AN INTRODUCTION TO THE DESIGN OF THE CASSINI SPACECRAFT

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Abstract. In October of 1997 NASA launched its largest interplanetary spacecraft to date. The Cassini spacecraft will arrive at Saturn in July of 2004 and begin a four year tour of that planetary system. After the spacecraft arrives it will separate into an orbiter and a probe. The Huygens Probe, developed by the European Space Agency, will follow a ballistic trajectory into the atmosphere of the moon Titan. The orbiter will relay signals received from the probe back to Earth and then begin the tour. This article provides an introduction to the design of the Cassini spacecraft. The major engineering functions of mechanical configuration, power generation and distribution, telecommunications, information system, pointing and course correction, and some other miscellaneous design features are discussed. A description of the engineering elements of the Huygens Probe is also provided.

1. Introduction

The Cassini spacecraft was launched on October 15, 1997 from Cape Canaveral aboard the Titan IVB rocket shown in Figure 1.

This article describes the engineering elements of the Cassini spacecraft. Separate sections describe the following subjects: spacecraft system overview, mechanical configuration, power generation and distribution, telecommunications, information system, pointing and course correction, miscellaneous design features, and the Huygens Probe.

2. Spacecraft System Overview

The Cassini spacecraft is the largest interplanetary spacecraft built by NASA. It stands 6.8 m (22.3 ft) tall with a high gain antenna 4 m ($\sim$13 ft) in diameter. At launch the spacecraft weighed about 5655 kg (12 470 pounds) of which 2523 kg was dry mass and 3132 kg was propellant. Table 1 lists the 12 instruments carried by the orbiter and the 6 instruments carried by the probe. Figure 2 shows the spacecraft in the cruise configuration, that is, the configuration of the spacecraft after the last Earth flyby on its way to Saturn. The cruise configuration is noteworthy because the boom for the magnetometer and antenna for the Radio and Plasma Wave Science experiment are deployed and the Huygens Probe is still attached. Figure 3 shows the spacecraft during preparations for launch.

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Figure 1. Titan IVB Rocket (photo courtesy of NASA).

3. Mechanical Configuration

Figure 4 shows several of the major structural elements of the Cassini orbiter. The orbiter has 8 major mechanical assemblies:

High Gain Antenna	HGA (not shown in Figure 4)
Electronics Bus	BUS
Magnetometer Boom	not shown in Figure 4
Upper Shell Structure	USS
Remote Sensing Pallet	RSP
Fields and Particles Pallet	FPP
Propulsion Module	PMS (not shown in Figure 4)
Lower Equipment Module	LEM.

The HGA sits atop the spacecraft. It is a 4 m diameter, parabolic, cassegrain feed antenna constructed of graphite epoxy layers and an aluminum honeycomb core. The HGA assembly also includes the low gain antenna #1 (LGA1), a pair of sun sensors, several feeds for the Radar instrument, and a receive path for the Huygens Probe signal. The HGA serves as the primary antenna for orbiter communications. Additional information about the performance characteristics of the HGA is provided in the section about the spacecraft telecommunications system.

TABLE I

Cassini spacecraft science instruments

INSTRUMENT (SENSORS)	ABBREVIATION
Orbiter Instruments	
Cassini Plasma Spectrometer	CAPS
Composite Infrared Spectrometer	CIRS
Cassini Radar	RADAR
Radio Frequency Instrument Subsystem	RFIS
Magnetometer	MAG
Imaging Science Subsystem	ISS
(Wide Angle Camera)	(WAC)
(Narrow Angle Camera)	(NAC)
Visible and Infrared Mapping Spectrometer	VIMS
Radio and Plasma Wave Science	RPWS
Ion and Neutral Mass Spectrometer	INMS
Magnetospheric Imaging Instrument	MIMI
(Charge-Energy Mass Spectrometer)	(CHEMS)
(Low-Energy Magnetospheric Measurement System)	(LEMMS)
(Ion and Neutral Camera)	(INCA)
Cosmic Dust Analyzer	CDA
Ultraviolet Imaging Spectrograph	UVIS
Huygens Probe Instruments	
Aerosol Collector and Pyrolyzer	ACP
Descent Imager and Spectral Radiometer	DISR
Doppler Wind Experiment	DWE
Gas Chromatograph/Mass Spectrometer	GCMS
Huygens Atmospheric Structure Instrument	HASI
Surface Science Package	SSP

Below the HGA is the electronics bus. The electronics bus features 12 bays and a ‘penthouse’. Each bay consists of an outboard shear plate, an inboard shear plate, and stringers connecting these plates. Within the bays are housed the majority of the spacecraft electronics. The penthouse is mounted to the top of bus bay #11. It is another electronics bay housing additional electronics for the Radar instrument. Bay #1 is on the spacecraft +X axis, i.e., behind the remote sensing pallet. The bay numbers increase in the clockwise direction if viewed from above. The contents of the 12 bays are listed in Table 2.

Figure 2. Cassini Spacecraft in Cruise Configuration.

TABLE II
Contents of orbiter electronics bays

Bay #	Assemblies Within Bay
Bay 1	attitude control computers, sun sensor electronics
Bay 2	power distribution
Bay 3	power subsystem remote engineering units, power control, shunt regulator, pyro
Bay 4	science calibration, magnetometer, and radio & plasma wave electronics
Bay 5	radio system amplifiers
Bay 6	transponders, command detectors, ultra-stable oscillator electronics, telemetry control units
Bay 7	radio frequency instrument electronics
Bay 8	command and data subsystem electronics
Bay 9	solid state recorders, backdoor ALF injection loader
Bay 10	reaction wheel electronics
Bay 11	remote sensing pallet remote engineering units, radar electronics
Bay 12	imaging science electronics, accelerometer
Penthouse	additional radar electronics

Figure 3. Cassini Spacecraft Being Prepared for Launch (photo courtesy of NASA).

The electronics bus is constructed of aluminum. The top of the electronics bus is covered by an electrically conductive cap to create a Faraday cage.

The magnetometer boom (see Figure 2) is mounted to the top of the electronics bus. It is the same design used on the Galileo spacecraft. After the last Earth flyby it was deployed from a canister to its full length of 10.5 m. The boom is constructed of 3 fiberglass longerons and supporting cross members and has a triangular cross-section. The boom is home to inboard and outboard magnetometer sensors. The inboard sensor is located about half way out the boom.

The upper shell structure (USS) connects the bottom of the electronics bus to the propulsion module. It is a conic section of aluminum to which the following assemblies are mounted: remote sensing pallet, the CDA instrument, an articulated reaction wheel assembly, the RPWS antenna assembly, the MIMI INCA and electronics, the ultra stable oscillator, and the probe support electronics. The com-

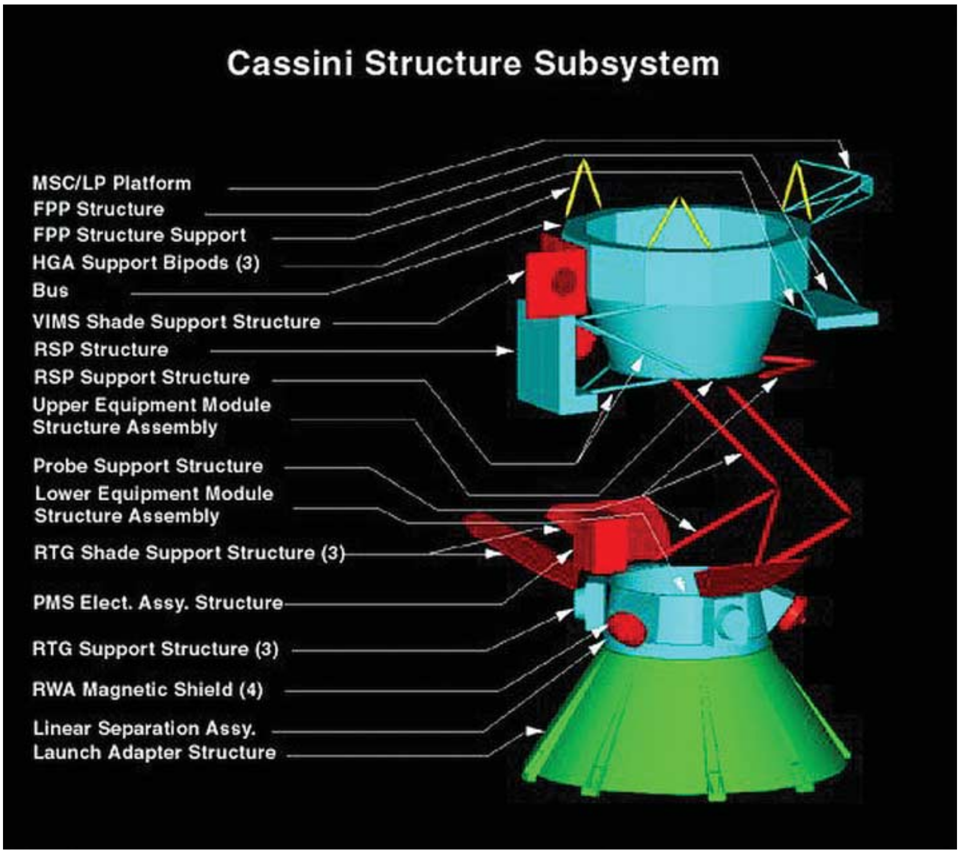

Figure 4. Major Mechanical Assemblies.

bination of the electronics bus and upper shell structure is referred to as the upper equipment module (UEM).

The remote sensing pallet (RSP) shown in Figure 5 and fields and particles pallet (FPP) shown in Figure 6 are two aluminum structures attached to the USS that support science instruments. The RSP supports the ISS NAC, ISS WAC, VIMS, CIRS, UVIS, and two stellar reference units. The fields and particles pallet supports the INMS, the CAPS, MIMI CHEMS, and MIMI LEMMS.

The propulsion module subsystem (PMS) shown in Figure 7 attaches to the bottom of the USS. The primary structure is a cylindrical, semi-monocoque, aluminum shell. Housed within this shell are 2 tanks for bipropellants. Attached to the outside of the shell are a helium tank, spherical monopropellant tank, four thruster booms, two main engines, two pressurant control components assemblies, two propellant isolation components assemblies, and an electronics bay.

The bottom of the spacecraft is named the lower equipment module (LEM). It is also made of aluminum. Attached to the LEM are 3 radioisotope thermoelectric

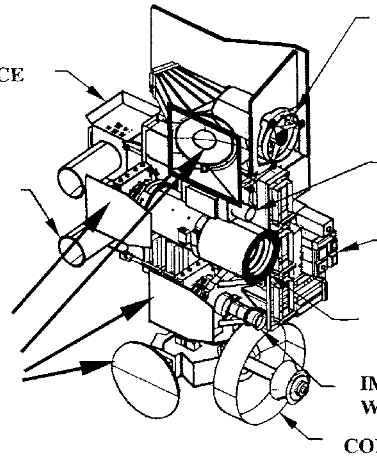

Figure 5. Remote Sensing Pallet with Instruments.

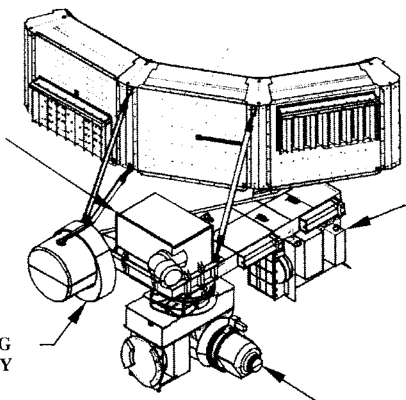

Figure 6. Fields and Particles Pallet with Instruments.

generators (RTG), 3 reaction wheels (RWA), low gain antenna #2 (LGA2), and a deployable and retractable cover for the main engines.

The spacecraft structural coordinate system origin is on the spacecraft centerline in the plane defined by the interface between the electronics bus and the upper equipment module. The +X axis points radially outward in the direction of the stellar reference units' boresights (which is perpendicular to the remote sensing boresights). The +Y axis points in the direction of the magnetometer boom. The +Z axis points down toward the main rocket engines.

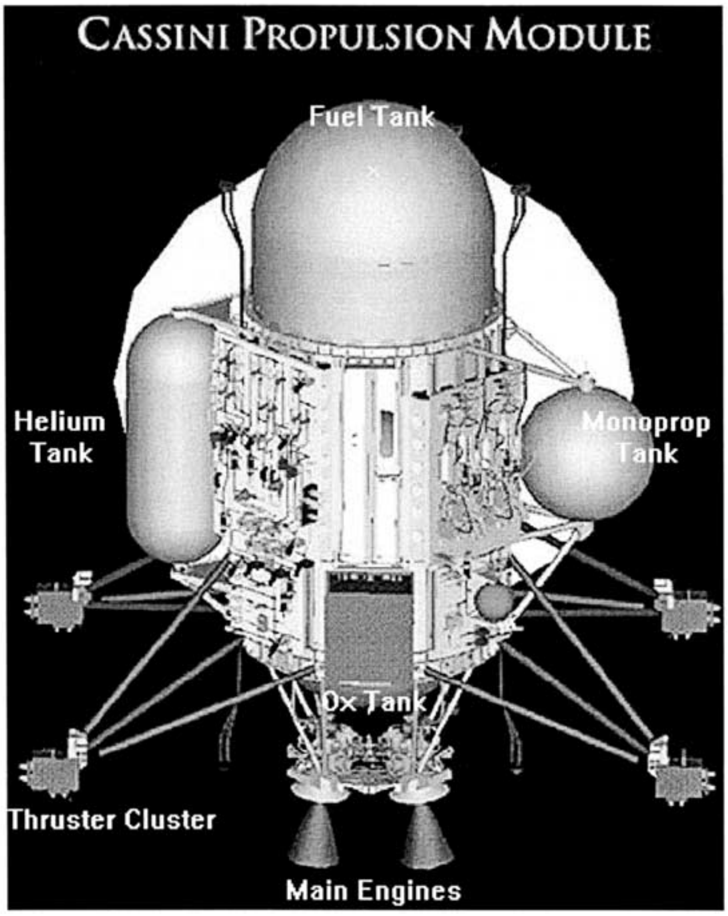

Figure 7. Propulsion Module Subsystem.

4. Power Generation and Distribution

Power generation and distribution is provided by the Power and Pyrotechnics Subsystem (PPS). Figure 8 is a block diagram of the PPS.

The source of electrical power is 3 radioisotope thermoelectric generators (RTGs). Figure 9 shows a cutaway drawing of an RTG. Each unit uses the heat produced by 10.9 kg of plutonium dioxide to generate about 300 W of electrical power at beginning of mission. By the end of the 11 year nominal Cassini mission the output power will degrade to a total of about 640 W.

Power is distributed around the spacecraft by a single power bus. The power bus is controlled to a nominal 30 volts by a shunt regulator. Excess power is directed to a shunt radiator mounted on top of the electronics bus. Both the high and low (return) rails of the bus are isolated from spacecraft chassis by 2 kohms.

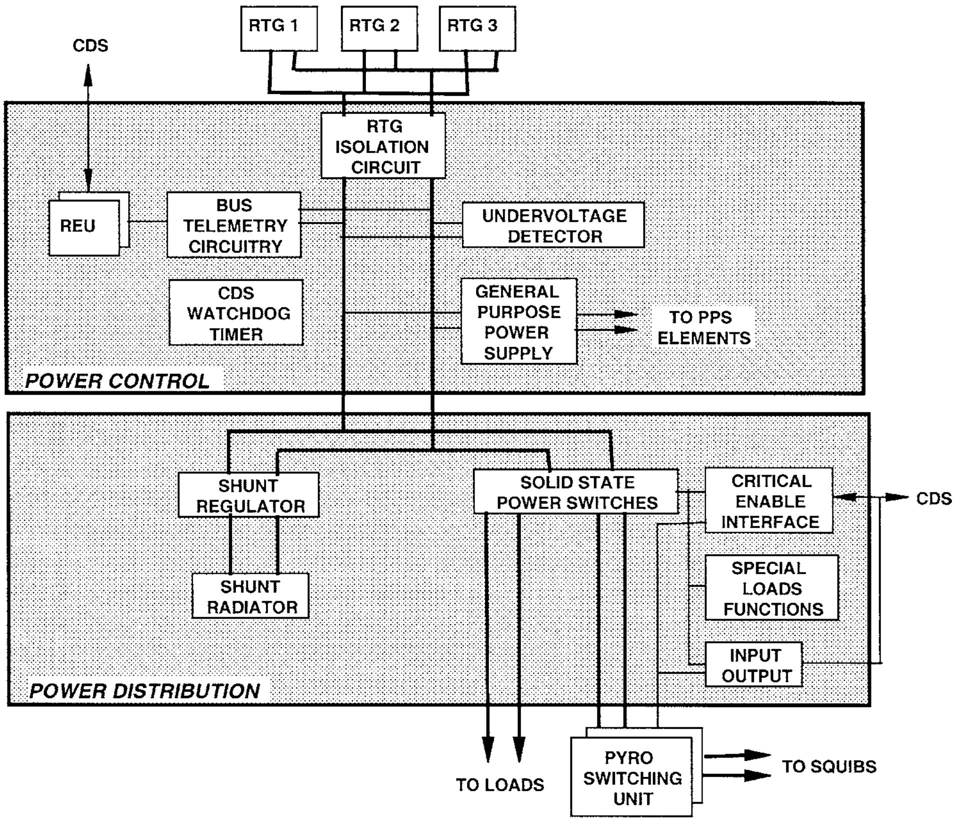

Figure 8. Power and Pyrotechnics Subsystem.

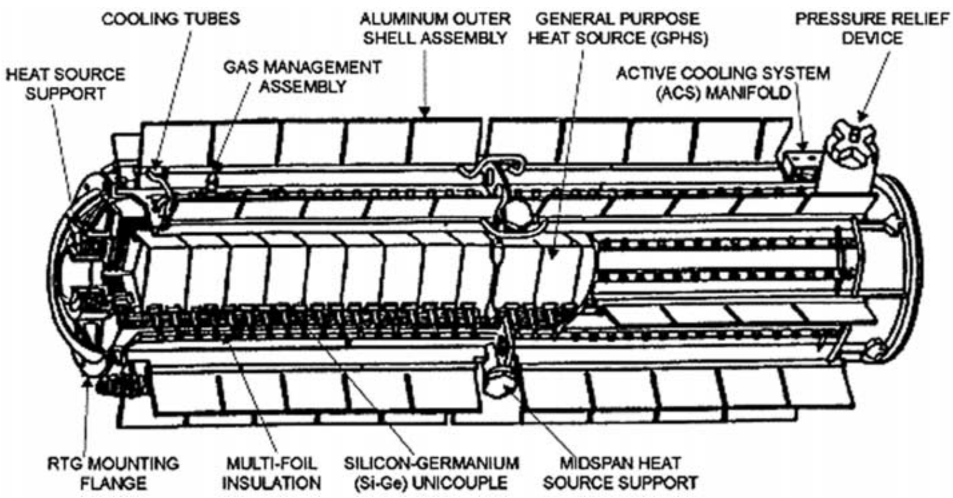

Figure 9. Radioisotope Thermoelectric Generator.

TABLE III

Antenna functions

Antenna	Mode	Frequency (GHz)	Function
HGA			
X-band	Transmit	7.175 ±0.025	Telecommunications
	Receive	8.425 ±0.025	Telecommunications
Ka-band	Transmit	32.028 ±0.1	Science (RFIS)
	Receive	34.316 ±0.1	Science (RFIS)
Ku-band	Transmit & Receive	13.7765 ±0.005	Science (RADAR)
	Receive	13.800 ±0.1	Science (RADAR)
S-band	Transmit	2.298 ±0.005	Science (RFIS)
	Receive	2.097 ±0.005	Probe Relay
	Receive	2.118 ±0.005	Probe Relay
LGAs			
X-band	Transmit	7.175 ±0.025	Telecommunications
	Receive	8.425 ±0.025	

This feature makes the bus tolerant to single shorts to chassis. Transient loads are accommodated by filter capacitors connected to the bus.

Electrical loads are connected to the power bus by solid state power switches (SSPS). These switches feature controlled ramp-up of voltage to loads and hardware controlled automatic shutoff in the event of overcurrent. Each switch can deliver up to 3 amps. Monitors of the on/off state, trip state, and load current through each switch are available in telemetry. The switches switch both the high and low (return) rails of the circuit. In addition, no single failure within a switch can prevent its being switched off. There are 192 SSPSs on the spacecraft.

The PPS also supplies power for pyrotechnics. Two block redundant pyro switching units consisting of capacitor banks and associated electronics perform this function. A silicon controlled rectifier on the high side and an enable relay on the return side switch each pyro circuit. Potentially mission catastrophic events are inhibited by additional critical enable relays that can be positioned only by hardware decoded uplink commands.

The PPS includes several fault protection features. The PPS control electronics include hardware to detect and recover the spacecraft from system undervoltages. The switches to many loads are automatically switched off if the bus voltage drops below a preset threshold. Switches for mission critical loads are left on following undervoltages and are automatically switched on if both elements of a redundant set are off. An example of this is the watchdog timer for the Command and Data

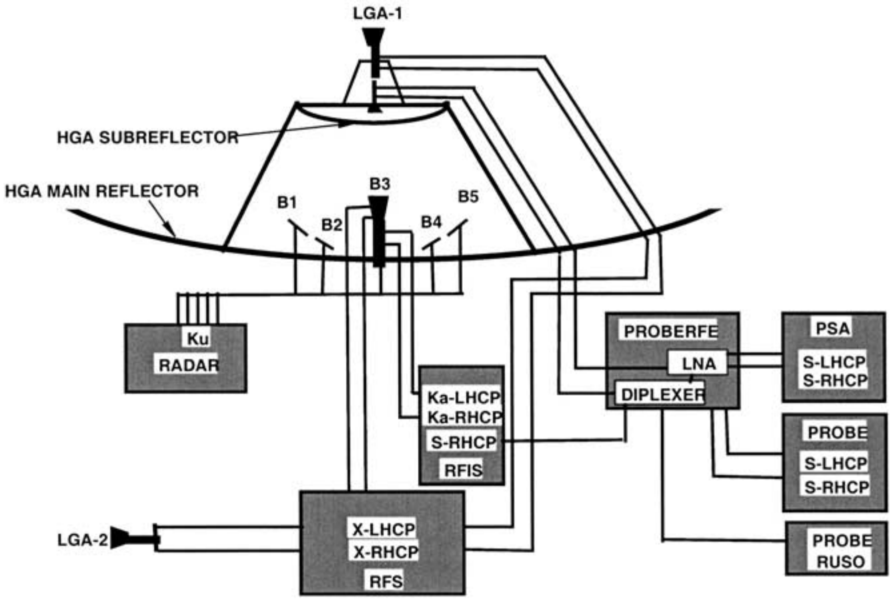

Figure 10. Antenna System.

Subsystem (CDS) processors. The control electronics also include hardware to isolate an RTG from the power bus in the event the RTG suffers an internal short.

Power for the PPS electronics is provided by block redundant general purpose power supplies that put out power at ±5 V and ±12 V.

5. Telecommunications System

The combination of the Antenna (ANT) and Radio Frequency Subsystems (RFS) provides the telecommunications function for the spacecraft.

The ANT subsystem consists of the HGA, LGA1, LGA2, and associated waveguides. Figure 10 is a block diagram of the ANT subsystem. The HGA and LGA1 are mounted on top of the spacecraft facing along the −Z axis. The LGA2 is mounted on the lower equipment module facing along the −X axis. The HGA provides both engineering and science functions. The LGAs serve only engineering functions. The functions and associated frequencies are summarized in Table 3.

The HGA and LGAs feature right hand and left hand circularly polarized ports for the telecommunications function. Key performance parameters of the antennas are shown in Table 4.

The HGA provided by the Agenzia Spatiale Italiana is an example of the international cooperation within the Cassini program.

TABLE IV

Antenna performance parameters

Parameter	Value
HGA on axis gain (X-band transmit)	>46.1 dB
HGA on axis gain (X-band receive)	>44.3 dB
HGA half power beam width (X-band transmit)	9.77 ± 0.35 mrad
HGA half power beam width (X-band receive)	11.34 ± 0.35 mrad
LGA1 on axis gain (X-band transmit/receive)	>7.3 dB
LGA1 gain 90 degrees off axis (X-band transmit/receive)	>−14.0 dB
LGA2 on axis gain (X-band transmit/receive)	>4.2 dB
LGA2 gain 90 degrees off axis (X-band transmit/receive)	>−13.5 dB

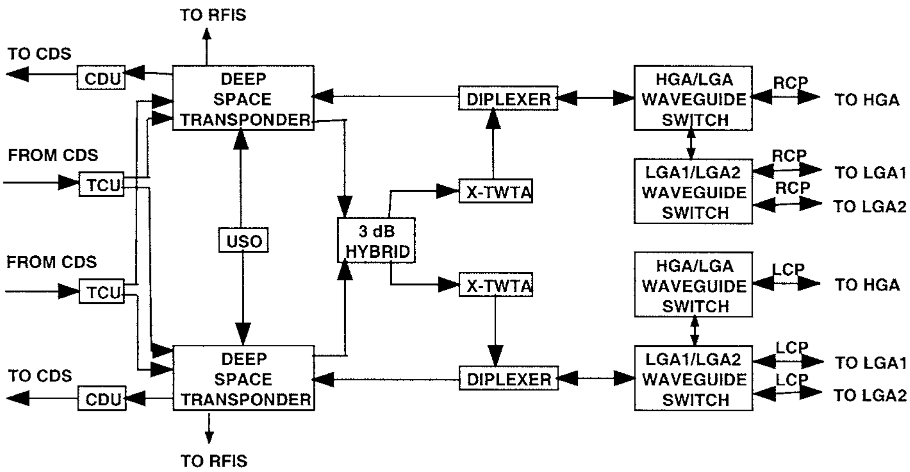

Figure 11. Radio Frequency Subsystem.

The RFS translates radio signals to digital data and digital data to radio signals. It also generates one-way or turns around ranging signals used for navigation. Figure 11 is a block diagram of the RFS. The RFS can receive data, transmit data, receive ranging, and transmit ranging simultaneously or perform any of these functions alone.

Uplink data arriving from an antenna pass through waveguide transfer switches and a diplexer before reaching the receiver within the Deep Space Transponder (DST). The receiver demodulates the 7175 MHz uplink signal and passes the 16 kHz subcarrier signal to the Command Detector Unit (CDU). The CDU demodulates the

TABLE V
Radio frequency subsystem parameters

Parameter	Value
Uplink data rates	7.8125 to 500 bps
Receiver threshold	−156 dBm
Receiver noise figure	2.5 dB
Uplink frequency tracking range	200 kHz
Uplink frequency acquisition range	±650 Hz
CDU threshold (Eb/No ratio)	12.59 dB
RF output power	20 W

subcarrier signal and outputs digital command data, clock, and lock signals to the CDS for interpretation.

Downlink data in non-return-zero digital format is received by the Telemetry Control Unit (TCU) from the CDS at rates from 5 bps to 249 kbps. The TCU modulates this data onto a 22.5 kHz, 360 kHz, or biphase-L subcarrier. An optional convolutional code of constraint length 7 and rate 1/2 or length 15 and rate 1/6 can also be added by the TCU. An exciter within the DST receives the output of the TCU and modulates the subcarrier onto an X-band carrier signal. There are 3 frequency sources for the exciter: auxiliary oscillator, ultrastable oscillator, or receiver controlled oscillator. The DST outputs to a 3 dB hybrid power splitter to both traveling wave tube amplifiers (X-TWTA). The amplified signal is then passed from the powered TWTA through the diplexer and waveguide switches to the selected antenna.

The DST's exciter can generate differential one-way ranging signals used for navigation. There are two tones that can be selected for this function. The DST can also perform turnaround or two-way ranging. The exciter receives a ranging signal from the ground and feeds this signal to the exciter. The exciter remodulates the received ranging signal at one of two indices onto the downlink carrier signal.

Some key RFS parameters are listed in Table 5.

6. Information System

Figure 12 is a block diagram of the spacecraft information system. The assemblies listed in Table 6 are distributed throughout the spacecraft and combine to form the information system.

Elements of the information system communicate over the active portion of redundant MIL-STD-1553B serial data buses. The data buses are redundant pairs of twisted shielded wires. The bus bandwidth is 610 kbps. The clock rate is 1 MHz.

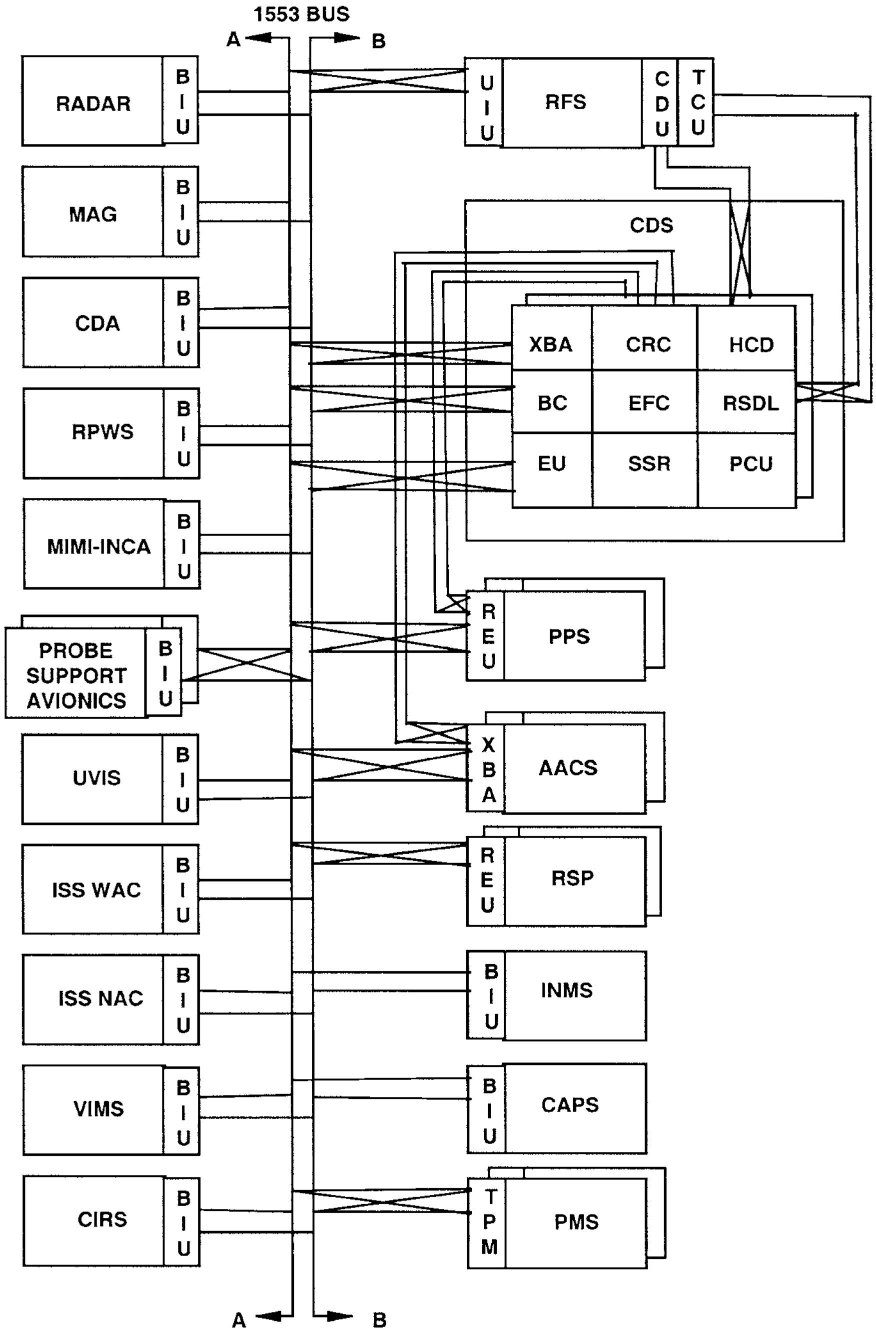

Figure 12. Spacecraft Information System.

TABLE VI
Assemblies of spacecraft information system

Assembly	Abbreviation
Command and Data Subsystem	CDS
Bus Controller	BC
Critical Controller	CRC
Engineering Flight Computer	EFC
Engineering Unit	EU
Hardware Command Decoder	HCD
Reed Solomon Downlink	RSDL
Solid State Recorder	SSR
Cross Strapped Bus Adapter	XBA
Remote Engineering Unit	REU
Bus Interface Unit	BIU
Unique Interface Unit	UIU
Telemetry Processing Module	TPM
Data Bus	

TABLE VII
Engineering flight computer parameters

Parameter	Value
Type	1750A
Software language	Ada
Word size	16 bits
Total RAM	8.2 Mbits
PROM	131 kbits
Throughput	1.28 MIPS

Data on the bus are Manchester II biphase encoded. Signal amplitudes range from 11.5 volts (logic ‘1’) to −11.5 volts (logic ‘0’).

The 1553 standard calls for a single master of the data bus. This function is performed by the CDS BC. The BC orchestrates the distribution of commands, telemetry, and spacecraft intercommunication messages by outputting commands or calling for inputs over the bus. All of the remote terminals connected to the bus are slaves to the BC. The BC’s are in turn controlled by the CDS EFCs over a CDS internal bus. Some key parameters of the EFC are listed in Table 7.

Some remote terminals such as science instruments are connected to the bus through BIUs. The BIU comes in several forms. In addition to a standard BIU there are the RFS UIU and the CDS and AACS XBAs that have all the features of the standard BIU plus some high rate data transfer capabilities. Other remote terminals such as the PPS are connected to the bus through REUs (a.k.a., TPM for the propulsion subsystem's interface and EU for the command and data subsystem's interface). Both the BIUs and REUs have the following interfaces: digital interface to the 1553B bus, discrete status lines to the users, discrete command lines to the users. The REUs also have voltage and temperature interfaces with users.

Data for realtime transmission are formatted by the EFC into transfer frames, optionally encoded with Reed-Solomon code within the RSDL, and sent to the RFS for translation into a radio signal. Over 9000 telemetry channels have been defined.

Data to be recorded is formatted into unencoded transfer frames by the EFC and sent to one of two SSRs. Each SSR has a capacity of 2 Gbits in the form of dynamic random access memory. Because this memory is vulnerable to radiation effects, the SSRs are encased in vaults of half-inch thick aluminum.

The CDS/SSR interface is capable of exchanging data at up to 1.5 Mbps although no software data mode has been defined to operate at this speed. The SSR memory space can be divided by command into as many as 16 partitions. The partitioning protects data from being overwritten and allows for segregation of incoming data into types, e.g., engineering and science data. Data can be recorded onto the SSRs by four different modes: read-write to end, circular buffer, first in first out, and direct memory addressing. Different modes can be used in different partitions simultaneously. Data can be simultaneously recorded and read out from the same partition. All science data are buffered in the SSR before transmission. Instrument health measurements known as 'housekeeping telemetry' are transmitted realtime.

The SSR is also used to store multiple copies of the flight software loads for the CDS, attitude control computers, and instruments. These copies are used in the event radiation corrupts a RAM load of any of these elements.

The CDS also coordinates uplinked commands. Most commands are processed by CDS software. Commands can stand alone and be tagged for delayed or immediate distribution. Commands can also be embedded in sequences. Approximately 2.4 Mbits of RAM is reserved in each CDS for storage of sequences. Up to 256 sequences can be stored. Up to 64 sequences can be simultaneously active. A maximum of 128 commands per second can be sent on the data bus; although, not all of these can originate from sequences as some of this capability has been reserved for fault responses. Over 1100 command stems have been defined.

A few commands need no software interpretation. These commands are interpreted by the HCD. Commands recognized by the HCD are routed to the CRC where latching relays are set. Such commands enable or disable critical events such as release of the probe release or powering off of critical hardware.

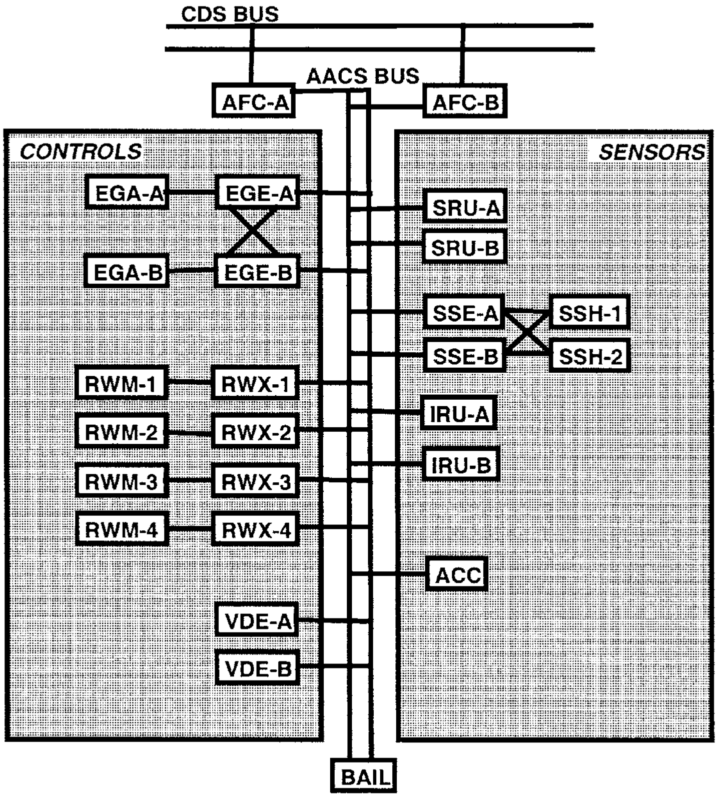

Figure 13. Attitude and Articulation Control Subsystem.

7. Pointing and Course Correction

Figure 13 is a block diagram of the Attitude and Articulation Control Subsystem. Figure 14 is a block diagram of the Propulsion Module Subsystem (PMS). Both subsystems feature block redundancy of critical elements including the main rocket engine. Together, these two subsystems provide the pointing and course correction functions for the spacecraft. The orbiter is designed for 3-axis stable operation. Some key system parameters are listed in Table 8.

The brain of the attitude control loop is the Attitude Flight Computer (AFC). The computer selected is the same computer used in the CDS. This computer controls a separate subsystem 1553 data bus for the AACS sensors and control mechanisms. Table 9 lists the principal flight modes of the AACS software. The normal mode during cruise is Home Base. Transitions are made to various other modes to perform maneuvers or fine pointing operations. The AFC software includes an extensive set of fault protection monitors and responses to manage the configuration of the subsystem.

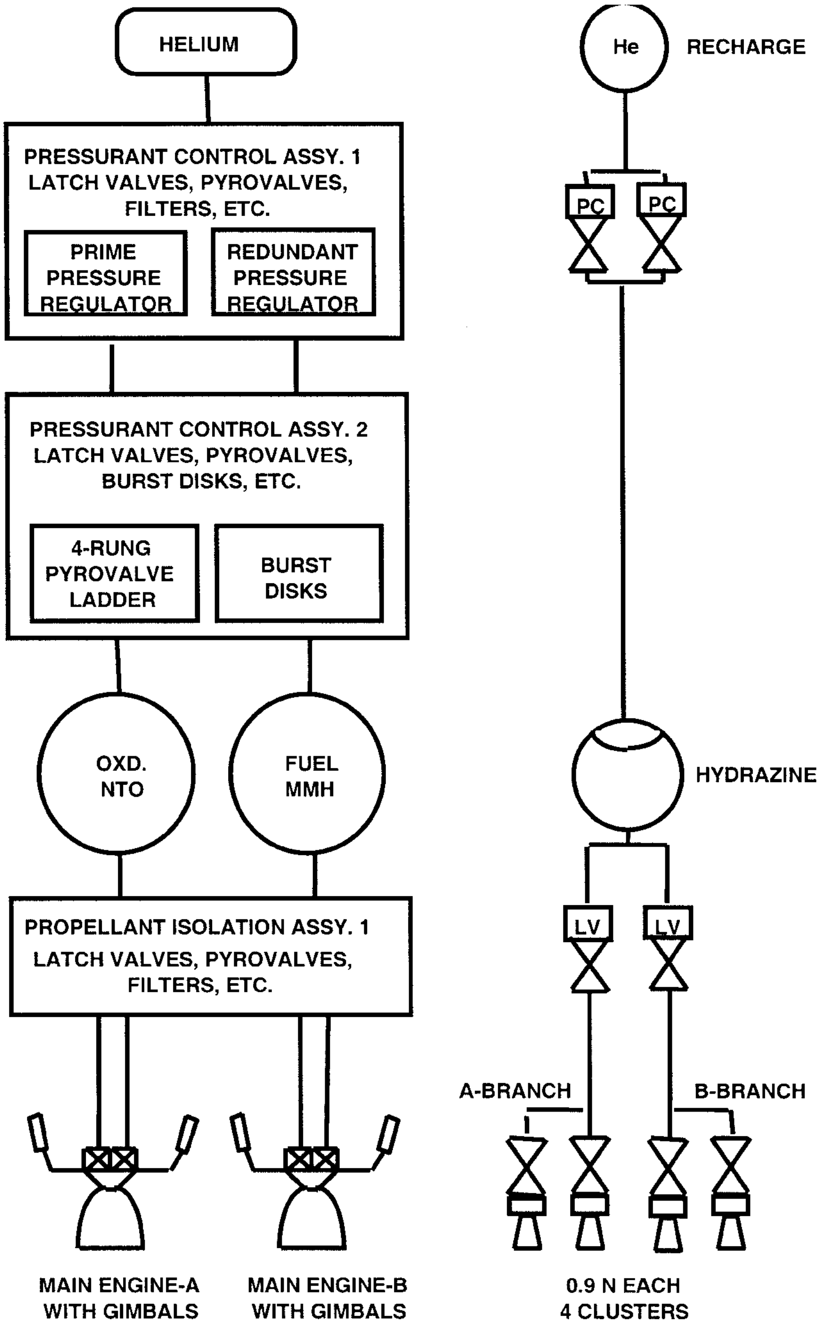

Figure 14. Propulsion Module Subsystem.

TABLE VIII
Pointing and course correction parameters

Parameter	Value
Probe release pointing	30.2 mrad
HGA pointing	4.0 mrad
Probe relay pointing	3.5 mrad
Imaging science pointing	2.0 mrad
Radar pointing	1.3 mrad
Pointing stability (1 second)	8 microrad
Pointing stability (1 hour)	280 microrad

TABLE IX
AACS software modes

Mode	Means of Attitude Determination	Attitude Control
Pause	None	Idle
Coast	IRU	Idle
Detumble	IRU	RCS
Find_Sun	IRU, SSA	RCS
Center_Sun	IRU, SSA	RCS
Find_Stars	IRU, SSA, SRU	RCS
Home_Base, Inertial_RCS	IRU	RCS
Home_Base, Inertial_RWA	IRU	RWA
Home_Base, Cruise_RCS	SRU	RCS
Home_Base, Cruise_RWA	SRU	RWA
ME_DeltaV	IRU	ME/RCS
RCS_DeltaV	IRU	RCS

An important inclusion on the AACS data bus is the valve drive electronics (VDE). This allows the AFC to command the propulsion elements without having to use the spacecraft data bus, thereby improving the response time of the control loop.

Attitude positions and rates are determined using several AACS sensors. Stellar reference units (SRU) provide the primary source of attitude knowledge. Sun sensors (SSA) seed the star identification algorithm for the trackers and furnish an emergency attitude reference. Inertial reference unit gyros (IRU) measure rates during major course correction burns and turns. A single accelerometer (ACC)

TABLE X
AACS sensor parameters

Parameter	Value
SRU field of view	15 deg $\times$ 15 deg
SRU-based attitude knowledge	1 mrad (3 axis, rates $<$ 0.3 deg s^{-1})
Sensitivity	5.6 Mv stars
Stars in catalog (in AFC)	5000
SSA field of view	60 deg $\times$ 60 deg
SSA range	0.6 to 10.06 AU
SSA-based attitude knowledge	26 mrad (2 axis)
IRU type	hemispherical resonating
IRU drift rate	$<$1 deg h^{-1}
IRU drift rate stability	$<$0.06 deg h^{-1} (8 h period)
IRU resolution	0.25 microrad
IRU full performance range	$<$2 deg s^{-1}
IRU degraded performance range	$<$15 deg s^{-1}
ACC range	$>\pm$ 1.12 g
ACC resolution	0.002 g

provides closed loop time out of major burns. Table 10 lists some of the key parameters of the AACS sensors.

Attitude is controlled using either reaction control thrusters (RCS) or reaction wheels (RWA). The thrusters are used for coarse pointing control, the reaction wheels for fine control (2 mrad, 99% radial). There are block redundant branches of eight thrusters pointed in the spacecraft +Z and +Y directions mounted in four clusters. There are three primary and one backup reaction wheels. The backup wheel is mounted on an articulating platform so that it can be positioned to compensate for the loss of any of the three primary wheels. Each wheel is capable of storing greater than 36 Nms of momentum and producing greater than 0.14 Nm of torque under certain conditions.

Table 11 lists some key parameters of the thruster system. The propellant is hydrazine. The feed system is a blowdown type with a single recharge. Helium is used as the pressurant. A membrane is employed within the hydrazine tank to serve as the propellant management device.

For course corrections of magnitude less than 1 m s^{-1} to 5 m s^{-1} the thrusters described above are used. For course corrections greater than 5 m s^{-1} such as orbit injection, one of two redundant main engines is used. These engines burn the bipropellants nitrogen tetroxide and monomethyl hydrazine. Each engine is gimbaled to keep the thrust through the spacecraft center of mass. The propellants

TABLE XI
Thruster system parameters

Parameter	Value
Fuel load at launch	132 kg
Thrust per thruster	1 N (0.2 lbf)
Maximum burn duration	120 min
Thruster cycles	270 000
Throughput per thruster	25 kg
Minimum impulse bit	<0.015 N-s
Fuel tank volume	11 350 in^3
Fuel tank maximum operating pressure	420 psig
Fuel tank pressure at launch	367 psia
Fuel tank pressure at recharge	237 psia
Pressurant tank volume	14 340 in^3
Pressurant tank maximum operating pressure	3720 psig
Recharge tank volume	418 in^3
Recharge tank maximum operating pressure	3600 psig

are stored in two tanks. Each tank features a vane-type propellant management device. The system can be operated in a blowdown or pressure regulated mode. The pressurant is helium and is supplied from a single tank. The pressurant system features pyrovalve ladders to prevent the mixing of fuel and oxidizer vapors. Mixing of propellant vapors is one of several possible causes for the loss of the Mars Observer spacecraft. Table 12 lists some key parameters of the main engine system.

8. Miscellaneous Design Features

Several miscellaneous features of the orbiter are worth noting. Attached to the bottom of the orbiter is a cover for the main engines. This cover is deployable and retractable like a baby carriage cover. It is used to protect the main engines from micrometeoroid strikes. The insides of the engine nozzles have thin thermal coatings that are particularly vulnerable to micrometeoroid damage.

Several approaches are taken to control temperatures on the orbiter. Electrically conducting and grounded multilayer thermal blankets are used extensively. Many of these blankets include a special layer of material to provide protection from micrometeoroids. A number of special paints are used. A particularly challenging painted surface is the HGA that has an allowable temperature range of about −200 to +125 degrees Celsius. Thermostatically controlled louvers are mounted

TABLE XII
Main engine system parameters

Parameter	Value
Total delta-V capability	$>$2360 m s^{-1}
Oxidizer load at launch	1132 kg
Fuel load at launch	1868 kg
Thrust per engine	445 N (100 lbf)
Specific impulse	$>$3016 N-s/kg
Maximum allowable burn duration	170 min
Engine starts	200
Throughput	3000 kg
Pressurant tank volume	14 340 in^3
Pressurant tank maximum operating pressure	3720 psig
Propellant tank volume	84 670 in^3
Propellant tanks maximum operating pressure	330 psig

on the outboard shearplates of many electronics bus bays to control the temperature swings associated with cycling power. Electrical heaters are used in many places. Software algorithms control some of these heaters. Radioisotope heater units, 1 W pellets of plutonium dioxide, are used in other places such as the thruster clusters to limit the amount of electrical power required. Waste heat from the RTG's is channeled onto the exteriors of the main propellant tanks. Finally, there are several surfaces that are used to shade other parts of the spacecraft. The HGA is the most important of these. Whenever the distance from the spacecraft to the Sun is less than 2.7 AU, the HGA is pointed at the Sun to shade the spacecraft. This attitude is maintained except for short periods for trajectory correction maneuvers. The Huygens Probe is pointed at the Sun to shade the spacecraft during maneuvers.

9. Huygens Probe

The Huygens probe weighs 319 kg and is 2.7 m in diameter. During flight to Saturn it receives power from the orbiter for occasional checkouts. On November 6, 2004 it will be pointed at Titan and released by orbiter-initiated pyrotechnics. A spin eject device will spin up the probe to about 7 rpm as it is released. Probe support equipment (PSE) remains onboard the orbiter to receive and translate the radio signals from the Probe.

On January 14, 2005 it is expected that the Probe will enter Titan's atmosphere. A coast timer set by the orbiter just before orbiter/probe separation will awaken the probe avionics about 15 min before encountering the atmosphere. A series

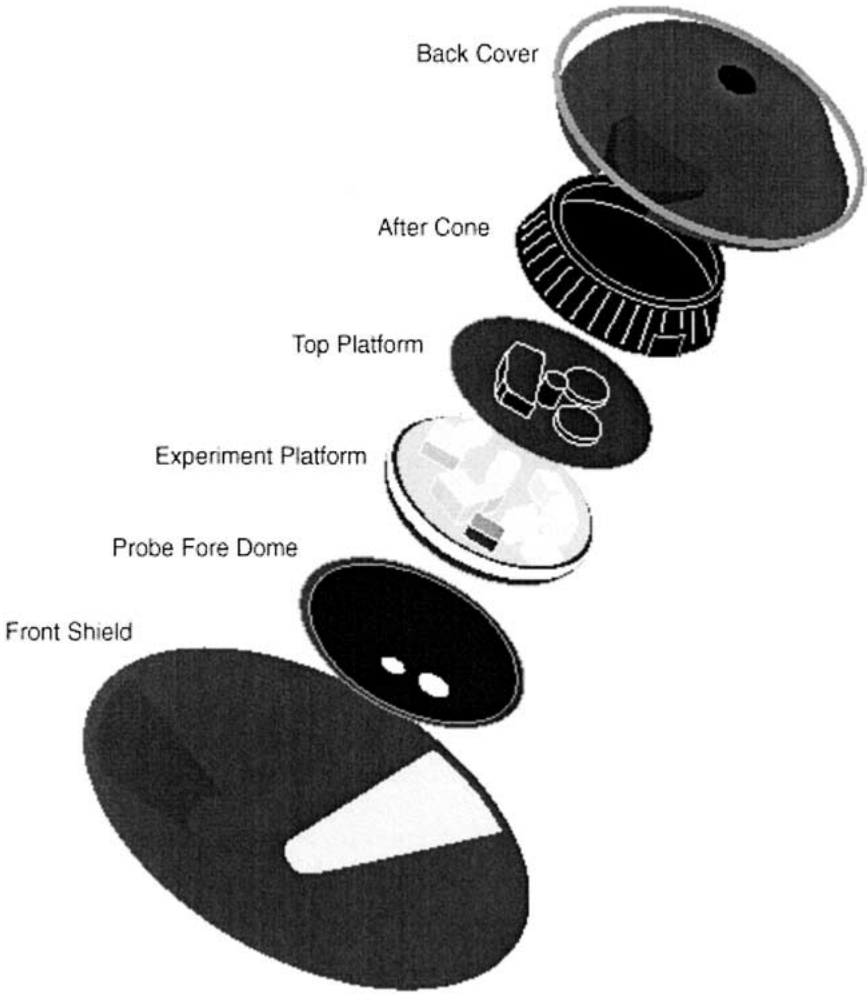

Figure 15. Huygens Probe Structure Assemblies.

of deployments and jettisons of parachutes and engineering assemblies follows eventually culminating with a landing on Titan's surface. The descent is expected to take 20 to 150 min. The probe may survive another 30 min on the surface.

As shown in Figure 15 the probe has six major structural assemblies. The front shield is made of tiles of silica fibers attached to a honeycomb shell with insulating foam sprayed on the back of the shell. The front of the shield ablates to provide thermal protection from the 1 MW m^{-2} flux as the probe decelerates from about Mach 20 to Mach 0.6. The front shield is jettisoned at around 160 km altitude. The fore dome provides thermal protection for the rest of the descent. Spin vanes attached to the fore dome provide a controlled spin rate for camera observations. The experiment platform and top platform are aluminum honeycombs supporting

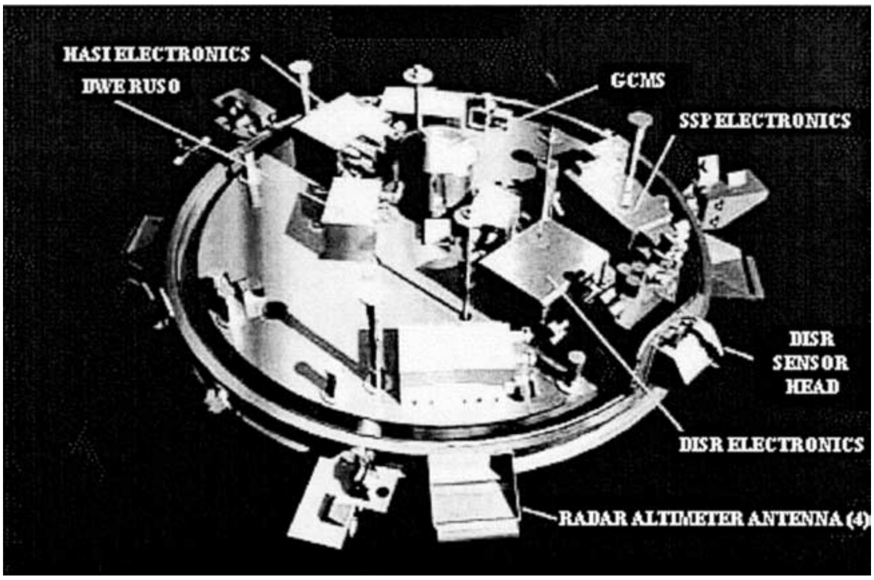

Figure 16. Top of Probe Experiment Platform.

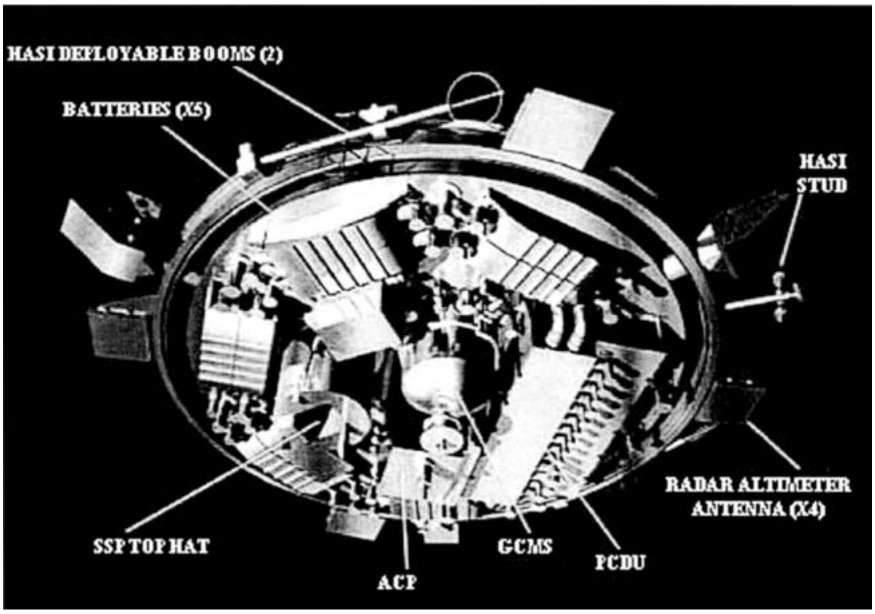

Figure 17. Bottom of Probe Experiment Platform.

the instrument and engineering subsystem packages. The instrument locations on the experiment platform are shown in Figure 16 and Figure 17. The after cone and back cover are aluminum structures completing the package. The back cover is detached by the deployment of the first of three parachutes.

The principal elements of the Descent Control Subsystem are three parachutes. The first or pilot chute is 2.59 m in diameter and pulls the back cover away. The 8.3 m main parachute is deployed as the back cover is pulled away. This chute decelerates the probe to about Mach 0.6. The third drogue parachute is only 3.0 m in diameter so that the probe drops to the surface before running of power.

The probe sequence is controlled by block redundant Command and Data Management Units (CDMUs). Total memory is about 20 kbits. Three redundant timer units and two redundant g-switches are used to trigger sequencing. Software is in the Ada programming language.

Five lithium-sulphur dioxide batteries power the probe avionics. Total capacity is about 1600 W-hr. One of these batteries can fail without compromising the mission. The Power Conditioning and Distribution Unit (PDCU) supplies power at 28 V. The peak load during descent is somewhere 300–400 W. A Pyro Unit provides redundant sets of lines for 13 devices.

The probe has redundant S-band transmitters and antennas. One is left hand circularly polarized, the other right hand. Identical data is transmitted from both transmitters; however, a six second delay is used between transmissions to reduce the chance of data loss due to link gaps. The orbiter HGA receives both signals. Probe support avionics on the orbiter convert the radio signals to digital data. Probe data is redundantly recorded on the orbiter SSRs.

10. Web Sites

Additional information about the Cassini Orbiter and Huygens Probe can be found at the following web sites:
http://www.jpl.nasa.gov/missions/current/cassini.html
http://sci.esa.int/huygens

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