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NASDA Technical Memorandum

Feasibility Study on Lunar and Mars Exploration

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Feasibility Study on Lunar and Mars Exploration

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Contents

Preface

Space development for lunar and Mars exploitation

This technical memorandum summarizes the results of in-house study on lunar and Mars drone explorations - observation, landing and mobile explorations and sample returns for lunar and Mars respectively. So far, lunar and planet explorations have been primarily performed by the United States and the Soviet Union. ISAS and ESA have also contributed to some extent. The main purpose has been scientific exploration. There are some arguments that lunar and planet explorations should be performed for scientific purposes and the exploitation of them is not necessary. However, most scientific researches involve the existence and survival of humankind, so it is not the fact that they cannot be organized from the side of exploitation. Especially, if NASDA makes approaches to lunar and Mars exploration, it should inevitably embrace exploitation plans. In this preface we provide the outline of lunar and Mars exploitation scenarios set up as premise of the review on lunar and Mars unmanned exploration plans.

Various reviews have been performed on whether lunar or Mars would allow for human activities or survival. Among them, ^3He mill, solar powered satellite material mill and construction project of relay station to Mars as well as Mars teraforming plan have important issues. These projects have not yet become feasible because their expected investments are too large to make them practical. However, the present time seems the most appropriate to get with lunar and Mars exploitation projects under international cooperation since the realization of space station is imminent and the international cooperation is being created with the participation of Russia.

The international space station project will be continued until the year of 2015. The post project has not yet been decided. Therefore, we expect that Japan would propose two projects as successive ones - one is to construct an orbital service station combining manned abilities of the station and orbital service system and the other to build a manned lunar base taking almost all functions of space station onto the lunar surface. The mission of lunar base is to perform experiments for lunar resource exploitation. Fig. 1 shows the image of the entire space development plan to be proposed by Japan based upon this concept. The following description outlines each item:

1. Manned lunar base

For the time, assuming that five people or so would stay only in the daytime (14 days), the manned facility of space station would be available without large change. For transportation system, the transportation facility up to LEO could be the same as the space station with only exception of OTV and lunar landing/takeoff aircraft development. Further international cooperation should be promoted following the space station project.

2. Orbital system

The orbital service center should be built combining manned abilities of space station and unmanned functions developed in the orbital service system. Applications would be a transportation relay center to the above-mentioned lunar base and the operation of free flyer group in orbital experiments/mills. The relay center would also be necessary for lunar revolution orbit.

3. Stationary orbit platform

When the orbital service center and manned OTV are created, maintenance services can be provided for stationary orbit platform. Therefore, conventional type satellites would operate as platforms and become permanently available. Also, aged platforms could be recovered to prevent debris.

4. Transportation system

For transportation system, five types - LEO freight, LEO manned, OTV freight, OTV manned and lunar landing types - would be required. LEO freight is the transport aircraft from the earth to LEO. Lunar landing type is recycled propulsion system to be used for landing connected with manned cabin or freight container.

5. Mars exploration

If we try to perform Mars sample return in one blast-off from the earth, it will be necessary that LEO transport aircraft has huge transportation capacity. Therefore, it seems feasible to make blast-off material several times and build a Mars sample return aircraft in the orbital service center.

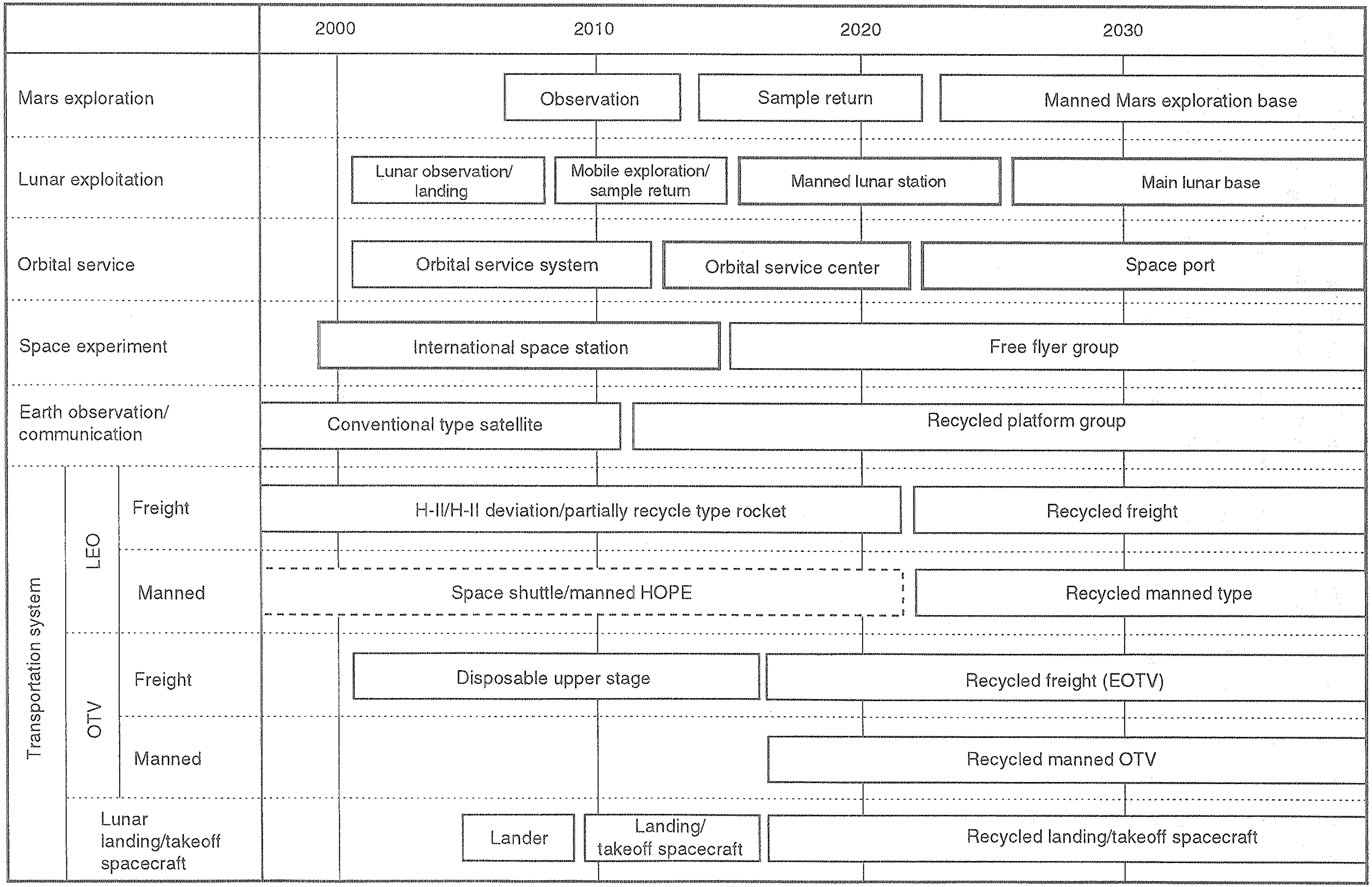


Fig. 1 Entire space development project with premise of lunar and Mars exploitation

The study in this memorandum is based upon the above-mentioned lunar and Mars exploitation project, however, it is also the fact that lunar and Mars exploitation procedures have been established through this study. As the nature of exploration planning, changes of exploitation project would not remarkably reduce the validity. Nevertheless, establishment of successive projects in early stage would help effective planning of exploration as a whole without fail. Study on post-projects of manned lunar base, orbital service system and future transportation system are scheduled and these reports will be provided as soon as possible.

1. Review on mission request

First, we would like to summarize the mission requirement for space craft based upon the accomplishment of past exploring aircraft as well as on-going projects (see attached information sheets 1 & 2) for lunar and Mars explorations, respectively.

1.1 Lunar exploration

(1) Exploration for resource exploitation and base construction

In order to help effective lunar development including lunar resource exploitation and base construction, data acquisition according to the procedure in fig. 1-1 should be performed as a lunar exploration mission.

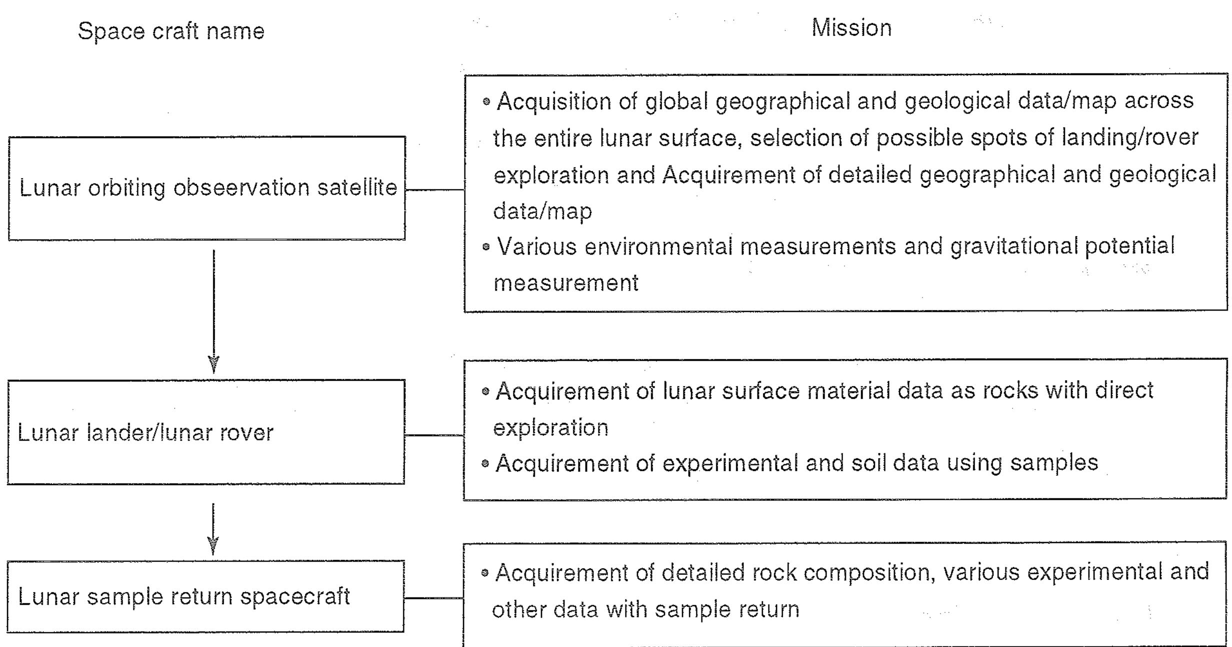


Fig. 1.1-1 Steps of unmanned lunar exploration mission

The following description indicates the mission requirement for each space exploration aircraft for lunar resource exploitation and base construction.[1]

(a) Lunar orbiting observation satellite

1) Creation of geological maps & mineral composition profiles as well as contour maps across the entire lunar surface with the spatial resolution of 10m (equivalent to that of remote sensing data on the earth) for the layout of lunar resource exploitation projects and base construction

2) Creation of element composition profile across the entire lunar surface for the layout of lunar resource exploitation projects

3) Survey of underground structure including regolith thickness required for the layout of lunar resource exploitation projects and base construction

4) Creation of digital geographical data and contour map with the spatial resolution of 1m or more (this resolution allows the identification of rocks) for landing points and exploration routes of lunar mobile exploration aircraft

5) Environmental measurement of radiation, solar wind, magnetic attraction and gravitational field, etc. required for practical lunar activities

(b) Electric propulsion orbit transfer mission spacecraft

1) Environmental measurement of long time and earth-moon wide area radiation, solar wind, magnetic attraction and gravitational field, etc. required for practical lunar activities

2) Data relay between lunar observation satellites and the earth for gravitational potential measurement

(c) Lunar lander/lunar rover

1) Composition observation and resource survey with direct observation of rock grains for the layout of lunar resource exploitation projects

2) Correlation of direct observation data with remote sensing data for the resource exploration across the entire lunar surface

3) Execution of the above-mentioned 1) & 2) activities for correlation with the exploration of back surface and polar areas geographically different from front surface

4) Execution of regolith heating and ^3He extraction experiments for the layout of lunar resource exploitation projects

5) Acquirement of lunar soil data for base construction

6) Environmental measurement of radiation, solar wind, magnetic attraction and gravitational field, etc. required for practical lunar activities

(d) Lunar sample return mission spacecraft

1) Sample return to the earth requiring detailed research for the layout of lunar resource exploitation projects

(2) Scientific exploration

Mission requests for scientific exploration vary depending on individual space crafts. Main requests are listed as follows.(2)

(a) Lunar observation satellite, electric propulsion orbit transfer mission spacecraft

1) Acquisition of scientific composition, magnetism and core data including the abundance of non-volatile elements, abundance ratio of ferrum-magnesium and abundance of metals for researching the lunar origin

2) Measurement of moonquakes, gravitational field, surface material composition and particulate indicating the lunar origin for researching lunar crust & mantle structures and the evolution

3) Measurement of moonquakes and gravitational field for analyzing the lunar core

4) Measurement of temperature and gravitational field as well as geographical observation for researching the thermal history of the moon

5) Measurement of magnetic field for researching the origin of lunar magnetism

(b) Lunar lander/rover

1) Confirmation of the fact reasoned out from remote sensing observation data

2) Detailed survey in the following regions:

- Regions suggesting water existence
- Regions in which lunar mantle appears at the bottom of craters, etc.
- Regions in which crust cross sections crop out on the surface
- Regions which seem to be the oldest lunar crust
- Volcanoes in crust forming era
- Regions still blowing

(c) Lunar sample return mission spacecraft

1) Taking material to the earth with scientific attraction such as newly discovered material and dating samples

(3) Exploitation and scientific exploration

Fig. 1.1-2 shows the mission request for observation satellites as the first stage of lunar exploration and the required observation equipment. As the figure indicates, there are many common missions and exploitage explorations greatly contribute to scientific researches including the lunar origin study.

[Reference]

(1) RESTEC, S 63 Report on the result of NASDA commission tasks "Survey on the remote sensing mission for lunar and planet exploitation": RESTEC, 1989

(2) ISAS, Lunar exploration mission planning: ISAS, 1987

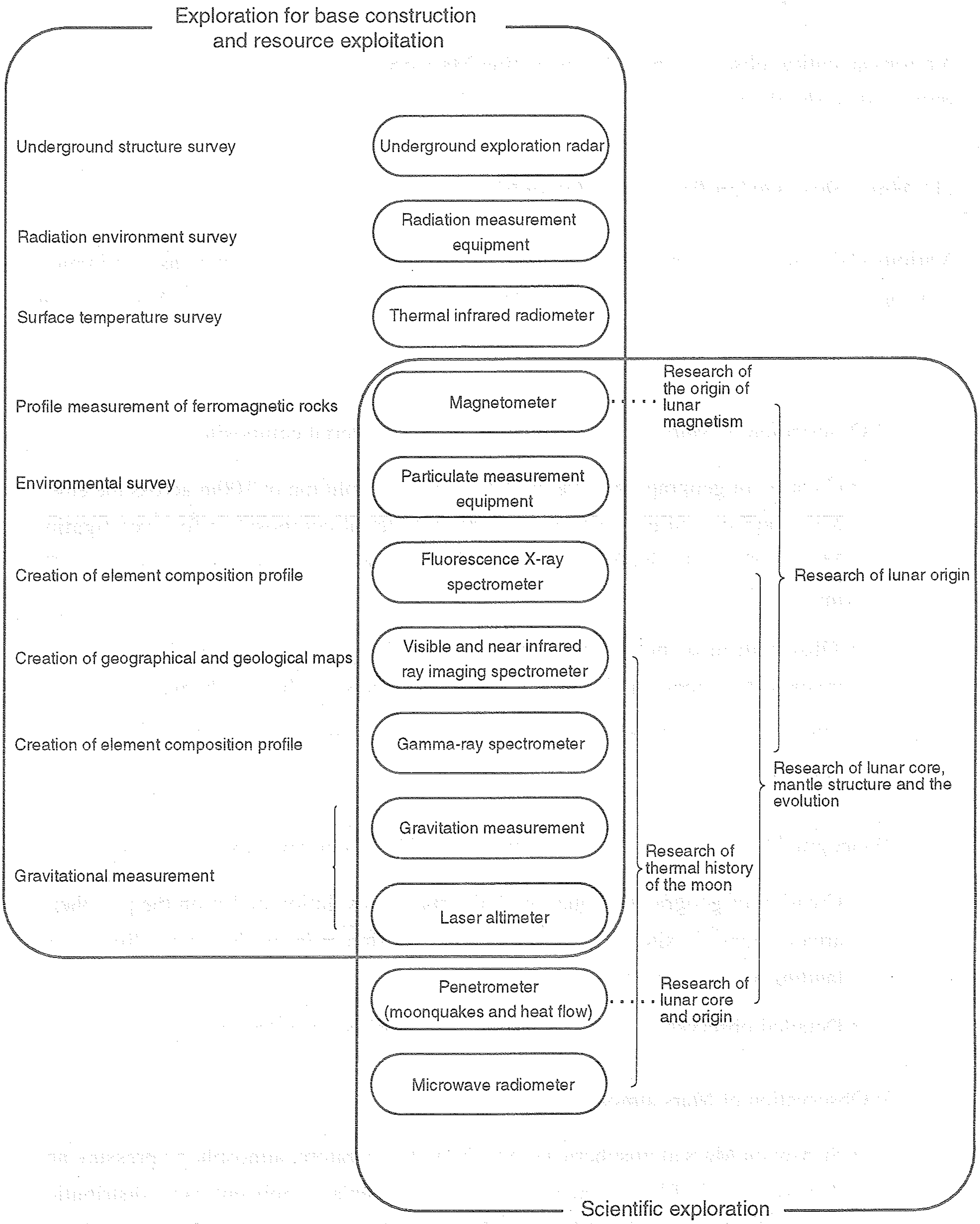


Fig. 1.1-2 Lunar remote sensing exploration items and required observation equipment

1.2 Mars exploration

Various scientific observation for investigating Mars origin, history (evolution process) and present state should be performed with exploration aircraft.

(1) Mars observation from circular orbit

Various global scientific observations of Mars surface and atmosphere as well as acquirement of geographical data of possible rover landing plans should be performed with Mars orbiting observation spacecraft.

1) Observation of Mars surface geography/surface material composition

- Creation of geographical map with the spatial resolution of 100m across the entire Mars surface and observation of surface material composition for investigating Mars surface's geographical profile, geological evolution and existence of volatile material
- Observation of polar cap's ingredient, thickness and seasonal changes for investigating polar cap forming era and the effects on climate changes
- Observation of thermal infrared radiation for investigating the weather and volcanism

2) Geographical and other observations on possible lander/rover exploration spots

- Creation of geographical map with the spatial resolution of 1m on the peripheral area of possible sites (20~40 km square) for determining the possibility of soft landing and movement
- Detailed observation on each possible site required for selection

3) Observation of Mars atmosphere

- Survey on Mars atmospheric composition, temperature, atmospheric pressure and their vertical profiles, cloud and storm observation and moisture vapor distribution survey for investigating Mars weather/climate changes, atmospheric circulating system and the existence of volatile gases

4) Measurement of charged particle environment around Mars

- Observation of energy distribution of charged particles for investigating Mars magnetic field

(2) Mars lander/rover exploration

Detailed and complex observational researches should be performed with Mars landing/mobile exploration aircraft on the exploration area selected according to Mars orbiting observation satellite data

1) Mars weather/climate change observation

- Surface environment (temperature, pressure, composition, wind direction/speed) observation for investigating dynamics of Mars weather/climate changes

2) Survey on vertical profile of Mars atmosphere

- Observation of temperature, pressure and composition changes during down landing for investigating the vertical profile of Mars atmosphere
- Execution of detailed analysis of atmospheric composition (rare gas measurement)

3) Material composition observation on Mars surface

- Analysis of soil or rock's element/mineral composition with several sites among landing spots and moving paths for detailed investigation of surface material distribution
- The analyzed data should be used for the compensation of orbiting observation aircraft data

4) Mars quake observation

- Execution of quake observation for investigating inner structure and the presence of quake activities

5) Life existence survey on Mars

- Organic qualitative/quantitative analyses for investigating detailed process of organic chemistry evolution (generation, stability and corruption) on Mars
- Execution of various biological experiments (photosynthesis and metabolism reactions) for investigating the existence of lives on Mars

6) Acquirement of Mars surface images

- Acquirement of surface image data for observing the surface geography and the evolution process, weather/climate changes and polar cap evolution process

(3) Mars sample return

Sample recovery with Mars sample return aircraft and detailed analysis investigation on earth should be performed. Possible return samples are listed as follows:

1) Soil/rock samples

- Investigation of Mars geological evolution and hydrosphere through detailed analysis of element and mineral compositions
- Confirmation of SNC meteorite origin by performing the composition comparison with SNC meteorite(*) which is considered to be derived from Mars

2) Lives or trace of lives

- Life existence investigation by detailed analysis of samples

3) Polar cap ice

- Investigation of polar cap forming era and evolution by detailed analysis

4) Mars atmosphere

- Rare gas ingredient qualitative/quantitative analyses by detailed method

(*) SNC (Shagotti, Nakura, Cassini) meteorite:

This meteorite is clearly different from others in that its data such as element composition, isotope composition and dating data can be grouped. Gas ingredients trapped within the meteorite can be analyzed as compound of Mars ingredients serveyed by Viking and earth atmospheric ingredients. In addition, characteristics of formed minerals are similar to those of Viking data. Therefore, it is considered to be derived from Mars. (The meteorite is supposed to be broken out from Mars due to meteorite collision into Mars.)

[Reference]

(1) RESTEC, S 62 Report on the result of NASDA commission tasks "Survey on the remote sensing technique for lunar and planet resource exploitation": MRC88-331, March, 1988

(2) RESTEC, S 63 Report on the result of NASDA commission tasks "Survey on the remote sensing mission for lunar and planet exploitation": MRC89-437, March, 1989

(3) RESTEC, H 1 Report on the result of NASDA commission tasks "Survey on the remote sensing mission for lunar and planet exploitation (Second issue)": March, 1990

(4) RESTEC, H 1 Report on the result of NASDA commission tasks "Survey on the remote sensing mission for lunar and planet exploitation (Third issue)": March, 1991

(5) RESTEC, H 1 Report on the result of NASDA commission tasks "Survey on the remote sensing mission for lunar and planet exploitation (Fourth issue)": March, 1992

(6) RESTEC, H 1 Report on the result of NASDA commission tasks "Survey on the remote sensing mission for lunar and planet exploitation (Fifth issue)": March, 1993

(7) "Geochemistry of Mars" by Togashi (Investigation committee documents on observation analysis technique for lunar and planet exploitation): February, 1994

2. Study on lunar observation plans

2.1 Lunar observation scenario

Sufficient observation data should be collected using lunar observation satellites and electric propulsion orbit transfer mission aircraft (See fig. 2.1-1). Especially, electric propulsion orbit transfer mission aircraft should perform relay functions for measuring gravitational potential on lunar back surface and should acquire space environmental data including wide-area and long-time radiation and magnetic field data from the earth to the moon.

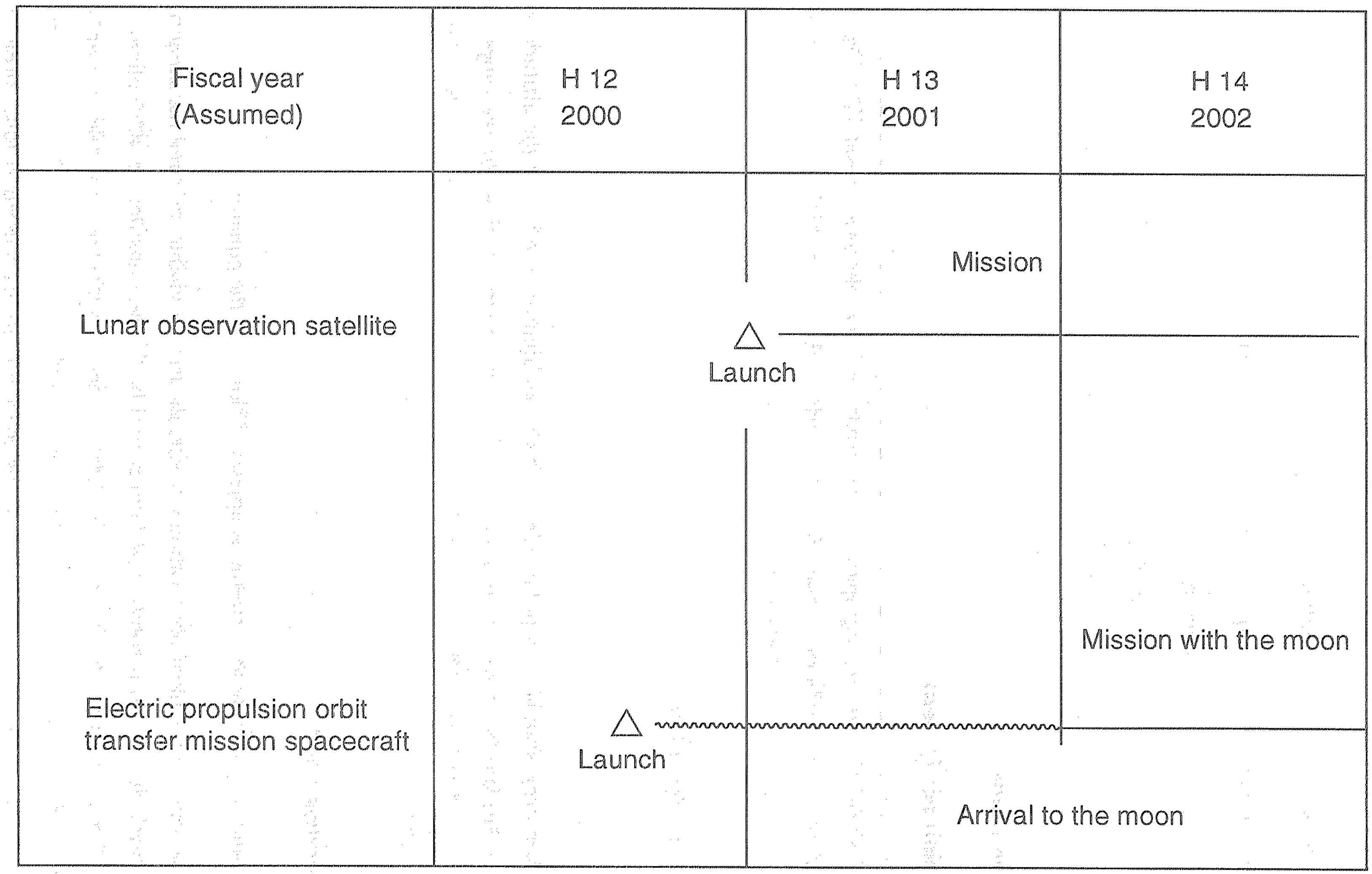


Fig. 2.1-1 Lunar observation scenario

2.2 Study on lunar observation satellite

2.2.1 Objective

Objectives of lunar observation satellite are as follows:

1) Resource survey and mapping across the entire lunar surface

2) Acquirement of lunar environmental data

3) Acquirement of scientific data on the moon

4) Mastering of lunar orbit entry technique as well as lunar remote sensing technique

2.2.2 Mission plans

2.2.2.1 Mission equipment

Table 2.2. 2-1 shows the mission equipment (proposed) for lunar observation satellite. Direst lunar surface observation can be performed due to low orbital altitude and high spatial resolution of sensor enabled by no atmospheric condition on the moon.

2.2.2.2 Observation plans

(1) Orbit

The observation orbit should be a polar orbit with the inclination of 85° and the altitude of 100km. This value has been set taking account of lunar revolving orbit's change (approx. ±30km per orbital altitude of 100km) affected by gravitational potential. In this case, the orbital cycle is 118 minutes.

(2) Mission management

Intraorbit distance adjacent to the satellite is approx. 32km near the equator.

Among mission devices for lunar observation satellite, the trim of visible infrared ray imaging spectrometer with the spatial resolution of 10m would be 40km, considering 4096 element CCD is used. Therefore, the establishment of three data reception earth stations would constantly allow the visual observation of the moon. As a result, in the shortest case, the entire lunar surface observation can be performed in one month required for one circuit of lunar orbit by satellite. In case of one earth station, the period would be 3~6 months. After the entire surface observation, local area observation required for lunar landing/mobile exploration aircraft should be performed with the spatial resolution of 1m.

The fluorescence X-ray spectrometer with the shortest trim of 10km would allow the entire lunar surface observation in four months, however, the measurement accuracy should be improved by performing the same spot observation several times and extending observation time.

In the latter part of mission period, gravitational potential measurement should be performed for about six months. Accordingly, the mission period should be two years.

(3) Data transmission

The period when a observation satellite goes to the lunar back surface and becomes invisible is 48 minutes max. The observation data during this time should be logged by recorder and transmitted in the visible time. The storage data capacity is 86.4G bits. Assuming the earth station's performance as DSN level of NASA, data transmission with the rate of 60Mbps would be possible at 20W output using high gain antenna with the diameter of 0.8m. The line design table is included in attached information 3.

Table 2.2.2-1 Lunar observation satellite equipped sensors (proposed)

Sensor name		Observation purpose	Main characteristics (proposed)	Weight	Power	Data capacity
Visible shortwave infrared ray imaging spectrometer		Stereo mapping with spatial resolution of 10m, 1m across the entire lunar surface; creation of geographic and mineral composition maps	Number of low spatial resolution bands: 8 Observation wavelength: 0.4~2.0 μm High spatial resolution band: panchromatic	200kg	300W	30Mbps
Fluorescence X-ray (*) spectrometer		Creation of major element composition profile across the entire lunar surface	Detection range: 0.5~10 keV Spatial resolution: 10km	15kg	20 W	1Kbps
Gamma-ray (*) spectrometer		Creation of isotope element composition profile across the entire lunar surface	Detection range: 0.1~3 MeV Spatial resolution: 60km	30kg	10 W	2.5Kbps
Geographic and underground exploration radar		Mapping across the entire lunar surface including polar caps and underground structure	Frequency L band: VHF and others Spatial resolution: 20m	150kg	500 W	30Mbps
Laser altimeter (lider)		Measurement of orbital altitude, sensor calibration	Spatial resolution: 10m Altitude resolution: 1m	50kg	150 W	2Kbps
Gravitational potential measurement		Gravitational potential measurement with orbit determination	Data relay with electric propulsion lunar revolving aircraft	—	—	—
TEDA	Radiation measurement device	Acquirement of radiation environmental data on lunar revolving orbit	Space radiation, solar wind and other radiation measurement	25kg	40 W	100bps
TEDA	Magnetometer	Acquirement of magnetic data on lunar revolving orbit	Measurement range: 256,65536nT	7kg	10 W	20bps
TEDA	Corpuscular measurement device	Acquirement of corpuscular data on lunar revolving orbit	Fine suspended matter, neutral gas and other substance measurement	18kg	50 W	150bps
TEDA	Part material deterioration device	Acquirement of part material deterioration data on lunar revolving orbit	Sample: Memory, MPU, solar cell, etc.	15kg	20 W	100bps
AO sensor		Microwave radiometer, thermal infrared radiometer, etc		40kg		
			Total	550kg	1100 W	60Mbps

(*) Elements observed by fluorescence X-ray spectrometer: Mg, Al, Si, Fe, Ti, etc.
Elements observed by gamma-ray spectrometer: U, Th, K, etc.

2.2.3 System study

(1) System analysis

Fig. 2.2.3-1 shows the flight profile of lunar observation satellite. Table 2.2.3-1 indicates the flight sequence. For propellant quantity, the margin of 5% should be allowed. One H-II rocket would enable a satellite of 2t class to enter a lunar revolving orbit.

(2) System study

Fig. 2.2.3-2 and 3 show the system plot and fairing storage, respectively. Because the lunar revolving orbit cannot achieve the solar synchronization, solar cell panels should be 2 axis driven and the yaw-around should be performed every six months.

Table 2.2.3-2 and 3 show the system overview and main functions of subsystems, respectively. The development of bus aircraft seems possible with the conventional earth observation satellite technology.

Weight and electric power estimation is indicated in table 2.2.3-4. (For details, refer to attached information 3.)

(3) Rocket system

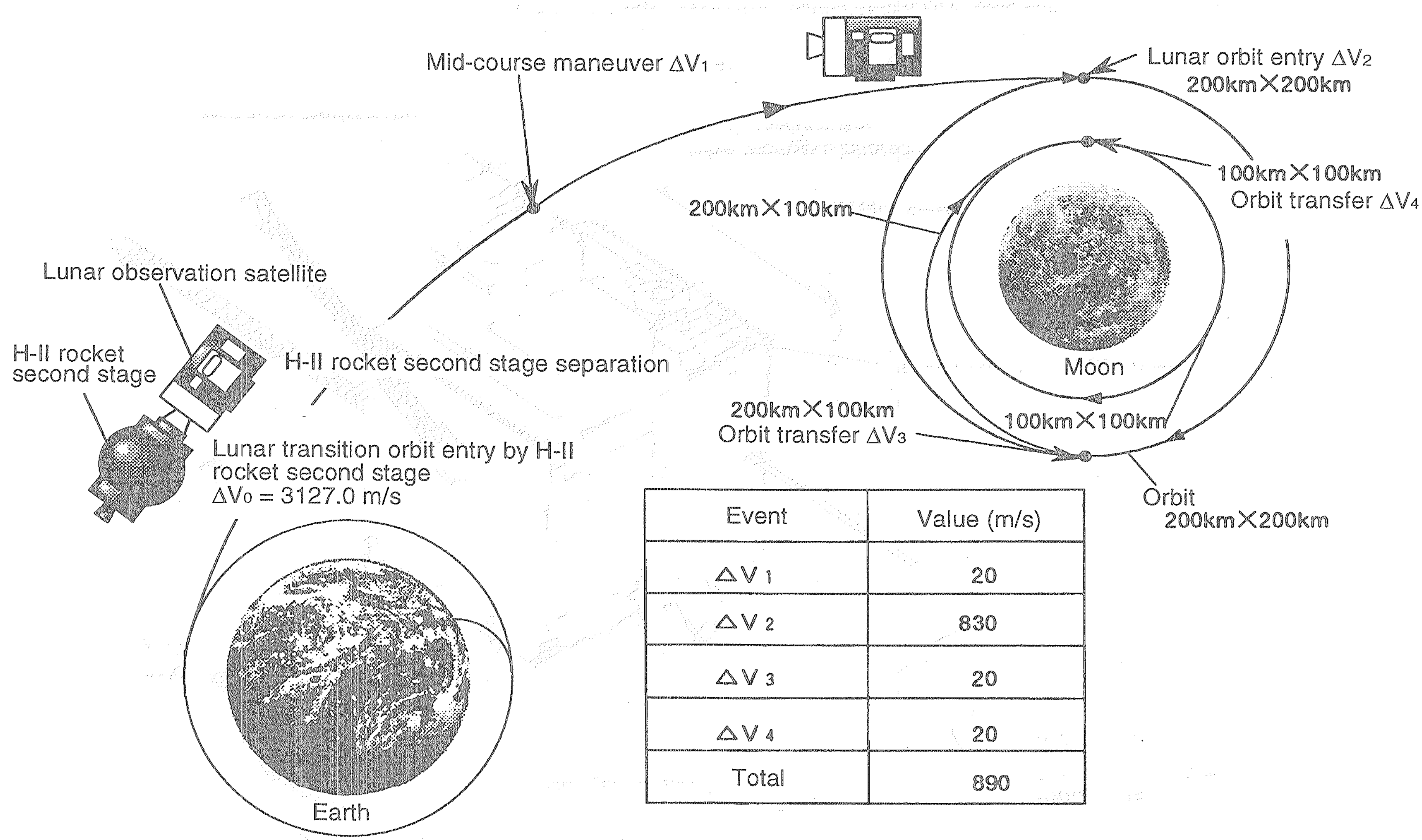
Lunar transition orbital entry should be achieved with H-II rocket.

Table 2.2.3-1 Flight sequence of lunar observation satellite

Event	Period	ΔV (m/s)	Propellant weight (kg)	Current weight (kg)
Lunar transition orbital entry with H-II	Standard	3150	—	2,800
Mid-course maneuver	Approx. 13 hours	20	20	2,780
Lunar orbit entry to 200 × 200 km	Approx. 90 hours	830	650	2,110
Orbit transfer to 200 × 100 km	Approx. 100 hours	20	15	2,095
Orbit transfer to 100 × 100 km	Approx. 110 hours	20	15	2,080
No orbit control	—	100	70	2,010

Table 2.2.3-2 System overview of lunar observation satellite

Item	Contents
Launch vehicle	H-II rocket (fairing=4m diameter)
Entry orbit	Lunar orbit with the altitude of 100km (inclination=85°)
Mission period	Two years
Weight	Approx. 2t
Mission equipment	Visible shortwave infrared ray imaging spectrometer, fluorescence X-ray spectrometer, TEDA gamma-ray spectrometer, underground exploration radar, laser altimeter, etc.



Event	Value (m/s)
ΔV_1	20
ΔV_2	830
ΔV_3	20
ΔV_4	20
Total	890

Fig. 2.2.3-1 Flight profile of lunar observation satellite

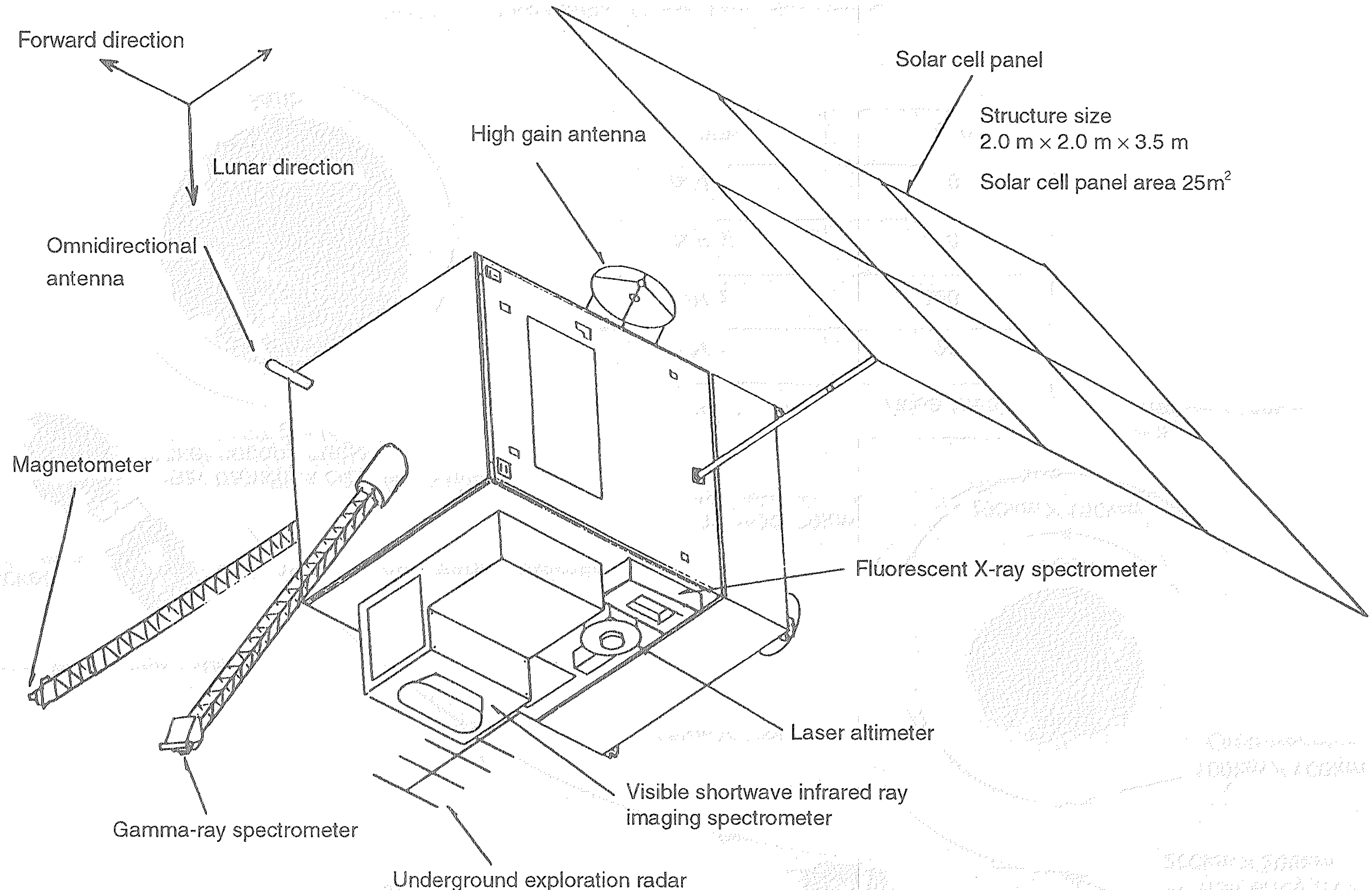


Fig. 2.2.3-2 Lunar observation satellite system plot (proposed)

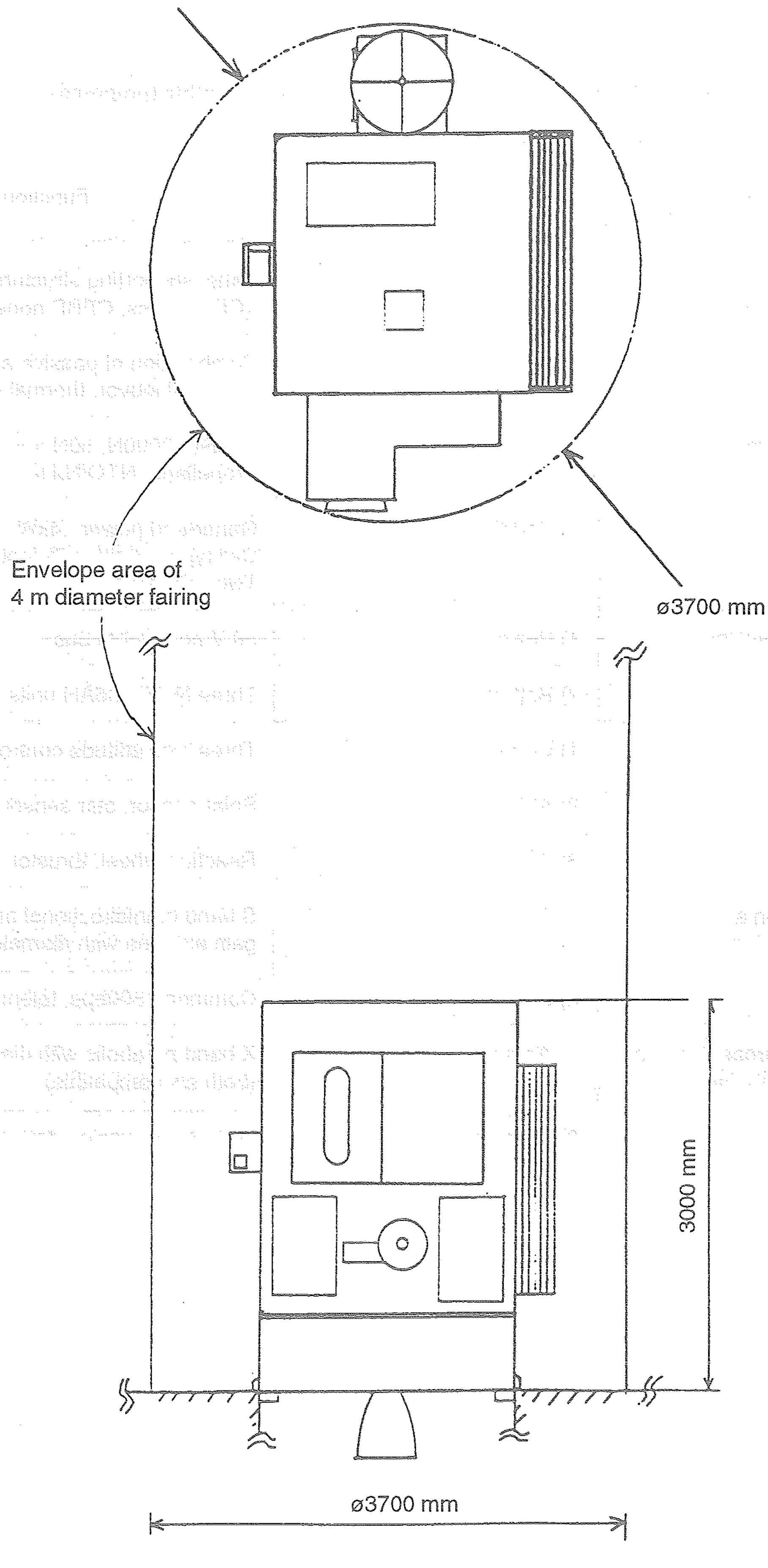


Fig. 2.2.3-3 Lunar observation satellite fairing storage

Table 2.2.3-3 Main functions of lunar observation satellite (proposed)

Section	Item	Function
Structure system	1) Structure	Panel supporting structure (CFRP truss, CFRP honey-comb panel)
Thermal control system	1) Method	Combination of passive and active (Thermal louver, thermal blanket, etc.)
Propulsion system	1) Thruster	Thrust: 2000N, 50N $\times$ 4, 5N $\times$ 12 Propellant: NTO/N_2H_4
Solar cell paddle system	1) Paddle	Generated power: 4kW Cell type: Highly efficient Si cell Two axis drive
Power supply system	1) Power supply	50 V non-stable bus
	2) Battery	Three Ni-MH 35AH units
Attitude control system	1) Control method	Three axis attitude control
	2) Attitude sensor	Solar sensor, star sensor, IMU
	3) Actuator	Reaction wheel, thruster
Communication and data processing system	1) Antenna	S band omnidirectional antenna + high gain antenna with diameter of 0.8m
	2) Data rate	Command 500bps, telemetry 1kbps
Mission data processing and transmission system	1) Antenna	X band parabolic with diameter of 0.8m (both s/x compatible)
	2) Transmission output	20W (transmission data rate: 60Mbps)

Table 2.2.3-4 Weight and electric power estimation of lunar observation satellite

Item	Weight (kg)	Power consumption (W)
Lunar transition orbit entry weight	2,800	
Satellite	2,030	2,244
Dry	1,843	2,244
Mission equipment	550	1,130
Bus equipment	1,293	1,114
Structure system	300	—
Thermal control system	80	200
Propulsion system	180	280
Solar cell paddle system	143	40
Power supply system	127	18
Attitude control system	87	150
Communication and data processing system	83	125
Mission data processing system	159	314
Mission data transmission system	54	267
Instrumentation system	80	—
Margin	187	—
Propellant	770	—

2.3 Study on electric propulsion orbit transfer mission spacecraft

2.3.1 Objective

Objectives of electric propulsion orbit transfer mission aircraft are listed as follows:

1) Long-time and wide-area space environmental data necessary for actual lunar surface activities should be acquired in the flight period of approx. two years required to reach lunar orbit with low thrust flight.

2) Ranging data of lunar revolving observation satellite should be relayed to measure the gravitational potential of the moon.

3) The basic technology of electric propulsion OTV should be established which becomes essential as efficient and economical transportation means in lunar exploitation requiring bulk transportation such as lunar base construction.

2.3.2. Mission plans

Measurement should be performed during entire orbit transfer period from the initial entry orbit(*1) to the final orbit(*2) with each of mounted TEDA (Technical Data Acquisition Equipment) devices. Details of these devices are as shown in table 2.3.2-1. For data transmission, mission data acquired in the visible range for the earth station should be directly transmitted. Invisible data should be recorded in data recorder to be regenerated and transmitted in visible range. Total transmission amount would be 1.6 kbps as a sum of mission system's real data rate of 320 bps and regenerated data rate. Bus system data should be acquired and transmitted in visible time with the transmission amount of 512 bps. Thereby, the transmission to TACS with USB should be performed setting the transmission rate to 3 kbps. Detailed circuit design is indicated in attached information 4.

In addition, the gravitational potential measurement on almost the entire lunar surface would be allowed by relaying the ranging data of lunar observation satellite during three-month flight. This flight would be performed on a circular orbit with the distance of 6000km to the moon core and orbital angle of 90°.

(*1) Earth revolving orbit with the orbit altitude of 200km, orbital angle of 30° and 1.5 hour cycle
(*2) Lunar revolving orbit with the orbit altitude of 100km, orbital angle of 90° and 1.4 hour cycle

2.3.3 System study

(1) System analysis

Table 2.3.3-1 shows the flight sequence and fig. 2.3.3-1 indicates the flight profile.

The electric propulsion transfer mission spacecraft should be launched by H-II and entered into low earth orbit. Then, it should raise the orbital altitude by performing spiral orbit transfer using ionic engine to enter the lunar orbit taking advantage of lunar gravitational capture. The required flight time would be approx. two years.

(2) System study

Fig. 2.3.3-2 and 2.3.3-3 illustrate mission spacecraft images and fig. 2.3.3-4 indicates the fairing storage.

The system overview is indicated in table 2.3.3-2, the subsystem specifications are in table 2.3.3-3 and the weight and electric power estimation is in table 2.3.3-4. Detailed explanation is described in attached information 4.

(3) Rocket system

- H-II rocket dual launch

  Weight upon lower orbit entry (altitude of 200km): 1.3t

Table 2.3.2-1 Mission list of electric propulsion orbit transfer mission aircraft

Mission name	Device name	Purpose	Main characteristics	Transmission rate (bps)	Weight (kg)
TEDA	Radiation absorption monitor	Measurement of radiation absorption by semiconductors	Detector: Silicon semiconductor detector Count number: 103 counts/sec or less	16	5
	Heavy ion observation device	Heavy ion type, energy, nuclear mass, intensity, direction distribution measurement	Detector: Position detector, PIN type semiconductor, Li drift type semiconductor	80	15
	Part and material deterioration device	Device for deteriorating parts and material	Samples: Memory, gate array, solar cell, etc.	100	15
	Magnetometer	Measurement of magnetic field intensity	Detector: Flux gate type Resolution: 0.125nT	20	2
	Charged potential monitor	Measurement of charged potential and leak current on the surface material of mission aircraft	Measurement item: Potential: Tuning fork modulation type Current : Electrometer Measurement accuracy: Within ±5%	100	3
			Total	316	40

Table 2.3.3-1 Mission sequence of electric propulsion lunar flight mission aircraft

Event	Orbital altitude	Required ΔV (m/s)	Flight time (day)(Note)	Mission: TEDA	Mission: Lunar gravitational potential measurement	Remarks
Launch						
Earth orbit entry	200km		—			Initial entry weight: 1.3 t
Earth spiral orbit transfer	200km ~ 150,000km from the earth core	6,473	475			200km ~150,000km from the earth core Because of large air resistance in the initial orbit transfer (200 ~350km), RCS should be used in the orbit transfer. ΔV (170m/s) required for this is included.
Lunar transition orbit entry	150,000km from the earth core to 250,000km from the earth core	860	50			
Lunar transition	250,000km from the earth core to 45,000km from the moon core	0	12 (537)			Lunar transition should be achieved with lunar gravity capture flight
Lunar orbit entry	45,000km from the moon core to 20,000km from the moon core					
Lunar spiral orbit transfer #1	20,000km from the moon core to 6,000km from the moon core	474	50 (587)			
Gravitational potential measurement mission	6,000km from the moon core	0	90 (677)			Ranging data of lunar observation satellite should be relayed for three months.
Lunar spiral orbit transfer #2	6,000km ~100km from the moon core	1050	40			
Total		7,333	717			

(Note): Numbers in parentheses indicate the time period from launch to each event termination.

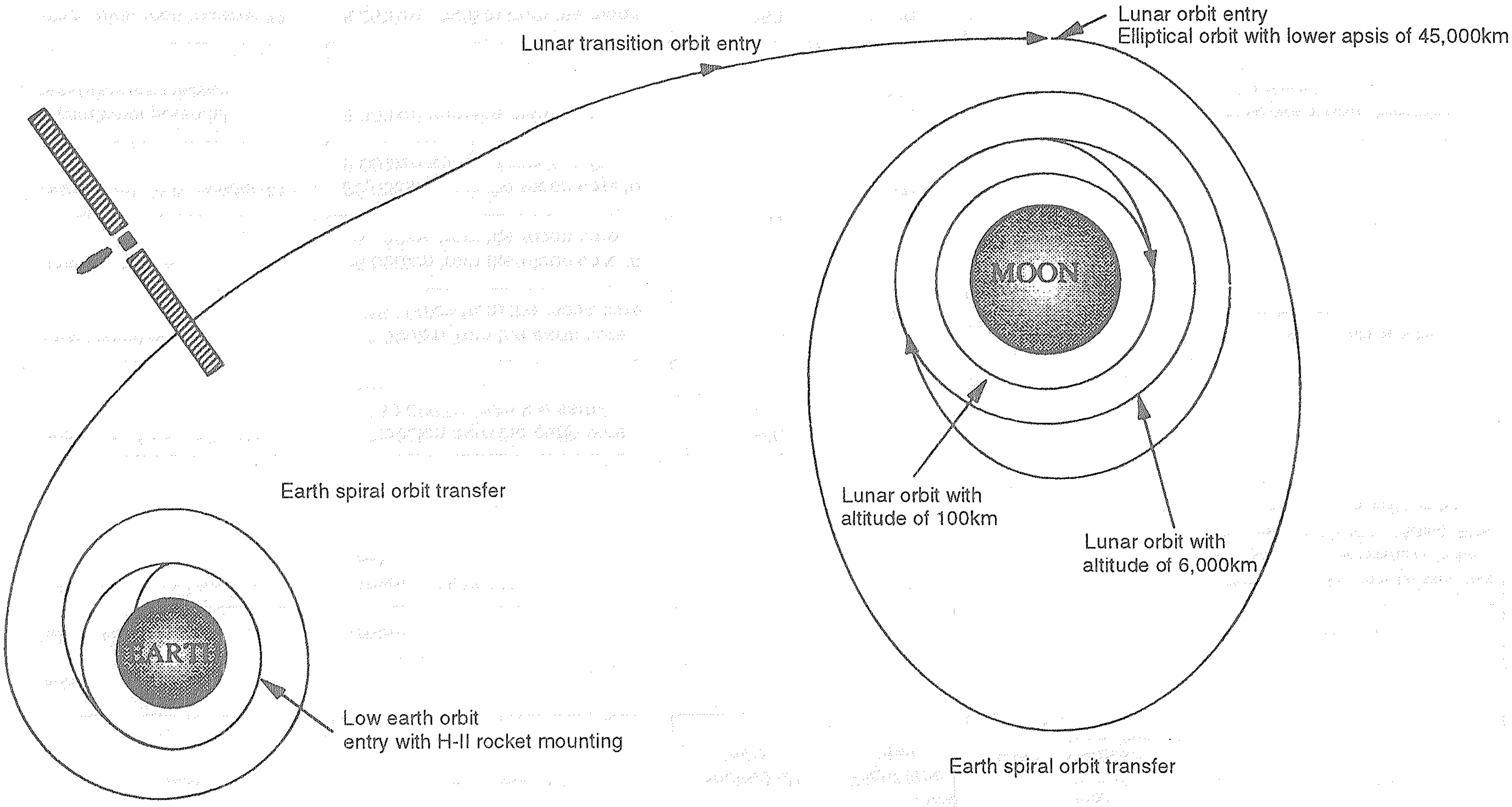


Fig. 2.3.3-1 Electric-propulsion orbit transfer mission aircraft profile

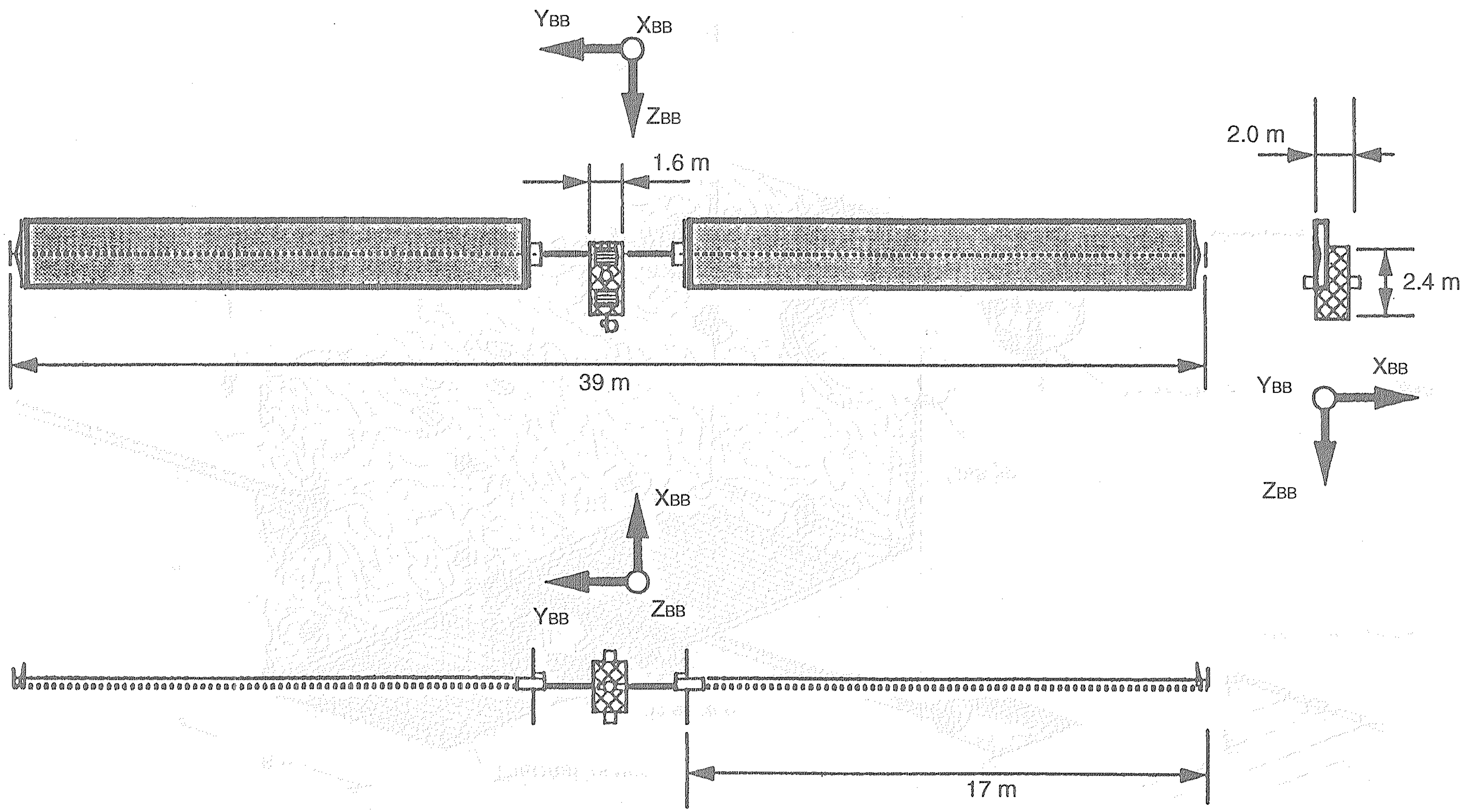


Fig. 2.3.3-2 Electric-propulsion orbit transfer mission aircraft plots

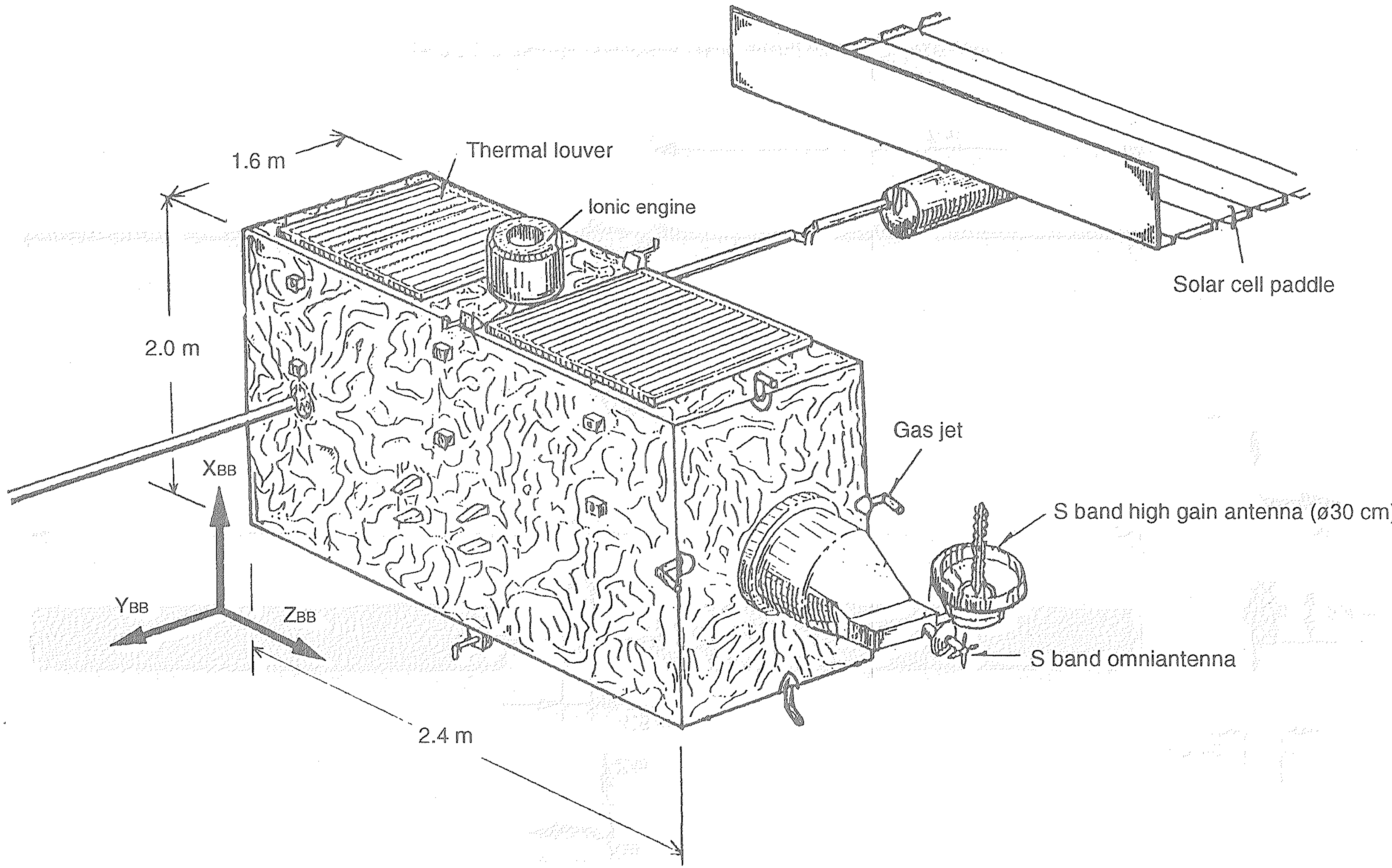


Fig. 2.3.3-3 Electric-propulsion orbit transfer mission aircraft body plots

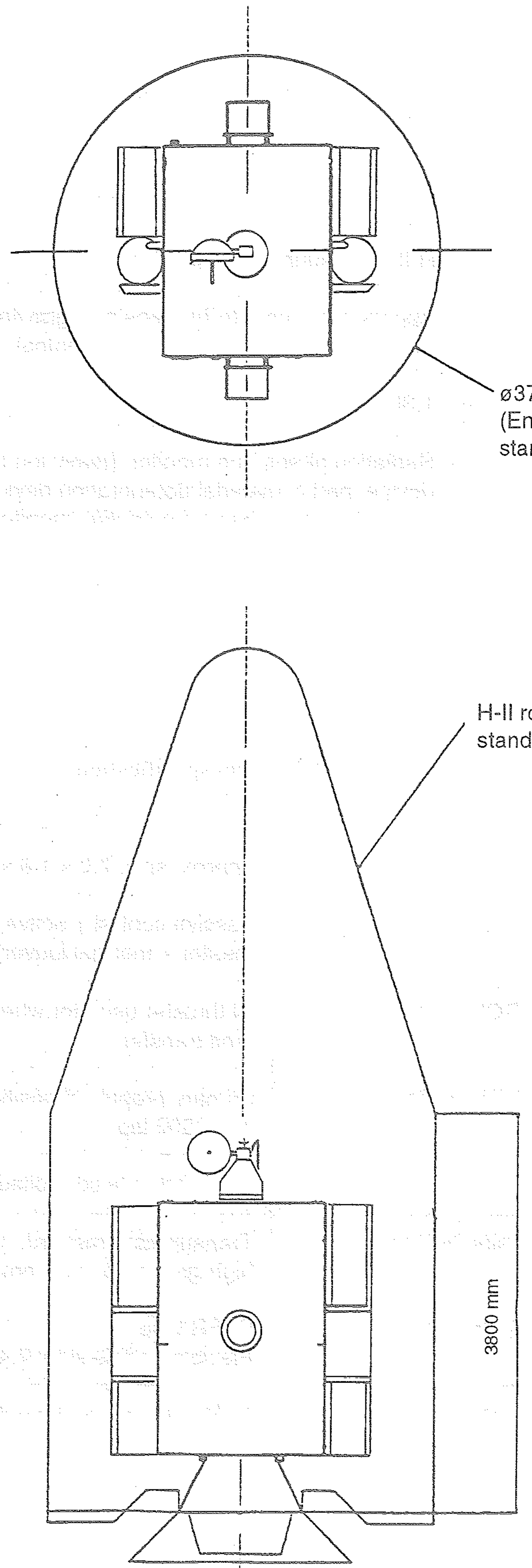


Fig. 2.3.3-4 Fairing storage

Table 2.3.3-2 System overview

Item	Contents
Launch vehicle	H-II dual launch (fairing: ø4m dual)
Mission period	Approx. two years (orbit transfer + gravitational potential measurement mission for three months)
Weight	1.3t
Mission equipment	Radiation absorption monitor, heavy ion observation device, part & material deterioration device, magnetometer, charged potential monitor

Table 2.3.3-3 Subsystem specifications

Structure system		Approx. size: 2.0 × 1.6 × 2.4m, box-type
Thermal control system		Passive control + active control (heater + thermal louver)
Propulsion system	RCS system	IN thruster used for wheel's unloading and initial orbit transfer
	Ionic engine system	Primary propulsion system with 200mN thrust and 3500 Isp
Derivative control system		3 axis attitude controlled zero momentum type
Communication data processing system		Transmission method: USB, Antenna: ø30 cm high-gain antenna + omnidirectional antenna
Power system	Solar cell	BSFR type Flexible paddle size: 2.4 × 1.7m, 2 airfoils
	Battery	Ni-MH, power consumption: approx. 1500Wh

Table 2.3.3-4 Weight and power estimation for electric propulsion mission aircraft

Item		Weight (kg)	Power (w)	Remarks
Mission equipment	Radiation absorption monitor	5	12	
	Heavy ion observation device	15	18	
	Part & material deterioration device	15	20	
	Magnetometer	2	4	
	Charged potential monitor	3	8	
	Subtotal	40	62	
Mission aircraft bus equipment	Structure system	80	—	
	Thermal control system	50	100	Heater power of propulsion system should be included in the thermal control system.
	RCS system	26	—	
	Ionic engine system	150	5300	
	Attitude control system	76	149	
	Communication & data processing system	50	85	
	Power system	380	30	
	Subtotal	812	5623/323	Maximums of electric power during ion engine operating/nonoperating
Dry weight of mission aircraft		858	—	
Propellant weight	Ionic engine	274	—	
	RCS	82	—	
	Subtotal	356	—	
Design margin		86	—	
Total		1300	5685/385	Maximums of electric power during ion engine operating/nonoperating

3. Study on lunar landing and mobile exploration spacecraft

3.1 Objectives

Objectives of lunar mobile exploration spacecraft are as follows:

1) Unmanned landing and resource survey on the lunar surface

2) Acquirement of lunar environmental data

3) Execution of various experiments on the lunar surface

4) Acquirement of scientific data on the moon

5) Mastering of landing/mobile exploration technique for moon and Mars

3.2 Mission plans

There are two kinds of systems for lunar mobile exploration spacecraft - one type is limited to perform lunar observation and experiments and the other is to collect samples and pass them to the sample return spacecraft besides the first type task. The following description explains the case of only performing lunar observation and experiments. For the case of including sample collection, refer to attached information 5.

3.2.1 Mission equipment

Table 3. 2-1 shows the mission equipment (proposed) for lunar mobile exploration spacecraft No. 1. The weight of mission equipment is 50kg.

Table 3.2-1 Lunar mobile exploration mission aircraft equipped sensors (proposed)

Sensor name	Observation purpose	Main function
Remote secondary ion mass spectrometer (Remote SIMS)	Analysis of chemical composition of lunar surface material, especially accessory element composition	Measurement range: M = 1~210 amu
Gamma-ray spectrometer	Element composition analysis and general classification of rocks	Measured energy range: 0.1~10 MeV
Fluorescence X-ray spectrometer	Element composition analysis	Measured energy range: 0.5~10 keV
Close-up camera	Zoom-up observation of lunar surface material to presume mineral types	Magnification: 100 times (TBD) Pixels: 512 × 512
Regolith (lunar mantlerock) heater	Regolith heating for volatile ingredient extraction experiments as well as resource exploitation experiments with melting/consolidation	Remote SIMS is also used in analysis.

3.2.2 Observation plans

(1) Route of rover

Since crust thickness and geological structure are different between the front surface and back surface of the moon, possible exploration spots for lunar mobile exploration spacecraft would be widely scattered. Therefore, it seems that several spacecraft would be required. For the exploration route of the first one, a route from the Apennines to the Copernicus crater is proposed as shown in fig. 3.2-1, because the most recently formed Copernicus crater and the older Mare Imbrium can be observed at one opportunity and risks can be decreased by known landing spots. For the second spacecraft, it seems effective to explore those spots different from the first one (the back surface or polar area) and the study should be performed in the future.

(2) Mission management

The mission of lunar lander spacecraft is to deploy rover on the lunar surface. In this case, lunar observation tasks are excluded.

Lunar rover should be assumed to run approx. 5km a day with the average day speed of 1km/h and the maximum of 4km/h (operation time: several hours a day) with the total mobile distance per year of approx. 1,000km. Observations and experiments should be performed on 100 spots or so including those determined by remote sensing data and those considered to be important looking at image data.

(3) Data transmission

Since rover move on the front surface of the moon, they would be always visible from the earth. Therefore, mission data recorders are not required. Considering the data rate up to 21Mbps and assuming the earth station to be DSN class of NASA, the line can be established with the transmission output of 5W using high gain antenna of 0.6m diameter. For the line design table, refer to attached information 5. If the second rover explores on the far-side surface, lunar orbiting data relay satellite will be required.

3.3 System study

(1) System analysis

Fig. 3.3-1 shows the flight profile of lunar lander/rover.

In the propulsions system of lunar lander, 5 units of 2,000N thrusters using storable propellant (ETS-VI mounted LAPS equivalent) should be clustered with throttling of approx. 40% being performed on one unit.

Table 3.3-1 indicates the flight sequence. One H-II rocket can transport a rover of 500kg to the moon if storable propellant is used in the propulsion system of lander.

(2) System study

Fig. 3.3-2 and 3.3-3 show the system plot and fairing storage, respectively. Table 3.3-2 and 3.3-3 indicate the system overview and main characteristics of subsystems.

Weight and electric power estimation is indicated in table 3.3-4. For details, refer to attached information 5.

(3) Rocket system

Lunar transition orbital entry should be achieved with H-II rocket.

Table 3.3-1 Flight sequence of lunar lander/rover

Event	Period	ΔV (m/s)	Propellant weight (kg)	Current weight (kg)
Lunar transition orbital entry with H-II	Standard	3,150	—	2,800
Mid-course maneuver	Approx. 13 hours	20	20	2,780
Lunar orbit entry with 100 × 100 km	Approx. 90 hours	860	695	2,085
Orbit transfer with 100 × 15 km	Approx. 100 hours	20	15	2,070
Orbit determination, equipment check out	Approx. 100 hours	—	—	2,070
Final landing	Approx. 110 hours	1,890	950	1,120
Attitude control, margin	—	150	50	1,070
Lunar rover separation	Approx. 120 hours	—	—	570+500 *

* Lunar landing spacecraft weight (570kg) + lunar rover weight (500kg)

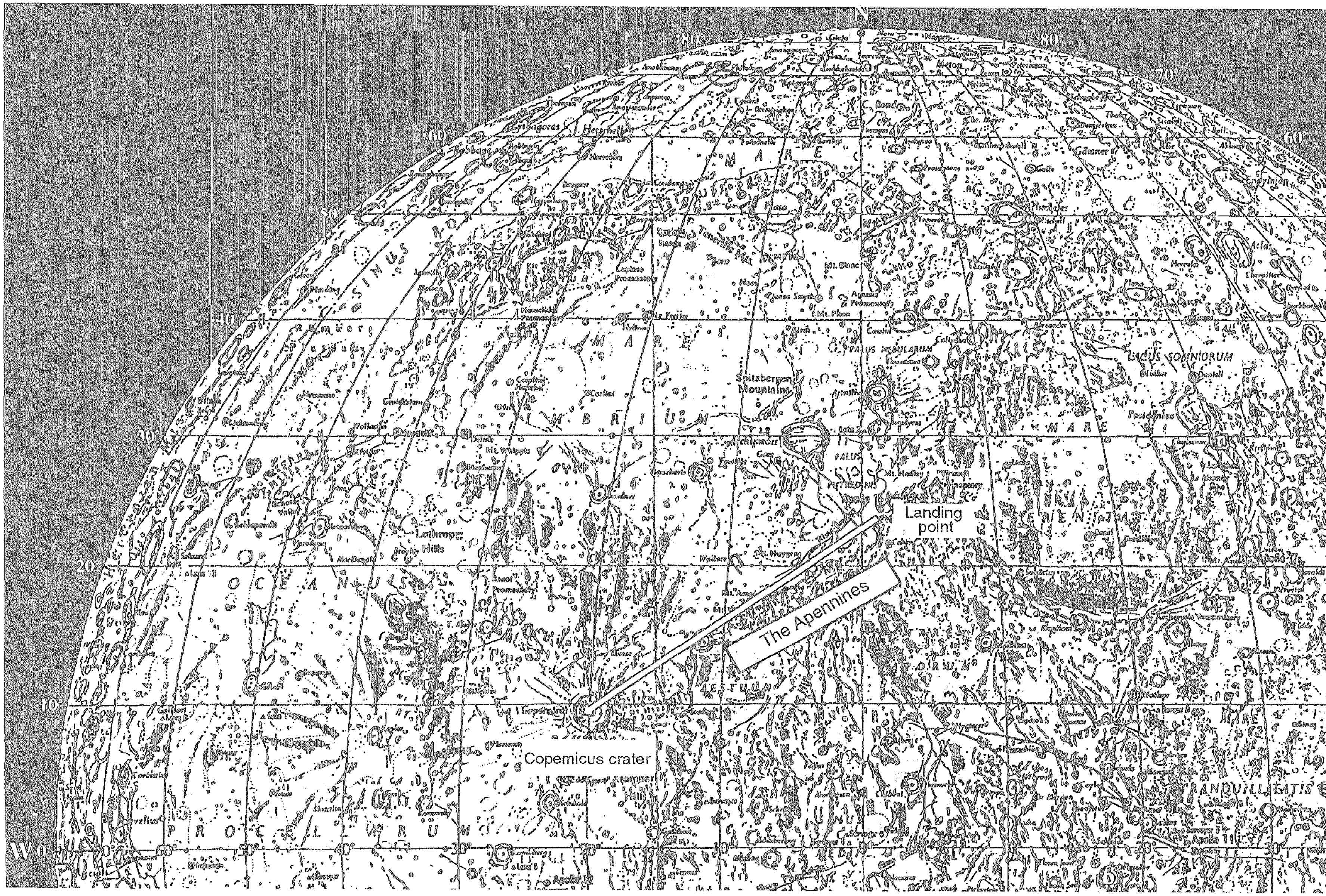
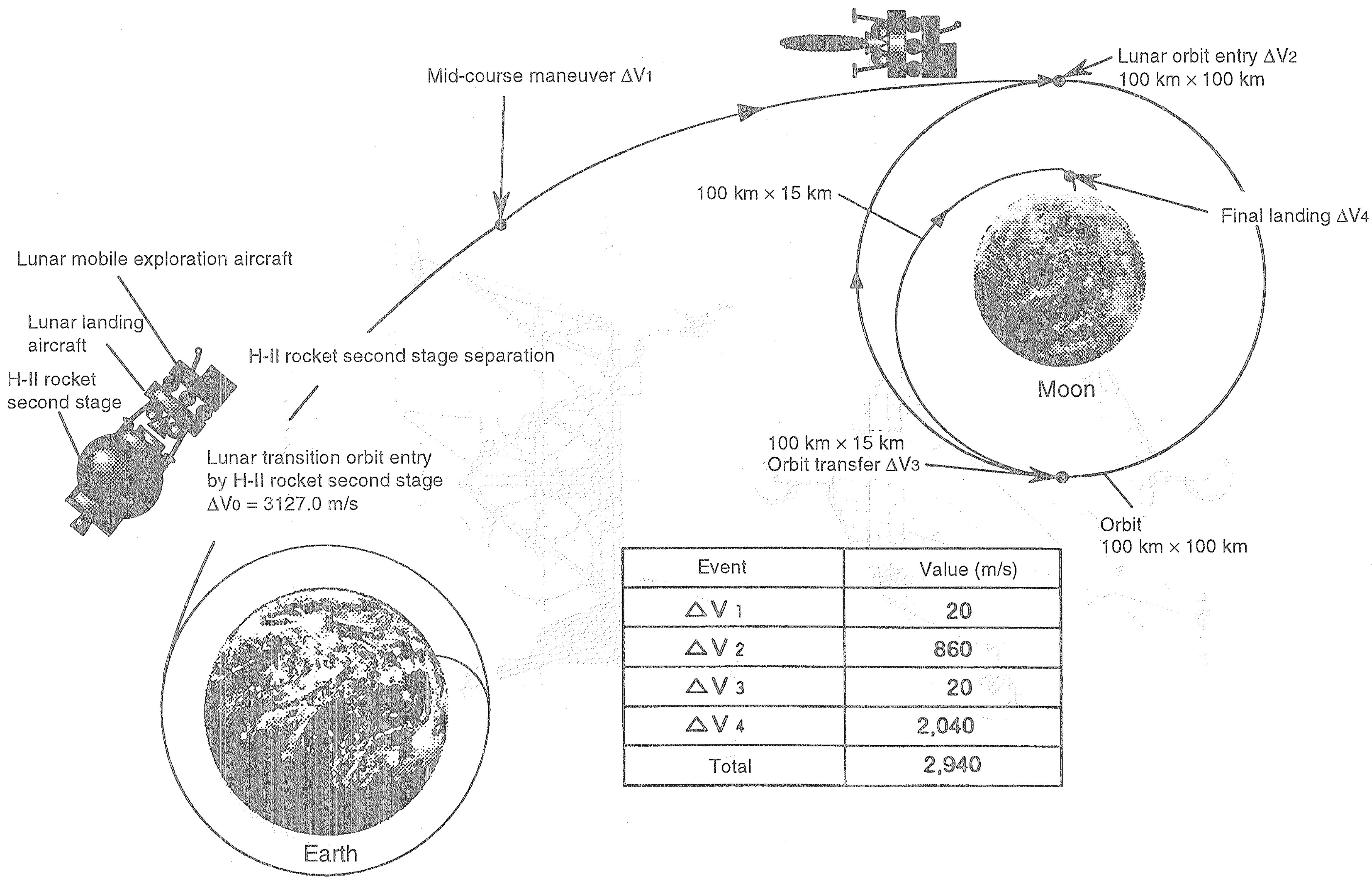


Fig. 3.2-1 Route of lunar rover (proposed)



Event	Value (m/s)
ΔV_1	20
ΔV_2	860
ΔV_3	20
ΔV_4	2,040
Total	2,940

Fig. 3.3-1 Flight profile of lunar landing/mobile exploration aircraft

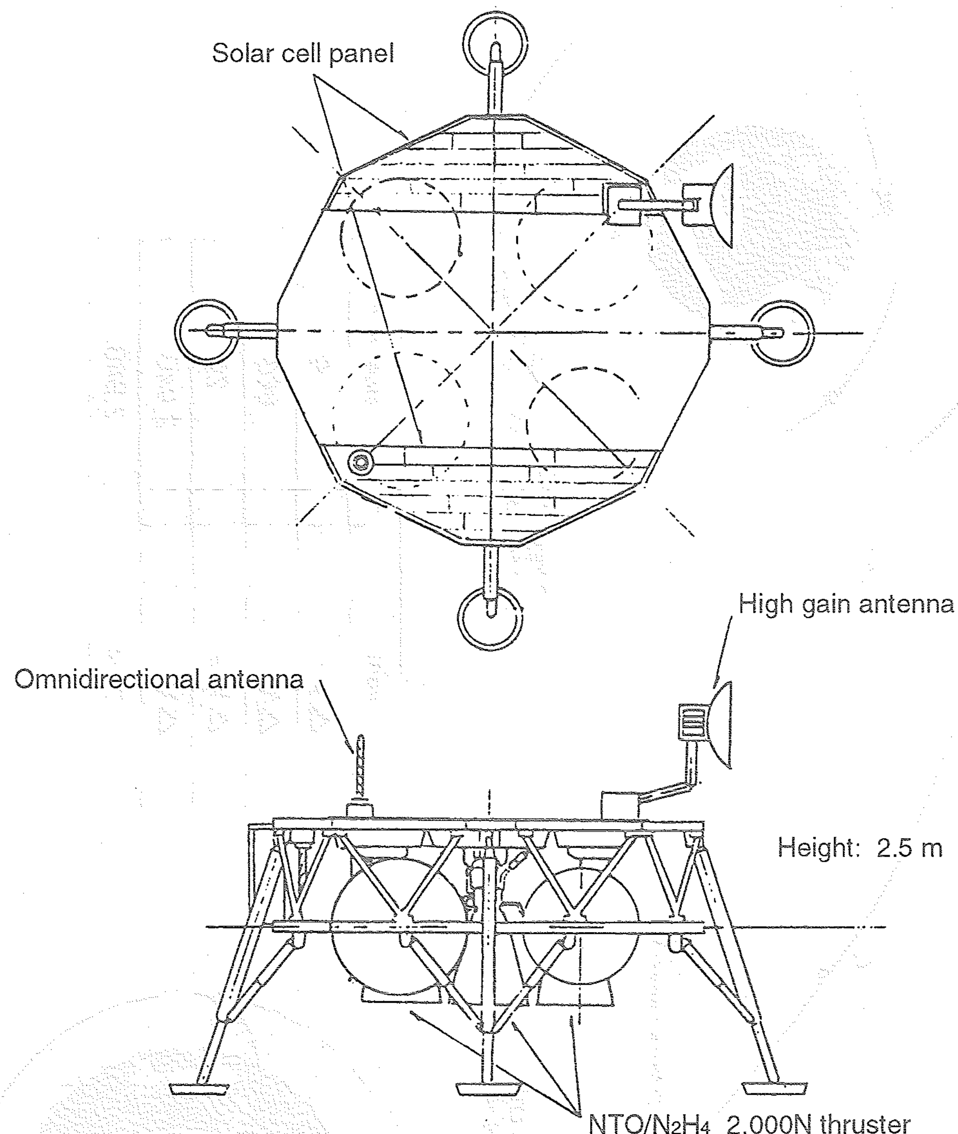


Fig. 3.3-2 (a) Lunar lander system plot

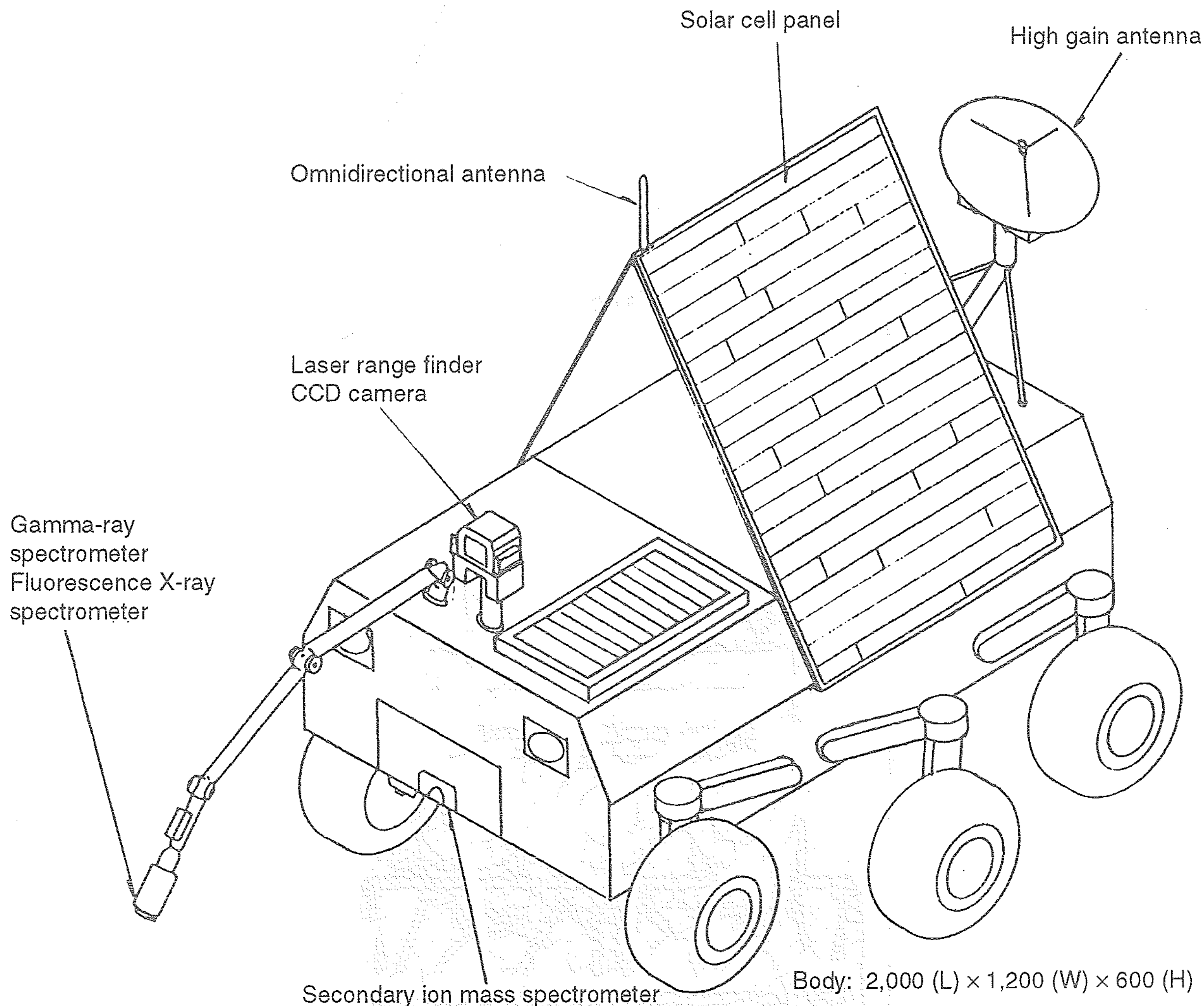


Fig. 3.3-2 (b) Lunar rover system plot

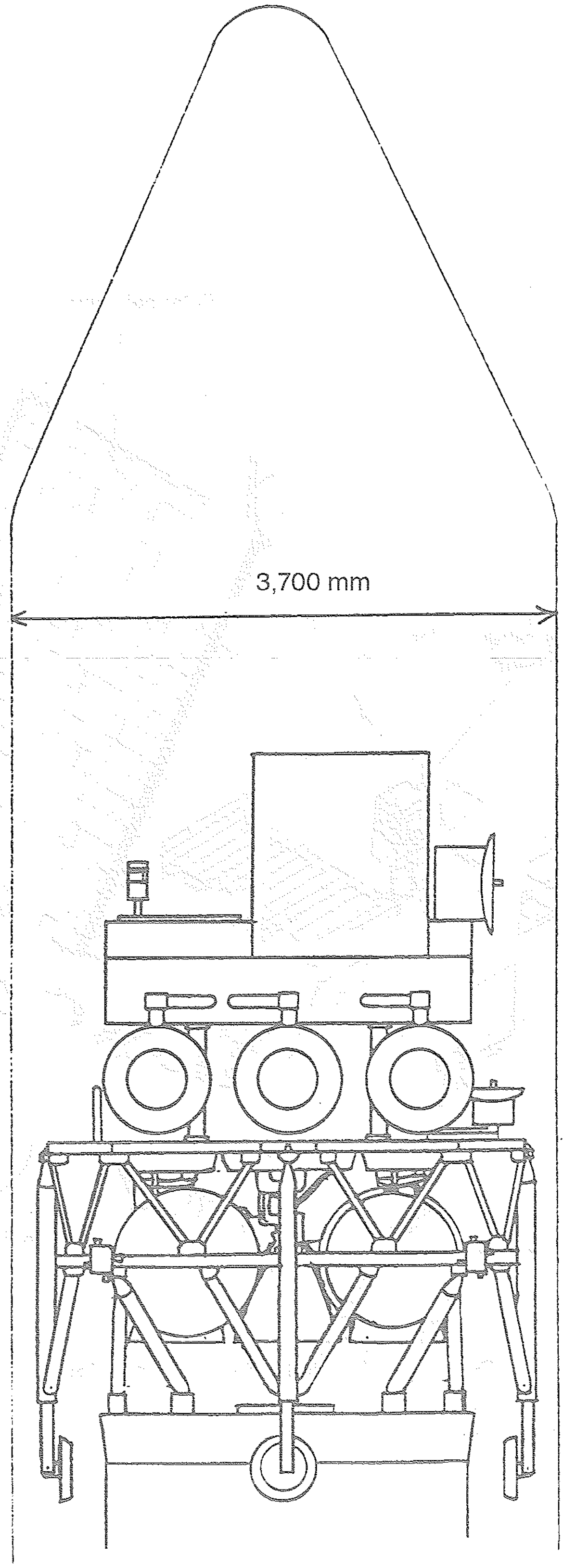


Fig. 3.3-3 Lunar lander/rover fairing storage

Table 3.3-2 (a) System overview of lunar landing aircraft

Item	Contents
Launch vehicle	H-II rocket (fairing = 4m diameter)
Landing point	Vicinity of Apollo 15 landing point (West edge of Mare Imbrium)
Mission period	Until the separation of lunar mobile exploration aircraft after landing
Weight	570kg

Table 3.3-2 (b) System overview of lunar mobile exploration aircraft

Item	Contents
Launch vehicle	H-II rocket (fairing = 4m diameter)
Place of activity	Between the Apennines and Copernicus crater (approx. 1,000km) (See fig. 3.2-1.)
Mission period	One year
Weight	Approx. 500kg (Weight of mission equipment: 50kg)
Mission equipment	Remote mass spectrometer, fluorescence X-ray spectrometer, gamma-ray spectrometer, regolith heater, close-up camera
Running performance	Maximum up-hill angle: 30° Maximum stable angle: 35° Maximum step height for climbing: 30cm Running speed: 1km/h (average)

Table 3.3-3 (a) Main functions of lunar landing aircraft subsystem (proposed)

Subsystem	Item	Function
Structure system	Structure	Truss + panel structure
	Shock absorption upon landing	Plastic deformation of aluminum honeycomb
Thermal control system	Method	Combination of passive and active types
Propulsion system	Main propulsion engine	NTO/N_2H_4 engine (LAPS equivalent) with thrust of 2,000N × 5 units; 40% throttling is performed on one unit.
	Thruster	50N × 8, IN × 16
Power supply system	Solar cell	Generated power: 200W Cell type: Highly efficient Si cell
	Battery	Capacity: 800Wh Type: Ag-Zn primary cell
Attitude control system	Attitude Control method	Three axis attitude control
	Landing method	Semi-automatic landing evading obstacles based on image data
	Mounted sensor	Solar sensor, star sensor, IMU, landing radar, video camera
Communication and mission data processing system	Frequency	Telemetry, command: USB Image data: X band
	Data rate	Telemetry: 1,024bps, command: 500bps Image data: 4.6MHz (transmission output: 5W)
	Antenna	Omnidirectional antenna: USB High gain antenna with diameter of 30cm: X/S band

Table 3.3-3 (b) Main functions of lunar mobile exploration aircraft subsystem (proposed)

Subsystem	Item	Function
Structure system	Style	Frame + panel structure
Thermal control system	Method	Thermal louver + passive type thermal control material
	Night time incubatory method	Heater + water latent heat utilization
Running system	Wheel	Wire mesh
	Steerage	Method: 2 front wheel steerage
	Driving method	Brushless DC motor + harmonic driving decelerator
Power system	Solar cell panel	Generated power: 200W (max.) Cell type: Highly efficient Si cell
	Battery	Type: Ni-MH Capacity: 1,400Wh
Flight system	Aviation	Inertial aviation + geographic recognition aviation
	Control	Remote control + automatic risk avoidance control
	Mounted sensor	CCD camera, laser range finder, wheel revolution indicator, solar sensor, clinometer, touch sensor
Communication and mission data processing system	Frequency	Telemetry, command: USB Mission data: X band (transmission output: 5W)
	Data rate	Telemetry: 1,024bps, command: 500bps Mission data: 21Mbps
	Antenna	Omnidirectional antenna: USB High gain antenna with diameter of 60cm: X/S band

Table 3.3-4 Weight and electric power estimation of lunar landing/mobile exploration aircraft

Item	Weight (kg)	Power consumption (W)
Lunar transition orbit entry weight	2,800	
Lunar landing aircraft	2,300	700
Dry	570	662
Bus equipment	512	662
Structure system	114	—
Thermal control system	20	30
Propulsion system	200	360
Power supply system	49	15
Attitude control system	61	140
Communication and data processing system	48	117
Instrumentation system	20	—
Margin	58	38
Propellant	1,720	—
Lunar mobile exploration aircraft	500	800
Mission equipment	56	100
Bus equipment	407	658
Structure system	66	—
Thermal control system	94	50
Running system	48	279
Power system	67	16
Flight system	52	108
Communication and data processing system	35	45
Mission data processing and transmission system	35	160
Instrumentation system	10	—
Margin	37	42

4. Study on lunar sample return system

4.1 Objective

Objectives of lunar sample return are listed as follows:

(1) Detailed analysis of lunar material

(2) Validation of remote sensing data with orbiter

(3) Mastering of lunar shuttle technology

(4) Development of Mars sample return technique

4.2 Sample recovery method

1) Sample recovery should be performed by the lunar mobile exploration spacecraft separately blasted-off.

2) Weight of recovered sample should be 50kg.

4.3 System study

4.3.1 Study case

Lunar sample return system described in this chapter consists of capsule, landing/takeoff module, lunar orbiter and external tank. The function of each part is listed as follows:

- Lunar orbiter:

  Capsule with samples should be entered into the earth transition orbit. This module should be separated from the body before landing on the moon and parked on the lunar orbit while the landing/takeoff module loads samples. For propellant, storable type should be used.

- Landing/takeoff module:

  This module should land on the moon and collect samples, then take off the moon with collected samples, which should be transshipped into a lunar orbiter on the orbit - at this point the mission of the landing/takeoff module will be terminated. The thrust system of this module should be used in lunar orbiter entry and lunar landing/takeoff. The propellant should be of storable type.

  Upon takeoff, landing legs, landing sensor, sample transshipment device should be left on the lunar surface.

- External tank:

  Upon lunar orbit entry, the propulsion system of landing/takeoff module should be used. This external tank should store the propellant required for entry and supply it to the landing/ takeoff module.

- Capsule:

  It should receive samples on the lunar orbit and enter the earth atmosphere to be recovered.

The comparison with other cases is included in attached information 8.

4.3.2 System study

(1) System analysis

Fig. 4.3-1 indicates the flight profile and table 4.3-1 shows the flight sequence.

Event	Symbol	Incremental speed (m/s)
Lunar transition orbit entry	ΔV_1	3,127
Mid-course maneuver	ΔV_2	20
Lunar revolving orbit entry	ΔV_3	860
Lunar landing	ΔV_4	2,050
Lunar takeoff	ΔV_5	1,910
Rendezvous docking	$\Delta V_{5'}$	140
Earth transition orbit entry	ΔV_6	860
Mid-course maneuver	ΔV_7	20

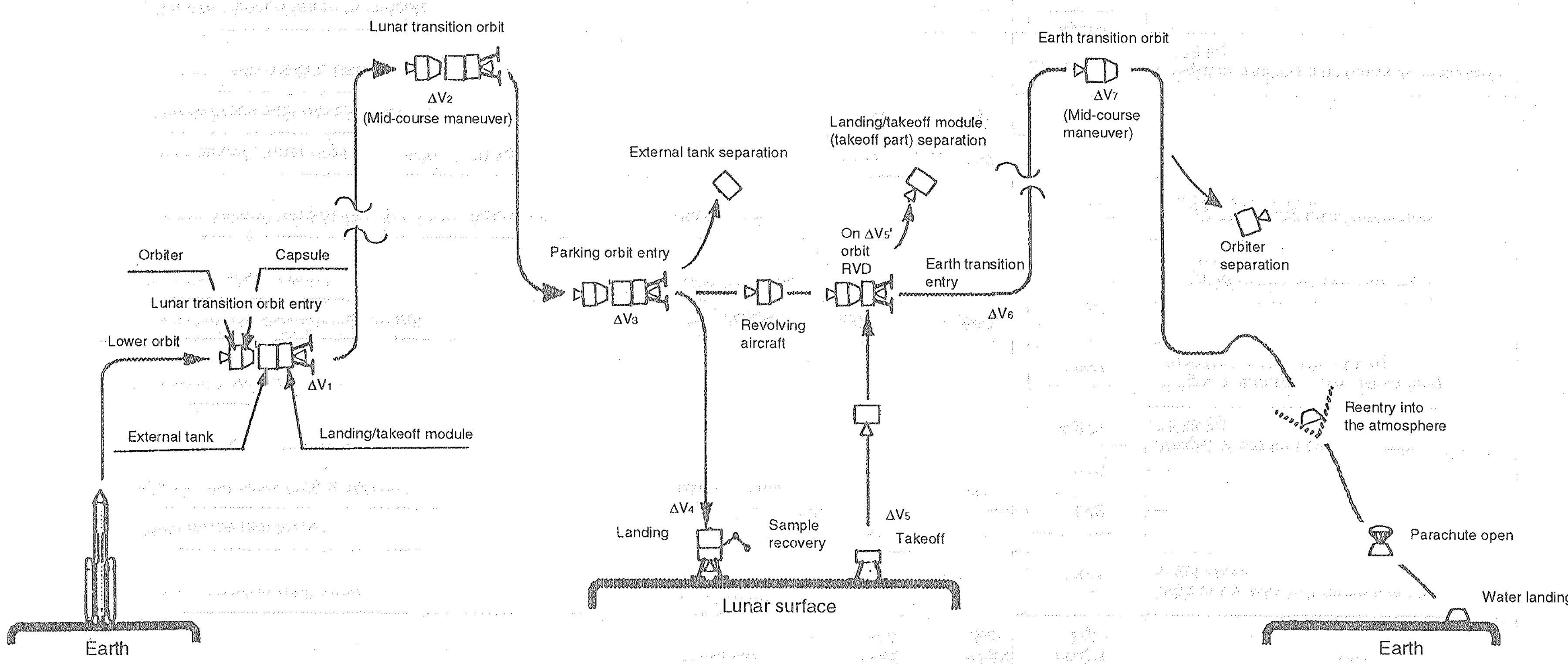


Fig. 4.3-1 Flight profile of lunar sample return system

Table 4.3-1 Flight sequence of lunar sample return aircraft

	Flight time	Incremental speed (m/s)	Propellant weight (kg)	Current weight (kg)	Remarks
1 Lunar transition orbit entry	Blast-off date	—	—	7,691	Orbit entry with H-II derivative type rocket of 20t class
2 Mid-course maneuver	↓	20	49	7,642	
3 Lunar orbit entry (100 × 100 km)	After 4.5 days	860	1,832	5,810	
4 Revolving part separation	↓	—	—	4,272	Weight of orbiting part (capsule included): 1,538 kg
5 External tank separation	↓	—	—	3,837	Weight of external tank (remaining propellant included): 414 kg
6 Lunar landing (hovering included)	After 5 days	2,050	1,840	1,997	
7 Sample transshipment	After 5~10 days	—	—	2,067	Weight of collected samples (box included): 70 kg
8 Landing/takeoff module (landing part) separation	After 10 days	—	—	1,320	Weight of landing part (remaining propellant included): 747 kg
9 Lunar takeoff, lunar orbit entry (100 × 100 km)	↓	1,910	602	718	
10 Rendezvous with orbiting part	↓	140	31	687	
11 Docking with orbiting part	↓	—	—	2,225	Weight of orbiting part (capsule included): 1,538 kg
12 Sample transportation to capsule	↓	—	—	2,225	
13 Landing/takeoff module (takeoff part) separation	↓	—	—	1,608	Weight of takeoff part (remaining propellant included): 617 kg
14 Earth transition orbit entry	After 10.5 days	860	385	1,223	
15 Mid-course maneuver	↓	20	8	1,215	
16 Orbiter separation	After 15 days	—	—	770	Weight of orbiter (remaining propellant included): 445 kg
17 Atmospheric entry	↓	—	—		
(Total weight of propellant)			4,747		

Specific thrust (for NTO/N_2H_2 engine): 320

(2) System study

Fig. 4.3-2 illustrates the configuration on the lunar transition orbit. Table 4.3-2 ~ 4.3-4 indicate main functions of entire system, subsystem structure and weight estimate, respectively.

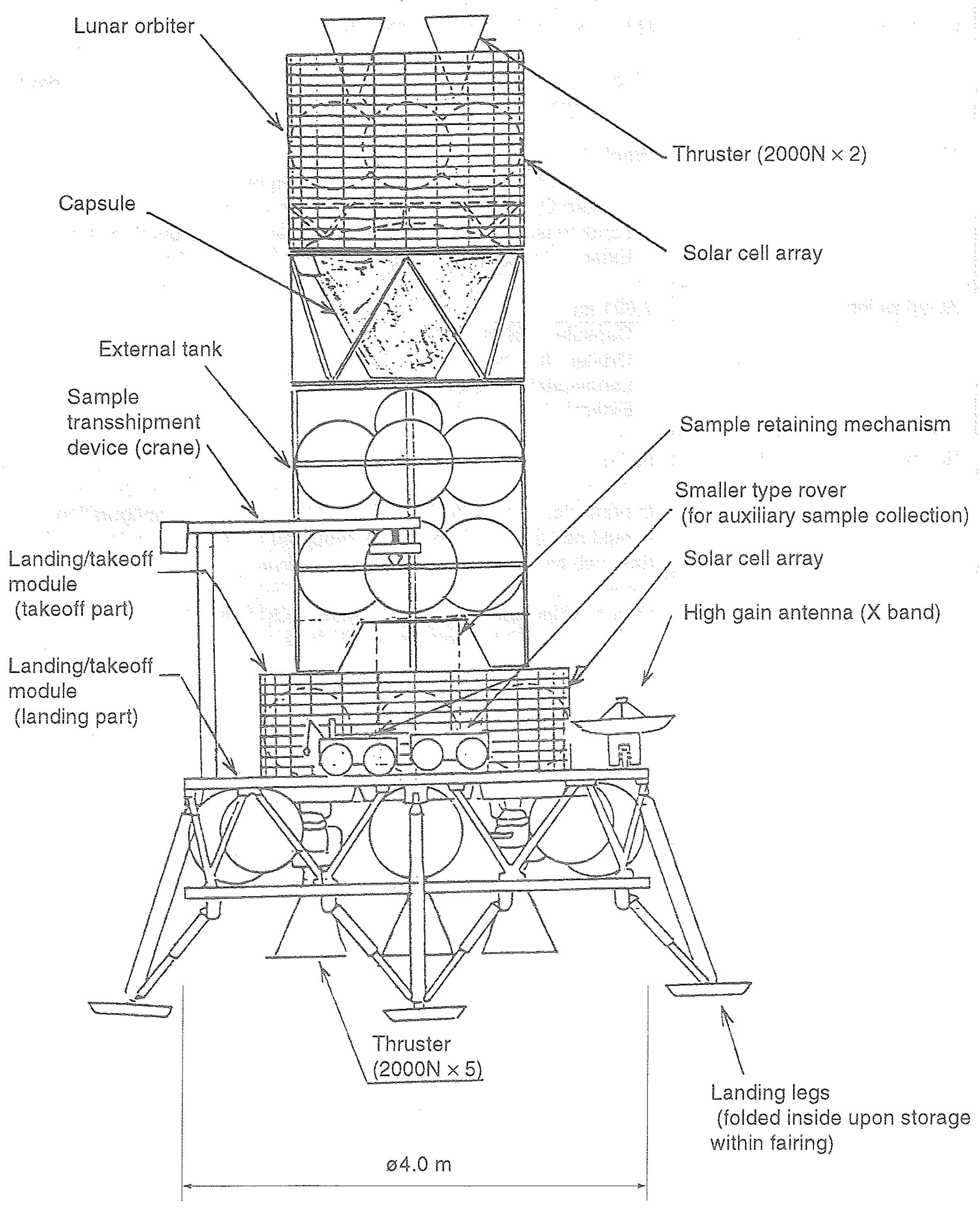


Fig. 4.3-2 Configuration on the lunar transition orbit

Table 4.3-2 Main functions of lunar sample return aircraft

Item	Contents
Launch vehicle	H-II rocket derivative type (20t class)
Mission period	15 days (10 days for shuttling between the earth and moon + 5 days for stay-over on the moon)
Main size	Whole length Capsule: Max. diameter of 1.6m/length of 1.1m Orbiter: Outer diameter of 1.6m/length of 2m Landing/takeoff module: Max. diameter of 4.0m/length of 4m External tank: Outer diameter of 1.6m/length of 3.5m
All-up weight (excluding samples)	7,691 kg Capsule: 700 kg Orbiter: 838 kg Landing/takeoff module: 3,837 kg External tank: 2,316 kg
Recovery sample weight	50 kg
Redundancy configuration	In principle, dual-system and stand-by redundancy configuration should be adopted. The CPU of mounted computer should be of IFO (tri-level) type taking account of single event.
Earth station	• For tracking control, NASDA station (KATSUURA, etc.) and ISAS station (USUDA) supports should be assumed.

Table 4.3-3 System configuration for each module

	Capsule	Lunar revolving aircraft	Landing/takeoff module
Structure system	• Apollo-style capsule • Adiabatic structure with ablater • Cover closing gear (for sample storage) • Data communication with revolving aircraft and power connector	• Cylindrical structure • Coupling with capsule via upper separation unit • Data communication with capsule and power connector	• Cylinder (takeoff part) + truss (landing part) structure • Landing part should be separated upon takeoff
Atitude control system	Computer, IMU, GPS receiver, RV sensor (target appulse system), valve driving mechanism	Star sensor, RVD sensor (target distant system) (Capsule should be used for IMU, etc.)	Obstacle detection system (landing part), RVD sensor (chaser), computer, IMU
Communication system	CU-RIU, omnidirectional antenna, transponder, data recorder	RIU (2 style) (Capsule should be used for CU, etc.)	• CU-RIU (4 style)
Power system	[Return module separation ~ recovery] Power supply with mounted battery [Except for the above-mentioned case] Power supply from lunar revolving aircraft	Power supply with solar cell and battery (Solar cell should be fixed on the external boarding.)	Power supply with solar cell and battery (Solar cell should be fixed on the external boarding.)
Propulsion system	RCS for attitude control	Main propulsion system: NTO/N_2H_4 of 2,000N (× 2) RCS: 50N (× 4), 4N (× 12)	Main propulsion system: NTO/N_2H_4 of 2,000N (× 5) RCS: 50N (× 4), 1N (× 12)
Mission system	Retaining mechanism of 50kg sample		Crane for sample transshipment: one set Sample retaining mechanism: one set Monitor camera (for crane top monitoring): one set (for entire task monitoring): one set Smaller type rover: one set Sample box (spare): one set
Thermal control system	Capsule's heat removal and incubatory should be performed with heat pipe, radiator and heater.		

Table 4.3-4 Weight estimation of each module

	Capsule	Revolving aircraft	Landing/takeoff module		External tank
			Landing	Takeoff	
Structure system	268	130	305	165	130
Derivative control system	66	28	138	127	0
Communication and data processing system	41	19	83	41	0
Power system	35	49	55	55	0
Propulsion system	38	131	292	81	128
Thermal control system	40	25	75	45	10
Mission system	30	0	191	10	0
Recovery system	120	0	0	0	0
(Margin)	49	38	121	66	32
Dry weight	687	420	1,260	590	300
Propellant weight	13	416	2,568	657	2,008
GHe	0.1	2	9	3	8
Weight of collected samples	50	0	0	50	0
Sample box	20	0	0	20	0
All-up weight (samples included)	700 (770)	838 (—)	3,837 (—)	1,250 (1320)	2,316 (—)

4.4 Rocket system

Lunar sample return system should be launched aboard using H-II derivative rocket (20t class). For the comparison study on the transportation system, refer to attached information 6.

5. Study on Mars observation satellite

5.1 Objectives

(1) Acquirement of wide-area Mars observation data

Wide-area Mars observation should be performed to acquire the fundamental data for the successive Mars atmosphere entry/landing mission and Mars sample return mission.

Detailed missions are described as follows:

a) Geographical/surface material composition observation on Mars

- Creation of geographical map across the entire Mars surface with the spatial resolution of 100m and observation of surface material composition for investigating geological profile, geological evolution and volatile material presence on the Mars surface
- Observation of polar cap ingredients, thickness and seasonal changes for investigating polar cap forming era and influence on climate change
- Observation of thermal infrared emission for investigating weather and volcanism

b) Geographical and other observation on possible landing spots

- Creation of geographical map with the spatial resolution of 1m on the peripheral area of possible sites (20~40 km square) for determining the possibility of soft landing and movement
- Detailed observation on each possible site required for selection

c) Observation of Mars atmosphere

- Survey on Mars atmospheric composition, temperature, atmospheric pressure and their vertical profiles, cloud and storm observation and moisture vapor distribution survey for investigating Mars weather/climate changes, atmospheric circulating system and the existence of volatile gases

d) Measurement of charged particle environment around Mars

- Observation of energy distribution of charged particles for investigating Mars magnetic field

(2) Establishment of planet exploration technique

Transition orbit entry technique and tracking control & operation technique should be established.

5.2 Mission plans

Table 5.2-1 shows the mission equipment. The orbital altitude should be 400km taking account of orbital altitude decrease caused by atmosphere and observation resolution, etc. According to solar synchronization conditions, inclination would be 92 degrees and regression cycle would be 79 days. The distance between adjacent orbits on the equator would be 24km. Therefore, for visible near infrared emission system with the resolution of 100m, observation should be performed using CCD with 300 elements and setting the trim to 30km. In case that the distance between the earth and Mars is 7.48×10^7km or less (for 79 days), realtime observation of sunshine data for visible near infrared radiometer, shooting camera and visible thermal infrared radiometer as well as constant observation for other sensors should be performed to simultaneously transmit Mars back surface data recorded in data recorder and realtime data. The data rate at this time should be 85kbps. (See attached information 7.) In case that the distance is more than 7.48×10^7km, transmission should be performed with lower rates depending on distance, efficiently combining real-time data and data recorder data.

5.3 System study

(1) System analysis

Table 5.3-1 indicates the flight sequence and fig. 5.3-1 shows the flight profile. After the earth revolving lower orbit entry completion using H-II derivative rocket of 15t class, Mars transition orbit entry should be achieved with reignition of LE-5A. H-II second stage separation should follow the transition orbit entry and mid-course maneuver as well as Mars revolving orbit entry should be performed with satellite mounted propulsion system (LAPS equivalent).

(2) System study

Fig. 5.3-2 and 5.3-3 illustrate system plots and fairing storage, respectively. Table 5.3-2 and 5.3-3 indicate system overview and subsystem specifications. Weight and power estimates are as shown in table 5.3-4. For details, refer to attached information 3.

(3) Rocket system

H-II derivative type of 15t class

Table 5.2-1 Mars observation satellite mission device list

Sensor name	Purpose	Main function	Transmission rate (kbps)	Weight (kg)
Visible near infrared radiometer	Investigation of surface condition as well as geography, rocks and mineral resource distribution, creation of geographic map with stereovision observation	Observation wavelength: 0.4~0.91μm, 4 bands One band stereovision Scan width: 30km Resolution: 100m	43.0	73
Radar altimeter	Inequality inspection, orbital altitude measurement, orbital information compensation, shape and gravitational potential compensation, inner density profile measurement, isostasy and viscosity measurement	Frequency: 13.8GHz Beam width: 1.6° Altitude accuracy: ± several meters Antenna: Parabolic antenna with diameter of 1m	1.0	25
Gamma-ray spectrometer	Investigation of regolith element composition	Detector: Germanium semi-conductor Observation band: 0.05~10MeV Energy resolution: Approx. 2keV Field of view: 60km × 60km	2.5	30
Shooting camera	Detailed shooting of local area	Resolution: 1m or less	10.0	15
Visible thermal infrared radiometer	Investigation of cloud distribution and atmospheric circulation, temperature profile on the surface, superstratum water vapor	Observation wavelength: 0.5~12.5μm, 4 bands Observation width: Approx. 320km	5.0	30
Microwave radiometer	Investigation of water vapor amount within the atmosphere and polar caps	Observation frequency: 23GHz band, 31GHz band	0.1	50
Ultraviolet spectrometer	Atmospheric measurement, concentration measurement	TBD	8.0	20
Radiation monitor	Measurement of temporal and spatial changes of cosmic ray intensity in the ambient of Mars and on the transition orbit	TBD	2.0	5
		Total	71.6	248

Table 5.3-1 Mars observation satellite mission sequence

Event	Orbit	Required ΔV (m/s)	Weight (kg)	Required time (Day)	Remarks
Launch					Launch with H-II derivative type rocket of 15t class
Mars transfer orbit entry	Hohmann orbit	3,700	6,541	(Reference)	Entry with LE-5A reignition Specific thrust: 452 s
H-II second stage separation			3,690		
Mid-course maneuver		200			For the following phases, the engine with storable propellant should be used. Specific thrust: 320 s Structural efficiency: 0.85 assumed
Mars orbit entry	Circular orbit with orbital altitude of 400km	2,500	1,560	240 ~ 280	Entry in the following two steps should be performed. Elliptical orbit entry (periapsis altitude of 400km, apoapsin altitude of 36,200km) Circular orbit entry (orbital altitude of 400km)

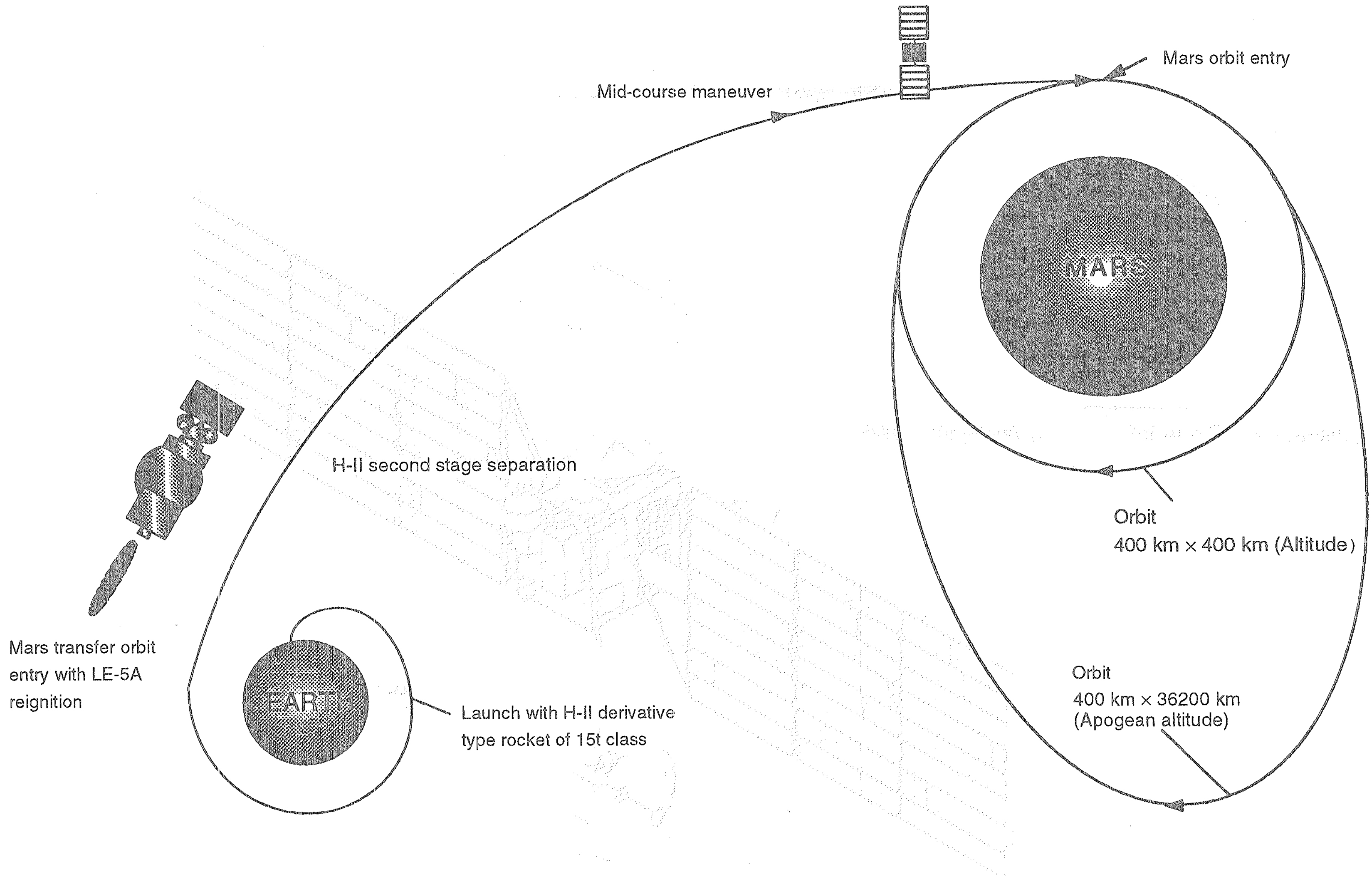


Fig. 5.3-1 Flight profile of Mars observation satellite

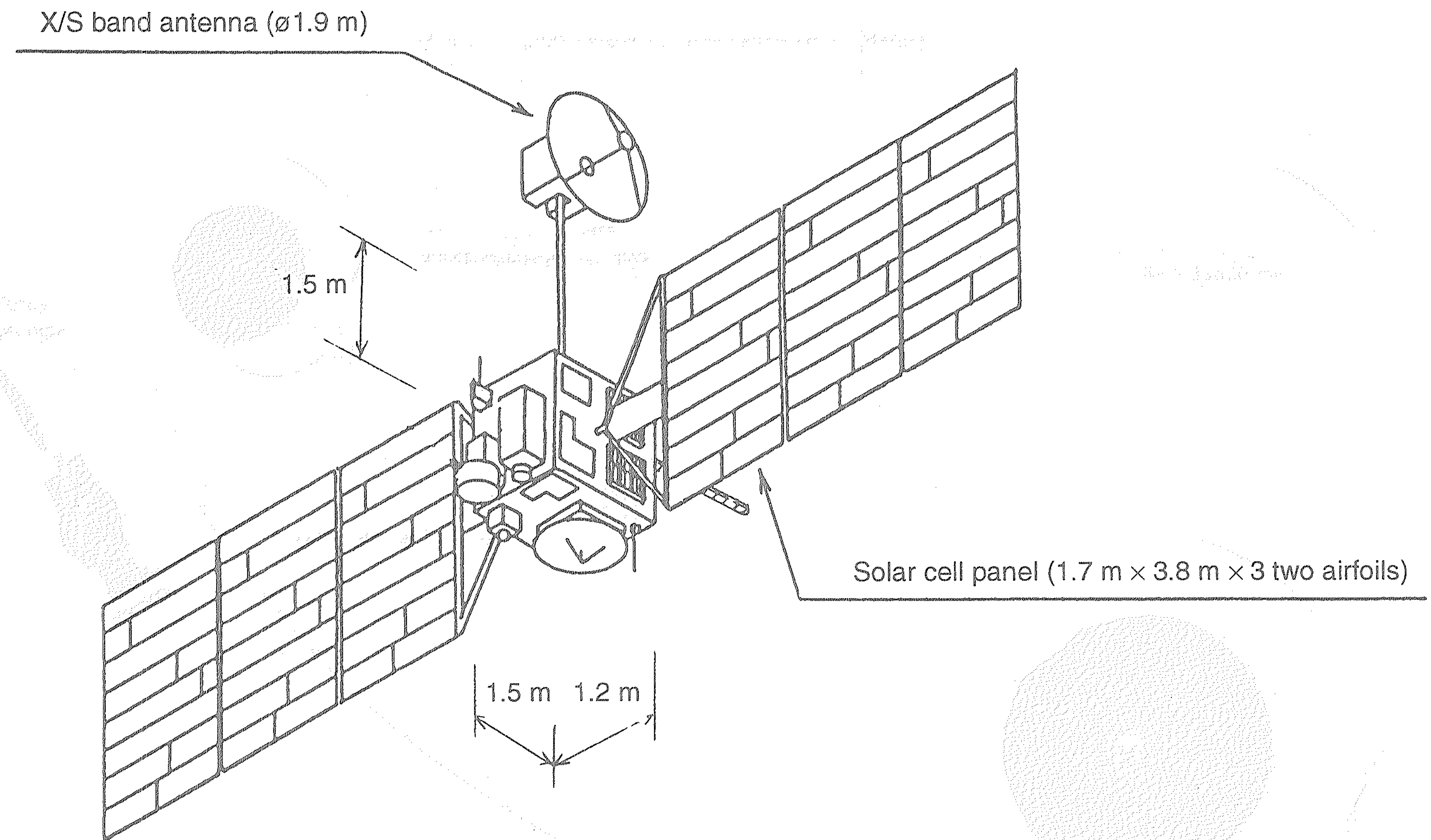


Fig. 5.3-2 Mars observation satellite plans

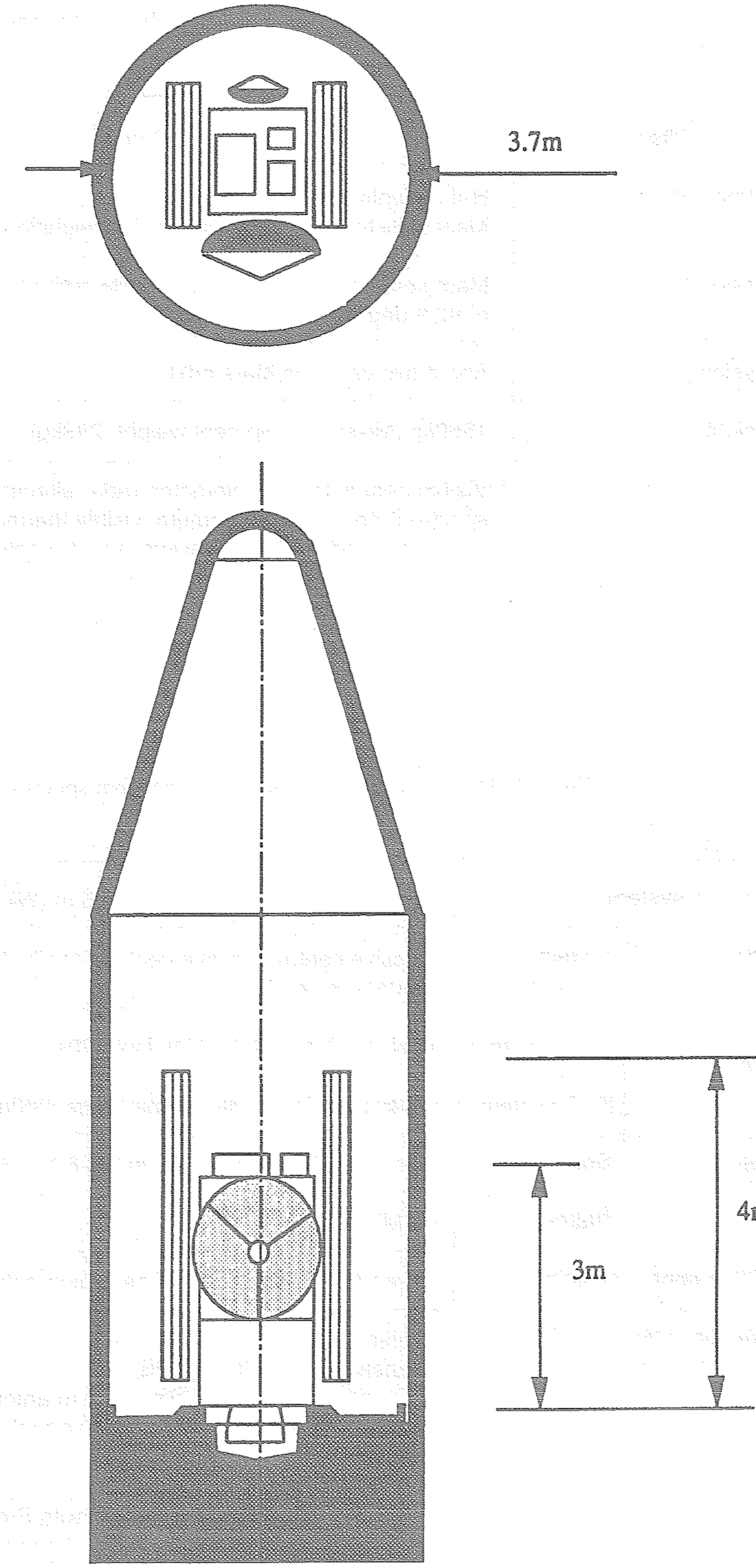


Fig. 5.3-3 Fairing storage (fairing of 4m)

Table 5.3-2 System overview of Mars observation satellite

Item	Contents
Launch vehicle	H-II derivative type of 15t class Mars transfer orbit entry with LE-5A reignition
Mission orbit	Mars solar-synchronized orbit, orbital altitude of 400 km, inclination of 92.9 degrees
Mission period	About two years on Mars orbit
Weight	1560kg (Mission equipment weight: 248kg)
Mission equipment	Visible near infrared radiometer, radar altimeter, gamma-ray spectrometer, shooting camera, visible thermal infrared radiometer, microwave radiometer, ultraviolet spectrometer, radiation monitor

Fig. 5.3-3Mars observation satellite subsystem specifications

Structure system		Frame + panel method, size: 1.5 m (W) × 1.2 m (D) × 1.5 m (H)
Thermal control system		Passive control + active control (conductance heat pipe + thermal louver + heater)
Propulsion system	LAPS system	N_2H_4/NTO, thrust: 2000N, Isp: 320s
	RCS system	Hydrazine, IN thruster, control rate: 200m/s
Power system	Solar cell	Semi-rigid BSFR type (1.7 m × 3.8 m × 3, two airfoils)
	Battery	Ni-MH 35Ah × 2
Attitude control system		Three-axis attitude control zero-momentum method
Communication and data processing system		(Telemeter) Transmission method: USB Transmission output: 20W, ø 1.9 m antenna Transmission amount should be variable depending on distance. (Mission data) X-band, transmission power: 40W, ø 1.9 m antenna shared with S-band Transmission amount should be changed depending on distance.

Table 5.3-4 Weight and electric power estimation of Mars observation satellite

			Weight (kg)	Power consumption (W)
Mission equipment	Visible near infrared radiometer		73	130
	Radar altimeter		25	35
	Gamma-ray spectrometer		30	20
	Shooting camera		15	20
	Visible thermal infrared radiometer		30	45
	Microwave radiometer		50	60
	Ultraviolet spectrometer		20	15
	Radiation monitor		5	5
	Subtotal		248	330
Bus equipment	Structure system		90	
	Thermal control system		35	91
	Propulsion system	LAPS system	200	
		RCS system	40	
	Solar cell paddle system		203	30
	Power supply system		142	18
	Attitude control system		91	119
	Communication data processing system		73	111
	Mission data processing and transmission system		136	581
	Instrumentation system		90	3
	Subtotal		1,100	953
Satellite dry weight			1,348	
Propellant weight (for RCS system)			115	
Margin			97	17
Total			1,560	1,300

6. Study on Mars atmospheric entry and landing mission spacecraft

6.1 Objectives

Mars atmospheric entry and landing mission spacecraft should acquire the technical data on aerobrake due to Mars atmosphere and perform reentry to land on Mars surface, acquiring the basic technical data for successive Mars sample return mission as well as environmental data on Mars surface. Their objectives are listed as follows:

(1) Mastering of Mars atmospheric aerobrake technique

(2) Mastering of Mars atmospheric reentry technique

(3) Observation of Mars atmosphere

(4) Acquirement of environmental data on Mars surface

(5) Analysis of Mars soil

6.2 Mission plans

6.2.1 Mission equipment

Table 6.2.1 lists proposed mission devices.

6.2.2 Observation plans

(1) Landing spot

Possible landing spots are polar area where water (ice) may exist, regions with dark green patterns which may allow vegetation and valleys and shoals which seem to have ever been rivers, lakes and oceans. Table 6.2.2 lists proposed landing spots.

(2) Data transmission

Observation data, etc. should be directly transmitted to the earth. Since the distance between the earth and Mars varies from 70 Mkm to 400 Mkm, data transmission rate should change depending on the distance. Assuming the earth station to be the DSN of NASA (antenna diameter of 70m), the transmission rate can be 128bps in most closeness and 2bps with the distance of 380 Mkm at the transmission output of 10W and frequency range of S band. However, during this mission period (three months), max. 10bps can only be achieved.

6.3 System study

(1) System analysis

Fig. 6.3-1 shows the mission profile of Mars atmospheric entry and landing mission spacecraft.

Sequence of events is indicated in table 6.3-1. For propellant weight, a margin of 5% is allowed for each event.

(2) System study

Fig. 6.3-2 shows the system plot. As the main objective is to acquire Mars atmospheric entry technique, its aeroshell shape is analogous to that of Mars sample return system. In order that the aerodynamic heating ratio becomes equivalent to that of Mars sample return spacecraft, scale ratio is set to 0.42.

Table 6.3-2 and 3-3 indicate the system overview and main characteristics of subsystems, respectively.

Weight and electric power estimation is as shown in table 6.3-4.

(3) Rocket system

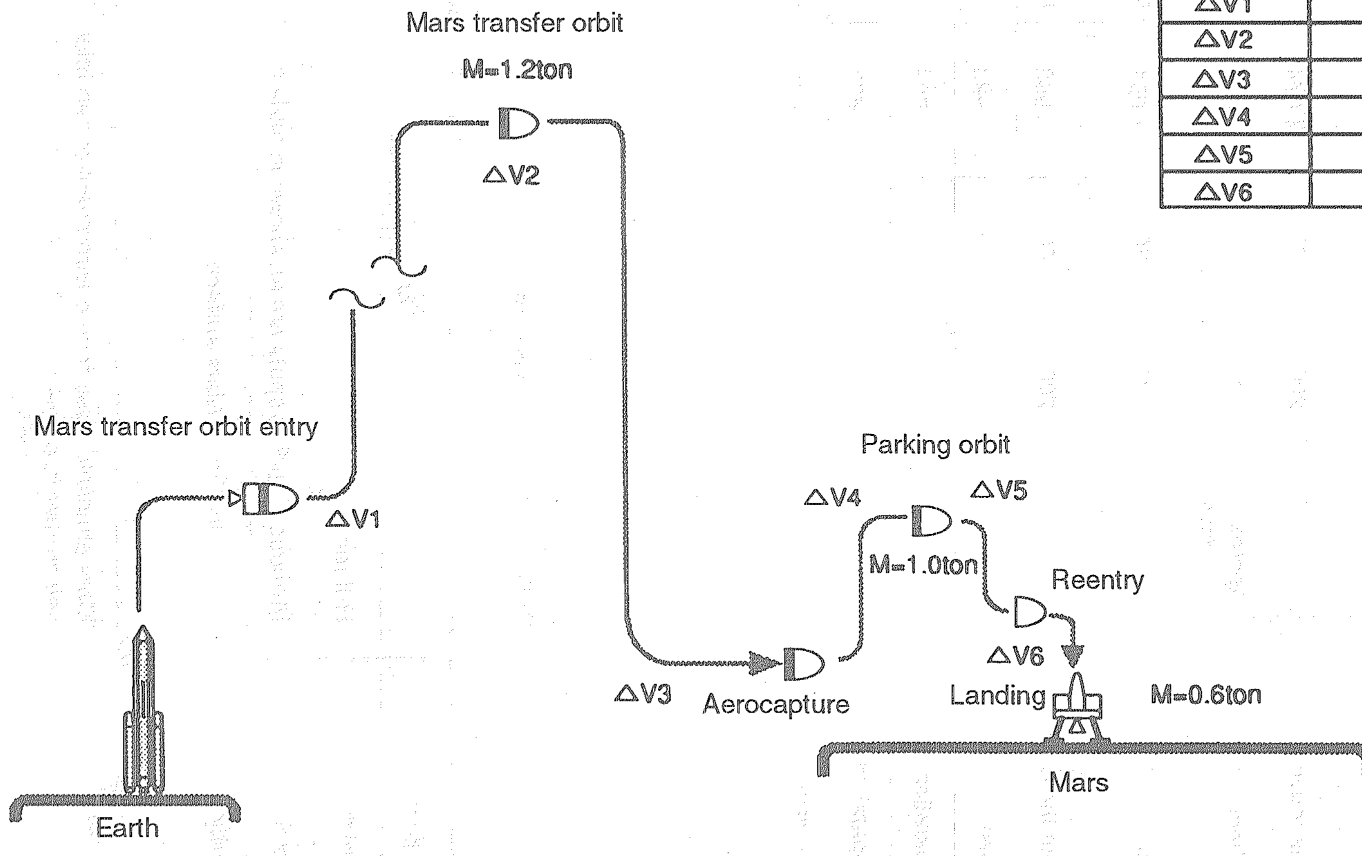
H-II type rocket should be assumed as the rocket system.

Table 6.2-1 Proposed mission devices for Mars atmospheric entry and landing mission spacecraft

Mission device	Observaton purpose	Weight (kg)	Power consumption (W)	Data capacity
Atmospheric observation devices (pressure, temperature, acceleration)	Observation of Mars atmospheric structure	1.5	6.2	65bps
Surface environment observation devices (pressure, temperature, acceleration)	Observation of Mars surface environment	1.0	0.1	10kbits/day
Seismometer	Observation of Mars quakes	1.5	2.0	10Mbits/day
α-P-X spectrometer	Major element composition analysis	1.0	0.5	100kbits
Thermal analyzer/gas analyzer	Analysis of Mars soil ingredients and constituent gases	2.0	12.0	3Mbits
Camera (for use upon landing)	Geographical observation from upper air	0.5	4.0	12Mbits/12 images
Camera (for use after landing)	Mars surface observation	1.5	21.0	25Mbits/one revolution

Table 6.2-2 Proposed landing spots

Landing spot name	Position	Characteristics
North polar cap	North pole	Climate investigation in polar area
Cydonia Mensae	latitude: 30-43° longitude: 0-20°	Regions where dark green patterns appear, indicating the possibility of vegetation. They had the first priority as proposed landing spots for Viking 2 rocket.
Valles Marineras	latitude: -18-1° longitude: 24-113°	Valleys which can be considered as previous rivers with the possibility of creature fossil discovery
Terra Meridiani	latitude: -15-0° longitude: 341-17°	Regions where dark green patterns appear, indicating the possibility of vegetation. They were auxiliary landing spots for Viking 1 & 2 rockets.
Isidis Platinia	latitude: 4-20° longitude: 255-279°	Regions with bay-like shapes, separating those areas which seem dark green. They had the second priority as proposed landing spots for Viking 2 rocket.
Margaritifer Terra	latitude: -27-2° longitude: 12-45°	Regions where dark green patterns appear, indicating the possibility of vegetation. They were auxiliary landing spots for Viking 1 & 2 rockets.
Terra Sabaea	latitude: -20-0° longitude: 348-3°	Regions where dark green patterns appear, indicating the possibility of vegetation.



Incremental speed request

Symbol	Incremental speed (m/s)
ΔV1	3614
ΔV2	200
ΔV3	20
ΔV4	100
ΔV5	200
ΔV6	270

Fig. 6.3-1 Entire mission profile of Mars atmospheric entry and landing mission aircraft

Table 6.3-1 Sequence of events

Event	Period	Incremental speed (m/s)	Propellant weight (kg)	Current weight (kg)	Remarks
(1) Mars transfer orbital entry		3,614		1,170	H-II second stage reignition
(2) Mid-course maneuver		200	76	1,094	ISP = 320 sec
(3) Mars atmospheric entry	Approx. 300 days	20	7	1,087	ISP = 320 sec
(4) Periapsis ascent maneuver		100	36	1,051	ISP = 320 sec
(5) Departure from the orbit		200	68	983	ISP = 320 sec
(6) Parachute open		—	—	983	
(7) Aeroshell separation		—	—	723	
(8) Parachute separation		—	—	683	
(9) Landing power flight		270	59	624	ISP = 320 sec
(10) Touch down		—	—	624	

Table 6.3-2 System overview

Item	Contents
Launch vehicle	H-II rocket
Landing point	Proposed landing spots are as shown in table 6.2-2.
Mission period	Three months on Mars surface
Weight	1.2t
Mission equipment	Mars atmosphere/surface environment observation devices, seismometer, camera, α-P-X spectrometer, thermal analyzer/gas analyzer

Table 6.3-3 Main characteristics of subsystem

Subsystem	Main characteristics
Structure system	Main structure: Truss + panel structure Aeroshell: Ablator + honey-comb panel structure
Thermal control system	Combination of passive and active types
Attitude control system	Three-axis attitude controlled zero-momentum method
Communication system	S band omniantenna
Propulsion system	Main propulsion system: Thrust of 2000N, propellant of NTO/N_2H_4 RCS: 4N × 16 units, propellant of N_2H_4
Power system	Solar cell: BSFR type Si cell, generated power of 200W Battery: NiH_2, capacity of 2200Wh
Deceleration system	Parachute aperture area: 4000m^2 (diameter of 70m)

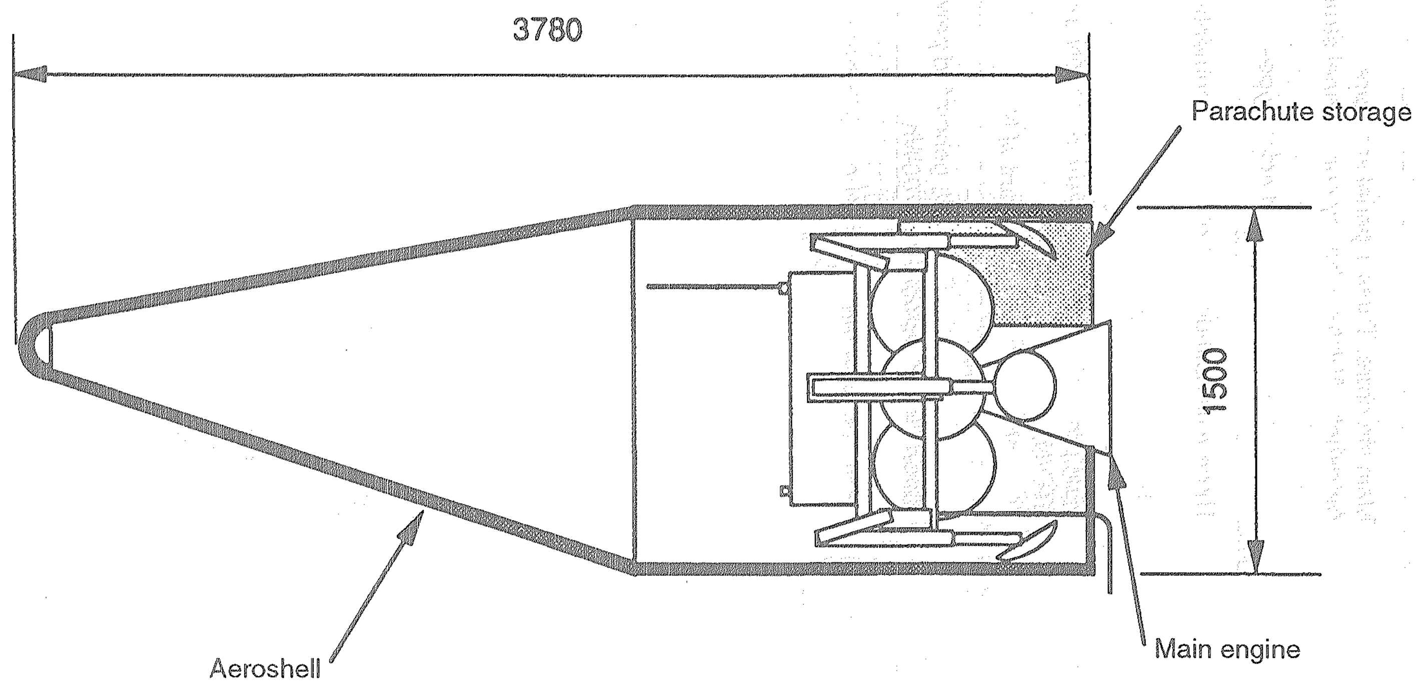


Fig. 6.3-2 System plot for Mars atmospheric entry and landing mission spacecraft

Table 6.3-4 Weight and electric power estimation

Item		Weight (kg)	Power consumption (W)
Mission equipment	Atmospheric observation device	1.5	6.2
	Surface environment observation device	1	0.1
	α-P-X spectrometer	1	0.5
	Thermal analyzer/gas analyzer	2	12
	Seismometer	1.5	2
	Camera (for use upon landing)	0.5	4
	Camera (for use after landing)	1.5	21
	Fixture	2	—
	Subtotal	11	46
Bus equipment	Attitude control system	91	73
	Communication system	66	29
	Propulsion system	90	100
	Power system	149	77
	Thermal control system	50	10
	Structure system	80	—
	Aeroshell	260	—
	Parachute	40	—
	Subtotal	826	289
Dry weight		837	—
Propellant	NTO/N_2H_4	246	—
	GHe	1	—
Margin		86	—
Total		1,170	335

7. Study on Mars sample return system

7.1 Preface

This chapter provides study on Mars sample return mission focusing on the mission profile and submits requirements for transportation system in near-earth area. For Mars sample return mission, various studies have been performed in many countries. The description in this chapter is based on the mission profile of "A MARS SAMPLE RETURN MISSION USING A ROVER FOR SAMPLE ACQUISITION" (AAS 84-159) studied by NASA.

7.2 Scope of study

7.2.1 Premise for study

The following conditions were included as premise for study according to "A MARS SAMPLE RETURN MISSION USING A ROVER FOR SAMPLE ACQUISITION" by J.P. de Vries and H.N.Norton, AAS84-159, summarizing a collaboration research of JPL, JSC and SAI.

- Recovered sample weight: 5kg
- Mars rover (weight of 400kg) should performs in-situ analysis on Mars surface to select and collect valuable samples.
- Mars orbital entry should be performed by aerocapture.
- Mission sequence should be as per fig. 7.2-1. (Dates are indicated as examples.)

Phase	Departure date	Arrival date	Period
Earth → Mars	Nov. 18, '96	Sep. 17, '97	303 days
Stay on Mars			401 days
Mars → earth	Oct. 23, '98	Sep. 14, '99	326 days
		Total	1030 days

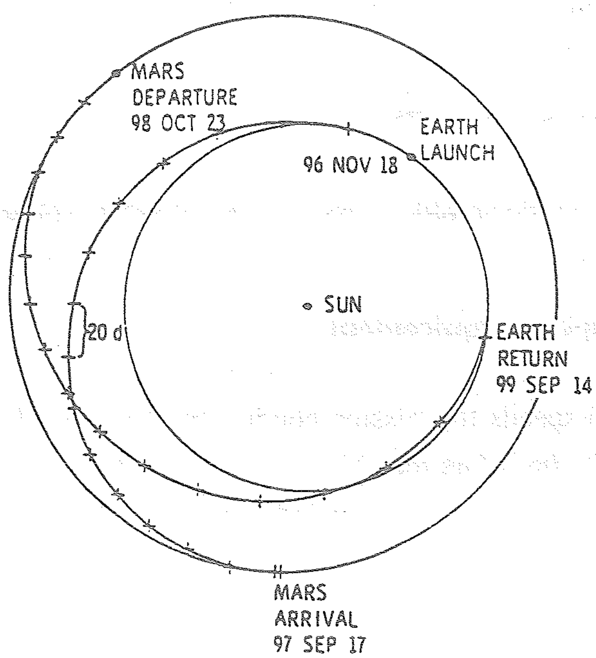


Fig. 7.2-1 Mission sequence

7.3 Result of study

7.3.1 Mission profile

(1) Entire profile

Fig. 7.3-1 shows the entire mission profile.

Various mission profiles for Mars sample return can be considered depending on forms upon departure from the earth, sample recovery methods upon return to the earth and Mars aerocapture vehicle shapes, etc. In this chapter, a representative reference proposal is provided. (For the study on each form, refer to attached information 8.)

(2) GTO recovery spacecraft profile

Fig. 7.3-2 indicates the mission profile for recovery spacecraft traveling to GTO to fetch samples.

7.3.2 Incremental speed requirement

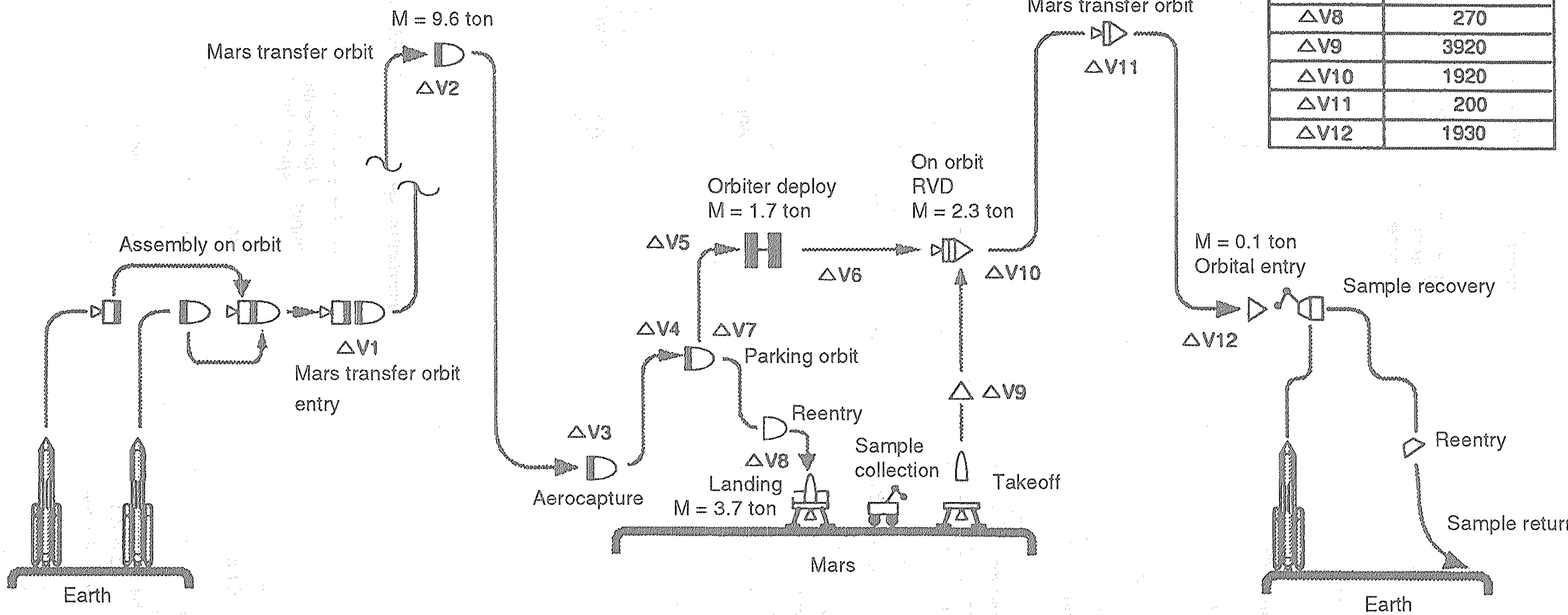
Required incremental speeds for mission conducting systems (all systems of transporting samples to earth GTO after Mars transition) are listed in table 7.3-1. Required incremental speeds for GTO recovery spacecraft are included in table 7.3-2.

Table 7.3-1 Required incremental speeds for mission conducting systems

Event	Symbol	Incremental speed (m/s)	Period	Remarks
(1) Mars transfer orbit entry	ΔV1	3610		Entry should be started with h=500 km and i=28.5.
(2) Mid-course maneuver	ΔV2	200		
(3) Mars aerocapture	ΔV3	20	Approx. 300 days	1900 m/s in case of using propulsion system
(4) Periapsis ascent maneuver	ΔV4	100		560 km × 2000 km
(5) Mars circular orbit entry	ΔV5	250		Circular orbit with 560 km diameter, M = 1.7t
(6) On-orbit attitude control	ΔV6	150		
(7) Departure from Mars elliptical orbit	ΔV7	200		
(8) Mars landing maneuver	ΔV8	270		M = 3.7t
(9) Mars takeoff - Mars circular orbit entry	ΔV9	3920		Three stage type
(10) Earth transition orbit entry	ΔV10	1920	Approx. 700 days	M = 0.5t
(11) Mid-course maneuver	ΔV11	200		
(12) Earth parking orbit	ΔV12	1930	Approx. 1000 days	Elliptical orbit of 280 × 40200 km In case of LEO entry: 3700 m/s, M = 0.1t

Table 7.3-2 Required incremental speed for GTO recovery spacecraft

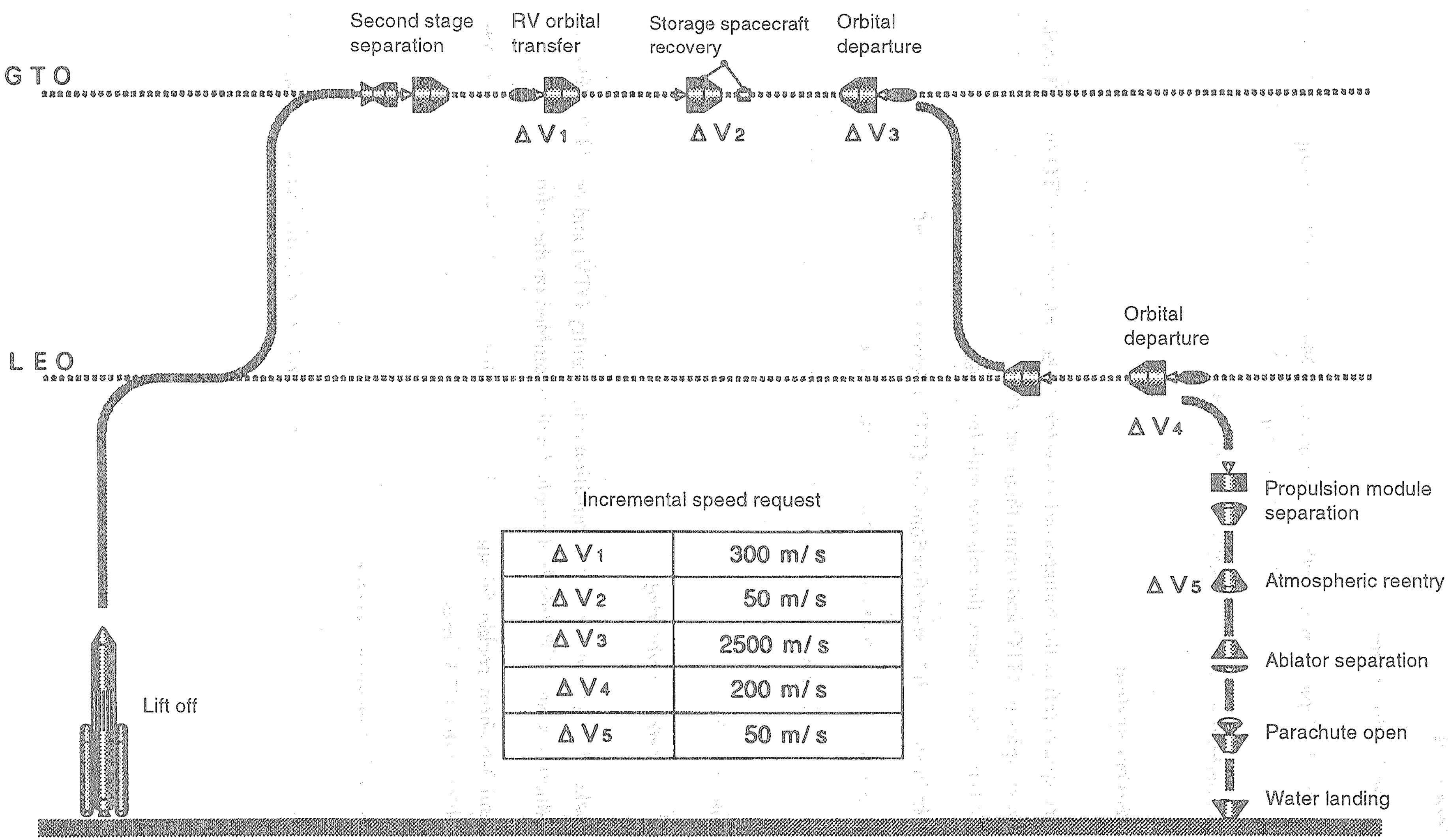
Event	Symbol	Incremental speed (m/s)	Remarks
(1) RV orbital transfer	ΔV_1	300	
(2) Access, berthing	ΔV_2	50	
(3) GTO orbital departure/LEO entry	ΔV_3	2500	Elliptical orbit of 280 × 40200km
(4) LEO orbital departure	ΔV_4	200	
(5) Attitude control upon reentry	ΔV_5	50	



Incremental speed request

Symbol	Incremental speed (m/s)
ΔV1	3610
ΔV2	200
ΔV3	20
ΔV4	100
ΔV5	250
ΔV6	150
ΔV7	200
ΔV8	270
ΔV9	3920
ΔV10	1920
ΔV11	200
ΔV12	1930

Fig. 7.3-1 Entire mission profile of integrated GTO recovery type



Incremental speed request

ΔV_1	300 m/s
ΔV_2	50 m/s
ΔV_3	2500 m/s
ΔV_4	200 m/s
ΔV_5	50 m/s

Fig. 7.3-2 Mission profile of GTO recovery aircraft

7.3.3 System study

(1) Mission conducting system

This system should depart lower earth orbit to arrive at Mars and collect samples which are recovered at GTO. The table 7.3-3 shows the system configuration and its weight.

The mission conducting system is indicated in fig. 7.3-3.

(2) GTO recovery system

This recovery spacecraft is the transportation system to be entered into GTO by H-II rocket, recover Mars samples on GTO and return them onto the earth. The recovery spacecraft should be of capsule type easy to carry the mission with low cost.

Table 7.3-4 indicates weight specifications for GTO recovery spacecraft.

The GTO recovery spacecraft plot is indicated in fig. 7.3-4.

The study on GTO recovery system is provided in attached information 8, addendum 3.

7.3.4 Rocket system

(1) Launch of mission conducting system

One LEO rocket of 20t class (for Mars transition orbital entry OTV) and one H-II rocket (for mission conducting system) should be launched and assembled on the orbit.

Fig. 7.3-5 and 3-6 show rocket fairing storages for mission conducting system and Mars transition orbital entry OTV, respectively.

(2) Launch of GTO recovery system

GTO recovery system should be launched by H-II rocket. H-II rocket fairing storage is as shown in fig. 7.3-7.

(3) Mars transition orbital entry OTV

Thrust	: 12.5t
Specific thrust	: 452 seconds
Weight	: 18880kg
Propellant weight	: 16048kg
Structural efficiency	: 0.85

Since the propellant weight is almost the same as that of H-II rocket's second stage, this stage should be used as it is for Mars transition orbital entry OTV. However, small changes such as addition of RVD mechanism for on-orbit assembly would be required.

Table 7.3-3 System configuration and weight

Component								Weight (kg)	Remarks
Departure time	Mission conducting system	Mars orbit parking & earth return system	Mars orbit parking satellite	Earth transfer orbit parking spacecraft	Earth return spacecraft	SCA	Sample	5	
							Case	15	
						Hardware		118	See attached information 8, addendum 2
						Propellant		117	NTO/N_2H_4, ISP = 320 sec
						Subtotal		255	
					Hardware			208	
					Propellant			541	Solid, ISP = 280
					Subtotal			1004	
				Hardware				592	
				Propellant				1133	NTO/N_2H_4, ISP = 320 sec
				Subtotal				2729	
			Mars entry aeroshell					617	
			Subtotal					3346	
		Adapter						177	
		Mars landing/takeoff system	Mars lander	Takeoff spacecraft	Hardware			690	
					Propellant			1242	Solid, ISP = 280
					Subtotal			1932	
				Booster	Hardware			250	
					Propellant			551	Solid, ISP = 280
					Subtotal			801	
				Mars rover				400	
				Hardware				617	
				Propellant				339	NTO/N_2H_4, ISP = 320 sec
				Subtotal				4089	
			Mars entry spacecraft	Aeroshell				890	
				Parachute				327	
				Hardware				272	
				Propellant				368	NTO/N_2H_4, ISP = 320 sec
				Subtotal				1857	
			Bioshield					148	
			Subtotal					6094	
		Subtotal						9617	
	Adapter							276	
	OTV	Hardware						2832	Structural efficiency: 0.85
		Propellant						16048	LOX/LH_2, ISP = 452 sec
		Subtotal						18880	
	Total							28773	
Recovery spacecraft launch upon return time								4254	See attached information 8, addendum 3

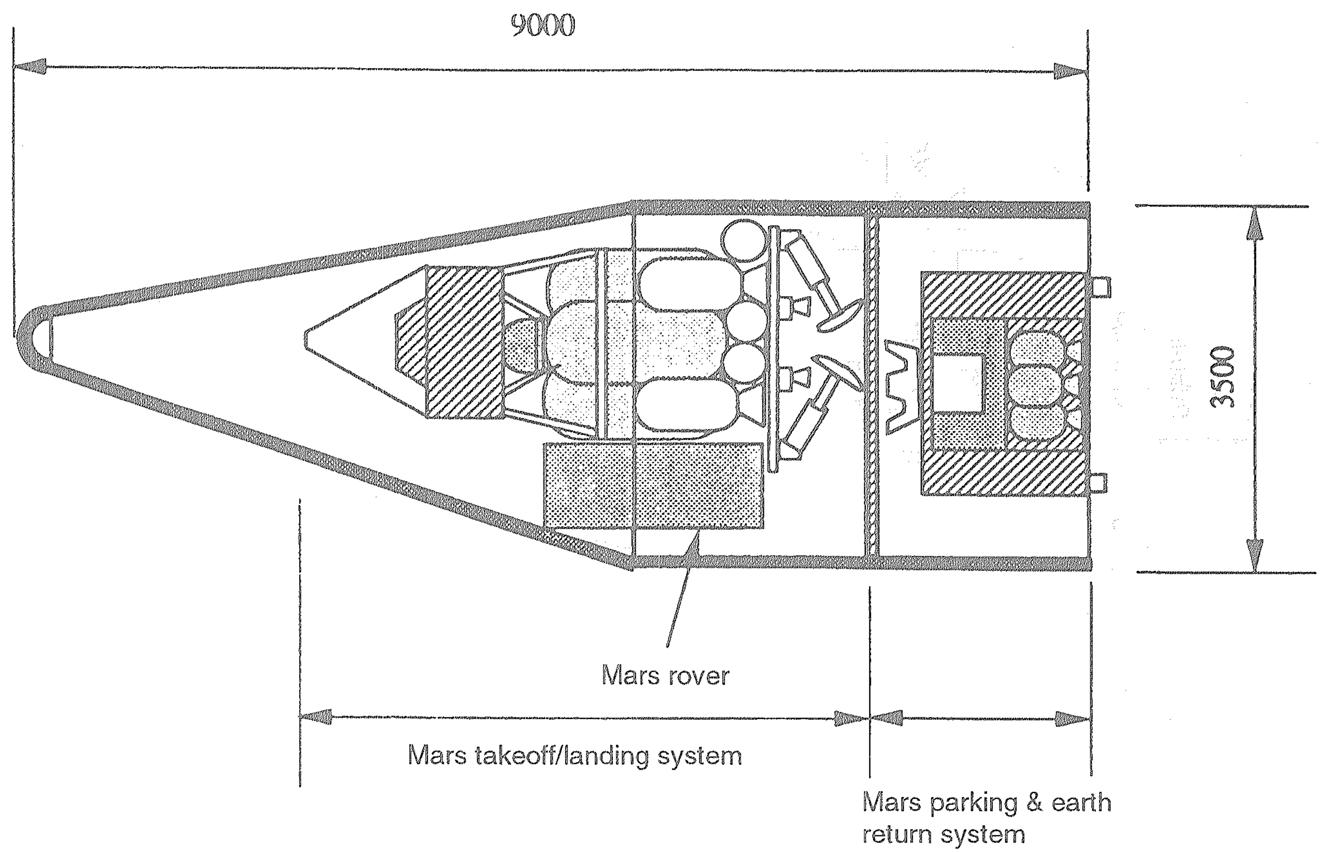


Fig. 7.3-3 Mission conducting system plot

Table 7.3-4 Weight specifications for GTO recovery spacecraft

Module	System	Weight (kg)
Capsule	Dry weight	1163
	Propellant	27
	GHe	0.1
	Subtotal	1190
Propulsion module	Dry weight	737
	Propellant	2319
	GHe	8
	Subtotal	3064
	Total	4254

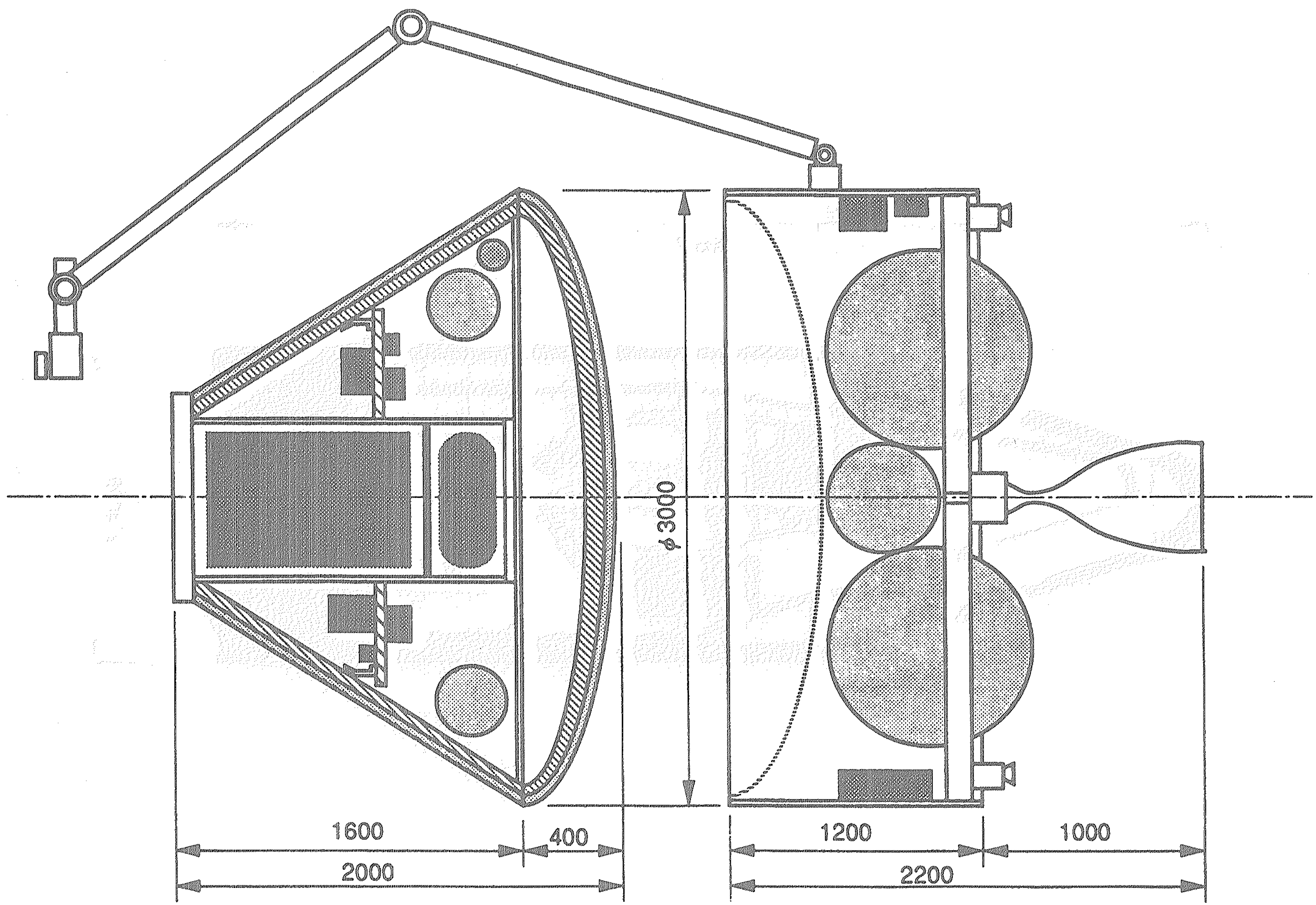


Fig. 7.3-4 GTO recovery spacecraft plot

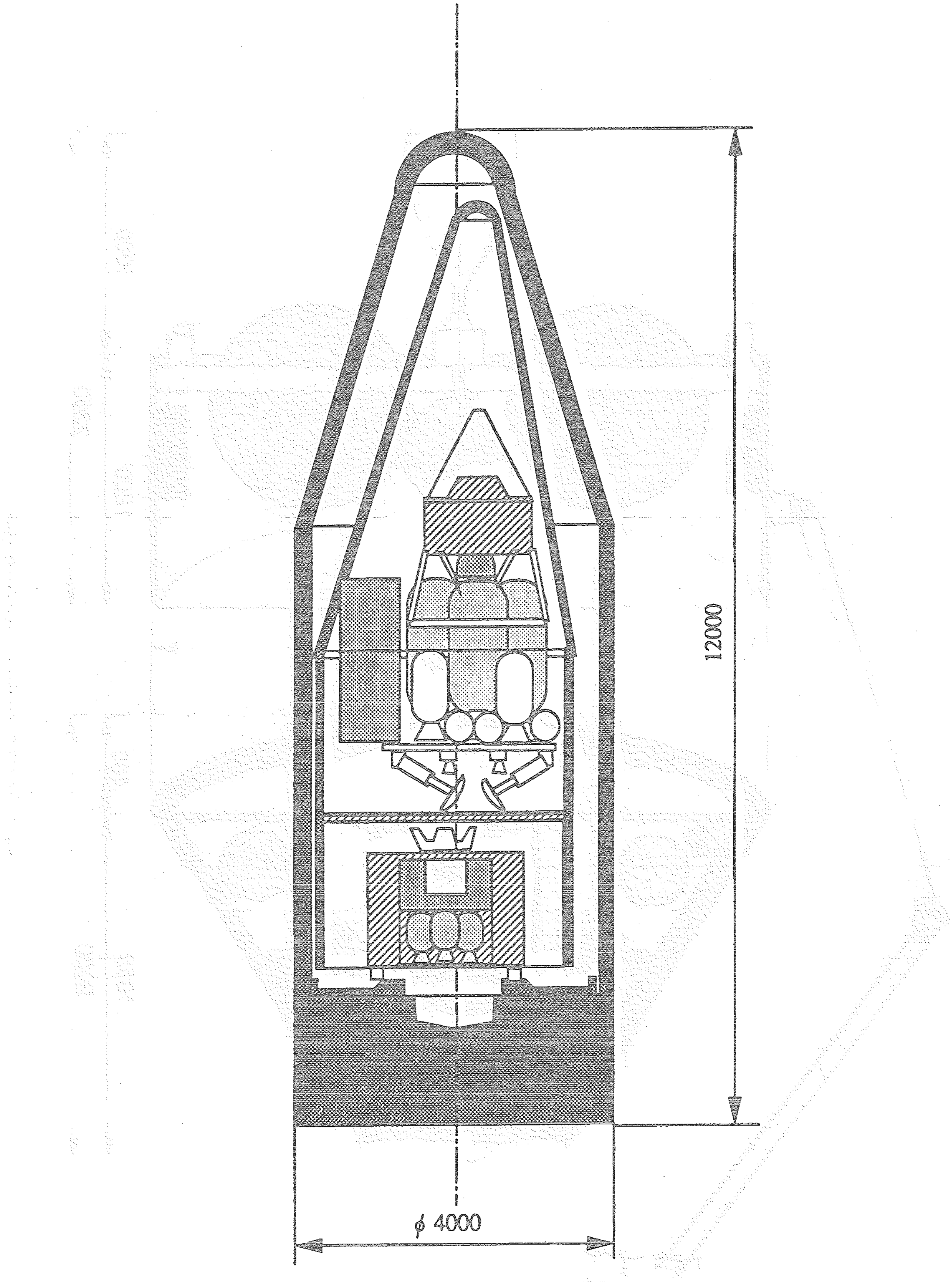


Fig. 7.3-5 Mission conducting system fairing storage

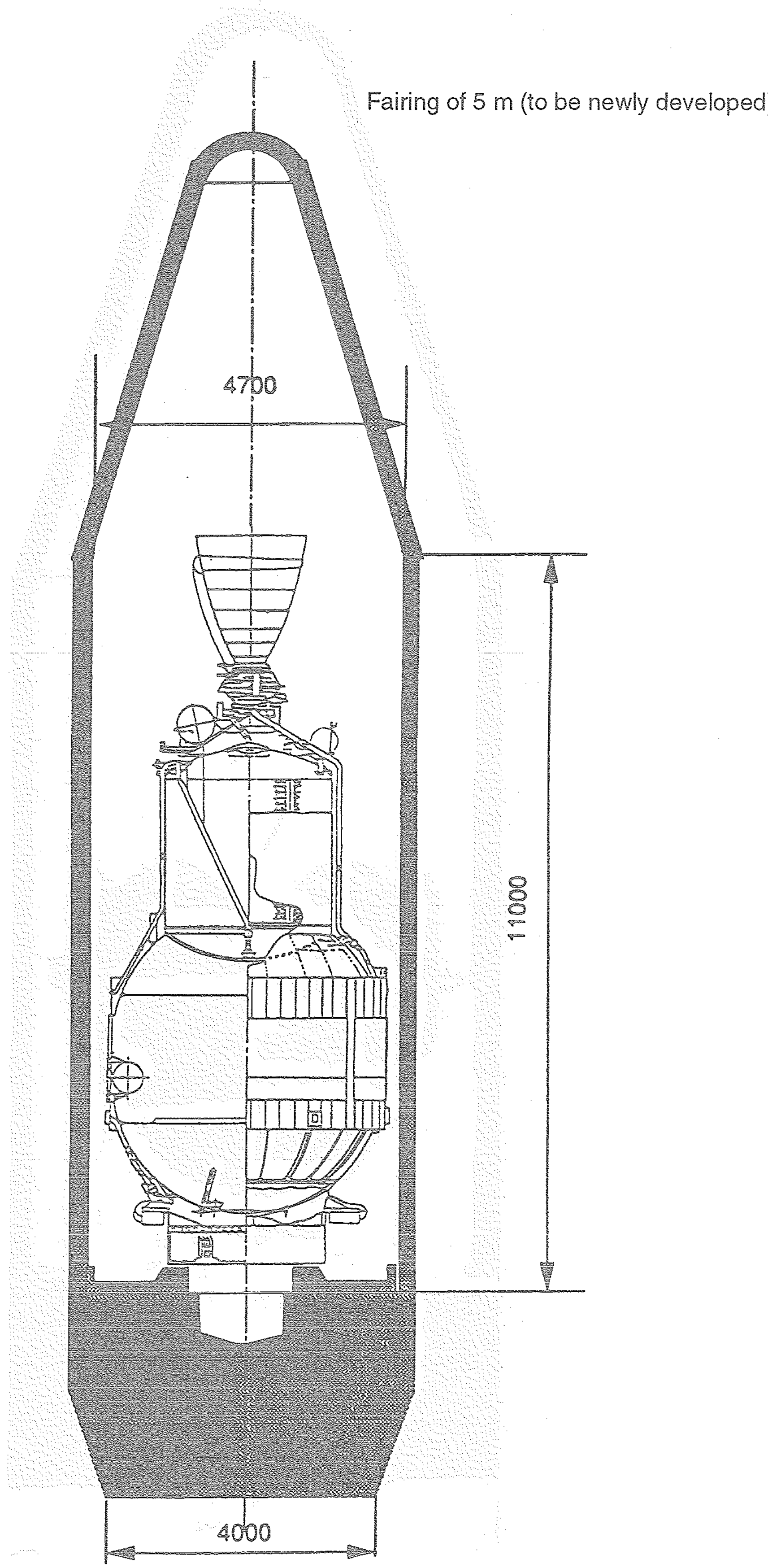


Fig. 7.3-6 Upper stage OTV fairing storage

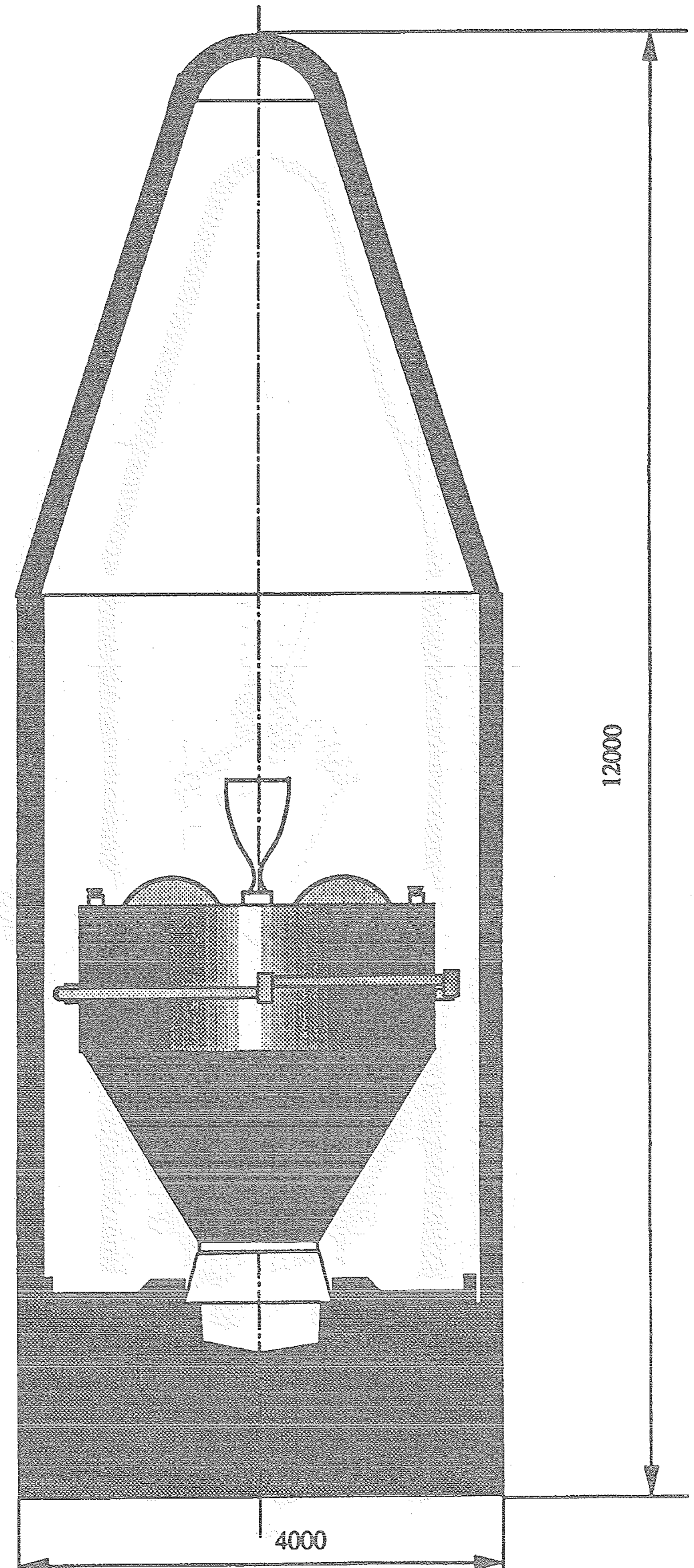


Fig. 7.3-7 GTO recovery spacecraft fairing storage

7.3.5 Review on reliability

Primarily studyed outline on the reliability of this system is described as follows:

Mission conducting system	: 0.357
GTO recovery system	: 0.855
Entire system	: 0.31

The reliability of entire system has proved to be very low. It is caused by the fact that this sample return system consists of many spacecrafts. This study is a basic one requiring further detailed analysis, however, it would provide guidelines for such a complexed system's reliability.

Detailed study is included in attached information 8.

[Reference]

1) J.P.de Vries, H.N.Norton "A MARS SAMPLE RETURN MISSION USING A ROVER FOR SAMPLE AQUISITION" (AAS 84-159)

2) GLENN E.CUNNINGHAM, DONALD G.REA, DONNA PIVIROTTO, JOHNNY KWOK, MARK K.CRAIG, MICHAEL H.CARR "MARS ROVER SAMPLE RETURN MISSIONS" (IAF-88-398)

3) James E. Pavlosky, Leslie G.St.Leger "APOLLO EXPERIENCE REPORT - THERMAL PROTECTION SUBSYSTEM" (NASA-TN-D-7564)

4) Hidehiko Mori, Keiji Nitta, Tatsuo Yamanaka "REVIEW ON CONCEPT OF GEMINI TYPE CAPSULE RECOVERY AIRCRAFT", May 1981

8. Transportation system for lunar and Mars exploration

8.1 Preface

In order to conduct lunar and planet exploring missions as discussed in previous seven chapters, those rockets whose abilities exceed H-II rocket currently largest in Japan would be necessary depending on the case. In this technical memorandum, currently proposed H-II derivative type rockets are used, and upper stage OTVs should be used in case of requiring further ability. The upper stage OTV means a disposable transportation spacecraft which helps the mission conducting system enter from lower orbit into lunar or Mars transfer orbit.

8.2 Each mission request

Table 8.2-1 lists transition orbital entry weight requests for each lunar and Mars exploring mission.

Table 8.2-1 Weight on transfer orbit

Mission	Weight on transfer orbit (ton)
(1) Lunar observation satellite	2.8
(2) Electric propulsion orbital transfer mission spacecraft	1.3
(3) Lunar lander/rover	2.8
(4) Lunar sample return	7.7
(5) Mars observation satellite	3.7
(6) Mars atmospheric entry & landing mission spacecraft	1.2
(7) Mars sample return	9.6

8.3 Launch abilities of rockets

Table 8.3-1 shows launch abilities of currently proposed rockets.

Abilities for lunar transfer orbit entry and Mars transfer orbit entry are those obtained from reignition of second stage.

Table 8.3-1 Blast-off abilities of rockets

Rocket name	LEO launch ability (ton)	Lunar transfer orbit entry ability (ton)	Mars transfer orbit entry ability (ton)
H-II rocket	10.5	3.0	2.0
H-II derivative type rocket of 15t class	15	5.6	3.7
H-II derivative type rocket of 20t class	20	7.7	5.3

8.4 Proposed transportation system for each mission

Table 8.4-1 lists proposed transportation systems for each mission.

Table 8.4-1 Proposed transportation system for each mission

Mission	Launch vehicle	Transfer orbit entry form
(1) Lunar observation satellite	One H-II rocke	Second stage reignition
(2) Electric propulsion orbital transfer mission aircraft	One H-II rocket	Ionic engine
(3) Lunar landing/mobile exploration aircraft	One H-II rocket	Second stage reignition
(4) Lunar sample return	One H-II derivative type rocket of 20t class	Second stage reignition
(5) Mars observation satellite	One H-II derivative type rocket of 15t class	Second stage reignition
(6) Mars atmospheric entry & landing mission spacecraft	One H-II rocket	Second stage reignition
(7) Mars sample return	One H-II rocket (for mission) One H-II derivative type rocket of 20t class (for upper stage OTV)	Upper stage OTV (H-II second stage)

9. Development plan

In this chapter, lunar and Mars unmanned exploration plan proposed as the first period program for lunar and Mars exploitation project is studyed from the technical point of view.

9.1 Technical development scenario

9.1.1 Development issues

(1) Lunar observation satellite

- Lunar transfer orbit and lunar orbit entry
- Tracking control/operation
- Mission equipment

(2) Electric propulsion orbital transfer mission spacecraft

- Lunar transfer orbit and lunar orbit entry by low thrust/gravity capture
- Tracking control/operation
- Large size ionic engine
- Mission equipment

(3) Lunar lander/rover

1) Lunar lander

- Aviation, guidance and control upon landing including hovering
- Obstacles avoidance upon landing
- Throttleable engine

2) Lunar rover

- Nighttime (for 14 days) thermal control technique
- Lunar surface running mechanism and driving system including bearing and decelerator
- Remote control with time delay and semi-automatic control
- Mission equipment

(4) Lunar sample return

- Lunar takeoff and earth return orbit entry
- Capsule design
- Ablator
- Water landing and recovery system

(5) Mars observation satellite

- Mars transfer orbit and Mars orbit entry
- Tracking control/operation
- Mission equipment

(6) Mars atmospheric entry and landing mission spacecraft

- Mars atmospheric aerobrake
- Mars atmospheric entry and landing
- Mission equipment

(7) Mars sample return

- Assembly on earth orbit
- Highly automatic controlled rover
- Measurement positions for Mars takeoff spacecraft and orbiter
- Highly automatic controlled rendezvous docking on Mars orbit
- Mars takeoff and earth return orbit entry
- Mission equipment
- Large size launch vehicle

9.1.2 Development steps

(1) Both lunar and Mars explorations should be steadily developed step by step as follows:

- Lunar exploration step

  Lunar observation satellite → Lunar lander/rover → Lunar sample return

- Mars exploration step

  - Atmospheric entry and landing mission should be performed for technical development of Mars atmospheric aerobrake and atmospheric reentry as well as landing.

  - Mars observation satellite → Mars atmospheric entry and landing mission spacecraft → Mars sample return

(2) Mars exploration should reflect the technical results of lunar exploration.

- Lunar observation satellite → Mars observation satellite
- Lunar rover → Mars rover
- Lunar sample return → Mars sample return

(3) Highly automatic controlled rendezvous docking should be performed based on the technique developed in the on-orbit service system.

9.2 Development schedule

Table 9-1 summarizes the study on the entire schedule in the following concept. In this schedule, each project operation indicated in development steps is technically sorted out and comprehensive planning should be designed including political view points.

1) Even in the latest case, CDR should reflect the result of previous step project.
2) The development of lunar observation satellite should be started (start of basic design) in the year of Heisei 7.
3) Mars exploration project should be started in early Heisei 8 year. It would allow the result of lunar sample return to be reflected in Mars sample return.
4) The window of Mars transfer from the earth should be set to approx. two years, vice versa.

9.3 Development cost

Development cost is indicated in technical information RS-S94007 "Study on the development cost for lunar and Mars explorations".

Table 9-1 Review on the entire schedule of lunar and Mars explorations

Heisei year	6				10					15					20					25		
Dominical year	'94						2000					'05					'10					'15
1. Lunar exploration																						
(1) Lunar observation satellite		△ SRR		△ PDR	△ CDR		△ LO		△ EOM													
(2) Electric propulsion orbital transfer mission Spacecraft		△ SRR		△ PDR	△ CDR		△ LO	△ LOI	△ EOM													
(3) Lunar lander/rover																						
∘ No.1						△ SRR		△ PDR	△ CDR		△ LO	△ EOM										
∘ No.2							△ SRR		△ PDR	△ CDR		△ LO	△ EOM									
(4) Lunar sample return								△ SRR	△ PDR		△ CDR		△ LO	△ ER								
2. Mars exploration																						
(1) Mars observation satellite			△ SRR	△ PDR		△ CDR		△ LO	△ MOI		△ EOM											
(2) Mars atmospheric entry and landing mission spacecraft						△ SRR		△ PDR		△ CDR		△ LO	△ ML									
(3) Mars sample return									△ SRR		△ PDR		△ CDR			△ LO	△ ML	△ MLO	△ ER			

Note: Abbreviation

LO: Lift Off
ML: Mars Landing
EOM: End of Mission
MLO: Mars Lift Off

LOI: Lunar Orbit Insertion
ER: Earth Return
MOI: Martian Orbit Insertion

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