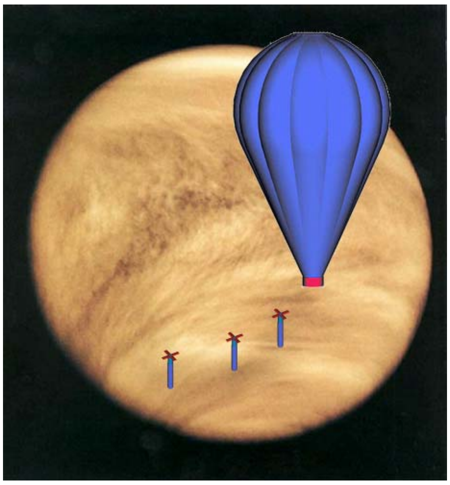
STUDY OVERVIEW OF THE VENUS ENTRY PROBE



An ESA Technology Reference Study

Planetary Exploration Studies Section (SCI-AP)
Science Payload and Advanced Concepts Office (SCI-A)

prepared by/*préparé par*	Marcel van den Berg and Peter Falkner
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European Space Agency
Agence spatiale européenne

ESTEC
Keplerlaan 1 - 2201 AZ Noordwijk - The Netherlands
Tel. (31) 71 5656565 - Fax (31) 71 5656040

Venus Entry Probe Technology Reference Study – Study Team		
ESA	Marcel van den Berg Peter Falkner Andre Schiele Arnaud Boutonnet	VEP TRS study manager Technical officer VEP TRS activities Technical officer microprobe project Mission analysis validation
EADS Astrium Ltd.	Steve Kemble	Mission analysis
Surrey Satellite Technology Ltd.	Andy Phipps Adrian Woodroffe Dave Gibbon Peter Alcindor Craig Clark Nadeem Ghafoor Alex Cropp Carlos Lovett Lineares John Buckley Yoshi Hashida Tanya Vladimirova Adam Baker Jackie Brooks Alex da Silva Curiel Jim Clemmet Andrew Cawthorne Syed Husnain Phil Whittaker	Study manager OBDH Propulsion Power Power Payloads/Science ACS Communications Projects Trajectories Advanced Technologies Structure/Micro-power technologies Project Assistant Research and Development Structure/Configuration Thermal Thermal Navigation
Vorticity Ltd.	Steve Lingard John Underwood Nick Bown	Systems Engineering Systems Engineering (entry vehicle and aerobot) Space inflatables
Fluid Gravity Engineering	Arthur Smith Gavin Parnaby	Thermal Protection System analysis validation Thermal Protection System analysis validation
Cosine Research B.V.	Stefan Kraft Joe Moorhouse	Project manager Payload assessment and definition
Swiss Space Technology	Julian Harris	Payload electronics assessment and design
QinetiQ Ltd. ***(Microprobe design)***	Nigel Wells Andrew Ballard John Doherty Andrew Eldridge	Project manager Microprobe communications Microprobe aerodynamics Microprobe aerodynamics

Venus Entry Probe Technology Reference Study – Mission Summary				
Key scientific objectives	• Detailed study of the Venus atmosphere: o Origin and evolution of the atmosphere o Composition of the lower atmosphere o Atmospheric dynamics and thermal balance o Aerosol analysis			
Strawman reference payloads assumed for this study	• ***Venus Polar Orbiter (VPO):*** Sub-mm wave sounder, visible-NIR imaging spectrometer, UV spectrometer, IR Fourier transform spectrometer, UV-visible-NIR camera • ***Aerobot:*** Gas chromatograph mass spectrometer (with aerosol inlet), nephelometer, solar and IR flux radiometers, meteorological and inertial packages, radar altimeter • ***Microprobes:*** Thermometer, pressure sensors, solar flux sensors, (wind velocity)			
Launch and transfer	• Launch of 1509 kg into direct Venus escape by Soyuz-Fregat 2-1B (Kourou)(2-11-2013) • Type II transfer (160 days) and Venus capture by chemical propulsion • VPO and VEO (Venus Elliptical Orbiter) interplanetary cruise as separate modules • System level mass margin 20%			
Entry and descent	• Entry probe released from VEO (90 to 180 days after Venus arrival) • Thermal protection system based on a high density ablator (entry angle ~ 40˚) • Parachute deployment at 1.5 Mach			
Aerobot	• Hydrogen filled superpressure balloon • During flight, the balloon will drop atmospheric microprobes			
VPO science acquisition	• Remote sensing science acquisition concurrent with aerobot operational phase • Almost continuous monitoring of the Venus atmosphere (duty cycle 99.9%)			
S/C Modules	**VPO**	**VEO**	**Aerobot**	**15 microprobes**
Stabilization	3-axis	3-axis	-	-
Orbit/Altitude	2,000 km × 6,000 km	400 km × 215,000 km	55 km	55 – 10 km
Initial inclination	90˚	64˚	Deployment: 20±2˚ N	
S/C ΔV requirements	3.5 km/s	1.7 km/s		
Operational lifetime	> 2 years	> 2 years	15 – 22 days	< 1 hr
Platform dry mass (excl. P/L)	222 kg	183 kg	15 kg (gondola)	115 g (each)
P/L mass	25 kg	91 kg (entry vehicle)	4 kg (P/L) + 4 kg (microprobes)	< 10 g
Total wet mass (incl. 20% system margin)	905 kg	558 kg (incl. entry vehicle)	32 kg (aerobot)	4 kg (incl. comms)
Power (peak) Power (average)	155 W	112 W	26 W 25 / 5 W (day/night)	2.3 W 0.1 W
Telemetry band	X/Ka	X/Ka	X	S
Continuous compressed science bit rate	50 kbps	-	2.5 kbps	100 bps
Key mission drivers	• ΔV requirements for VPO • Aerobot power (primary batteries and solar cells) • Highly integrated P/L suite for aerobot			
Key critical technologies	• Heat shield for entry vehicle • Balloon envelope for Venus environment • Triple-junction amorphous silicon solar cells for Venus environment • Fully miniaturized low resource in situ atmospheric instruments package • Atmospheric microprobe system (including localization)[1]			

[1] Breadboard currently under development under a TRP contract (17946/03/NL/PA)

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1 INTRODUCTION

This document provides an overview of the Venus Entry Probe system design study. The Venus Entry Probe is one of ESA's Technology Reference Studies (TRS), which provide a focus for the development of strategically important technologies that are of likely relevance for future scientific missions [Falkner05, Peacock06]. This is accomplished through the study of several technologically demanding and scientifically interesting mission concepts, which are not part of the ESA science programme. The TRSs subsequently act as a reference for possible future technology development activities.

Venus has been targeted for a TRS because an in-situ planetary atmospheric mission is both scientifically interesting and technologically challenging. The mission profile for the Venus Entry Probe study consists of two small-sats and a long-duration aerobot. The first satellite enters a polar Venus orbit. The Venus Polar Orbiter (VPO) contains a remote sensing payload suite primarily dedicated to support the in-situ atmospheric measurements by the aerobot. The second small-sat enters a highly elliptical orbit, deploys the aerobot and subsequently operates as a data relay, data processing and overall resource allocation satellite. The aerobot itself consists of a long-duration balloon, which will analyse the Venusian middle cloud layer. The balloon also deploys a swarm of active ballast probes, which determine vertical profiles of selected properties of the lower atmosphere.

In order to optimize the cost-efficiency, available components and subsystems have been used as much as possible. New technologies have only been baselined if they are enabling or significantly reduce the overall cost. For those technologies, the technology horizon has been set to five years (TRL of 5 before end of 2010).

This technical report is a considerably extended version of the refereed articles published in *Acta Astronautica* and *Advances in Space Research* (both available from sciencedirect.com)[Berg06a, Berg06b].

2 VENUS

In order to set the Venus Entry Probe TRS into perspective, a background on Venus as well as a summary of previous, proposed and planned missions is provided in this section.

2.1 Venus properties

Venus resembles Earth in many ways. It has 80% of the Earth's mass and is less than 30% closer to the Sun. Yet it has evolved completely differently resulting in a planet with a very hot surface (460 ºC) with no diurnal variation and a very dense atmosphere (92 bars at the surface), consisting mainly of carbon dioxide (~96%). The planet is heated by a runaway greenhouse effect caused by the carbon dioxide, as well as by cloud particles and minor atmospheric constituents that play a significant role in the atmospheric chemistry. The lack of water (~10^5 times less than on Earth) is considered to be the major cause of the hostile environment on Venus. Although the total amount of carbon dioxide is quite similar on Earth and Venus, most of the carbon dioxide in the Earth's atmosphere is dissolved in raindrops and transported in the form of bicarbonate ions to the oceans where the ions are converted into carbonate rocks. Measurements of the abundance of deuterium suggest that in the past a much larger quantity of water was present on Venus. Consequently, the prevailing theory is that Venus' primordial atmosphere has been (partially) lost and that the present atmosphere has been formed by crustal outgassing and/or by impacts from comets and meteorites.

The planet is completely covered with dense highly acidic clouds from an altitude of about 40 to 70 km, with haze below and above (up to 90 km). The clouds have a complex layered structure and exhibit a variable opacity. The cloud tops move at about 100 m/s (relative to the surface of the planet) in the longitudinal direction, thus circling the planet in four days. The cause of the atmospheric superrotation is still under debate. It is in strong contrast with the slow rotation rate of the planet itself, which has a period of 243 Earth days in retrograde direction. At the poles, the atmosphere displays a dynamical rotating dipole vortex feature, surrounded by a broad ring of circulating cold air, known as the 'polar collar.'

Due to the dense cloud deck, the surface of the planet can only be studied by radar or on the night side using the 1 μm IR spectral window. From an orbiter, the spatial resolution of the latter is limited by light scattering to about ~100 km. Radar mapping (down to 120 m spatial resolution) and altimetry has revealed that the topography follows a unimodal distribution (unlike Earth's bimodal distribution). The surface of Venus primarily consists of plains with an elevation between -2 and 2 km; smooth lowland plains (-2 – 0 km) and slightly rougher rolling plains (0 – 2 km). Only 15% of the surface reaches altitudes above 2 km. These highlands can be classified as tesserae (cm to m-scale rough terrain), volcanic rises, and Ishtar Terra, which is a unique and complex mountainous terrain. It contains Maxwell Montes with an altitude of 12 km, the highest feature on Venus.

Based on the crater distribution, which appears uniformly distributed, an average surface age of 300 – 500 My has been derived. This can either be explained by a continuous resurfacing of the craters by uniformly distributed global volcanic activity or by a single global catastrophic resurfacing event that occurred less than 500 My ago.

Though the surface exhibits tectonic features, such as long and narrow sinuous features (so-called wrinkle ridges) and deformation belts (ridge and fracture belts), there is no evidence of global plate tectonics as on Earth. The heat conduction through the relatively thick lithosphere is considered insufficient to release all the heat generated by radioactive decay in the interior. Current theories on how the internally produced heat is released include periodic mantle overturning scenarios.

By making use of imaging, X-ray fluorescence spectroscopy and gamma-ray spectroscopy, seven Venera and two VEGA landers have revealed that the surface primarily consists of basaltic rock plates. The mineralogical composition has not been determined directly, but it is expectedly dominated by surface-atmosphere chemistry. Due to the absence of water, the stable surface temperature and the low surface wind speeds, erosion and transport of surface material is negligible, with the exception of crater impact events. In contrast, chemical weathering is expected to play a major role. A prominent example of this process is the high surface reflectance at radio wavelengths in the Maxwell Montes region, which is indicative of an electrically conducting coating on the surface.

Venus has no intrinsic magnetic field, which different models attribute to either a completely solid core or (current) absence of core formation. The absence of a magnetic field has important consequences for the ionosphere of Venus, such as e.g. a direct interaction of the solar wind with the upper atmosphere/ionosphere, an induced magnetosphere, the absence of radiation belts, and significant plasma induced atmospheric escape processes).

Some basic planetary data on Venus are listed in Table 1 and Table 2. More details on Venus background and science can be found in [Bougher97a, Fegley04, Hunten83, Luhmann97].

Table 1: Venus solid body data (compared to Earth).

Parameter	Venus	Earth	Ref.
Mass (kg)	4.869E+24	5.974E+24	[NSSDC]
Equatorial radius (km)	6051.8	6378.1	[NSSDC]
Oblateness (Re-Rp)/Rp where Re and Rp are equatorial and polar radii, respectively.	0.0000	0.00335364	[Allen99]
Density (kg/m^3)	5243	5515	[NSSDC]
Surface gravity g (m/s^2)	8.87	9.78	[Allen99]
Equatorial escape velocity (km/s)	10.36	11.18	[Allen99]
Surface characteristics	nearly uniform surface level; few continental-scale highlands	land-sea contrasts	
Magnetic dipole field at surface (Tm^3)	< 1E11	7.84E15	[Allen99]

Table 2: Venus orbital and rotational data (compared to Earth).

Parameter	Venus	Earth	Ref.
Mean distance from Sun J2000 (km) (AU)	1.08209E+08 0.72333199	1.496E+08 1	[Allen99]
Minimum distance from Earth (km) (AU)	3.82E+07 0.255		[NSSDC]
Maximum distance from Earth (km) (AU)	2.61E+08 1.74		[NSSDC]
Eccentricity J2000	0.006773	0.0167	[Allen99]
Mean orbital velocity (km/s)	35.02	29.79	[Allen99]
Inclination of equator to orbit, obliquity (deg)	177.3	23.45	[Allen99]
Inclination to Ecliptic (deg)	3.39	0.00005	[Allen99]
Orbital (sidereal) period (d)	224.701	365.256	[NSSDC]
Sidereal rotation period (d)	-243.0187	0.99726968	[Allen99]
Length of solar day (d)	116.75	1	[NSSDC]
Overhead motion of Sun	west to east	east to west	

2.2 Missions to Venus

Venus has been explored by ground-based observations, flybys, orbiters and in-situ probes (descent probes, landers, aerobots). The ground-based observations and missions have provided a basic description of the planet, its atmosphere and ionosphere as well as a complete mapping of the surface by radar. The recently launched comprehensive planetary orbiter ESA's Venus Express (launched 2005)[Lebreton01] and the upcoming Planet-C mission from ISAS (launch 2010)[Oyama02], will further enrich our knowledge of the planet. These satellite observatories will perform an extensive survey of the atmosphere and plasma environment, thus practically completing the global exploration of Venus from orbit. For the next phase, detailed in-situ exploration will be required, expanding upon the very successful Venera atmospheric and landing probes (1967 - 1981), the Pioneer Venus 2 probes (1978), and the VEGA balloons (1985).

2.2.1 PAST, CURRENT AND PLANNED MISSIONS

The table below provides a sample of the more than twenty missions that have flown to Venus, are on its way to Venus or are currently planned. For more details or a more complete overview of past, current and planned missions to Venus, see e.g. [Hunten83, Shirley97] or the following web sources:

- http://nssdc.gsfc.nasa.gov/planetary/planets/venuspage.html
- http://www.mentallandscape.com/V_Venus.htm
- http://www.solarviews.com/eng/craft2.htm#venus
-

Table 3: Overview of past, current and planned missions to Venus.

Launch date	Mission	Type	Primary objective
August 1962	Mariner 2	Flyby	Atmosphere and plasma environment
June 1967	Venera 4	Flyby and descent probe	Ionosphere and in situ atmospheric measurements
June 1967	Mariner 5	Flyby	Plasma environment, ionosphere and UV absorption features in the upper cloud layer
January 1969	Venera 5 Venera 6	Flyby and descent probe	In situ atmospheric measurements and plasma environment
August 1970	Venera 7	Descent probe	In situ atmospheric measurements down to the surface
March 1972	Venera 8	Flyby and descent probe/lander	In situ atmospheric investigation and surface composition
November 1973	Mariner 10	Flyby to Mercury	Plasma environment, atmosphere and characterization of solid body
June 1975	Venera 9 Venera 10	Orbiter and descent probe/lander	Remote sensing of atmosphere, clouds and surface (radar). In situ investigations of atmosphere, clouds and surface, including a panoramic camera
May 1987	Pioneer Venus Orbiter	Orbiter	Comprehensive investigation of ionosphere, atmosphere and surface

Launch date	**Mission**	**Type**	**Primary objective**
August 1987	Pioneer Venus Multiprobe	Multiple descent probes	In situ atmospheric investigation at various locations across the planet, including a high latitude probe
September 1978	Venera 11 Venera 12	Flyby and descent probe/lander	Atmospheric chemistry, cloud composition and thermal balance
October 1981	Venera 13	Flyby and descent probe/lander	Atmospheric chemistry, cloud composition, thermal balance and surface composition
November 1981	Venera 14	Flyby and descent probe/lander	See Venera 13
June 1983	Venera 15 Venera 16	Orbiter	Atmosphere (IR spectroscopy) and surface (Synthetic Aperture Radar)
December 1984	Vega 1 Vega 2	Flyby (to comet Halley), descent probe/lander and balloons	Descent probe focussed on atmospheric and surface composition, balloons on atmospheric physics
May 1989	Magellan	Orbiter	Comprehensive radar investigations of the surface
October 1989	Galileo	Flyby (gravity assist to Jupiter)	Plasma environment and atmosphere (Near-IR mapping spectrometer)
October 1997	Cassini	Flyby (gravity assist to Saturn)	Plasma environment (including search for lightning signatures) and atmosphere
November 2005	Venus Express	Orbiter	Detailed remote sensing investigations of the plasma environment, atmosphere, and surface

2.2.2 MISSION CONCEPT STUDIES

Table 4 provides a literature overview of mission concepts for Venus exploration that have been studied or proposed in the past. As can be seen, the concept of a long duration aerobot with ballast probes is not new and has been considered before, see e.g. [Cutts99, Kerzhanovich00, Klaasen03]. The objective of the Venus Entry Probe Technology Reference Study is not so much to come up with a completely new conceptual approach to in-situ exploration of Venus, but rather to establish a technically feasible mission concept that is able to fulfil a set of reference mission objectives for lowest cost. The detailed system design study subsequently provides an overview of the mission drivers and is used to identify the critical technologies.

Table 4: Mission concept studies for Venus exploration.

Study/mission concept	Type	Short description	Reference
Venus Ionospheric Science Probe	Small spinning orbiter	Comprehensive investigation of the plasma environment Small spinning subsatellite as part of larger mission	[Blomberg06]
Venus Environmental Satellite	Orbiter	Science focus: -Atmospheric dynamics -Atmospheric composition -Atmospheric and surface chemistry -Meteorology Circular 30,000 km orbit (21h) with 45° inclination	[Baines95]
Atmospheric Composition Orbiter	Orbiter	Science focus: -spatial and temporal variations in clouds and trace gases	[Crisp02]
Global Geological Process Mapping Orbiter	Orbiter	Science focus -Surface mapping with 25-50 m horizontal resolution using a stereo or interferometric radar	[Crisp02]
Atmospheric Dynamics Explorer	Orbiter with multiple balloons	Science focus: -In-situ atmospheric dynamics -In-situ atmospheric structure 12-24 balloons deployed at different altitudes and latitudes Orbiter provides support by balloon tracking and global measurements	[Crisp02]
Venus stratospheric sounder	Aerobot	In situ measurements in the upper cloud region Slowly ascending zero-pressure balloon (altitude range 55 to 80 km)	[Kerzhanovich03]
Low altitude balloon	Aerobot	Science focus: -surface imaging -atmospheric composition -atmospheric dynamics Aerobot altitude 13 km Aerobot lifetime 10-30 days	[Izutsu00]
Venus Exploration of Volcanoes and Atmosphere	Descent module and balloon with drop sondes	Science focus: -atmospheric composition (<20 km) -surface vis-NIR imaging -basic atmospheric properties First item measured by descent module(s) and other items by smaller drop sondes Aerobot altitude 60 km Aerobot lifetime 7 days	[Cutts99] [Kerzhanovich00] [Klaasen03]

Study/mission concept	Type	Short description	Reference
Lavoisier	Three balloons and a descent probe	Science focus: -in situ atmospheric investigation -surface NIR imaging/spectroscopy Aerobot altitude 10 km Mission proposal to ESA, 2000.	[Chassefière04]
Venus Entry Probe Technology Reference Study	Orbiter with long-duration aerobot and microprobes	Science focus: -atmospheric dynamics -global in situ exploration of the atmosphere (altitude 55 km) -atmospheric structure Aerobot lifetime 15-22 days Study to assess technology requirements for a typical in situ exploration of Venus	[Berg06a] [Berg06b]
Venus Geoscience Aerobot Study	Altitude controlled aerobot	Science focus: -atmospheric dynamics -atmosphere-surface interaction -high resolution surface imaging -surface mineralogy Reversible fluid balloon Aerobot altitude range 1 – 60 km Aerobot lifetime 100 days	[Bachelder99]
Directed Aerial Robot Explorers	-Long duration aerobot with trajectory control -Microprobes	Trajectory control by wing hanging on its side below the balloon on a very long (several km) tether	[Pankine04]
Application of aerobot technology for Venus	Two aerobot technology studies: - long duration aerobot oscillating between 40 and 60 km -Aerobot descending to Venus surface	Science application: -atmospheric science -surface investigations	[Gilmore05]
Venus aircraft	Solar aircraft	Venus in situ atmospheric exploration by a solar-powered aircraft	[Landis02]
Noble gas and trace gas explorer	Descent probe	Science focus: -Noble gas abundance and isotopic ratios -Atmospheric composition -Atmospheric structure	[Crisp02]
Venus microprobes	Microprobes (0.3 – 5 kg)	Vertical profiles of atmospheric properties and surface imaging	[Lorenz98]

Study/mission concept	Type	Short description	Reference
Venera-D	Lander with - small long-living station (~5-30 days) - balloons (optional) - microprobes (optional)	Science focus: -near-surface atmosphere -surface composition -age dating -seismic activity Under study by IKI/Lavochkin Launch beyond 2015	[Korablev06]
Surface and Interior Explorer	Multiple landers	Science focus: -surface composition -surface mineralogy -seismometry -meteorological conditions at surface 3 or more long-lived landers (more than several months)	[Crisp02]
Venus sample return	Several studies for sample return mission concepts, including: -surface sample return -atmosphere sample return	Concepts include: -Atmosphere skimmer with hypersonic velocity -High altitude atmospheric sampler -Surface sampling with rocket launched from balloon	[Coradini98] [Rodgers00] [Crisp02] [Sweetser03]

3 MISSION SCENARIO

3.1 Mission objectives

The objective of the Venus Entry Probe Technology Reference Study is to establish a feasible mission profile for a cost-efficient in-situ exploration of the atmosphere of Venus. In order to obtain a scientifically meaningful mission profile, an extensive literature survey has been performed, resulting in a typical set of key scientific objectives for Venus atmospheric investigation. From this survey, the following set of mission objectives (MO) for the Venus Entry Probe study has been derived (with references to review articles):

[MO1] *Origin and evolution of the atmosphere*

It is of great interest for comparative planetology to understand why and how the atmosphere has evolved so differently compared to Earth. This can only be investigated by in-situ measurements of the isotopic ratios of the noble gases [Moroz02, Titov02].

[MO2] *Composition and chemistry of the lower atmosphere*

Accurate measurements of minor atmospheric constituents, particularly water vapour, sulphur dioxide and other sulphur compounds, will improve our knowledge of the runaway greenhouse effect on Venus, atmospheric chemical processes and atmosphere-surface chemistry as well as the possible existence of volcanism [Moroz02, Titov02].

[MO3] *Atmospheric dynamics*

Venus has a very complicated atmospheric dynamical system. The driving force behind the zonal superrotation, the dynamics of the polar vortices and the meridional circulation as well as the origin of the temporal and spatial variations of the cloud layer opacity are all rather poorly understood [Moroz02, Taylor02, Titov02].

[MO4] *Aerosols in the cloud layers*

Measurements of the size distribution, temporal and spatial variability as well as the chemical composition of the cloud particles is of interest for better understanding the thermal balance as well as the atmospheric chemistry [Moroz02]. Furthermore, it has been suggested that the unidentified large (~ 7 μm diameter) cloud particles might contain microbial life [Cockell99, Schulze-Makuch02].

3.2 Derived mission requirements

In order to address above mission objectives, the following mission requirements have been imposed on the Venus Entry Probe TRS:

[MR1] In-situ atmospheric exploration at an altitude between 40 and 57 km at all longitudes by means of an aerobot [MO1-4].

[MR2] Vertical profiles of selected properties of the lower atmosphere at varying locations across the planet by means of atmospheric microprobes [MO3].

[MR3] Remote atmospheric sensing to provide a regional and global context of the in-situ atmospheric measurements (also concurrent with the aerobot operational phase) [MO2-4].

[MR4] Remote sensing of the polar vortices with a large field of view and a temporal resolution of at least 5 hours [MO3].

[MR5] Remote sensing of the Venus atmosphere at all longitudes and latitudes [MO2-4].

3.3 Mission concept

The mission configuration that is able to fulfil the mission requirements consists of a pair of small-sats and an aerobot, which drops active ballast probes. Two orbiting satellites are required in order to commence the remote sensing atmospheric investigations prior to the aerobot deployment [MR3]. One satellite, the Venus Polar Orbiter (VPO), contains a remote sensing payload suite to support the in situ atmospheric measurements of the aerobot as well as to address the global atmospheric science objectives. The second small-sat, the Venus Elliptical Orbiter (VEO) enters a highly elliptical Venus orbit, deploys the aerobot (after the VPO has reached its final orbit and the VPO instrument commissioning phase has been completed). The VEO subsequently operates as a data relay satellite. The use of a dedicated data relay satellite enables the Venus Polar Orbiter to practically continuously monitor the Venus atmosphere because data transmission to Earth is carried out by the VEO.

The aerobot consists of a long-duration balloon, which will analyse the Venusian middle cloud layer. During flight, the balloon deploys a swarm of active ballast probes, which determine vertical profiles of pressure, temperature, solar flux levels and wind velocity in the lower atmosphere.

Table 5 gives an overview of the mission baseline scenario, including a reference model payload suite. This representative set of payload instruments has been assumed in order to study the impact of typical payload interface and resource requirements on the mission concept design. The selection is based on a literature study and does not imply any endorsement of specific science instruments for a possible future mission to Venus

Table 5: Mission baseline scenario.

S/C Module	Measurements	Reference payload suite	Requirements
Venus Polar Orbiter (VPO)	- Atmospheric composition - Atmospheric dynamics - Atmospheric structure	- Sub-mm wave sounder -Visible-NIR imaging spectrometer - UV spectrometer - IR Fourier transform spectrometer - UV-visible-NIR camera	- Large FOV (~5,000 km) - Frequent visit of poles (at least every ~5 hours) - Resolution ~ 5 km - Operational (just) before aerobot deployment
Venus Elliptical Orbiter (VEO)			- Entry probe deployment - Data relay to Earth
Aerobot	- Isotopic ratios noble gases - Minor gas constituents - Aerosol analysis - Atmospheric structure - Thermal balance - Tracking and localization of microprobes	- Gas chromatograph mass spectrometer (with aerosol inlet) - Nephelometer - Solar/IR flux radiometers - Meteorological package - Inertial package - Radar altimeter	- Long duration (different longitudes) - Altitude 40- 57 km (aerosols) - Microprobe deployment - Tracking and localization of microprobes
Atmospheric microprobes	- Pressure - Temperature - Light level (up and down) - Wind velocity	- P/L fully integrated with probe	- Operational down to ~ 10 km

4 MISSION ENVIRONMENT

Table 6 shows the main environmental conditions for the orbiters. The radiation environment around Venus is rather benign because Venus has no trapped radiation belts. The thermal environment is more challenging due to higher solar flux level (compared to Earth) and the high planetary albedo. For missions to Venus (orbiters, flybys and landers) no planetary protection requirements need to be fulfilled.

Table 6: Mission environmental parameters for the Venus orbiters.

Parameter	Value	Remarks
Radiation	24 krad (Si) for 2 mm Al shielding 10 krad (Si) for 4 mm Al shielding	- Solar maximum conditions - 2 year lifetime and ~160 days transfer - Mainly solar protons and galactic cosmic rays
Thermal (at Venus)	Solar constant: 2.62 kW/m^2 Bond albedo: 0.76 Geometric albedo: 0.65	From [Moroz85] and [Allen99]
Ionosphere	Av. peak electron density ~10^4 cm^{-3} Ionopause between 300-2000 km (day/night) Peak electron density at 140-180 km	From [Bauer85]
Planetary protection	No protection required	VEP is a COSPAR category I mission [Cospar02]

In Table 7 the environmental parameters for the in-situ mission elements are given. The aerobot and microprobes will have to withstand a highly corrosive environment and the microprobes will experience high temperatures and pressures during descent. Additionally, the variation in solar flux as a function of local solar time will need to be considered in the balloon and gondola design.

According to [Aplin06], a global electric circuit on Venus is unlikely, though electrical processes do occur in the atmosphere (e.g. cosmic ray ionization, ion-aerosol interaction). The Cassini flybys detected no high-frequency signatures of lightning, thus putting severe constraints on the occurrence and/or nature of lightning (i.e. only very weak cloud-cloud or cloud-ionosphere discharges) [Grebowsky97, Gurnett01].

Table 7: Key mission environmental parameters for the Venus aerobot and microprobes.

<table>
<tr><th>Parameter</th><th colspan="4">Value</th><th>Remarks</th></tr>
<tr><td>Radiation</td><td colspan="4">5.5 krad (Si) for 2 mm Al shielding
2.5 krad (Si) for 4 mm Al shielding</td><td>- Solar maximum conditions
- During ~160 days transfer</td></tr>
<tr><td rowspan="10">Atmosphere</td><td>Height (km)</td><td>T (˚C)</td><td>P (bar)</td><td>Zonal wind speed (m/s) avg (min-max)</td><td rowspan="10">- Composition: 96.5% CO_2, 3.5% N_2
- T, p, v for latitudes up to 30˚ [Seiff85]
- North-South wind speed: -10…+10 m/s
- Dense cloud layer between 45-70 km
- Wind speed in East-West direction for latitudes up to 40˚ [Kerzhanovich85]</td></tr>
<tr><td>70</td><td>-43</td><td>0.037</td><td>92 (62 – 124)</td></tr>
<tr><td>60</td><td>-10</td><td>0.24</td><td>77 (53 – 110)</td></tr>
<tr><td>55</td><td>29</td><td>0.53</td><td>60 (39 – 90)</td></tr>
<tr><td>50</td><td>77</td><td>1.1</td><td>61 (38 – 80)</td></tr>
<tr><td>40</td><td>144</td><td>3.5</td><td>41 (28 – 59)</td></tr>
<tr><td>30</td><td>224</td><td>9.6</td><td>36 (22 – 49)</td></tr>
<tr><td>20</td><td>308</td><td>23</td><td>28 (12 – 41)</td></tr>
<tr><td>10</td><td>385</td><td>38</td><td>5 (-2 – 11)</td></tr>
<tr><td>0</td><td>462</td><td>92</td><td>0.5 (-1 – 1)</td></tr>
<tr><td rowspan="9">Solar and thermal flux</td><td>Height (km)</td><td colspan="2">Solar flux (kW/m^2)</td><td>Thermal flux (kW/m^2)</td><td rowspan="9">- Fluxes quoted as average (up/down)
- Solar flux for SZA 0 degrees from [Moroz85, Tomasko80]
- Thermal flux (interpolated) from D. Crisp thermal model in [Jones97]</td></tr>
<tr><td>60</td><td colspan="2">1.52</td><td>0.28</td></tr>
<tr><td>55</td><td colspan="2">1.16</td><td>0.46</td></tr>
<tr><td>50</td><td colspan="2">0.88</td><td>0.79</td></tr>
<tr><td>40</td><td colspan="2">0.66</td><td>1.74</td></tr>
<tr><td>30</td><td colspan="2">0.58</td><td>3.44</td></tr>
<tr><td>20</td><td colspan="2">0.44</td><td>6.31</td></tr>
<tr><td>10</td><td colspan="2">0.25</td><td>10.3</td></tr>
<tr><td>0</td><td colspan="2">0.05</td><td>16.0</td></tr>
<tr><td rowspan="3">Atmospheric electricity</td><td colspan="3">Electrical conductivity (@ 55 km)</td><td>$\sim 10^{-14}$ S/m</td><td>Predicted values from [Borucki82]</td></tr>
<tr><td colspan="3">Electric field intensity (10 kHz – 90 kHz)</td><td>< 300 µVolt/m/sqrt(Hz)</td><td rowspan="2">From [Ksanfomality83]</td></tr>
<tr><td colspan="4">Global electric circuit unlikely [Aplin06]

Existence and nature of lightning unresolved, but not expected to occur from cloud to surface [Grebowsky97, Gurnett01]</td></tr>
<tr><td>Other</td><td colspan="4">Highly corrosive environment</td><td>- Cloud droplets: 75% H_2SO_4 * 25% H_20
- Reactive gases, such as HF, HCl, and H_2SO_4
- See also [Fegley97]</td></tr>
<tr><td>Planetary protection</td><td colspan="4">No protection required</td><td>Aerobot and microprobes fall under a COSPAR category I mission.</td></tr>
</table>

5 MISSION ANALYSIS

5.1 Launch window analysis

A standard high thrust heliocentric transfer scenario from Earth to Venus has been baselined for the study, because it is the most cost-efficient and flexible transfer option for a mission to Venus. Table 8 provides a summary of the optimized Δ-V requirements for different launch windows for a half-revolution transfer from Geostationary Transfer Orbit (GTO, defined as 250 km × 35,786 km) to a Venus capture orbit (400 km × 215,000 km)[Boutonnet07]. GTO has been used as the departure orbit so that the table is independent of launch vehicle performance. The transfer time for a half solar revolution transfer is typically between 110 and 180 days.

The launch opportunities are clearly driven by the Earth-Venus synodic period of 1.6 years. The 3.4º inclination of Venus' orbit to ecliptic causes a variation in the Earth-Venus distance, so that the total delta-V requirements vary at successive optimum launch windows. After five synodic periods, i.e. 8 years, the optimum transfers approximately repeat, as can be seen by comparing the first and last two launch dates.

Table 8: Summary of ΔV-requirement for high-trust transfers to Venus (conservative reference launch date is indicated with gray background, possible launch opportunities in bold + italics)[Kemble03].

Launch date	ΔV escape from GTO (km/s)	Arrival date	ΔV insertion (km/s)	Total ΔV (km/s)	Transfer time (days)	Time to next launch window (months)
12/06/2010	***1.49***	***15/12/2010***	***0.59***	***2.08***	***186***	***41***
15/08/2010	1.14	09/12/2010	1.25	2.39	116	-
15/01/2012	1.71	31/07/2012	0.77	2.48	198	-
04/04/2012	1.49	23/07/2012	1.09	2.58	110	-
02/11/2013	***1.11***	***09/04/2014***	***1.14***	***2.25***	***158***	***0.5***
15/11/2013	***1.54***	***05/03/2014***	***0.66***	***2.2***	***110***	***17***
25/04/2015	***1.17***	***27/10/2015***	***0.93***	***2.1***	***185***	***1.5***
08/06/2015	***1.23***	***19/11/2015***	***0.55***	***1.78***	***133***	***18***
06/12/2016	***1.28***	***18/05/2017***	***0.5***	***1.78***	***163***	***1***
06/01/2017	***1.1***	***10/05/2017***	***0.91***	***2.01***	***124***	***17***
12/06/2018	***1.47***	***13/12/2018***	***0.58***	***2.05***	***184***	***-***
11/08/2018	1.13	07/12/2018	1.25	2.38	118	-

A typical worst-case launch date, 2 November 2013, has been selected as the baseline, which allows the basic mission concept to be flown in 3 out of 4 successive launch windows, which occur every 19 months. The maximum time between two successive launch opportunities, in any eight year period, is 3.5 years. The baselined Type-II interplanetary transfer trajectory is depicted in Figure 1.

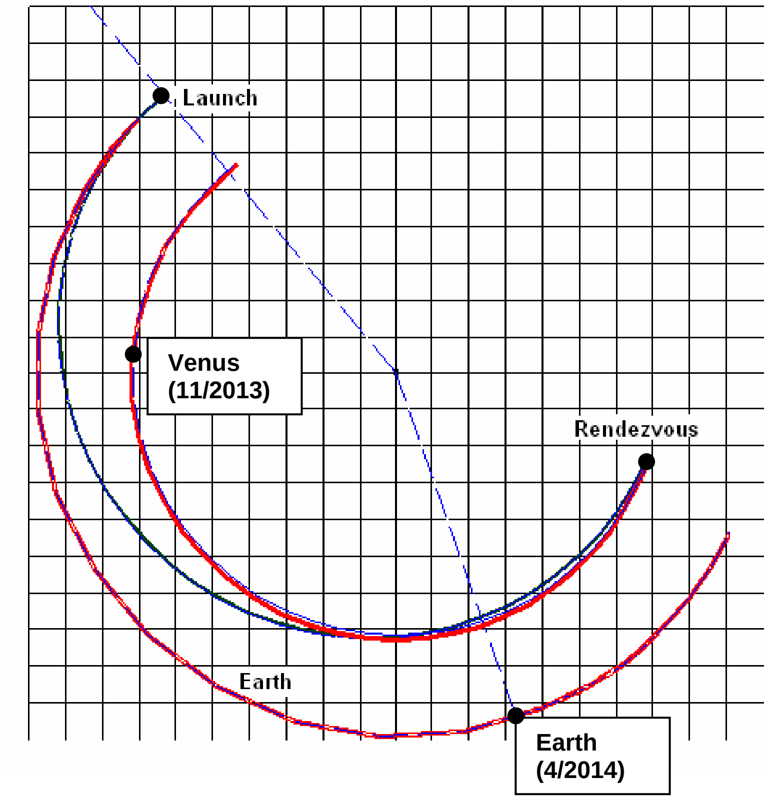


Figure 1: Interplanetary transfer trajectory from Earth to Venus for reference launch date 02/11/2013.

5.2 Launch vehicle

A Soyuz-Fregat 2-1B launch from Kourou has been selected as the baseline for the Venus Entry Probe TRS because it is a cost-efficient and highly reliable launch vehicle with sufficient mass capability. For the reference launch date, 2 November 2013, the mass capability for direct escape to Venus is 1509 kg (using standard circular parking orbit, including a 20-day launch window), which becomes 1464 kg after subtraction of the launch adapter[2]. For the alternative launch dates in 2015/2016/2017/2018, the mass capability of the Soyuz-Fregat to Venus escape trajectory as well as the S/C propellant and propulsion system mass (needed to perform Venus orbit insertion) will be different, but the 'useful in-orbit spacecraft mass' that can be achieved should be similar (or higher) compared to the reference launch date. The key specifications of the launch vehicle are summarized in Table 9. The volumetric constraints for the fairing are depicted in Figure 2 [Soyuz01].

[2] The quoted values for Soyuz-Fregat performance to Venus escape are based on data available end of 2005.

Table 9: Baseline launch vehicle parameters.

Parameter	Value	Notes
Launch site	Kourou	Guiana Space Centre (CSG)
Launch vehicle	Soyuz-Fregat 2-1b	
Escape performance to Venus (2013/2015/2016/2017/2018)	~1100 – 1600 kg	Depending on launch date
Mass to Venus escape (2/11/2013)	1509 kg	Reference launch date, including launch window
Launch adapter mass	45 kg	937-SF
Fairing dimensions	Diameter: 3.8 m Height: 5.0 – 9.5 m	ST-Fairing (S-fairing expected to be unavailable at CSG)
Cost	~40 M€	FY2005

5.3 *Operational orbits*

The operational orbits for the spacecraft, shown in Table 10, have been chosen such that the scientific return and spacecraft mass are optimized. The operational orbit of the Venus Polar Orbiter is tailored to allow a comprehensive investigation of the Venus atmosphere and its dynamics, particularly the study of the polar vortices which require a large field of view and a high revisit frequency, whereas the required spatial resolution is modest (see also Table 5). The Venus Elliptical Orbiter stays in the highly elliptical Venus capture orbit, which is energetically most favourable for deployment of the entry probe. The inclination and argument of periapse for the VEO have been selected to minimize the ΔV required for the entry probe release manoeuvre as well as for 5 years of VEO orbital maintenance [Boutonnet07].

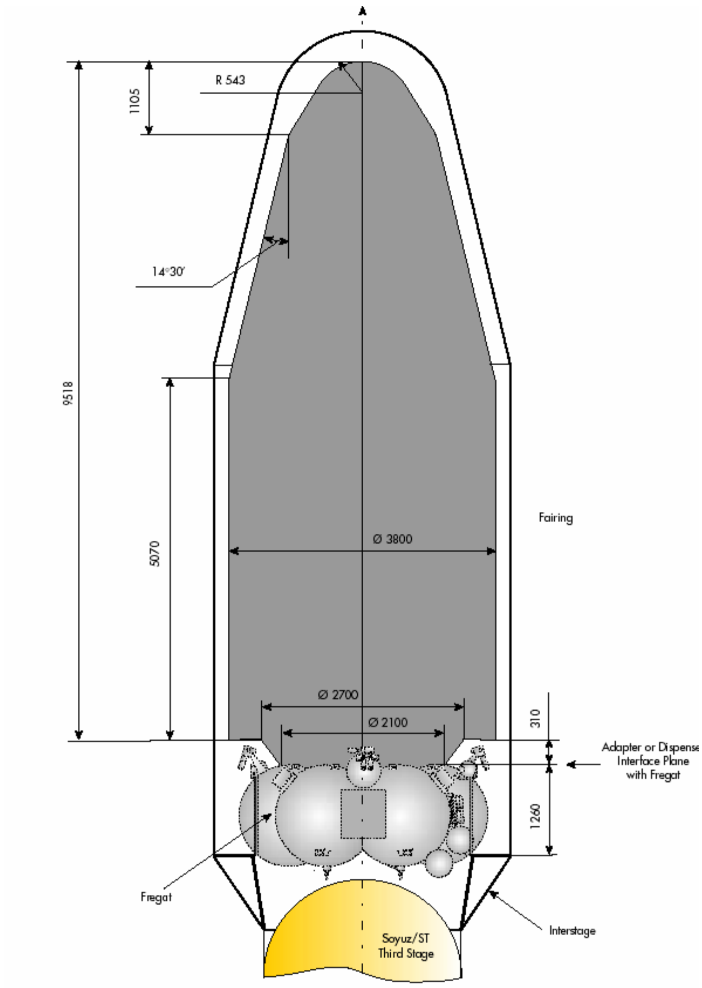


Figure 2: Volumetric constraints for Soyuz-Fregat ST-fairing (from [Soyuz01]).

Table 10: Operational orbits for the Venus Polar Orbiter and Venus Elliptical Orbiter spacecraft.

Parameter	VPO	VEO
Initial periapse (km)	2,000	400
Initial apoapse (km)	6,000	215,000
Initial inclination	90°	64°
Argument of periapse	No requirements	103°
Period (hr)	3.1	117 (4.9d)
Remarks	For atmospheric science, particularly for studying polar dynamics [MR4], a large coverage and a high repeat frequency are required. Required spatial resolution is modest (see also Table 5).	Inclination and argument of periapse selected to minimize ΔV for entry probe release and VEO orbital maintenance [Boutonnet07].

5.4 *ΔV summary*

The spacecraft ΔV requirements for the reference launch date 2/11/2013 (or 8 years later) are summarized in Table 11. The Earth departure ΔV is provided by the Soyuz-Fregat launch vehicle and requires no propulsive burn from the spacecraft. After launch, a mid-course correction burn of 55 m/s ensures that any launch errors are corrected and facilitates accurate targeting of the final planetary insertion point. Venus orbit insertion into an initial capture orbit of 400 km × 215,000 km requires 1250 m/s (excluding gravity losses). After a period of functional checkout and commissioning, the Venus Polar Orbiter is transferred to its operational orbit, which requires 2000 km/s. The associated gravity losses of 17 m/s are based on a two-burn apocentre lowering. 90 to 180 days after Venus Orbit Insertion (VOI), the entry probe is deployed. The total ΔV associated with the deployment of the entry probe from the Venus Elliptical Orbiter is less than 216 m/s. A provision of 70 m/s is made for orbital maintenance, though this is not required for the operational orbits provided in Table 10 [Boutonnet07].

Table 11: Mission ΔV summary.

Event	Venus Polar Orbiter			Venus Elliptical Orbiter			Notes
	ΔV (m/s)	Margin (%)	Total ΔV (m/s)	ΔV (m/s)	Margin (%)	Total ΔV (m/s)	
Earth departure	0	0	0	0	0	0	2/11/2013
Mid course correction	50	10	55	50	10	55	
Safe mode correction	7	15	8	13	15	15	5 days before VOI
Venus orbit insertion	1190	5	1250	1190	5	1250	
Gravity loss	110	15	127	60	15	70	
Pericentre rotation				3	10	4	
Pericentre lowering				55	10	60	Maximum
De-orbit burn				71	10	78	Entry probe release
Re-orbit burn				71	10	78	
Operational orbit	1905	5	2000				
Gravity losses	15	15	17				
Orbit maintenance	35	100	70	35	100	70	Not essential
Momentum dumping	5	100	10	5	100	10	
Total			**3537**			**1691**	

The VEO and VPO spacecraft can travel as a composite or individually. The composite configuration is likely more cost-efficient (mission operations), but it offers less flexibility in choosing the individual operational orbits. For this reason, in this study, individual transfer to Venus has been assumed.

6 SPACECRAFT DESIGN

This section is largely based on a mission design study performed by Surrey Satellite Technology Ltd under an ESA contract [Phipps06]. It starts with a system overview and top-level mass budget. In subsequent subsections, the design of the mission elements is detailed.

6.1 Margins

During the spacecraft design study, the margins listed in Table 12 have been used. The ΔV margins are provided in Table 11. The margins largely comply with the ESA margin philosophy for assessment studies [Atzei05]. The nominal mass and power budgets are determined after application of the subsystem margins. All subsystems are sized to accommodate any other subsystem with subsystem margin applied (e.g. the entry probe heat shield is sized to accommodate the aerobot with subsystem margin; the balloon is sized to lift the gondola with subsystem margin).

The nominal propulsion subsystem (including tanks), as well as the propellant, is sized to accommodate the mass budget after application of the system level margin. Likewise, the nominal power subsystem is designed and sized to provide the spacecraft required power, including system level power margin.

Table 12: Margin overview.

Item	Margin
Subsystem mass margin	
Off-the-shelf equipment	5%
Off-the-shelf equipment requiring minor modifications	10%
New designs/major modifications	20%
Power subsystem margin	
Off-the-shelf equipment	5%
Off-the-shelf equipment requiring minor modifications	10%
New designs/major modifications	20%
Data processing	
On-board memory capacity margin	50%
Processing peak capacity margin	50%
Communications	
Communication link	3 dB
Telecommand and telemetry data rates	3 dB
System level	
System level mass margin	at least 20%
System level power margin	at least 20%

Table 13: Spacecraft top level mass budget.

Item	VPO mass (kg)	VEO mass (kg)	Remarks
Science instruments	25.2		Highly integrated P/L suite for atmospheric science
Entry probe		91.1	
Communications	20.2	20.3	X/Ka-band
Structure & harness	78.0	76.5	Thrust tube concept
Propulsion	63.7	42.3	Chemical bipropellant system. Tanks sized for maximum separated mass plus 5% margin.
ACS	9.5	9.5	CMG, 2 star trackers, 3 sun sensors
OBDH	4.2	4.2	Leon processor, hard-disk and peripherals
Power	25.5	13.6	
Environment	21.4	16.7	Primarily thermal
Nominal dry mass	**247.7**	**274.2**	
System level margin (20%)	49.5	54.8	
S/C mass incl. system level margin	*297.2*	*329.0*	*For propellant calculation and propulsion system sizing*
Propellant	607.7	229.1	Both high and low thrust manoeuvres included
Wet mass	**904.9**	**558.1**	Including system level margins
Launch adapter	45		937-SF (Venus Express)
Total launch mass	**1508**		Including 20% system level margin
Launch vehicle capacity	**1509**		Reference launch date 2/11/2013. Incl. launch window margin.

6.2 *System overview*

The functional architecture for the system design is presented in Figure 3. The spacecraft are designed for maximum commonality on platform and subsystem level. Existing space qualified components have been baselined as far as practicable. The two orbiter spacecraft have a large degree of redundancy, particularly the communication, attitude control system (ACS), and on-board data handling system (OBDH). Table 13 summarizes the overall mass budget. The mass of the composite spacecraft is compliant with the 20% system level margin requirement for ESA assessment studies. Clearly, the high ΔV requirement for the Venus Polar Orbiter is a significant mass and design driver.

The launch configuration is shown in Figure 4, with the Soyuz-Fregat ST-fairing to scale in the background. The stacked configuration easily fits into the ST-fairing and would also fit into a Soyuz-Fregat S-fairing (payload envelope diameter of 2.3 meters for a height up to 3.9 meters [Soyuz01]), but it is not expected that the S-fairing will be available at CSG.

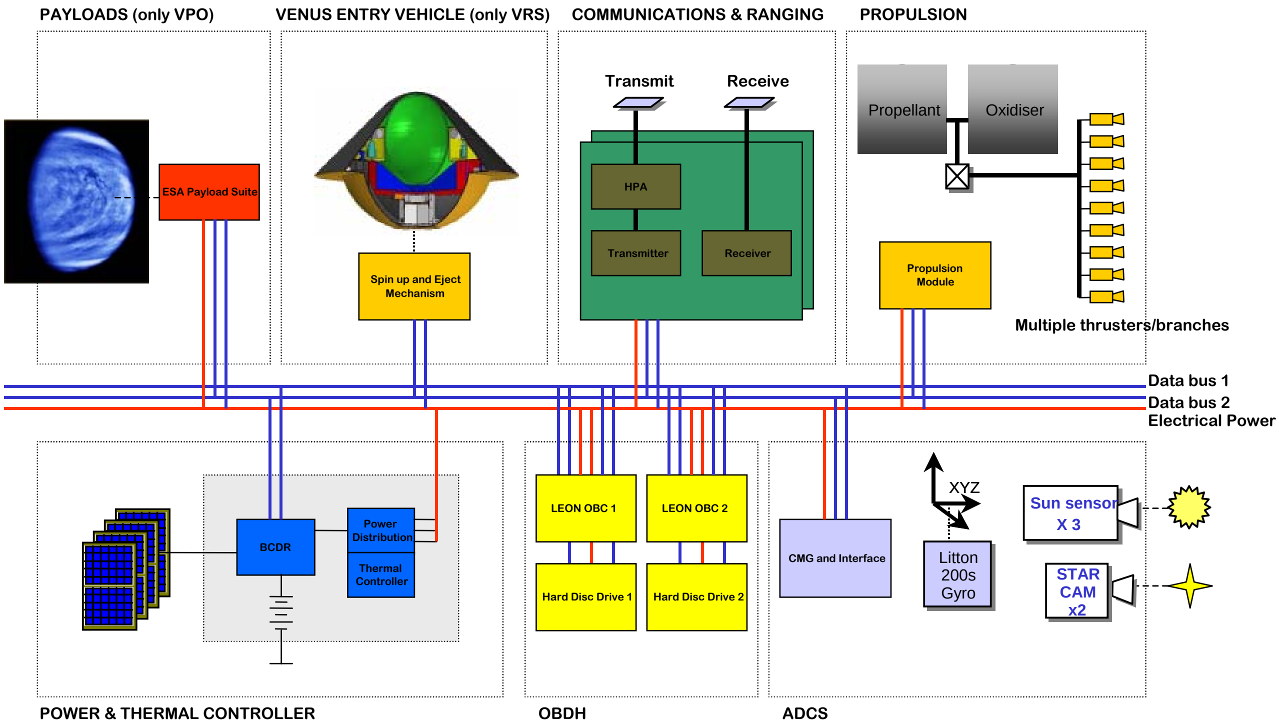


Figure 3: System diagram of the Venus Entry Probe system.

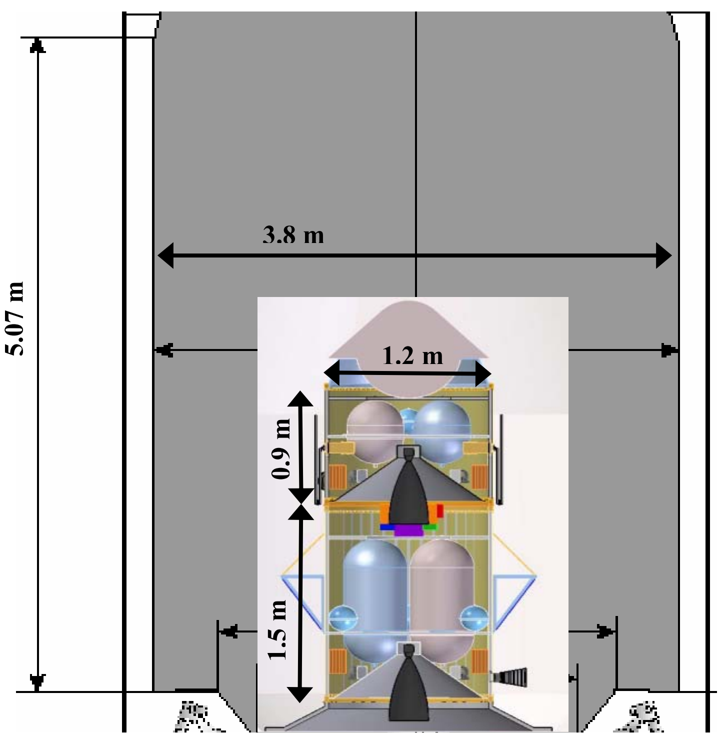


Figure 4: Spacecraft in launch configuration with Soyuz-Fregat ST-fairing in the background.

6.3 Orbiters

6.3.1 REMOTE SENSING REFERENCE PAYLOAD SUITE

Only the Venus Polar Orbiter carries a remote sensing payload suite. The objective of this remote sensing reference payload suite, as assumed for the Venus Entry Probe study, is to perform a global investigation of the Venus atmosphere, particularly the Venus atmospheric dynamics, as well to support the in-situ measurements of the aerobot (which also provide ground truth calibration of the remote sensing instruments). The reference payload suite that can fulfil the mission objectives is listed in Table 14, together with the allocated resources.

The typical viewing and accommodation requirements of the instruments for the baseline polar orbit of 2,000 × 6,000 km are summarized in Table 15. The field of view (FOV) is defined as the solid angle within which the instrument accepts and images incoming radiation. For several instruments, scanning mechanisms are implemented to increase the coverage to enable investigation of the extended region of the polar vortices (see also [MR4] in section 3.2). The Visible-NIR imaging spectrometer is complemented by a nadir-looking high resolution spectrometer.

Table 14: Resource specifications of the reference remote sensing payload suite assumed for the TRS.

Instrument	Key measurements	Mass (kg)	Peak power (W)	Raw data rate (kbps)	Compressed data rate (kbps)
Sub-mm wave sounder (540-660 GHz)	Atmospheric structure and circulation, temperature, composition and chemistry	6.0	30.7	5	5
Visible-NIR imaging spectrometer (and nadir non-imaging spectrometer) (0.7 – 2.5 μm)	Composition and dynamics of the lower atmosphere	4.0	15.8	4,000	20
UV spectrometer (50 – 600 nm)	Composition and dynamics of the upper atmosphere	4.5	4.0	7	7
IR Fourier transform spectrometer (6- 16 μm)	Temperature, cloud structure and composition	4.0	2.8	2,000	30
UV-visible-NIR camera (0.4 – 1 μm)	Global circulation	1.0	2.5	50	2
Centralized power supply unit		1.0	8.4		-
Central processing unit		0.5	1.9		-
Subtotal		**21.0**	**66.3**		-
Margin (20%)		4.2	13.2		-
Total (peak)		**25.2**	**80.0**	**6,062**	**64**
Total average			**60.0**		**50**

Table 15: Viewing and accommodation requirements for the remote sensing reference payload instruments.

Instrument	**Instrument field of view [degrees]** (along track × across track or diameter)	**Aperture diameter [mm]**	**Pointing direction**	**Scanning strategy**
Sub-mm wave sounder	Ø < 0.3°	100	Limb and nadir	120° scanning mechanism to cover nadir, limb and cold space
Visible-NIR imaging spectrometer	15° × 0.05°	30	Nadir scanning	Whisk-broom scanning to achieve across track coverage of 50°
Visible-NIR non-imaging spectrometer	Ø < 1°	30	Nadir	-
UV spectrometer	2° × 0.1°	40	Primarily limb	Limb scanning mechanism also allows nadir
IR Fourier transform spectrometer	15° × 0.05°	45	Nadir scanning	Scanning mechanism to increase across track coverage to 60° as well as to achieve cold space view for calibration
UV-visible-NIR camera	70° × 70°	5	Nadir	-

An assessment study has been performed by Cosine Research to integrate the strawman science instruments into a so-called highly integrated payload suite [Moorhouse05]. By merging individual instruments onto one platform and sharing resources on a system architecture level, mass and

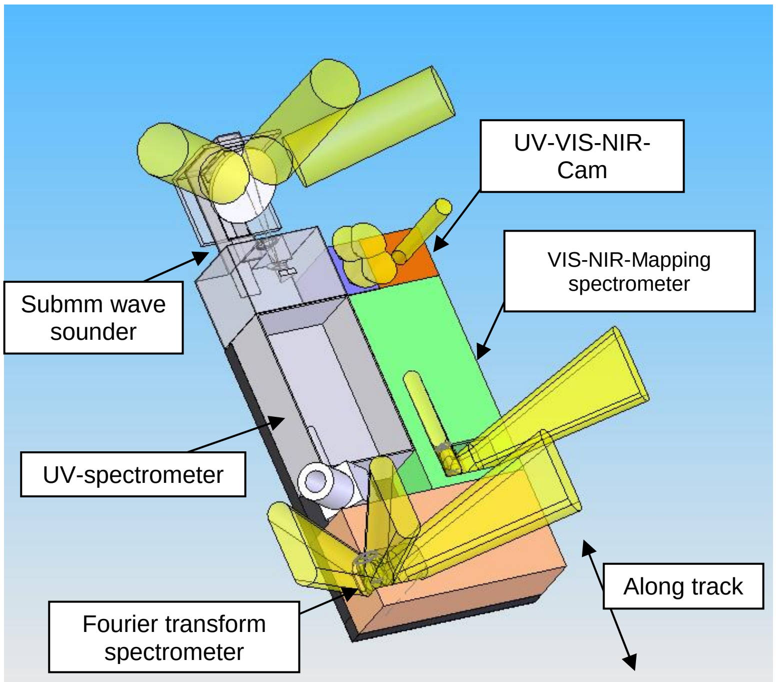


Figure 5: Conceptual layout of a highly integrated payload suite for the Venus Polar Orbiter. The fields of view of the individual instruments are shown in yellow.

power reductions can be achieved without sacrificing instrument performance. Additionally, the number of spacecraft interfaces is reduced. A possible implementation of a highly integrated payload suite for the Venus Polar Orbiter is schematically shown in Figure 5.

The size of the reference payload suite is approximately 40 × 80 × 20 cm^3. The UV-visible-IR cameras and spectrometers share the same isothermal optical bench[3], while the sub-mm wave sounder is supported off the bench. The simple design allows all fields of view to be accommodated when the bench is attached to the spacecraft nadir panel. The sub-mm wave sounder, based on an ESA Concurrent Design Facility study [Henderson04], requires views to both limbs and extrudes over the side of the spacecraft. The IR Fourier transform spectrometer and UV-spectrometer also require a limb view.

The electronic boxes, including the central power supply and central processing unit, are located on the opposite side of the optical bench. All payload instruments share the same centralized power supply and central processing unit, which is based on a dual LEON processor core implemented on an Actel FPGA. For the interface between the instruments and the payload processing unit, as well as between the payload processing unit and the spacecraft OBDH, a SpaceWire architecture has been baselined.

6.3.2 SPACECRAFT

6.3.2.1 Mechanical configuration and structure

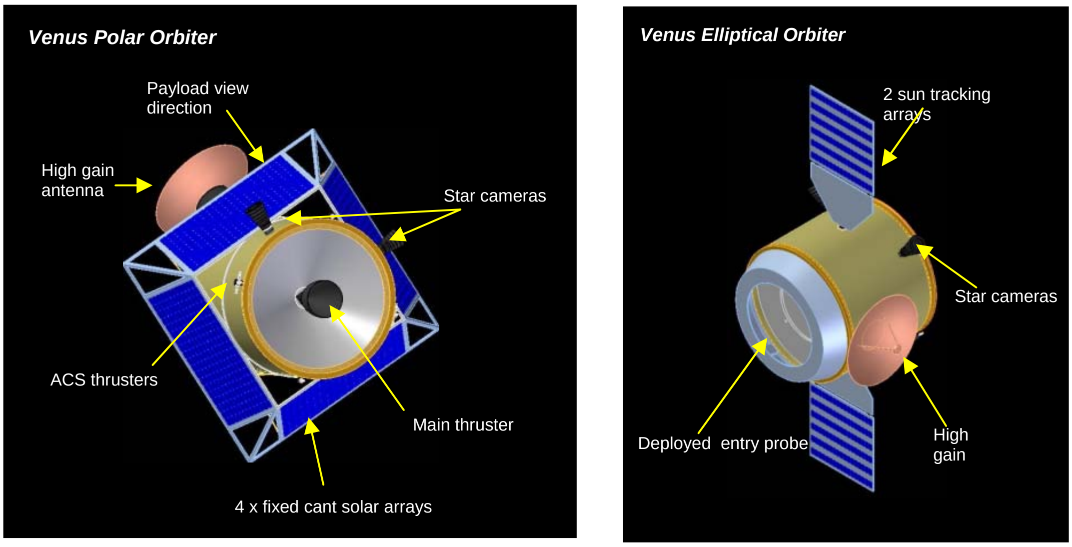


Figure 6: Conceptual design of the Venus Polar Orbiter and the Venus Elliptical Orbiter.

The spacecraft configurations for the Venus Polar Orbiter and the Venus Elliptical Orbiter are depicted in Figure 6. The 3-axis stabilized design is based on a cylindrical central thrust tube structural concept because of its low mass and simplicity of design. The spacecraft structure

[3] The mass allocation for the common structure is distributed over the individual instruments in Table 14.

consists of an aluminium honeycomb sandwich with Carbon Fibre Reinforced Plastic skins, similar to the Cluster spacecraft. The thrust tubes have a diameter of 1.2 metre and a height of 1.5 metre (VPO) and 0.9 metre (VEO).

6.3.2.2 Thermal Design

The thermal environment at Venus is much harsher than at Earth due to the relatively short distance from the sun as well as the significant planetary albedo (see also section 4). In order to limit the heat input, the majority of the external surfaces of the spacecraft are covered with Second Surface Mirror tape, which combines low absorptivity with a high emissivity. Additionally, the solar panels are thermally decoupled from the main spacecraft body, which allows excess heat to be radiated from the back of the panels without significantly affecting the spacecraft body. A first order thermal analysis has shown that a largely passive thermal control system can be used for both spacecraft. A limited number of heaters will have to be operated during eclipses for those subsystems that are sensitive to low temperatures.

6.3.2.3 Propulsion system

The propulsion system consists of a conventional bipropellant system, using Monomethyl Hydrazine and Nitrogen Tetroxide. For the main engine, a third generation EADS Astrium 500 N engine, with an Isp of 325 s., has been baselined, while the low thrust manoeuvres are carried out with EADS Astrium's 10 N bipropellant thrusters (Isp of 290 s.). The propellant is stored in four 140 litre tanks (VPO) and four 50 litre tanks (VEO), respectively. The tank configuration is shown in Figure 7. The propellant tanks are thermally shielded from the main engine by a conical heat shield with titanium multilayer insulation.

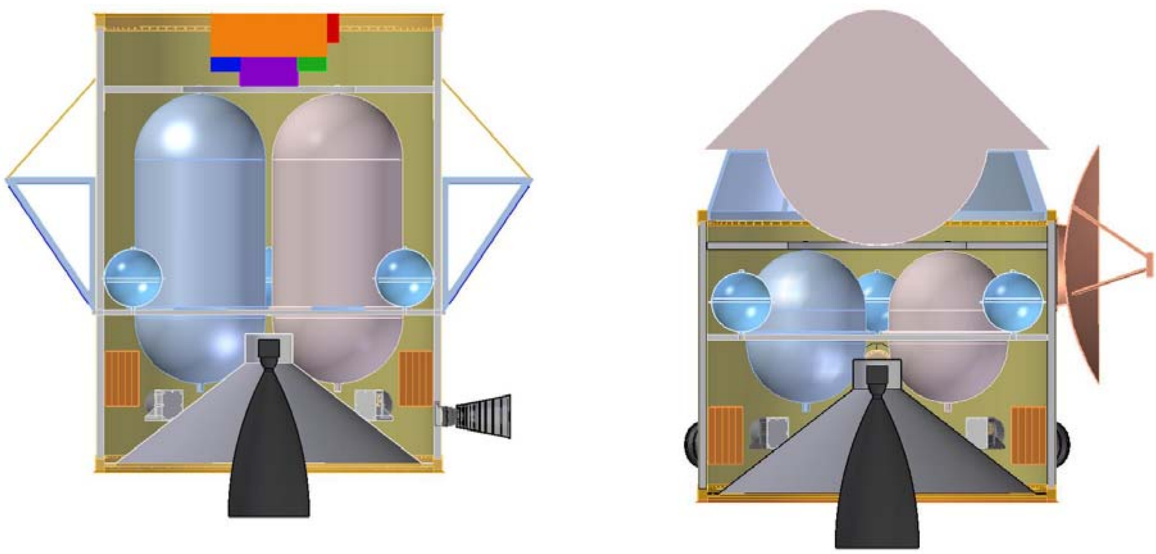


Figure 7: Cross-sectional view of the tank configuration for Venus Polar Orbiter (left) and the Venus Elliptical Orbiter (right).

6.3.2.4 Power

Table 16 list the power requirements for the two orbiters. For both spacecraft, power is generated by GaAs multi-junction solar arrays with an assumed efficiency of 32% (BOL at 28°C and 1 AU).

Table 16: Spacecraft average power budgets (including system level margins)

Operational mode	VPO (W)	VEO (W)	Remarks
Transfer phase	123	75	
Communication mode	112	112	VPO payload <20 W power (quiescent mode)
Science mode	154	-	
Eclipse	111	105	VPO payload 100% operative, VEO communication continues

To allow continuous nadir pointing of the science instruments, the Venus Polar Orbiter has four fixed cant solar panels, each with an area of 0.5 m^2. The Venus Elliptical Orbiter has two deployable 0.16 m^2 solar array panels, which can be maintained orthogonal to the sun vector by a 1 DOF solar array drive mechanism. During eclipses, power is provided by 120 W/kg Li-ion batteries with a capacity of 810 Whrs for VPO and 270 Whrs for VEO. The maximum eclipse times are 38 minutes for VPO and 58 minutes for VEO. The power system architecture is based on an unregulated Maximum Power Point Tracking bus [Clark02].

6.3.2.5 Attitude Control System

The key pointing and stability requirements are listed in Table 17. For the Venus Polar Orbiter, the reference science payload is the primary driver. For the baseline polar orbit of 2,000 × 6,000 km, the pointing stability and knowledge requirements correspond to an atmospheric pixel resolution of 5 km for a 30° FOV (see also Table 5). The Venus Elliptical Orbiter does not carry science payload and the requirements for entry probe release are not demanding. Consequently, the pointing requirements are driven by the communication subsystem.

For commonality, both spacecraft have the same ACS system. Attitude and orbit determination is achieved by two star trackers, with a redundant solution of three sun sensors in combination with fibre-optic gyros. Slewing capability is provided by a four-axis control moment gyro.

Table 17: Key pointing and stability requirements for the orbiter spacecraft.

Spacecraft	Pointing control (arc min)	Pointing stability (arc sec / 30 s)	Pointing knowledge (arc sec)	Remarks
VPO	10	85	85	Science payload
VEO	30	~600		Antenna pointing for Earth communication

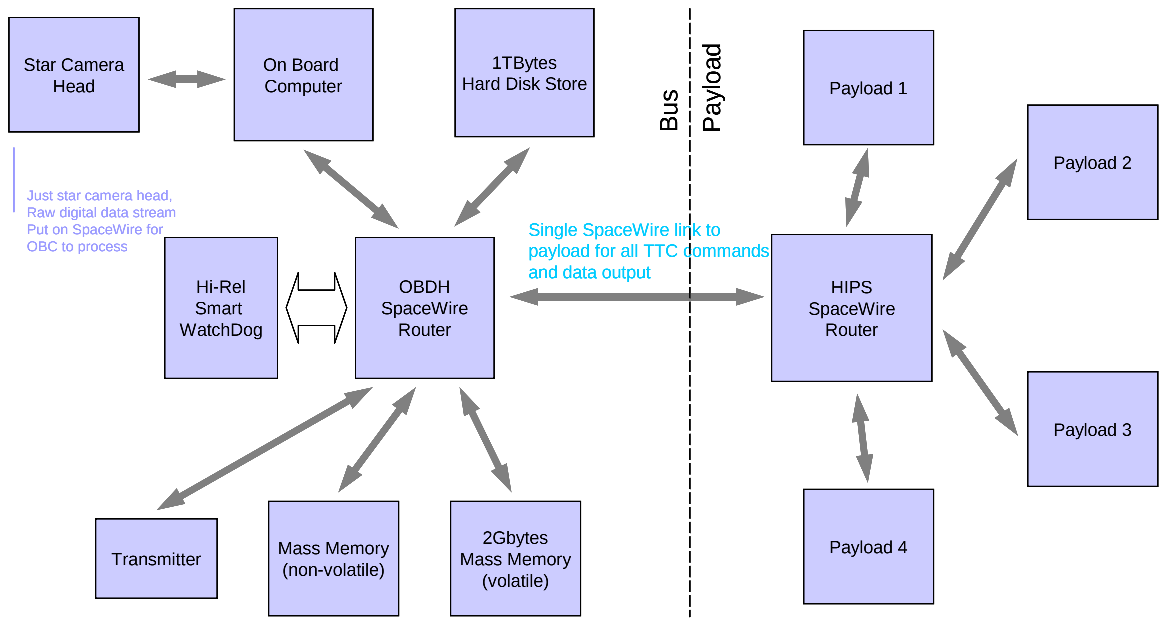


Figure 8: Schematic of the orbiter On-Board Data Handling system.

6.3.2.6 On-Board Data Handling System

The on-board data handling system for both orbiters is schematically shown in Figure 8. It is based on a SpaceWire architecture with a radiation hardened FPGA based LEON processor as on board computer. The OBDH system is completely dual redundant, including the routers. Each module has two connections; one to each router.

Though some amount of solid state memory will be available, the majority of the science data will be stored on dual redundant hermetically sealed COTS hard disk drives. Using currently available COTS hard disk drives, 200 Gbyte could be stored. Over the next five years this is likely to at least double, with a 1Tbyte disk available in the foreseeable future. The large amount of onboard data storage capacity allows varying the downlink data rate depending on Earth Venus distance. It also brings about the possibility to store high resolution data on-board, parts of which can be downloaded after analysis of the medium resolution has shown it to be of scientific interest. Hard disk drives have already been flown on several missions, with as the most recent example SSTL's Disaster Monitoring Constellation. However, in order to use this mission enhancing technology for an ESA science mission, a qualification programme will need to be carried out.

6.3.2.7 Communication

This section assesses the communication requirements and provides the baseline communication architecture for the VEP TRS. Table 18 shows the link ranges for the difference space elements and between Earth and Venus. Clearly, for down-linking the aerobot data, the link distance to the Venus Polar Orbiter is much more favourable than to the Venus Elliptical Orbiter.

Table 18: Link ranges for the Venus Entry Probe space elements.

Distance between	Maximum	Average	Minimum	Units	Notes
VEO – Aerobot	204.4×10^3	141.0×10^3		km	Average distance taking into account direct visibility.
VPO – Aerobot	11.3×10^3	8.4×10^3		km	
VPO – VEO	213.9×10^3	152.3×10^3		km	
Earth - Venus	261.0×10^6 1.74	170.5×10^6 1.14	38.1×10^6 0.25	km AU	

The link rate requirements are detailed in Table 19. The basic mission level assumptions are that the aerobot transmits its data to the Venus Polar Orbiter. The science data generated by the remote sensing payload suite on the VPO are relayed, together with the aerobot data, to the VEO for transmission to Earth. A back-up communications link between VPO and the Earth ground-station will be available at the expense of a significant reduction in science acquisition duty cycle. Both spacecraft will also downlink their telemetry data (either directly or indirectly), and be able to receive telecommand data (either directly or indirectly).

The dominant factor for the communication links between the spacecraft themselves and between the spacecraft and the ground segment are the sizes of the antenna dishes. To achieve the required data rate the use of high gain parabolic antennas is required on both the space and ground segments. For the ground segment, the ESA facilities at New Norcia (35 m aperture antenna) have been baselined.

Table 19: Orbiters and aerobot communication link requirements.

Space element		Data source	Continuously generated data rate (bps)	Notes
From	**To**			
Aerobot	VPO	Entry/descent data	100	Critical performance data during probe entry and descent
		Science data and telemetry	2,552	Ref. section 0
VPO	VEO	VPO science	50,000	Ref. section 6.3.1
		Aerobot data	2,552	Relayed from aerobot to VPO
		VPO telemetry	100	
		TOTAL	**52,652**	
VEO	Earth	VPO data	52,652	Relayed from VPO to VEO
		VEO telemetry	100	
		TOTAL	**52,752**	
Earth	VEO	Command uplinking	1,000	Normal mode
Earth	VPO	Command uplinking	1,000	Failure mode
VPO/VEO	Aerobot	Command uplinking	1,000	Optional

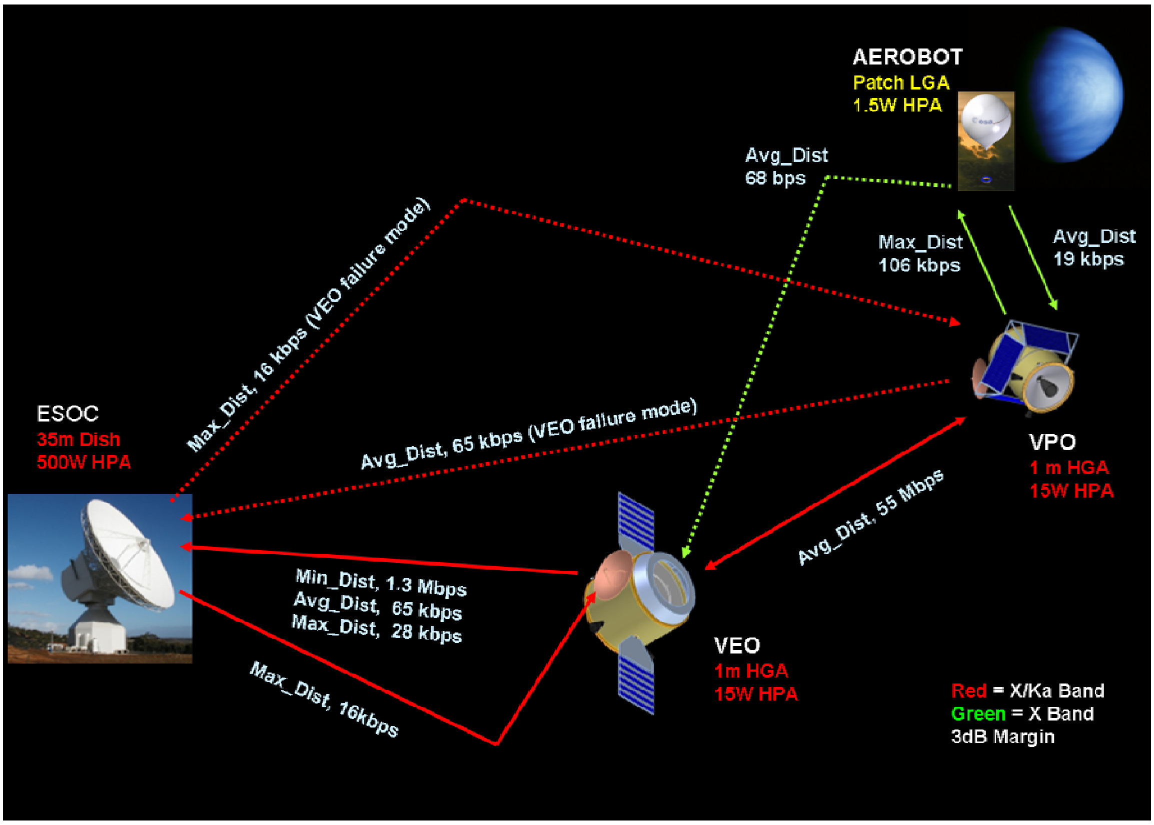


Figure 9: Communication link architecture for the Venus Entry Probe TRS.

A schematic of the baselined communication architecture is provided in Figure 9. Because of the availability of large memory storage on both spacecraft, the data downlink budgets are sized for mean Earth-Venus and VPO-VEO distances. For the command uplink budget, maximum distances have been used. Ka-band will be used for communications to/from Earth as well as for VPO-VEO communications. Aerobot-VPO and aerobot-VEO communications will be carried out at X-band frequencies. Both orbiters have a 1 m diameter high gain antenna and an X/Ka band transponder with a 15 W high power amplifier (HPA). The orbiters also carry omnidirectional X-band antennas for the near Earth phase as well as for emergency mode operation.

Comparing the data rate requirements in Table 19 with the average data link budgets in Figure 9, the communication duty cycles can be determined. Due to the short distances between the spacecraft, the VPO only needs to communicate with the VEO spacecraft for 0.1% of the time, leaving 99.9% for nadir pointing science acquisition. The dedicated VEO data relay spacecraft subsequently transmits all data to Earth, requiring a communication duty cycle of 81% to transmit all generated data at average Venus-Earth distance. At large Earth-Venus distances, not all acquired science data can be transmitted straight away, but this is more than compensated by the high link rates available at shorter Earth-Venus distances. The aerobot-VPO communication windows and duty cycle are discussed in section 6.4.6.7.

6.4 Venus Entry Vehicle

The Venus Entry Vehicle (VEV) system design study has been carried out by Surrey Satellite Technology Ltd, with Vorticity Ltd as subcontractor [Phipps05b]. The assessment and definition of the in situ payload assumed for this study has been performed by Cosine Research [Moorhouse05].

6.4.1 DESIGN REQUIREMENTS AND KEY TRADES

The goal for the Venus Entry Vehicle system design is to fulfil the mission objectives, outlined in section 3.1, with minimal mass and complexity. The key mission requirements have been listed in Table 5. The resulting entry probe design requirements and main trades that have determined the entry probe conceptual design are listed in this section.

6.4.1.1 Atmospheric entry

The entry probe will be released from the Venus Elliptical Orbiter after Venus orbit insertion as this is the simplest way to fulfil the study requirement of concurrent in-situ and remote sensing atmospheric investigations [MR3] (section 3.2). The alternative, direct entry from the interplanetary transfer hyperbola, would require a complicated interplanetary transfer trajectory or orbit insertion scenario.

In order to avoid a complex entry vehicle design, a passive entry has been baselined with the Venus Elliptical Orbiter providing the required propulsive manoeuvres and probe orientation.

6.4.1.2 Aerobot

The aerobot is designed to float at an altitude of 55 km (29 °C and 0.53 bars [Seiff85]), which is at the higher end of the desirable altitude range of 40 – 57 km [MR1] (section 3.2). At lower equilibrium float altitudes, the ambient temperature quickly increases (see Table 7), which would necessitate the use of either thermal insulation or instruments and electronics that can operate at high temperatures. The former limits the balloon flight time and adds mass, while the latter adds significant complexity.

The design goal for the aerobot operational mission duration is to travel at least twice around Venus, so that all longitudes are visited twice [MR1]. Taking the average speed of 67.5 m/s from the VEGA balloons that flew at a similar altitude [Andreev86], one obtains a minimum flight duration of 14 days. The latitudinal coverage that can be achieved for a cruise altitude of 55 km is expected to be minimal and difficult to predict; one of the VEGA balloons (deployed at 7°N) experienced a negligible polewards drift of 0.2 ± 1.3 m/s, while the second (deployed at 6°S) floated towards the equator with a velocity of 2.5 ± 1.2 m/s [Crisp90]. Also at higher latitudes the meridional winds, at an altitude of 55 km, are weak [Gierasch97]. Latitudes above 60° are not recommended for long duration ballooning concepts due to the expectedly dynamically unstable environment caused by the polar vortices and the ‘polar collar.’ In view of the above, an entry latitude of 20° N has been baselined for this study. The optimum insertion longitude is close to the morning terminator as this maximizes the aerobot lifetime (power provided by sunlight, see below).

A light gas balloon with slight overpressure is considered the most suitable candidate for the Venus aerobot, because such a balloon complies best with the operational requirements for a long duration mission. As gas leaks out of the super pressure balloon, the float altitude will gradually increase from 55 km up to 55.5 km until there is insufficient gas for positive buoyancy (and the balloon sinks to the surface). A carefully selected microprobe drop scenario partially counterbalances the loss of balloon gas and thus maximizes the operational lifetime. Additionally, a gas release mechanism and gas replenishment system are included in the design. The gas replenishment system not only increases the mission lifetime, but can also compensate for temperature changes in the balloon gas due to gradients in solar radiation at the day/night terminator. The gas venting system safeguards the balloon from bursting due to unforeseen events.

Hydrogen has been selected as the baseline for the balloon inflation gas, with helium as a backup option. Though the mass of gas storage systems for hydrogen and helium are similar, the main advantage of hydrogen is that it generally has a lower gas leakage rate compared to helium, which is a monatomic gas. The main disadvantage of using hydrogen is its hazardousness.

Several options exist for the storage of hydrogen, such as a conventional high pressure gas cylinder, chemical storage, cold gas generators, glass spheres and carbon nanotubes. A pressurised gas storage system has been baselined for the VEP as this is currently the most mass-efficient mature technology for storing hydrogen. The main drawbacks are the associated hazards due to the high pressure as well as the constraints on the entry probe accommodation. Chemical systems, based on e.g. Lithium Hydride (LiH) or Lithium Borohydride ($LiBH_4$) which react with water, offer advantages in terms of packaging and volume, but at the expense of a heavier system with less mature technology. Cold gas generators and storage in thin glass microspheres that are crushed by a pyrotechnic charge do not provide mass advantages over a conventional gas tank, while the development of technologies for efficient storage of hydrogen in nanospheres and nanotubes is only in its early stages.

6.4.1.3 Gondola

The selection of materials and material-to-material connections will require careful attention. Due to the corrosive nature of the atmosphere, the gondola must be resistant to the reactive chemicals present in the Venus atmosphere or its exposure adequately limited by some means of protection. Atmospheric electromagnetic activity could induce galvanic coupling between the structure and the atmosphere and between the structure and mounted equipments.

The power system and the science instruments are the key mass drivers for the gondola design. During the day, the aerobot will be powered by amorphous silicon solar cells, which are mounted on the gondola surfaces. For the night primary batteries are baselined. Secondary batteries have been considered, but with current technology, this would add ~4 kg to the gondola mass. The measurement and communication duty cycles need to be optimized to minimize average and peak power during the nighttime.

6.4.2 ENTRY VEHICLE SYSTEM OVERVIEW

Figure 10 shows a conceptual drawing of the entry vehicle. The 45˚ sphere-cone entry probe, with a diameter of 1.2 m, is designed to be stable in the hypersonic and supersonic regimes, so that no active control is required. Most of the volume of the entry probe is taken up by the spherical gas storage tank, which is surrounded by the ring-shaped gondola.

Table 20 summarizes the top-level mass budget for the Venus entry vehicle. The design of the subsystems will be detailed in the next sections.

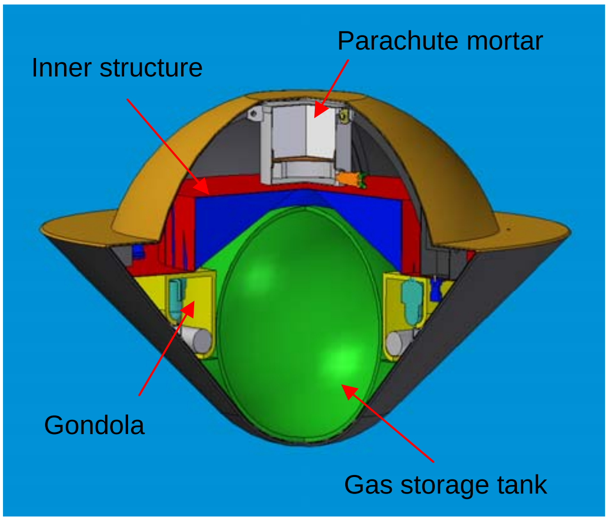


Figure 10: Entry probe geometry.

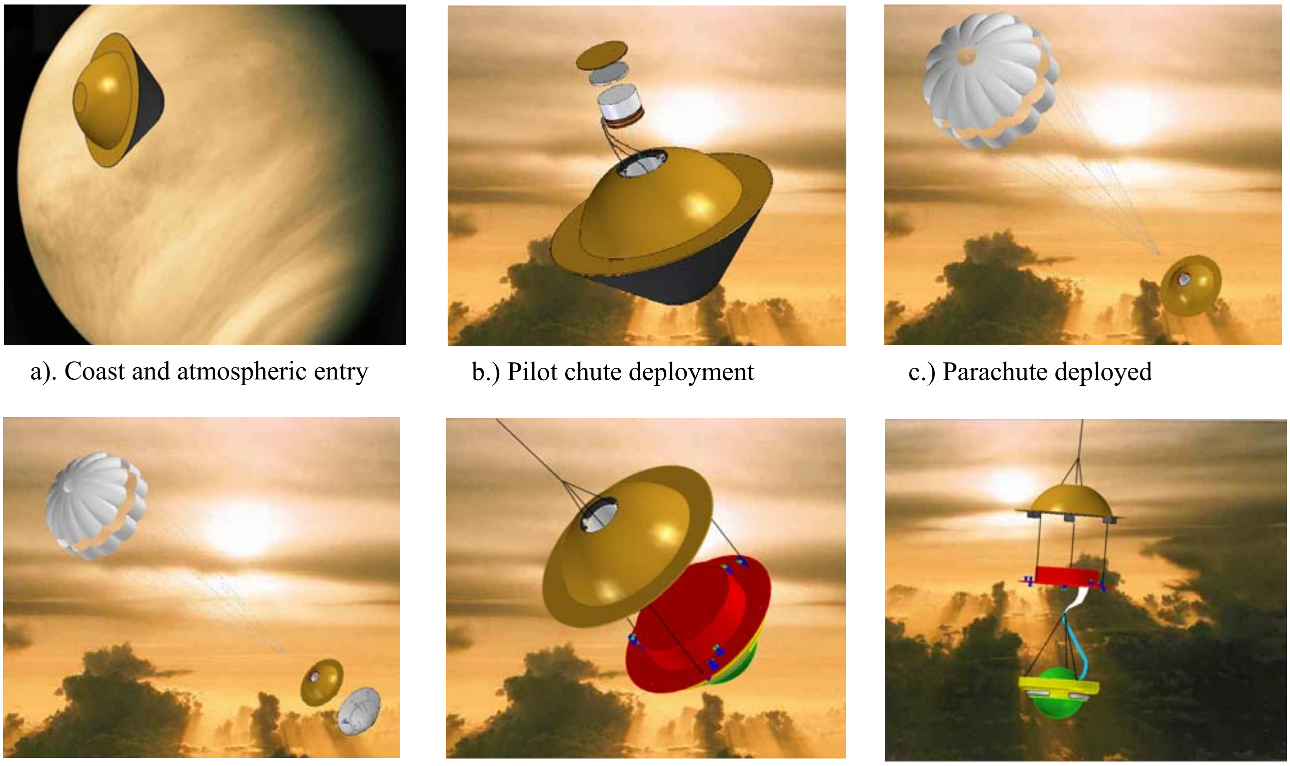
Table 20: Entry vehicle mass budget.

Item	Mass (kg)	Remarks
Gondola	22.7	Incl. science payload
Balloon	9.1	Incl. gas replenishment system
Gas storage system	16.8	
Parachute system	4.3	Incl. parachute mortar
Inner structure	4.2	
Back cover	8.0	Norcoat Liege ablator
Front shield	26.0	High-density ablator
Total mass Entry Vehicle	**91.1**	

6.4.3 ENTRY AND DEPLOYMENT SCENARIO

The deployment of the entry probe from the VEO spacecraft is initiated after the VPO has reached its final operational orbit and the instrument calibration phase has been completed (90 – 180 days after Venus orbit insertion). The deployment sequence is depicted in Figure 11 and the key events are tabulated in Table 21.

The entry probe will be released by means of three pyrotechnic release nuts. Separation will be effected by springs between the orbiter release plane and the probe. After a coast period of approximately 2.5 days, the probe enters the outer limits of the Venus atmosphere. The probe enters the dense Venus atmosphere with a velocity of 9.8 km/s and a flight path angle of -40˚, as this scenario yields a good overall system mass. The steep entry angle ensures a short duration entry and allows a quick release of the aeroshell, thus minimizing the time for the absorbed heat to



a). Coast and atmospheric entry | b.) Pilot chute deployment | c.) Parachute deployed

d.) Front shield release | e.) Rear aeroshell separation | f.) Inner structure release and start of balloon deployment.

Figure 11: Entry and deployment sequence.

soak through the heat shield. Drawbacks are the high spacecraft ΔV-requirement (2 × 70 m/s) and the high g-loads.

The probe velocity is quickly reduced by aerodynamic drag, which generates a maximum g-load of 242g (at 3σ). Just above Mach 1.5, a disk-gap-band parachute is deployed by a pyrotechnic mortar. The parachute stabilizes the probe as it decelerates through the transonic regime. Only 2.5 seconds later, the subsonic regime has been reached and the front aeroshell is released. It is essential to release the aeroshell as soon as possible in the sequence since the heat absorbed during entry soaks through the aeroshell, raising its temperature for a significant time after peak heating. To minimize heating from the back cover, the rear aeroshell is separated from the aerobot by a tether.

At a velocity of ~14 m/s and altitude of 54.5 km, the balloon is deployed. The trigger to start the deployment is provided by a (redundant) pressure switch, as deployment and inflation at a too low ambient pressure (high altitude), could potentially cause the superpressure balloon to burst. The gondola and gas storage tank are released from the parachute / rear aeroshell / inner structure combination. As the inner structure moves away from the gondola, the balloon is extracted from its stowage. Lightweight break-ties separate the fully deployed balloon from the inner structure. The parachute is designed with a small amount of glide to ensure lateral separation between the aerobot and the inner structure, which stays connected to the rear aeroshell and the parachute.

Table 21: Entry and descent event sequence and key characteristics.

Event	Time (s)	Height (km)	Velocity (m/s)	Notes
Release from spacecraft	-2.5d	-	-	Spacecraft de-orbit and re-orbit burn 70 m/s each
Atmosphere interface	0.0	120	9.8×10^3	Flight path angle -40° Ballistic coefficient of 65 kg/m^2
Maximum heat flux	4.78	90	9.1×10^3	Peak heat flux 14 MW/m^2 (3σ) Total absorbed heat ~55 MJ/m^2
Maximum deceleration	5.38	87	5.8×10^3	Maximum deceleration 242g (3σ)
Parachute deployment	15.4	73	359	Mach 1.5 Dynamic pressure less than 3800 Pa (3σ), dynamic force less than 30 kN
Aeroshell release	17.9	72	140	Mach 0.57
Start balloon deployment	725	54.5	14	
Minimum altitude	757	54.3	0	
Equilibrium altitude	23 min.	55.0		

Inflation is started after the balloon is fully deployed. A small drogue is incorporated into the apex of the balloon to control its shape during the first phase of the inflation. The inflation time of the balloon is a trade between aerodynamic loads on the balloon and the minimum altitude. Currently, an inflation duration of 15 seconds and a minimum altitude of 54.3 km are foreseen. The gas storage tank is released directly after inflation is completed. Because the gas cools as it expands into the balloon, it will take another 10 seconds before the balloon has reached its equilibrium volume. The aerobot subsequently rises in about 10 minutes to its cruise altitude.

6.4.4 DESIGN OF THE ENTRY AND DESCENT SYSTEM

6.4.4.1 Front heat shield

The critical design parameters for the selection of the front heat shield material are the peak heat flux and the peak stagnation pressure, which are primarily determined by the entry flight path angle and the probe ballistic coefficient. The Venus Entry Vehicle has a relatively low ballistic coefficient of 65 kg/m^2 due to the large volume of the spherical gas storage tank. For the baselined entry sequence with a flight path angle of 40°, the peak heat flux is ~14 MW/m^2 and the peak stagnation pressure around 2.3 atmosphere. Because these values are slightly above the surface spallation threshold of mid-density ablators [Laub04], a high density ablator, such as fully dense Carbon-Phenolic, has been selected for the front heat shield. Carbon-phenolic heat shield material, as used on previous NASA atmospheric entry probes, can withstand up to 300 MW/m^2. However, though carbon-phenolic is in general still available, the specific U.S. rayon fabric which was used as the source of the heat shield material is not available anymore. Therefore, an activity for qualification and optimization of currently European available Carbon-Phenolic materials will be required.

The thickness of the front shield thermal protection system largely depends on the total absorbed heat during the atmospheric entry. For the baselined steep entry sequence, the soak time is less than 20 seconds and the total absorbed heat around 55 MJ/m^2. Using the material properties of fully dense carbon-phenolic ablators, a thickness of 7 mm will be required. The ablator thickness has been sized conservatively using a margin on radiative flux of 50% and on convective flux of 20%. Additionally, as a carbon-phenolic heat shield material would need to be (re)developed, a 20% subsystem margin has been applied.

The carbon-phenolic ablator is bonded to the front aeroshell, which is manufactured from a carbon-carbon composite. Thermocouples and recession sensors will be incorporated in the front shield ablator in order to assess the performance during actual entry [Martinez04].

As a possible alternative, (improved) medium density ablators can be considered. This would likely require a shallower entry angle (and thus higher total absorbed heat), but this disadvantage might be compensated by the significantly lower entry probe g-loads and lower spacecraft ΔV requirements, the inherently better insulating properties of medium density ablators as well as the lower mass density of the ablator.

6.4.4.2 Rear aeroshell

The rear aeroshell structure is manufactured from carbon-carbon composite. A patch antenna is mounted on the back structure to allow transmission of critical performance data during entry and descent (subject to communication blackouts due to plasma interference). A lightweight ablator (e.g. Norcoat Liege) is bonded to the outer surface of the back cover. If the baselined ablator is not transparent at radio frequencies, a different ablator, such as e.g. PTFE, will be used on top of the antenna.

6.4.4.3 Parachute system

At the top of the back cover, a circular opening exists for the deployment of the parachute, which is sealed by a breakout patch. A low-pressure mortar, similar in design to the Huygens and Beagle-2 probes, will deploy the parachute through the breakout patch.

The parachute will be a 3.57 m reference diameter Disk-Gap-Band design, as this type of parachute exhibits good supersonic opening and flight characteristics and has also been successfully used for the Huygens probe. The parachute size is determined by the requirement to separate the probe from the front aeroshell. In order to achieve this, the ballistic coefficient of the VEP under the parachute must be no more than 70% of the ballistic coefficient of the released front shield. The lines and canopy will be manufactured from polyester for compatibility with the Venusian atmosphere.

The deployment Mach number for the parachute is not sensed directly, it is inferred from the acceleration of the probe. The range of Mach numbers at parachute deployment will lie between 1.36 to 1.61 (3σ) assuming a 5% inaccuracy in the accelerometer measurement (1σ) and typical variability in entry angles and aerodynamic coefficients.

6.4.4.4 Inner structure

The inner structure provides storage for the packed balloon and links the aeroshells with each other and with the gondola. The systems are bolted together with release nuts. Additionally, three short lanyards connect the inner structure to the back cover, which allows separation of the back cover from the entry probe payload (see Figure 11e).

6.4.4.5 Entry sequence control and detection system

A radiation hard FPGA-based sequencer controls the entry and deployment events and commands the firing of the parachute mortar, the pyrotechnic release nuts and pyrovalves. Accelerometers (g switch) and a barostat (pressure switch) provide signal inputs for the entry sequence. The FPGA also reads out the heat shield sensors and relays the critical atmospheric entry data to the orbiters and/or Earth via the gondola transponder. The system, including power generation and management, is completely dual redundant and is accommodated in the gondola (see section 6.4.6.1).

6.4.5 AEROBOT

6.4.5.1 Balloon system design

A conceptual drawing of the balloon system with gondola is shown in Figure 12. The balloon has a spherical shape as this minimizes the envelope mass. The lower part of the balloon is tapered to provide an even load path from the envelope to the riser system. In order to distribute the stress evenly, the bridle assembly will pass over the top of the envelope. Three bridle legs are baselined.

A balloon diameter of 4.0 meters will be required, assuming hydrogen inflation gas, a cruise altitude of 55 km and a total aerobot mass of 31.8 kg (see Table 20). The length of the riser system is 2.0 m. The flying mass is not greatly influenced by the cruise altitude. The initial gauge pressure of the balloon was chosen to be 4000 Pa, representing a 7.0% overpressure of the balloon (at T_{gas} = 34° C). Increasing the initial pressure is not recommended as a higher initial pressure could permanently increase the size of any perforations.

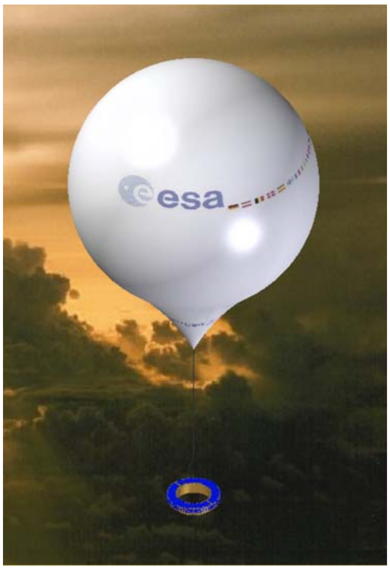


Figure 12: Drawing of the balloon with gondola.

6.4.5.2 Gas storage system

The balloon gassing system consists of a high pressure storage sphere with valves to control gas flow. The tank will be manufactured from a thin-wall Aluminium vessel with a Carbon Fibre Reinforced Plastic overwinding and will be designed to work at an internal pressure of about 300 bars. Release of the gas into the balloon, and separation of the gas tank with the fill line, will be effected by operation of pyrovalves. Once the balloon is fully inflated, the tank will be released from the gondola by means of three lightweight release nuts.

6.4.5.3 Balloon envelope

6.4.5.3.1 Requirements

The balloon envelope material will need to fulfil a substantial number of requirements:

- Resistance to temperature at functional altitudes
- Resistance to atmospheric constituents
- Strong
- Low creep
- Low absorptivity of thermal and solar radiation
- Low permeability to inflation gas
- Ease of balloon fabrication from base components
- Damage resistance
- Availability in discrete thicknesses and sufficiently large areas

During the lifetime of the balloon, the size of the envelope must remain constant. Any change in its size will alter the internal pressure, density and thus the cruise altitude. The balloon envelope should be made from a strong material (to minimize mass), which is stable against all environments encountered during the mission.

The material must be capable of surviving exposure to the corrosive Venusian atmosphere for the minimum mission lifetime of 15 days. The principal species of concern in the atmosphere is sulphuric acid droplets, which could either condense directly on the balloon or fall onto it in the form of rain. Protective coatings could be considered; however, since any disruption of the coating would result in the atmosphere penetrating to the underlying structure, either the coating must be completely robust or must only be required to improve the longevity of a slightly sub-optimal material. Materials laminated together are not 100% effective, especially around seams.

The lifetime requirements of the balloon mandates exceptionally low leakage from the balloon envelope. The major source of leakage in a pressurised envelope is usually the seams; a reliable jointing technique must be established, preferably by using a welded construction. Additionally, the number and length of the joints need to be kept to a minimum in the balloon design. Leakage also occurs through the material itself, requiring a material with extremely low permeability to both the inflation and atmospheric gasses. Often, a thin polymeric or metallic layer can significantly reduce the gas permeability of the base material.

The balloon will be packed tightly for a period of over a year between integration and deployment. It will then be deployed rapidly and inflated while descending through the atmosphere. Subsequently, it will be buffeted by wind gusts throughout the operation life. In order to withstand these environments the envelope material must be flexible, must not stick together or degrade on repeated folding / flexing. If the material is to be coated for protection against the environment it is essential that the coating must adhere without perforation for the lifetime of the mission.

Finally, in order to minimise mass, the material must be no thicker than is necessary to withstand the inflation forces (with margins). Any additional thickness simply adds mass to the system. Most commercially available materials are available only in discrete thicknesses so it is important that the chosen material can be obtained in the optimum thickness.

6.4.5.3.2 Material selection

Many polymers are available in the form of film and new chemical formulations are continuously under development. Table 22 shows a non-exhaustive overview of typical balloon envelope candidate materials against the requirements. Clearly, none of these materials are known to fulfil all requirements. Most promising are PPTA aramid and PET (polyester). The former would need to be better characterized, while the latter material might not be sufficiently resistant against highly concentrated sulphuric acid (depending on the exact PET-type). PTFE, used for the VEGA balloon mission, has excellent chemical characteristics, but its very low strength would pose an unacceptable mass penalty. The provisional baseline for the balloon is PET (polyester), possibly with a PTFE protective film.

Table 22: Selected properties of candidate balloon envelope materials.
PBO = Polybenzoxazole (e.g. Zylon), PE = Polyethylene, PEN= Polyethylene napthalate (e.g. Kaladex, Kalidar), PPTA = Poly(p-phenylene terephthalamide) aramid (aka Aramica), PET = Polyethylene terephthalate (e.g. Mylar, Melinex, Hostaphan), PTFE = Polytetrafluoroethylene (aka Teflon), PVDF = Polyvinylidene fluoride (aka KYNAR). Brandnames for polymide are ULTEM PEI, UPILEX, Kapton and IMIDEX. Polysulphone is also sold as UDEL-P.

Property	**PTFE**	**PE**	**PET**	**PEN**	**PBO**	**PPTA-Aramid**	**Polyimide**	**PVDF**	**Poly-sulphone**
Max T (°C)	250	<90	150	190	200+	180+	230	135	150
Chemical resistance	√√	X	TBD	X	X	TBD	X	√	±
Strength per area density (N/m/[g/m^2])	10	5	140	140	3700	260	90	20	55
Creep	X		√	√	√	√	√		
Optical transmission	medium-high		high	medium-high		medium	medium	high	
Gas permeability	√		√	√	TBD	√	√	√	
Damage susceptibility	√√		√√	√		TBD			X
Many thicknesses	√	√	√	√	X	√	√	√	
Ease of balloon envelope fabrication	√		TBD	TBD	TBD	TBD	X	√	

An extensive development and qualification programme, starting with the detailed characterization of several candidate materials, will be required for the identification and manufacturing of the optimum balloon envelope material (or combination of materials).

6.4.5.4 Balloon envelope coating

Although the diurnal atmospheric temperature range on Venus is negligibly small, variation in solar radiation with local solar time can have a significant impact on the balloon temperature, depending on the thermal properties of the balloon envelope. The balloon temperature range not only determines the mass for the inflation gas, but is also critical in establishing the range of overpressures the balloon will have to endure during the operational phase, which sets the requirements for the strength of the balloon envelope.

A preliminary thermal analysis has been carried out to assess the temperature range that the balloon experiences at Venus. The thermal balance model used assumes a thermal equilibrium between the gas and the balloon envelope and thus only includes radiative and convective heat transfer between the atmosphere and the balloon.

Table 23 lists the minimum and maximum temperatures of the balloon at an equilibrium float altitude of 55 km for several different coatings, using the environmental parameters listed in Table 7. Clearly, in order to minimize the temperature variations, the balloon envelope should be as transparent as possible to solar and infrared radiation. If the balloon envelope material is not inherently transparent, a silver coating is recommended. It should be noted that the calculated balloon envelope temperatures strongly depend on the thermal and solar flux levels, which are not accurately known, as well as on the actual coating thermal characteristics, which will depend on the balloon envelope properties.

6.4.5.5 Gas replenishment system

In order to maximize the balloon lifetime, a gas replenishment system is included. Gas replenishment during the mission (as compared to a higher initial gauge pressure) offers the advantage of a stable gauge pressure during the mission, and allows optimization of the balloon envelope strength vs mass.

Table 23: Equilibrium temperatures for the Venus balloon at an altitude of 55 km (29° C) for several different thermal finishes.

Coating	Solar absorptance α	IR emittance ε	T_{MAX} [SZA = 0°] (°C)	T_{MIN} [at night] (°C)
Clear PET film	~0.02	~0.35	34	28
Silver	0.08	0.66	43	28
Aluminium	0.14	0.05	83	28
Gold	0.3	0.023	135	29

The requirements for a replenishment gas differ from those for an inflation gas. The function of the inflation gas is to inflate the balloon envelope fully at the ambient pressure for as little mass as possible, requiring a gas with a low molecular mass. The purpose of the replenishment gas is to keep the balloon envelope inflated against atmospheric pressure. Since the replenishment gas will provide only a small proportion of the overall gas in the balloon, it is less important that the replenishment gas is light. Therefore, also ammonia or formaldehyde (gasses which can easily be stored in liquid form) can be used.

A trade-off has shown that ammonia is the best option for the replenishment gas system. The baseline gas replenishment system consists of a ~1 kg liquid ammonia stored in a small pressure vessel (at a maximum temperature of 50°C, the vapour pressure is 20 bar), which is incorporated in the gondola body. The volume of the system will be approximately 1.5 litres. When replenishment is required, a small amount of ammonia gas will be released from the top of the replenishment system into the main balloon fill line. As gas is released from the replenishment system the internal pressure will fall, thus allowing more liquid to evaporate. The latent heat of evaporation will be obtained from the environment.

In case the balloon envelope is not resistant against the much diluted ammonia gas, chemically produced hydrogen can be used at the cost of a slight mass penalty.

6.4.5.6 Gas venting system

In order to prevent over-pressure in the case of an unexpected event, a pressure relief valve will be incorporated in the balloon to vent gas before the balloon structural limit is reached. While venting of gas will likely shorten the mission, this is preferable to the balloon bursting as a result of overpressure.

6.4.5.7 Balloon housekeeping sensors

A differential pressure transducer will be incorporated in the balloon fill line. The pressure sensor provides valuable engineering and science data, and is also used on-board to control the gas replenishment system and to ensure a safe the microprobe drop scenario (risk of overpressure).

6.4.6 GONDOLA

6.4.6.1 System overview

The following sections describe the concept design for the in situ payload instruments and the gondola. The functional architecture for gondola system diagram is depicted in Figure 13. The gondola subsystem mass budget is provided in Table 24.

A data processing unit (DPU), which is shared with the science instruments, forms the central controller for the aerobot and handles all data transfer and provides a limited data storage capacity. The data interfaces are based on a CAN bus architecture, as the data rates are sufficiently low. A dedicated memory storage unit for science data is also included. The aerobot payload, assumed for

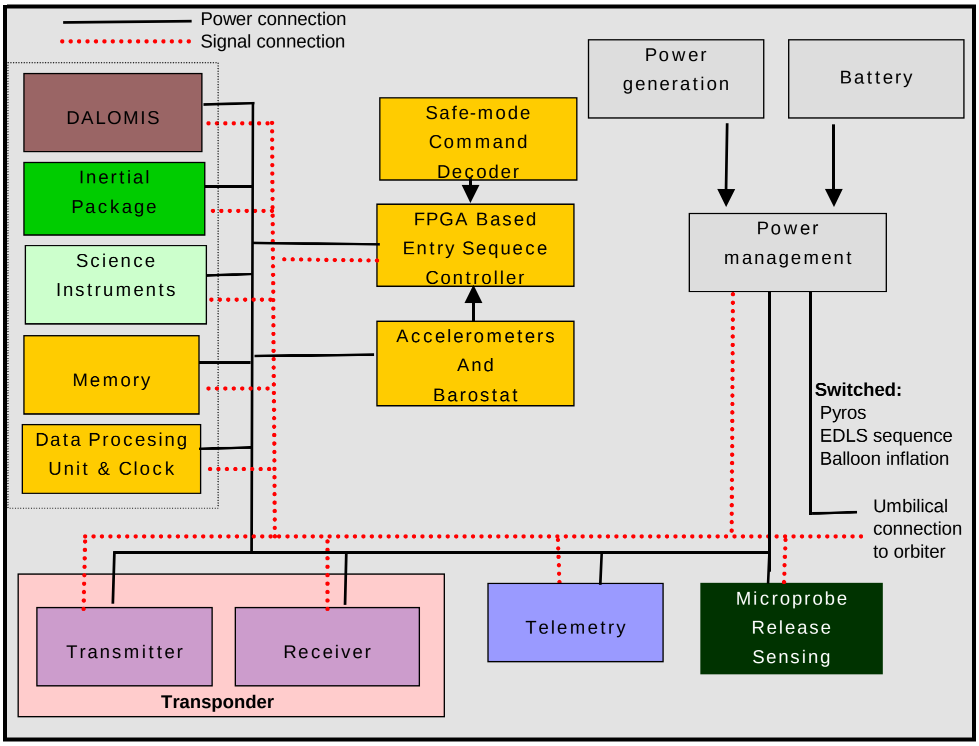


Figure 13: Gondola system diagram.

this study, consists of a set of science instruments, an inertial measurement package as well as a microprobe system, consisting of microprobes, a microprobe release system and a microprobe communication and ranging system (DALOMIS-C). All payloads directly interface with the central data processing unit. The entry sequence control and detection system has been discussed in section 6.4.4.5.

A power management unit controls the on-board power system. It interfaces through an umbilical connector to the entry vehicle (and to the VEO spacecraft prior to release). A transponder unit provides communications and ranging with the orbiters. The telemetry system collects and formats

Table 24: Gondola mass budget.

Item	Mass (kg)	Remarks
Science instruments and inertial package	3.65	Two packages
Microprobe system	4.40	Microprobes, deployment, localization and communication
Communications	1.65	
Structure & harness	6.85	
DPU and memory	0.75	Incl. g-switch and barostat
Power	5.40	
Total gondola mass	**22.70**	

data pertinent to the health and status of the entry system during entry and the gondola and balloon during the aerobot operational phase.

Due to mass and system constraints, the gondola system design is single redundant in many areas with elements of redundancy where possible. Dual redundant platform subsystems include:

- Entry sequence controller
- DPU system
- Power storage (multiple cells)
- Pyros (dual initiators, except on microprobe release system)

6.4.6.2 Configuration

Figure 14 shows a layout of the mechanical configuration of the gondola with subsystems. The solar cells will be mounted on the top and side panels of the gondola structure. As can be seen, the fifteen microprobes take up most of the volume and are accommodated in three slots, each containing five microprobes. The gas replenishment system is not shown in the picture below, but could possibly be located next to one of the payload instrument suites.

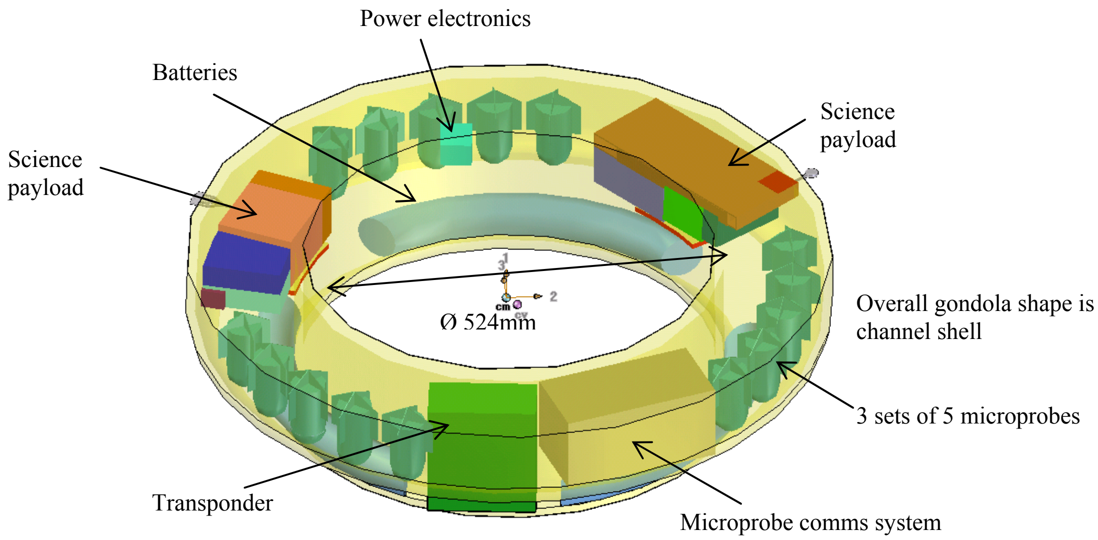


Figure 14: Gondola mechanical layout.

6.4.6.3 Structure

The overall structural design of the gondola is quite demanding. The gondola has important structural interfaces with the entry probe inner structure (see section 6.4.4.4) and with the gas storage tank. These interfaces will transmit significant dynamic and static loads during launch and entry but also have the requirement to separate during the entry and descent sequence. The stiffness characteristics of the complete entry vehicle system will need to be optimized for the dynamic launch loads with special attention to the transfer of the dynamic launch loads into the high pressure balloon gas tank. The gondola structure likely will act as a primary load path during launch, contributing stiffness to the integrated entry vehicle.

The gondola structure will be manufactured from low oxygen grade titanium or titanium – SiC fibre reinforced composite, both highly resistant against concentrated sulphuric acid. The latter material has excellent properties (ultimate strength of ~ 1700MPa, and Young's modulus of ~200GPa) and has been recognised as promising structural material for space and other applications [Eaton94]. To date it has only been used in a limited number of demanding applications due to difficulties with manufacturability, fibre-matrix incompatibility, as well as poor transverse properties [Bednarcyk01]. However, it is projected that in the next ten years, the difficulties may be overcome, which will make this material the optimum choice for the gondola structure. Estimated mass savings, compared to a titanium structure, is 1 kg. Figure 15 shows the structural dimensions of the gondola.

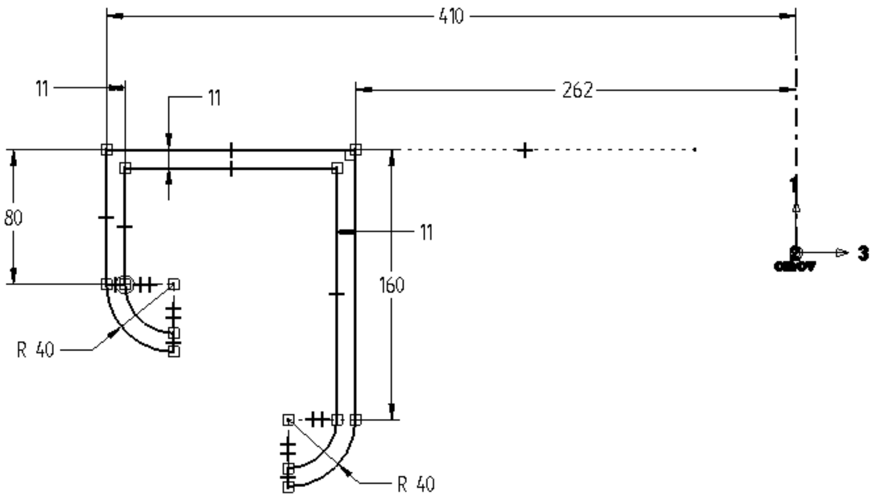


Figure 15: Dimensions of the gondola structure (in mm).

6.4.6.4 Power system

The key power requirements, as used for the sizing of the gondola power system, are summarized in Table 25. During the day, most power will be used when the gondola is transmitting data to the orbiter. In order to save battery mass, no data will be transmitted during the night, though the transponder will regularly send out a low power 'life' signal. At night, the science instruments will be operated with a reduced duty cycle to minimize the power demand from the primary batteries.

Electrical power will be provided by triple-junction amorphous-silicon solar cells, which are mounted on the gondola surfaces, yielding sufficient power during the day. For the night, primary batteries have been baselined.

Due to the practically omni-directional nature of the solar flux within and below the Venus cloud layers, the solar cells can be accommodated on the top and side surface areas of the gondola (see Figure 16). This yields an area of approximately 0.5 m^2. If required, the top surface area available for power generation can be significantly increased by an overhanging structure.

Table 25: Summary overview of gondola power requirements.

Operational mode		Day / night	Science P/L (W)	Microprobe system (W)	DPU and memory (W)	Comms (W)	Total (W)
Microprobe drop campaign (only day time)		Day	6.0	18.0	0.85	0.05	24.9
Communication mode (day time only, limited science)		Day	10.7	-	0.85	14.8	26.4
Science mode	peak value	Day	21.8	-	0.85	0.05	22.7
	peak value	Night	15.5	-	0.85	0.05	16.4
	average value	Night	3.9	-	0.85	0.25[4]	5.0

The peak power generated by the solar cells is estimated as 40 W for a SZA of 45°, assuming a solar cell efficiency of 11% at 85° C. It should be noted that the actual solar flux levels are uncertain and could depend on local weather patterns. In addition, the solar array power yield varies with local solar time (and aerobot latitude). A detailed autonomous operational strategy for the different operational modes might be required to cope with these uncertainties. Alternatively, the primary batteries can be used to provide additional power when necessary, e.g. for communication.

The primary battery system is sized to provide power for a maximum of 8 (Earth) days, thus limiting the operational lifetime of the aerobot to 15-22 days (depending on local time of entry). Lithium-thionyl chloride batteries, which have excellent energy storage capacity for low/medium current applications, will provide power during the night. For operation of the high current devices, such as the pyros and the microprobe release system, a separate lithium sulphur dioxide battery system has been baselined. The baselined batteries will need to be qualified for space as well as for the high g-loads and shocks during launch and atmospheric entry.

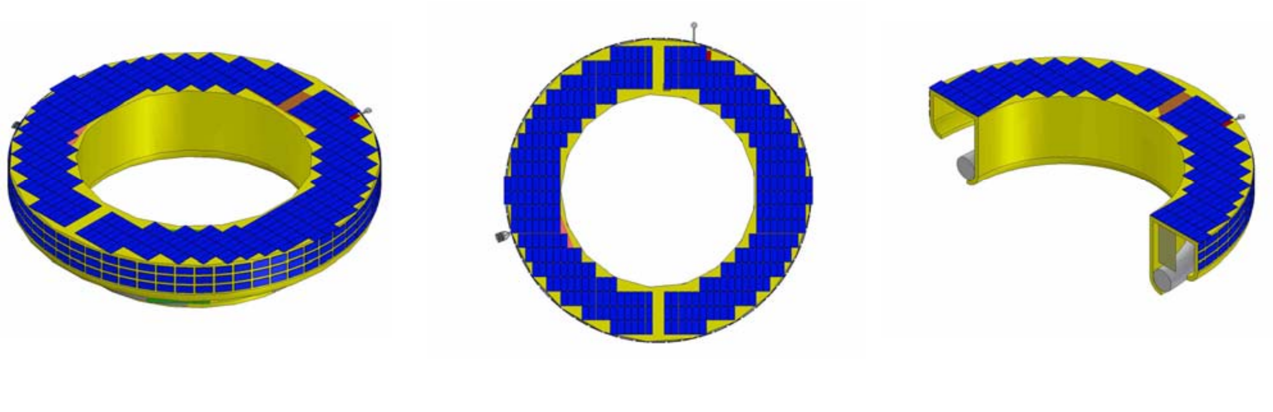


Figure 16: Solar cell accommodation on the gondola structure.

[4] During the night, the transmitter will send out a low power 'life' signal with a low duty cycle.

Table 26: Gondola surface temperatures as a function of local time and surface finish.

Local time	Surface finish	Peak internal power (W)	Surface temperature (°C)
Noon (SZA = 20°)	Ti/solar cells	70	100 – 109
Night	Ti/solar cells	40	35 – 39
Noon (SZA = 20°)	White paint/solar cells	70	70 – 75
Night	White paint/solar cells	40	27

6.4.6.5 Thermal design

During the operational phase, the gondola temperature will be driven by the Venus thermal flux environment along with internal power dissipation. A simple thermal flux balance model with empirical relations for free convective heat transfer has been used to estimate the gondola surface temperature, assuming it is partly covered with solar arrays. The day and nighttime extremes of the gondola surface temperature at the nominal cruise altitude at 20° N are given in Table 26. For most cases, the equilibrium temperature depends on the model (laminar or turbulent convective heat transfer) as well as the surface orientation (vertical/horizontal). The calculations clearly show that it is recommended to paint the gondola surface that is not covered with solar cells white.

More detailed thermal design and analysis would be able to optimize the solution to further reduce the temperatures extremes observed – particularly important for the batteries and payloads. The external solar flux and material properties are the dominant thermal drivers; internal power dissipation has limited effect.

6.4.6.6 On-Board Data Handling System

The On-Board Data Handling System (OBDH) consist of a dual cold redundant central processing unit based on a LEON processor core implemented on a 200krad radiation hard Actel RTAX FPGA with 6 Mbytes of SRAM. A conceptual layout of a single board is shown in Figure 17. A separate flash memory unit provides 512 Mbytes of data storage with low power consumption.

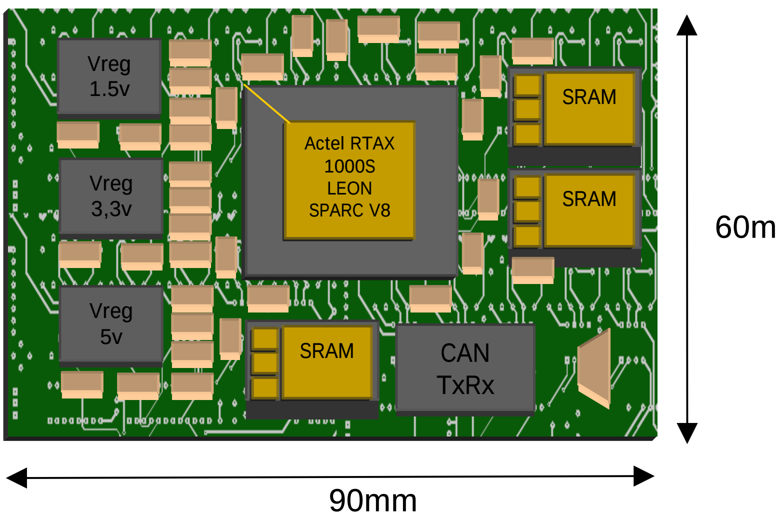


Figure 17: Physical layout for the aerobot data processing unit.

Table 27: Aerobot communication windows and link budget summary.

Link	Sized for (distance)	Link rate (bps)	Average access interval (hr)	Average access duration (hr)
Aerobot to VPO	Average	18,900	2.15	1.05
Aerobot to VEO	Average	68	144	22
Aerobot to 35 m ground station	Maximum	< 1	-	-

6.4.6.7 Communication

The data rate requirement during entry and descent is estimated to be 100 bps, amply sufficient to transmit critical performance data. During the aerobot operational phase, science data is continuously generated at 2.5 kbps (compressed, see section 6.4.6.8), while 52 bps is reserved for housekeeping data (e.g. balloon monitoring, see section 6.4.5.7). The microprobe generated data at 100 bps for less than one hour per microprobe is included in the science data rate. The total aerobot data rate thus becomes 2.55 kbps.

The aerobot communication system consists of a low mass 1.5 W X-band transponder with ranging capabilities. Several substrate antennas are present on the gondola top surface, which combined together form a complete hemispherical coverage antenna. During entry, patch antennas on the back cover will be used (see 6.4.4.2) to transmit critical performance data.

A link access analysis was undertaken using *Satellite Tool Kit* to establish the communications visibility from the two orbiters to the aerobot floating in the Venus atmosphere. Table 27 provides a summary of the communication windows and the achievable down-link rates. As the Venus Elliptical Orbiter has infrequent accesses (approximately every 6 days) at long average distances, the aerobot will downlink all its data to the Venus Polar Orbiter.

For a VPO-aerobot link rate of 19 kbps (at the average distance of 8,400 km, see Table 18), a communication duty cycle of 13.5% is required to transmit all data generated by the aerobot (2.55 kbps). Since the (day and night) communication windows permit a duty cycle of 33%, it is possible for the aerobot to operate the transmitter only during the daytime, thus saving power during the night. During the night a low-power, low duty cycle life signal will be transmitted (see 6.4.6.4).

Achievable aerobot uplink rates at maximum distances are 106 kbps and 322 bps, for VPO-Aerobot and VEO-Aerobot respectively. From a 35 meter ground station, at maximum Earth-Venus distance, a commanding link rate of 8 bps can be achieved. At minimum distances, this increases to 320 bps. Aerobot downlink to a single 35 m ground station is not viable due to the limited aerobot power.

6.4.6.8 Aerobot reference payload suite

The aerobot payload suite assumed for this study consists of a comprehensive set of typical instruments that can perform a detailed in-situ investigation of the atmosphere and additionally provide attitude and altitude knowledge. The instruments and their allocated resource budgets are listed in Table 28. As mass and power are key drivers for the Venus aerobot, an assessment study has been carried out to minimize these resources by integrating the reference payload instruments into two highly integrated payload suites. The most challenging of these instruments is a fully miniaturized Gas Chromatograph Mass Spectrometer with aerosol analysis package, which not only requires a miniature sensor front end, but also an extremely low mass gas and aerosol handling system (requiring silicon-micromachined injection valves, microbore capillary columns, and small ion pumps).

In order to comply with the stringent power constraints, particularly during the night, the instruments can not be operated continuously. This will only moderately affect the overall science return due to the long operational lifetime (required to investigate spatial and temporal variations). During the night, duty-cycled instrument operation reduces the science payload power consumption from its maximum of 22 W to an average of 3.9 W. During day-time, an average of 15 W is expectedly available for the science instruments.

A conceptual layout of the two highly integrated payload suites (HIPS) as accommodated in the gondola structure is shown in Figure 18. The payload suites also accommodate the dual redundant DPU with memory storage, which has been detailed in section 6.4.6.6. The payload has been divided into two sections, 120° apart, as this has several advantages for the payload instruments:

- An almost 4π field of view for the solar flux measurements (using two sensor heads)
- Large separation for the two radar altimeter antennas (one at the bottom of each HIPS)

Table 28: Gondola instrument suite and resource requirements assumed for this study.

Instrument	Key measurements	Mass (kg)	Peak power (W)		Compressed data rate (kbps)	Duty cycle (night) (%)
Gas chromatograph / Mass spectrometer with aerosol inlet	Isotopic ratios of the noble gases, minor gas constituents, aerosol chemical analysis	1.1	6 - 10		1.6	17
Gas chromatograph/Mass spectrometer (sleep mode)		-	1		-	83
Solar and IR flux radiometers	Radiative balance (up- and down-welling solar flux levels, IR net flux levels)	0.30	2.3		0.3	10
Meteorological package	Pressure, temperature	0.07	0.6		0.1	10
Inertial package	Acceleration and attitude	0.06	0.6		0.1	20
Polarization nephelometer	Aerosol size distribution and particle density	0.20	1.7		0.2	20
Radar altimeter	Altitude (baroclinic instabilities)	1.0	3.0		0.2	10
Structure		0.3	**Avg**[5]	**Peak**	-	
Subtotal		**3.03**	**3.25**	**18.2**		
Margin (20%)		0.62	0.65	3.6		-
Total		**3.65**	**3.9**	**21.8**	**2.5**	

[5] Average power during night-time.

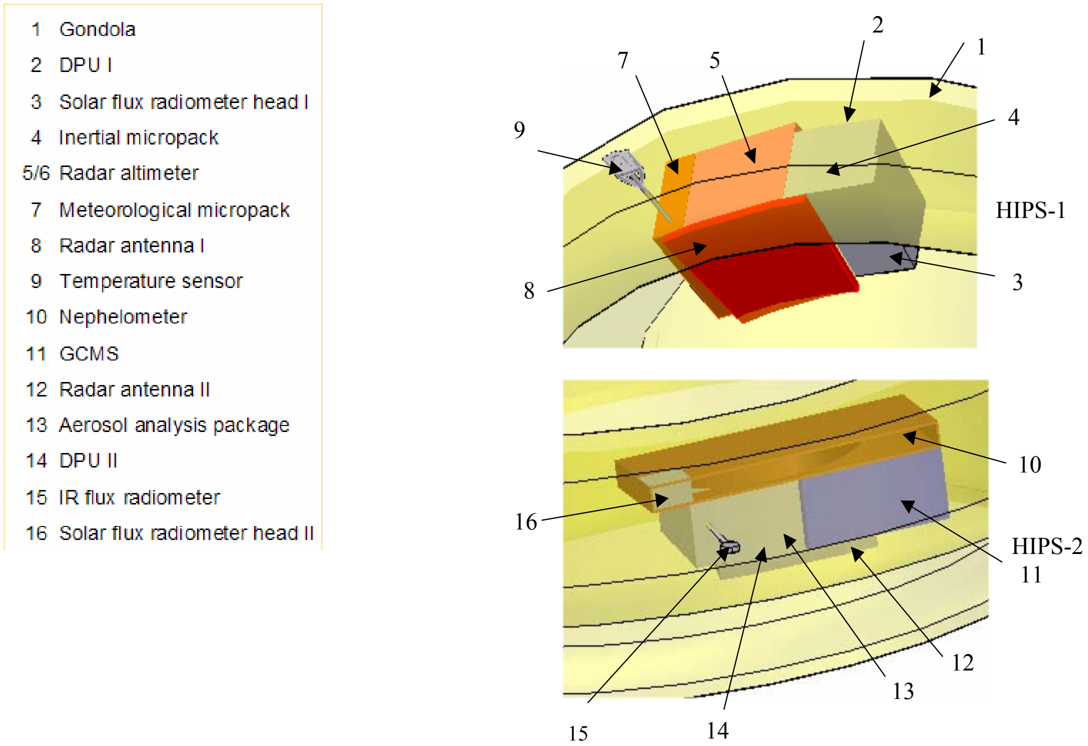


Figure 18: Conceptual design of the two highly integrated payload suites for the Venus aerobot.

6.4.6.9 Atmospheric microprobe system

The system design of the atmospheric microprobes has been carried out under an ESA TRP contract by QinetiQ with the University of Oxford and Laben S.p.A as subcontractors. The technical details provided in this section have been taken from [Wells04] and [Schiele05]. SSTL has investigated the deployment mechanism and the gondola accommodation [Phipps05].

6.4.6.9.1 Introduction

The atmospheric microprobes serve a twofold purpose:

- Perform scientific meaningful measurements (ref. [MO3] in section 3.1)
- Drop ballast in order to increase the aerobot operational lifetime (ref. section 6.4.1.2)

The aerobot will carry up to 15 atmospheric microprobes. These microprobes will determine in-situ vertical profiles of selected properties of the lower atmosphere from the aerobot float altitude down to at least 10 km altitude. Due to the stringent mass constraints of the aerobot, the choice of atmospheric properties that can be measured is limited to basic measurements such as pressure, temperature, and solar flux levels. The horizontal wind velocity can be deduced from the trajectory of the microprobes.

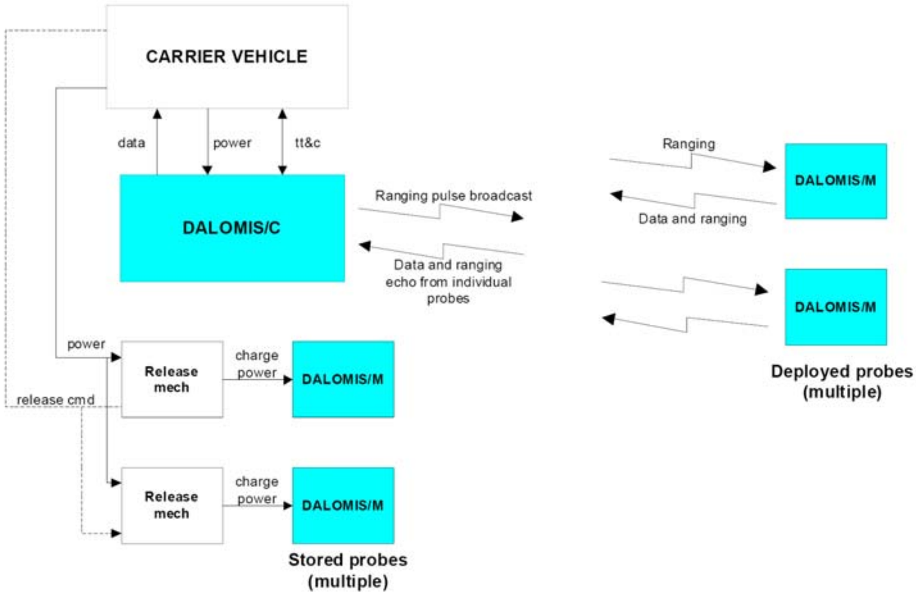


Figure 19: Functional diagram of the microprobe system.

In order to investigate both the local weather patterns on Venus as well the global atmospheric dynamics, the 15 microprobes will be dropped during daytime in 5 separate drop campaigns, spaced equally over the mission lifetime or the first local day (at different local solar times). The three probes in a drop campaign will be released with an interval of about 5 minutes.

6.4.6.9.2 System overview

The atmospheric microprobe system, also called DALOMIS (Data transmission And LOcalization system for Microprobe Swarms), consists of a communication and ranging system (DALOMIS/C), atmospheric microprobes (DALOMIS/M) and a release mechanism. The complete microprobe system is powered and controlled by the gondola. A functional diagram is shown in Figure 19 and the mass budget is detailed in Table 29.

6.4.6.9.3 Microprobe communication and localization system

The S-band communication and localization system is based on a two-way 'active echo' code ranging concept with direction of arrival measurement [Schiele05]. The microprobe actively returns a radar pulse received from the gondola communication and ranging system. A frequency compensated range correlator determines the gondola – microprobe distance with an accuracy of ~75 m, while the azimuth and elevation angles are determined with an accuracy of 0.5° by dual-axis phase interferometry. Simultaneous access to multiple probes is achieved with a Time Division Multiple Access scheme (TDMA).

Table 29: Microprobe system mass budget.

Item		Mass (kg)
Microprobe (× 15)		1.71
- Communication/OBDH	25 g	
- Power	5 g	
- Sensors	15 g	
- Packaging/harness	10 g	
- Structure/thermal	40 g	
Subtotal	**95 g**	
20% subsystem margin	19 g	
Microprobe mass	**114 g**	
Microprobe accommodation and deployment		1.24
Microprobe communication and ranging system (DALOMIS/C)		1.45
Total mass microprobe system		**4.40**

6.4.6.9.4 Microprobe design

The design of the microprobes is dictated by a complex interaction of microprobe mass, aerothermodynamic behaviour, timing and localization accuracy as well as link range requirements. To minimize heating time as well as the link range, the probe should descent as fast as possible. However, a fast descent requires a slender design, which is not optimal for protecting the sensitive electronics to the high ambient temperatures at lower altitudes (see Table 7). Additionally, a fast descent requires a high telemetry rate and a fast localization system.

The baseline design is shown in Figure 20. The microprobe has a diameter of about 4.5 cm and a total length of 11 cm. The aerodynamic shape ensures a quick descent, while the stabilizing fins as well as the low centre of mass provide aerodynamic stability against the strong vertical winds. The nose section is semi-spherical so that the static pressure location is stable and accurately known.

The microprobe structure consists of an open outer shell from boro-silicate glass or alumina and an internal pressure sphere, which contains the electronics and battery. The electronics and battery are thermally insulated by a light-weight foam with a high service temperature (e.g. Microsil microporous insulation), also located inside the pressure sphere.

Released from an altitude of 55 km, the 115 g probe descents in ~30 minutes down to 10 km altitude. The maximum slant angle and slant range to the aerobot is about 60° and 120 km [Henderson05]. The maximum descent speed is ~85 m/s, well below the transonic regime. The maximum electronics temperature will be less than 65° C.

The sensor suite, comprising of several pressure sensors, two solar flux sensors and two temperature sensors, is completely integrated with the microprobe. A multiple pressure sensor system is connected to a plenum chamber that has three openings to the nose at the static pressure locations. The sensor head contains three silicon diaphragm sensors to cover the full pressure range. A single pressure sensor system measures the difference between the probe stagnation pressure and the static pressure, from which the relative vertical probe speed can be derived. Two

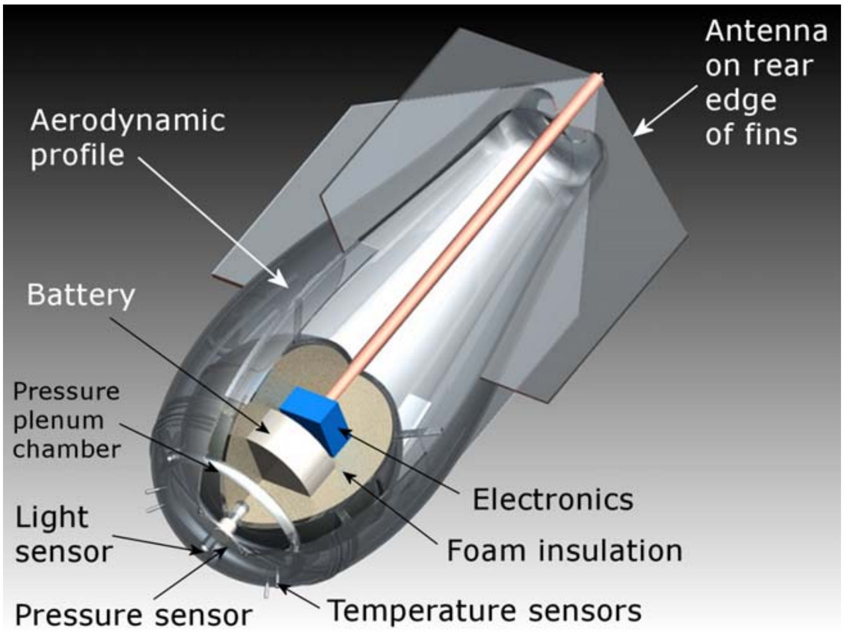


Figure 20: Conceptual design of the atmospheric microprobe.

silicon photodiodes and thermopiles measure both up- and down-welling fluxes, which are fed to the insulated electronics box through an optical fibre with a condensing lens. Two external thin-wire thermocouples determine the ambient temperature during the descent.

The microprobe communication system consists of a regenerative transceiver with BPSK demodulator/modulator units. A sloping dipole antenna is integrated into the rear edge of the tail fins. The OBDH system is time synchronized with the gondola at probe release and contains the TDMA schedule. It will be implemented in a mixed ASIC. The required average power of 0.1 W (peak 2.3 W) is provided by lithium-thionyl-chloride non-rechargeable batteries.

6.4.6.9.5 Accommodation and release mechanism

The microprobes are arranged in 3 ‘cartridges,’ each cartridge storing 5 microprobes (see Figure 14). The microprobes are retained and released by a *Frangibolt* connector. The *Frangibolt* is memory metal activated, about 40 seconds is required to heat the actuator. The actuator fractures the bolt releasing the microprobe. The time of release is sensed by a series of opto-switches or Radio Frequency Identification Devices.

7 CONCLUSION

Technology Reference Studies are detailed mission concept studies with the aim to identify enabling and enhancing technologies that are relevant for potential future science missions. The Venus Entry Probe TRS concentrates on in-situ atmospheric exploration of Venus and other planetary bodies with a significant atmosphere.

The VEP TRS mission concept study has been primarily driven by the objective to establish a technically feasible and scientifically meaningful mission profile for a cost-efficient in-situ exploration of the Venus atmosphere. This has resulted in a mission concept comprising of four building blocks: a comprehensive atmospheric orbiter, a long duration aerobot, an atmospheric microprobe system as well as a dedicated data relay satellite. For cost-efficiency reasons, the system design study has put a strong focus on reducing the number as well as the complexity and technology horizon of the critical technologies, and to baseline existing technologies whenever available. A summary of the key enabling and mission enhancing technologies is provided in Table 30. The definition of Technology Readiness Levels (TRL) can be found in Table 31.

Many other mission concepts for future exploration of Venus can be envisaged, as exemplified by the numerous concept studies (outlined in section 2.2.2). Different objectives (e.g. ionosphere, atmospheric dynamics, atmosphere-surface chemistry, volcanism, surface topology, tectonics) or constraints (e.g. no atmospheric remote sensing concurrent with in-situ investigation) will result in different mission concepts. The Venus Entry Probe TRS should therefore be considered as a reference concept, which aims to assist mission designers in assessing the technological complexity and challenges for Venus mission concepts that are tailored to fulfil a certain set of objectives.

Table 30: Summary of key enabling and enhancing technologies for the VEP TRS.

Space element	Technology	Criticality	TRL	Notes
Orbiter	On-board Computer	enhancing	3 – 4	Scaleable design based on FPGA with LEON processor core and SpaceWire architecture
	Hard disk drive	enhancing	6	European flight heritage exists.
	Highly efficient GaAs solar cells	enhancing	1 – 3	Efficiency increase from 27% to 32% assumed using Quad/triple junction cell technology.
	Light-weight solar array drive mechanism	enhancing	4	Not available commercially for small-sats.
Entry system	Protective heat shield for Venus entry	enabling	2 – 4	Development and qualification of a medium or high density ablator. Possibly requires upgrade / development of test facilities. A first step would be to verify applicability of currently available TPS materials.
	Heat shield monitoring sensors	enabling / enhancing	5 – 6 (U.S.)	Essential to comply with Beagle-2 recommendations. European TRL is ~2.

Space element	Technology	Criticality	TRL	Notes
Aerobot	Balloon envelope	enabling	1	Although a number of potential materials have been identified, none complies with all requirements. Further research and development into lightweight materials is required. A short test programme would allow the assessment of the most likely candidate materials (or combination of materials).
	Alternative gas generators	enhancing	1	Alternative gas generators might offer significant mass and accommodation advantages (see section 6.4.1.2)
Gondola	In-situ atmospheric instruments	enabling	3	A comprehensive fully miniaturized low resource instrument package.
	Thin-film solar cells	enabling	3 – 5	Acid resistant flexible thin film triple-junction amorphous silicon solar arrays with IRR cover glass and polymer substrate. High packing density is required.
	Primary batteries	enabling	3 – 4	Characterisation and qualification, particularly wrt g-loads and shocks
	DPU with memory	enabling	3 – 4	Highly miniaturized DPU based on FPGA with LEON3 architecture and CAN bus interface. Including flight qualified flash memory (or other low-power mass memory storage)
	Low-mass ranging transponder	enabling / enhancing	8 (U.S.)	No European availability (TRL ~2 for 1 kg transponder)
	Substrate antenna	enhancing	3 – 4	Hemispherical X-band antenna on PCB board. Available for aircraft.
	Gondola structural material	enhancing	4	Improved manufacturing techniques for titanium metal matrix composite or other light-weight structural materials
Microprobe	Localization and communication	enabling	3	Breadboard development under TRP contract
	Integrated microprobe	enabling	1	Full integration of all subsystems

Table 31: Definition of Technology Readiness Levels.

TRL number	Definition
1	Technology concept and/or application formulated
2	Analytical and experimental critical function and/or characteristic proof-of-concept
3	Component and/or breadboard validation in laboratory environment
4	Component and/or breadboard validation in relevant environment
5	(Sub)system model or prototype demonstration in a relevant environment (ground or space)
6	System prototype demonstration in a space environment
7	Actual system completed and "Flight qualified" through test and demonstration (ground or space)
8	Actual system "Flight proven" through successful mission operations

8 LIST OF ABBREVIATIONS

ACS	Attitude Control System
ASIC	Application-Specific Integrated Circuit
BPSK	Binary Phase-Shift Keying
CMG	Control Moment Gyroscope
COTS	Components of the Shelf
DPU	Data Processing Unit
DOF	Degree of Freedom
FOV	Field of View
FPGA	Field Programmable Gate Array
CSG	Guiana Space Centre
GTO	Geostationary Transfer Orbit (defined here as 250 km × 35,786 km)
HIPS	Highly Integrated Payload Suite
HPA	High Power Amplifier
IR	Infra-red
IRR	Infra-Red Reflective
NIR	Near-infrared
OBC	On-board Computer
OBDH	On-Board Data Handling
PBO	Polybenzoxazole (brand name Zylon)
PCB	Printed Circuit Board
PE	Polyethylene
PEN	Polyethylene napthalate (brand name Kaladex, Kalidar)
PPTA	Poly(p-phenylene terephthalamide) aramid (brand name Aramica)
PET	Polyethylene terephthalate (brand name Mylar, Melinex, Hostaphan)
PTFE	Polytetrafluoroethylene (brand name Teflon)
PVDF	Polyvinylidene fluoride (brand name KYNAR)
ROM	Rough order of magnitude
SZA	Solar Zenith Angle (The angle between the local zenith and the line of sight to the sun)
TBD	To be determined
TPS	Thermal Protection System
TRS	Technology Reference Study
TRL	Technology Readiness Level
TRP	Technology Research Programme
UV	Ultra-violet
VEO	Venus Elliptical Orbiter
VEP	Venus Entry Probe, an ESA Technology Reference Study
VEV	Venus Entry Vehicle (entry system + aerobot + atmospheric microprobes)
VOI	Venus Orbit Insertion
VPO	Venus Polar Orbiter

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