Pergamon

Planet. Space Sci., Vol. 46, No. 9/10, pp. 1245–1256, 1998

0032–0633/98/$—see front matter

PII: S0032–0633(98)00065–8

The microwave sensing in the Cassini Mission: the radar

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Received 24 January 1998; accepted in revised form 15 April 1998

Abstract. The spacecraft of Cassini Mission has been launched towards Saturn on October 1997 to study the physical structure and chemical composition of Saturn as well as all its moons. To this end many instruments have been mounted on the spacecraft; one of these is the Cassini Radar.

Cassini Mission is a cooperative mission between NASA (National Aeronautics and Space Administration), ESA (European Space Agency) and ASI (Agenzia Spaziale Italiana) to acquire scientific knowledge about the Saturnian system. Through many different scientific instruments, the Cassini Radar objective will be to investigate the nature of Titan: its optically opaque atmosphere and surface. Part of the Radar, the Radio Frequency Electronic Subsystem (RFES) has been developed by Alenia Aerospazio while the other parts have been developed from Jet Propulsion Laboratory (JPL).

Cassini Radar is a multimode instrument able to operate in altimeter mode (4.25 MHz), an imaging mode (0.85 and 0.425 MHz bandwidth), a scatterometer mode (0.106 MHz bandwidth), and a radiometer mode (100 MHz bandwidth). These modes will be used to acquire images, topographic profiles, backscatter reflection coefficient, and sense brightness temperatures of the surface of Titan.

A passive mode, i.e. radiometer, has been implemented to measure Titan's surface emissivity in the ku-band.

In the development of such an hardware, designers faced many requirements coming from the "deep space environment" and the specific features of a spacecraft designed to cruise in the Solar System to reach Saturn and its moons: e.g. reduced mass, low available room, low power consumption, severe environmental conditions, specific thermal control and on-ground test accessibility. The structure has been designed to survive high levels of vibrations at high frequencies (pyroshocks). Thermal design has to withstand wide range of temperature. Design avoids the generation of magnetic fields, which could disturb magnetic sensitive sensor. Electronics have been shielded from natural and artificial radiation.

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Introduction

Cassini Radar is one of the science instruments on the Cassini Mission—a joint NASA/ESA/ASI mission to carry out detailed study of Saturn and its many satellites in early 2000s. The Cassini spacecraft has been launched in 1997 and will begin orbiting Saturn in 2004. In order to study the surface properties and processes of the cloud enshrouded Titan, the largest satellite of Saturn; the spacecraft will make a number of close flybys of Titan during its 4-year mission. During these flybys, the Cassini Radar and other instruments onboard the spacecraft will conduct intense observations. The Cassini Radar is designed to operate in four observational modes at spacecraft altitudes below 100,000 km on both inbound and outbound tracks of each hyperbolic flyby of Titan. They include: the imaging mode which provides medium-to-high resolution imaging, the altimeter mode which measures the relative surface elevation of the suborbital tracks, the scatterometer mode which measures Titan's surface backscatter coefficients, and the radiometer mode which measures the surface emissivity of Titan as an adjunct to the active radar measurements throughout the entire pass.

The Cassini Radar Instrument has been developed jointly between NASA/JPL, Italian Space Agency (ASI) and its contractor, Alenia Aerospazio (ALS). The flight instrument consists of four major components: the Radio-Frequency Electronics Subsystem (RFES), the Digital Sub-

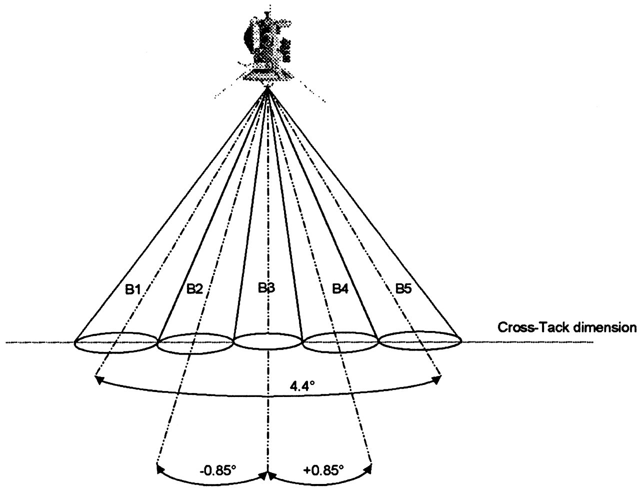

Fig. 1. Antenna beam configuration for the Cassini Radar

system (DSS), the Energy Storage Subsystem (ESS), and the Ku-band antenna (ANT).

Radar mode description

Tradeoffs made between atmospheric absorption, measurement resolutions, and detection sensitivity have led to the selection of 13.78 GHz as the radar operating frequency. Due to the weight and volume constraints, the Cassini Radar will use the spacecraft's high-gain, 4-m telecommunications antenna during radar operation. To extend the imaging coverage, multiple radar feed structure has been mounted on the antenna reflector to generate five antenna beams, which are adjacent to one another in the cross-track dimension. This beam configuration is graphically illustrated in Fig. 1.

The motivation for the Cassini Radar multimode design is to be able to accommodate the potentially different types of surfaces on Titan. Given the uncertainties in ephemeris and radar backscatter, the RADAR system performance must be robust. Since the radar range vary within a single Titan flyby pass as well as from pass to pass, the radar parameters, such as pulse width, bandwidth, receiver gain, pulse repetition frequency, and other timing parameters, must be updated at regular intervals in order to maintain sufficient signal-to-noise ratio on the radar echoes. The key system parameters for each radar mode are listed in Table 1.

Imaging mode

During radar imaging, the spacecraft will rotate to either the left or the right side of the sub-satellite track according to the pre-determined command sequence, and all five radar antenna beams will be utilized one at a time (80–200 ms each) to obtain the maximum possible cross-track swath coverage. The total cross-track swath created by combining the five illuminated sub-swaths ranges from ~ 120 km at spacecraft altitude of 1000 km to ~ 460 km at spacecraft altitude of 4000 km. During a close flyby with $h_0 \leqslant 1250$ km, the Cassini Radar can image up to $\sim 1.1\%$ of Titan's surface.

The image azimuth resolution will be accomplished by unfocused SAR processing of echo bursts. For the burst-timing scheme to be used by the Cassini Radar, the azimuth resolution is estimated to be between 350 m and 720 m throughout the imaging period during each flyby. The image range resolution will be accomplished by range compression of the linear FM chirp signals and is proportional to the signal bandwidth and the angle of incidence. In the current design, an 850-kHz bandwidth will be used when the spacecraft altitude is 1600 km or less, and a 425-kHz bandwidth will be used at spacecraft altitudes between 1600 and 4000 km in order to enhance the SNR. The corresponding range resolution is estimated to be between 480 m and 640 m at $h \leqslant 1600$ km, and between 420 m and 2.7 km at 1600 km $< h \leqslant 4000$ km.

Altimeter mode

This mode will be used to study the relative topographic change of Titan's surface along the sub-satellite track. Operating at spacecraft altitudes between 4000 km and 9000 km, this mode will utilize the central antenna beam (Beam 3 which will now be nadir-pointing by rotating the spacecraft) for transmission and reception of chirp pulse signals at a system bandwidth of 4.25 MHz. The altimetric measurements collected are expected to have horizontal resolutions (pulse-limited radar footprints) ranging from 24 km to 27 km, and vertical resolutions of about 50 m. Given such vertical resolution and the anticipated spa-

Table 1. System parameters for the Cassini Radar modes

Parameters	Imaging	Altimeter	Scatterometer	Radiometer
Frequency (GHz)	13.78	13.78	13.78	13.78
Peak power (W)	63	63	63	N/A
Look angle (deg):				
Cros-track	5–20	0	± 6–± 12	± 6–± 12
Along-track	0	0	± 6–± 12	± 6–± 12
PRF (kHz)	1.8–6.0	4.7–5.6	1.0–3.0	N/A
Pulsewidth (μs)	200–400	150	500	N/A
Bandwidth (kHz)	425,850	4250	106	100,000

cecraft navigation and pointing uncertainties, we estimate the overall relative height measurement accuracy to be of the order of 150 m.

Scatterometer mode

This mode is intended to map the radar reflectivity of the potentially different types of Titan's surfaces at different incidence angles. This mode will operate at altitudes between 9000 km and up to 100,000 km (the exact altitude will be SNR limited and determined after the first flyby) and will require spacecraft manoeuvers to scan the central antenna beam over the entire Titan disk. To ensure sufficient signal detection, a smaller bandwidth of 106 kHz will be used by this mode. Both surface backscatter and noise-only measurements will be collected with this mode so that the surface σ^0 can be estimated accurately. Depending on the range distance, this mode can detect surface backscatter coefficient values as low as -35 dB.

Radiometer mode

The objective of the radiometer mode is to measure Titan's surface emissivity at 13.78 GHz. The data collected will be complementary to those collected by the active radar modes since it provides an additional input into the radar backscatter models. The radiometer mode has a nominal bandwidth of 100 MHz. The radiometer can operate alone or in conjunction with any of the active modes. When operating in conjunction with the scatterometer mode, for example, this mode may use the central beam (Beam 3) for maximum resolution or up to all five beams in order to increase the surface coverage. Cold-space, hot-load and resistive-load calibration procedures are included in the Radiometer mode in order to improve the temperature measurement accuracy. At 100,000 km the central beam footprint is approximately 600 km. Scenario temperatures will range from 2.7°K of the deep space cold reference to the expected 80–120°K.

Radar antenna

Due to the limited spacecraft weight and volume, the Cassini Radar is required to use the spacecraft's high gain 4-m diameter telecommunications, parabolic-reflector antenna for signal transmission and reception. In order to achieve the desired surface coverage during the limited number of Titan flybys, a multiple feed structure at 13.8 GHz is mounted on the antenna reflector to generate five antenna beams, which overlap in the range direction.

The center circular beam (beam 3) is generated by illuminating the entire reflector with a feed located at the reflector's focal position, while the four off-center beams are generated by partially illuminating the reflector with feeds that are offset from the reflector's focal axis by the appropriate amounts. Due to feed symmetries the two outer most beams (1 and 5) and the two inner beams (2 and 4) have identical patterns except that they are rotated 180° with respect to one another. Beams 1 through 5 will illuminate swaths that are progressively further from the satellite's nadir track. Expected parameters for the 5 beams are shown in Table 2.

Table 2. Cassini Radar antenna beams parameters

Parameter	Beams 1 & 5	Beams 2 & 4	Beam 3
Peak Gain (dBi)	43.0	43.1	50.2
Azimuth Beamwidth (deg)	0.3	0.35	0.35
Range Beamwidth (deg)	1.3	1.35	0.35
Angle from Focal Axis (deg)	± 2.20	± 0.85	0.0
Peak Sidelobe (dB)	-15.3	-13.3	-18.1

Radar electronics

Digital storage subsystem

The Digital Storage Subsystem (DSS) is the portion of the Cassini Radar which processes the digital commands from the spacecraft to set the operational modes of the RADAR, collect and buffer the data to and from the RFES, acquire and process the analog telemetry channels, and reformat and buffer the data for pickup and storage by the Cassini spacecraft data recording system. The DSS consists of six parts: the Flight Computer Unit (FCU), the Control and Timing Unit (CTU), the Science Analog to Digital Converter (SADC), the Science Data Buffer (SDB), the Telemetry ADC (TADC), and the Power Converter Unit (PCU).

The FCU handles the interfaces with the spacecraft and the rest of the DSS. The Flight Software runs in the FCU which processes all commands from the spacecraft and operates the RADAR through "instructions" which are loaded for each pass from the spacecraft. The CTU is the timing heart of the RADAR in that it sends signals to all other units including the RFES, which set the RADAR operational states. The FCU, through software sets registers in the CTU, which defines the timing. The CTU consists of a number of Field Programmable Gate Arrays (FPGA) which allow a complex set of functions to operate within the confined space and power limitations of this radar. The SADC receives the analog video radar echo data from the RFES and digitizes it before sending it on to the SDB for short-term storage prior to processing by the FCU. The TADC acquires approximately 80 channels of telemetry data for inclusion with the science data for transfer to the spacecraft. The PCU takes the 30 VDC from the spacecraft power subsystem and puts out regulated voltages to the other units.

Radio frequency electronic subsystem

Radio Frequency Electronic Subsystem (RFES) is composed of seven units whose purpose is to provide active signals (scatterometer, altimeter and imaging) and a passive signal (radiometer) to DSS for data formatting and storage. It is a subsystem, which is able to receive commands from DSS, and is able to give its housekeeping information.

The Power Supply (PS) is the unit which generates the regulated voltage from the main bus of the spacecraft (+30 Vdc). Two converters (half bridge topology) are used to generate/switch-on separately the voltage from master oscillator (+15 V) from the others. Regulators are obtained both through Pulse Width Modulation voltage loop and series techniques. An electronic switch protects main bus from failure propagation. Input (main bus overcurrent and undervoltage) and output (overvoltage, undervoltage and overcurrent on each line) protections have been designed in order to disconnect PS from main bus by opening the input switch driving. Protection can be by-passed if required by an external command.

The Frequency Generator (FG) is the unit, which generates the reference signal for the whole radar (coherence) and the local oscillator signals for all RFES up and down frequency conversion. The unit is manufactured using packaged components on alumina (over 1 GHz) or FR4 fiberglass. Thick-film hybrid technology has been used for size and mass minimization. The Ultra Stable Oscillator (USO) at 10 MHz is placed into a thermally controlled oven (+85°C) in order to maintain a stable environment. Telemetry monitors the temperature to verify frequency accuracy. Two Harmonic Phase Locked Loop Oscillators (HPLLOs) provides 30 MHz and 90 MHz. Two Sampling Phase Locked Loop Oscillators (SPLLOs) provide at 720 MHz and 12960 MHz.

The Digital Chirp Generator (DCG) allows the RADAR to meet the great flexibility requested by the mission design. In fact it is a complete programmable signal generator which is able to synthesize at baseband (10 MHz) pulsed linearly frequency-modulated signals. Their pulsewidth, bandwidth and center frequency can be varied by simply sending command data. It is based on Direct Digital Synthesis (DDS) techniques and uses a Numerically Controlled Oscillator (NCO) as core for the generation of the sinusoidal samples. A phase modulator addresses the reading of correct values. A Digital to Analog Converter (DAC) converts data in an analog signal. The following low pass filter attenuates out the spurious spectral components.

The Chirp Up Converter and Amplifier (CUCA) is the unit which translates in frequency the signal generated at baseband to Ku-band (13.8 GHz) by a three mixing process. The unit is manufactured using packaged components on alumina substrate (Ku-band) or fiberglass PCBs. Use has been made of thick film and surface mount (for passive components) technology for mass and size minimization. In order to compensate the holding effect due to the DAC in the DCG the very last stage of CUCA is operated in a soft saturation region.

The High Power Amplifier (HPA) is the unit, which energizes the radio-frequency signal. It can work both in pulsed or in CW mode. The TWT has a magnesium alloy housing with three depressed collector stages plus a spike. Its gun is a metal ceramic technology with mixed metal matrix cathode and molibdenum grid. The main bus (+35 V) regulator is of boost type with a power push-pull converter. Only one PCB is required to support all high voltages assembled in a single shielded potted module (no wire connections are inside the high voltage section). Main bus power is protected from failure propagation. Input (main bus overcurrent and undervoltage) and output (overvoltage, helix current, overcurrent and modulation of the grid with no transmission mode confirmation) protections have been designed in order to disconnect HPA from main bus.

The Front End Electronic (FEE) is the unit, which routes the signal for the transmission/reception. It consists of a ferrite latching circulator network to route the transmission signal from the HPA to one out of the five antenna ports. On reception a mirror operation is done by addressing echo signals into the receiver of RFES. In calibration mode, the circulators are driven in a way to prevent the input of external signals and letting low loss path only to the internal calibration signals. It is manufactured using waveguide technology.

The Microwave Receiver (MR) is the unit that frequency down converts the signal to baseband from Ku-band. The major functions are: low noise reception (Low Noise Amplifier, LNA) of echo signals, level conditioning, filtering; low noise reception and detection of environmental natural emissions (radiometer); generation of a high stable noise signal to route to the calibration path. Radiometer path passes through from a bandpass filter (100 MHz), which sets the spectral components, a linear detector, which determines the power reading, and an integrator, which smooths the signal. Signals from DSS establish the integration period. A noise diode, with a high Excess Noise Ratio, is placed into MR and is used mainly for the calibration of the radiometer path. The receiver path for the active signals passes through one of four selection SAW filters (117 kHz, 465 kHz, 935 kHz and 4675 kHz are the −3 dB points bandwidths) and a set of step attenuators (1 dB resolution) to set the return signal within the proper dynamic range.

Energy storage subsystem

The Energy Storage Subsystem (ESS) is a battery substitute for the HPA of the RFES and is used to reduce the peak-power required by the RADAR during transmission. The ESS does this by boosting the 30 VDC of the spacecraft to approximately 80 VDC for storage in a large capacitor bank and then bucks the 80 VDC to 35 VDC for use by the HPA. The input to the ESS is limited to 34 W while the output is approximately 200 W. The RADAR transmits at a 75% duty cycle for 10 to 50 msec and then waits for approximately 10 times this time until the next transmission occurs. During this "quiet" time the voltage in the ESS builds back up.

Performance

Evaluation of performance of RFES has been done by using two different approaches. Use of standard instrumentation has been made extensively whereas the test setup complexity did not affect the accuracy of measurements; on the contrary signal data processing has been used (analog signals, properly conditioned, has been sampled at 90 MHz with an external digitizer and processed

Table 3. Performance of main parameters of the transmission chain of Cassini Radar

TRANSMISSION CHAIN				
Parameter	Requirement	0°C	+23°C	+55°C
USO accuracy	< 0.05	0.018 ppm	0.017 ppm	0.023 ppm
Bandwith Accuracy	< 2%	0.62%	0.62%	0.61%
Pulsewidth Accuracy	< 0.5%	0.063%	0.035%	0.023%
Peak Power	> 46.65 dBm	47.52 dBm	47.37 dBm	47.19 dBm
Amplitude Ripple	< 0.8 dBpp	0.34 dBpp	0.32 dBpp	0.36 dBpp
Phase Ripple	< 15 degpp	10.6 degpp	10.6 degpp	11.25 degpp

Table 4. Performance of main parameters of the receiver chain of Cassini Radar

RECEIVING CHAIN				
Parameter	Requirement	0°C	+23°C	+55°C
Gain	> 109 dB	113.1 dB	112.9 dB	110.2 dB
Noise Figure (@ 25 attenuation)	< 5 dB	4.15 dB	3.6 dB	4.9 dB
Amplitude Ripple	< 0.8 dBpp	0.23 dBpp	0.3 dBpp	0.33 dBpp
Phase Ripple	< 12 degpp	2.4 degpp	2.5 degpp	2.4 degpp

off-line). In the following (Table 3 and Table 4) they are reported performance of main parameters obtained during Thermal Vacuum (TV) tests: at ambient temperature (+23°C) and at the extreme of the acceptance range (0°C and +55°C).

In the radar received a dedicated radiometer section has been derived. It is composed of a low noise amplifier (shared from the radar receiver too), a detector diode and integrator circuit. It is able to linearly detect low level signals (from 2.7°K up to 300°K) with programmable integration period (from 5 ms to 75 ms). Radiometer operation foresees four types of modes: Cold Sky Calibration, Hot Load Calibration, Resistive Load Calibration and Temperature measurement. In particular, Cold Sky Calibration is an external calibration, as the antenna spacecraft will be pointed in the outer space, apart from every possible source of interference; to have an absolute reference temperature at 2.7°K. In Table 5, they are shown main parameters of the radiometer chain.

In order to exactly verify performance of the radiometer chain a special test set-up has been studied. A cryogenic source has been taken as absolute reference in order to verify the radiometer capability. Such a device is able to guarantee low temperature with a high degree of stability. From such type of tests it results that the overall measured error is +1.95°K and −1.85°K. Overall inaccuracy, taking into account the set-up inaccuracy and the radiometer sensitivity can be estimated to be: $\leqslant \pm 2.15$°K. In order to have the complete sensitivity of the radiometer the contribution of the digital part has to be added. However, it is not expected a significant degradation with respect to the shown figure. In comparing to the expected scale of temperature variations of components on Titan, achieved sensitivity should allow to gain significant science results and, basing on the fact that the original design of RFES was tailored to radar mode only, it can be considered as a good result from the engineering point of view.

Finally, in Table 6, is shown the summary budget relevant to the instrument.

Hardware qualification

In the following, it is reported the development and solutions that have been carried out to space-qualify the hard-

Table 5. Main parameters of the radiometer mode

RADIOMETER CHAIN				
Parameter	Requirement	0°C	+23°C	+55°C
Bandwidth	> 95 MHz		127.9 MHz	
Noise Figure	< 5 dB	3.2 dB	3.1 dB	3.4 dB
Amplitude Flatness	< 2 dBpp	1.2 dBpp	1.3 dBpp	1.4 dBpp
Gain Stability	< 1.10E-3	6.10E-4	5.10E-4	3.10E-4

Table 6. Cassini Radar summary budget

CASSINI RADAR BUDGET		
Parameter	Requirement	Measurement
Mass	56.73 kg	43.27 kg
Envelope:		
RFES	490.2 mm × 533.4 mm × 237 mm	490.2 mm × 533.4 mm × 237 mm
DSS + ESS	419 mm × 418 mm × 178 mm	419 mm × 418 mm × 178 mm
Power Consumption	120 W	85.4 W

ware. Main attention is dedicated to RFES just because its location on the spacecraft is peculiar with respect to the disposition of the other electronic bays.

In fact, main constraints are due to spacecraft configuration: location outside electronic bus (main body of spacecraft), low available room, reduced mass allocation, low power consumption and on-ground test accessibility when spacecraft fully integrated.

Mechanical design

The RFES is located in a self standing box named "Penthouse", between High Gain Antenna and the Electronic Bus of Cassini spacecraft, as depicted in Fig. 2.

The particular location "produced" several constraints and requirements for mechanical design of RFES assembly:

(a) Penthouse must be attached to support structure only in three points, the apexes of triangle as defined by the "bipod support" of HGA;
(b) the stiffness needed to support electronics and meet dynamic requirement should be provided by "attachment feet" of Penthouse to bipod support, with a small contribution from spacecraft structure;
(c) the room occupied by RFES should be compacted as much as possible;
(d) the Penthouse structure should provide enough shielding against radiation and solid particle penetration (micrometeoroids);
(e) the Penthouse structure should shield the magnetic emissions produced by electronics inside the assembly;
(f) the RFES harness (RF cables + DC wires) the electrical interfaces (DC and RF connectors) with others subsystem of Radar, and the interface with wave guide paths towards High Gain Antenna should be hold by Penthouse;
(g) the Penthouse structure should hold and guarantee the proper operation of devices needed for thermal control purpose;
(h) the location of electronics inside Penthouse should be compatible with location of thermal control devices.

To have the needed stiffness and meet the mass allocation constraints, Alenia Aerospazio adopted an electronic package configuration based upon previous experience in spacecraft design of JPL: the electronic units was designed in such a way to create a "sandwich structure" with structural panels of Penthouse, with this approach the electronic equipment play an active role to share the loads imposed to Penthouse, and to avoid buckling problems.

Mechanical design of RFES assembly took into account also the needs of integration and test activities when the spacecraft is fully integrated. In fact the current design allows the assembly and disassembly of Bay 11 outboard shearplate (below the Penthouse) without removal of Penthouse, to access the electronics of DSS and ESS (Digital and Energy Storage subsystems of Radar). The Penthouse can be installed and removed from its location without any impact to other spacecraft's hardware. The access to test connectors on sidewalls of Penthouse is easy. Furthermore in case of inspection to electronics inside Penthouse after the full assembly of it, the current configuration allows to accomplish it without the need to take apart all the Penthouse.

Thermal control

Due to location of Penthouse outside Electronic Bus of spacecraft, a thermal control system dedicated to RFES was designed.

The scope of this system is to maintain the average temperature of Penthouse's outboard shearplate within the flight temperature limits: 5°C to 50°C, either with RFES in "operating mode" or in "non-operating mode". The average temperature on Penthouse's outboard shearplate is the main boundary condition for thermal design of electronics of RFES. Thermal control system was aimed to control the wide variation in the electronics power dissipation of RFES (57 watt while operating 0 watt when off) and to maintain 5°C as minimum average temperature on outboard shearplate at 10 AU from Earth, near Saturn when temperature of the antenna is ca −200°C. To modulate temperature due to wide variation of RFES power dissipation, a full set of louvres is located on outboard shearplate of Penthouse, this device allows the radiating panel to withstand larger power variations than the panel would be able to handle if it only had passive radiators. This capability is the result of louvres being able to vary the emissivity for the area associated with it.

Due to low power consumption constraints, electrical heaters dedicated to RFES are not available, then to

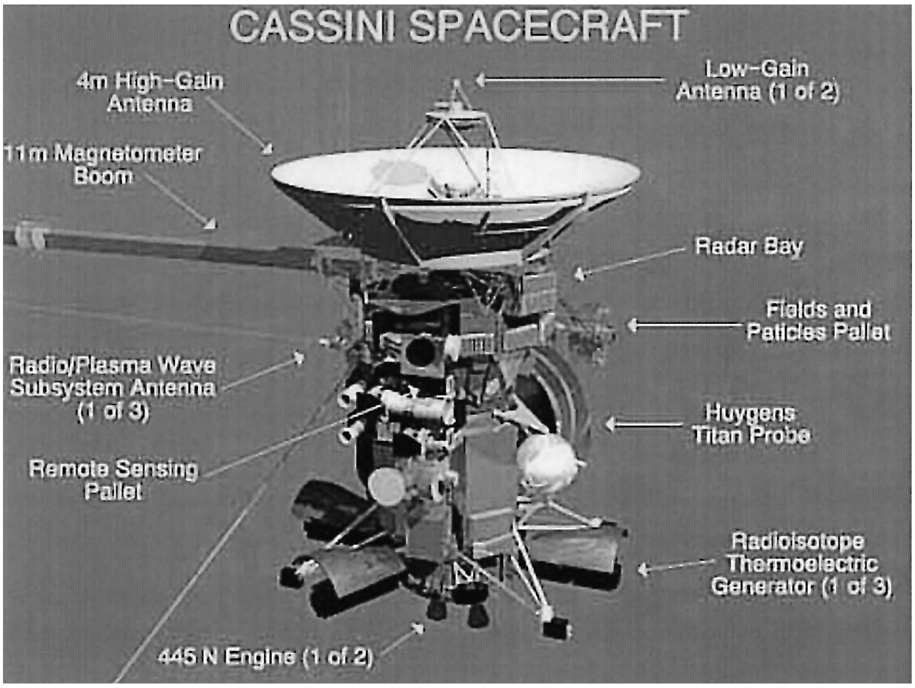

Fig. 2. Cassini Spacecraft configuration

guarantee the minimum temperature two ways were followed: to use heat sources that don't affect the available power for RFES and to reduce as much as possible the heat dissipation.

The provisions adopted to maintain the minimum temperature at 10 AU from Earth is:

(a) heating by 15 watt from RHU's located in five places on Penthouse sidewalls;
(b) upper attachment of Penthouse thermally isolated from High Gain Antenna;
(c) lower attachments of Penthouse conductively coupled to bus upper ring;
(d) back and bottom sides of Penthouse radiatively coupled with bus upper ring and top half of Bay 11 outboard shearplate;
(e) wave-guide flex section included in each of the five wave guide paths from RFES to Antenna;
(f) the whole RFES assembly is covered by a thick thermal blanket;
(g) the back and bottom sides of Penthouse are black painted to enhance the radiation coupling.

RHU is the acronym of Radioisotope Heater Unit, this device provides heat to associated hardware by means of radioactive decay of its plutonium fuel, and it permits to save large amount of heater power that can be utilised to power science instrument.

Enviromental design

The overall aim of Cassini enviromental program was to design and demonstrate during the spacecraft development phase that the flight hardware was environmentally reliable and could function as required throughout the mission's environments. This objective is pursued by satisfying two specific tasks:

1. demonstrate that the design capability has margin over the expected mission environmental requirements;
2. demonstrate that the flight hardware is representative of the qualified design and workmanship integrity to function as required throughout the mission's environments.

Since the RFES is one-of-a-kind hardware, the enviromental tests are the only opportunities before flight to verify acceptable performance under anticipated environmental conditions with appropriate margins. In the next paragraphs the environmental requirements of different space mission are compared.

Vibration test

In the Table 7 are compared levels and duration for sine and random test of electronic units for different types of space missions. The approach to define the sine and random test amplitude is the same for all missions, but for "deep space mission", as shown, it requires larger safety factors than others. The test duration, too, for random tests are 1/3 longer.

Finally, for a "deep space mission", the design is required to be qualified with more severe test amplitude and duration.

Furthermore, the Cassini program required to accomplish the vibration test with unit under test powered even if it is not powered during spacecraft launch, the aim of this requirement is to increase the capability of detecting any failure that could occur to electronics.

For commercial telecommunications satellites in the acceptance test campaign of electronic units, sine test is not usually required.

Thermal design

In Figs 3, 4 and 5 are compared thermal design and test philosophies of: a "deep space mission", a telecommunication satellite and an ESA Earth observation mission.

The largest margins adopted for Cassini mission are used to improve the reliability of electronics assemblies and to increase the probability of mission success in case problems affect thermal control system. This approach is based upon previous missions managed by JPL, where assemblies violated their allowable flight temperature range without impacting the success of the mission.

Thermal/vacuum test

Temperatures of boards and piece-parts of electronic units are significantly lower (about 15–20°C) during atmosphere tests due to convective cooling. Gradients are also reduced due to convection. Therefore, the demonstrated reliability is significantly reduced for an atmospheric test, because of these effects, vacuum test is always preferred.

Cassini program required a single cycle dwell test instead of thermal cycling. Thermal cycling is done only when the assembly is expected to thermal cycle in flight more than 10 cycles over a ΔT equal or greater than 20°C. Much of industry performs thermal cycling as workmanship and defect screen for satellites orbiting around Earth. It's JPL's belief that cycling assemblies may degrade and reduce their life. JPL's experience in "deep space" mission states that workmanship and defect problems are effectively exposed during a dwell test.

Pyroshock

This requirement comes from spacecraft separation shock and from shocks of pyrotechnic events of deployable appendages. The environment created by these shock events consists of very high acceleration level at high frequencies. The electronics located on the spacecraft should proof the capability to withstand to these dynamic loads, because the literature documents several failures caused by pyroshock events.

Each unit passed successfully pyroshock test; this event represents a qualification for mechanical design, materials and processes used in the electronic packaging. In Fig. 6 is reported the pyroshock requirement for qualification test of Cassini compared with that of ENVISAT, the new

Table 7. Levels and duration for sine/random test of electronic units in different types of space missions

	CASSINI Deep space mission		Telecommunication Commercial Satellite		ESA scientific mission ERS-1 and 2	
Test Description	Acceptance	Qualification	Acceptance	Qualification	Acceptance	Qualification
Sine vibration						
Amplitude	⩾95th Percentile	1.5 × FA	⩾95th Percentile	1.45 × FA	⩾95th Percentile	1.45 × FA
Sine sweep	6 Oct/min	2 Oct/min	4 Oct/min	2 Oct/min	4 Oct/min	2 Oct/min
Random vibration						
Amplitude	⩾95th Percentile	FA + 4 dB	⩾95th Percentile	FA + 3.5 dB	⩾95th Percentile	FA + 3.2 dB
Duration	1 minute	3 minutes	1 minute	2 minutes	1 minute	2 minutes

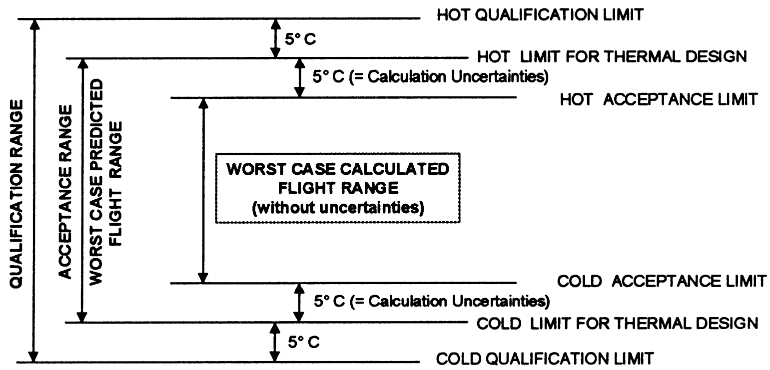

Fig. 3. Thermal design philosophy for TLC commercial satellite

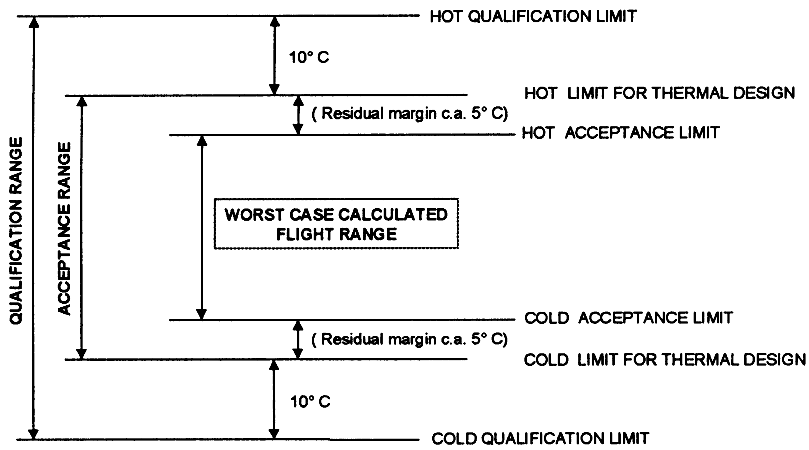

Fig. 4. Thermal design philosophy for ERS – 1 (ESA Earth observation mission)

Earth observation mission of ESA, also in this case the environment specified for CASSINI is more severe.

Radiation

RFES has been designed to withstand ionization effects and displacement damage resulting from the flight radiation environment. The radiation environment for Cassini mission consists of the following contributions:

(a) naturally occurring proton and electron radiation;
(b) neutrons and gamma radiation produced by RTG's (Radioisotope Thermoelectric Generator) the "power source" of Cassini spacecraft;
(c) neutrons and gamma radiation produced by RHU's.

The last two contributions are not present in a mission "around the Earth" because in that case the needed power and heat are generated by solar cells that are not effective beyond Mars.

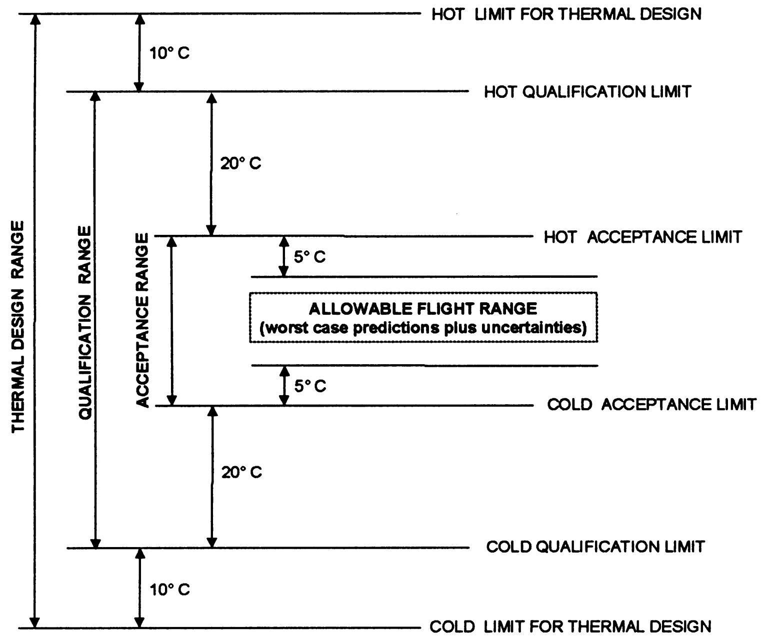

Fig. 5. Thermal design philosophy for CASSINI

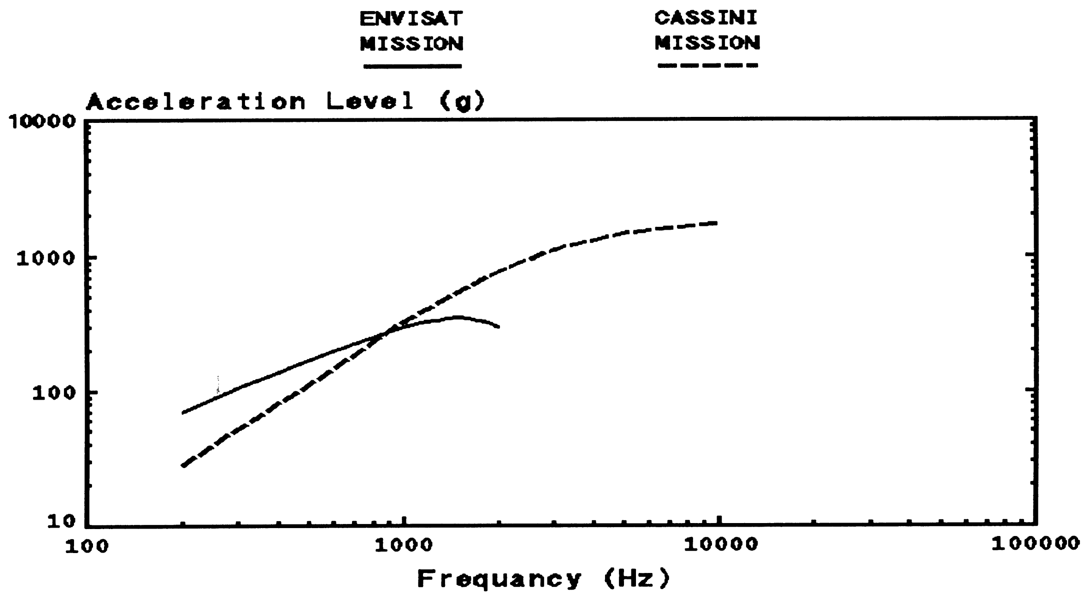

Fig. 6. Pyroshock qualification test

Magnetics

The presence of magnetic sensitive sensors, on Cassini spacecraft, requires RFES's electronics to avoid the generation of magnetic field, which could disturb those sensors.

The RFES Radar Subsystem shall not generate DC magnetic fields in excess of 5 nT maximum radial field at one meter. The RFES magnetic properties are largely affected by magnetic properties of TWTA; to meet the above requirement was necessary to bond three magnets on Penthouse sidewalls.

Solid particle penetration

The flight instruments shall be designed to ensure a 95% probability of mission success in the solid particle environment, accounting for penetration damage.

This environment is composed by three main sources: micrometeoroids, Earth orbiting debris and Saturn ring particles. The requirement affects the thickness of aluminium panels exposed to this environment; thermal blankets properly spaced from the sensitive surface provide an additional protection.

Conclusions

The Cassini mission has been designed to send an orbiter spacecraft to the ringed planet Saturn, and deploy an instrument probe, Huygens, that will descend to the surface of Saturn's moon Titan. The experiment that has been conceived thanks to the radar instrument will bring to scientific community information to study the surface properties and processes of the cloud enshrouded Titan, the largest satellite of Saturn. During its 4-year mission the Cassini Radar will conduct intense observations. They include: the imaging mode which provides medium-to-high resolution imaging, the altimeter mode which measures the relative surface elevation of the suborbital tracks, the scatterometer mode which measures Titan's surface backscatter coefficients, and the radiometer mode which measures the surface emissivity of Titan as an adjunct to the active radar measurements throughout the entire pass.

Alenia Aerospazio has carried out this work under Italian Space Agency (ASI) contract and JPL has been supported under a contract with the National Aeronautics and Space Administration.

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