiter scenarios, while the aerosynchronous scenario had a moderate fault tolerance with three satellites in its network. Future study should consider further development into a recovery program to fulfill mission objectives caused by one or more satellite failures in the network.

Spacecraft Reliability

Two general reliability items of interest are basic reliability and mission reliability. Basic reliability is the probability that all elements of a spacecraft system will survive, whereas the latter is the probability that at least the essential mission elements of a spacecraft will survive. These two measures can differ greatly in spacecraft that employ extensive fault tolerance. Since general reliability is far from being an exact science, this subfactor would contribute to relatively low weighting of the overall network lifetime design driver.

A relative comparison in overall spacecraft reliability is investigated in this portion of this study. This section on reliability is subdivided into the following spacecraft subsystems: propulsion, structure, thermal, attitude determination and control, communications, command and data handling, and the electrical power subsystems. In general spacecraft reliability is enhanced when several physically identical units serve the same spacecraft function. As stated in the fault tolerance section of this study, no attempt is made to obtain absolute values in spacecraft reliability. Absolute values with respect to various aspects in the reliability of spacecraft systems is largely subjective, serves little added value, and is beyond the scope of this study. Since reliability is one of the more subjectively determined factors relative among the parameters evaluated in this study, this parameter will be lightly weighted when used toward the Cost Model.

Table 53 shows spacecraft reliability by spacecraft subsystem among the spacecraft class. Reliabilities with a range of values indicate variation among scenarios in the spacecraft class, while singular reliability values indicate that the subsystem reliability is scenario-independent in that spacecraft class. Singular reliability values are shown for the propulsion, thermal, structure and the command and data handling subsystems.

Reliability values are largely derived from Wertz and Larson and are rough order estimates. From information derived from Wertz and Larson, for the small spacecraft class

an overall spacecraft reliability of 0.6 is assumed based on few spacecraft component redundancies in its spacecraft design.⁸ A higher spacecraft reliability rating of 0.65 is used for the large spacecraft class with its added component redundancies in its spacecraft design. A lower reliability rating is used for the small spacecraft design with its lower budget and lower redundancy in the small spacecraft design compared to the large spacecraft design. In this study the payload reliability is considered as the communications reliability, and again the communications reliability values are of rough order estimates. Several additional items have been identified in influencing spacecraft reliability. Each subsystem is further discussed below with the exception of the structure subsystem reliability.

Table 53. Spacecraft Reliability Evaluated by Spacecraft Subsystem among Spacecraft Class. See above text for reliability information.

	Small Spacecraft Class	Large Spacecraft Class
Propulsion	0.95	0.95
Structure	0.99	0.99
Thermal	0.99	0.99
Attitude Determination and Control	0.88 - 0.95	0.90 - 0.95
Communications/Payload	0.80 - 0.90	0.86 - 0.97
Command & Data Handling	0.93	0.93
Electrical Power Subsystem	0.80 - 0.95	0.80 - 0.95
Overall Spacecraft Reliability	0.60	0.65

⁸ Wertz, J. R., and Larson, W. J., Space Mission Analysis and Design, 302.

Reliability - Attitude Determination and Control

Attitude determination and control reliability are similar among scenarios with each spacecraft class with exception of the Mars lunar lander scenario since the Mars lunar lander scenario does not use attitude control upon landing in their respective lunar surfaces. However the Mars lunar lander relay spacecraft would need some attitude determination knowledge in controlling its uplink and downlink antenna gimbals. Table 54 shows the Mars lunar lander relay with a higher reliability rating over the other scenarios in each spacecraft class due to the absence of an attitude control mechanism. For the small spacecraft class all scenarios except for the Mars lunar scenario uses spin-stabilized attitude control that keeps the uplink antenna Earth-pointed. As reflected in the reliability values in Table 54, the spin-stabilized attitude control design for the small spacecraft class has a relatively lower pointing accuracy compared to the general three-axis stabilized attitude control design for the large spacecraft, again with the exception of the Mars lunar lander scenario. Compared among scenarios the multiple low Mars orbit and the medium-orbit dedicated communication orbiter inherently have greater attitude variations in their orbits that are reflected in their lower reliability values.

Table 54. Reliability of Attitude Determination and Control Compared among the Scenarios for the Small and Large Spacecraft Class.

Reliability Rating	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Small Spacecraft Class	0.90	0.90	0.89	0.88	0.88	0.95
Large Spacecraft Class	0.93	0.93	0.92	0.90	0.90	0.95

Reliability - Communications Payload

Reliability from the communications design among scenarios and spacecraft class is studied in this section. Communication reliability design considerations include the option of an antenna gimbaling design for the uplink and the downlink antenna design, and communications switching. The uplink path is the Mars-to-Earth data transfer while the downlink path is the Earth-to-Mars data transfer. Communication switching complexity among scenarios is detailed from Chapter 7 in this study.⁹ The communications payload design for the small spacecraft class and for the large spacecraft class are discussed next.

⁹ Danehy, M., "Martian Communications Network Design Trade Study," Masters Thesis, San José State University, (December 1997).

Communications payload - small spacecraft class. The uplink and downlink antennas have no antenna gimbaling mechanism and hence are of fixed antenna design. A standard baseline design of a one-meter parabolic uplink antenna and UHF downlink antenna used among all scenarios for the small spacecraft class. Thus the reliability rating among scenarios are nearly the same. The only exception is the Mars lunar lander scenario where a gimbaled antenna design for the uplink and the downlink is needed to enhance antenna-pointing accuracy. Antenna pointing accuracy also depends largely upon successful landing and positioning of the respective Mars lunar landers. Based on these findings using the standard antenna baseline with exception of the Mars lunar lander scenario, a reliability rating of 0.90 is used among scenario while a lower 0.80 rating is used for the Mars lunar scenario. Table 55 shows the reliability ratings from antenna design among scenarios.

Table 55. Reliability of Communications Design is compared among the Scenarios for the Small Spacecraft Class.

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Reliability	0.90	0.90	0.90	0.90	0.90	0.80

Communications payload - large spacecraft class. The decision to include a gimbal mechanism in an antenna design is largely determined by the Mars altitude of the scenario of interest. As opposed to the small spacecraft class, both the uplink and downlink antennas can include an option of an antenna gimbal along with various antenna designs

customized to each scenario. Another option to consider is to have antenna gimbaling on either the uplink or the downlink, and to use the spacecraft attitude to steer the fixed antenna to its pointing accuracy. This option may be applicable to all but the Mars moon scenario for reasons detailed below. And this option may be restricted from use due to thermal limitations.

For the aerosynchronous orbiter scenario, a gimbaling mechanism is only needed for the downlink antenna. The uplink antenna does not need to be gimbaled since the relay spacecraft ground trace and spacecraft attitude can be fixed with respect to the Martian surface. The Martian moon landers, with relatively similar orbit altitude as the aerosynchronous orbiters, will need a gimbaling mechanism for both the uplink and downlink antenna designs. Though the same side of each Martian moon faces the Mars surface, the initial decision of a fixed uplink antenna would be sufficient in taking advantage of the attitude of the Martian moons. However, since the individual uplink and downlink antenna pointing accuracies depend largely upon successful landing and positioning of the respective Mars lunar landers, a gimbaled antenna design can help compensate for any landing and positioning errors from the Mars lunar landing operation. But the gimbal feature would be of limited utility for the Mars lunar lander with the feature used only once at landing and would unlikely be used again.

The halo orbiter scenario is to have a gimbaled downlink 1.8-meter diameter parabolic antenna and a fixed five-meter uplink antenna option. Though a gimbaled antenna for both the uplink and the downlink antenna could be used for greater pointing accuracy and reduced signal loss, the single gimbal antenna design is sufficient in relaying information from Mars to the Earth. With an orbital period of about 9.2 years for each halo orbit, the orbital motion relative to Mars is relatively slow and manageable for antenna pointing accuracy via attitude control and antenna gimbaling.

The multiple low Mars orbiters and the dedicated communication orbiter scenarios show similar traits in that only the uplink antenna gimbal would be required. The downlink antenna does not need to be gimbaled since the downlink antenna could be pointed to specific areas of the Martian surface via spacecraft attitude control in their respective orbits. Also the amount of relay switching may be high due to limited visibility duration as a relay satellite speeds through the horizon as observed by any point on the Martian surface.

For the common period orbiter a gimbaled uplink antenna and a fixed horn and helix downlink design is utilized. The horn antenna, with less scatter energy compared to a parabolic antenna design, is used for most of the relay spacecraft orbit with exception of the periapse region. At the periapse region of the orbit, the less focused helix antenna

design is used. The horn and helix antenna pointing accuracy to the Mars surface of interest is maintained through attitude control of the relay spacecraft in accommodating its periodically dynamic elliptical orbit. A future feasibility study of such an antenna system design in this highly elliptical Mars orbit is to be considered.

Table 56. Reliability of Uplink and Downlink Communications Designs among the Scenarios are shown for the Large Spacecraft Class.

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Downlink (to Mars)	1.8 meter parabolic	horn	horn and helix	helix	helix	helix(Phobos) horn(Deimos)
Gimbaled?	No (possible)	No	No (possible)	No	No	Yes
Uplink (to Earth)	5 meter parabolic	5 meter parabolic	5 meter parabolic	5 meter parabolic	5 meter parabolic	5 meter parabolic
Gimbaled?	Yes	Yes	Yes	Yes	Yes	Yes
Reliability	0.97	0.93	0.90	0.89	0.88	0.86
Rank	1	2	3	4	5	6

Estimates in the reliability rating and ranking in antenna designs are shown in Table 56. Considerations in the determination of each reliability rating include general hardware reliability, an estimate of the system performance of the antenna design in its scenario operation, and the data transfer performance effects in the event of a failed gimbal

mechanism. The effect of the loss of the downlink gimbaling mechanism but with otherwise operational downlink antenna typically results in varying degrees of loss in ground commanding or different levels of ground command downtime. A loss of uplink antenna gimbal mechanism can range from delays in collecting data from Mars to the Earth, to complete loss of data collection capabilities.

Based on the above discussion the ranking from most favorable to least favorable are the halo orbit, the aerosynchronous orbiter, the common period inclined orbiter, the multiple low Mars orbit relay orbiter, the dedicated Mars orbiter and finally the Martian lunar lander relay. The halo orbiter is most favorable due to its relatively manageable antenna pointing with the slow relative motion between the relay spacecraft, Mars and the Earth. The aerosynchronous orbiters are next favorable with a gimbaling mechanism pointing the uplink antenna to the Earth in its relay orbit about Mars. The downlink antenna does not need to be gimbaled since the relay spacecraft ground trace and Mars nadir-pointing spacecraft attitude can be fixed with respect to the Martian surface. The common period inclined orbiter is similar in communications hardware compared to aerosynchronous scenario with the exception of the added horn and helix combination in downlink antenna design and with its relatively greater antenna pointing management from the dynamic elliptical orbit of the relay spacecraft. The multiple low Mars has a relatively dynamic antenna pointing management due to the faster relay orbit about Mars. Similarly

the dedicated communication orbiter has dynamic downlink antenna pointing, but with the added risk of a single point failure with a single satellite situation compared to the multiple low Mars orbiter scenario. The Mars lunar lander is least favorable since its relay performance depends largely upon successful landing and positioning of the respective Mars lunar landers. There is the possibility of a relay spacecraft landing on either the near or far side of the lunar surface rendering either the respective downlink or uplink communications relay useless. In such an event little or no antenna gimbaling is possible in recovering from an incorrect Mars lunar landing.

Reliability - Electrical Power Subsystem

The battery depth of discharge (DOD) and the amount of power cycling a relay spacecraft experience throughout its mission largely affect reliability in the electrical power subsystem. Both the DOD and the number of eclipse or power cycling periods can affect the life of the batteries and hence the life of the spacecraft. The greater the number of power cycles and the lower the DOD, the more the likelihood of a reduced life on the spacecraft batteries and a shorter life on the spacecraft and network. In this study the battery designs include the traditional nickel-cadmium design for the small spacecraft class and nickel-hydrogen for the large spacecraft class. Table 57 shows a comparison of the

electrical power subsystem reliability values based on estimated lifetime power cycle data from Table 49 and DOD values from Appendix I and J.

Table 57. The Reliability of the Electrical Power Subsystem is compared among the Scenarios for both Spacecraft Classes.

	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Comm. Orbiter - Lander	Martian Moon Landers (Phobos/Deimos)
Estimated Lifetime Power Cycles	0	1800	600	27500	24000	2200 / 1500
Depth of Discharge (DOD)	0.95	0.95	0.95	0.50	0.50	0.50 / 0.95
EPS Reliability	0.95	0.89	0.93	0.80	0.81	0.88 / 0.90
Rank	1	3	2	6	5	4

Preliminary findings – spacecraft reliability. From Table 57 the halo orbiter scenario achieves the most favorable electrical power subsystem reliability rating with the relay spacecraft being in constant solar visibility and having a high depth of discharge rating. The least favorable scenarios are the multiple low Mars orbiters and the medium-orbit dedicated communications orbiter scenarios with the high number of orbit eclipses and low depth of discharge experienced by both scenarios. Again as mentioned at the beginning of this section, spacecraft reliability is far from being an exact science, and this subfactor would contribute to relatively low weighting of the network lifetime design driver.

Graceful Degradation

Graceful degradation is the ability of a network to provide communication services for the user at a reduced level of service over time. Graceful degradation considerations are more likely significant towards the end of the communication relay mission life. In the pursuit of science collection the continued functionality of a relay network beyond its intended design life is desired. But beyond a mission design life, spacecraft operation will likely degrade in the form of reduced power levels or the failure of one or more satellites in a Mars relay network. Various subsystem malfunctions can be caused by any number of possible factors could lead to a complete loss of service to a relay network, a functional but reduced level of service can be obtained though with effort and planning needed to reconfigure the relay network. Not only would this be beneficial in obtaining more data, but also to promote continued exploration in the future.

In general there is a progressive tendency in component design towards an electronic design with fewer moving parts as opposed to a mechanical design from a cost and overall reliability standpoint. Design trends such as the use of multiple solid state communications amplifiers as opposed to the traditional Traveling Wave Tube Amplifier (TWTA) would allow greater graceful degradation of transmitter power as amplifiers

degrade or fail over time. However, when mechanical components were used failures were relatively expected; electronic components, while lasting longer, tend to have more random failures. Random failures are becoming a greater concern from the observation of the progressive use from mechanical components to electronic components used in spacecraft design. Thus graceful degradation among scenarios and spacecraft classes is rather subjective and is of rough order estimates in this study.

Preliminary findings – graceful degradation. With more redundant components inherent in the large spacecraft class, the large spacecraft has a greater graceful degradation rating than the small spacecraft class. Preliminary findings show that the halo scenario displays the greatest level of graceful degradation with its very low stationkeeping requirement over the life of the mission, relative slow orbital motion relative to Mars, and virtually constant visibility of each relay spacecraft to the Martian surface. The Mars lunar lander relay would have the next favorable graceful degradation qualities provided that the initial lunar landing was successful and ideally positioned. The remaining scenarios have relatively lower levels of graceful degradation from stationkeeping, orbital motion relative to Mars, and communication visibility considerations. The dedicated communications relay scenario would have the lowest graceful degradation rating with its solitary relay spacecraft. Table 58 shows the

comparison of factors in graceful degradation among scenarios using normalized values.

The lower the normalized value, the more favorable the graceful degradation.

Table 58. Graceful Degradation Risk Factors compared among Scenarios.

Graceful Degradation Rating (normalized)	Halo Orbit Orbiters	Aero-synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Small Spacecraft Class	0.30	0.50	0.55	0.60	0.65	0.45
Rank	1	3	4	5	6	2
Large Spacecraft Class	0.25	0.40	0.45	0.50	0.60	0.35
Rank	1	3	4	5	6	2

Evaluation and Summary of Spacecraft Lifetime Driver

The collection of the nine lifetime subfactors with their respective normalized values among scenarios is shown in Appendix K. The values shown in Appendix K for each of the nine lifetime subfactors are that of rough order estimates, and relatively low weighting is used in evaluation of the network lifetime portion of the Cost Model. As observed from the analysis shown in Appendix K-4, the small spacecraft class has a slightly greater network lifetime characteristic. However, the lifetime subfactors show consistent ranking among scenarios that are independent of spacecraft class.

As shown in the sample calculation in row 81 in Appendix K-4, the evaluation of the lifetime subfactors is calculated with equal weight towards the network lifetime design driver. For the fault tolerance subfactor there is an added penalty imposed on the halo orbiting, dedicated communication, and the Mars lunar lander scenarios. These scenarios could not readily accommodate a loss of satellite capability and hence exhibit a poor fault tolerance capability. Based on findings in this section, the scenario ranking starting with the most favorable spacecraft lifetime characteristic to least favorable is as follows:

1. The common period inclined orbiter scenario
2. The Mars halo orbiter scenario
3. The Martian moon lander scenario
4. The aerosynchronous orbiter scenario
5. The multiple low Mars orbiter scenario
6. The dedicated communication orbiter with medium Mars orbit scenario.

CHAPTER 8

Mission Preparation and Planning Timeline Parameter

$$J = w_{\$}(\$) + w_{\text{Size}}(\text{Size}) + w_{\text{Comm}}(\text{Comm}) + w_{\text{Life}}(\text{Life}) + w_{\text{Plan}}(\text{Plan})$$

This chapter assesses for each scenario studied the planning timeline element involved towards establishing a Mars relay network. Of all parameters involved in the cost model of this study, the network timeline is perhaps the most difficult to determine objectively. Forces such as public pressure and the amount of congressional funding for aerospace programs may drive the network planning timeline despite all intentions of completing such a network to Mars in a timely manner. Another concern is the element of public interest. In a consumer-driven supply and demand economy, the public needs to perpetually see accomplishments in the space program. Therefore maintaining public interest in the space program is vital for the survival of the space program in likely receiving either private investments or government funding which would otherwise go to other programs.

Table 59 below shows the time-related factors compared among the scenarios in this study, with a lower value being more favorable. Two principal factors were determined to affect the amount of time needed to establish and operate a Mars communications network: the network constellation size, and the individual relay spacecraft size affecting

the network establishment schedule. These two factors are used toward determination of the likelihood of the network design to be established at Mars within three years from initial funding. Note that favorable scenarios have a lower value for the three-year likelihood factor. The determinations of the scoring criteria for each of the factors including the three-year likelihood factor are described next.

Table 59. Time-Related Subfactors in Completing Communications Relay Constellation compared among the Scenarios.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Network Size Affecting Network Establishment	0.2	0.3	0.4	0.6	0.15	0.2
(Number in Network)	(2)	(3)	(4)	(6)	(1)	(2)
Spacecraft Design Affecting Time						
Small Spacecraft Class	0.25	0.25	0.25	0.25	0.25	0.3
Large Spacecraft Class	0.45	0.35	0.4	0.35	0.35	0.45
Launch Vehicle Lead Time						
Small Spacecraft Class	0.2	0.25	0.25	0.5	0.2	0.2
Large Spacecraft Class	0.25	0.2	0.2	0.5	0.25	0.25

Table 59. Time-Related Subfactors (cont.).

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Likelihood of Network Established at Mars in 3 years						
Small Spacecraft Class	0.25	0.3	0.4	0.5	0.2	0.3
Large Spacecraft Class	0.35	0.4	0.5	0.6	0.3	0.35
Network Planning Establish Rating						
Small Spacecraft Class	0.225	0.2625	0.325	0.425	0.1875	0.2375
Large Spacecraft Class	0.3125	0.35	0.4125	0.5	0.2625	0.3

Network Size Affecting Network Establishment

This section investigates the communications relay network establishment of each scenario towards constellation completion and communications relay operation. From Tables 7, 12 and 13, and the normalized cost section of Appendix K, the favorable launch vehicles for the small spacecraft class scenarios studied include the Taurus, Athena, and China's CZ-2C Long March launch vehicles. Russia's Proton and China's CZ-2E and CZ3 launch vehicles are the favorable applicable launch vehicles for the scenarios studied for the large spacecraft class. In general, the larger network constellation the more time

likely is needed for the network to be established and readied for launch. Functional operation of the complete network is also dependent on the amount of time required to launch the entire network using one or more launch vehicles.

The traditional dedicated communication orbiter with its one satellite requirement will only require one launch for either spacecraft class and with a heritage approach to mission operations prior to launch would make this scenario most favorable from a mission planning standpoint. The Mars halo orbiter and the two Martian moon lander spacecraft have similar prelaunch activities, with two satellites used for this network requiring one launch for either spacecraft class. The aerosynchronous orbiters with its three satellites require one launch for small spacecraft class, two launches for the large spacecraft class. Likewise four satellites are needed for the common period inclined orbiter scenario; one launch is needed for the small spacecraft class, two launches for the large spacecraft class. Six satellites are needed for the multiple low Mars Orbit Orbiters; two launches are needed for the small spacecraft class, three launches for the large spacecraft class.

Program planning tools such as GANTT charts can be used to schedule key milestones throughout a mission program from System Requirements Review (SRR), System Design Review (SDR), Software Specification Review (SSR), Preliminary Design Review (PDR), and Critical Design Review (CDR), to Test Readiness Review (TRR) before

launch. Future study should consider use of detailed scheduling charts in determining and monitoring program progress prior to vehicle launch.

Preliminary findings – network constellation size subfactor. Hence based on findings in this section, the scenario ranking starting with the most favorable network establishment characteristic to least favorable is as follows:

1. The dedicated communication orbiter with medium Mars orbit
2. The Mars halo orbiters
3. The Martian moon landers
4. The aerosynchronous orbiters
5. The common period inclined orbiters
6. The multiple low Mars orbit orbiters.

Individual Relay Spacecraft Size Affecting Network Establishment

This section investigates the influence of spacecraft sizing on the network construction timeline. The evaluation of the spacecraft sizing is based on the spacecraft sizing design parameter information presented in Chapter 5. For the small spacecraft class, the spacecraft design is relatively simple and similar among the scenarios and is reflected in the *Spacecraft Design Affecting Time* section in Table 59. The only exception is for the

Mars lunar lander relay, which needs a gimbaled antenna design compared to the simple fixed one-meter parabolic antenna design for the other scenarios. The reason for a gimbaled antenna design for the Mars lunar lander is to account for the uncertainty in the final landing attitude of each lander on their respective Mars lunar surface. Greater data transfer effectiveness can be achieved with a gimbaled antenna design. Since all the other scenarios are not of the relay lander variety, a fixed antenna design can be used with attitude control ensuring adequate antenna pointing accuracy. For the large spacecraft class, there is greater design variation, complexity and redundancy compared to that of the small spacecraft design. Specifically, the downlink antenna designs, the relay spacecraft to Mars communication link, vary among scenarios. These variations are described and compared among scenarios in Table 60 below. The uplink antenna design is common among scenarios and consists of a gimbaled five-meter antenna.

Table 60. Downlink Antenna Design in Individual Spacecraft Size Affecting Network Establishment Compared among the Scenarios.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter - Lander	Martian Moon Landers
Small Spacecraft Class	1.0 m (fixed)	1.0 m (fixed)	1.0 m (fixed)	1.0 m (fixed)	1.0 m (fixed)	1.0 m (gimbaled)
Large Spacecraft Class	1.8 m parabolic	horn	horn/helix	helix	helix	horn (gimbaled)

Preliminary findings - vehicle design affecting network establishment timeline. In general a favorable vehicle design influencing the planning timeline includes simplicity in vehicle design. Added complexities such as antenna gimbaling compared to a fixed antenna design entails complexity and additional testing to the antenna hardware and software design that affects the mission planning timeline. As the vehicle design timeline section in Table 60 shows, the aerosynchronous orbiter, the multiple low Mars orbit orbiters, and the dedicated communication orbiter with medium Mars orbit relay spacecraft scenarios with similarly favorably low influence on the planning timeline. Conversely the Mars halo orbiters and Mars lunar relay lander scenarios have similarly large influence in the planning timeline. Preliminary findings in this section and from Table 60 show the scenario ranking from the most favorable spacecraft design influencing network establishment to least favorable is as follows:

1. The aerosynchronous orbiters
1. The multiple low Mars orbit orbiters
1. The dedicated communication orbiter with medium Mars orbit
4. The common period inclined orbiter
5. The Mars halo orbiters
5. The Martian moon landers.

Multiple 1 and 5 rankings of the scenarios shown above indicate tie scores.

Launch Vehicle Lead-Time

The launch vehicle lead-time is the amount of time specified by the launch vehicle vendor to have a launch vehicle produced, completed, and readied for mission launch in the mission planning timeline. Table 61 shows the launch vehicle lead-time for the small relay and the large relay spacecraft classes used in this study. From Table 61, the launch vehicle lead-time for the small relay spacecraft class ranges from 18 months to 33 months, whereas the launch vehicle lead-time for the large relay spacecraft class ranges from 18 months to 36 months.

Table 61 is representative of the launch vehicle selection available as of the middle 1990s with their associated estimated launch vehicle lead-times. The launch vehicle selection is constantly being updated with new and potentially cheaper launch vehicles, with the older and cost-ineffective launch vehicles being phased out. And with the introduction of new launch vehicle comes new risks and uncertainty that may lead to greater launch vehicle lead-times. Despite these limitations, use of the new launch vehicles clearly leads in the potential of lower cost and greater launch vehicle performance that ultimately benefits the aerospace industry. As stated in the previous two sections, the favorable launch vehicles for the small spacecraft class scenarios studied

include the Taurus, Athena, and China's CZ-2C Long March launch vehicles, while launch vehicles for the large spacecraft class include Russia's Proton and China's CZ-2E and CZ3 launch vehicles.

Table 61. Launch Vehicle Lead-Times for the Small Relay and Large Relay Spacecraft Classes Used in this Study.^{1, 2}

Launch Vehicle	Launch Vehicle Lead Time for Small Relay Spacecraft Class (Months)	Launch Vehicle Lead Time for Large Relay Spacecraft Class (Months)
Pegasus	20	
Taurus	18	
LMLV - 2	18	
CZ - 2C	24	
CZ - 2E		24
CZ - 3	24	24
Delta II (6925)	30	30
Delta II (7925)	30	30
Atlas I	33	33
Atlas II		36
Atlas IIA		36
H II		36
Atlas IIAS		36
Ariane 44L		29
Titan III & TOS		33
Titan III & IUS		33
Ariane 54L		36
Proton		18
Titan IV (SRMU)/Centaur		33

¹ Kelley, J., "US Expendable Launch Vehicle Data for Planetary Missions," <http://www.jsc.nasa.gov/bu2/USELV.html> (September 25, 1996) website.

² Kelley, J., "International Expendable Launch Vehicle Data for Planetary Missions," <http://www.jsc.nasa.gov/bu2/IELV.html> (September 25, 1996) website.

Preliminary findings – launch vehicle lead-time. In summary, information from Table 59 and Table 61 of the scenario ranking from the launch vehicle selection influencing network establishment show that the Chinese launch vehicle lead-times of 24 months compared to about an 18-month lead time for the Russian and LMLV launch vehicles. The relatively larger variation in multiple low Mars orbit orbiter between the small spacecraft class and the large spacecraft class is a result of the greater difference in the number of components and number of spacecraft in the network requiring resources. Conversely, little variation is observed between the small spacecraft class and the large spacecraft class in the dedicated communication orbiter with its singular spacecraft, and the remaining scenarios show slightly more variation in that respect. Thus a higher rating is used for the scenarios where a Chinese launch vehicle is used and a lower value for use of the other launch vehicle with one exception. The common period inclined and low Mars orbit scenarios which both utilize multiple launches are rated poorer than their single launch counterparts. However, the launch vehicle lead-times values noted in Table 61 are estimated normalized values and because of the relatively moderate uncertainty in these values, moderate weighting is imposed in the evaluation of this effect towards Cost Model evaluation. Additions of new launch vehicles with its associated launch vehicle lead-times will need to be updated and included into Table 61.

Planning Timeliness Subfactor - The Likelihood of a Complete Network**To be Established within Three Years from Initial Funding**

Realistically, as demonstrated time and time again in the aerospace industry, a comprehensive spacecraft mission that includes flight software in its spacecraft design rarely is completed from the initial funding stage to cruise phase after launch within three years. However, in the spirit of striving towards achieving this ambitious goal, a rating of the likelihood of a network design to be established within three years from initial funding is investigated. This rating applies through the last vehicle launch towards relay network completion and does not necessarily include the cruise phase of the interplanetary journey to Mars. The planning stages prior to launch entails a highly coordinated effort with various subcontractors interacting with one or more primary contractors along with the customer.

The actual amount of time needed to establish a relay network through post-launch is dependent on numerous factors of uncertain time duration. One factor is the total number of components supplied by numerous vendors. The greater the number of components that make up the spacecraft, the greater the risk of component delivery delays by one or more vendors. This factor more likely impacts the large spacecraft class with its greater number of components compared to the small spacecraft class. Another factor is the amount of time needed to complete the countless number of testing phases such as

satellite integration and testing to be performed in a timely but thorough manner. This also involves fundamental development of testing methods that are unique to an individual program. A third factor is the number of launches that is dependent on the total number of spacecraft needed to complete a network. Many more factors are involved in a relay satellite program, but even with this oversimplification of a program timeline various uncertainties affecting only these three factors can have a dramatic effect on the amount of time needed to produce a finished spacecraft product to be integrated to the launch vehicle. Each test phase is usually performed via a serial process where delays in each test phase affect the start of the next test phase as well as the timeline of the remaining test phases. Future study is to consider detailed program planning timeline.

Preliminary findings – the planning timeliness subfactor. Table 62 shows the preliminary findings of the planning phase scheduling factors from initial funding to the interplanetary cruise phase after launch using the three general factors outlined in the above section and that are considered in the three-year likelihood network establishment value. The lower values shown in Table 62 indicate a greater likelihood of completion of the scenario of interest within a three-year period. Values noted are estimated normalized values and because of the relatively large uncertainty in these values, low weighting is imposed in the evaluation of this effect towards Cost Model evaluation. The relatively larger variation in multiple low Mars orbit orbiter between the small spacecraft class and the large spacecraft class is a result of the greater difference in the number of components

and number of spacecraft in the network requiring resources. Conversely, little variation is observed between the small spacecraft class and the large spacecraft class in the dedicated communication orbiter with its singular spacecraft, and the remaining scenarios show slightly more variation in that respect.

Table 62. Planning Phase Scheduling Factors compared among the Scenarios.

	Halo Orbit Orbiters	Aero- synchronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit Orbiters	Dedicated Communication Orbiter	Martian Moon Landers
Likelihood of a Network Established through Post-Launch within 3 years						
Small Spacecraft Class	0.25	0.30	0.40	0.50	0.20	0.30
Large Spacecraft Class	0.30	0.36	0.47	0.75	0.22	0.35

Evaluation and Summary of the Planning Timeline Driver

The collection of the four subfactors with their respective normalized values among scenarios is shown in Appendix K-5. Each subfactor is evaluated and factored into the planning driver equally in this study. The average normalized value of the subfactors was evaluated in the mission preparation and planning timeline driver. From the planning timeline standpoint, the small spacecraft class showed a more favorable

planning timeline compared to that of the large spacecraft class. But for both spacecraft classes the halo-orbiting scenario showed the most favorable followed by the Mars lunar lander scenario. A relative low weighting would be used in this design driver since many uncertainties affecting the planning timeline design driver have not been accounted for and would be difficult to incorporate into the overall Cost Model.

CHAPTER 9

Evaluation and Initial Findings from the Cost Modeling Method

$$J = w_{\$}(\$) + w_{\text{Size}}(\text{Size}) + w_{\text{Comm}}(\text{Comm}) + w_{\text{Life}}(\text{Life}) + w_{\text{Plan}}(\text{Plan})$$

The preceding Chapters 4 through 8 focused on the quantitative values of each term in the Cost Model which are then normalized by dividing each value by the largest or least favorable value described in the respective tables. These are the normalized design parameter values n_i studied in Chapters 4 through 8 used towards the Cost Model evaluation. The evaluation of favorable scenarios is selected using the lowest and hence most favorable normalized values and values within five percent of the most favorable value from Appendix K-5. The terms “design drivers” and “parameters” will be used interchangeably throughout this study. Appendix K shows the normalized values derived from Tables 12 and 13 from Chapter 4, Tables 16 through 18 from Chapter 5, Tables 33 and 34 from Chapter 6, Tables 38 through 41 from Chapter 7, Tables 16 through 18 from Chapter 8. This chapter focuses on the weighting factors W_i that are multiplied with its corresponding n_i used in the evaluation of the Cost Model. Initially the individual design drivers to the Cost Model are studied to investigate the relative sensitivities of each design driver. Then different permutations of weighting factors are evaluated starting with single drivers dominating the Cost Model followed by evaluation of multiple drivers. The findings of the single and multiple design driver domination are used to rate

sensitivities of each design driver. The ultimate goal in this study is to determine favorable scenarios using realistic weights or coefficients with each design parameter based upon current conditions in the aerospace industry.

Evaluation of the Individual Parameters in the Cost Model

The equation is based on the summation of five design parameters studied in Chapters 4 through 8. The weighting factors W_i are then multiplied with the normalized design parameter values n_i and applied to Equation (2-1) and Equation (2-2) towards the total cost J . The following design drivers are discussed individually with their effects on the total cost J . An important general observation is that a scenario selected based on one driver alone instead of a scenario selection based on thorough consideration of all the drivers in this study would most likely result in a short-term solution. Appendix K-1 through Appendix K-6 shows the normalized values for various design driver emphases. The normalized values for each design driver are based on the scenario with the highest value. For the design of interest the lower the normalized value the more favorable the scenario. The goal in this study is to determine and then to incorporate current realistic conditions into the Cost Model. In evaluating the Cost Model with current conditions the

network lifetime and the planning timeline design drivers were determined to have lower weightings due to the relatively large level of uncertainty in these drivers.

Appendix K-1 shows the total mission cost evaluation among scenarios for both the small and large spacecraft classes. The number of required launches largely influences total mission costs. The mission costs vary from the minimum single launches needed for the dedicated relay scenario, small and large spacecraft class, to the maximum of six launches needed to establish the network for the low Mars orbiter scenario, large spacecraft class, using the H-2 launch vehicle. As expected the small spacecraft class showed lower normalized values with a tendency towards smaller launch vehicles needed compared to the large spacecraft class. As expected the dedicated relay scenario for both spacecraft classes has the lowest and most favorable normalized value. Though no other scenario was within five percent of the dedicated relay scenario, the halo orbiter and the Mars lunar scenario were the next favorable scenarios for both spacecraft classes.

Appendix K-2 shows the evaluation of the spacecraft ΔV sizing among scenarios for both spacecraft classes. The individual ΔV requirements for the halo orbiter and the Mars lunar lander are shown, and the average normalized value was used in evaluation of the Cost Model. The L_2 halo orbiter of this scenario has the higher ΔV requirement due to the slower velocity the L_2 relay spacecraft that would have to achieve upon arrival at

the Sun-Mars L_2 heliocentric orbit. Among the remaining scenarios, the common-period inclined scenario exhibited the lowest ΔV with the highly elliptical orbit of the final destination requiring less ΔV compared to the scenarios with their relatively tight circular orbits in their final destination.

Appendix K-3 displays the evaluation of the communications performance driver with sample calculations using an Excel spreadsheet. The performance calculation is based on a combination of downlink data rate throughput, uplink data rate throughput, information on the percentage of Mars relay communication network visibility greater than a 30 degree elevation compared among scenarios obtained from Danehy, and the degree of complication with the communication system configuration among scenarios. An estimated uplink and downlink data rate throughput, in total bits, is calculated over a 74-hour period based upon the common-period scenario with the greatest period studied.

Generally the estimated throughput for both uplink and downlink includes the day data rate fraction and the night data rate fraction calculation applied to all scenarios with two exceptions; the common-period inclined and the halo orbiting scenarios. In this study the day data rate is defined as the data rate the communications relay is collecting data outside the eclipse portion of its orbit, while the night data rate is at the rate data is collect in the eclipse portion of its orbit. The night data rate is typically higher than the

day data rate collected by a communications relay spacecraft due to the lower noise temperature in collecting data with the communications relay in eclipse.

For the common-period inclined scenario the worst-case event of only calculating the day data rate for the data throughput calculation was used. Most of the common-period inclined relay orbit is spent collecting data towards its apoapse portion of its elliptical orbit, although in its highly elliptical orbit, the day data rate and night data rate fractions will vary. One observational note is that the day and night data rates obtained from Danehy are only for the worst case condition of apoapse, although the night data rate values are not used in this calculation. Thus conditions would be more favorable for the common-period inclined relay once its orbital operation is further investigated to include the variations of the day and night data collection in its highly elliptical orbit in future study.

The data throughput calculation for the halo orbiter is more straightforward. Only the day data rate is used for the data throughput calculation for the L_1 relay with the relay facing the day side of the Red Planet. Conversely the night data rate is used for the data throughput calculation for the L_2 relay with the relay facing the night side of the Red Planet. Thus the total data throughput is the sum of the data throughput from each relay. The day and night data rate calculation is also straightforward with the Mars orbiting scenarios. In general, most of the Mars relay orbit for the Mars orbiting scenarios is out

of Mars eclipse. Thus the majority of the throughput calculation is at the day data rate. Therefore the total data throughput is the sum of the data throughput of the eclipse portion and the non-eclipse portion from each relay. Finally, the communications performance is calculated using the average of the uplink data throughput, the downlink data throughput, and the communication system configuration rating. Based on the evaluation of the communications performance, the halo orbiter scenario was observed to be most favorable followed by the common-period inclined scenario. The inclusion of the variations of the day and night data collection in the common-period inclined highly elliptical orbit would improve its rating which may surpass the halo rating.

Appendix K-4 shows the evaluation of the network lifetime driver with sample calculations from an Excel spreadsheet. A total of nine subfactors relating to relay network lifetime among scenarios are studied. The spacecraft fuel requirement applies to both spacecraft classes and varies from a favorable negligible amount for the Mars lunar scenario to a relatively high amount of stationkeeping required for each aerosynchronous orbiter. The total power requirement and the battery recharge margin varies from a uniformly low power requirement among scenarios for the small spacecraft class to the greatest power requirement for the low Mars orbit relays for the large spacecraft class.

The quantitative battery power loss risk and the risk from orbit altitude applies to both spacecraft classes and varies from the relatively low risk from the halo orbit to the

relatively high risk from the low Mars orbit relay. The variation in thermal power requirement was low among scenarios in each spacecraft class. The network fault tolerance risk applied to both spacecraft classes ranged from low risk for the aerosynchronous scenario to the relatively high fault-tolerance risk for the dedicated relay scenario. Finally the variation from graceful degradation ranged from relatively low for the halo-orbiting scenario, large spacecraft class, to relatively high for the dedicated relay, small spacecraft class. Overall, the halo-orbiting relay scenario exhibited the most favorable characteristics for both classes. Conversely, the low Mars orbiter relay scenario showed the least favorable characteristics for both classes.

Appendix K-5 shows the evaluation of the subfactors comprising of mission preparation and planning timeline driver among scenarios for both spacecraft classes. From observation of the cumulative evaluation of the planning timeline subfactors in completing communications relay constellation, the small relay class is favored over the large spacecraft class, and the dedicated relay scenario is favored among scenarios.

Appendix K-6 shows the Cost Model evaluation spreadsheet. Rows 2 through 15 in Appendix K-6 shows the parametric values of the individual design driver alone. Thus for example, when considering the monetary cost design driver, the weighting for this driver is set to 1.0, while the weighting for the remaining drivers is each set to 0.0. The values in parentheses, as shown in the halo orbiter and the Mars lander scenarios, are

average values of the two relay spacecraft in those scenarios. Also sample calculations are periodically shown for selected cells throughout Appendix K-6. From the above sections of Appendix K-6 the communications performance driver did appear to be a more significant driver among the design drivers studied with the lowest normalized value for the most favorable scenario, the halo orbiter relay.

The Evaluation of Multiple Parameters in the Cost Model

Rows 16 through 30 in Appendix K-6 show the Cost Model values with one term dominating the Cost Model. Thus for example, when considering the monetary cost design driver, the weighting for this driver is set to 1.0, while the weighting for the remaining drivers is each set to 0.1. Similarly rows 31 through 60, 61 through 90, 91 through 102 show the Cost Model values with two, three, and four terms dominating the Cost Model respectively. All possible two, three and four design-driver-dominating combinations are explored to observe sensitivities in various permutations. From mention of the communications performance driver appearing to be a more significant driver in the previous section of Appendix K-6, no one driver appeared to be more significant than any other driver among the design drivers studied.

Rows 104 through 105 in Appendix K-6 show equal weight and are referred to as the “straight” calculation. These calculations show a consistent favoring of the halo orbiting and the common period inclined scenarios for the small spacecraft class, and the halo-orbiting scenario for the large spacecraft class. Rows 107 and 108 in Appendix K-6 show the evaluation of the Cost Model based on weightings using current realistic conditions. This calculation shows favoring of the halo orbiting and the common period inclined scenarios for the large spacecraft class, and the halo-orbiting scenario for the small spacecraft class. The evaluation of the Cost Model is considered with the weightings set equally to 1.0 for each design driver.

The Evaluation of the Cost Model using Current Conditions

Appendix K-7 shows a plot of the Cost Model totals among scenarios using weightings of current conditions. Note that the totals for each scenario are shown in pairs. The left pair of points for each scenario represents the small spacecraft class, while the right point in the pair represents the large spacecraft. The weighting factor W_i for each driver are determined by the current conditions in the aerospace industry, although the weightings are to a degree also influenced by the amount of confidence and uncertainty in the evaluation of each driver. For the mission monetary cost driver, the weighting was determined to be high with mission cost being a very important factor in

considering most planetary missions. Moderately high weighting was used in the spacecraft sizing design driver since there is room for optimization in spacecraft design for both the small and large spacecraft thereby leaving room in this portion of this study for refinement in trajectory science and spacecraft design. Moderately high weighting was also used in the communications performance driver stressing the importance for consistent performance, reliability, and the timeliness of information transfer suited ideally for universal use by mission scientists and end users. Low weighting was used in determining the spacecraft lifetime driver given the relative uncertainty in spacecraft reliability, graceful degradation, and fault tolerance, coupled into the various fuel and power requirements. Moderate weighting is used for the mission preparation and planning timeline driver where the general uncertainty lies in the actual outcome of product deliverance, spacecraft integration and test in the planning and prelaunch timeline.

Thus for the current or realistic case, the following relative weightings are used for each parameter for both large and small spacecraft class:

Mission monetary cost driver – high (0.75)

Spacecraft sizing design driver – moderately high (0.6)

Communications performance driver – moderately high (0.65)

Spacecraft lifetime driver – low (0.4)

Mission preparation and planning timeline driver – moderate (0.5)

Summary of the Cost Model

In this study two spacecraft classes and six general scenarios were compared in the determination of a favorable scenario. Initial findings of the parametric values of each design driver in the Cost Model show that the mission cost and the planning timeline drivers favor the dedicated communications relay scenario. The communications performance driver and the network lifetime designs favor the halo-orbiting scenario, and the ΔV sizing driver favors the common-period inclined scenario. No weighting considerations were performed during the parametric value evaluation. As mentioned previously the investigation of sensitivities in the design drivers revealed that no single design driver is observed dominating the Cost Model.

From the evaluation of the single and multiple dominating design drivers, the halo orbiter scenario is consistently favored overall. However, an important item to consider is that the common-period inclined scenario was evaluated based on worst case conditions and may actually be the favored scenario. Future evaluation may consider the common-period inclined scenario incorporating actual conditions. Initial findings using even weightings from the “straight” case from this trade study show that the halo-orbiting

and common period inclined scenarios for both spacecraft classes as the most favorable various weightings, and the Mars lunar scenario is favored for the large spacecraft class.

A review of the results using weightings with current conditions supports findings similar to the “straight” case, with the halo orbiter and the common period inclined scenarios exhibiting favorable characteristics in both spacecraft classes, as well as the Mars lunar scenario, large spacecraft class. However strictly from an absolute evaluation standpoint, the small spacecraft class halo orbiting or the small spacecraft class common-period inclined scenarios are the most favored. Based on realistic conditions in this study, the small spacecraft class halo orbiting or the small spacecraft class common-period inclined scenarios are best suited as communication relays for universal data transmission.

CHAPTER 10

Conclusion: Only the Beginning...Mars or Bust!

A Mars relay satellite network has the potential to be an important element in future Mars missions. It can reduce the communications burdens on landers or other spacecraft and can provide an enhanced communications link to the Deep Space Network. The Mars Relay Satellite network enables continuous communications with landers or orbiters, and allows the mission designer the greater freedom in the exploration of Mars and in the pursuit of knowledge of the Red Planet.

A Mars relay satellite network may be an important and cost-effective element in future Mars missions. An effective relay network can reduce the communications burden on future spacecraft design and can provide an enhanced communications link to the Deep Space Network. The Mars relay network enables continued communications with landers or orbiters, and allows lander locations to be chosen based on Martian features of interest and serving science rather than limited to Mars-Earth visibility. The selection of a Mars Relay Satellite design has a significant impact on overall mission performance for future Mars missions that will subsequently rely on this network. And with a well-designed Mars communications relay network future Mars spacecraft will become

indispensable, popular and reliable not unlike current telecommunications systems on Earth.

In this design trade study a simple but effective Cost Model was developed in determining favorable scenarios. This Cost Model comprises of a Cost Estimating Relationship (CER) consisting of a series of weighted design drivers. With a Cost Model established several possible scenarios representative of the possible variety of available scenarios were used as inputs to the Cost Model and for comparison, although other possible scenarios and hybrid scenarios of those scenarios studied could be investigated in future study. Each design driver in the Cost Model was discussed in detail, then evaluated as objectively as possible among scenarios. Also with established Cost Model relative weightings for each design driver was ultimately evaluated based on user conditions with minimal uncertainty in the weighting values. Some limitations encountered in this study include the lack of inclusion of industry or proprietary information that would otherwise be of benefit in this study, the limited documentation that is specific to this study, and items of inherent uncertainty such as reliability that were included in the Cost Model.

Further determination of Mars communication network specifications will depend greatly on numerous factors that can be controlled as well as uncontrollable factors. Future Mars missions will benefit and upon complete operation of Mars communications

relay network will depend on such a network. Based on this study using the current and most realistic conditions, the common-period inclined orbiting network scenario was observed to have the most favorable characteristics of an effective Mars communication relay network followed by the Mars halo orbiting network scenario. The Mars halo orbiting network scenario achieved the lowest overall mission cost as calculated from current factors reflecting realistic conditions. However, the evaluation of the common-period inclined orbiting network was generally based on the worst-case conditions, and upon improved evaluation techniques for this scenario the evaluation may indeed favor the common-period inclined orbiting network.

The Unfinished Journey...

Useful trade studies prove to be challenging since changes are periodically needed to adapt to periodically changing conditions. As an example, strong lobbying efforts benefiting from the aerospace industry as well as public attitude can change the influential climate in the aerospace industry. This study is an attempt to determine the most favorable Mars scenario from current economic and technological conditions. Political and other conditions that are less predictable have been applied in this study on a more restricted and limited basis. Future considerations in this study are to adjust values in the Cost Model evaluation reflecting changes in industry, economy and policy trends

affecting the aerospace industry. These include adding new launch vehicles being introduced that are coming on-line, and incorporating new technologies. Also information from the Government Release of NAFCOM could benefit in the spacecraft development of this study. Since the author is not a NASA employee, the Government Release of NAFCOM, complete with access to specific projects' names, costs, weights, and sample printouts, are not available for use in this study. Mars science, atmospheric and environmental information from Mars Pathfinder and Mars Global Surveyor can be used in determining spacecraft lifetime considerations in the Cost Model. Other future considerations include exploring hybrid scenarios that are combinations of the six scenarios investigated in this trade study, as well as technological improvements such as ion propulsion that can be incorporated in spacecraft or launch vehicle design.

Bibliography

Chao C., "Long-Term Orbit Perturbations of the Draim four-Satellite Constellations," Journal of Guidance, Control, and Dynamics 15, no. 6, (Nov. - Dec. 1992).

Covault, C., "Russian 'InterBall' Mission Launched," Aviation Week & Space Technology, (August 14, 1995): 64.

Danehy, M., "Martian Communications Network Design Trade Study," Masters Thesis, San José State University, (December 1997).

Draim, J. E., "A Common-Period Four-Satellite Continuous Global Coverage Constellation," Journal of Guidance, Control and Dynamics 10, no. 5, (Sept.- Oct. 1987): 492 - 499.

Draim, J. E., "Continuous Global N-Tuple Coverage with $(2N+2)$ Satellites," Journal of Guidance and Control, and Dynamics 14, no. 1, (Jan.-Feb. 1991): 1.

"Expendable Launch Vehicles Summary for JPL Mission Planning," Jet Propulsion Laboratory internal document D-6936, Rev. A, Pasadena, CA (February 1991).

Farquhar R. and others, "Trajectories and Orbital Maneuvers for the First Libration-Point Satellite," Journal of Guidance, Control, and Dynamics, vol. 3, (Nov. - Dec. 1980): 549 - 555.

Grondine, E., "Mars on the Cheap," Space News, (18-24 November 1996): 12.

Hamblin, W., Christiansen, E., Exploring the Planets, (New York: Macmillan Publishing Company, 1990).

Hamilton, C., "Phobos 1 and 2 to Mars," NASA National Space Science Data Center, Washington, D. C. (1994).

Hastrup, R. and others, "Mars Relay Satellite Orbit Design Considerations for Global Support of Robotic Surface Missions," AAS Paper no.93-564, AAS/AIAA Astrodynamics Specialist Conference (16-19 August 1993).

Hopkins, R., "Long-Term Revisit Coverage Using Multi-Satellite Constellations." AIAA Paper no.88-4276, AIAA/AAS Astrodynamics Conference, Minneapolis, MN, (August 15-17, 1988).

Hubbard, S. and others, "Concepts for a Small Mars Sample Return Mission Using Microtechnology," Jet Propulsion Laboratory internal document D-8822, Pasadena, CA (October 1991).

Hubbard, S. and others, "Mars Environmental Survey (MESUR) Science Objectives and Mission Description," AMES Research Center study report, Moffett Field, CA (July 19, 1991).

Iannotta, B., "NASA Chooses Infrared Sensor," Space News, (July 31-Aug 6, 1994): 1.

Kelley, J., "International Expendable Launch Vehicle Data for Planetary Missions," <http://www.jsc.nasa.gov/bu2/IELV.html> (September 25, 1996) website.

Kelley, J., Mission and Data Analysis (MODA), <http://www.jsc.nasa.gov/bu2/MODA.html> (October 20, 1997) website.

Kelley, J., Mission Operations Cost Model, <http://www.jsc.nasa.gov/bu2/MOCM.html> (June 27, 1997) website.

Kelley, J., "US Expendable Launch Vehicle Data for Planetary Missions," <http://www.jsc.nasa.gov/bu2/USELV.html> (September 25, 1996) website.

McKay, D., and others, "Search for Past Life on Mars: Possible Relic Biogenic Activity in Martian Meteorite ALH84001," Science 273, no. 5277, (August 16, 1996): 924-930.

Microsoft FORTRAN, computer software, version 5.0, (Microsoft Corporation, 1982-1992).

Middour J., "An Efficient Technique for Computation of Satellite Earth Coverage," AIAA Paper no. 89-0452, 27th Aerospace Sciences (July 19, 1988).

Orbital Workbench, computer software, version 1.1, (Cygnus Engineering, 1990-1992).

Palamarik T. and others, "Mission Design for an Aerosynchronous Orbiter," AAS Paper no. 89-197, AAS/AIAA Conference (1989).

Pearson, M., "Scientists Find NASA Well-Positioned to Pursue Mars Life", Aerospace Daily, (Aug 22, 1996): 1.

Pernicka, H., lecture notes from Aerospace Engineering 110 course, San José State University, (spring 1994).

Pernicka, H. and others, "Use of Halo Orbits to Provide a Communication Link between Earth and Mars," AIAA Paper no.92-4585-CP, AIAA/AAS Astrodynamics Conference (1992).

"PLANET-B (FY 1998)", <http://www.isas.ac.jp/info/future/planetB-e.html#Tag:0> (1998) website.

Rogers, T., "The People Payload: Does the U. S. Space Transportation Industry Take Space Tourism Seriously?" Ad Astra, (March/April 1995): 45-55.

Rokey, M. J., Dodd, S. R., "Mars Observer Mission Plan," Jet Propulsion Laboratory internal document D-2518, Revision F, (July 1992).

Sergeyevsky, A. and others, "Interplanetary Mission Design Handbook, Volume I, Part 2," JPL publication no. 82-43, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California (September 15, 1983).

Sietzen, F., "1998: A Key Year in the Commercialization of Space," Ad Astra, November/December 1997, 32-36.

"Space FAQ 13/13 - Orbital and Planetary Launch Services,"
<http://www.landfield.com/faqs/space/launchers/.html> (February 5, 1998) website.

Stevenson, S. M., "An Evolutionary Communications Scenario for Mars Exploration," The CASE for Mars III: Strategies for Exploration-Technical, Extract AAS no. 87-268, vol. 75, (1990).

Stokley, C., "Mars Mission Schedule," Mars Underground, (March 6, 1996): 5.

Svitek, T., and others, "Mars Relay Spacecraft: A Low Cost Approach," Ninth Annual AIAA/Utah State University Conference on Small Satellites, Logan, Utah, (September 18-21, 1995).

Truly, R. H., "America at the Threshold, Report of the Synthesis Group on America's Space Exploration Initiative," NASA report, U. S. Government Printing Office, Washington, D. C., (May 3, 1991).

Truly, R. H., "Report of the 90-Day Study on Human Exploration of the Moon and Mars," internal NASA report, (November 1989).

Veverka, J., "The Moons of Mars," AAS Paper no. 86-157, NASA Mars Conference, National Academy of Sciences, Washington, D.C., (July 21-23, 1986).

“Weekly Compendium,” Lockheed Martin corporate weekly publication, (24 March 1995).

Wertz, J. R., Larson, W. J., Space Mission Analysis and Design, 2nd ed. (Torrance, CA: Microcosm Press, and The Netherlands: Kluwer Academic Publishers, 1992).

Wertz, J. R., Larson, W. J., Space Mission Analysis and Design Workbook, (Torrance, CA, Microcosm Press, 1993).

Willoughby, A., “Data Relay Solar Electric Utility Space Ship (DRSEUSS),” forum, Mars Forum IV, NASA/Ames Research Center (12 - 14 December 1994).

Zubrin, R. M., "The Significance of the Martian Frontier- Mars as the Final Hope for Earth." Ad Astra, (September/October 1994): 29-37.

Appendix A: Descriptions of Various Restricted Forms of the Three-Body Problem.

I. The Elliptic Restricted Three-Body Problem

The elliptic restricted three-body problem is a simplified version of the general “Three-Body Problem.” In the general instance, each of the three bodies of finite mass move under the influence from the gravity fields of the other two bodies. Unfortunately there is no general closed-form solution to the equations governing their motion. Of the required eighteen integrals of motion required in order to completely determine a solution, only ten have been found. But particular solutions can be found using some simplifying assumptions. One is to restrict one of the bodies to be of infinitesimal mass. In this instance the infinitesimal mass is the spacecraft, such that it has a negligible gravitational effect on the motion of the other two bodies or primaries, in this instance the Sun-Mars system, thereby simplifying the problem to the restricted three-body problem. Furthermore if the motion of the primaries is limited to elliptic orbits about their center of mass (barycenter), the problem then reduces to the elliptic restricted three-body problem.

II. The Circular, Restricted Three Body Problem (CR3BP)

The motion of two bodies orbiting each other can be described using Kepler’s laws, known as the Two-Body Problem. The Two-Body Problem was first solved analytically by Isaac Newton. Similarly the Three-Body Problem investigates the motion of three bodies. However, unlike the Two-Body Problem, there is no known closed-form solution for the Three-Body Problem despite many efforts devoted to such a solution. But some

simplification to the problem leads to several particular solutions. Restricting the mass of the third body small enough that the third body would not disturb the motion of the other two primary bodies such that $M_3 \ll M_2 < M_1$, and having only circular orbits to the motion of the primary bodies, the Three Body Problem becomes known as the Circular Restricted Three Body Problem (CR3BP). This simplified Three-Body Problem has been useful in describing spacecraft motion about a large body, as in the case of describing the motion of an Earth-orbiting spacecraft or of a Mars-orbiting spacecraft. The CR3BP also exposes another fundamental observation made by the mathematician Lagrange who determined that the CR3BP has five “equilibrium” points, also known as libration points or L-points, as illustrated in Figure 1(A) from Chapter 3 of this study.

L_1 , L_2 , and L_3 are collinear with L_1 lying between the two primary bodies, and closer to M_2 . L_2 lies nearly the same distance on the opposite side of M_2 . L_3 lies opposite M_2 , on the far side of M_1 . L_4 and L_5 each lie a distance equal to M_2-M_1 from M_1 , exactly 60 degrees ahead of L_4 and behind L_5 . At each of these points, the gravitational and centrifugal forces acting on M_3 balance and the body will remain at this position. However, these are unstable equilibrium points, and the slightest disturbance from equilibrium would send the body at an L-point away from that L-point. In this study, the Sun-Mars L_1 and L_2 points are considered.

Appendix B: Cost Estimating Relationships (CER) (cont.)

Ib. A CER is shown for the Small Spacecraft Class - Options 4, 5 and 6.

Mission Cost Parameters (Cost in FY96\$K)	RDT&E Param. (Unit)	1st Unit Para. (Unit)	RDT&E Cost	First Unit Cost	Option 4 Multiple LowOrb.	RDT&E Cost	First Unit Cost	Option 5 Dedicated Comm	RDT&E Cost	First Unit Cost	Option 6 Mars Moons
Payload											
Antennas	5 (m)	5 (m)	2623.36	614.46	3228.93	2623.36	614.46	614.46	2623.36	614.46	1167.47
Comm Electronics	8 (kg)	8 (kg)	3931.27	1432.00	7525.07	3931.27	1432.00	1432.00	3931.27	1432.00	2720.80
Spacecraft Bus	27.48 (kg)	27.48 (kg)	19275.80	2372.55	12467.58	19275.80	2372.55	2372.55	19275.80	2372.55	4507.84
Aerospace Ground Equip (\$K)	75000	n/a	8250.00			8250.00			8250.00		
Launch Ops & Orbital Support	n/a	100 (kg)		208.00	1093.03		208.00	208.00		208.00	395.20
Program Level (\$K)	35000	3000	12600.00	990.00	5202.39	12600.00	990.00	990.00	12600.00	990.00	1881.00
Software Dev Cost (\$K, 1750A Architecture)	n/a	n/a	7330.00			7330.00			7330.00		
Subtotal RDT&E Cost (FY 96\$M)			45.76			45.76			45.76		
Subtotal Cost 1stUnitCost				5.41			5.41			5.41	
Subtotal Cost Constellation					28.42			5.41			10.28
Space Segment= RDT&E + Constellation TotalCost('96\$M)					74.18			51.17			58.04

Appendix B: Cost Estimating Relationships (CER) (cont.)

II. A CER is shown for the Large Spacecraft Class (Halo Orbit and Aerosynchronous Scenarios)

Mission Cost			RDT&E 1st Unit		Option 1	RDT&E 1st Unit			Option 2
Parameters	RDT&E	1st Unit	Cost		Halo Orbit	RDT&E	1st Unit	Cost	Aero-synchronous Constellation
*** (Cost in FY96\$K)***									
Payload									
Antennas	127.95	127.95	17767.23	4046.073	7687.538	127.95	127.95	17767.2	4046.073
Comm Electronics	40.95	40.95	12329.55	7330.05	13927.1	40.95	40.95	12329.5	7330.05
Spacecraft Bus	512.1	512.1	72584	22561.86	42867.53	427.1	427.1	63234	19619.12
Apogee Kick Motor									
3-Axis Stabilized	3648540	3648540	56917.22	18972.41	7602.878	3648540	3648540	3801.43	3801.439
Aerospace Grd Equip	100000		11000			100000		11000	
Launch Ops and Orbital Support		450		712	1352.8		450		712
Program Level	100000	45000	36000	14850	28215	100000	45000	36000	14850
Software Dev Cost	Size (Kwords)		Size (Kwords)		Size (Kwords)		Size (Kwords)		
(1750A Archit.)									
Communications	5					5			
Attitude Sensor Processing	2					3			
Attitude Determination and Control	8					9			
Autonomy	3					3			
Fault Detection	3					3			
Power Management	0.5					0.5			
Thermal Control	0.5					1			
	Lines of code(Kwords)		Cost(\$ M)			Lines of code(Kwords)		Cost(\$ M)	
	22		17.52			24.5		17.52	
			1st Unit Cost	Option 1				1st Unit Cost	Option 2
				Halo Orbit					Aerosynchronous Constellation
			RDT&E	(\$M)				RDT&E	
			120.200	34.6499	65.83497			110.850	31.7072
Space Segment Cost (FY96 \$M)					186.0357				87.69452
Number of spacecraft					2				3
S/C dry mass (kg)					723.6				625.3

Appendix B: Cost Estimating Relationships (CER) (cont.)

II. A CER is shown for the Large Spacecraft Class (Common Four Period Orbit and Multiple Low Mars Orbit Scenarios)

Mission Cost		RDT&E		1st Unit	Option 3	RDT&E		1st Unit	Option 4
Parameters	RDT&E	1st Unit		Cost	Common	RDT&E	1st Unit	Cost	Multiple
*** (Cost in FY96\$K)***					Four Period				
Payload									
Antennas	127.95	127.95	17767.23	4046.07	14606.32	127.95	127.95	17767.23	21261.86
Comm Electronics	40.95	40.95	12329.55	7330.05	26461.48	40.95	40.95	12329.55	38518.96
Spacecraft Bus	508.1	508.1	72144	22428.0	81448.32	529.1	529.1	74454	118561.2
Apogee Kick Motor									
3-Axis Stabilized	3648540	3648540	3801.439	3801.43	13723.2	3648540	3648540	3801.439	19976.33
Aerospce Grd Equip	100000		11000			100000		11000	
Launch Ops and Orbital Support		450		712	2570.32		450		712
Program Level	100000	45000	36000	14850	53608.5	100000	45000	36000	14850
Software Dev Cost (1750A)	Size (Kwords)		Size (Kwords)						
Architecture	5					5			
Attitude Sensor	2					2			
Processing									
Attitude	10					15			
Determination and Control									
Autonomy	3					3			
Fault Detection	3					3			
Power Management	0.5					1			
Thermal Control	0.5					1			
	Lines of code(Kwords)	Cost(\$M)				Lines of code(Kwords)	Cost(\$M)		
	24	17.52				30	17.52		
			1st Unit	Option 3				1st Unit	Option 4
			Cost	Common				Cost	Multiple
				Four Period					Low Orbit
			RDT&E					RDT&E	
			119.7608	34.5141	125.0864			122.070	35.224
					244.8472				182.0835
Space Segment Cost (FY96 \$M)									304.1543
Number of spacecraft					4				6
S/C dry mass (kg)					638.4				804.2

Appendix B: Cost Estimating Relationships (CER) (cont.)

II. A CER is shown for the Large Spacecraft Class (Dedicated Communication and Mars Moons Scenarios)

Mission Cost	RDT&E		1st Unit	Option 5		RDT&E		1st Unit	Option 6	
Parameters	RDT&E	1st Unit	Cost	Dedicated	RDT&E	1st Unit	Cost	Mars Moons		
*** (Cost in FY96\$K)***										
Payload										
Antennas	127.95	127.95	17767.23	4046.073	127.95	127.95	17767.23	4046.073	7687.53	
Comm Electronics	40.95	40.95	12329.55	7330.05	40.95	40.95	12329.55	7330.05	13927.1	
Spacecraft Bus	529.1	529.1	74454	22561.86	410.1	410.1	61364	19015.02	42867.53	
Apogee Kick Motor										
3-Axis Stabilized	3648540	3648540	3801.439	3801.439	3648540	3648540	3801.439	3801.439	7602.878	
Aerospace Grd Equip	100000		11000		100000		11000			
Launch Ops and Orbital Support		450	712	712		450		712	1352.8	
Program Level	100000	45000	36000	14850	100000	45000	36000	14850	28215	
Software Dev Cost (1750A	Size (Kwords)		Size (Kwords)		Size (Kwords)		Size (Kwords)		Size (Kwords)	
Architecture)										
Communications	5				5					
Attitude Sensor	2				2					
Processing										
Attitude	11				1					
Determination and Control										
Autonomy	3				3					
Fault Detection	3				3					
Power Management	1				1					
Thermal Control	1				1					
	Lines of code(Kwords)		Cost(\$ M)		Lines of code(Kwords)		Cost(\$M)			
	26		17.52		16		17.52			
			1st Unit Cost	Option 5			1st Unit Cost	Option 6		
				Dedicated				Mars Moons		
			RDT&E	Communication			RDT&E			
			122.070	35.2245	35.22452		108.9808	31.10315	65.8349	
Space Segment Cost (FY96 \$M)					157.2953				174.815	
Number of spacecraft					1				2	
S/C dry mass (kg)					734.7				625	

Appendix C: Software Cost Estimation Packages that Exist to Date but are not used in this Study.

Some currently available software estimation products found in this study are shown below. Due to the proprietary nature associated with these software packages, these software packages were unavailable for use in this study. Future trade studies may utilize these or future methods in evaluating mission operations and support costs.

Basic COCOMO software cost model: This is a model for estimating the number of person-months required developing the software. The model also estimates the development schedule in months. This model is based on Barry Boehm's Constructive Cost Model (COCOMO).

PRICE-S: The Martin Marietta (Now Lockheed Martin) Price subsidiary developed this model originally. Developed in 1977 by F. Freiman and Dr. R. Park, it was the first commercially available detailed parametric software cost model to be extensively marketed and used.

REVIC: The Revised Intermediate Constructive Cost Model (REVIC) model was developed by Ray Kile and the U.S. Air Force Cost Analysis Agency. The model predicts the development costs for software development from requirements analysis through completion of the software acceptance testing and maintenance costs for fifteen years.

SASET : A sophisticated software cost estimating model previously built by Martin Marietta, Astronautics Division during 1986-1990 under contract with the Naval Center for Cost Analysis under auspices of the Office of Naval Research. The model combines database information from over 500 successfully completed software projects with "expert" factors to determine software development and maintenance costs, schedules, and sizing numbers used by engineers, estimators, and top-level management.

SEER-SEM: Part of a family of software and hardware costs, schedule and risk estimation tools. SEER-SEM is used throughout the aerospace and defense industry on two continents. All issues found in today's software environments are addressed.

SLIM : Quantitative Software Management, Inc. (QSM) developed SLIM for Windows 3.1. This was derived from the Rayleigh-Norden model and has been validated over a 15 year time period with thousands of real, completed projects.

SOFTCOST-R & SOFTCOST-ADA: The SoftCost-R model was developed by Dr. Don Reifer. This model was based on the work of Dr. Robert Tausworthe of the NASA Jet Propulsion Laboratory. SoftCost-R is applicable to all types of programs and considers all phases of the software development cycle.

Appendix D: Description and Flowchart of FORTRAN Program in Determining the ΔV Budget.

Appendix D-1: Brief Description of the FORTRAN Program.

Once the launch and arrival vectors with their corresponding dates are obtained, these vectors are inputs to a FORTRAN program. With these inputs placed into the program, the transfer angle from Earth to Mars in the heliocentric reference frame is computed using Kepler's Laws of Motion. The transfer angle computation involves the use of Newton's method in iteratively computing the eccentric anomaly variable. However, the solution to the transfer angle equation involves solving for a cosine equation where two possible solutions satisfy the cosine equation. This ambiguity is resolved using a second equation that involves the cross product and using two equations and two unknowns. Then the following orbital parameters of the transfer orbit are found by solving Lambert's equations numerically: the semi-major axis, a ; the orbit eccentricity, e ; the latus rectum, p ; and the velocity vector at launch. Ultimately the following orbital characteristics are found:

- The hyperbolic excess velocity at Earth launch (V_∞);
- The launch energy (C_3);
- The arrival ΔV (ΔV_2) in vector and scalar form; and
- The total ΔV for various Earth-to-Mars, Earth-to Mars L_1 , and Earth-to-Mars L_2 trajectories using various departure and arrival dates.

Appendix D: Description and Flowchart of FORTRAN Program in Determining the ΔV Budget (cont.)

Appendix D-2: Flowchart of FORTRAN Program Used in Determining the ΔV Budget.

Appendix E: Derived Equations in Obtaining the ΔV Budget.

The general energy equation for orbital trajectories of a spacecraft is given by:

$$\mathbf{E} = \frac{\mathbf{V}^2}{2} - \frac{\mu}{r} \quad (\text{E-1})$$

For interplanetary missions, the launch energy, C_3 , is defined as twice the specific mechanical energy of a satellite

$$C_3 = 2\mathbf{E} = \mathbf{V}^2 - \frac{2\mu}{r} \quad (\text{E-2})$$

In the FORTRAN program, the Mars mission was modeled with a launch and arrival date selected. Then the following launch and arrival vectors (**note**: vector values are shown in bold letters) were obtained from the Earth and Mars positions corresponding to these dates respectively:

$$\mathbf{R}_{\text{Earth}} = A_{\text{Earth}} \mathbf{x} + B_{\text{Earth}} \mathbf{y} + C_{\text{Earth}} \mathbf{z} \quad (\text{E-3})$$

$$\mathbf{R}_{\text{Mars}} = D_{\text{Mars}} \mathbf{x} + E_{\text{Mars}} \mathbf{y} + J_{\text{Mars}} \mathbf{z} \quad (\text{E-4})$$

Gravtiational Parameters ($\mu \equiv GM$) of the Sun, the Earth, and Mars are used in this study.

Time of flight (TOF) is the difference between Earth departure epoch and Mars arrival epoch calculated from the input file.

The transfer angle (TA) of Earth to Mars interplanetary trajectory defined as:

$$\text{TA} = \cos^{-1} \left(\frac{A_{\text{Earth}} \mathbf{x} D_{\text{Mars}} \mathbf{x} + B_{\text{Earth}} \mathbf{y} E_{\text{Mars}} \mathbf{y} + C_{\text{Earth}} \mathbf{z} J_{\text{Mars}} \mathbf{z}}{\mathbf{R}_{\text{Earth}} \mathbf{R}_{\text{Mars}}} \right) \quad (\text{E-5})$$

Obtaining the velocity vector (F and G are LAGRANGIAN coefficients which are functions of $\mathbf{R}$ and θ^* , and are also functions of a , e , and p):

$$V(x) = (1/G) D_{\text{Mars}} - F A_{\text{Earth}} \quad (\text{E-6})$$

$$V(y) = (1/G) E_{\text{Mars}} - F B_{\text{Earth}} \quad (\text{E-7})$$

$$V(z) = (1/G) J_{\text{Mars}} - F C_{\text{Earth}} \quad (\text{E-8})$$

$$V_{\text{departure}} = \sqrt{V^2(x) + V^2(y) + V^2(z)} \quad (\text{E-9})$$

$$V_{\text{Earth}} = \sqrt{(\mu_{\text{Sun}} / R_{\text{Earth}})} \quad (\text{E-10})$$

$$\cos \gamma_{\text{departure}} = \sqrt{(a_{\text{Sun-Earth}}(1 - e^2) / (V_{\infty}^2 R_{\text{Earth}}))} \quad (\text{E-11})$$

$$C_3 = V_{\text{Earth}}^2 + V_{\text{departure}}^2 - (2 V_{\text{Earth}} V_{\text{departure}} \cos \gamma_{\text{departure}}) \quad (\text{E-12})$$

For hyperbolic departure from Earth @ 150 km altitude:

$$R_{\text{perigee}} = 6378 + 150$$

$$a_{\text{hyperbolic}} = -\mu_{\text{Earth}} / C_3 \quad (\text{E-13})$$

$$V_{\text{hyperbolic}} = \sqrt{(2 \mu_{\text{Sun}} / r_{\text{perigee}} - \mu_{\text{Sun}} / a_{\text{hyperbolic}})} \quad (\text{E-14})$$

$$\Delta V_1 = V_{\text{hyperbolic}} - V_{\text{cir}} \quad (\text{E-15})$$

For hyperbolic arrival to Mars:

$$V_{\text{arrival}} = \sqrt{(2 / R_{\text{Mars}} - 1/2a)} \quad (\text{E-16})$$

$$\cos \gamma_{\text{arrival}} = \sqrt{(a_{\text{Sun-Mars}}(1 - e^2) / (V_{\text{arrival}}^2 R_{\text{Mars}}))} \quad (\text{E-17})$$

$$V_{\text{Mars}} = \sqrt{(\mu_{\text{Sun}} a (1 - e^2) / R_{\text{Mars}})} \quad (\text{E-18})$$

$$\Delta V_2 = V_{\text{Mars}}^2 + V_{\text{arrival}}^2 - (2 V_{\text{Mars}} V_{\text{arrival}} \cos \gamma_{\text{arrival}}) \quad (\text{E-19})$$

$$M = -7.4895722 * 2 \pi \text{ radians/degree} \quad (\text{E-20})$$

$$n = \sqrt{\mu_{\text{Sun}} / a^3} \quad (\text{E-21})$$

$$t = M / n \quad (\text{E-22})$$

$$\text{time} = \text{TOF} + t \quad (\text{E-23})$$

$$r = a (1 - e \cos(E)) \quad (\text{E-24})$$

$$V = \sqrt{(2 \mu_{\text{Sun}} / r_{\text{perigee}} - \mu_{\text{Sun}} / a_{\text{hyperbolic}})} \quad (\text{E-25})$$

$$\text{transfer angle} = \sqrt{\mu_{\text{Sun}} / a^3} (\text{TOF}) (86400 \text{ seconds} / \text{day}) \quad (\text{E-26})$$

$$\theta^*_{\text{launch}} = \cos^{-1} ((1-e^2)*a/(e*ry(4)) - (1/e)) \quad (\text{E-27})$$

$$\theta^*_{\text{arrival}} = \cos^{-1} ((1-e^2)*a/(e*rz(4)) - (1/e)) \quad (\text{E-28})$$

(Note: Need to calculate respective θ^* at proper quadrants using subroutine ‘LAMBER’)

$$\theta^*_{\text{arrival}} - \theta^*_{\text{launch}} = \text{transfer angle} \quad (\text{E-29})$$

Subroutine LAMBER is used to determine the type of transfer orbit (TYPE I or II, TYPE A or B, and with the selection of hyperbolic, parabolic or elliptical transfer orbit), and a , e , and latus rectum, p . Subroutine LAMBER Uses Lambert’s Theorem of Time of Flight between departure and arrival points being dependent upon the semi-major axis, a , rather than the eccentricity, e , of the transfer orbit.

(GIVEN RY, RZ, TOF AND INITIAL GUESSES FOR X AND Y)

Find TA (Transfer angle in radians) -
 Use DOT Product to obtain between 0 and 180 degrees,
 then check to see if between 180 and 360 degrees
 using Z-component of cross-product of Ry and Rz

$$TA = \cos^{-1}((RY(1)*RZ(1)+RY(2)*RZ(2)+RY(3)*RZ(3))/(RY(4)*RZ(4))) \quad (E-30)$$

$$CRZ = RY(1)*RZ(2) - RY(2)*RZ(1) \quad (E-31)$$

$$IF (CRZ.LT.0.) TA = 2 * \pi - ta$$

$$TA1 = TA * 180 / \pi \quad (E-32)$$

(FIND CHORD)

$$C = \sqrt{RY(4)^2 + RZ(4)^2 - 2*RY(4)*RZ(4)*\cos(TA)} \quad (E-33)$$

(FIND SEMI-PERIMETER)

$$S = (RY(4)+RZ(4)+C)/2 \quad (E-34)$$

(FIND LAMBDA)

$$LAMBDA = (\sqrt{RY(4)*RZ(4)} * \cos(TA/2))/S \quad (E-35)$$

(FIND W)

$$W = \text{ATAN}((RZ(4)/RY(4))^{0.25}) - (\pi/4) \quad (E-36)$$

(FIND L)

$$L = ((\tan(2*W))^2 + (\sin(TA/4))^2) / ((\tan(2*W))^2 + (\sin(TA/4))^2 + \cos(TA/2)) \quad (E-37)$$

(FIND M)

$$M = (8*\mu*((TOF*86400)^2))/((S^3)*((1+LAMBDA)^6)) \quad (E-38)$$

(BEGIN ITERATION TO CONVERGE UPON CORRECT X AND Y)

USE JJ INPUT TO DEFINE INITIAL CONDITION:

JJ=0 MEANS HYPERBOLA OR PARABOLA (X=0),

JJ=1 MEANS ELLIPSE (X=L)

$$DXOLD = 10000.00$$

IF (JJ.EQ.0) THEN

$$X = 1 \times 10^{-14}$$

IF (JJ.EQ.1) THEN

$$X = L$$

(Find PHI Variable)

$$PHI = \frac{a}{(3+(1/(5+N+(9N/7)/(1+((+16N/63)/(1+((25N/99)/(1+((36N/143)/(1+b)))))))} \quad (E-39)$$

where

$$N = X/(((DSQRT(1+X))+1)**2)$$

$$a = 8 * (DSQRT(1+X)+1)$$

$$b = 49/195*N/(1+64/255*N / (1+81 /323*N/(1 +(100/399*N))))$$

(FIND H1)

$$H1 = ((L+X)**2*(1+3*X+PHI))/((1+2*X+L)*(4 *X+PHI*(3+X))) \quad (E-40)$$

(FIND H2)

$$H2 = (M*(X-L+PHI))/((1+2*X+L)*(4*X+PHI*(3+X))) \quad (E-41)$$

(FIND B)

$$B = (27*H2)/(4*((1+H1)**3)) \quad (E-42)$$

(FIND U)

$$U = -B/(2*(DSQRT(1+B)+1)) \quad (E-43)$$

(FIND KU)

$$KU = (1/3)/(1-(4U/27/(1-(8U/27/(1-(2U/9/(1-(22U/81/(1-(208U/891/A)))))))))) \quad (E-44)$$

where

$$A = 1-(((418/ 1755)*U)/(1-598/22950*U/1-(700/2907)*U))$$

(SOLVE FOR NEW Y)

$$YNEW = ((1+H1)/3)*(2+(SQRT(1+B)/(1-(2*U*(KU**2))))) \quad (E-45)$$

(SOLVE FOR NEW X)

$$XNEW=SQRT(((1-L)/2)**2+(M/YNEW**2))-((1+L)/2) \quad (E-46)$$

(ITERATE UNTIL XNEW EQUALS X)

*** SOLVE FOR ORBIT PARAMETERS a, p, and ecc ***

$$a = M*S*(1+LAMBDA)**2/(8*X*Y**2) \quad (E-47)$$

$$p = (2*RY(4)*RZ(4)*Y^2*(1+X)^2*(SIN(TA/2)^2))/(M*S*(1+LAMBDA)^2) \quad (E-48)$$

$$\text{ecc} = \sqrt{1. - (p/a)} \quad (\text{E-49})$$

(Solve for functions F and G, THE LAGRANGIAN COEFFICIENTS)

$$F = 1 - (RZ(4)/P) * (1 - \cos(TA)) \quad (\text{E-50})$$

$$G = RY(4) * RZ(4) * \sin(TA) / \text{SQRT}(\text{MU} * P) \quad (\text{E-51})$$

(VELOCITY COMPONENTS AT LAUNCH)

$$V(1) = (1/G) * (RZ(1) - (F * RY(1))) \quad (\text{E-52})$$

$$V(2) = (1/G) * (RZ(2) - (F * RY(2))) \quad (\text{E-53})$$

$$V(3) = (1/G) * (RZ(3) - (F * RY(3))) \quad (\text{E-54})$$

VELOCITY MAGNITUDE (Spacecraft velocity)

$$V(4) = \sqrt{V(1)^2 + V(2)^2 + V(3)^2} \quad (\text{E-55})$$

(End of LAMBERT subroutine)

(Beginning of subroutine KEPLER)

THIS SUBROUTINE ITERATIVELY COMPUTES THE ECCENTRIC ANOMALY (E) USING NEWTON'S METHOD TO COMPUTE THETA

$$f = m - (E - \text{ecc} * \sin(E)) \quad (\text{E-56})$$

Initialize the following variables and program counter:

$$tp = 0.0$$

$$n = \sqrt{\text{mu}/a^3}$$

$$m = (t - tp) * n$$

$$j = 0$$

$$e = m$$

If the absolute value of f in equation (E-56) is greater than 1.0×10^{-15} ,

solve for the following two equations:

$$f' = ecc * \cos(E) - 1.0 \quad (E-57)$$

$$E = E - (f/f') \quad (E-58)$$

Then reiterate equations (E-57) and (E-58) as needed

$$\theta = 2. * \text{atan}(\text{sqrt}((1.+ecc)/(1.-ecc)) * \tan(E/2.)) \quad (E-59)$$

(EARTH HELIOCENTRIC VELOCITY)

$$V_{\text{earth}} = \sqrt{\mu/R_Y(4)} \quad (E-60)$$

$$\cos \gamma_{\text{Dep}} = \sqrt{(\mu * a * (1. - ecc^2)) / (V(4) * R_Y(4))} \quad (E-61)$$

(Launch Energy C_3 is defined as $(V_{\infty})^2$)

$$C_3 = (v_{\text{earth}}^2) + (v(4)^2) - (2 * v_{\text{earth}} * v(4) * \cos \gamma_{\text{Dep}}) \quad (E-62)$$

$$V_{\infty} = \sqrt{C_3} \quad (E-63)$$

Solving for ΔV_1 , the hyperbolic departure from Earth at 150km-altitude perigee.

$$R_{\text{perigee}} = 6378 + 150$$

$$\mu_{\text{Earth}} = 398600$$

$$\mu_{\text{Mars}} = 42,828$$

$$\mu = 132712439935.48$$

$$R_p = 3,950 \text{ kilometers}$$

$$\text{HypA} = -\mu_{\text{Earth}}/C_3 \quad (E-64)$$

$$\text{HypV} = \sqrt{\mu * (2/R_{\text{perigee}} - 1/\text{HypA})} \quad (E-65)$$

$$V_{\text{cir}} = \sqrt{\mu/R_{\text{perigee}}} \quad (E-66)$$

$$\Delta V_1 = \text{HypV} - V_{\text{cir}} \quad (E-67)$$

Solving for the hyperbolic arrival to Mars.

$$V_{\text{arr}} = \sqrt{(2 \mu_{\text{sun}} (1/R_Z(4) - 1/(2*a)))} \quad (E-68)$$

$$\text{Cosy} = \sqrt{(\mu * a * (1 - \text{ecc}^2)) / (\text{Varr} * \text{RZ}(4))} \quad (\text{E-69})$$

Solving for the hyperbolic arrival velocity at Mars towards evaluating ΔV_2 .

$$v_{\text{Mars}} = \sqrt{(2 * \mu / \text{RZ}(4)) - (\mu/a)} \quad (\text{E-70})$$

Calculating for V_{∞} :

$$V_{\infty} = \sqrt{((v_{\text{Mars}})^2 + (\text{Varr})^2 - (2 * v_{\text{Mars}} * \text{Varr} * \text{Cosy}))} \quad (\text{E-71})$$

Finding spacecraft velocity at periapse:

$$V_p = \sqrt{(2 * \mu_{\text{Mars}} / R_p) + ((v_{\text{Mars}})^2 + (\text{Varr})^2 - (2 * v_{\text{Mars}} * \text{Varr} * \text{Cosy}))} \quad (\text{E-72})$$

Finding semi-major axis at hyperbolic Mars arrival:

$$A_{\text{HypAr}} = \sqrt{(2 / R_p) - (V_p^2 / \mu_{\text{Mars}})} \quad (\text{E-73})$$

Mars circular orbit velocity:

$$V_{\text{cir}} = \sqrt{(\mu_{\text{Mars}} / R_p)} \quad (\text{E-74})$$

Mars elliptical orbit velocity (42560 km for Draim 74-hour orbit):

$$V_{\text{ellipse}} = \sqrt{(2 * \mu_{\text{Mars}} / R_p) - (\mu_{\text{Mars}}/42560)} \quad (\text{E-75})$$

ΔV (circular orbit):

$$\Delta V_{\text{circular}} = V_p - V_{\text{cir}} \quad (\text{E-76})$$

ΔV (elliptical orbit)

$$\Delta V_{\text{elliptical}} = V_p - V_{\text{ellipse}} \quad (\text{E-77})$$

Total ΔV (circular orbit)

$$\text{TotDelVC} = \Delta V_1 + \Delta V_{2C} \quad (\text{E-78})$$

Total ΔV (elliptical orbit)

$$\text{TotDelVE} = \Delta V_1 + \Delta V_{2e} \quad (\text{E-79})$$

Mars L_1 ΔV_2 calculations

Hyperbolic Arrival to Mars L_1 (in Vector Form)

$$\Delta V_{2 L_1} = V_{\text{Halo}} - V_{\text{arr}} \quad (\text{E-80})$$

where:

$$V_{\text{Halo}} = \dot{x}X + \dot{y}Y + \dot{z}Z + (dL_1/R_{\text{MARS}} * e_{\text{Mars}} * \sin(E/R_{\text{star}}) * V_{\text{star}}X + dL_1 * \Theta \cdot Y$$

$$V_{\text{arr}} = (V_{\text{arr}} * \cos \gamma) X + (V_{\text{arr}} * \sin \gamma) Y$$

A Halo Data file for Mars L_1 is obtained from Danehy used to determine the minimum ΔV_2 using velocity components selected from the four relative maximum and minimum Y halo positions (labeled as A, B, C, D) from the Sun - Mars position along one Mars heliocentric orbit.

$$V_{\text{star}} = \sqrt{G M_{\text{Star}} / L_{\text{star}}}$$

where:

G (the Gravitational Constant) = $6.67 \times 10^{-11} \text{ m}^3 / (\text{kg sec}^2)$,

M_{Star} (the mass of the Sun - Mars system) = $1.989100642 \times 10^{30} \text{ kg}$,

L_{star} (Sun - Mars Mean distance) = 227940974 km.

Thus $V_{\text{star}} = 24.130393219 \text{ km/s}$

$$\dot{x}_A = -0.0014349278 * V_{\text{star}} \quad (\text{E-81a})$$

$$\dot{y}_A = -0.0000290458 * V_{\text{star}} \quad (\text{E-81b})$$

$$\dot{z}_A = 0.0012607976 * V_{\text{star}} \quad (\text{E-81c})$$

$$\dot{x}_B = -0.0000177086 * V_{\text{star}} \quad (\text{E-82a})$$

$$\dot{y}_B = 0.0043018427 * V_{\text{star}} \quad (\text{E-82b})$$

$$\dot{z}B = 0.0000136673 * V_{star} \quad (E-82c)$$

$$\dot{x}C = 0.0014324387 * V_{star} \quad (E-83a)$$

$$\dot{y}C = -0.000030760 * V_{star} \quad (E-83b)$$

$$\dot{z}C = -0.0012600003 * V_{star} \quad (E-83c)$$

$$\dot{x}D = -0.0000135431 * V_{star} \quad (E-84a)$$

$$\dot{y}D = -0.0047515270 * V_{star} \quad (E-84b)$$

$$\dot{z}D = -0.0000100992 * V_{star} \quad (E-84c)$$

$$V_{arr} = \sqrt{2\mu (1/RZ(4) - 1/(2.a))} \quad (E-85)$$

$$\text{Cosy} = \sqrt{(\mu * a * (1 - \text{ecc}^2)) / (V_{arr} * RZ(4))} \quad (E-86)$$

$$V_{arrX} = V_{arr} * \text{Siny} \quad (E-87)$$

$$V_{arrY} = V_{arr} * \text{Cosy} \quad (E-88)$$

$$\text{Cosy}_1 = \text{Cosy} * \text{dpr} \quad (E-89)$$

$$vL1x = RZ(4)0.093368 / (r_{mars} * 1.496 \times 10^8) * \sin(EA / r_{mars}) * V_{star} \quad (E-90)$$

$$vL1y = \sqrt{\mu 1.523691 * 1.496 \times 10^8 (1 - .093368^2) / (RMARS(1.496 \times 10^8) RZ(4))} \quad (E-91)$$

$$\Delta V_{2A} = \sqrt{(vL1x + \dot{x}A - V_{arrX})^2 + (\dot{y}A + vL1y - V_{arrY})^2 + \dot{z}A^2} \quad (E-92a)$$

$$\Delta V_{2B} = \sqrt{(vL1x + \dot{x}B - V_{arrX})^2 + (\dot{y}B + vL1y - V_{arrY})^2 + \dot{z}B^2} \quad (E-92b)$$

$$\Delta V_{2C} = \sqrt{(vL1x + \dot{x}C - V_{arrX})^2 + (\dot{y}C + vL1y - V_{arrY})^2 + \dot{z}C^2} \quad (E-92c)$$

$$\Delta V_{2D} = \sqrt{(vL1x + \dot{x}D - V_{arrX})^2 + (\dot{y}D + vL1y - V_{arrY})^2 + \dot{z}D^2} \quad (E-92d)$$

$$vL1 = \sqrt{(vL1x^2 + vL1y^2)} \quad (E-93)$$

$$\Delta V_2 = \sqrt{((vL1)**2.d0 + (V_{arr})**2.d0 - (2.d0 * vL1 * V_{arr} * \text{Cosy}))} \quad (E-94)$$

$$\text{Total } \Delta V_A = \Delta V_1 + \Delta V_{2A} \quad (E-95a)$$

$$\text{Total } \Delta V_B = \Delta V_1 + \Delta V_{2B} \quad (E-95b)$$

$$\text{Total } \Delta V_C = \Delta V_1 + \Delta V_{2C} \quad (E-95c)$$

$$\text{Total } \Delta V_D = \Delta V_1 + \Delta V_{2D} \quad (E-95d)$$

Mars L₂ ΔV₂ calculations

Hyperbolic Arrival to Mars L_2 (in Vector Form)

$$\Delta V_2 L_2 = V_{Halo} - V_{arr} \quad (E-96)$$

where:

$$V_{Halo} = \dot{x}X + \dot{y}Y + \dot{z}Z + (dL_2/R_{MARS} * e_{Mars} * \sin(E/R_{star}) * V_{star}X + dL_2 * \theta \dot{Y}$$

$$V_{arr} = (V_{arr} * \cos \gamma) X + (V_{arr} * \sin \gamma) Y$$

A Halo Data file for Mars L_2 is obtained from Danehy used to determine the minimum ΔV_2 using velocity components selected from the four relative maximum and minimum Y halo positions (labeled as A, B, C, D) from the Sun - Mars position along one Mars heliocentric orbit.

$$V_{star} = \sqrt{G M_{Star} / L_{star}} ,$$

where

G (the Gravitational Constant) = $6.67 \times 10^{-11} \text{ m}^3 / (\text{kg sec}^2)$,

M_{Star} (the mass of the Sun - Mars system) = $1.989100642 \times 10^{30} \text{ kg}$,

L_{star} (Sun - Mars Mean distance) = 227940974 km.

Thus $V_{star} = 24.130393219 \text{ km/s}$

$$\dot{x}_A = -0.0014462396 * V_{star} \quad (E-97a)$$

$$\dot{y}_A = 0.0000228267 * V_{star} \quad (E-97b)$$

$$\dot{z}_A = 0.0013787231 * V_{star} \quad (E-97c)$$

$$\dot{x}_B = -0.0000060014 * V_{star} \quad (E-98a)$$

$$\dot{y}_B = 0.0047781274 * V_{star} \quad (E-98b)$$

$$\dot{z}_B = 0.0000255570 * V_{star} \quad (E-98c)$$

$$\dot{x}_C = 0.0014438404 * V_{star} \quad (E-99a)$$

$$\dot{y}_C = 0.0000136483 * V_{star} \quad (E-99b)$$

$$\dot{z}_C = -0.0013781477 * V_{star} \quad (E-99c)$$

$$\dot{x}_D = -0.0000366945 * V_{star} \quad (E-100a)$$

$$\dot{y}_D = -0.0043002611 * V_{star} \quad (E-100b)$$

$$\dot{z}_D = 0.0000224213 * V_{star} \quad (E-100c)$$

See Equations (E-85) through (E-95d) for similar derivations for the L_2 case.

Appendix F: Sample Upper Stage Mass Budget Spreadsheets for Small and Large Spacecraft Classes. The spreadsheet is used towards subsystem mass margin comparison towards the sizing term of the Cost Model.

Appendix F-1: Sample Spreadsheet Calculations for the Small Spacecraft Class.

Applicable column and row headings included for the occasional Excel cell calculations (i.e. E19- B18) provided for spreadsheet calculation cross-reference.

	B	C	D	E	F	G	H	I	J
2	Isp (s/c)	Isp		Rocket Equation					
3	225	292		R2 =exp(ΔV2+ΔVstationkeepng/g*Isp)					
4				<u>Minitial</u> = <u>Mpayload</u>					
5	ΔV ₁	Mars Approach		M final M final					
6	(km/s)	Velocity							
7	3.6								
8									
9	R ₁ (Δv ₁)	EXP((ΔV ₁)/ g *1000/Isp)							
10	3.5140002	EXP((B7)/9.81*1000/C3)							
11									
12	From MRS paper: Medium Mars Orbit								
13	ΔV ₂	ΔVStn Keepng							
14	1.76	0							
15	R2(ΔV2+ΔVStnKp) =			EXP((ΔV2+Vstnkp)/ g *1000/ Isp)					
16	2.2197005			EXP((B14+C14)/9.81*1000/225)					
17	Mfinal(advertised)			Mpayload=M initial =R2 * Mfinal					
18	45			98advertised					
19				99.8865 A15 * A18					
20				Mfuel s/c=Mpayload-Mfinal					
21				56advert 54.8865 E19 - B18					
21	fpla(given)	Mpio (given)	Mfe	fs	fpi				
22	0.0408	0.3	5	0.02	0.047				

Appendix F-1: Sample Upper Stage Mass Budget Spreadsheets , Small Spacecraft Class (cont.)

	B	C	D	E	F	G	H	I	J
23	Mprop								
24	Upper Stage	$[M_{pay} + f_{pi} M_{pay} + M_{pio} + M_{fe}] (R1 - 1) \cdot (1 - f_s) M_{margin} / [1 + f_{pi} - R1 (f_{pi} + f_s)]$							
25	348advert								
26	338.4638941	$((\$E\$19 + \$B23 * \$E\$19 + \$C23 + \$D23) * (\$B\$10 - 1) + ((1 - \$E23) * \$G\$23)) / (1 + \$F23 - (\$B\$10 * (\$F23 + \$E23)))$							
27	Mupper								
28	stage(dry)	$M_{fe} + M_{pio} + (f_{pi} * M_{prop}) + (f_s * M_{wet})$							
29	28.9advertised								
30	30.66971306	$\$D23 + \$C23 + (\$F23 * B28) + (\$E23 * B37)$							
31	Total								
32	Mass								
33	(Mwet)	$M_{prop} + M_{pay} + M_{pia} + M_{pio} + M_{fe} + (f_{pi} * M_{prop}) + f_s / (1 - R) * (M_{margin} - R * M_{prop})$							
34	479advert.								
35	473.0955018	$B28 - \$E\$19 - (\$B23 * \$E\$19) + \$C23 + \$D23 + (\$F23 * B28) + \$E23 / (1 - \$B\$10) * (\$G\$23 - (B28 * \$B\$10))$							

Appendix F-2: Sample Upper Stage Mass Calculations, Large Spacecraft Class.

	B	C	D	E	F
2	Rocket Equation				
3	$R_2 = e^{(\Delta V_2 + \Delta V_{stationkeeping}/g \text{ Isp})}$				
4	Minitial/Mfinal = Mpayload/Mfinal				
5					
6	Isp				
7	292				
9	ΔV_1				
10	(km/s)				
11	2.43				
13	$R_{1(\Delta V_1)}$	EXP((ΔV_1)/ g *1000/Isp)			
14	2.3356954	EXP((B11)/9.81*1000/B7)			
16	For Aerosynchronous Scenario :				
17	ΔV_2	ΔV_{StaKp}			
18	1.5	0.1			
19	$R_2(\Delta V_2 + \Delta V_{StaKp})$		EXP(($\Delta V_2 + \Delta V_{stnkp}$)/ g*1000/ Isp)		
20	1.7481495		EXP((B18 +C18)/9.8 (1000/B7)		
22	Mfinal(advertised)		Mpayload = M initial = R2 * Mfinal		
23	674.2		1062advertised		
24			1178.602 B10 * B13		
25			Mfuel s/c = Mpayload - Mfinal		
26			387.7 advertised 504.4024 E13 - B13		

**Appendix F-2: Sample Upper Stage Mass Calculations, Large Spacecraft Class
(cont.).**

	B	C	D	E	F	
27	fpla	Mpio	Mfe	fs	fpi	
29	0.01	20	20	0.022	0.015	
31	Mprop					
32	Upper Stage					
33	(PAM-D)					
34	1877.5advert.					
36	1767.9335					
38	Mupper					
39	stage(dry)					
40	132advertised					
42	134.56181					
44	Total					
45	Mass					
46	(Mwet)					
47	3271.4advert.					
50	3092.8547					

Appendix G: FORTRAN Source Code Used in Determining the ΔV Budget.

```

program Mars Observer
c    4/6/96
c
c **** This program computes the launch energy ****
c    **** for Earth to Mars departure ****
c
c    implicit double precision (a-h,o-z)
c    double precision mu, mue, n, masc
c
c    DIMENSION RY(4),RZ(4),v(4)

c***** Open the data file and output file. *****

      OPEN (UNIT=1,FILE='in.txt', STATUS = 'OLD')
      OPEN (UNIT=2,FILE='CPI.out',STATUS='NEW')

      WRITE(2,12)'Earth  Mars  C3  DELTA DELTA TOTAL  De
+hta  TOTAL Delta  TOTAL Delta  TOTAL Delta  TOTAL Transfe
+r'
12 FORMAT (1X, A)
      WRITE(2,13)'Depart  Arr          V1  V2AERO DV2A  V2Ph
+obo DVPhobo V2Deimo DVDeimo Hyp_V  Vcir  V2Dra74 DVDra74 Angle'
13 FORMAT (1X, A)
      WRITE(2,14)'-----
+-----'
14 FORMAT (1X, A)

      DO WHILE (.TRUE.)

      READ (1,15,END = 99) EarDep,rearth,ty,epoc1,EMarAr,rmars,tz,epoc2
15 FORMAT(A8,1X,F6.4,1X,F6.2,1X,F9.1,A10,1X,F6.4,1X,F6.2,1X,F9.1)

c ***** Calculates Time of Flight *****
      tof = epoc2 - epoc1

      pi = 4.d0 * datan(1.d0)
      rpd = pi / 180.d0
      dpr = 180.d0 / pi

C    write(*,*)'Earth/Mars Angles',ty,tz
C    write(*,*)'Magnitudes',rearth,rmars

c *** input Earth heliocentric vector ***

```



```

      RY(4) = rearth * 1.496d8
      RY(1) = RY(4) * dcos(ty*rpdl)
      RY(2) = RY(4) * dsin(ty*rpdl)
      RY(3) = 0.d0
c *** input Mars heliocentric vector ***
      RZ(4) = rmars * 1.496d8
      RZ(1) = RZ(4) * dcos(tz*rpdl)
      RZ(2) = RZ(4) * dsin(tz*rpdl)
      RZ(3) = 0.d0
C   write(*,*)'R Earth,R Mars',RY(4),RZ(4)
c
c *** 1 A.U. ***
      au = 1.496d8
c *** mu of the Sun ***
      mu = 1.3271243993548d11
c *** mu of the Earth ***
      mue = 3.986d5
c *** mu of Mars ***
      muMars = 4.2828d4
      jj = 1
c
c *** calls LAMBERT subroutine to solve for ***
c   the orbital parameters p and a, and computes
c   the velocity vectors and delta V
c
      CALL LAMBER (RY, RZ, TOF, MU, JJ, TA, V, A, P, ECC)
c
c *** masc is mean anomaly of spacecraft at Earth launch ***
      masc = -7.48957220*rpdl
      n = dsqrt( mu/(a**3.d0) )
      t = masc/n
c
c *** calculating eccentricity ***
      ecc = dsqrt(1.d0-(p/a))
c
c
      time = tof * 24.d0 * 3600.d0 + t
      pi = 4.d0 * datan(1.d0)
c
c *** calls KEPLER subroutine to find the eccentric, E ***
      call kepler(time,a,ecc,mu,ea,th)
      au = 1.496d8
      rsmo = a*(1.-ecc*dcos(ea))
      vmo = dsqrt(2.d0*mu/rsmo - mu/a)
      tae = dsqrt(mu/au**3)*tof*86400.d0
      alpha = tae - ta
c
      remo = dsqrt(au**2 + rsmo**2 - 2.*au*rsmo*dcos(alpha))

```

```
mars= 2.279441736d8
```

```

c
c      ***** Investigating Transfer Angle *****
c      Calculate theta star at spacecraft launch and
c      uses algorithm to eliminate cosine ambiguity
c
  thetay1 = dacos( (1-ecc**2)*a/(ecc*ry(4)) - (1/ecc) )
  thetay2 = -thetay1
c
c
c      ( Calculate Theta Star at spacecraft arrival )
c
  thetaz1 = dacos( (1-ecc**2)*a/(ecc*rz(4)) - (1/ecc) )
  thetaz2 = -thetaz1
c
c
c      ( Calculate Theta Star at proper quadrants )
c
  trangle1 = thetaz2 - thetay2
  if(trangle1.lt.0 .and. thetaz2.le.thetay2) then
    thetaz2 = 2.d0*pi + thetaz2
    trangle1 = thetaz2 - thetay2
  endif
c
  trangle2 = thetaz2 - thetay1
  if(trangle2.lt.0 .and. thetaz2.le.thetay1) then
    thetaz2 = 2.d0*pi + thetaz2
    trangle2 = thetaz2 - thetay1
  endif
c
  trangle3 = thetaz1 - thetay2
  if(trangle3.lt.0 .and. thetaz1.le.thetay2) then
    thetaz1 = 2.d0*pi + thetaz1
    trangle3 = thetaz1 - thetay2
  endif
c
  trangle4 = thetaz1 - thetay1
  if(trangle4.lt.0 .and. thetaz1.le.thetay1) then
    thetaz1 = 2.d0*pi + thetaz1
    trangle4 = thetaz1 - thetay1
  endif
c
c
c
  compare1 = dabs(trangle1 - ta)
  compare2 = dabs(trangle2 - ta)
  compare3 = dabs(trangle3 - ta)
  compare4 = dabs(trangle4 - ta)

```

```

c
c
c
  if(compare1.le.1d-8) then
    thetas1 = thetay2*dpr
    thetas2 = thetaz2*dpr
  endif
c
  if(compare2.le.1d-8) then
    thetas1 = thetay1*dpr
    thetas2 = thetaz2*dpr
  endif
c
  if(compare3.le.1d-8) then
    thetas1 = thetay2*dpr
    thetas2 = thetaz1*dpr
  endif

c
  if(compare4.le.1d-8) then
    thetas1 = thetay1*dpr
    thetas2 = thetaz1*dpr
  endif
c
  theta = thetas2 - thetas1
c  write(*,*)'Theta * @ launch/arrival:',thetas1,thetas2
c
C (EARTH HELIOCENTRIC VELOCITY)
  Vearth=dsqrt(mu/RY(4))
C  write(*,*)'S/C Vel=',V(4),' V Earth=',Vearth
  CosGamDp= dsqrt(mu*a*(1.d0-ecc**2))/(V(4)*RY(4))
C
c *** Launch Energy defined as (V infinity)^2 ***
  C3=(vearth**2.d0)+(v(4)**2.d0)-(2.d0*vearth*v(4)*cosgamdp)
C  write(*,*)'C3=',C3,' V inf + :',dsqrt(C3)

c *** DeltaV1, Hyperbolic departure from Earth at 170km altitude perigee ***
  Rperigee= 6378.d0 + 170.d0
c  write(*,*)'mue'.mue OK
  mue= 3.986d5
  muMars= 4.2828d4
  mu= 1.3271243993548d11
  Rp = 3.950d3
  HypA = -mue/C3
  HypV = dsqrt(mue*(2.d0/Rperigee - 1.d0/HypA))
  Vcir= dsqrt(mue/Rperigee)
  DeltaV1 = HypV - Vcir
C  write(*,*)'Delta V1=',DeltaV1,' HypV=',HypV

```

```

C *****
C Mars Aerosynchronous, Phobos and Deimos ΔV2 calculations
C *****

c *** Aerosynchronous Altitude ***
  Rp = 20.431d3
c *** Phobos Altitude ***
  Rpphob = 9.380d3
c *** Deimos Altitude ***
  RpDeim = 23.449d3

c *** Hyperbolic arrival to Mars ***
  Varr = dsqrt( 2.d0*mu* (1.d0/RZ(4) - 1.d0/(2.d0*a)) )
  CosGamma= dsqrt(mu*a*(1.d0-ecc**2.d0)/(Varr*RZ(4)))
c
c *** Hyperbolic arrival to Aerosync Mars Orbit-> Delta V2***
  vMars = dsqrt( (2.d0 * mu / RZ(4)) - (mu/a) )
  Vsquar = ((vMars)**2.d0+(Varr)**2.d0-(2.d0*vMars*Varr*CosGamma))
  Vp = dsqrt( (2.d0 * muMars / Rp) + (Vsquar) )
  AHypAr = ( (2.d0 / Rp) - (Vp**2.d0/muMars) ) ** (-1.d0)
  Vcir = dsqrt( muMars/Rp )
  ADraim74 = 42560.d0
  VDraim74 = dsqrt( (2.d0 * muMars / Rp ) - (muMars/ADraim74) )
  DelV2A = Vp - Vcir
  DelV2Dr74 = Vp - VDraim74
  TotDelVA = DeltaV1 + DelV2A
  TotDelVD74 = DeltaV1 + DelV2Dr74
c
c *** Hyperbolic arrival to Phobos Orbit-> Delta V2***
  Vpphob = dsqrt( (2.d0 * muMars / Rpphob) + (Vsquar) )
  AHypArp = ( (2.d0 / Rpphob) - (Vpphob**2.d0/muMars) ) ** (-1.d0)
  Vcirp = dsqrt( muMars/Rpphob )
  DelV2Cp = Vpphob - Vcirp
  TotDelVCp = DeltaV1 + DelV2Cp
c
c *** Hyperbolic arrival to Deimos Orbit-> Delta V2***
  VpD = dsqrt( (2.d0 * muMars / RpDeim) + (Vsquar) )
  AHypArD = ( (2.d0 / RpDeim) - (VpD**2.d0/muMars) ) ** (-1.d0)
  VcirD = dsqrt( muMars/RpDeim )
  DelV2CD = VpD - VcirD
  TotDelVCD = DeltaV1 + DelV2CD

  WRITE (2,11) eardep,Emarar,C3,DELTA V1,DEL V2A,TotdelvA,DEL V2Cp,
+TOTDELVCp,Delv2CD,TotDelvCD,DelV2Dr24,TotDelVD24,DelV2Dr74,
+TotDelVD74,theta
11 FORMAT (1X,a8,1X,a8,1x,F7.4,1x,F7.4,1X,F7.4,1X,F7.4,1X,
+F7.4,1x,F7.4,1X,F7.4,1x,F7.4,1x,F7.4,F7.4,1x,1x,F7.4,1X,F5.1 )
C *****

```

C End of Mars Aerosynchronous, Phobos and Deimos ΔV_2 calculations
 C *****

C *****

C Start of Mars L_1 ΔV_2 calculations

C *****

c ***** Hyperbolic Arrival to Mars L_1 (in Vector Form) *****

c $\Delta V_2 = V_{Halo} - V_{arr}$

c where:

$$V_{Halo} = xdotX + ydotY + zdotZ + (dL1/RMARS * eMars * \sin(E/Rstar) * VstarX + dL1 * ThetaDotY$$

$$V_{arr} = (V_{arr} * \cos\gamma) X + (V_{arr} * \sin\gamma) Y$$

c ***** Halo Data file *****

VStar = 24.130393219

xdotA = -0.0014349278 * VStar

ydotA = -0.0000290458 * VStar

zdotA = 0.0012607976 * VStar

xdotB = -0.0000177086 * VStar

ydotB = 0.0043018427 * VStar

zdotB = 0.0000136673 * VStar

xdotC = 0.0014324387 * VStar

ydotC = -0.000030760 * VStar

zdotC = -0.0012600003 * VStar

xdotD = -0.0000135431 * VStar

ydotD = -0.0047515270 * VStar

zdotD = -0.0000100992 * VStar

Varr = dsqrt(2.d0*mu* (1.d0/RZ(4) - 1.d0/(2.d0*a)))

CosGamma = Dsqrt(mu*a*(1.d0-ecc**2.d0))/(Varr*RZ(4))

SinGamma = Dsin(DAcos(CosGamma))

VarrX = Varr * SinGamma

VarrY = Varr * CosGamma

CosGamma1 = CosGamma * dpr

c

vL1x=RZ(4)*0.093368/(rmars*1.496d8)*dsin(EA/rmars)*VStar

write(*,*)EA,vL1x,RZ4,rmars,EA,vL1x,RZ(4),rmars

vL1y = dsqrt(mu*1.523691*1.496d8*(1.d0-(.093368**2.d0))) /
 + ((RMARS*1.496d8)**2.d0) *RZ(4)

c

DeltaV2A =Dsqrt((vL1x+xdotA-VarrX)**2.d0+(ydotA+vL1y-VarrY)**2.d0

```

+ +zdotA**2.d0 )
c
  DeltaV2B=Dsqrt((vL1x+xdotB-VarrX)**2.d0+(ydotB+vL1y-VarrY)**2.d0
+ +zdotB**2.d0 )
  DeltaV2C=Dsqrt((vL1x+xdotC-VarrX)**2.d0+(ydotC+vL1y-VarrY)**2.d0
+ +(zdotC**2.d0) )
  DeltaV2D=Dsqrt((vL1x+xdotD-VarrX)**2.d0+(ydotD+vL1y-VarrY)**2.d0
+ +(zdotD**2.d0) )

  vL1 = dsqrt ( vL1x ** 2.d0 + vL1y ** 2.d0 )
  DeltaV2 = dsqrt((vL1)**2.d0+(Varr)**2.d0-(2.d0*vL1*Varr
+ *CosGamma))
  TotDelV = DelV1+DeltaV2
c  write(*,*)deltav2,totdelv
  TotDelVA= DelV1+DeltaV2A
  TotDelVB= DelV1+DeltaV2B
  TotDelVC= DelV1+DeltaV2C
  TotDelVD= DelV1+DeltaV2D
c

  WRITE (2,11) eardep,EarMarAr,cosgamdpl,cosgamma1,Varr,vL1y,C3,
+ DELV1,DELTA V2A,DELTA V2B,DELTA V2C,DELTA v2D,TotdelvA,TotdelvB,
+ TotdelVC,TotdelVD,theta
11 FORMAT (1X,a8,1x,a8,1x,F7.4,1x,f7.4,1x,f9.4,1x,f7.4,1x,f7.3,1x,
+ f7.3,1x,F7.4,1x,F7.4,1x,F7.4,1x,F7.4,1x,F7.4,1x,F6.4,1x,
+ 1x,F6.4,1x,F5.1)
C *****
C End of Mars L1 ΔV2 calculations
C *****

C *****
C Start of Mars L2 ΔV2 calculations
C *****
c
c ***** Hyperbolic Arrival to Mars L1 ( in Vector Form ) *****
c   ΔV2 = VHalo - Varr
c where:
c   VHalo=xdotX+ydotY+zdotZ+(dL1/RMARS*eMars*sin(E/Rstar)*VstarX+dL1*ThetaDotY

c   Varr = ( Varr * Cosγ ) X + ( Varr * Sinγ ) Y
c
c
c   Varr = ( Varr * CosGammaX + Varr * SinGammaY )
c
c ***** Halo Data file *****
VStar = 24.130393219

```

```

x_dotA = -0.0014462396 * VStar
y_dotA = 0.0000228267 * VStar
z_dotA = 0.0013787231 * VStar

x_dotB = -0.0000060014 * VStar
y_dotB = 0.0047781274 * VStar
z_dotB = 0.0000255570 * VStar

x_dotC = 0.0014438404 * VStar
y_dotC = 0.0000136483 * VStar
z_dotC = -0.0013781477 * VStar

x_dotD = -0.0000366945 * VStar
y_dotD = -0.0043002611 * VStar
z_dotD = 0.0000224213 * VStar

Varr = dsqrt( 2.d0*mu*(1.d0/RZ(4) - 1.d0/(2.d0*a)) )
CosGamma = Dsqrt(mu*a*(1.d0-ecc**2.d0)/(Varr*RZ(4))
SinGamma = Dsin(DAcos(CosGamma))
VarrX = Varr * SinGamma
VarrY = Varr * CosGamma
c  write(*,*)'Varr X,Y',VarrX,VarrY
CosGamma1 = CosGamma * dpr
c
vL2x=RZ(4)*0.093368/(rmars*1.496d8)*dsin(EA/rmars)*VStar
write(*,*)'EA,vL2x,RZ4,rmars',EA,vL2x,RZ(4),rmars
vL2y = dsqrt( mu*1.523691*1.496d8*(1.d0-(.093368**2.d0)) ) /
+ ((RMARS*1.496d8)**2.d0) *RZ(4)
c
DeltaV2A=Dsqrt((vL2x+x_dotA-VarrX)**2.d0+(y_dotA+vL2y-VarrY)**2.d0
+ +z_dotA**2.d0 )
c
DeltaV2B=Dsqrt((vL2x+x_dotB-VarrX)**2.d0+(y_dotB+vL2y-VarrY)**2.d0
+ +z_dotB**2.d0 )
DeltaV2C=Dsqrt((vL2x+x_dotC-VarrX)**2.d0+(y_dotC+vL2y-VarrY)**2.d0
+ +(z_dotC**2.d0) )
DeltaV2D=Dsqrt((vL2x+x_dotD-VarrX)**2.d0+(y_dotD+vL2y-VarrY)**2.d0
+ +(z_dotD**2.d0) )

vL2 = dsqrt ( vL2x ** 2.d0 + vL2y ** 2.d0 )

DeltaV2 = dsqrt((vL2)**2.d0+(Varr)**2.d0-(2.d0*vL2*Varr*CosGamma))
TotDelV = DelV1+DeltaV2
c  write(*,*)deltav2,totdelv
TotDelVA= DelV1+DeltaV2A
TotDelVB= DelV1+DeltaV2B
TotDelVC= DelV1+DeltaV2C
TotDelVD= DelV1+DeltaV2D

```

```

WRITE (2,11) eardep,Emarar,cosgamdp1,cosgamma1,Varr,vL2,C3,
+ DELV1,DELTA V2A,DELTA V2B,DELTA V2C,DELTA v2D,TotdelvA,TotdelvB,
+ TotdelVC,TotdelVD,theta
11 FORMAT (1X,a8,1x,a8,1xF7.4,1xf7.4,1xf9.4,1xf7.4,1xf7.3,1x,
+ f7.3,1X,F7.4,1x,F7.4,1xF7.4,1x,F7.4,1x,F7.4,1x,F7.4,1x,F6.4,1x,
+ 1x,F6.4,1x,F5.1)
C *****
C End of Mars L2 ΔV2 calculations
C *****

END DO
C **** End of File record was read ****
99 CONTINUE
C
C ***** CLOSE FILES *****
CLOSE (UNIT = 1)
END FILE (UNIT = 2)
CLOSE (UNIT = 2)

stop
end

C *****Subroutine Lambert*****
C
SUBROUTINE LAMBER (RY,RZ,TOF,MU,JJ,TA,V,A,P,ECC)
C
IMPLICIT DOUBLE PRECISION (A-I,L-Z)
C
DIMENSION RY(4), RZ(4), V(4)
C
DOUBLE PRECISION REAL TA, C, S, PI, TOF, MU, A,
+ PHI, B, U, KU, P, N,
+ LAMBDA, L, M, X, Y, H1, H2, W,
+ XNEW, YNEW, DXNEW, DXOLD, F, G,
+ TA1, TA2, TA3, TA4, DTA, CROSS
INTEGER J, K, JJ
C
C (GIVEN RY, RZ, TOF AND INITIAL GUESSES FOR X AND Y)
C
C Find TA (RAD) -
C Use DOT Product to obtain between 0 and 180 degrees,
C then check to see if between 180 and 360 degrees
C using Z-component of cross-product of Ry and Rz
C
pi = 4.d0 * datan(1.d0)
TA = DACOS((RY(1)*RZ(1)+RY(2)*RZ(2)+RY(3)*RZ(3))/
+ (RY(4)*RZ(4)))
CRZ = RY(1)*RZ(2) - RY(2)*RZ(1)

```



```

      IF (CRZ.LT.0.) TA = 2.d0*pi-ta
      TA1 = TA*180.d0/pi

c   WRITE(*,*)'TRANSFER ANGLE, DEG',TA1
C
C (FIND CHORD)
      C = DSQRT(RY(4)**2 + RZ(4)**2 - 2.0D0*RY(4)*RZ(4)*
      + DCOS(TA))
C
C (FIND SEMI-PERIMETER)
      S = (RY(4)+RZ(4)+C)/2.0D0
C
C (FIND LAMBDA)
      LAMBDA = (DSQRT(RY(4)*RZ(4)) * DCOS(TA/2.0D0))/S
C
C (FIND W)
      W = DATAN((RZ(4)/RY(4))**0.25D0)-(PI/4.0D0)
C
C (FIND L)
      L=((DTAN(2.0D0*W))**2.d0+(DSIN(TA/4.0D0))**2.d0)/
      + ((DTAN(2.0D0*W))**2.d0 + (DSIN(TA/4.0D0))**2.d0 +
      + DCOS(TA/2.0D0))
C
C (FIND M)
      M = (8.0D0*MU*((TOF*86400)**2))/((S**3)*((1.0D0+LAMBDA)**6))
c
C (BEGIN ITERATION TO CONVERGE UPON CORRECT X AND Y)
C USE JJ INPUT TO DEFINE INITIAL CONDITION:
C JJ=0 MEANS HYPERBOLA OR PARABOLA (X=0),
C JJ=1 MEANS ELLIPSE (X=L)
C
      DXOLD = 10000.00
      IF (JJ.EQ.0) THEN
        X = 0.1D-15
      ELSE
        IF (JJ.EQ.1) THEN
          X = L
        ELSE
          WRITE(*,*)'JJ NOT DEFINED CORRECTLY'
          END IF
        END IF
      END IF
C
      DO 50 J=1,100
C (Find PHI Variable)
      N = X/(((DSQRT(1.0D0+X))+1.0D0)**2)
c
      aa = 9.d0/7.d0
      bb = 16.d0/63.d0
      cc = 25.d0/99.d0

```

```

dd = 36.d0/143.d0
ee = 49.0D0/195.0D0
ff = 64.0D0/255.0D0
gg = 81.d0/323.d0
hh = 100.0D0/399.0D0
aaa = 8.0D0*(DSQRT(1.0D0+X)+1.0D0)
bbb = gg*N/(1.0D0+(hh*N))
ccc = ff*N / (1.0D0+bbb)
ddd = ee*N/(1.0D0+ccc)
c
c ( Finally solve for PHI variable )
PHI = aaa/(3.0D0+(1.0D0/(5.0D0+N+(aa*N)/(1.0D0+((
+ bb*N)/(1.0D0+((cc*N)/(1.0D0+((dd*N)/(1.0D0+ddd )))))))))
c
C (FIND H1)
H1 = ((L+X)**2*(1.0D0+3.0D0*X+PHI))/((1.0D0+2.0D0*X+L)
+ *(4.0D0*X+PHI*(3.0D0+X)))
c
C (FIND H2)
H2 = (M*(X-L+PHI))/((1.0D0+2.0D0*X+L)*(4.0D0*X+PHI*
+ (3.0D0+X)))
c
C (FIND B)
B = (27.0D0*H2)/(4.0D0*((1.0D0+H1)**3))
c
C (FIND U)
U = -B/(2.0D0*(DSQRT(1.0D0+B)+1.0D0))
c
C (FIND KU)
eee = 1.0D0-(700.0D0/2907D0)*U
fff = 598.0D0/2295.0D00*U/eee
ggg = 1.0D0-(((418.0D0/ 1755.0D0)*U)/(1.0D0-fff))
c
KU = (1.0D0/3.0D0)/(1.0D0-(((4.0D0/27.0D0)*U)/
+ (1.0D0-(((8.0D0/27.0D0)*U)/(1.0D0-(((2.0D0/
+ 9.0D0)*U)/(1.0D0-(((22.0D0/81.0D0)*U)/(1.0D0-
+ (((208.0D0/891.0D0)*U)/ggg) ))))))))
C
C (SOLVE FOR NEW Y)
YNEW = ((1.0D0+H1)/3.0D0)*(2.0D0+(DSQRT(1.0D0+B)
+ /(1.0D0-(2.0D0*U*(KU**2)))))
C
C (SOLVE FOR NEW X)
XNEW=DSQRT(((1.0D0-L)/2.0D0)**2+(M/YNEW**2))
+ -((1.0D0+L)/2.0D0)
C
C (COMPARE XNEW WITH X TO ENSURE CONVERSION)
DXNEW = DABS(XNEW - X)
IF (DXNEW.LE.0.1D-15) THEN

```

```

      GO TO 50
C
      ELSE
      X = XNEW
      Y = YNEW
      ENDIF
50 CONTINUE
C
C *** THE FOLLOWING SOLVES FOR ORBIT PARAMETERS a, p, and ecc ***
C (Semi-major axis of interplanetary trajectory, a)
  a = M*S*(1.0D0+LAMBDA)**2/(8.0D0*X*Y**2)
  p = (2.0D0*RY(4)*RZ(4)*Y**2*(1.0D0+X)**2*(DSIN
+ (TA/2.0D0)**2))/(M*S*(1.0D0+LAMBDA)**2)
  ecc = dsqrt(1.d0-(p/a))
C
C (Solve for functions F and G, THE LAGRANGIAN COEFFICIENTS)
  F = 1.0D0-(RZ(4)/P)*(1.0D0-DCOS(TA))
  G = RY(4)*RZ(4)*DSIN(TA)/DSQRT(MU*P)
C
C (FIND VELOCITY AT LAUNCH)
  DO 65 K=1,3
    V(K)=(1.0D0/G)*(RZ(K)-(F*RY(K)))
  65 CONTINUE
C
  V(4)=DSQRT(V(1)**2.d0+V(2)**2.d0+V(3)**2.d0)
C
  RETURN
  END
C
C *****END OF LAMBERT SUBROUTINE *****
C
C ***** Subroutine Kepler *****
C
  subroutine kepler(t,a,ecc,mu,e,theta)
    implicit double precision (a-h,o-z)
    double precision m,n,mu
C
    pi = 4.0d0*datan(1.0d0)
C
C INITIALIZE VARIABLES AND COUNTER
    tp = 0.0
    n = dsqrt(mu/a**3)
C    write(*,*)t:',t OK
    m = (t-tp)*n
    j = 0
    e = m
C
C BEGIN LOOP TO ITERATIVELY COMPUTE E (e) USING NEWTON'S METHOD
10  f = m - (e -ecc*dsin(e))

```

```
c
  if (abs(f) .lt. 1.d-15) go to 20
    fprime = ecc*dcos(e) - 1.0
    e = e - (f/fprime)
    j = j + 1
    go to 10
c
20  theta = 2.*datan(dsqrt((1.+ecc)/(1.-ecc))*dtan(e/2.))

return
end
```

Appendix H: Mars and Earth Positions Relative to the Solar System at Various

Times. The following is obtained using the Orbital Workbench software program. The leftmost latitude and longitude information refer to that of the Earth, the rightmost information refer to that of Mars. The Earth and Mars Positions Indicated in Bold-Type are the Respective Earth Launch and Mars Arrival positions for the Mars Observer Mission.

Date / Julian Launch Date	Sun-Earth distance	Sun-Mars distance
01 Jan 1992 / 2448622.5	D = 0.9832 A.U.	D = 1.4924 A.U.
Latitude = 0.0 ⁰	Latitude = -0.748 ⁰	
Longitude = 99.81 ⁰	Longitude = 253.31 ⁰	
01 Jul 1992 / 2448804.5	D = 1.0167 A.U.	D = 1.3954 A.U.
Latitude = 0.0 ⁰	Latitude = -1.323 ⁰	
Longitude = 279.41 ⁰	Longitude = 3.9751 ⁰	
01 Aug 1992 / 2448835.5	D = 1.0149 A.U.	D = 1.42 A.U.
Latitude = 0.0 ⁰	Latitude = -0.831 ⁰	
Longitude = 308.99 ⁰	Longitude = 22.953 ⁰	
01 Sep 1992 / 2448866.5	D = 1.0091 A.U.	D = 1.4535 A.U.
Latitude = 0.0 ⁰	Latitude = -0.271 ⁰	
Longitude = 338.8 ⁰	Longitude = 41.159 ⁰	
25 Sep 1992 / 2448890.5	D = 1.0028 A.U.	D = 1.4833 A.U.
Latitude = 0.0⁰	Latitude = 0.1652 ⁰	
Longitude = 2.1745⁰	Longitude = 54.642 ⁰	
01 Oct 1992 / 2448896.5	D = 1.0011 A.U.	D = 1.491 A.U.
Latitude = 0.0 ⁰	Latitude = 0.271 ⁰	
Longitude = 8.0668 ⁰	Longitude = 57.927 ⁰	
24 Aug 1993 / 2449223.5	D = 1.011 A.U.	D = 1.5983 A.U.
Latitude = 0.0 ⁰	Latitude = 0.6248⁰	
Longitude = 330.84 ⁰	Longitude = 209.82⁰	

Appendix I: Spreadsheet for the Small Relay Spacecraft Baseline Design

Appendix I –1. Baseline Relay Design (Excluding Power Design) Applicable to All Scenarios

Baseline Design (all scenarios)					
Payload					
	Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)
Antennas					
X-band - Earth (~1 m)	1	2		2	
UHF - Mars	1	1		1	
Transponder	2	2		4	6.5
Receiver	2				
Transmitter	2				
Filters, Diplexers	2	1		2	
Amplifiers				0	
Solid state 5 W	2	1	40	2	40
Payload Totals	8			11	46.5
SubSystem					
	% s/c dry mass	% of payload mass	Subsystem mass (kg)	Subsystem Pwr (W)	
Payload	36.9%	100.0%	11.00	46.50	
Structure	14.2%	38.5%	4.23		
Thermal	6.3%	16.9%	1.86	0.10	
Power	11.4%	30.8%	3.38	10.00	
TT&C	2.8%	7.7%	0.85	0.20	
ADCS	7.4%	20.0%	2.20	0.10	
Propulsion	21.0%	56.9%	6.26	0.10	
Batt Recharge Pwr Mgn				7.50	
	100.0%				
		Total s/c mass (kg)	29.78	Total s/c pwr (W)	57.00
		Mass w/ Margin (15%)	34.25	Pwr w/ Margin (15%)	65.55

Appendix I: Spreadsheet for the Small Relay Spacecraft Baseline Design (cont.)

Appendix I -2: Baseline Relay Power Design Among Scenarios

Halo Orbiter Baseline Power Design			
Batteries		Solar panels	
Eclipse Power Load (W)	23.00	Solar Panel Eff	18%
Max duration for recharge(sec)	7200	Mars Solar Constant (W/m**2)	586
Battery drain eff	90%	Reqd Solar Panel Area (m**2)	0.62
		Available Solar Panel Area (m**2)	1.00
Desired DOD	95%	Solar panel Mass (kg)	2.62
Reqd Battery Cap (W*Hr)	44		
Desired # Batteries	3	Power Control Unit mass (kg)	1.31
Batt Capacity (W*Hr)	15	Regulator mass (kg)	1.64
Battery Mass {Li Ion} (kg)	0.968	Regulator Pwr Loss (W)	10.90
Recharge time (Hr)	5.81	Wiring mass (kg)	1.37
		Wiring Pwr Loss (W)	2.73
		Power Capability	70.759
Aerosynchronous Baseline Power Design			
		Batt Recharge Pwr Mgn	5.00
		Total s/c pwr (W)	62.00
		Pwr w/ Margin (15%)	71.30
Batteries		Solar panels	
Eclipse Power Load (W)	23.00	Solar Panel Eff	18%
Max duration (sec)	5000	Mars Solar Constant (W/m**2)	586
Battery drain eff	90%	Reqd Solar Panel Area (m**2)	0.68
		Available Solar Panel Area (m**2)	1.00
Desired DOD	95%	Solar panel Mass (kg)	2.85
Reqd Battery Cap (W*Hr)	30		
Desired # Batteries	3	Power Control Unit mass (kg)	1.43
Batt Capacity (W*Hr)	10	Regulator mass (kg)	1.78
Battery Mass {Li Ion} (kg)	0.673	Regulator Pwr Loss (W)	10.40
Recharge time (Hr)	6.05	Wiring mass (kg)	1.42
		Wiring Pwr Loss (W)	2.60
		Power Capability	71.384

Appendix I-2: Spreadsheet for the Small Relay Spacecraft Baseline Design (cont.)

Low Orbit Baseline Power Design			
		Batt Recharge Pwr Mgn	30.00
		Total s/c pwr (W)	87.00
		Pwr w/ Margin (15%)	100.05
Batteries		Solar panels	
Eclipse Power Load (W)	23.00	Solar Panel Eff	18%
Max duration (sec)	2600	Mars Solar Constant (W/m**2)	586
Battery drain eff	90%	Reqd Solar Panel Area (m**2)	0.95
		Available Solar Panel Area (m**2)	1.00
Desired DOD	50% SMAD Table 10-26	Solar panel Mass (kg)	4.00
Reqd Battery Cap (W*Hr)	30		
Desired # Batteries	3	Power Control Unit mass (kg)	2.00
Batt Capacity (W*Hr)	10 ea batt	Regulator mass (kg)	2.50
Battery Mass {Li Ion} (kg)	0.664 total	Regulator Pwr Loss (W)	15.40
Recharge time (Hr)	1.00	Wiring mass (kg)	1.62
		Wiring Pwr Loss (W)	2.85
		Power Capability	66.139
Medium Orbit Baseline Power Design			
		Batt Recharge Pwr Mgn	30.00
		Total s/c pwr (W)	87.00
		Pwr w/ Margin (15%)	100.05
Batteries		Solar panels	
Eclipse Power Load (W)	23.00	Solar Panel Eff	18%
Max duration (sec)	2700	Mars Solar Constant (W/m**2)	586
Battery drain eff	90%	Reqd Solar Panel Area (m**2)	0.95
		Available Solar Panel Area (m**2)	1.00
Desired DOD	50% SMAD Table 10-26	Solar panel Mass (kg)	4.00
Reqd Battery Cap (W*Hr)	31		
Desired # Batteries	3	Power Control Unit mass (kg)	2.00
Batt Capacity (W*Hr)	10.350 ea batt	Regulator mass (kg)	2.50
Battery Mass {Li Ion} (kg)	0.690 total	Regulator Pwr Loss (W)	15.40
Recharge time (Hr)	1.04	Wiring mass (kg)	1.62
		Wiring Pwr Loss (W)	3.85
		Power Capability	65.134

Appendix I-2: Spreadsheet for the Small Relay Spacecraft Baseline Design (cont.)

Common Period Inclined Baseline Power Design			
		Batt Recharge Pwr Mgn	5.00
		Total s/c pwr (W)	62.00
		Pwr w/ Margin (15%)	71.30
Batteries		Solar panels	
Eclipse Power Load (W)	23.00	Solar Panel Eff	18%
Max duration (sec)	7200 2 Hr Max eclipse time	Mars Solar Constant (W/m**2)	586
Battery drain eff	90%	Reqd Solar Panel Area (m**2)	0.80
		Available Solar Panel Area (m**2)	1.00
Desired DOD	95% SMAD Table 10-26 ~ 1000 Cycles	Solar panel Mass (kg)	2.85
Reqd Battery Cap (W*Hr)	44		
Desired # Batteries	3	Power Control Unit mass (kg)	1.43
Batt Capacity (W*Hr)	15 ea batt	Regulator mass (kg)	1.78
Battery Mass {Li Ion} (kg)	0.968 total	Regulator Pwr Loss (W)	10.40
Recharge time (Hr)	8.72	Wiring mass (kg)	1.87
		Wiring Pwr Loss (W)	2.60
		Power Capability	71.384

Appendix J: Spreadsheet for the Large Relay Spacecraft Baseline Design among Scenarios.

Appendix J-1: Spreadsheet for Halo Orbiter. Specific scenario designs are each labeled in bold type, and column and row headings included for the occasional Excel cell calculations (i.e. F44-F41-F42) for use in cross-reference.

	A	B	C	D	E	F	G	H
8.	Halo Orbiter			Payload Specs (from SMAD pg 379, 543, 805)				
9.	Payload	Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)		
10.	Antennas							
11.	Parabolic - Earth (5m)	1	102.2		102.2			
12.	Parabolic - Mars 1.8m	1	13.25		13.25			
13.	Cross-link	1	9.2		9.2			
14.	Waveguide	2	1.4		2.8			
15.	Omnidirectional	2	0.25		0.5			
16.	Coax	2	0.5		1			
17.	Transponder	3	4.45		13.35			
18.	Receiver	3		5		15		
19.	Transmitter	3		10		30		
20.	Filters, Diplexers	3	1.2		3.6			
21.	Amplifiers							
22.	r-e 5 W	10	1.2	30	12	300		
23.	xl 2 W	10	1.2	20	12	200		
24.	Payload Totals	10			169.9	545		
32.	Spacecraft SubSystem	% s/c dry mass	% of payload mass	Subsystem mass (kg)		Subsystem Pwr (W)	% Payload Pwr	
35.	Payload	24.6%	100.0%	169.90	Payload	545.00	100%	
36.	Structure	28.0%	114.1%	193.90	Structure			SMAD pg 300
37.	Thermal	2.8%	11.3%	19.23	Thermal	38.15	7%	Subsystem power numbers from SMAD
38.	Power	26.4%	107.3%	187.65	Power	242.76	30%	
39.	TT&C	4.1%	16.6%	28.16	TT&C	54.50	10%	
40.	ADCS	6.2%	25.1%	42.68	ADCS	245.25	45%	
41.	Propulsion	6.9%	28.2%	47.87	Propulsion	38.15	7%	
42.					Batt Recharge Power Mgn (W)	50.00		
43.		98.9%						
44.			Total s/c mass (kg)	689.4	Total s/c pwr (W)	1213.81		
45.			Mass w/ Margin (15%)	792.8	Pwr w/ Margin (15%)	1395.88		
46.			2 Satellites	1585.6				
48.	Batteries							
50.	Operational Power Load (W)	1125.66	F44-F41-F42		Solar panels Solar Panel Eff 18%			
51.	Max duration (sec)	2000	0.56 Hr tumble recovery		Mars Solar Const.. (W/m ²)	586		
52.	Battery drain eff	90%			Reqd Solar Panel Area (m ²)	13.23		

Appendix J-1: Spreadsheet for Halo Orbiter (cont).

	A	B	C	D	E	F	G	H
53.	Desired DOD	95%			Solar panel Mass	55.84		
					(kg)			
54.	Reqd Batt Cap (W*Hr)	592	B50*B52*(B51/3600)/B53					
55.	Desired # Batteries	3			Power Control Unit mass (kg)	27.92	0.02*F45	SMAD pg 319
56.	Batt Capacity (W*Hr)	197	B54/B55		Regulator mass (kg)	34.90	0.025*F45	
57.	Battery Mass {NiH2} (kg)	13.17	B55*(B56/45)		Regulator Pwr Loss (W)	194.21	0.2*(F35+F36+F37+F39+F40+F41+F42)	
58.	Recharge time (Hr)	11.85	B54/F42		Wiring mass (kg)	55.84	0.02*F45	
59.					Wiring Pwr Loss (W)	48.55	0.05*(F35+F36+F37+F39+F40+F41+F42)	

Appendix J-2: Spreadsheet for the Aerosynchronous Orbiter, Large Spacecraft Class.

	A	B	C	D	E	F	G	H
66.	Aerosynchronous							
67.	Payload							
68.								
69.		Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)		
70.	Antennas							
71.	Parabolic	1	102.2		102.2			
72.	Horn	1	0.5		0.5			
73.	Waveguide	2	1.4		2.8			
74.	Hemispherical	2	0.4		0.8			
75.	Coax	2	0.5		1			
76.	Transponder	2	4.45		8.9			
77.	Receiver	2		5		10		
78.	Transmitter	2		10		20		
79.								
80.								
81.	Filters, Duplexers	3	1.2		3.6			
82.	Amplifiers							
83.	r-e 5 W	10	1.2	30	12	300		
84.	xl 2 W	10	1.2	20	12	200		
85.								
86.	Payload Totals	10			143.8	530		
87.								
88.								
89.	SubSystem	% s/c dry mass	% of payload mass	Subsystem mass (kg)		Subsystem Pwr (W)	% Payload Pwr	
90.								SMAD
91.	Payload	24.6%	100.0%	143.80	Payload	530.00	100%	pg 300
92.	Structure	28.0%	114.1%	164.12				Subsystem
93.	Thermal	2.8%	11.3%	16.28	Thermal	37.10	7%	power
94.	Power	26.4%	107.3%	200.61	Power	238.93	30%	numbers

Appendix J-2: Spreadsheet for the Aerosynchronous Orbiter, Large Spacecraft Class (cont.).

	A	B	C	D	E	F	G	H
95.	TT&C	4.1%	16.6%	23.83	TT&C	53.00	10%	from SMAD
96.	ADCS	6.2%	25.1%	36.13	ADCS	238.50	45%	
97.	Propulsion	6.9%	28.2%	40.52	Propulsion	37.10	7%	
98.					Batt Recharge Pwr Mgn	60.00		
101			Total s/c mass (kg)	625.26	Total s/c pwr (W)	1194.63		
102			Mass w/ Margin (15%)	719.07	Pwr w/ Margin (15%)	1373.82		
103			2 Satellites	1438.14				
104	Batteries				Solar panels			
105	Eclipse Power Load (W)	1097.53			Solar Panel Eff	18%		
106	Max duration (sec)	4500	1.25 hrs		Mars Solar Constant (W/m**2)	586		
107	Battery drain eff	90%			Reqd Solar Panel Area (m**2)	13.02		
108	Desired DOD	95% SMAD	< 1000 Cycles		Solar panel Mass (kg)	54.95		
		Table 10-26						
109	Reqd Battery Cap (W*Hr)	1300						
110	Desired # Batteries	3			Power Control Unit mass (kg)	27.48		
111	Batt Capacity (W*Hr)	433			Regulator mass (kg)	34.35		
112	Battery Mass {NiH2} (kg)	29			Regulator Pwr Loss (W)	191.14		
113	Recharge time (Hr)	21.66	(23.37 Hrs sunlite) Avail		Wiring mass (kg)	54.95		
114	Relay Orbital Period (Hr)	24.624			Wiring Pwr Loss (W)	47.79		

Appendix J-3: Spreadsheet for the Large Relay Spacecraft Baseline Design, Phobos Lander.

	A	B	C	D	E	F	G	H
117	Phobos (Baseline from Aerosynchronous Design)				Batt Recharge Pwr Mgn	180		
118			Subsystem - power	424.67	Total s/c pwr (W)	1314.63		
119			power mass(kg)	217.86	Pwr w/ Margin (15%)	1511.82		
120			Total S/C Mass	642.53				
121	Batteries		Mass+15%Margin	738.91	Solar panels			
122			2 Satellites	1447.81				
123	Eclipse Power Load (W)	1217.53			Solar Panel Eff	18%		
124	Max duration (sec)	2050			Mars Solar Constant (W/m**2)	586		
125	Battery drain eff	90%			Reqd Solar Panel Area (m**2)	14.33		
126	Desired DOD	50% SMAD Table 10-26		~ 10000 Cycles	Solar panel Mass (kg)	60.47		
127	Reqd Battery Cap (W*Hr)	1248						
128	Desired # Batteries	3			Power Control Unit mass (kg)	30.24		
129	Batt Capacity (W*Hr)	416			Regulator mass (kg)	37.80		
130	Battery Mass {NiH2} (kg)	28			Regulator Pwr Loss (W)	191.14		
131	Recharge time (Hr)	6.93	(7.09 Hrs Avail sunlite)		Wiring mass (kg)	60.47		
132	Relay Orbital Period (hr)	7.66			Wiring Pwr Loss (W)	47.79		

Appendix J-4: Spreadsheet for the Large Relay Spacecraft Baseline Design, Deimos Lander.

	A	B	C	D	E	F	G	H
133	Deimos(Baseline from Aerosynchronous Design)				Batt Recharge Pwr Mgn	56		
134			Subsystem - power	424.67	Total s/c pwr (W)	1190.63		
135			power mass(kg)	200.22	Pwr w/ Margin (15%)	1369.22		
136			Total S/C Mass	624.89				
137	Batteries		Mass+15%Margin	718.62	Solar panels			
138			2 Satellites	1437.24				
139	Eclipse Power Load (W)	1093.53			Solar Panel Eff	18%		
140	Max duration (sec)	5533			Mars Solar Constant (W/m**2)	586		
141	Battery drain eff	90%			Reqd Solar Panel Area (m**2)	12.98 F135/(F139*F140)		

Appendix J-4: Spreadsheet for the Large Relay Spacecraft Baseline Design, Deimos Lander (cont.).

	A	B	C	D	E	F	G	H
142	Desired DOD	95% SMAD	< 1000 Cycles	Solar panel Mass (kg)	54.77	0.04*F13		
		Table 10-26				5		
143	Reqd Battery Cap (W*Hr)	1592 B139*B141*(B140/3600)/B142						
144	Desired # Batteries	3		Power Control Unit mass (kg)	27.38	0.02*F13		
						5		
145	Batt Capacity (W*Hr)	531 B143/B144		Regulator mass (kg)	34.23	.025*F13		
						5		
146	Battery Mass {NiH2} (kg)	35 B144*(B145/45)		Regulator Pwr Loss (W)	191.14	0.2*(F92+F93+F94+F96+F97+F98+F99)		
147	Recharge time (Hr)	28.43 (28.75 Hrs Avail sunlite)		Wiring mass (kg)	54.95	0.04*F135		
148	Relay Orbital Period (Hr)	30.28 (2 * PI() * ((23449)^3/42828)^0.5)/3600		Wiring Pwr Loss (W)	47.79	0.05*(F92+F93+F94+F96+F97+F98+F99)		

Appendix J-5: Spreadsheet for the Large Relay Spacecraft Baseline Design, Low Mars Orbit Orbiter.

	A	B	C	D	E	F	G	H
150.	Low Orbit							
151.								
152.	Payload							
153.								
154.		Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)		
155.	Antennas							
156.	Parabolic	1	102.2		102.2			
157.	Horn	1	0.5		0.5			
158.	Waveguide	2	1.4		2.8			
159.	Hemispherical	2	0.4		0.8			
160.	Coax	2	0.5		1			
161.	Transponder	2	4.45		8.9			
162.	Receiver	2		5		10		
163.	Transmitter	2		10		20		
164.								
165.								
166.	Filters, Diplexers	3	1.2	0	3.6	0		
167.	Amplifiers							
168.	r-e 5 W	10	1.2	30	12	300		
169.	xl 2 W	10	1.2	20	12	200		
170.								
171.	Payload Totals	10			143.8	530		
172.								
173.								
174.	SubSystem	% s/c dry mass	% of payload mass	Subsystem mass (kg)	Subsystem Pwr (W)	% Payload Pwr		
175.								
176.	Payload	24.6%	100.0%	143.80	Payload 530.00	100%	SMAD	pg 300

Appendix J-5: Spreadsheet for the Large Relay Spacecraft Baseline Design, Low Mars Orbit Orbiter (cont.).

	A	B	C	D	E	F	G	H
177.	Structure	28.0%	114.1%	164.12				Subsystem power numbers from SMAD
178.	Thermal	2.8%	11.3%	16.28	Thermal	37.10	7%	
179.	Power	26.4%	107.3%	379.51	Power	517.68	30%	
180.	TT&C	4.1%	16.6%	23.83	TT&C	53.00	10%	
181.	ADCS	6.2%	25.1%	36.13	ADCS	238.50	45%	
182.	Propulsion	6.9%	28.2%	40.52	Propulsion	37.10	7%	
183.					Batt Rchg Pwr Mgn	1050.00		
185.		Total s/c mass (kg)		804.17	Total s/c pwr (W)	2432.13		
186.		Mass w/ Margin (15%)		924.80	Pwr w/ Margin (15%)	2796.94		
187.		2 Satellites		1849.60				
188.	Batteries				Solar panels			
189.	Eclipse Power Load (W)	1376.28	F153-F151-F150		Solar Panel Eff	18%		
190.	Max duration (sec)	2600			Mars Solar Constant (W/m**2)	586		
191.	Battery drain eff	90%			Reqd Solar Panel Area (m**2)	28.52		
192.	Desired DOD	50%	SMAD Table 10-26	> 10000 Cycles	Solar panel Mass (kg)	111.88		
193.	Reqd Battery Cap (W*Hr)	1789						
194.	Desired # Batteries	3			Power Control Unit mass (kg)	55.94		
195.	Batt Capacity (W*Hr)	596 ca batt			Regulator mass (kg)	69.92		
196.	Battery Mass {NiH2} (kg)	30 total			Regulator Pwr Loss (W)	389.14		
197.	Recharge time (Hr)	1.28	(1.34 Hr Avail sunlite)		Wiring mass (kg)	111.88		
198.	Relay Orbital Period (Hr)	1.89			Wiring Pwr Loss (W)	97.29		

**Appendix J-6: Spreadsheet for the Large Relay Spacecraft Baseline Design,
Medium Mars Orbit orbiter.**

	A	B	C	D	E	F	G	H
203.	Medium Orbit							
204.								
205.	Payload							
206.								
207.		Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)		
208.	Antennas							
209.	Parabolic	1	102.2		102.2			
210.	Helix	1	0.5		0.5			
211.	Waveguide	2	1.4		2.8			
212.	Hemispherical	2	0.4		0.8			
	Coax	2	0.5		1			
213.	Transponder	2	4.45		8.9			
214.	Receiver	2		5		10		
215.	Transmitter	2		10		20		
216.								
217.								
218.	Filters, Diplexers	3	1.2		3.6			
219.	Amplifiers							
220.	r-e 5 W	10	1.2	30	12	300		
221.	xl 2 W	10	1.2	20	12	200		
222.								
223.	Payload Totals	10			143.8	530		
224.								
225.	SubSystem	% s/c dry mass	% of payload mass	Subsystem mass (kg)		Subsystem Pwr (W)	% Payload Pwr	
226.								
227.	Payload	24.6%	100.0%	143.80		530.00	100%	SMAD pg 300
228.	Structure	28.0%	114.1%	164.12				Subsystem power numbers from SMAD
229.	Thermal	2.8%	11.3%	16.28		37.10	7%	
230.	Power	26.4%	107.3%	310.04		392.68	30%	
231.	TT&C	4.1%	16.6%	23.83		53.00	10%	
232.	ADCS	6.2%	25.1%	36.13		238.50	45%	
233.	Propulsion	6.9%	28.2%	40.52		37.10	7%	
234.	Batt Recharge Pwr Mgn					675.00		
235.		Total s/c mass (kg)		734.71	Total s/c pwr (W)	1963.38		
236.		Mass w/ Margin (15%)		844.91	Pwr w/ Margin (15%)	2257.88		
237.		2 Satellites		1689.80				

**Appendix J-6: Spreadsheet for the Large Relay Spacecraft Baseline Design,
Medium Mars Orbit orbiter(cont.).**

	A	B	C	D	E	F	G	H
238.	Batteries				Solar panels			
239.	Eclipse Power Load (W)	1251.28	F236-F234-F233		Solar Panel Eff	18%		
240.	Max duration (sec)	2000	0.56 hr		Mars Solar Constant (W/m**2)	586		
241.	Battery drain eff	90%			Reqd Solar Panel Area (m**2)	21.41		
242.	Desired DOD	50% SMAD Table 10-26		> 10000 Cycles	Solar panel Mass (kg)	90.32		
243.	Reqd Battery Cap (W*Hr)	1251						
244.	Desired # Batteries	3			Pwr Cntrl Unit mass (kg)	45.16		
245.	Batt Capacity (W*Hr)	417 ea batt			Regulator mass (kg)	56.45		
246.	Battery Mass {NiH2} (kg)	28 total			Regulator Pwr Loss (W)	314.14		
247.	Recharge time (Hr)	1.85	(1.90 Hr Avail sunlite)		Wiring mass (kg)	90.32		
248.	Relay Orbital Period (Hr)	2.45			Wiring Pwr Loss (W)	78.54		

Appendix J-7: Spreadsheet for the Large Relay Spacecraft Baseline Design Common-Period Inclined orbiter.

	A	B	C	D	E	F	G	H
255.	Common Period Inclined							
256.								
257.	Payload							
258.								
259.		Quantity	Mass (kg)	Power (W)	Total (kg)	Total Power (W)		
260.	Antennas							
261.	Parabolic	1	102.2		102.2			
262.	Horn	1	0.5		0.5			
263.	Helix	1	0.5		0.5			
264.	Waveguide	2	1.4		2.8			
265.	Hemispherical	2	0.4		0.8			
266.	Coax	2	0.5		1			
267.	Transponder	2	4.45		8.9			
268.	Receiver	2		5		10		
269.	Transmitter	2		10		20		
270.								
271.	Filters, Diplexers	3	1.2		3.6			
272.	Amplifiers							
273.	r-e 5 W	10	1.2	30	12	300		
274.	xl 2 W	10	1.2	20	12	200		
275.	Payload Totals	10			144.3	530		
276.								
277.	SubSystem	% s/c dry mass	% of payload mass	Subsystem mass (kg)		Subsystem Pwr (W)	% Payload Pwr	
278.	Payload	24.6%	100.0%	144.30		530.00	100%	SMAD pg 300
279.	Structure	28.0%	114.1%	164.69				Subsystem power
280.	Thermal	2.8%	11.3%	16.33		37.10	7%	numbers
281.	Power	26.4%	107.3%	212.23		231.43	30%	from SMAD
282.	TT&C	4.1%	16.6%	23.91		53.00	10%	
283.	ADCS	6.2%	25.1%	36.25		238.50	45%	
284.	Propulsion	6.9%	28.2%	40.66		37.10	7%	
285.	Batt Recharge Pwr Mgn					30.00		
286.								
287.		Total s/c mass (kg)		638.38	Total s/c pwr (W)	1157.13		
288.		Mass w/ Margin (15%)		734.13	Pwr w/ Margin (15%)	1330.69		
289.		2 Satellites		1468.26				
290.	Batteries				Solar panels			
291.	Eclipse Power Load (W)	1090.03	F287-F285-F284		Solar Panel Eff	18%		
292.	Max duration (sec)	7200 2 Hr Max eclipse time			Mars Solar Constant (W/m**2)	586		
293.	Battery drain eff	90%				Reqd Solar Panel Area (m**2)	12.62	

**Appendix J-7: Spreadsheet for the Large Relay Spacecraft Baseline Design
Common-Period Inclined orbiter (cont.).**

	A	B	C	D	E	F	G	H
294.	Desired DOD	95% SMAD	~ 1000 Cycles	Solar panel Mass (kg)	53.23			
		Table 10-26						
295.	Reqd Battery Cap (W*Hr)	2065		Power Control Unit mass (kg)	26.61			
296.	Desired # Batteries	3		Regulator mass (kg)	33.27			
297.	Batt Capacity (W*Hr)	688 ea batt		Regulator Pwr Loss (W)	185.14			
298.	Battery Mass {NiH2} (kg)	46 total		Wiring mass (kg)	53.23			
299.	Recharge time (Hr)	68.84		Wiring Pwr Loss (W)	46.285			
300.	Relay Orbital Period (Hr)	74.05						

Appendix K: Cost Model Evaluation. Normalized values are derived from the total mission cost information in Tables 12 and 13 shown in Chapter 4, ΔV information in Table 15 shown in Chapter 5, communications design data from Tables 38 through 43 shown in Chapter 6, lifetime factors from Tables 46 through 58 shown in Chapter 7, and planning factors from Tables 59 through 62.

Appendix K-1a: Normalized Total Mission Cost (Small Spacecraft Class)

	Halo Orbit	Aerosynchronous	Common Period Inclined	Multiple Low Mars Orbit(6)	Dedicated (Med Mars Orbit)	Mars Moons
United States						
Atlas I	0.096	0.100	0.104	0.110	0.092	0.096
Delta II 6925	0.091	0.094	0.098	0.105	0.087	0.091
Delta II 7925	0.095	0.098	0.102	0.109	0.091	0.095
Athena	0.061	0.065	0.086	0.092	0.057	0.061
Pegasus						
Taurus	0.056	0.071	0.075	0.093	0.052	0.056
Japan						
MV	0.099	0.102	0.106	0.113	0.095	0.099
China						
CZ-2C	0.060	0.063	0.067	0.089	0.056	0.060
CZ3	0.070	0.073	0.077	0.084	0.066	0.070

Appendix K-1b. Normalized Total Mission Cost (Large Spacecraft Class)

Launch Vehicle	Halo Orbit	Aerosynchronous	Common Period Inclined	Multiple Low Mars Orbit(6)	Dedicated (Med Mars Orbit)	Mars Moons
United States						
Atlas IIA	0.228	0.293	0.332	0.648	0.194	0.215
Atlas IIAS	0.236	0.234	0.348	0.472	0.202	0.222
Delta II 6925	0.255	0.301	0.387	0.530	0.174	0.242
Delta II 7925	0.263	0.313	0.352	0.554	0.178	0.250
Titan IV - Centaur	0.357	0.356	0.395	0.444	0.323	0.344
France						
Ariane 4	0.267	0.266	0.410	0.566	0.233	0.254
Japan						
H-2	0.287	0.411	0.450	1.000	0.252	0.273
Russia						
Proton D1e	0.214	0.213	0.252	0.354	0.180	0.200
Zenit 3	0.224	0.223	0.324	0.437	0.190	0.211
China						
CZ-2E	0.193	0.223	0.262	0.343	0.159	0.179
CZ3	0.213	0.238	0.302	—	0.153	0.200

Appendix K-2: Evaluation of the Spacecraft Sizing Term. Normalized values were derived using data from Table 15 and apply to both spacecraft classes. The average normalized values were computed for the halo orbit and the Martian moon lander scenarios towards the Cost Model evaluation.

	Halo Orbit Orbiters (L ₁)	Halo Orbit Orbiters (L ₂)	Aero-synchronous Orbiters	Drain Orbiters (74 hr)	Multiple Low Mars Orbit Orbiters	Dedicated Comm. Orbiter (Med Mars Orbit)	Martian Moon Landers (Phobos)	Martian Moon Landers (Deimos)
Total ΔV [(km/s)]	0.935	1	0.859	0.812	0.933	0.909	0.877	0.858

Appendix K-3: Evaluation of the Communications Performance Driver.
Normalized values derived from Tables 38 through 41.

1.		Halo Orbiters	Aerosyn- chronous	Common Period	Multiple (6)	Dedicated	Mars Moon
2.	Data Rates						
3.	Downlink (to Mars)						
4.	Small Spacecraft Class						
5.	Day Data Rate(kbps)	0.1	0.7	3	500	350	0.6
6.	Night Data Rate (kbps)	0.15	0.9	6	950	650	0.8
7.							
8.	Large Spacecraft Class						
9.	Day Data Rate(kbps)	10000	13000	1500	8000	6000	8667
10.	Night Data Rate (kbps)	10000	25000	3000	11000	10000	16667
11.							
12.	Uplink (to Earth)						
13.	Small Spacecraft Class						
14.	Day Data Rate(kbps)	0.03	0.7	3	100	80	0.47
15.	Night Data Rate (kbps)	0.03	0.7	3	100	80	0.47
16.							
17.	Large Spacecraft Class						
18.	Day Data Rate(kbps)	2500	15000	1500	3000	1500	10000
19.	Night Data Rate (kbps)	2500	15000	1500	3000	1500	10000
20.							

Appendix K-3: Evaluation of the Communications Performance Driver (cont.).

Normalized values derived from Tables 38 through 41. A sample Excel calculation appears on Row 29 and applies to the halo orbiter calculation, small spacecraft class.

		Halo Orbiters	Aerosyn- chronous	Common Period Inclined	Multiple (6)	Dedicated	Mars Moon
21.	Percentage of Mars Relay Communication Network Visibility Greater than 30 Degree Elevation Compared among Scenarios.						
22.	Polar	18.4	0.01	29.1	11.7	10	0.01
23.	Temperate	42.3	39.8	66.3	2.5	2.5	30
24.	Equatorial	59.5	84.3	77.8	1.7	2	60
25.							
26.	Estimated Data Throughput over 74 hour period						
27.	Eclipse fraction	0.5	0.0496	0.5	0.5	0.5	0.03216243
28.	Downlink (Marsbound)						0.05538362
29.	Small Spacecraft Class	2*(B\$5+B\$6/2*B22/100*74*3600	3*(((1-C\$27)*C\$5)+C\$27*C\$6)*C22/100*74*3600	4*D\$5*D22/100*74*3600	6*(E\$5+E\$6)/2*E22/100*74*3600	(F\$5+F\$6)/2*F22/100*74*3600	2*(((1-G\$27)*G\$5)+G\$27*G\$6+(1-G\$28)*G\$5)+G\$28*G\$6)*G22/100*74*3600
30.	Polar	12254.4	56.73683219	930268.8	135584280	13320000	64.8688907
31.	Temperate	28171.8	225812.5921	2087510.4	28971000	3330000	194606.672
32.	Equatorial	39627	478291.4953	2487110.4	19700280	2664000	389213.344
33.							
34.	Large Spacecraft Class						
35.	Polar	980352000	1086529.93	465134400	1776621600	213120000	960835.628
36.	Temperate	2253744000	4324389126	1043755200	379620000	53280000	2882506883
37.	Equatorial	3170160000	9159447321	1243555200	258141600	42624000	5765013766
38.							
39.	Uplink (Earthbound)						
40.	Small Spacecraft Class						
41.	Polar	2941.056	55.944	930268.8	18701280	2131200	49.728
42.	Temperate	6761.232	222657.12	2087510.4	3996000	532800	149184
43.	Equatorial	9510.48	471607.92	2487110.4	2717280	426240	298368
44.							
45.	Large Spacecraft Class						
46.	Polar	245088000	399600	465134400	561038400	39960000	1065600
47.	Temperate	563436000	1590408000	1043755200	119880000	9990000	3196800000
48.	Equatorial	792540000	3368628000	1243555200	81518400	7992000	6393600000

**Appendix K-3: Evaluation of the Communications Performance Driver
(cont.).**

		Halo Orbiters	Aerosyn- chronous	Common Period Inclined	Multiple (6)	Dedicated	Mars Moon
49.	Normalized Data Throughput over 74 hour period						
50.	Downlink (Marsbound)						
51.	Small Spacecraft Class	C35/B35					
52.	Polar	0.00463	1	6.1E-05	4.1846E-07	4.2595E-06	0.87463855
53.	Temperate	0.00201	0.000251	2.72E-05	1.9584E-06	1.7038E-05	0.00029155
54.	Equatorial	0.00143	0.000119	2.28E-05	2.88E-06	2.1298E-05	0.00014577
55.	Average	0.00269	0.3334	3.6994E-05	1.7523E-06	1.4198E-05	0.29169195
56.							
57.	Large Spacecraft Class	\$C\$35/ B40					
58.	Polar	5.8E-08	5.22E-05	1.22E-07	3.1935E-08	2.6622E-07	5.9049E-05
59.	Temperate	2.5E-08	1.31E-08	5.44E-08	1.4946E-07	1.0649E-06	1.9683E-08
60.	Equatorial	1.8E-08	6.19E-09	4.56E-08	2.1979E-07	1.3311E-06	9.8416E-09
61.	Average	3.39E-08	1.74E-05	7.3988E-08	1.3373E-07	8.874E-07	1.9693E-05
62.							
63.	Uplink (Earthbound)						
64.	Small Spacecraft Class	G46/ B46					
65.	Polar	0.0169	0.889	5.35E-05	2.6591E-06	2.3333E-05	1
66.	Temperate	0.00735	0.000223	2.38E-05	1.2444E-05	9.3333E-05	0.00033333
67.	Equatorial	0.00523	0.00010	2.00E-05	1.8301E-05	0.00011667	0.00016667
68.	Average	0.0098	0.29640589	3.2424E-05	1.1135E-05	7.7778E-05	0.3335

Appendix K-3: Evaluation of the Communications Performance Driver (cont.)

		Halo Orbiters	Aerosyn- chronous	Common Period	Multiple (6)	Dedicated	Mars Moon
69.	Large Spacecraft Class	G46/B51					
70.	Polar	2.029E-07	0.000124	1.0691E-07	8.8636E-08	1.2444E-06	4.6667E-05
71.	Temperate	8.83E-08	3.127E-08	4.7643E-08	4.1481E-07	4.9778E-06	1.5556E-08
72.	Equatorial	6.27E-08	1.48E-08	3.9989E-08	6.1002E-07	6.2222E-06	7.7778E-09
73.	Average	1.17E-07	4.15E-05	6.4848E-08	3.7116E-07	4.1481E-06	1.5563E-05
74.	SUBTOTAL						
75.	Small Spacecraft Class	0.00980	0.611	4.522E-05	2.20568E-06	1.877E-05	0.6458
76.	Large Spacecraft Class	0.0000001	0.0000295	0.0000001	0.0000003	0.0000025	0.0000176
77.							
78.	Communication System Configuration						
79.	Small Spacecraft Class	0.35	0.45	0.4	0.55	0.7	0.6
80.	Large Spacecraft Class	0.3	0.4	0.35	0.5	0.7	0.55
81.							
82.	Communications Performance	(B60+B70 + (B82* B86))/3					
83.	Small Spacecraft Class	0.123200	0.557449	0.133363	0.183335	0.233346	0.630564
84.	Large Spacecraft Class	0.100000	0.133353	0.116667	0.166667	0.233335	0.183345

Appendix K-4: Network Lifetime Subfactors compared among the Scenarios. A
sample calculation for cell B82 is included below. Normalized values
derived from Tables 46 through 58.

1.		Halo Orbit Orbiter	Aerosyn- chronous Orbiters	Common Period Orbiters	Multiple (6)	Dedi- cated Comm Orbiter/ Lander	Phobos	Deimos
2.	Mission Life Cycle Risk Parameters							
3.								
4.	Spacecraft Fuel Requirement							
5.	Small Spacecraft Class (ΔV)	0.04	2.9	0.03	1.5	1.5	0.001	0.001
6.	normalized	0.0138	1	0.01034	0.5172	0.5172	0.000345	0.0003
7.	Large Spacecraft Class (ΔV)	0.04	2.9	0.03	1.5	1.5	0.001	0.001
8.	normalized	0.01379	1	0.01034	0.5172	0.5172	0.000345	0.0003
9.								
10.	Total Power Requirement							
11.	Small Spacecraft Class (watts)	71.3	71.3	71.3	100.05	100.05	75	71
12.	normalized	0.024	0.02395	0.02395	0.03361	0.03361	0.0252	0.0239
13.	Large Spacecraft Class (watts)	1395.9	1373.8	1330.7	2976.6	2811.3	1534.8	1362.3
14.	normalized	0.4689	0.4615	0.4471	1	0.9445	0.5156	0.45767
15.								
16.	Battery Recharge Margin							
17.	Small Spacecraft Class (watts)	7.5	7.5	7.5	7.5	7.5	7.5	7.5
18.	normalized	1	1	1	1	1	1	1
19.	Large Spacecraft Class (watts)	50	60	30	1050	675	180	56
20.	normalized	0.04762	0.05714	0.02857	1	0.6429	0.1714	0.0533
21.								
22.	Quantitative Battery Power Loss Risk Items (Both Satellite Classes)							
23.	Estimated Max. Eclipse Duration (secs)	0	4160	1200	1800	1800	2500	3500
24.	normalized	0	1	0.2885	0.4327	0.4327	0.6010	0.8413
25.	Estimated Lifetime Number of Power/Eclipse Cycles	0	1800	600	27500	24000	2200	1500
26.	normalized	0	0.0654	0.02182	1	0.8727	0.08	0.0545
27.	Battery Power Loss Impact (watts)	0	0.25	0.125	0.75	0.75	0.25	0.25
28.	normalized	0	0.3333	0.1667	1	1	0.3333	0.3333
29.								
30.	Risk from Orbit Altitude (Both Spacecraft Classes)							
31.	Average Orbit Altitude (km)	1000000	17040	45000	300	1000	5989	20058
32.	normalized	0.0003	0.0176	0.0066	1	0.3	0.05009	0.0149
33.								
34.	Thermal Power Requirement (both Spacecraft Classes.)							
35.	Estimated Lifetime Thermal Cycling	0	1800	600	27500	24000	2200	1500
36.	normalized	0	0.0654	0.0218	1	0.8727	0.08	0.0545
37.								
38.	Thermal Power (Watts)	0	0.1	0	0.1	0.1	0	0
39.	normalized	0	0.002695	0	0.0027	0.0027	0	0
40.	Large Spacecraft Class	38.2	37.1	37.1	37.1	37.1	37.1	37.1
41.	normalized	1	0.9712	0.9712	0.9712	0.9712	0.9712	0.9712

Appendix K-4: Network Lifetime Subfactors Compared among the Scenarios (cont.)

42.		Halo Orbit Orbiter	Aerosyn- chronous Orbiters	Common Period Orbiters	Multiple Low Mars Orbit(6)	Dedi- cated Comm Orbiter/ Lander	Phobos	Deimos
43.	Network Fault Tolerance Risk Factors Relating to Constellation Configuration (Both Spacecraft Classes)							
44.								
45.								
46.								
47.	Coverage Reduction from One Spacecraft Loss in Network	50%	33%	25%	17%	100%	50%	50%
48.	normalized	0.50	0.33	0.25	0.17	1.00	0.50	0.50
49.	Approximate ΔV Effort from Network Rephasing(km/s)	na	0.25	0.2	0.1	na	na	na
50.	normalized		1	0.8	0.4			
51.	Coverage Reduction from Network Rephasing	na	5%	10%	0%	na	na	na
52.	normalized		0.50	1.00	0.00			
53.	Fault Tolerance Penalty	0.6				0.9	0.7	0.7
54.	Spacecraft Reliability Evaluated by Spacecraft Subsystem							
55.								
56.								
57.								
58.	Propulsion Both Spacecraft Classes	0.95	0.95	0.95	0.95	0.95	0.95	0.95
59.	Structure Both Spacecraft Classes	0.99	0.99	0.99	0.99	0.99	0.99	0.99
60.	Thermal Both Spacecraft Classes	0.99	0.99	0.99	0.99	0.99	0.99	0.99
61.	ADCS Small Spacecraft Class	0.9	0.89	0.88	0.88	0.95	0.95	0.95
62.	Large Spacecraft Class	0.93	0.92	0.9	0.9	0.95	0.95	0.95
63.	Reliability of Comm Designs							
64.	Reliability Rating (small s/c class)	0.9	0.9	0.90	0.9	0.9	0.8	0.8
65.	Reliability Rating (large s/c class)	0.97	0.93	0.90	0.89	0.88	0.86	0.86
66.	Command & Data Handling							
67.	Both Spacecraft Classes	0.93	0.93	0.93	0.93	0.93	0.93	0.93
68.	EPS Reliability							
69.	Both Spacecraft Classes	0.95	0.89	0.93	0.80	0.81	0.88	0.9
70.	Graceful Degradation							
71.	Small Spacecraft Class	0.7	0.5	0.45	0.40	0.35	0.55	0.55
72.	Large Spacecraft Class	0.75	0.6	0.55	0.50	0.45	0.65	0.65
73.	Network Lifetime Subtotal							
74.	Small Spacecraft Class	0.2977	0.4574	0.3124	0.5505	0.4980	0.3203	0.3242
75.	$B82 = (B78 + (B74 * B71 * B69 * B65 * B62 * B60 * B58)) + (B48 + B50 + B52 + B53) / 4 + (B43 + B38) / 2 + B33 + (B29 + B27 + B25) / 3 + B20 + B14 + B8) / 9$							
76.	Large Spacecraft Class	0.3109	0.4709	0.3181	0.7235	0.6229	0.3526	0.3373

Appendix K-5: Evaluation of Mission Preparation and Planning Timeline Driver.
 Normalized values are from Table 59.

	Halo Orbiters	Aerosynchronous	Common Period Inclined	Multiple (6)	Dedicated	Mars Lunar Landers
Network Size Affecting Network Establishment	0.2	0.3	0.4	0.6	0.2	0.2
Spacecraft Size Affecting Time						
Small	0.25	0.25	0.3	0.4	0.2	0.25
Large	0.45	0.5	0.55	0.6	0.4	0.45
Launch Vehicle Lead Time						
Small	0.2	0.2	0.2	0.25	0.2	0.2
Large	0.25	0.2	0.2	0.5	0.2	0.25
Likelihood of Network Established at Mars in 3 years						
Small	0.25	0.3	0.4	0.5	0.2	0.3
Large	0.3	0.36	0.47	0.75	0.22	0.35
Planning Timeline Factors in Completing Communications Relay Constellation						
Small Spacecraft Class	0.225	0.2625	0.325	0.4375	0.2	0.2375
Large Spacecraft Class	0.3	0.34	0.405	0.6125	0.255	0.3125

Appendix K-6: Cost Model Evaluation. Rows 2 through 15 show the parametric values of each individual design driver obtained or derived from Appendices K-1 through K-5. See text for explanation and the next page for the continuation of Cost Model values.

	A	B	C	D	E	F	G
1.		Halo Orbiters	Aerosynchronous	Common Period Inclined	Multiple (6)	Dedicated	Mars lunar landers
2.	Mission Cost						
3.	Small	0.056	0.063	0.067	0.064	0.062	0.056
4.	Large	0.193	0.213	0.252	0.343	0.163	0.179
5.	ΔV Sizing						
6.	both classes	(0.967)	0.859	0.812	0.933	0.909	0.877
7.	Communications Performance						
8.	Small	0.123	0.557	0.133	0.183	0.233	0.631
9.	Large	0.100	0.133	0.117	0.167	0.233	0.183
10.	Network Lifetime						
11.	Small	0.263	0.457	0.323	0.573	0.531	(0.311)
12.	Large	0.255	0.449	0.307	0.723	0.640	(0.312)
13.	Planning Timeline						
14.	Small	0.225	0.2625	0.325	0.4375	0.200	0.2375
15.	Large	0.3	0.34	0.405	0.6125	0.255	0.3125

Appendix K-6: Cost Model Evaluation (cont.). Rows 16 through 30, 31 through 60 show the Cost Model values with one and two selective terms dominating the Cost Model respectively. A sample calculation for cell B17 is included below. See text for explanation and the next page for the continuation of Cost Model values.

	A	B	C	D	E	F	G
16.	Mission Cost						
	B17=(B3+ 1*B6+ 1*B8+ 1*B11+ 1*B14)/5						
17.	Small	0.042	0.055	0.046	0.059	0.048	0.052
18.	Large	0.071	0.078	0.083	0.117	0.071	0.070
19.	ΔV Sizing						
20.	Small	0.207	0.199	0.179	0.212	0.202	0.200
21.	Large	0.210	0.194	0.184	0.224	0.207	0.195
22.	Communications Performance						
23.	Small	0.055	0.144	0.057	0.077	0.081	0.156
24.	Large	0.054	0.064	0.059	0.086	0.086	0.070
25.	Network Lifetime						
26.	Small	0.078	0.126	0.091	0.147	0.134	0.098
27.	Large	0.082	0.121	0.093	0.186	0.159	0.083
28.	Planning Timeline						
29.	Small	0.073	0.091	0.092	0.123	0.075	0.085
30.	Large	0.090	0.103	0.112	0.163	0.090	0.094
31.	Cost & Sizing						
32.	Small	0.217	0.210	0.191	0.227	0.211	0.210
33.	Large	0.245	0.233	0.229	0.285	0.235	0.227
34.	Sizing & Comm Performance						
35.	Small	0.229	0.299	0.203	0.245	0.244	0.314
36.	Large	0.228	0.218	0.205	0.254	0.249	0.228
37.	Cost & Comm Performance						
38.	Small	0.065	0.156	0.069	0.092	0.090	0.166
39.	Large	0.089	0.102	0.104	0.147	0.113	0.103
40.	Cost & Network Lifetime						
41.	Small	0.088	0.138	0.103	0.162	0.143	0.108
42.	Large	0.117	0.159	0.138	0.247	0.186	0.126
43.	Cost & Planning Timeline						
44.	Small	0.083	0.103	0.104	0.138	0.084	0.095
45.	Large	0.125	0.139	0.156	0.227	0.117	0.126
46.	Sizing & Network Lifetime						
47.	Small	0.252	0.281	0.238	0.315	0.298	0.256
48.	Large	0.063	0.103	0.077	0.167	0.141	0.076
49.	Sizing & Planning Timeline						
50.	Small	0.247	0.246	0.238	0.291	0.238	0.243
51.	Large	0.071	0.084	0.095	0.147	0.072	0.076
52.	Comm Performance & Network Timeline						
53.	Small	0.100	0.227	0.115	0.180	0.176	0.212
54.	Large	0.081	0.127	0.098	0.197	0.183	0.109
55.	Comm Performance & Planning Timeline						
56.	Small	0.095	0.192	0.116	0.156	0.117	0.198
57.	Large	0.089	0.108	0.116	0.177	0.114	0.109
58.	Network Timeline & Planning Timeline						
59.	Small	0.119	0.174	0.150	0.226	0.170	0.141
60.	Large	0.117	0.165	0.150	0.277	0.187	0.132

Appendix K-6: Cost Model Evaluation (cont.) Rows 61 - 90, 91 - 102 show the Cost Model values with three and four selective terms dominating the Cost Model respectively. A sample calculation for cell B62 is included below. See text for explanation.

	A	B	C	D	E	F	G
	Parametric Values	Halo Orbiters	Aerosynchronous	Common Period	Multiple (6)	Dedicated	Mars lunar landers
61.	Cost / ΔV / Comm Performance $B62=(B3+B6+B8+0.1*B11+0.1*B14)/5$						
62.	Small	0.239	0.310	0.216	0.260	0.253	0.324
63.	Large	0.263	0.257	0.250	0.315	0.277	0.260
64.	Cost/ ΔV / Network Lifetime						
65.	Small	0.262	0.292	0.250	0.330	0.307	0.266
66.	Large	0.291	0.314	0.285	0.415	0.350	0.283
67.	Cost/ ΔV / Planning Lifetime						
68.	Small	0.257	0.257	0.250	0.306	0.247	0.253
69.	Large	0.106	0.122	0.140	0.209	0.099	0.108
70.	ΔV / Comm Perf / Network Lifetime						
71.	Small	0.274	0.381	0.262	0.348	0.340	0.370
72.	Large	0.081	0.127	0.098	0.197	0.183	0.109
73.	ΔV / Comm Perf / Planning Lifetime						
74.	Small	0.269	0.346	0.262	0.324	0.280	0.356
75.	Large	0.089	0.108	0.116	0.177	0.114	0.109
76.	Comm Perf / Network Lifetime / Planning Lifetime						
77.	Small	0.141	0.274	0.174	0.259	0.212	0.254
78.	Large	0.135	0.189	0.171	0.307	0.229	0.165
79.	ΔV / Network Lifetime / Planning Lifetime						
80.	Small	0.293	0.328	0.296	0.394	0.334	0.299
81.	Large	0.117	0.165	0.150	0.277	0.187	0.132
82.	Cost / Comm Perf / Network Lifetime						
83.	Small	0.110	0.238	0.127	0.195	0.185	0.222
84.	Large	0.116	0.166	0.143	0.259	0.210	0.141
85.	Cost / Comm Perf / Planning Lifetime						
86.	Small	0.105	0.203	0.128	0.171	0.126	0.208
87.	Large	0.124	0.146	0.161	0.239	0.141	0.141
88.	Cost/ Network Lifetime / Planning Lifetime						
89.	Small	0.129	0.185	0.162	0.241	0.179	0.151
90.	Large	0.152	0.203	0.195	0.339	0.214	0.164
91.	Cost / ΔV / Comm Performance / Network Lifetime						
92.	Small	0.284	0.393	0.274	0.363	0.349	0.380
93.	Large	0.309	0.338	0.306	0.445	0.392	0.316
94.	Cost / ΔV / Comm Performance / Planning Timeline						
95.	Small	0.279	0.358	0.274	0.339	0.289	0.366
96.	Large	0.317	0.318	0.323	0.425	0.323	0.317
97.	ΔV / Comm Performance / Network Lifetime / Planning Timeline						
98.	Small	0.315	0.429	0.320	0.427	0.376	0.412
99.	Large	0.135	0.189	0.171	0.307	0.229	0.165
100.	Cost / ΔV / Network Lifetime / Planning Timeline						
101.	Small	0.303	0.340	0.308	0.409	0.343	0.309
102.	Large	0.152	0.203	0.195	0.339	0.214	0.164

Appendix K-6: Cost Model Evaluation (cont.). Rows 104 and 105 show the Cost Model values with equal weighting among scenarios. Rows 107 and 108 reveal the evaluation of the Cost Model based on weightings of current realistic conditions. A sample calculation for cell B107 is included below. See text for explanation.

	A	B	C	D	E	F	G
	Parametric Values	Halo Orbiters	Aerosynchronous	Common Period	Multiple (6)	Dedicated	Mars lunar landers
103.	Straight						
104.	Small	0.325	0.440	0.332	0.442	0.385	0.422
105.	Large	0.363	0.399	0.375	0.556	0.438	0.373
106.	Current Realistic Conditions B107=(B67*B3+B68*B6+B69*B8+B70*B11+B71*B14)/5						
107.	Small	0.183	0.248	0.183	0.238	0.210	0.244
108.	Large	0.208	0.222	0.215	0.304	0.239	0.213

Appendix K-7: Plot of Cost Model Totals among Scenarios using Current Conditions for Parametric Weights. The left pair of points for each scenario represents the small spacecraft class, and the right point represents the large spacecraft class.

IMAGE EVALUATION TEST TARGET (QA-3)

APPLIED IMAGE, Inc.
1653 East Main Street
Rochester, NY 14609 USA
Phone: 716/482-0300
Fax: 716/288-5989

© 1993, Applied Image, Inc., All Rights Reserved