Mars Xpress
Summary of the Group Design Project
MSc in Astronautics and Space Engineering
1997/98
Cranfield University

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Abstract

This is a summary of the Group Design Project of the MSc course in Astronautics and Space Engineering in the College of Aeronautics at Cranfield University for the academic year 1997/98. Executive summaries from all the individual reports are contained in an Appendix to this report.

The project represents about 8000 hours' effort by the students of the course directed by staff, and takes the form of a preliminary mission feasibility study. The project was based on ESA's Mars Express mission.

The proposed mission is for a Martian lander composed principally of a rover equipped to search for signs of past or present life on Mars. A controlled descent is required to ensure landing close to sites of particular interest. The surface exploration is planned to last 250 Martian days. A Mars orbiter (also part of the Mars Express mission) is used as a relay for the rover to communicate with Earth.

The mission appears feasible as far as the study goes. Topics requiring further study were identified and include thermal design, communications with Earth, achieving the required landing precision, and mobility on the Martian surface.

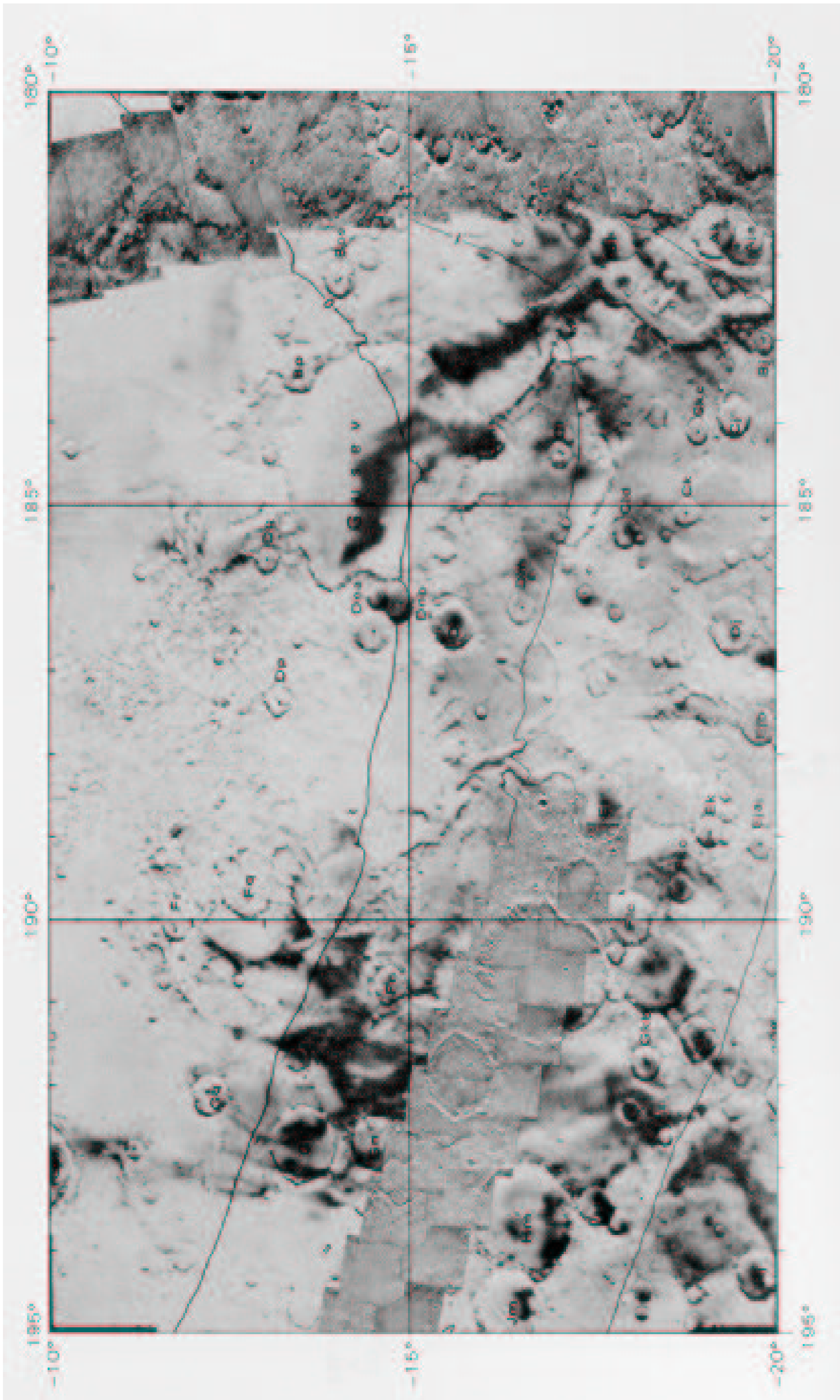

Figure 1: The proposed Mars Xpress landing site, Gusev Crater (186° W, 14.5° S) [17].

Acknowledgements

Many others have helped with this project. Particular thanks go to Cedric Seynat, Susan Jason and Tom Bowling, and the many contacts in the space industry worldwide who responded to questions and requests for information from the students of the project team. The project directors would like to thank the team for being such an enthusiastic and stimulating group to work with.

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Chapter 1

Introduction

This report summarises the results of the group design project undertaken by students of the MSc in Astronautics and Space Engineering at Cranfield University for the academic year October 1997 to September 1998. The project concerned a feasibility study of a robotic exploration mission to Mars, and was based loosely on ESA's Mars Express. The report gives a summary of the mission and contains all the executive summaries written by the students in an appendix (Appendix C). The full reports [1] - [16] are available in the College of Aeronautics at Cranfield University.

1.1 Organisation of the Project

The project runs over the first two terms (October to Easter) of the year long MSc course in Astronautics and Space Engineering at Cranfield University. The students work as one team, usually broken into several subgroups, and each contribute about 500 hours' effort to the project; the total resource represented by the project is thus approximately 8000 hours' work.

Appendix A contains diagrams showing the work packages identified and the corresponding subgroups and their membership. The project was directed by Mr. R.F. Turner (Chief Engineer, Space, Rutherford Appleton Laboratory) and Dr. S.E. Hobbs (Course Director). Several research students helped run the weekly progress meetings and provided support to the MSc students during the project.

1.2 Why study Mars?

There are two main reasons for studying Mars: for education and for scientific interest. The educational purpose is that the group project is used to train students in team working on a challenging technical subject relevant to their future careers. Basing the group project on current or planned ESA missions (Mars Express in this case) satisfies these requirements and generally ensures that good background material is available. An additional requirement is that the project should be one the students can relate to and work on with enthusiasm; Mars is an excellent subject for this purpose.

The scientific interest of Mars is unquestioned. Mars is a close neighbour in the Solar System and, above all, may once have been a home to life. Pons [15] summarises the key points of the scientific rationale for studying Mars: "The development of life is likely to occur in the presence of water and atmosphere. According to our current knowledge of Mars, it appears to be the most susceptible of all the (other) planets in our Solar System of having supported life. With its 95% carbon dioxide atmosphere giving a nominal surface pressure of 0.7% of that of Earth, the presence of water will fulfill the parameters required for organism developments. Thus there is great interest in its scientific exploration, which may give clues to the origin of life, reveal the differences between the planets of the Solar System, and allow their common origin to be understood."

1.3 Previous and Planned Missions to Mars

Pletinckx [13] quotes a useful summary of past and planned missions to Mars.

"Most of our current knowledge of Mars is the result of investigations conducted by a fleet of spacecraft beginning with the Mariners in the mid-1960's. The Mariner 4, 6 and 7 fly-by missions returned photographs and weather data from the southern hemisphere of Mars that put to rest hopes of finding a civilisation, and that gave the impression that Mars, like the Moon, has long been geologically inactive. The data from the 1971 Mariner 9 orbital mission created quite a different picture. Looking at the entire planet, Mariner 9 revealed huge volcanic mountains in the northern Tharsis region, so large that they deformed the planet's sphericity. One of these, Olympus Mons, at more than 26 km high, remains the largest volcano observed in our Solar System. Mariner 9 also revealed the awesome Vallis Marineris, a gigantic equatorial rift valley deeper and wider than the Grand Canyon and longer than the distance from New York to Los Angeles.

Although Mariner 9 photographs showed none of the fabled irrigation canals, the mission did disclose evidence of surface erosion and dried riverbeds, indicating that the planet was once capable of sustaining liquid water. This fuelled the possibility that life may be (or have been) possible on Mars. To investigate, two Viking spacecraft were dispatched to Mars in 1975. Each consisted of an orbiter and a lander. The orbiters surveyed the planet while the landers monitored surface weather conditions, took pictures, and tested the soil for signs of life. Viking 1's photographs revealed reddish desert-like drifts of dust. Some 5000 km away, Viking 2 observed a slightly more rolling duneless landscape, where patches of frost covered the ground in the Martian winter. From the weather stations, we quickly learned that these regions of Mars are too cold, and the atmosphere too thin, for liquid water to exist. The experiments designed to test for life showed some intriguing chemistry, but no signs of life.

In 1996, Mars Pathfinder and Mars Global Surveyor launched the next wave of Mars exploration. The Pathfinder approach demonstrates new, lightweight, low-cost lander, rover and imaging technologies while characterising Martian soils and rocks in the vicinity of the landing site. Mars Global Surveyor inaugurates an ambitious programme of orbital science to recapture the science lost with the Mars Observer spacecraft. Martian weather, seasonal change, surface features, and composition will be studied in detail over Mars Global Surveyor's two-year mapping phase, providing our first comprehensive, high-resolution look at the near-surface and surface phenomena on Mars. These missions set the stage for the Mars Surveyor series, which will send similarly lightweight orbiters and landers to Mars every two years into the first decade of the next century. Orbiters will provide synoptic coverage of areas and phenomena of interest, while acting as data relay stations for landers. Landers will probe the soils and test the rocks in search of clues regarding the origins and evolution of the Red Planet, and will look for tell-tale signs of life forms, past and present. We envision the Mars Surveyor programme as the linchpin for NASA participation in all future international Mars exploration programmes."

1.4 Starting Point of the Project

A brief project outline was developed over the summer of 1997 and issued to the students at the start of the academic year in October 1997. The project outline gave an introduction to previous exploration of Mars and identified some of the key areas of research interest.

1.5 Structure of this Report

Following this introduction there is an overview of the technical work carried out by the team. The overview considers each technical area of the project in turn and concludes with a synthesis based on the system studies. The final chapter (Conclusions) states the projects main findings and outlines areas requiring further study.

Appendices contain all the executive summaries written by the students and summarise the mission and the project organisation.

Chapter 2

Technical Work

This chapter provides an overview of the technical work carried out in the project. Further details are given in Appendix C (containing the Executive Summaries of all the reports) and in the reports themselves [1] - [16] (available for reference at the College of Aeronautics). The reports have been examined and any major errors that were identified have been corrected. However, it is not possible to guarantee that no errors remain; users of the summaries and reports should bear this in mind.

Table B.1 provides a summary of the whole mission.

2.1 Science

(Refer to the reports by Butler [1], Frew [5], and Hannington [7].)

Several different classes of scientific objective (such as planetary geology, exobiology) were considered, and then a primary mission aim identified. Factors considered included the geographical distribution of sites, complementarity with other missions and identified scientific requirements of various research communities (geology, meteorology, biology). The eventual choice was a primary aim of exobiology, with geology and meteorology as secondary objectives.

A target landing site was identified (Crater Lake, Fig. 1) which had good scientific potential and which was practical for landing (altitude not too high, surrounding terrain relatively level, suitable latitude).

Detailed specifications were developed for the scientific payload, and mission timelines have been proposed.

2.2 Mission Overview

(Refer to the reports by Fereday [4] and Pabon [12].)

Once the scientific objectives had been defined the next task was to develop a baseline mission. The system subgroup was responsible for developing the baseline mission design (drawing on technical input from all other groups) and then ensuring integration of the various mission subsystems.

The final baseline design was developed by performing a semi-quantitative trade-off between a wide variety of candidate mission designs.

2.3 Launch and Trajectory

(Refer to the reports by Dhiri [3], Poetro [14] and Pons [15].)

The launch and trajectory group were responsible for considering the launch options (launcher and launch site), and considering the possible interplanetary trajectories from Earth to Mars (including achieving a suitable circular orbit at Mars). The option of a lunar swing-by was considered in some detail because of the potential ΔV gain, and only discarded because of the very narrow launch window (approximately 4 minutes) it imposed. Figure 2.1 shows the Mars Xpress spacecraft stowed in the launcher fairing.

2.4 Descent

(Refer to the reports by Chameau [2], Pletinckx [13] and Rossignol [16].)

The task of the descent group was to design an appropriate strategy for leaving the circular orbit about Mars and achieving a safe landing sufficiently close to the desired landing site. The strategy chosen was an atmospheric entry using a heat shield, and then a parachute to decelerate the spacecraft, followed by a controlled soft landing (to ensure adequate landing accuracy). Figure 2.2 shows the descent module which performs the atmospheric entry and descent.

2.5 Lander Design

(Refer to the reports by Greenway [6], Holmes [8], Matakidis [9], McGrath [10] and McKown [11].)

Several "lander" options were considered (including a balloon), but the final choice was a large rover (with the lander proper being no more than the minimum structure necessary to deliver the rover safely to the surface). The rover carries all the scientific payload and is designed for a range of about 10 km over a lifetime of 250 Martian days. The rover design is based on an evaluation of previous rover designs for planetary exploration and is a "conventional" wheeled vehicle (Fig. 2.3).

2.6 Operations and Telemetry

(Refer to the reports by Butler[1], Chameau[2], Fereday[4], Hannington[7], Holmes[8], Matakidis[9], and Pabon[12].)

Aspects of mission operations and telemetry were considered by several students. Chameau and Fereday both consider the sterilisation policy and procedures to avoid contamination of the Martian surface from Earth or by the lander. Pabon discusses the project organisation.

Hannington, Holmes and Matakidis all looked at aspects of the data handling, storage and transmission (including an orbiter relay). A mission constraint is the bandwidth of the link back to Earth and it appears that the main constraint is in the link from the relay orbiter to Earth (40 Mbit / day) rather than the surface to orbiter link (even though that can only operate for two short periods during orbiter overpass each day).

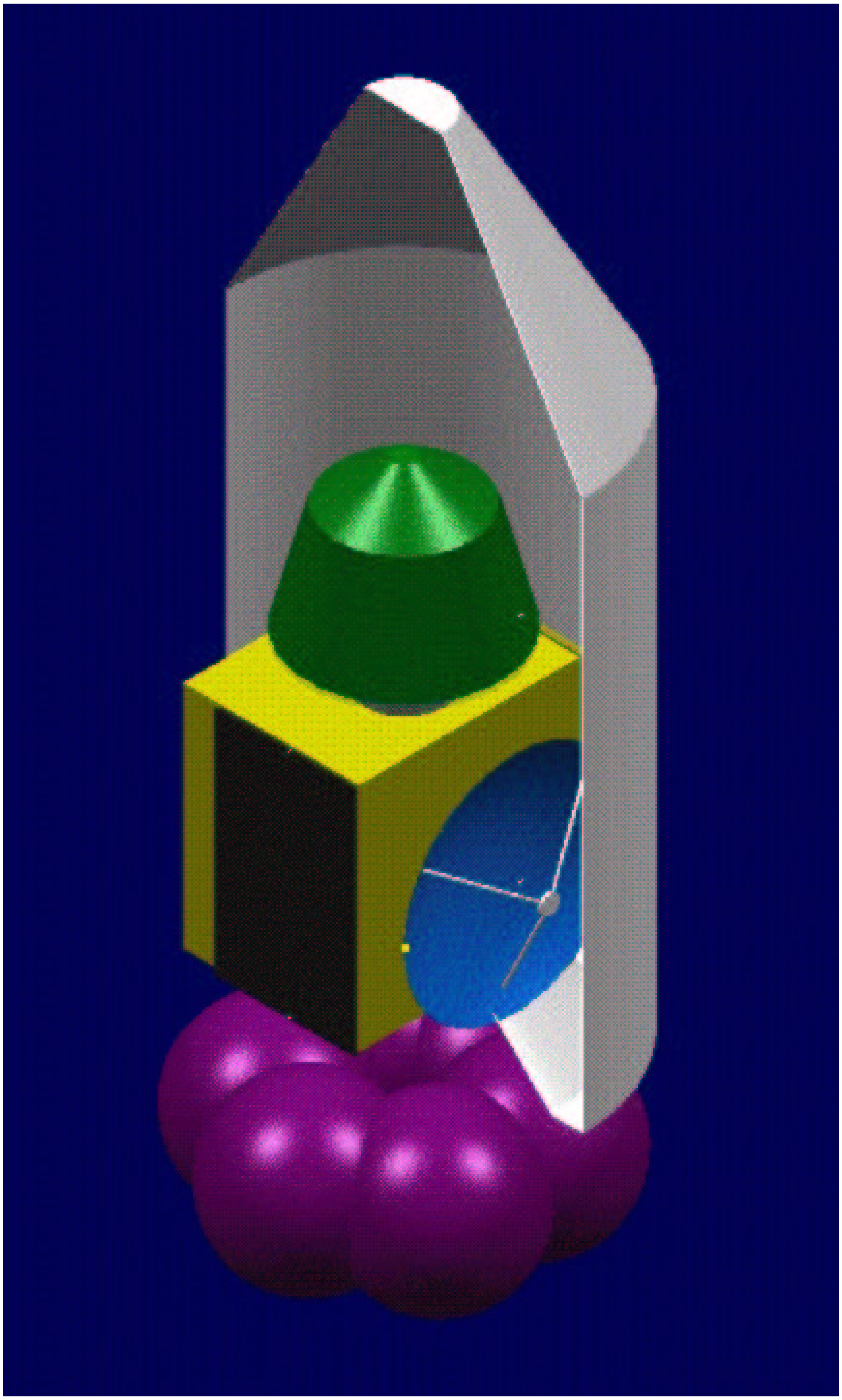

Figure 2.1: The Mars Xpress spacecraft stowed for launch, with the lander on top of the orbiter [6].

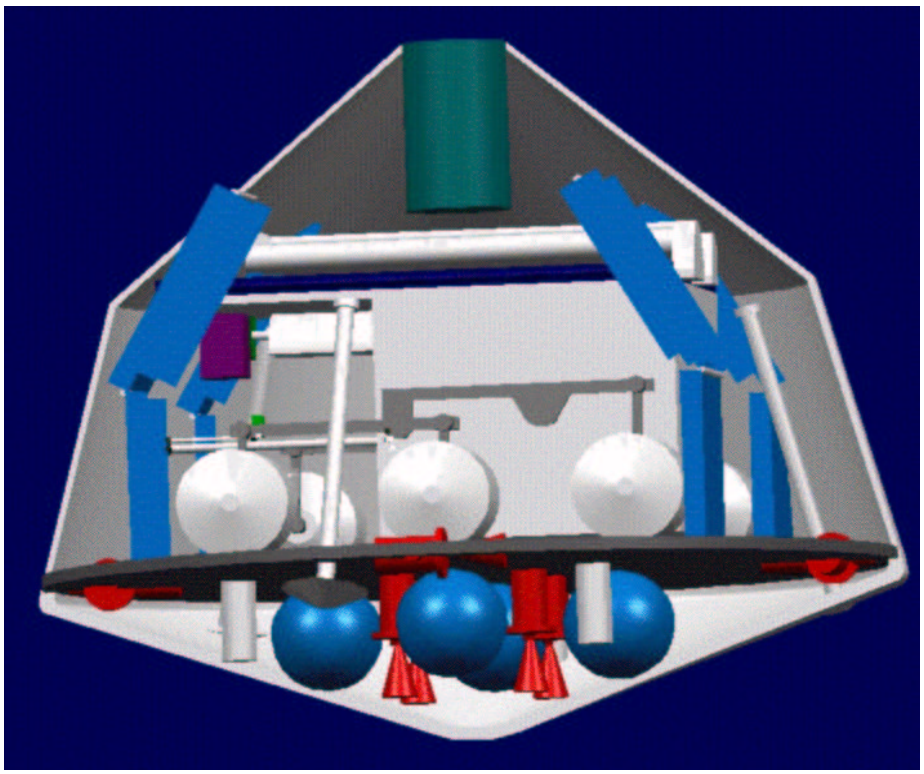

Figure 2.2: A cut-away view of the lander showing its stowed configuration [10].

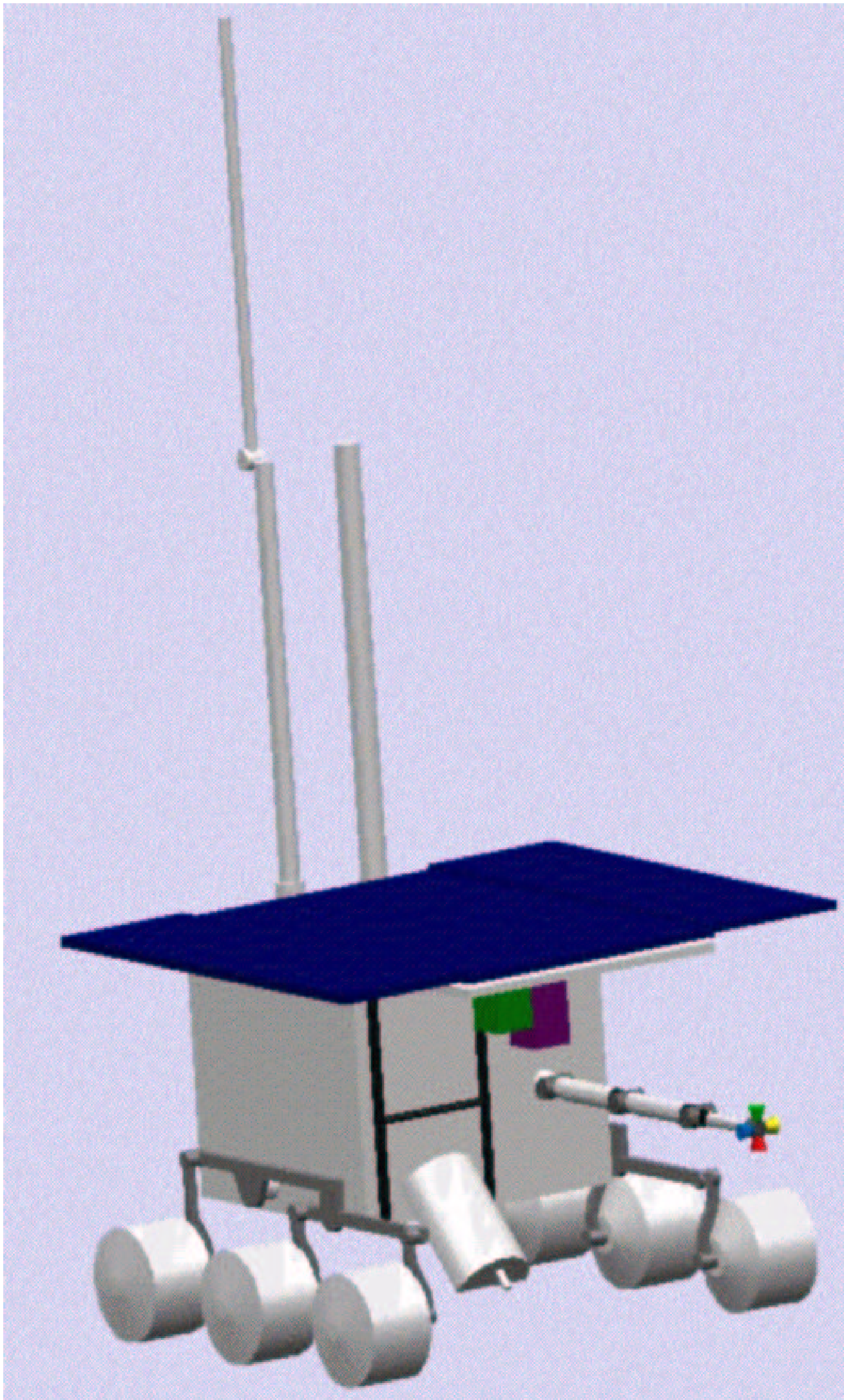

Figure 2.3: The Mars Xpress rover in its fully deployed configuration [10].

Butler considers surface operations, and describes mission planning. Two main experimental phases are envisaged: a main phase of 250 Martian days (sols) during which the primary research objectives will be tackled. The mission design lifetime is for the whole of Phase I. Tentative plans are proposed for a continuation phase on the assumption that there is a reasonable possibility that the system will still be operational beyond 250 sols.

2.7 Discussion

The conclusion of the study was that as far as the team were able to take their research the proposed mission appeared feasible. The search for life or indicators of life would clearly be a major achievement if successful, and is a goal which will not be directly addressed prior to the proposed mission data of 2003.

Several areas of uncertainty remain, some of which could have a major impact on the mission's feasibility if further work identifies significant problems. Some of the main areas requiring clarification are:

- Choice of landing site: Crater Lake appears to be a good target since it is feasible to land nearby and there are good reasons to expect to be able to detect evidence of life here if any existed on Mars. If for some reason the landing site has to be changed this could have a major impact on the mission.
- The descent strategy, in particular the final stage of the descent must achieve a demanding landing accuracy of better than 10 km. If this proves to be impossible then an alternative landing site may be needed where the mission can tolerate the reduced landing accuracy. As mentioned above, a change of landing site could have major implications for the mission.
- Rover design. The rover design's strength is its mobility, reducing the risk of missing important discoveries because of an unlucky landing. However, its mobility is also its most challenging feature, since the terrain is relatively unknown and it is quite possible that the rover will be incapable of reaching desired objectives. Further information about the landing site may help clarify this issue. Other issues raised by the study are the rover's thermal management (is it necessary to use radioactive sources?), whether it can achieve its design lifetime and range, and its instrument complement and means of handling samples.
- Data communications. There is an apparent data bottleneck in the communications link from the orbiting relay to Earth. Careful design of data collection, processing and transmission strategies is necessary to make best use of the available link.

The detailed work described in the individual reports provides the background material to the topics summarised here and documents the various choices made through the project. The executive summaries in Appendix C give a good indication of the scope of the work performed by each student.

Chapter 3

Conclusions

The proposed mission (summarised in Table B.1) appears to be feasible and offers an exciting opportunity to search for signs of past, or even present, life on Mars - one of the most intriguing scientific issues for mankind. The proposed mission is relatively low mass and low cost (expected cost of 160 MAU excluding scientific payload instruments), and could be achieved in a short timescale (the target date for arrival at Mars is 2003) using currently available technology.

3.1 Future Work

The study reported here is only an initial feasibility study which would have to be evaluated in detail before any actual mission could be designed. Several particular areas requiring further work have been identified:

- Achievable landing accuracy
- Rover design, especially thermal design and mobility relative to expected terrain
- Communication link bandwidth

These are judged to be the most significant areas of uncertainty in the mission as proposed although the whole proposal should be re-evaluated in detail if the work were to be taken further.

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Appendix A

Organisation of the Project

All the course students work as one team on the group project during Terms 1 and 2 (October to the end of March). Formal weekly progress meetings are held with staff present and are minuted with any necessary actions noted. Subgroups meet as necessary between these main weekly meetings.

During the first few weeks a general investigation of the science requirements was carried out by all students. After this initial phase students chose one of the four technical subgroups (launch and trajectory, descent, lander, or payload and operations) and started work on specific areas to allow a baseline mission to be defined. A fifth subgroup (systems) was formed and until Christmas had delegates from each of the technical subgroups. After Christmas the systems group had a permanent membership (Fereday, Pabon) with responsibilities only at system level.

The initial task of the team is to define a baseline mission, and then once this is done the team work to refine aspects of detailed system design. The baseline mission definition was achieved by January. The system group coordinates development of the baseline mission and then is responsible for integration of the detailed technical work of other team members into the mission.

Research was structured around a set of work packages. The following figures (Fig. A.1 to Fig. A.6) show the work packages defined and their relation to the five subgroups. The individual reports (references [1] to [16]) and their executive summaries (Appendix C) all refer to this common work package structure.

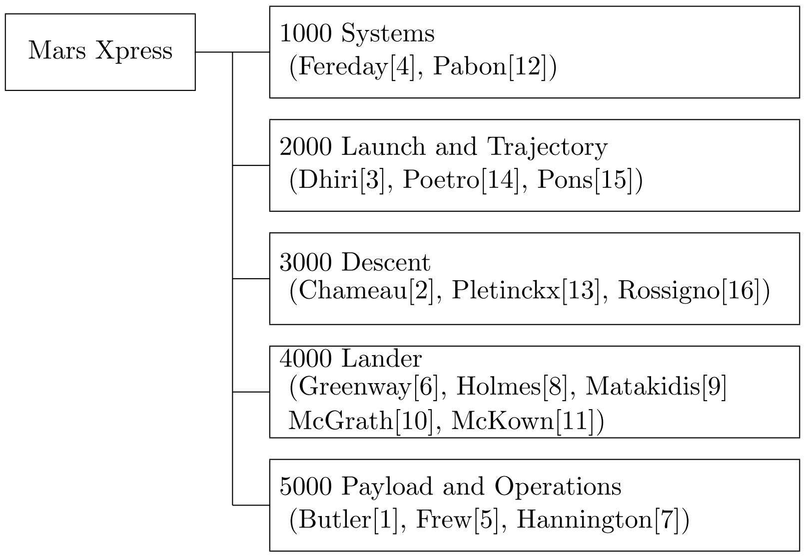

Figure A.1: Project organisation diagram showing the main subgroups and their members.

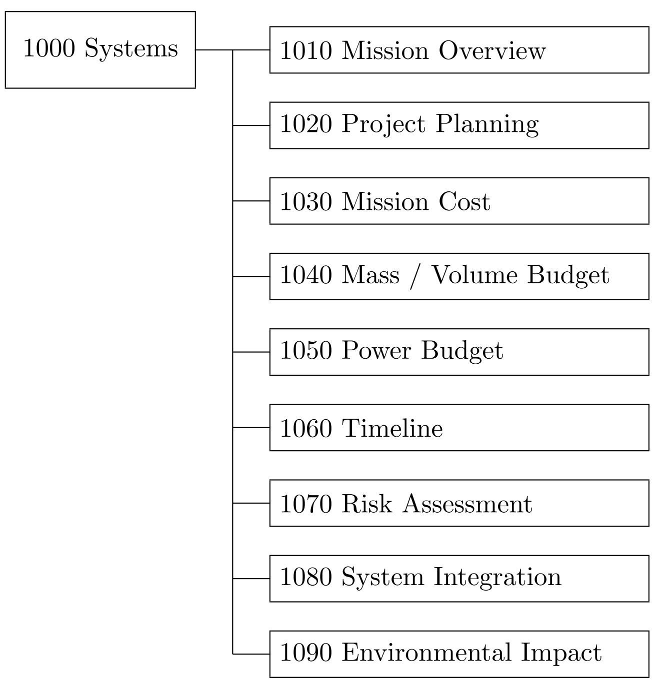

Figure A.2: Systems group work packages.

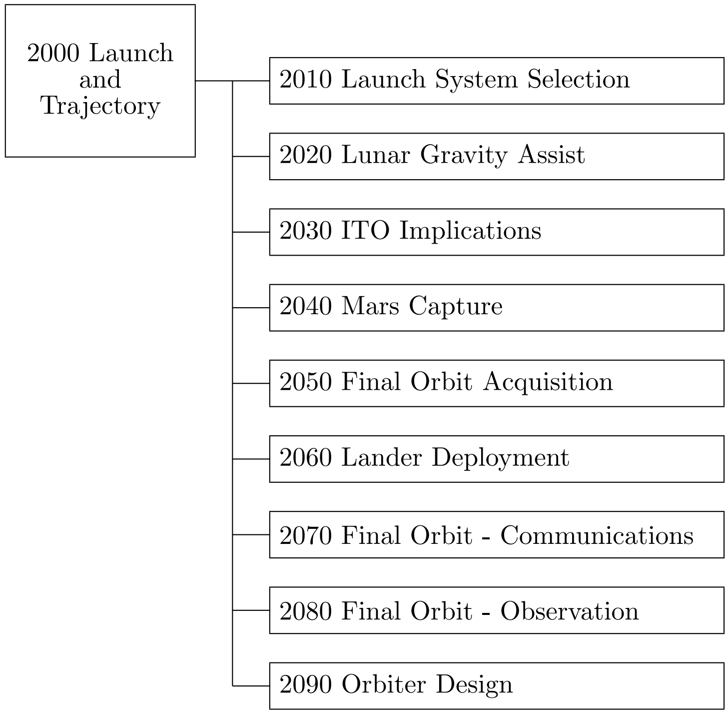

Figure A.3: Launch and trajectory group work packages.

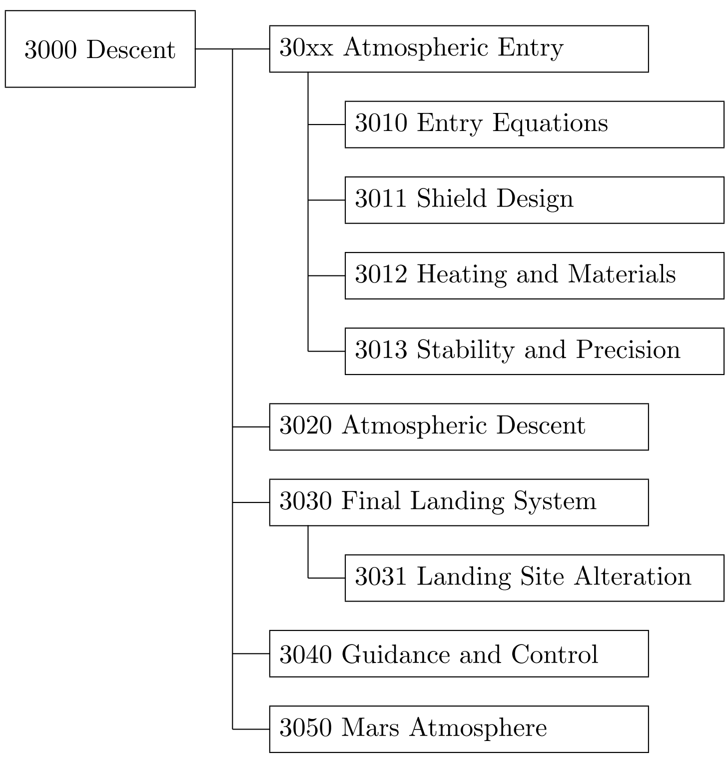

Figure A.4: Descent group work packages.

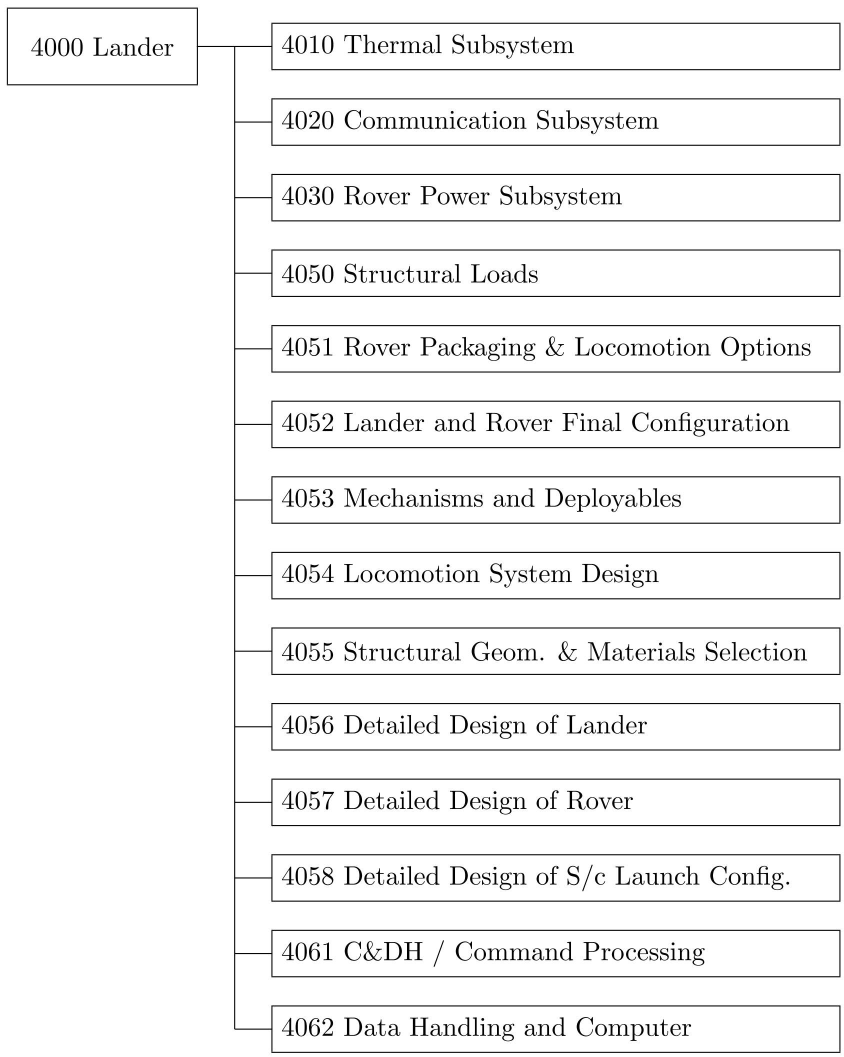

Figure A.5: Lander group work packages.

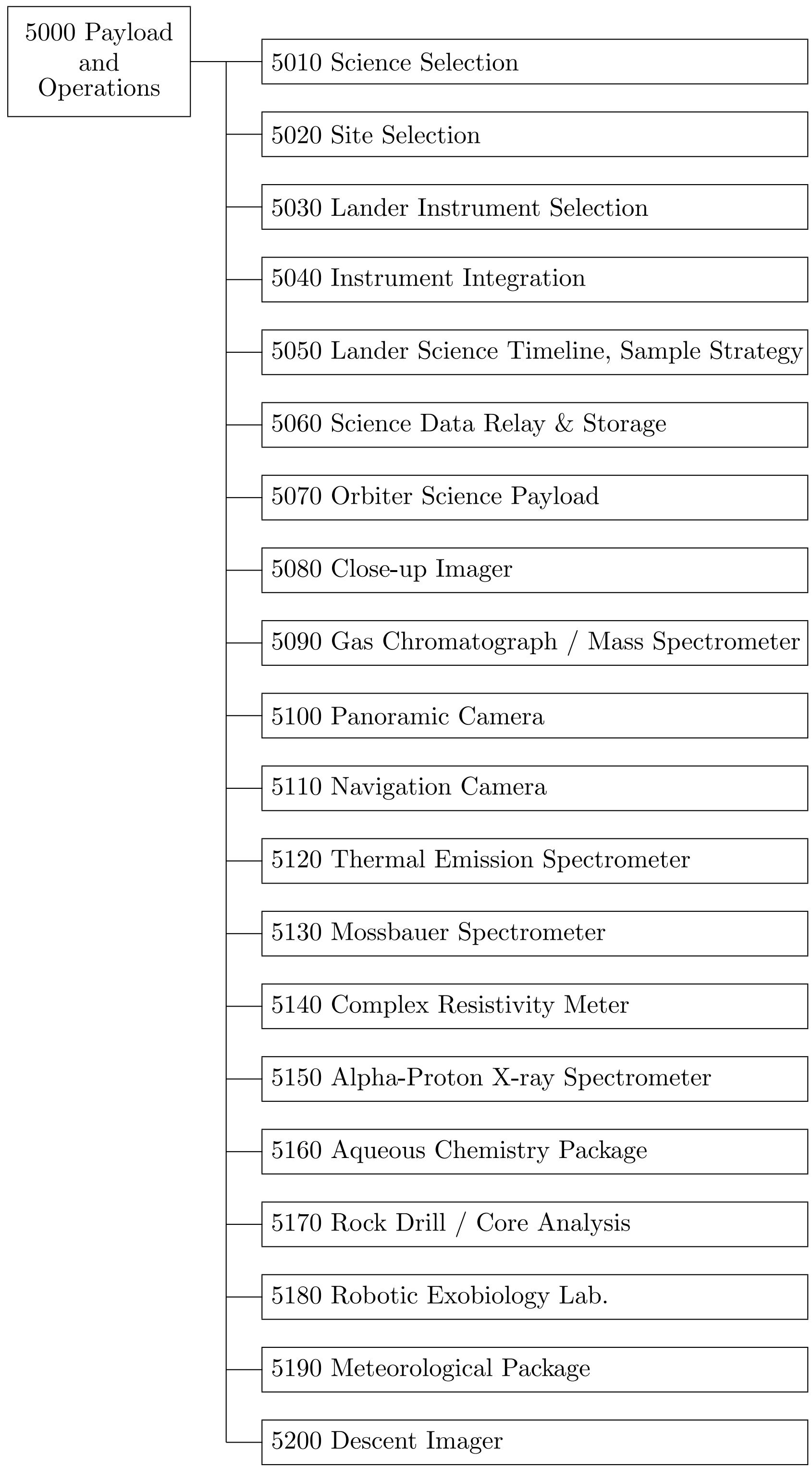

Figure A.6: Payload and operations group work packages.

Appendix B

Mission Summary

The main objective of the mission is to perform exobiology research (the study of life apart from that on Earth) on the Martian surface. The mission takes advantage of favourable planetary positions to obtain a relatively low energy transfer from Earth to Mars in 2003. On arrival at Mars, the spacecraft separates into a lander and orbiter; the main focus of this study is the lander. The lander is designed for a controlled final descent to ensure a relatively high precision landing close to significant features on the Martian surface.

A rover is the main element of the lander (once the rover has deployed from the lander the lander structure plays no further part in the mission). The rover carries a range of payload instruments to help search for signs of past or present life on Mars and is able to search to a range of about 10 km from the landing site. The initial phase of the mission is 250 Martian days which is the planned life of the mission. After this period any further exploration is a bonus and the rover can attempt to explore further from the landing site.

The orbiter is used as a data relay back to Earth as well as carrying out its own science using a variety of imaging payloads.

Table B.1 [4] summarises the whole mission.

Science goals	Primary	Exobiology
	Secondary	Geology
		Meteorology
		Orbiter mapping
Launch and orbit	Soyuz-2 launch vehicle	
	Baikonur cosmodrome	
	Parking orbit	
	Lunar gravity assist	
	Elliptic transfer orbit	
	Release of lander before MOI	
	Capture using aerobraking, aerocapture and burns with a duration of 47 days	
	Final orbit - 600 km altitude, near polar, circular, 2 overhead passes per day with total duration of 12 min	
Spacecraft	Launch mass	1171.5 kg
	Propellant	327.1 kg
	Orbiter dry mass	603.2 kg (inc. 20% margin)
	Orbiter instruments	101.6 kg
	Orbiter bus	418 kg
	Landing module	241.2 kg (inc. 10% margin)
	Rover	66.8 kg (inc. 12.5 kg instruments)
	Lander module	45 kg
	Descent system	107.5 kg
Descent system	Atmospheric shield plus cover	45 kg
	Drag increaser	17.5 kg
	Final landing system	45 kg (inc. hydrazine propellant)
	Heat shield volume capacity	2 m dia., 1.3 m high
Rover	Lander module mass	45 kg
	Descent camera	0.5 kg
	Rover mass	Instruments 12 kg
		Communications 4.3 kg
		Structure & mechanism 25 kg
		Thermal accommodation 5 kg
		Power 17.5 kg
		C&DH 2.5 kg
	Solar array size	1.6 m^2, GaAs
	Battery capacity	11 Ahr
	Thermal	Heating by RHU's and solar array or battery powered heaters
	Dimensions	2 m x 1.3 m
Operations	Landing site - Crater Lake within Gusev Crater (14.5° S, 186° W)	
	Orbiter to Earth - X-band - 40 Mbit per day	
	Rover to orbiter UHF omnidirectional helix antenna	
	Data uplink 128 kbps for 12 minutes - 92 Mbits per day	
	Payload power allocation - 15W daytime, 6W night	
	Nominal mission duration - one Martian year	
	Primary mission phase duration - 250 sols	
	Traverse distance - maximum 10 km	
	Secondary mission phase - for as long as the rover survives	

Table B.1: Mars Xpress mission summary table [4].

Appendix C

Individual Report Executive Summaries

Executive summaries for all the project reports are given in this appendix. Full copies of the reports may be referred to at the College of Aeronautics, Cranfield University, UK.

The summaries have been only lightly edited. The reports have been examined and any major errors that were identified have been corrected. However, it is not possible to guarantee that no errors remain; users of the summaries and reports should bear this in mind.

C.1 Lander scientific payload selection and surface science timeline (Simon Butler)

The planet Mars provides us with the clearest opportunity to study another terrestrial planet other than our own, at a relatively modest cost. The exploration of Mars could yield vital scientific data on the origins of our own planet Earth and lead to a more constrained definition of the planetary conditions that constitute a 'window of opportunity' for the emergence of life. Further study of Mars, with its record of abundant liquid water early in its history, is of major importance in regard to the origin of life, further exploration of the planet will hopefully enable us to answer the question "Does life, either extinct and extant, exist anywhere else in the solar system apart from Earth?". At present the only viable means of conducting in-situ experiments on Mars is robotically, the risk and complexity of manned missions makes them too dangerous and expensive.

Previous successful lander missions to Mars have failed to provide any evidence that life has ever existed on Mars. The Viking missions in the 1976 did carry instrumentation capable of detecting life but results from these experiments proved inconclusive, oxidation due to the high UV flux would have probably destroyed any organic compounds at the Viking sample sites anyway. Future missions by NASA are aimed primarily at technology demonstration, leading up to a sample return mission in 2005. This NASA strategy does not appear to endorse the in-situ exobiological exploration of Mars, the MarsXpress mission hopes to fill this apparent niche.

The MarsXpress mission will be launched aboard a Soyuz 2 rocket from Baikonur, Kazakstan during June, 2003. After a three month lunar assisted transfer to Mars a lander will be deployed to the Martian surface. The lander will make a soft landing within Gusev Crater (15.5° S , 184.6° W). The primary science focus is exobiology, secondary science focus is meteorology and geoscience. The surface science mission is based around a highly mobile (10km) rover only strategy.

C.1.1 Science Timeline

The surface science phase of the MarsXpress mission will last for one Martian year and will be split into two parts as shown in the table below.

Parameter	Primary mission phase	Secondary mission phase
Duration	Sol 1 to 250	Sol 251 and on
Traverse distance	10 km	N/A
Traverse days	100 (200 m / day)	N/A
Sampling sites	15 (10 sites / day)	Meteorological science only
Science days	150	N/A
Science power budget / W	15 (day) / 6 (night)	TBD
Uplink data rate	77 Mbit / day	TBD
Orbiter to Earth data rate	40 Mbit / day	40 Mbit / day

Table C.1: Surface science mission phases

Instrument	Mass / kg
Alpha-Proton-X-ray Spectrometer	0.40
Aqueous Chemistry Package	0.80
Close-up Imager	0.50
Descent imager	0.50
Drill / Core sampler	1.70
Gas Chromatograph / Mass Spectrometer	5.00
Meteorological Package	0.20
Mossbauer Spectrometer	0.40
Panoramic Camera	2.00
Complex Resistivity Meter	0.30
Thermal Emissions Spectrometer	0.70
Total mass	12.50

Table C.2: Science payload instruments.

Parameter	Value
Mass (sensor head) / kg	0.10
Mass (electronics) / kg	0.30
Dimensions (sensor head) / mm	65 (dia) x 40
Dimensions (electronics) / mm	80 x 70 x 60
Power requirement / W	0.30
Data rate	16.4 kbit / measurement
Duty cycle	10 hr / measurement
Thermal envelope (sensor head) / deg C	-120 to 10
Thermal envelope (electronics) / deg C	-50 to 50

Table C.3: Alpha Proton X-ray Spectrometer data.

C.1.2 Science Payload

The lander will carry a total of eleven instruments, see table below. Of the eleven instruments, ten will be mobile aboard the rover and one, the descent imager will be left at the landing site.

An Alpha-Proton-X-ray Spectrometer (APXS) instrument will be used to support the primary science objective of exobiology by determining the bulk quantities of all biogenic elements, (C,N,O,P,S), except H. The APXS technique can also determine the elemental chemical composition of planetary soil and rock samples. The performance capabilities of the APXS instrument have been successfully demonstrated in work extending back almost three decades, which lead to the inclusion of APXS instruments on the recent Mars96 and Mars Pathfinder missions. APXS instrument characteristics are shown in the table below below.

The APXS sensor head is mounted on a robotic arm which allows it to be placed against soil and rock samples in arbitrary positions, ranging from vertical to horizontal, in order to perform in-situ analysis.

A Descent imager data will provide the first images of the Martian surface sent back by the lander phase of the MarsXpress mission. The descent imager will take loosely nested panochromatic images of the Martain surface during the final stages of the lander descent phase. The descent imager will compliment

Parameter	Value
Mass (sensor) / kg	0.30
Mass (electronics) / kg	0.10
Dimensions (sensor) / mm	50 x 50 x 40
Dimensions (electronics) / mm	50 x 46 x 13
Power requirement / W	2.0 / 0.1 standby
Data rate	40 Mbit total
Duty cycle	60 s during descent
Thermal envelope / deg C	-50 to 50

Table C.4: Descent imager characteristics.

the ground based rover imaging suite and orbiter camera by aquiring pictures of local and regional geography in the immediate vicinity of the MarsXpress landing site from a new perspective which as well as supporting the science objectives of the mission will also provide very good PR.

The descent imager will address some of the problems previous lander missions to Mars have had in showing the landing site in context of the regional topography. Descent imager instrument characteristics are shown in the table below below.

Parameter	Value
Mass (sensor) / kg	0.10
Mass (electronics) / kg	0.40
Dimensions (sensor) / mm	(50 x 20
Dimensions (electronics) / mm	30 x 30 x 40
Power requirement / W	4.0 / 0.1 standby
Data rate (after compression)	512kbit / measurement
Duty cycle	10min / measurement
Thermal envelope (sensor) / (C	-120 to 10
Thermal envelope (electronics) / (C	-50 to 50

Table C.5: Close-up imager characteristics.

A deployable drill / core sampler will be used to obtain rock and soil samples for use in the gas chromatograph/mass spectrometer and the aqueous chemistry package. The core sampler will obtain rock and soil samples that have not been oxidised by the high UV radiation levels at the Martian surface. Target areas that are thought to be protected from UV include material at shallow depths within rock and material buried at depths of greater than 1.5 to 2m in the ground.

A close-up imager will provide very high resolution close-up images of soils and rock mineral grains. These observations will be used to augment the detailed mineralogical and elemental analyses carried out by other rover instruments. A vast amount of information can be obtained by studying soils and rocks with an imager of sufficiently high resolution that enable detailed characterisation of coatings, weathering rinds, individual mineral grains, clasts, or other particles. With very high spatial resolutions there is also the possibility, however remote, of finding fossil evidence of past life, an exobiology objective. Image data can be used to identify small-scale veins of precipitated minerals like the carbonates that contain possible microfossils in Martian meteorite ALH84001.

The close-up imager will be mounted on a robotic arm which allows it to be placed against soil and rock samples in arbitrary positions, ranging from vertical to horizontal, in order to perform in-situ analysis.

C.2 Final Descent System (Olivier Chameau)

The aim of this executive summary is to present all the work I carried out about the Group Design Project. It encloses an history of the design of the final landing system, and shows the role that my work played in the whole Mars Xpress project.

C.2.1 Mars Xpress Mission

The aim of the Mars Xpress project is to land on the surface of Mars a scientific payload of 12.5 kg using the lowest possible lander mass ejected from a mother spacecraft, following an optimal launch date. The scientific objectives of the mission are the research for life. Originally, the mission could include several landers, but in the final design, only one lander will be considered. It will be able to put safely a rover containing all the necessary instruments for the experiments it has to carry out. The mission was divided into 4 main parts :

- The launch and trajectory
- The descent
- The lander
- The science and operations

A fifth part, the System, deals with the integration of the 4 first parts in a homogeneous system. The system group was responsible for trading off the possible solutions and had to take the right decisions concerning the orientation of the project. In the design of a space mission, one of the the main drivers is the cost. We had to keep in mind certain figures, like the mass allowed for the whole mission, insofar as the overall cost grows dramatically with the entire mass of the spacecraft. At the earliest stage of the Group Design Project, the mass breakdown was not very well known, as several parameters had to be decided. But what we knew is that the launch mass was not to exceed 1100 kg, including spacecraft(s) plus fuel. Following the progress of our work, some precisions came out, as the rough mass budget allowed for the mission :

- Rover : 69 kg
- Lander : 54 kg
- Descent system : 74 kg
- Total : 197 kg

C.2.2 Insertion of the descent group in the mission

The descent phase is one of the more complex of the mission. It involves some very complex phenomena : mechanical, thermodynamical, electrical, chemical... The main problem comes from the fact that the descent system is cornered between the Lauch and Trajectory phase, and the Lander phase. The descent team has to meet the requirements coming from both groups. Then, the design of the descent system becomes picky. On the one hand, the launch and trajectory team ends up with an entry altitude and velocity whereas on the other hand the lander team requires a minimum accuracy in landing site, a maximum G-load at touchdown, maximum mass and dimensions for the descen system, no pollution of the landing site, etc...All the difficulty in the Descent System design is precisely to meet simultaneously all these requirements. But it is also what makes itself so interesting. In the descent team, we divided the work into 3 parts :

- The atmospheric entry
- The atmospheric descent
- The final landing system

Anthony Rossignol studied the first one, whereas Olivier Pletinckx was responsible for the second one. I was in charge of the last one. Even if these parts seem different from each other, we worked in great collaboration, and always took important decisions together, so that the descent system remains homogeneous.

C.2.3 My personal work in the project

Study and modelling of Mars atmosphere

The role of the descent team was to design a system aimed at decreasing the velocity of the spacecraft before it touches the surface of Mars. By increasing the drag in the atmosphere, such a velocity reduction can be achieved. As the differential equations that describe the behaviour of the spacecraft during the descent involve several parameters of the atmosphere, such as pressure, temperature, density, composition,... it was necessary to compute a simple model that approaches the actual values. Thus, integration of differential equations became much easier. This study uses the data brought back by the Viking missions in the seventies. The two landers of Viking 1 & 2 recorded the parameters of Mars atmosphere during their descent towards the ground. By analysing these data, I could check that a perfect gas model, using a 100 % carbon dioxide composition at 160 K, gave some results that matched the actual values for temperature, density and pressure. This work was fully used by my colleagues of the descent team, insofar as they were in great demand for such parameters.

Study of possible solutions for the Final Landing System

Since the beginning of the project, the descent team investigated all possible solutions for the descent scenario. Hard landing, semi-soft landing, soft-landing... several solutions were possible, each of them offering advantage and drawbacks. I was responsible for comparing the different solutions available for the final landing system. The Pathfinder scenario, using a system of airbag to damp the final shock, had many advantages, but the accuracy and safety requirements forced us to go for a soft landing. That is why I drawed inspiration from the Viking lander to design the final landing system.

Design of the Soft Landing System

The soft landing system uses several retrorockets to reduce the vertical and horizontal velocities of the lander before touchdown. I was in charge of the study, design and sizing of the controlled retrorockets that operate during the landing. Several parts of the system needed consideration. After having chosen the retro-rockets technology (mono-propellant hydrazine engines), some modifications were required in the nozzle, so that it is adapted to Mars atmosphere. Then, a specific control system for the retrorockets had to be design. I went for some inclinable rockets, thanks to electrical engines. These engines are controlled by the flight computer, which receives altitude informations from a radar and attitude information from gyros.

Landing site alteration

One of the main drivers of our mission is the science. We want to make some scientific operations on the surface of Mars, the aim of which is to detect any form of life. To investigate, some exobiology experiments will be carried out. Therefore, we do not want the hydrazine thrusters to pollute the landing site, or to damage any potential micro-organisms. Some soil samples should also be taken to perform chemical analysis. The products of decomposition of hydrazine in the thrusters must not be mixed with the samples, otherwise the measures would be nonsense. It is therefore important to study the chemical, but also physical impact of the hydrazine thrusters on the landing site.

C.3 Launch and Trajectory (Viney Dhiri)

This Astronautics & Space Engineering (ASE) group design project aims to investigate and design a mission to Mars. Science requirements are the drivers to the mission although cost efficiency is a high priority. Development speed is also an issue but is not covered in great detail in this report.

The primary science objective is to conduct extensive surface science at Mars in the area of exobiology. Secondary objectives are to study geology, meteorology and to map the Martian surface. To this end a mobile lander will cover the primary and some of the secondary objectives and an orbiter will serve as a mapping tool as well as a communications relay between lander and Earth ground stations.

The initial objectives outlined here are based on that of the current Mars-Express mission being undertaken by the European Space Agency (ESA). The addition of seismology in the ESA mission was dropped in the ASE mission due to priority shifting to mobile surface science.

C.3.1 Mission Breakdown

The mission is sectioned into four areas of investigation as follows;

- Launch & Trajectory
- Descent Systems
- Lander Design
- Payload & Operations

This report deals with the Launch and Trajectory aspects of the mission. The subjects covered are as follows;

- Launcher Selection
- Lunar Gravity Assist
- Interplanetary Transfer
- Lander Release
- Mars Aerocapture
- Mars Aerobraking
- Orbiter Design
- Final Orbit Communications
- Final Orbit Science

The strategy options were investigated in each subject. Some aspects are covered in this report. The remainder are covered in the work completed by M X.Pons and Mr R.Poetro.

C.3.2 Launcher Selection

The crucial first step in the mission. There are around 11 launcher vehicle systems stated as capable of putting a payload into interplanetary transfer. Cost was discovered to be the main driver in the selection process and therefore from the 11 launcher systems two launchers were found to stand in a league of there own, the Molnyia and Soyuz-2 Russian class-A vehicles at \$ 12-25M. It was also discovered that the Soyuz-2 vehicle has an increased payload mass capacity over Molniya by using a more powerful third stage called a Fregat. This increased payload volume and mass capability was found to be necessary to the mission and so Soyuz-2 became the final choice. The location of the launch site also offered advantages in a reduced minimum inclination parking orbit (45.6°). Meaning less fuel is required to change planes. The Soyuz-2 vehicle is still under development and as of yet still unproven with the Fregat incorporated.

The performance capability of the Soyuz-2 launch vehicle was evaluated using the customised Launcher Performance Model (LPM) program. Results are given in Table C.6.

Max Payload Mass	1200 kg
Parking Orbit altitude	200 km
Parking Orbit type	Circular
Min inclination	45.60°
3rd Stage ΔV available	4.05 kms^{-1}

Table C.6: Launch performance model results for Soyuz-2.

The LPM program used to model the launcher performance is basic and would need significant updating to improve the reliability of the results.

C.3.3 Mars Aerocapture

The aerocapture technique uses the atmosphere of a planet, in much the same way as aerobraking, to slow the velocity of a spacecraft down. Atmospheric drag decelerates the spacecraft that is entering at a hyperbolic escape velocity. A deeper penetration of the atmosphere is called for then in aerobraking. This increases the need for additional thermal protection due to aerodynamic surface heating of the s/c. Hence, the mass and cost of the vehicle is increased. A trade-off is born between cost/thermal mass losses and fuel/mass gains.

The advantage of aerocapture is that if complete capture to a specified orbit can be accomplished without 'frying' the vehicle then significant fuel savings can be made. Traditionally this manoeuvre would be performed in space with thruster burns. A customised Aerocapture Model (AM) program was designed and utilised to give the characteristics of the final orbit acquired. The data collected for the final configuration are given in Table C.7.

The final captured orbit acquired is similar to that of the US Mars Global Surveyor that used traditional burns to capture. Deceleration values compared well with other similar models. The excessive surface heating that has been calculated is initially put down to the evaluation methods adopted. The method of analysis needs to be verified and tested. Due to the dubious nature of the temperature the capture orbit, inputs were selected to give a minimum capture configuration. This was a minimum negative entry angle and the least duration

Inputs	Entry Velocity	9 kms^{-1}
	Atmospheric Entry Angle	-30°
	Time Within Atmosphere	327 s
	Altitude Depth	100 km
Results	Max deceleration	4.37 kms^{-1}
	Max surface temperature	6680 K
	Periapsis	3738 km
	Apoapsis	55313 km
	Semi-Major Axis	29526 km
	Eccentricity	0.873369

Table C.7: Mars aerocapture model inputs and results.

spent within the atmosphere. This was in order to minimise any surface heating induced at the cost of the resultant need for longer aerobraking duration.

C.3.4 Orbiter Design

The orbiter design was a low priority factor is this mission and therefore rudimentary estimates have been taken in this part of the investigation. The various sub-systems considered in the analysis were as follows;

- Payload
- Spacecraft Subsystems
- Propulsion Attitude Control
- Communications Command and Data Handling
- Thermal
- Power
- Structure and Mechanisms
- Margin
- Propellant

Each sub-system was considered to give a mass breakdown of the orbiter. This led to the creation of an overall mass breakdown program that allowed an iteration process to occur each time the mass of a component in the MX spacecraft was revised. The program allows the injected mass of the mission to be estimated. The final mass breakdown is illustrated and discussed in the work completed by MrB.Pabon. The main values and some configuration detail are given in Table C.8.

Main Apoapsis engine and multiple thrusters based on Royal Ordinance LEROS 1 with the features given in Table C.9. The use of a dual-mode motor allows one tank and feed system to supply all thrusters and the main engine. It is lighter and is cost effective in development.

Mass of orbiter Dry	603.2 kg
Mass of orbiter Full	625 kg
Ballistic Coefficient	62.18 kg/m^2
Area of orbiter	4.57 m^2
Volume	6.25 m^3

Table C.8: Principal parameters for the Mars orbiter.

Dual Mode	
Propellants	MON (mixed oxides of Nitrogen) and Monomethylhydrazine (MMH)
Thrust	490 N main motor, 22.2 N multiple Thrusters (12)
Nominal Isp	315s
Mass	4.15 kg

Table C.9: Characteristics of Royal Ordnance LEROS-1 thrusters.

C.3.5 Final Orbit Science

Mapping is the secondary objective outlined by the payload and operations team. The primary objective involves the science being conducted by the lander and thus the design of the final orbit geometry is driven by communications of the data being gathered by the lander.

The limits to the planetary science and observation that can be conducted from the final orbit around Mars specified was briefly investigated. The requirement for a sun-synchronous orbit was also investigated, as were the resultant perturbations for the specified orbit. A more detailed analysis of the science instrument performance available from the specified orbit is covered in Mr M.Hanningtons work. The main results are given in Table C.10.

The main results of importance for the instrument studied are given in Table C.11.

It was discovered that for the specified orbit sun-synchronism was impossible, as orbit precession is too rapid for sun-synchronisation. There is a requirement to either use active control systems to maintain the orbit or to change the altitude or inclination of the orbit to accommodate sun-synchronism. Although the investigation was not taken further, changing the inclination of the orbit is the more attractive option to solve the problem due to the increase in fuel requirements for active control. These changes call for a trade-off with science requirements (mapping and communications).

Precession required for Sun-Synchronism	0.0008123 rad/orbit
Actual precession in specific orbit	0.0134 rad/orbit
Solar radiation pressure acceleration	6.68E-08 m/s^2

Table C.10: Mars orbit perturbations.

Instantaneous area coverage rate	3.473451742 km^2/s
Instantaneous area coverage rate	420.0494 km^2/s
Average area coverage rate	168.01976 km^2/s
Instantaneous access area	38369405 km^2
Area Access Rate	3.51E+04 kms^{-1}
Spacing between Successive passes	1847.7411 km

Table C.11: Imager coverage data for Mars orbiter.

C.4 Mars Xpress Mission Concept (Jayne Fereday)

Table B.1 gives a summary of the whole mission.

C.4.1 Mars Xpress Mission Concept

The mission objectives are to send a low cost mission to Mars with the aim of landing 12.5 kg of scientific instruments on a designated landing site, to conduct in-situ measurements. The mission will take advantage of the 2003 optimal launch date and, because of cost restrictions, uses a Molniya/Soyuz class launch vehicle. The launch vehicle and orbiter will be provided by ESA, but the lander payload will not be, hence a low cost approach is necessary.

C.4.2 Alternative Mission Architectures

Based on the mission objectives and requirements, a list of elements for each mission phase was drawn up. These elements were combined to give a list of possible mission scenarios. All mission scenarios involved joint launch of an orbiter and a lander module.

C.4.3 System level trade-off

Trade-offs were performed to select a mission baseline. The main driver was mass. A trade-off methodology was developed to give each option a final score. For each mission phase, every option was given a score out of five, with respect to high level parameters such as mass, complexity, reliability, and so on. Different mass and capability weightings for these parameters were used to give a range of scores for each option. These scores were then assessed and, if necessary, further analysis was performed. This resulted in a short list of 3 possible scenarios. A final preferred baseline was then selected. We retained the option of reverting to one of these three scenarios if the first choice proved unrealistic. A preliminary mass budget was drawn up and an initial allocation was made to each system.

C.4.4 Selected Baseline

The mission selected launches using a Soyuz 2 launch vehicle. After using the Moon for a gravity assist the spacecraft approaches Mars on an elliptic trajectory. The lander module is released before Mars orbit insertion and makes use of a soft landing system to put a rover down onto the surface. The rover carries just under 12.5 kg of scientific instruments and will conduct in-situ measurements using sample preparation tools. After lander release, the orbiter utilises a strategy combining aerocapture, aerobraking and limited thruster burns to achieve a near polar orbit. It will act as a communication relay for the lander and will also perform surface mapping to back up surface science. The main focus will be exobiology, but experiments studying geology and meteorology will also be conducted in order to optimise the science return and to back up the exobiology. The surface mission will have a nominal length of one Martian year.

C.4.5 Timeline

A timeline for the operational phase of the mission has been produced. It includes a summary of the phases beginning with launch, and going on to interplanetary transfer, descent and the landed science phase. Major mission phases are included, as are key events such as launch.

C.4.6 Power Budget

This chapter details the power budget for the surface mission. Initially, the power requirement for each component was estimated. The power system consists of solar arrays and a rechargeable battery for use during the night and during hostile conditions. The available size for the solar arrays and the battery was estimated. Power requirements during different operational modes were investigated. Using these preliminary figures and after refinement of the subsystem power requirements the array and battery were resized. This was an iterative process. Finally, a power profile was produced that shows power use during the different operational modes.

Subsystem	Average power / W	Peak power / W
Thermal	10	10
Communications	25	25
Computing	8.8	8.8
Housekeeping	2.5	2.5
Locomotion	20.31	20.31
Power	40	40
Instruments	37.33	62.33
Subsystems total	143.94	168.94

Table C.12: Power budget breakdown by subsystem.

C.4.7 Environmental Impact

The issue of planetary protection is also considered. As the primary objective of this mission is to study exobiology on the Martian surface it is particularly important that there is no contamination of samples by terrestrial organisms and organic compounds. It is necessary to prevent contamination of the Martian surface by terrestrial micro-organisms. A policy to prevent any forward contamination has been drawn up. It is based on the procedures used by past Mars missions such as Pathfinder and Viking, and follows guidelines laid down by International treaties.

Implementation Plan

- lander components cleaned with ethanol alcohol and bagged prior to assembly
- modules assembled in a class 100,000 clean room by workers in surgical dress
- bioassay samples taken from the interior of each module

- subsystem exterior surfaces cleaned by wiping, bioassayed and bagged
- lander assembled in class 100,000 clean room
- during testing surfaces protected using temporary covers
- bioassay done after testing - if contamination present, lander must be cleaned & reassembled
- another set of bioassays taken of exterior surfaces
- lander encapsulated in cleaned and bioassayed aeroshell subsystem
- surface cleaned after testing
- biobarrier sealed during transportation to launch site
- lander subjected to dry heat sterilisation cycle of approximately 30 hours at 111.7° C at a specified humidity of 1.3 mg/l

Some components, such as any electronics, are not capable of withstanding this kind of temperature and so other procedures must be followed. Most of the electronics will be housed within the warm electronics box on the rover which isolates them form the Martian environment. For some components it will be necessary to conduct pre-sterilisation procedures only: thorough cleaning with an agent such as ethanol, followed by extensive bioassaying to assess the level of contamination.

The second part of this section discusses some of the issues of surface impingement by hydrazine thrusters. NASA conducted tests at their White Sands facility that looked at this contamination before they launched the Viking missions in the 70s. The tests found that contamination by exhaust products such as ammonia was likely in the area immediately below the engines. There are also trace quantities of organic aniline present, but this can be reduced to below significant levels by using purified fuel. To avoid the effects of exhaust ammonia we must not sample from within 30 metres of the final landing site.

C.5 Site Selection and Instrument (David Frew)

The 1997/8 ASE Group Design Project 'Mars Xpress' mission is a low cost exobiology mission to a region of mars that may once have supported life.

C.5.1 Site Selection

The main scientific criteria for exobiological site selection are:

- A past means for supporting life
- A mechanism for preserving evidence of life

Other important considerations were:

- Sample Diversity
- Elevation
- Terrain

Crater lake was chosen as the final landing site as it fulfilled both the scientific and technical requirements of the mision.

Selection criteria for crater lake Science Merit

Science	Merit	
Exobiology	high	2 Gyr fluvio-lacrustine activity
		hydrologically dependent environment
		possible hydrothermal activity within Crater Lake
Sampling	high	sedimentary rocks
Diversity		igneous rocks
		ejecta material
		carbonates
		extinct life (perhaps even extant)
Technical		
Elevation	high	0 km elevation wrt Mars datum
		asymmetric
Terrain	unknown	good at Viking resolution
		predicted to be good for lake bed
		need MOC for higher resolution

Table C.13: Selection criteria for Crater Lake.

C.5.2 Instrument Integration

The integration of the Mars Xpress payload must focus on the the following instrument parameters.

- An assessment of power and thermal lifetime requirements
- Deployment requirements of the individual instruments, and any means of using existing mechanisms to simplify operations.

- Technical constraints imposed on the lander.

The operation of the rover is semi-autonomous - navigation, instrument deployment and sample retrieval are performed with minimum intervention from Earth.

Panoramic Camera

The panoramic camera is used in three main areas:

- Sample selection
- Navigation
- Geology

It is mounted on the elevation mast to provide the required range and perspective on the returned images.

Parameter	Value
Sensor mass	1.25 kg
Electronics mass	0.75 kg
Sensor dimensions	25 x 5 x 5 cm
Electronics dimensions	46 x 8 x 8 cm
Power requirement	4 W peak
Data volume	86 Mbit (full colour stereo)
	8:1 JPEG compression
Duty cycle	2 s / frame - stereo image
Thermal envelope	173-283 K

Table C.14: Main instrument parameters for the panoramic camera.

Thermal Emission Spectrometer (TES)

The main functions of the TES are:

- mineralogy
- meteorology
- thermal properties of the surface

It utilises the same mechanisms as the pancam allows full sweeps of the surrounding area as well as imaging of the martian atmosphere.

Mossbauer spectrometer

The exobiological uses of the Mossbauer spectrometer are:

- Mineralogy
- Microfossils at hydrothermal sites
- Biomediated material

The Mossbauer spectrometer is also used for climatology and for examining the magnetic properties of the martian dust.

Parameter	Value
Instrument mass	0.35 kg
Electronics mass	0.35 kg
Sensor dimensions	10 x 15 x 21 cm
Electronics dimensions	3 x 10 x 15 cm
Power requirement	5.5 W
Data volume	81 Mbit
Duty cycle	4 s / frame
Thermal envelope	173-283 K

Table C.15: Main instrument parameters for the TES.

Parameter	Value
Sensor mass	0.25 kg
Electronics mass	0.15 kg
Sensor dimensions	3.1 x 4.5 x 8 cm
Electronics dimensions	1 x 4.5 x 8 cm
Power requirement	0.6 W peak
Data volume	0.2 Mbit / measure
Duty cycle	10 hr day and night
Thermal envelope	173-283 K

Table C.16: Main instrument parameters for the Mossbauer spectrometer.

Meteorology pack

The Meteorological package is the lowest mass and least power consuming of the instruments in the scientific payload. At the chosen landing site, the parameters of the that are of most interest are:

- temperature
- pressure
- wind direction and wind velocity

Parameter	Value
Instrument mass	0.2 kg
Instrument dimensions	11 x 16 x 4 cm
Power requirement	0.8 W
Data volume	65 kbit / day
Duty cycle	semicontinuous
Thermal envelope	173-283 K

Table C.17: Meteorology pack for the Mars Xpress rover.

Navigation Camera

The main functions of the Navigation cameras are,

- Range finding and obstacle detection

- conformation of instrument deployment
- Self-inspection
- Additional imaging

Parameter	Value
Sensor mass	0.08 kg
Electronics mass	0.40 kg
Sensor dimensions	25 x 5 x 5 cm
Electronics dimensions	5 x 8 x 8 cm
Power requirement	2.8 W peak
Data volume	125 kbit / image, 6:1 JPEG compression
Duty cycle	3 s / image
Thermal envelope	173-283 K

Table C.18: Main instrument parameters for the Navigation cameras.

C.6 Structural Analysis and Materials (Paul Greenway)

This summary provides an overview of the personal work completed for the Group Design Project for the MSc in Astronautics and Space Engineering 1997/1998. This consists of a complete project with the aim of placing a 12.5 kg scientific payload on the surface of Mars. The primary objective of the mission is to provide valuable fresh scientific data regarding the exobiology of Mars. The aim of the study parallels the ESA Mars Express mission in planning all aspects of the mission.

C.6.1 Hard Lander Preliminary Research

Penetrators enter the planet's atmosphere at high velocity and implant scientific instruments a few (or perhaps several) meters beneath the surface. This means that the instruments and electronics have to be designed to survive the very high g impact forces. They are as yet unproven in space applications but would have been used in the ill-fated Russian Mars 96 mission. However, penetrators are to be used in the Japanese Lunar-A programme and two microprobes are to be launched on the NASA Mars Surveyor 98 mission. They have been used terrestrially for various military, scientific and remote sensing applications.

The basis of operation involves reducing impact kinetic energy on landing to be an exploitable asset. The design usually incorporates a two-body separable penetrator that allows a forebody section to detach from the afterbody. The two sections are connected by an umbilical. The forebody penetrates deeper than the afterbody as a function of the impact velocity and the surface and subsurface terrain. The subsurface instrumentation is contained in the forebody whereas the afterbody remains on the surface perhaps obtaining meteorological data and surface images.

MISSION	Typical Impact Speed (m/s)	Lifetime	Mass (kg)	Penetration Depth (m)
Mars 96	300	1 year (earth)	13	1 - 3
Lunar-A	80-100	1 year (earth)	45	5 - 6
Mars Surveyor Microprobe	200	50 hours - 20 days	2	0.3 - 2

Table C.19: Penetrator data.

Table C.19 summarises the primary characteristics of some of the penetrator missions considered. Structural Design, Configuration and Materials

C.6.2 Background to Structures and Mechanisms Subsystem

Overall, it is found that structural design is inherently dependent on other subsystems - for example thermal, communications and the power subsystem. Within spacecraft structural design it may seem relatively simple to solve the design problems and produce an optimum solution. The difficulty is producing an extremely mass efficient and highly reliable solution.

C.6.3 Structural Loads and Response of Spacecraft

The main aim is to assess all of the likely loads and environments that the spacecraft would encounter during the mission. This includes all of the applied conditions from manufacture until the end of the mission. Once they have been quantified, the results will dictate the spacecraft's geometry, stiffness, materials and required strength.

Launch Environment

The launch environment usually generates the highest loads for most spacecraft structures. These loads cannot be controlled which implies that the spacecraft has to be designed to withstand them. The launch vehicle generates random vibrations, acoustic vibrations and shocks which can all affect the payload. Random vibrations cause structures to vibrate randomly and are generated by mechanical parts which are moving, for example the turbopumps or components excited by the acoustic environment and combustion phenomena. Acoustic vibrations are sound pressure waves generated by engine and aerodynamic noise (especially around Mach 1), primarily propagating throughout the atmosphere within the spacecraft at launch. Structures that respond most are light-weight and large in surface area. Shocks are high intensity and high frequency vibrations which can be caused by pyrotechnic devices. They will occur principally during engine ignition, stage separation and lander/orbiter separation. Each of the load types generated by the launcher were analysed to quantify the effects on the spacecraft.

Performing the calculation gives the maximum spacecraft response displacement to random vibrations as 1.4 mm for a simple lump mass representation with a resonance frequency of 50 Hz.

The next stage would be to divide the spacecraft into the rover, lander and orbiter enabling a more accurate model to be produced. Ideally, a finite element analysis (FEA) would then be performed in which a dynamic simulation would assess natural frequencies and deflection displacements at different points in the structure can be calculated.

Cruise Phase Environment

The primary loads during the cruise to Mars will be experienced during the Moon gravity assist manoeuvre, course correction burns and the 47 day aerobraking procedure.

Entry into Martian Atmosphere

The descent group anticipated that the maximum load during entry would be 17-20 g which does not exceed the peak loading conditions during launch. In addition, a sudden impulsive force will occur when the parachute is deployed. Landing Loads and Martian Environment - The soft landing intends to place the lander on the surface with a maximum impact load of 5 g.

C.6.4 Packaging and Locomotion Options for Rover

Various configuration options are contemplated, each with the following common elements :

- Omnidirectional solar array panel(s)
- Structure containing electronics and internal instruments
- Locomotion system
- Communications antenna (helix type)
- Devices to allow mobile measurements
- Some structure to support the external fixed height instruments and navigation cameras

Considering the locomotion system, a planetary rover must traverse rugged and possibly unknown terrain, climb slopes and avoid boulders and crevasses. An autonomous rover must also be competent, reliable and efficient. Planetary exploration places stringent requirements on system performance and design. Surface terrain such as channels and boulders will affect stability; if the centre of mass is too low then it may be difficult to climb over rocks, but being too high may cause tipping. Various designs are considered in selecting a final locomotion system for the Xpress rover (a six wheeled system is finally chosen).

C.6.5 Locomotion System Design

The final design implements six wheels. This gives good stability characteristics and distributes the loads more effectively than a four wheeled configuration. Six wheels also offer better redundancy capabilities and can possibly give better obstacle clearing capability due to the simple fact that six wheels combined can provide more tractive force.

C.6.6 Structural Geometry and Material Selection

The objective is to state the materials that the major components of the lander and rover would be composed of and provide a mass estimation of each individual component.

C.6.7 Detailed Design of Spacecraft Configuration in Soyuz Fairing

The orbiter configuration is constant through all stages of the packaging process, for example the antenna and solar arrays remain as for the ESA specification.

The packaging considerations involve establishing that sufficient volume is available to contain the subsystem components. Appendages must also be packed surrounding the body (such as the Xpress orbiter solar arrays) within the fairing. Structurally, it must be verified that feasible load paths are transmitted between the spacecraft and launch vehicle. There must be compatibility between the spacecraft and the Soyuz mechanical interface. The finalised packaging configuration is modelled using CATIA.

C.7 Science Selection; Orbiter Science; Science Data Relay and Storage (Mark Hannington)

The current framework of United States, Japanese and Russian missions to Mars entails the most intense analysis of the planet since the spate of NASA Mariner and Viking missions of the 60's and 70's. Utilising the series of close conjunctions between Earth and Mars in the timescale 1996 to 2005 a wide range of orbital and landed investigations are planned. The Global Surveyor and Pathfinder missions have initiated the study period successfully, and the three NASA and one NASDA mission planned should provide some compensation for the disappointing failure of the Mars '96 mission.

The ESA Mars Xpress mission constitutes Europe's contribution to the investigation of Mars in this timescale.

C.7.1 Science Selection.

The selection of the science drivers for the Mars Xpress mission has been based on a rationale devised from a study of the science accomplishments of previous missions, the coverage expected by planned missions and the limitations imposed by Payload and Site Selection strategies. The Science Drivers are expressed in three resolutions, Science Focus, Science Aims and Science Objectives. It is intended that the specification of Science Objectives provides a detailed framework within which the mission can be designed at system and sub-system levels.

Mars Xpress Science Focus.

The mission will be carried out with a single Primary Focus and two Secondary Foci.

- PRIMARY SCIENCE FOCUS EXOBIOLOGY
- SECONDARY SCIENCE FOCUS GEOSCIENCE
- SECONDARY SCIENCE FOCUS METEOROLOGY / ATMOSPHERICS.

Mars Xpress Science Aims

Science Aims to be addressed by Mars Xpress have been specified for the three Science Foci. Mars Xpress Exobiology Science Aims.

Mission Science Aims within the Primary Focus are defined as;

- Sampling from target areas free of oxidants/ protected from ultraviolet radiation, detection and chiral classification of organic compounds,
- Identification of biomediated mineral/chemical fossil signature,
- Morphological evidence of microfossilisation,
- Isotopic analysis of biogenic elements

- Assessment of oxidative properties of Martian surface.

Mars Xpress Meteorology Science Aims are defined as;

- Atmospheric Structure.
- Meteorological Variation.
- Surface atmospheric Parameters and Variation.

Mars Xpress Geoscience Science Aims are defined as;

- Mineralogy
- Surface/Subsurface Morphology.
- Global Permafrost Presence.
- Subsurface Water.

Specification of these Aims allows a range of Mission Science Objectives for successful address of each Aim to be allocated. These Objectives provide the framework for Mars Xpress system and sub-system design.

The Principal Investigations for three of the twelve payload instruments (Gas Chromatograph / Mass Spectrometer, Aqueous Chemistry Package, Complex Resistivity Meter) provide a definition for each instrument in terms of its; rationale for inclusion, mechanism of operation, the physical characteristics of the instrument (mass/ volume/ structural integrity) the power requirements placed on the Rover Power Sub-System the thermal requirements placed on Rover Thermal Sub-System and the data requirements placed on Rover Command and Data Handling/Communications Sub-System

A Robotic Exobiology Package to identify active metabolism could be included in the Mars Xpress payload allowing address of Extant Biology as a Scientific Aim, providing a range of development and operating criteria can be met.

C.7.2 Orbiter Instruments and Mapping

From an initial draft of an Orbiter Science Rationale and a total payload mass of 120 kg a payload for the orbiter comprises;

- High Resolution Stereoscopic Camera
- Wide Angled Stereoscopic Camera
- Planetary Fourier Spectrometer
- Long Wave Radar.

A requirement for a final sun-synchronous mapping orbit of height of between 100 and 500 km is specified primarily from the resolution and time of exposure requirements of the two cameras. The final orbit height for the mission of 600 km is a trade off between resolution performance for these instruments and increased Rover - Orbiter data relay capability. Due to mass constraints late in the study an additional instrument - a Gamma ray/ Neutron spectrometer has been removed from the payload.

C.7.3 Data Relay Strategy

The flow of data collected by the science instruments is managed by use of the Science Data Administration Spreadsheet. The relay and storage strategy adopted meets the Mission Science Timeline sampling and return of data requirements within the Rover communications Sub-system performance specification.

C.8 Communications, Command and Data Handling (Andrew Holmes)

This report centres on two major subsystems of the Mars Express Mission and indeed any space mission, namely the Communications subsystem and the Command and Data Handling subsystem.

This report presented an analytic approach to defining a Rover - Orbiter communications subsystem. Not considered was the Orbiter - Earth communications link. The results achieved consisted of a baseline system design with final dimensions, mass and power requirements.

C.8.1 Communications Subsystem

A Low mars orbit (LMO) of 600 km was decided as the final orbit for the communications relay orbiter. This height allows a substantial communications window to transfer a necessary quantity of data and does not demand huge amount of transmitter power.

The final selected link frequency was UHF (Ultra high Frequency). The rationale behind this is that UHF can be supported by a LGA (Low Gain Antenna) which allows:-

- A smaller mass for the antenna
- An omnidirectional beam. This provides a greater coverage and does not require any steering mechanisms.
- Simpler Design
- Tried and tested technology. UHF gas been extensively used in the past.

The chosen antenna classifications was a helix antenna. It allowed a higher gain, than other similar LGA's for a relatively smaller antenna size. This in turn necessitated the requirement for less transmitter power.

A Travelling Wave Tube Amplifier (TWTA) was chosen since it requires less inout power due to higher efficiencies.

Rover Orbiter Antenna dimensions	1m length
	4 cm diameter
	mass = 1.8 kg
Rover Transmitter Power	3.7 W
Orbiter Transmitter Power	0.1W
Power Amplfier Type	TWTA
	Mass = 2.5 kg
Amplifier Input Power	18W

Table C.20: Baseline antenna design.

This mast supports a transmit data rate of 128Kbps at a frequency of 0.4371 GHz and receive data rate 8Kbps at a frequency of 0.4015 GHz

C.8.2 Command and Data Handling Subsystem

The onboard command and Data handling subsystem will consist of a centralised state-of-the-art Reduced Instruction Set computer (RISC) built with radiation-hard technology. Several aspects of the CPU were considered including Command Processing which supports:-

- A decision making ability
- Capability to store commands
- Data Processing and storage

Also discussed was the Telecommand Packet format in which commands are received and the steps executed to validate, decode and execute the instruction.

The extent and size of commands received from Earth is highly influenced by the level of onboard rover software complexity and autonomy. A high level of autonomy has the following advantages.

- Onboard data-sensing and programmed decision-making process.
- Highly survivable. No need for communications window restrictions
- Fast response time, real-time calculations and operations
- Less human errors, from complex lines of command codes.
- Reduces ground equipment and costs

a coding and modulation technique was selected to encode all data to reduce bit error rates (BER). Reed-Solomon and Rate 1/2 convolution codes were concatenated together to provide a better overall performance code and lower error probabilities. The modulation technique used to reduce the amount of data sent/received was Quadrature Phase Shift Keying (QPSK)

Data transmission capacity:

39 Mbits (per pass) of science/payload data and 2 Mbits (per pass) of housekeeping data

Data reception capacity:

2.4 Mbits (per pass) of science / payload data and 0.1 Mbits (per pass) of housekeeping data

Note that there are two overhead Orbiter passes per Martian day.

C.9 Rover's Electrical Power System, Data Handling and Computer System (Panagiotidis Matakidis)

This report documents the work that was carried out for the Mars Xpress mission study, during the two terms of the academic year 1997-1998. It summarises the preliminary mission drivers along with the system-level issues considered to identify and evaluate the alternative mission concepts; it also describes the design constraints, options, and rationale behind major design decisions, for the baseline design of the following systems:

- Rover's Power/Electrical Subsystem
- Rover's Data Handling and Computer Subsystem

C.9.1 Mission Overview

The Mars Xpress mission is envisioned as a low cost, low mass robotic mission, with the aim to conduct scientific experiments on the surface of Mars. A synopsis of the Mars Xpress mission is shown in Table B.1.The main mission drivers were mass, mobility, landing accuracy, cost (major driver for launcher selection), subsurface investigation capability, and complexity/reliability.

C.9.2 Power/Electrical Baseline Design

The power/electrical subsystem will provide the electrical power to the rover during the launch, cruise, descent and the surface-operation phases of the mission, as required by the scientifc objectives. The most demanding power requirement is during the surface operations. Table C.21, summarises the power requirements for the different operating modes of the rover for surface operations.

	Power Requirements (Whr)	
MODE	Day	Night
Traverse day	256	106
Science day	294	106
Science night	317	217
Hibernation	254	104

Table C.21: Mission power requirements.

Different power configurations were examined and evaluated against the physical (mass, volume, power), environment (radiation, vibration, temperatures) and political constraints of the mission. Photovoltaic systems offer the best option considering the rover design constraints (mass, area, volume, thermal), reliability (flight history, cost), flexibility and performance degradation. Gallium Arsenide (GaAs) solar cells will be used as they provide high conversion efficiencies. Li-Ion secondary batteries have been selected because the cell technology is mature, and have been developed especially for the Martian environment. The regulated power bus topology offers flexibility, simpler electromagnetic control and optimisation of the solar array and battery designs.

The proposed system consists of a fixed GaAs solar array, mounted at the top of the rover, with a physical area of 1.6 m^2. The mass of the array is 2.40 kg (the array support structure mass is not included). A dust removal mechanism is incorporated. Two lithium-ion batteries provide the required night power. The total capacity of the batteries is15Ahr. The total weight of the two batteries including the regulators interfaces is 7.70 kg. A regulated bus voltage of 28V (0.5 V is provided using shunt and battery charge/discharge regulators). The power consumption for power distribution and switching is 1.5 W. The total mass of the power/electrical subsystem is estimated to be 17.6 kg.

The annual and diurnal variation of the power output of the solar array is shown in the following figures. Degradation of the solar array performance due to dust deposition, radiation and temperature has been considered.

The solar array, with clear atmospheric conditions (Tau=0.5) can meet all the requirements of the primary mission up to about sol 216 (Ls=100). After 216 sols, power from the energy storage (i.e. the batteries) must be used. At the beginning of the mission the solar array can provide the required power levels from 7:00 until 17:00 local solar time. The batteries will be used during the early morning communication link with the orbiter to assist the array to provide the peak power of 42W. The power output of the array is reduced towards the end of the primary mission.

During the local storm conditions (Tau=1.0) the power generated from the solar arrays can meet the power requirements of the primary mission, up to about the aerocentric longitude of 350° (175 sols after landing). The 1.6 m^2solar array can provide the power required for the hibernation mode even if a global storm occurs at winter solstice (it is assumed that the storm will result in an optical depth Tau =2.0), ensuring the survival of the rover.

C.9.3 Data Handling and Computer Design

A number of different architectures for the data handling and computer subsystems were investigated. The proposed design provides an integrated data handling computer system, where data handling capabilities will be provided centrally through the main computer, with remote capabilities at selected systems. This proposed design relies on a partly decentralised architecture to allow for some degree of modularity and reduction in the power consumption of the computer during the night period.

The estimated processing capability of the main computer is 1-5 MIPS with a 10MHz clock rate. This estimation is based largely on similarity sizing. The onboard computer provides random access memory (RAM), programmable read only memory (PROM) and volatile memory. It is assumed that 0.8 Mbytes are required for application and operating system software code storage . Allocation for payload programme code must be made, but no data are available to allow for a baseline approximation. A total memory allocation of at least 119.7 Mbytes is required. To allow for contingencies, for example failure to communicate with the orbiter for consecutive sols, additional memory must be provided to store the payload data. A total size of 500 Mbyte of volatile memory has been allocated to the payload data.

The computer will operate at the nominal clock frequency (about 10Mhz) during the day operations where the processing requirements are high. During the night, the main computer will be powered down to support only the fundamental functions for the survival of the rover, such as health monitoring, thermal and power control. The payload processing functions of data storage and limited data processing will be carried out by the remote processors which will communicate with the buffers without the intervention of the main computer.

Radiation hardened components will be used to minimise the effect of radiation on the operation of the electronic circuits and the software code.

The performance of the main computer is summarised in Table C.22.

Processing capability	10 MIPS	(better estimate is needed)
Scaleable clock rate	10 MHz	
Programmable Read Only Memory (PROM)	0.8 Mbytes	application and operating system software
	TBD	for payload program code
Volatile memory	500 Mbytes	for payload data
(including buffer memory)	TBD	for mission & housekeeping data
Estimated total volatile memory	500 Mbytes	
Power:	12.8 W	day-mode
	0.8 W	night-mode (power-down)
Mass	2.5 kg	
Volume (cm^3)	9 x 15 x 2	

Table C.22: Performance specification of the main computer.

C.10 A robotic exploration mission to Mars (Justin McGrath.)

C.10.1 Introduction

This project is based around the ideals of the European Space Agency, its main aim being to produce a study which parallels the ESA express mission to Mars.

The initial study objectives that defined the mission were:

- To land a scientific payload of 12.5 Kg on the surface of Mars using the lowest possible Lander mass ejected from a mother spacecraft following an optimal launch date in the year 2003.
- The mission will consist of an orbiter plus one or more landers.

In order to cover all parts of the mission, various workgroups were set up. This report is from the Lander subgroup and it covers the structural design of the Lander and Rover systems.

The Lander group drew the following objectives from the work that the science group produced.

C.10.2 Primary Objectives

The experiments to be carried out on Mars are very extensive, so it is fair to say that the Lander would have to carry a large amount of scientific equipment. It was also noted that to give an accurate representation of Mars, the samples would have to be taken from a number of different sites. This of course means that the Lander would need to have some mobility. In order to get untouched samples, some kind of digging device would also have to accompany the mission. This device would need to be capable of either burrowing metres beneath the soil or drilling into rock.

C.10.3 Secondary Objectives

These secondary objectives give us some more guidelines into the design of the Lander system. In both Meteorology and Geoscience there is an obvious need for a visual camera of some kind. This would have to take distant and close up images. In order to have a moveable science platform there would also have to be some kind of navigation imaging system.

C.10.4 Summary

So initially a brief outline for the Lander Group was developed, the following factors were required by the Lander:

- Mobility,
- Digging device such as a mole or drill,
- Cameras.

C.10.5 System Level Requirements

The system level needs were established for the rover based on the primary and secondary Lander objectives. They are that the vehicle must:

- Provide mobility over tens of kilometres,
- Provide a drilling mechanism,
- Provide a robotic arm,
- Carry out experiments on the Martian environment,
- Provide views of the Martian environment,
- Provide views of the Rover while operating on Mars,
- Safely interact with the Martian environment,
- Operate reliably for a period of at least 1 Martian year,
- Follow all societal expectations with relation to launch and operations.

The result of this process was that the Lander should Primarily consist of a single roving vehicle which would soft land inside Crater Lake. This "Rover" would carry all of the scientific instruments in order to carry out the scientific objectives.

The Lander itself would therefore be made up of a descent system and frame for the rover. It would also carry a descent camera to capture images of the landing area.

C.10.6 Configuration Breakdown

The Descent System

It uses 8 thrusters feeding from four fuel tanks. The landing gear consists of three legs which are stowed vertically and they deploy by pyrotechnics. This system compresses into a height of less than 40 cm.

The Rover

It carries all instrumentation. It is attached to the Lander by bolts which are cut at the appropriate time.

The Locomotion system

This is divided into four design elements: chassis and suspension, steering, wheel and clearance. These elements have been selected from several examples of relevant precedence including European, Soviet and American design.

- Our Mars rover employs a "rocker-bogie" suspension system. The rocker and bogie links will be made from thin walled aluminium alloy tubing welded to aluminium housings for pivots and actuators.

- A four wheeled explicit steering system is chosen for this rover, with the four corner wheels explicitly steered about their vertical centres and the middle two wheels fixed in a forward position. The most compelling argument for the chassis and suspension system design choice is its characteristic of passively maintaining traction on all of the wheels. This philosophy is extended to the steering system in that the four-wheeled explicitly steered vehicle maintains the best traction. The result is a vehicle with the best possible performance.

- An elastic mesh wheel of titanium alloy will be the "tire" of the rover. This mesh will be stiff enough to retain most of the stability of the Marsakhod or Lunakhod non-elastic wheels, but will provide some of the shock absorption and conformability useful for the Rocky rover. This mesh is attached to an aluminium alloy hub which is connected to the reduction unit and motor, with the single seal of the entire wheel assembly formed between the hub and the motor housing. Also connected to this mesh tire are several grousers providing an effective ground coverage of 2/3.

Mechanisms

Various mechanisms were required they include a drill, robot arm and mast.

- The Mini-Corer can obtain intact rock samples from up to 5 cm within boulders and bedrock.

- It can replace worn drill bits with new ones.

- It can position acquired rock cores where they can be examined by instruments on the Instrument Arm.

- This mission requires an arm which can be as manoeuvrable as possible. The choice for our mission is a six degree of freedom arm. Situated at the end are three instruments and a wire brush.

- The Athena mast covers most of our objectives so it will be used as a basis for the design of our rover mast.

- The rover will need two masts mechanisms, one as described above and the other to be used as an antenna. They are both mounted at the rear of the rover and spring open when released. This in turn releases the solar array panels.

- Three spring joints on each ramp would lower the ramps once they had been released.

C.10.7 Final Packaging

Figures 2.2 and 2.3 show the lander stowed for atmospheric descent and the rover deployed respectively .

C.11 Rover Thermal Design (Simon McKown)

This report forms one element of a group design project aimed at landing a robotic Rover on the surface of Mars in 2003 and primarily performing Exobiology experiments in the search for past or present life. The Rover subsystems are covered by workpackages making up the Lander group. This report covers the thermal subsystem of the Rover. Other aspects of the mission are covered by the appropriate group; Systems, Launch & Trajectory, Descent, and Payload & Operations.

C.11.1 Mission Summary

The Mars Xpress mission intends to make use of the favourable launch date 2003, as suggested by ESA's Mars Express mission . Studies by the Launch and Trajectory group concluded that a Soyuz 2 launch vehicle would be the best option (See work package 2010 by the Launch and Trajectory group). The primary science objective is to perform exobiology experiments in the search for past or present life. The lander system consists of a mobile vehicle called a Rover which will carry all the instruments. The descent system will use retrorockets for a soft landing. The landing site is targeted as Crater lake, which has a diameter of about 20km. This lies within Gusev crater. It is thought this site held water at some time in its past and so holds the change that sediments and organisms could be found. To increase the chance of finding organisms areas protected from U.V radiation will be probed. These areas are below the regolith or under the surface of rocks. This means a drill is required to penetrate to these depths and mobility is required to sample different sites. The primary mission phase is estimated at one Martian year. The secondary phase is for as long as the Rover survives. The mission breakdown is given in Table B.1.

C.11.2 Group Summary

The project was split into five groups

- Systems
- Launch and Trajectory
- Descent
- Lander
- Payload and Operations

Each member of the project produced work packages for a particular aspect of their group. For example the lander group was split into the various subsystems associated with a Rover vehicle on Mars. These included power, thermal, structural, and communication subsystems. The initial steps of the project where to establish a baseline design for the mission. This involved researching different concepts, such as types of descent systems, types of Rovers, launcher selection, etc. A trade off for the best combination of systems that could achieve the mission objectives was performed by the Systems group. The results concluded that a single large Rover could achieve good mobility and carry out the exobiology requirements of subsurface sampling.

C.11.3 The Thermal Subsystem

With the baseline established work on the Rover subsystems could begin. This report looked at the Rover thermal subsystem. If all the different components that constitute the Rover such as the electronics and batteries could be operated at the environment temperatures then there would be no need to consider a thermal design. Unfortunately components such as batteries have relatively small temperature ranges over which they can operate efficiently, whilst other components such as solar cells have operating ranges which span over 200K. Therefore the individual components must be kept within certain temperature limits which could be dependent on whether they are in the operating state or not. This applies throughout the diurnal temperature change and through the changing seasons on Mars. Given the operating temperature ranges of all the components, it may be sensible to group together ones with similar temperature ranges wherever possible in order to reduce the complexity of the thermal subsystem. This is achieved in the Rover design by designating a Warm Electronics Box, WEB, into which all electronic components such as computers, instrument electronics, communications, etc., are strategically placed. The batteries also occupy the WEB but have additional thermal considerations due to their sensitive operating requirements. Any component exterior to the WEB that cannot operate at the environment temperature has to have its own thermal design considerations. The thermal subsystem report works through the steps required to reach a simple model for determining worst case hot and cold scenarios. If the WEB can keep the components within operating or survival temperatures during the worst case scenarios then the Rover survival is assured, at least in thermal terms, at temperatures between the extremes. The report then discusses the heat dissipation by components and RHUs in the WEB. The insulation requirements and different types of insulation that are envisaged for use on the Rover are compared. Finally heater requirements for components such as batteries and motors are considered. The physical properties of the WEB are shown in the table;

WEB Construction Material	Kevlar composite Sandwich
Dimensions	0.5 x 0.5 x 0.4 m
Mass	3 kg
Insulation thickness	6.5cm SiO_2 Aerogel

Table C.23: Warm Electronics Box (WEB) characteristics.

The WEB is mounted on an Aluminium honeycomb baseplate which carries the structural loads. See the Lander group Rover structural design reports by P.Greenway and J.McGrath for details. A composite material was chosen to keep down the mass of the box, whilst the dimensions were restricted by the availability of space within the descent module. See the picture of the rover stored in the descent module in the report by J.McGrath. The height of the box, 0.4m, was restricted by the parachute storage and deployment system located above the Rover. However the dimensions turned out to be satisfactory when all the WEB components were in-situ. There is enough space within the WEB to accommodate wiring harnesses and any ancillary components that could be added at a later development stage of the project.

Element	Mass
Insulation	1 kg
Heaters, control electronics & harness	0.5 kg
RHUs	10g + casing (0.75 kg)
Kevlar WEB material	3 kg
Total	5 kg

Table C.24: Thermal subsystem mass breakdown.

C.11.4 Conclusion

Aerogel is shown to out perform the foam type insulators and can maintain the temperature of the WEB close to its operating limits when considering some extreme cases of hot and cold conditions on Mars. The baseline design for the WEB incorporates about 6.5cm thickness of Aerogel between its walls and a constant 5 Watt heat input independent of the battery and solar panel from 5 RHUs. Wheel motor heaters are baselined but may be left out at later development stage as low temperature motors are now available. Other heaters inside the WEB are essential for battery temperature control.

C.12 Volume/Mass Budgets, Planning/Risk/Cost (Bertrand Pabon)

C.12.1 Project Planning

This project run under a semi-collegial management process which fits this kind of project, which has not a high-level time constraints. The main stages in the study are:

- Preliminary design: each group study possible options separately

Leading to the final design process:

- Lander release strategy
- Final orbit/Communication
- Structural and Thermal design
- Final landing system design
- Power design
- Volume management/CAD

C.12.2 Volume Budget

The volume budget is linked to the mass budget because the bigger the volume is, the bigger atmospheric shield ballistic coefficient is, the bigger heat transfer becomes, the thicker the heatshield must be, and the heavier it will be.

Finally the compromise is a 2.0 m diameter and 1.3 m height maximum dimensions for the heatshield and so for the accommodated systems.

C.12.3 Mass Budget

The launcher capacity is 1200 kg.

The back-up strategy in terms of mass was the orbiter mass since a major part of the budget is dedicated to spacecraft manoeuvres. Hence reducing its mass reduces drastically the propellant involved. The trajectories options chosen were following a mass optimisation policy, to decrease propellant mass and so enable either more payload or redundant systems. The mass budget has been tighten by the choice of a soft landing system (107.5 kg).

Actually the initial payload baseline had to be revised since the mass involved in the descent system was huge and endangered mission feasibility . An instrument has been dropped from the orbiter following science priorities (from 121.6 to 101.6 kg). A rough design of the orbiter was done assuming that the payload ratio is 12%. That gives 418.8 kg for the orbiter bus.

The lander design assuming redundancies in some critical sub-systems (power) gives a total mass of 66.3 kg plus 45.5kg for the landing platform. The margins taken are 10% for the landing module and obviously 20% for the orbiter bus leading to a total margin of 105.7 kg. The propellant mass(326.3 kg) was calculated through an iteration process since the heavier the spacecraft is the more propellant you need for burns.

The injected mass is then 1,175.8 kg which fits the Soyuz2 launcher capability.

C.12.4 Risk Analysis

Our mission is a technology demonstration and science driven mission. Its risk level is then quite high.

The areas of lower reliability are the descent system because of the complexity of soft landing systems, the power sub system because it involves GaAs solar array which is quite a new technology even if proven.

Some design features are reliable but have a huge impact on the mission success if they fail. The launch window narrowed by the choice of a Moon gravity assist cannot be missed, obviously any failure on the launcher or the propulsion system during trajectory up to the swing-by would be fatal for the mission. The release of the lander is a very important step in the mission since no release means no mission at all.

The major redundancies included yet in the design are the power subsystem's and the final landing system (GNC) redundant systems.

C.12.5 AIT

The model philosophy don't go for protoflight because of no real need for it and lots of disadvantages.

The models involved are a STM, EM, FM. Attention will be paid to the cleanliness of the operations (clean room class 100,000/ethanol cleaning).

The instruments will have to be qualified before integration. A special attention is paid to release system, descent propulsion system. Mock-up of the release system and simulation of the descent and landing phase dynamics should be required.

A final attention is drawn on the RHUs use. It should be procured and stored in Russia and integrated on the Launch site for ecological and political reasons.

C.12.6 Cost Estimation

No parametric cost could be performed at this early stage of the design so a top-down costing has been chosen. The overall cost is 160 MAU which can be divided in:

- Launch 30 MAU
- Lander : 60 MAU
- Staff cost: 36 MAU
- Hardware cost: 24 MAU
- Orbiter : 70 MAU

It is assumed that the instruments will be provided by national agencies.

Prime contractors are used when high reliability is required, basically for the release system.

C.13 Mars Xpress and the Group Design Project (Olivier Pletinckx)

During approximately 6 months we worked together on the project of landing a scientific payload on the surface of mars. This project which will take place in 2003 is called 'Mars Xpress Mission'.

Following the interest of each, we split up the Masters in 4 different groups:

- Launch and Trajectory group
- Descent group
- Lander group
- Payload and operations group

plus a systems group in charge to manage the different budgets, and to identify the specific mission options.

After collaboration work between each group, we finally decided on the final choice for the mission. The Mars Xpress Mission will consist of:

- Launcher: Soyuz II (Molniya + Fregat) maximum mass of the spacecraft to mars orbit = 1200 kg
- Transfer: Moon gravity assist, elliptic transfer orbit
- Mars orbit acquisition: aerobraking, aerocapture and burns
- Descent: soft - landing - containing a shield for the entry into the mars atmosphere - drag increaser : disk - gap - band parachute - final landing system : retro - rockets - ⇒ Lander will be equipped of 3 legs for its ground stability -accuracy at landing : 25 - 30 km
- Lander : rover only with 6 wheels containing a protection in terms of thermal subsystem and a power subsystem equipped

C.13.1 The Descent System

According to my interest, I chose to take in charge the study of the Descent System: in collaboration with 2 other students from the MSc course. We decided to split the Descent group in 3 different sub-groups - group covering all the descent process :

- The atmospheric entry (SHIELD) charged to study the different parameters of entry
- The atmospheric descent (PARACHUTE) charged to study a means to increase the drag during the descent

The final landing system (RETRO - ROCKETS) charged to study the landing process on Mars; surface

C.13.2 The Atmospheric Descent

To prevent crash landing of the lander on the surface of mars, a drag increaser was necessary. Two types of drag increasers were studied according to the previous mission to Mars:

- The Inflatable Decelerator System which consists of a conical shape airbag was especially studied for a landing on Mars. This systems was capable to withstand up to 1425° C indefinitely, was configured to provide terminal velocities of 9 m/s to 18 m/s
- The Disk - Gap - Band parachute which is an High Altitude Decelerator system; usually used to provide a highly stable, high drag area system packed in a minimum volume. This kind of parachute consists of a flat circular disk and a cylindrical band separated vertically by an open space.

The final choice of the drag increaser was the Disk - Gap - Band parachute, already used for the Viking and pathfinder missions. The characteristics who decided me to choose this parachute were that it:

- can operate at supersonic speeds (M2 - 2,5)
- can operate at very low dynamic pressure, at high altitude (5- 11 km)
- is designed to have better stability than solid flat canopy

The discontinuity in the surface shape allow to control the flow of air exiting from the interior of the canopy, maximising the drag while maintaining the required degree of stability.

C.13.3 Design of the parachute

According to the mass budget available for the lander and the scientific payload, the total mass to take in account during the descent was approximately equal to 160 kg.

The characteristics of the atmospheric descent were defined as following :

- Parachute deployment at an altitude between 4 and 11 km, and a speed of M 2
- Aeroshell separation at an altitude between 2 and 9 km, and a speed > 60 m/s
- Firing of retro - rockets at an altitude of 2 km, and a speed of 60 m/s
- Parachute separation at an altitude of 2 km

Taking in account those different parameters at chute deployment and release, the parachute was designed as following:

The deployment system used to eject the parachute out of its bag is the MORTAR system. It consists of a tube of 23,5 cm of diameter and 39,6 cm of height, containing a sabot and an aft cover. The sabot acts as a piston to propel the bag out of the tube, producing a reaction force which must be directed through the vehicle centre of gravity.

DO nominal diameter of canopy	10.5 m
WC mass of canopy	3.3 kg
NG number of gores	32
F drag force	593.46 N
Q dynamic pressure at chute deployment	800 kg / m.s^2
Lsusp length of suspension lines	17.85 m
Wtot total mass (canopy + suspension lines)	12.2 kg

Table C.25: Parachute design characteristics.

The minimum ejection velocity for which the suspension lines remain in tension throughout the deployment process was estimated to be about 30 m/s.

The total weight of the parachute - mortar system is 17,56 kg.

To implement the overall descent process, a guidance and control subsystem was necessary. This system, strongly influenced by the general requirements to soft-land on the surface of Mars, is composed of:

- a Guidance, Control and Sequencing Computer
- a Flight Program
- a Inertial Reference Unit
- a Radar Altimeter
- a Terminal Descent and landing Radar
- a Valve Drive Amplifier

The power will be furnished by a Ni - Cd battery capable of many charge - discharge cycles.

C.14 Lunar Gravity Assist and Mars Aerobraking (Ridanto Poetro)

C.14.1 Lunar Gravity Assist

The Mars Express mission has constraint in cost which leads to using any possible means and options for succeeding the mission objectives. One of the options which is considered is using Lunar Gravity Assist to let the spacecraft gaining velocity while deflected in proper inclination at the interplanetary trajectory.

Some scenarios are analysed to obtain the best performance possible for this mission since each scenario has its own characteristics which in turn can only give advantages for certain mission.

Basically, to achieve maximum velocity gain from an planetary gravity assist, a spacecraft has to arrive from the opposite direction of the planet movement and departs with direction as close as possible with the planet movement. This way can give velocity gain about 2 times the planet velocity.

On the other hand, to be able highly deflected as required for achieving the maximum velocity gain, the spacecraft velocity passing the planet must be low enough, depends on the planet gravity attraction.

For this Lunar Gravity Assist case, both requirements are in conflict. The velocity requirement for the interplanetary trajectory needs such velocity which drives too high velocity passing the Moon too be deflected by Moon gravity attraction.

The scenario finally taken for this mission is just design the hyperbolic departure trajectory to pass the Moon at such distance to be able gain the minimum velocity while deflected to the required inclination for the interplanetary trajectory.

The result indicates that depend on the inclination requirement of the interplanetary trajectory, the Lunar gravity assist can be useful or not useful. It is stated that it can be useful if the hyperbolic excess velocity provided is bigger than one provided by not employing Lunar gravity assist and otherwise.

For the chosen launcher, the maximum possible performance can be achieved using Lunar gravity assist is hyperbolic excess velocity of 4.74 km/s with inclination of 2.39° .

The same condition for not using Lunar gravity assist, a maximum hyperbolic excess velocity of 4.443 km/s can be achieved.

So the maximum ΔV gain can be obtained is about 300 km/s.

It is clearly described that there are two areas of concern. For requirement of heliocentric trajectory inclination lower than about 2.7° the Lunar gravity assist can offer a bigger hyperbolic excess velocity. But on the other hand, for requirements of inclination more than 2.7° there is no point of using Lunar gravity assist.

However, the maximum advantage of 300 km/s can not be chosen directly, since the requirement of hyperbolic excess velocity and inclination are related with the interplanetary trajectory chosen. And the interplanetary trajectory design also depends on the launch window given by using this Lunar gravity assist. Therefore, all of them, hyperbolic excess velocity, inclination achieved, the launch window and the interplanetary trajectory must be analysed and chosen together.

The window where this scenario can be performed is wide enough actually, which is a range of Moon position approachig its last quarter position. But each finite segment of the window corresponds with different burn out velocity and off course correspond with different deflection or final inclination at the heliocentric trajectory. So, once an interplanetary trajectory with certain inclination requirement is taken, basically just an instantaneous window of the Moon position can be used to launch our spacecraft.

However, if a certain error can be tolerated, which effects can be corrected later, the instanteous window can be widen. For example, if a 250 km distance error is allowed, with Moon velocity of 0.99 km/s, a window of 4.2 minutes will appear.

C.14.2 Mars Aerobraking

After Mars orbit insertion (MOI), the spacecraft is captured in a high elliptic orbit of Mars. The intended final orbit is a circular near polar sun-syncronous low orbit to perform Mars mapping and as a relay station to send data from lander back to earth.

To achieve the final orbit from captured orbit after MOI, a negative ΔV must be given to reduce the energy of the captured orbit down to the final orbit. The ΔV can be obtained by burning fuel for the simplest way, but to save fuel budget and in turn spacecraft dry mass, aerobraking method is used.

Aerobraking is the utilisation of atmospheric drag on the spacecraft to reduce the energy of the orbit. The friction caused by the passage of the spacecraft through the atmosphere provides a velocity change at periapsis, which results in the lowering of the apoapsis altitude. The rate at which the apoapsis altitude decreases is determined by how much drag is generated and the resulting velocity change at periapsis. Going deeper into the atmosphere will provide greater drag and reduce the orbit faster, but will consequently generate higher spacecraft temperatures and dynamic pressures.

There are three phases in the aerobraking process, e.g. Walk-In, main-phase and Walk-Out.

Walk-In represents the initial phase of aerobraking where the periapsis altitude is lowered to the desired main-phase altitude. This step will be done gradually due to uncertainty in the atmospheric density model of Mars. Basically, periapsis altitude of 112 km is determined as the main-phase altitude. During those steps a more accurate measurement of the atmospheric density which will drive the necessity of the next step is performed.

The main-phase is where the majority of the orbital energy is removed by aero friction. However, use of small propulsive manuvers are needed at apoapsis to maintain periapsis within a well-defined periapsis altitude corridor, which is low enough to produce enough drag to reduce the orbit and high enough to avoid spacecraft heating limits and maximum allowable dynamic pressure to maintain control authority over aerodynamic torque. Due to oblateness of the Mars, the altitude of periapsis tends to rise during the main-phase, so the burn in this phase is to keep the spacecraft down at its periapsis.

When the desired altitude of apoapsis is achieved (nearly), a gradual manoeuvres is performed to raise the peripasis altitude to the final orbit altitude. This step is also helped by the aerodynamic lift generated.

Because of time needed to do aerobraking is quite long, the oblateness of Mars causes a significant changes in line of apsides and line of node of the orbit. Having had that, the Mars Orbit Insertion properties, particularly the longitude of the ascending node and the argument of periapsis have to designed to be at certain angles displaced from the values require for final orbit in which the values will reach exactly the values required at the end of aerobraking. Hence the final orbit properties will be achieved at the end of aerobraking phase.

However, in practice, where error can occur, there are more than one walk-in and walk out phases. For example if MOI line of node obtained slightly shifted from the intended value, the aerobraking time must be increased or decreased to maintain achieving the final orbit properties at the same time.

The aerobraking altitude of 112 km is taken following value as used by Mars Global Surveyor mission. This value is related with heating rate of 0.38 W/cm^{2}which is about 50 % of the maximum tolerable value 79 W/cm^2.

During aerobraking, the lander already starts to do science mission and requires the orbiter to relay the data obtained to the earth. Basically it can be done except when the orbiter dragging in the mars atmosphere, which is just between 12-15 minutes each orbit. The period of orbit starts from 45 hours to 2 hours. But due to limitation in power of the lander transmitter, most of the time of the high elliptic orbits can not be used either.

Here are the properties of capture orbit and the final orbit which must be achieved.

Capture Orbit	
orbit altitude	348.3 x 45086.71 km
inclination	93.48°
Final Orbit	
altitude	600 km, circular
inclination	93.48°

Table C.26: Mars capture and final orbit parameters.

Having had range of atmosphere altitude bigger than 100 km is concerned, an exponential model is developed to be approximation of Viking-2 Mars atmosphere data at that range.

Based on the atmosphere model, ΔV calculation due to atmospheric drag by treating orbit trajectory passing atmosphere between 112 km to 200 km as N finite segments is performed on three representative orbits. They are the first orbit, a middle orbit and the final orbit in main-phase of aerobraking.

Then, an equivalent atmosphere density is defined which can represent one orbit passing the atmosphere. The relation is as seen below.

$$\Delta V = \frac{\frac{1}{2}\rho_m V_p^2}{B}\Delta t_m$$

The braking time and equivalent atmosphere densities correspond with the ΔV values above can be obtained.

The equivalent density is then modelled as function of semi major axis of the orbit concerned to be able used for the rest of the orbit with minimum calculation.

Some representative orbits are presented below to show the dynamic of aerobraking.

Event	Days from MOI	Orbit Number	Periapsis/Apoapsis Altitude / km
MOI	0.0	1	348.3/45086
Walk-In Manoeuvre 1	1.5	2	348.3/45086
Walk-In Manoeuvre 2	3.0	3	150/45086
Walk-In Manoeuvre 3	4.5	4	133/45086
Walk-In Manoeuvre 4	6.0	5	124/45086
Walk-In Manoeuvre 5	7.5	6	118/45086
Walk-In Manoeuvre 6	9.0	7	113/45086
Start Aerobraking	10.5	8	112/45086
Orbit Period = 30 hr	38	28	112/39665
Orbit Period = 24 hr	73	60	112/33261
Orbit Period = 12 hr	170	198	112/18399
Orbit Period = 6 hr	253	433	112/9049
Orbit Period = 3 hr	306	729	112/3138
Walk-out	331	978	112/603
Circularisation	331	979	600/603
Final Orbit	332	980	600/600

Table C.27: Illustration of Mars aerobraking (orbit height).

Event	Days from MOI	Orbit Number	Regression of Nodes / deg	Rotation of Apsides / deg
MOI	0.0	1	0.00	0.00
Start Aerobraking	10.5	8	0.12	-0.96
Orbit Period = 30 hr	38	28	0.47	-3.79
Orbit Period = 24 hr	73	60	1.04	-8.41
Orbit Period = 12 hr	170	198	3.69	-29.82
Orbit Period = 6 hr	253	433	8.97	-72.48
Orbit Period = 3 hr	306	729	17.71	-143.06
Final Orbit	332	980	28.61	-231.15

Table C.28: Illustration of Mars aerobraking (orientation).

C.15 Interplanetary Transfer and Mars Orbit Capture (Xavier Pons)

C.15.1 Interplanetary Transfer Summary

Objectives:

- Assess possibility of using it to achieve the required velocity and inclination at the interplanetary trajectory with the capability of the launcher chosen (Soyuz-2)
- Find out the launch window to use it and in turn drive the possible interplanetary trajectory

Methodology:

- Analysis is based on patched conic approximation, which considers only 2-Body in each conic defined as sphere of influence of the bigger body between the two.
- Work out velocity and inclination (outcoming) required at the moon sphere of influence to achieve velocity and inclination of the interplanetary trajectory specified
- Work out a point and velocity (incoming) at the moon sphere of influence to be targeted from earth to be able end up with the above outcoming values.
- Work out the trajectory from the launch site to get to the point.
- Work out the launch window based on the trajectory time and date where moon at the required position.
- Any conflict between launch window available from Lunar Gravity Assist and the planned one leads to changes in the arrival date plan.

Inputs:

- S/C velocity at the beginning of the heliocentric trajectory
- inclination of the heliocentric trajectory to the ecliptic
- Earth velocity (w/r to the Sun) = 29.77 km/s
- Moon velocity (w/r to the Earth) = 0.99 km/s
- Max. Capability of the launcher (burn out velocity at 200 km altitude) Vbo = 11.834 km/s
- Gravitational parameter of the Earth, $\mu_E = 398600.4$ km^3/s^2
- Gravitational parameter of the Moon, $\mu_M = 4902.8$ km^3/s^2
- Moon mean radius, RM = 1783 km
- Earth Sphere of influence radius, SE = 900000 km
- Moon Sphere of influence radius, SM = 70000 km

Outputs:

- Launch Window : at time where Moon approaching its last quarter position (in front of the Earth), the exact time depends on the requirements of the interplanetary transfer chosen
- Maximum V_∞ and inclination for the Soyuz-2 capability either $V_\infty = 4.741$ km/s for i = 2.4° or $V_\infty = 3.080$ km/s for i = 4°

C.15.2 Mars Aerobraking Summary Sheet

Objectives:

- Find out how much ΔV can be saved
- Find out what range of time will be used and its effect on line of nodes and line of apsides.

Methodology:

- State some approximation values of the S/C and Mars atmospheric properties.
- State the scenarios
- Calculate successive changes in orbit properties due to atmospheric drag and control burn from initial capture orbit to the final orbit
- Sum out the final ΔV and time required and also the regression of nodes and rotation of line of apsides appear.

Inputs:

- Gravitational parameter of the Mars, μ = 42828.3 km3/s2
- Zonal Coefficient of Mars, J2 = 0.001964
- Capture orbit properties, orbit altitude: 348.3 x 45086.71 km (altitude) inclination : 93.48°
- Final Orbit Properties, circular orbit, 600 km altitude, 93.48° inclination
- Atmospheric Density, a model based on viking-2 data
- Ballistic Coefficient, B = 62.18 kg/m^2
- aerobraking altitude (periapsis), h = 112 km

Outputs:

- ΔV burn required = 118.78 m/s
- ΔV saving (compare to w/o aerobraking) = 1106.2 m/s
- time required, t = 7938.4 hours (11 months)
- regression of nodes, = 28.65°
- rotation of line of apsides, = -231.53°

C.15.3 Lander Release

The principal aim of the study has been to analyse and attain a fully detailed design of the lander release. The separation of the spacecraft and the lander has to be optimised with respect to the landing accuracy within the constraints of the descent system and final orbit design.

The release scenario to be chosen influences directly the orbiter attitude and deployment at insertion into Mars' orbit, but also the lander design and its descent system. Therefore the release strategy has to be in agreement with the aerocapture, the final near polar orbit and the descent strategy of the lander. Those latter's are limiting parameters that leads the team to decide on a specific release method.

The particular necessary inputs are the following:

- Lander and spacecraft interface
- Landing site location
- Velocity at arrival
- Spacecraft mass breakdown
- Aerobraking orbit features
- Final orbit design

The work undertaken has been separated under different activities that will enable one to select the most suitable option for the entire mission.

A first appraisal of the requirement of the strategy shows that in order to optimise the landing accuracy the arrival of the lander must be within a plane parallel to Mars equatorial plane. In the other hand the orbiter should be inserted into a near polar orbit plane. Therefore the chosen strategy is to separate the two bodies before entering into Mars sphere of influence. The capability of such an option has been analysed in terms of velocity and direction, but alternative designs with different improvement lead to the final decision.

The best deployment strategy requires a burn from the spacecraft propulsion system in order to slow down both the orbiter and the lander from 10kms^{-1}to 9kms^{-1}. The slowdown occurs approximately 80,000 km before the moment when the spacecraft will experience any gravity influence from Mars (580,000 km away from Mars). Then a set of pyrotechnics nuts coupled with mechanic springs is instantly fired in order to deviate the lander into the equatorial plane and let the orbiter on a trajectory that target a point 7000 km above the North Pole of Mars.

Both the lander and the orbiter approach Mars' atmosphere with a hyperbolic trajectory. The aerocapture and the atmospheric entry workpackages state the angles at which each module enters the atmosphere. The orbiter will arrive into its first orbit around Mars approximately 10 hours before the soft landing of the rover. This will enable the first communication as soon as the lander enters the Martian atmosphere.

C.15.4 Interplanetary Transfer

There are many ways of launching a spacecraft into an interplanetary exploration mission. The study of the different transfer orbits has been achieved in order to gather all the information necessary for an appropriate selection.

The evaluation of the best option is made in respect with the primary drivers of the mission. Those are the competitiveness and efficiency of the mission. The aim is therefore to put the maximum payload at the vicinity of Mars for the lowest cost using the most reliable transfer orbit. The journey time and arrival date are some of the parameters that has to be provided by the final study.

The analysis is closely related with the strategy to escape from Earth sphere of influence: launch selection, parking orbit and moon gravity assist. It also rely upon astronomical data for the specific year and inputs given by the other team of the project:

- Availability of the launcher vehicle
- Velocity increments available from the launcher
- Velocity and plane change provided by the Moon gravity assist
- Solar System configuration along the year 2003
- Preferred arrival date

The study required a fully detailed modelisation of the Earth and Mars orbit, as well as their relative motion within the Solar System. An estimation of the low energy transfer opportunity dictates the best launch window during the year. As a first approximation the Hohmann transfer has been calculated that enables to compare any other possible transfer to the most efficient one. All the possibilities were assessed from their performances, their ΔV required, their reliability and accuracy.

Finally an elliptical transfer fulfil the most the mission requirements. This trajectory has a particular transfer time of 130 days and is to be launched between the 24th May and 1st June 2003. This orbit has been designed in order to use the full capability of the launcher and take advantage of the opportunity of low energy transfer in 2003.

C.15.5 Orbiter Application

Communication

One of the primary objectives of the orbiter is to perform all the communication with the lander. In order to achieve a sufficient data communication and handling the orbiter has to be put into a particular orbit where it will receive and transmit all the measurement results back to Earth. The final orbit can either be circular or elliptical, equatorial or polar, geostationary or sunsynchronous. All the requirements and dependencies of this orbit have to be analysed in order to choose the final design from all the possible options.

The orbit design in term of communication is one of the part of the final orbit study because the orbiter is mainly used for science and mapping.

This study rely upon different parts of the mission design, including the following:

- Lander design and instrumentation (antenna)
- Power allocation
- Final orbit acquirement strategy
- Lander release
- Final orbit investigation: Science and Observation
- Orbiter design
- Landing site

The final trade-off demands to analyse the communication time available per day, the power needed from the lander and the orbiter to achieve the communication requirements and the parameters to put the spacecraft in sunsynchronous orbit.

The best orbit in term of communication has been decided to be a near polar orbit. It will have 92° inclination that enable the orbiter to constantly face the sun and therefore receive the maximum power from its solar panels, and will hover above the planet at 600 km altitude. The 12 minutes communication per day will occur in two passages, one early morning and the other late evening. This orbit has an 80Mbits maximum data rate and 23.3m mapping resolution on the ground allowing 5 days for coverage. This decision satisfies the science and observation requirement and has been evaluated as being energy efficient.

Finally the orbiter is design to last at least one Martian year and must support the lander in its communication process during all the duration of the mission.

C.16 Mars Atmospheric Entry (Anthony Rossignol)

The executive summary presents the work accomplished by the whole Group Design project, focusing on the descent group and the Atmospheric entry to show how this work contributed to the whole mission.

C.16.1 Mars Xpress Mission

As first stated, our Mars Xpress mission was to design a mission to mars matching the same requirements as the ESA Mars Express mission, that was to land 12.5 kg scientific payload on the Mars surface, at the minimum cost and time. The main objective was the science and exobiology in particular.

Some weeks were spent before a satisfactory mission outline was found. At the beginning, several lander types have been considered, like penetrators, semi-soft and soft landers, or a combination of these. A final design with only one lander featuring a river was chosen, and a working method was set for the class.

It was decided to split the class into five groups, according to everyone's particular interest, in order to fulfill the different missions steps;

- Launch and trajectory
- Descent
- Lander design
- Science and operations
- System to co-ordinate the work effort

It is to note that the idea of a system group greatly eased communication and trade-offs between the different groups, allowing the project to be managed as a real one.

As our working group is a student one and out mission not subject to real cost imperative, the mission driver was shifted from COST to SCIENCE, which would allow for more flexible work and individual expression. Then the decision was made to use a soft lander for its qualities compared to semi-soft or hard lander, still trying to keep the cost aspect in mind.

C.16.2 The Descent System

The descent system comes between the Launch and Trajectory and the Lander design. Even if this part occupies only a small time in the whole mission life, this is a crucial event on which rely the life of the spacecraft. The descent system has 3 specific areas:

- Atmospheric entry
- Parachute descent
- Landing

As we were only 3 students in charge of this work, we chose to logically split the work, one person doing one part. It is to notice that this split worked very well, each of trying its best to do his work accordingly to the other's requirement. In that way, the descent part was quite solid and coherent.

Furthermore, because of this importance, the descent system was very important in the whole mission design, and had more than any other group to make many trade-offs and design with the other teams.

C.16.3 The Atmospheric Entry

Introduction

This part of the descent group, chronologically the first one to occur, tackles the issue of Atmospheric entry into the thin Mars atmosphere.

Goals

The goal of this part is to propose for the whole project, a relevant solution to the atmospheric problem, that is:

- Choosing the right entry trajectory
- Designing a shield to protect the lander during the entry

The problems for an atmospheric entry were that the spacecraft as to withstand incredibly devastating phenomena, such as acceleration and heating. The main requirements in the design were hence the following:

- Withstand very important deceleration
- Withstand tremendous heating of the lander and shield body
- Carefully designing the trajectory in accordance with Mars entry corridor
- Allow for correct parachute opening conditions, in terms of speed, altitude, stability and attitude
- Allow for a good ground precision, according to landing site requirements

Working Method

It was decided to do a design as much as possible specific to our Mars Xpress mission, involving heavy calculations. This was made, at least for dynamics calculation. Other sources of pre-calculated information were used and adapted as closely as possible to our mission design, specially for the heating calculations.

Final Solution

A final solution in accordance with mission requirements was found, and is presented in the table below:

Trajectory design	
Entry type	Direct from interplanetary path
Entry method	Ballistic
Entry velocity	7.6 km/s
Entry angle	15°
Entry altitude	150 km
Duration	3 minutes
Peak G-load	20 g
Aeroshell peak heating	2000° C
Precision landing	50 km
Parachute opening altitude	8 km
Parachute opening	Mach 2
Shield design	
Diameter	2 m
Cone Angle	70°
Height	1.3 m
Ballistic coefficient	40 kg/m^2
Drag coefficient	1.6
Surface material	SLA 561
Structure	Aluminium honeycomb
Mass	38 kg

Table C.29: Mars orbit insertion parameters.