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INSTITUTO DE PESQUISAS ESPACIAIS

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TÍTULO

SPACECRAFT SPECIFICATION

CÓDIGO O.T.

AC-E-100

GERENCIADO EM
CONFIGURAÇÃO
A PARTIR DE: 02/02/87
Emanuel Fernandes

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iii	3	03/08/87	conforme pedido nº PMS-19	
iv	01	09/06/86		
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02	01	09/06/86		
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27	01	09/06/86		
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iii	3	18/08/87		
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A3	1	27/01/87		

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A7	2	14/09/87		
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A31	1	27/01/87		
A32	1	27/01/87		
A33	1	27/01/87		

CONTENTS

1. INTRODUCTION AND SCOPE

This document specifies the performance, design and construction requirements of the MECB - S1 Data Collecting Satellite at system level, and furnishes the baseline for the establishment of the performance, design and construction requirements of all satellite subsystems.

2. APPLICABLE AND REFERENCE DOCUMENTS

2.1 - SPECIFICATION TREE

The specification architecture of the S1 spacecraft is indicated in the specification tree of Figure 2.1.1.

In case of conflict between the requirements of different documents the document with higher hierarchy shall prevail. The level of hierarchy is defined by the order of appearance in the specification tree.

2.2 - APPLICABLE DOCUMENTS

- AD1 - Spacecraft to Launch Vehicle Interface Definition A-EIF-0001
- AD2 - Spacecraft to Ground Interface Definition A-EIF-0002.

In case of conflict between the requirements of this specification and those of AD1 and AD2, those shall prevail.

- AD3 - Environmental Specification A-EAB-0001
- AD4 - Design and Construction Specification A-ERC-0001
- AD5 - Electromagnetic Compatibility Specification A-ECE-0001
- AD6 - Satellite Product Assurance Plan A-GQL-0006
- AD7 - Product Assurance for Subcontractors A-GQL-0007

In case of conflict between the requirements of the AD3, AD4 and AD5 and this specification, this specification shall prevail.

2.3 - REFERENCE DOCUMENTS

- Data Collecting Satellite System Specification A-ETC-0001
- Plano de Desenvolvimento e Teste dos Satélites de Coleta de Dados A-GRC-0010.

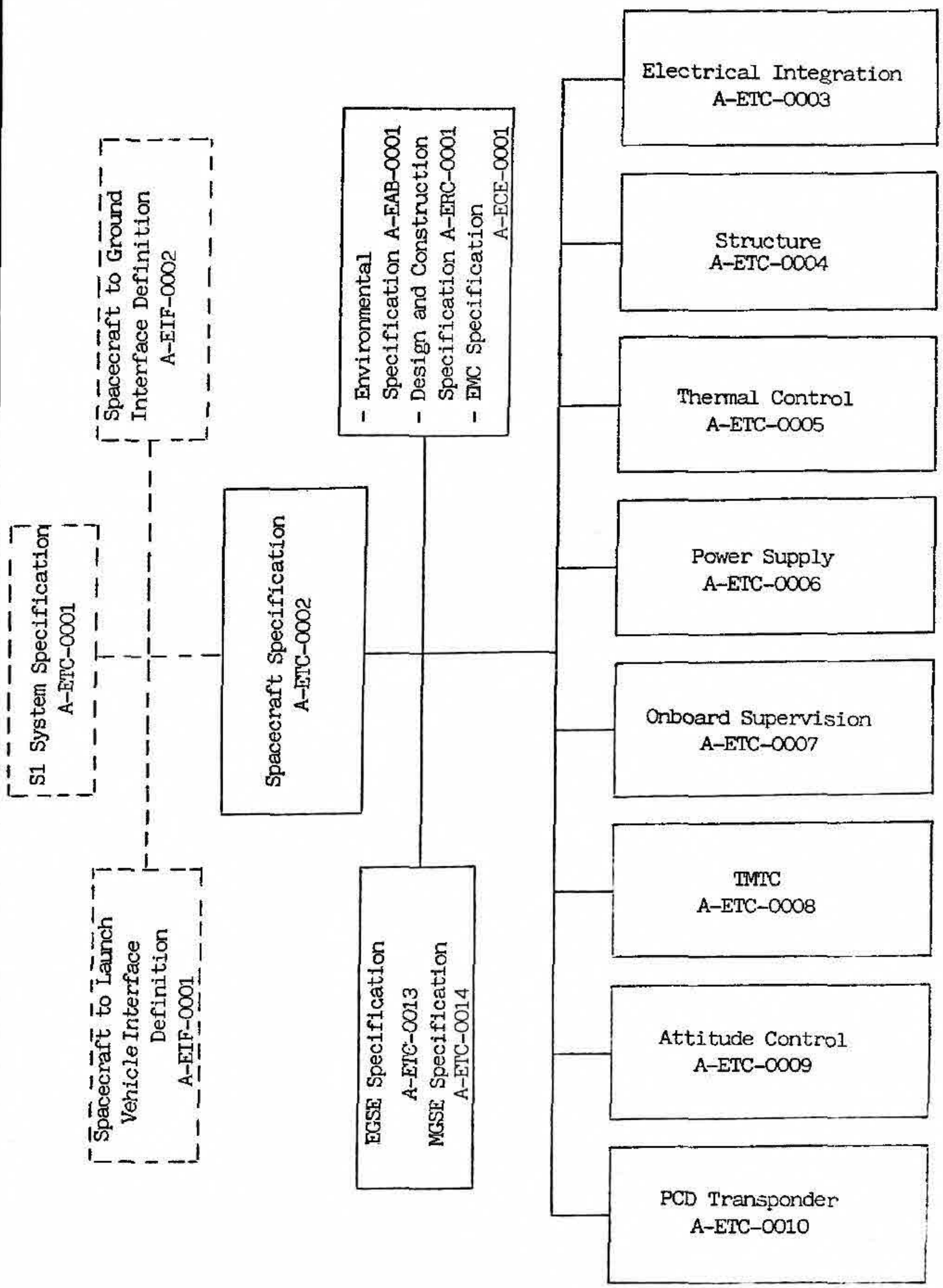

Fig. 2.1.1 - S1 Spacecraft Specification Tree

3. REQUIREMENTS

3.1 - SYSTEM DEFINITION

The general architecture of the Data Collecting Satellite System is defined in A-ETC-0001. The present specification concerns itself with the spacecraft containing the payload composed of a Data Collecting Platform (PCD) transponder.

3.1.1 - MISSION OF THE SPACECRAFT

The MECB Program's S1 spacecraft is an experimental low Earth orbit satellite for real time reception and retransmission of data on the environment. The satellite will fly in a nominally circular orbit inclined about 25^{o} with respect to the Earth's equatorial plane, at an average altitude near750km. During the passes visible from the Cuiabá tracking station, any Data Collecting Platform (PCD) within the coverage angle of the satellite payload antennas will have its UHF signal relayed by the satellite, in S-band. Up to 500 PCDs can be set up to operate in the experiment, at arbitrary locations over Brazil.

The spacecraft will be equipped with an onboard supervision subsystem which comprises a computer and a telemetry encoder. The computer will acquire, process and store data during so-called "experimental" operating modes, and can substitute for the real time telemetry encoder as well as distribute telecommands that are not to be executed in real time. It is part of the S1 mission, to flight qualify the computer.

The attitude stabilization of the spacecraft is achieved through the use of the rotation imparted to the last stage-spacecraft assembly. The attitude control subsystem is equipped with a magnetic torque coil to effect emergency and end-of-life manoeuvers. It is part of the S1 mission to flight qualify the manoeuver procedures.

The spacecraft will transmit house-keeping data in addition to receiving telecommands such as to validate the design, integration and test methods developed and to operate the ground segment.

3.1.2 - SYSTEM DESCRIPTION

The satellite will be spin stabilized by the rotation imparted by the upper stages of the launcher prior to separation. Thus the satellite will maintain an inertial attitude during the flight; the attitude with respect to any tracking station will be continuously changing. In order to allow for the effect of this varying attitude on communication with the ground, the satellite will carrry antennas for its housekeeping and payload communications on its top as well as on its bottom panels (top and bottom refer to the launch configuration). This antenna arrangement shall provide nearly omnidirectional radioelectric coverage for the housekeeping or service telecommunications and for the payload communications downlink. The UHF receiving antennas (payload uplink) shall provide at least 50% coverage for the data collecting experiment.

To fulfill the mission objectives the spacecraft is composed of the following seven subsystems together with the payload:

1. Structure
2. Thermal Control
3. Power Supply
4. Attitude Control
5. Onboard Supervision
6. TMTC
7. Electrical Integration.

The shape of the spacecraft mechanical structure is a right octogonal prism whose base fits within a 1m diameter circle. The main structural element is the central cylinder which supports the horizontal panels. Lateral panels covered with solar cells are attached to the horizontal panels.

The thermal control of the spacecraft is achieved using only passive means. Since all satellite surfaces but one are covered with solar cells, the ways to obtain thermal control are: selective painting of the interior surfaces and electronic boxes, use of insulation blankets, disposal of the excess heat through the bottom panel, and control of conduction heat paths.

The electronic equipments in the spacecraft are connected as shown in Figure 3.1.2.1, and have the following features:

The TMTC subsystem is responsible for the housekeeping telemetry, telecommand and ranging. It provides ESA standard S-band telecommunication between the ground segment and the satellite. The spacecraft command is performed through the cross-strapped systems in which the two receiver outputs drive redundant decoders. Telemetry information is provided by a Pulse Code Modulation (PCM) system. Redundant TM encoders modulate either of two telemetry transmitters via a cross-strap switch. The transmitter outputs are fed to two communication antennas located on the top and bottom panels of the spacecraft, thus forming a quasi-omnidirectional radio frequency coverage, so that communications are assured in all phases throughout the satellite life. The TMTC subsystem employs the verified S-band concept where a single RF carrier is used in each direction of transmission. The up-link and down-link carrier frequencies are related by the exact ratio 221/240 and the latter carrier is generated from the former by means of a coherent transponder which relays to ground the received signal modulated by the ranging tones thus allowing the determination of the distance from the satellite to the tracking station.

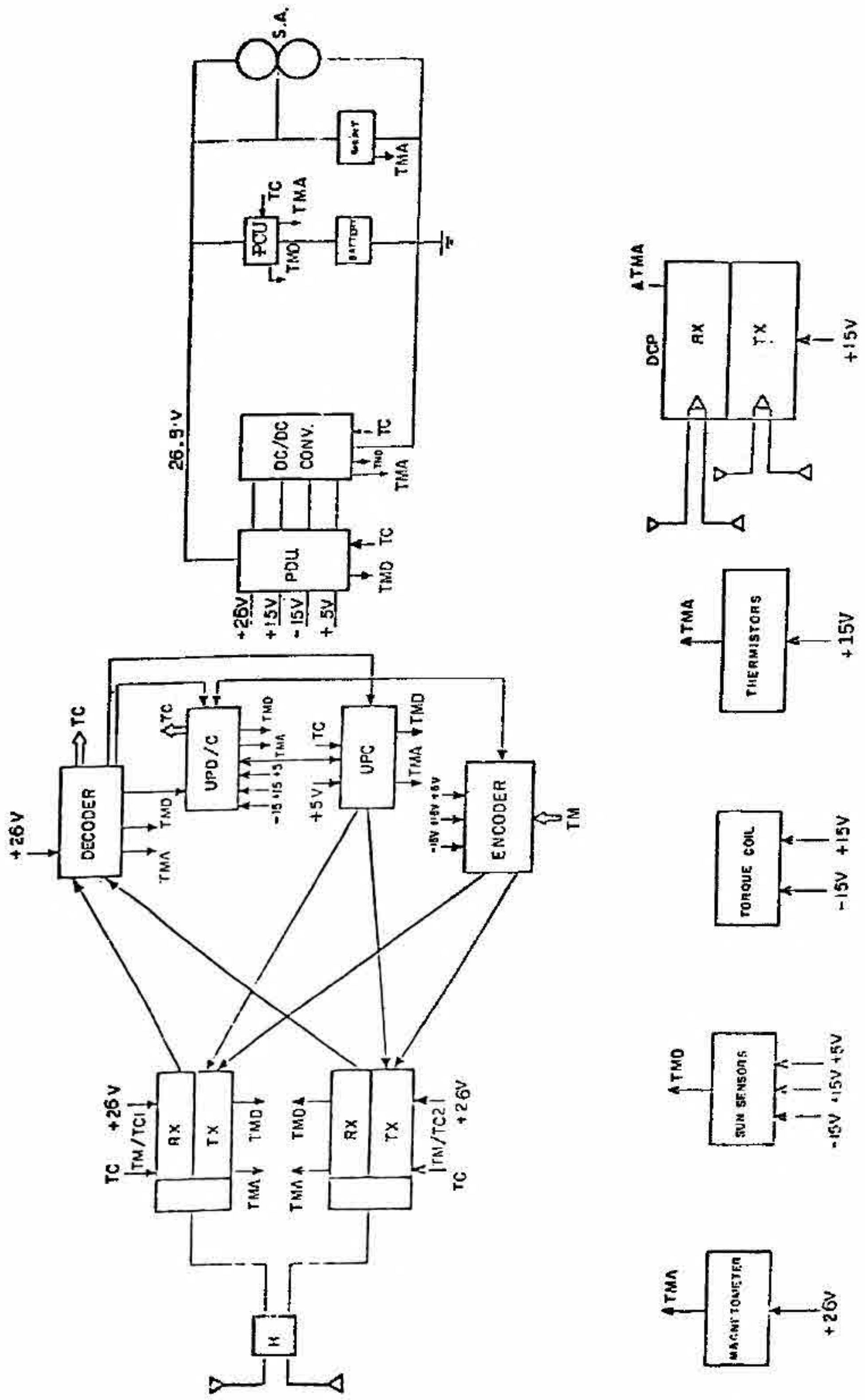

Figure 3.1.2.1 - Electrical Block Diagram

The sunlight illuminated spacecraft receives primary electrical power from its solar array. The launch window is chosen such that the solar aspect angle is always between 0^{o} and 90^{o}. A nickel-cadmium 7Ah battery supplies secondary electrical power to the spacecraft via a discharge controller. In order to increase the battery life use is made of a charge regulator that limits the charge current of the battery and allows charge only to 1/7 of its capacity. DC/DC converters will transform the main bus voltage into the secondary regulated bus voltages. A power distribution unit routes electrical power to the subsystems under telecommand action.

Sun and magnetic sensors are included for attitude determination. Magnetic and optical data are utilized on ground to estimate the satellite attitude. The attitude control subsystem is equipped with a magnetic torque coil for use during emergency and end-of-life manoeuvers. This torque coil, when energized produces a magnetic torque which, under the action of the earth magnetic field, causes a torque that changes the satellite attitude.

The onboard supervision subsystem comprises a computer and a telemetry encoder. Its purpose is to acquire, process and store data from the various subsystems during the experimental stand-by mode, for later transmission to the tracking station. The computer can substitute for the real time telemetry encoder, and can also distribute telecommands that are not to be executed in real time.

In the block diagram of Figure 3.1.2.1 the telemetry and command lines are indicated. All telemetry signals are routed to the TM encoder which connects these data, at the same time, to the computer's UPD/C. A previously selected transmitter will transmit the telemetry signal coming from the encoder or the computer, as required. The two receivers are permanently turned on and both receive signals from the two antennas. The received command sub-carrier is demodulated and decoded in the TC decoder which also distributes the direct command signals. The TC decoder feeds a control signal to the computer to load its memory and to control its telemetry and command capabilities. The DCP payload subsystem works with its own UHF receive and S-band transmit antennas.

3.1.3 - MISSION PHASES

The S1 spacecraft will be launched by the IAE's VLS from Alcântara, Maranhão. The complete mission is divided in six phases:

3.1.3.1 - COUNT-DOWN PHASE

This phase starts five days before the scheduled launch time, and corresponds, in the launch chronology to the closing of the shroud. During this phase the spacecraft will be powered and will undergo system tests via umbilical connector. At lift-off the umbilical providing external power is disconnected and transfer is made to internal battery power. At this moment only the receivers and the telecommand decoder and the telemetry encoder will be powered. The count-down phase ends at lift-off.

3.1.3.2- LAUNCH PHASE

This phase lasts approximately 11 minutes and covers the time period from lift-off until separation. The satellite receivers, telemetry encoder and TC decoder will remain powered. The aerodynamic fairing is jettisoned at about 100 seconds from lift-off at an altitude of 95 Km when the spacecraft begins to receiver solar power. The satellite/launch vehicle separation indication is provided by the action of a device located at the launcher interface, which turns on the spacecraft telemetry transmitter. A telecommand shall be provided to override the action of the separation device in case it malfunctions.

3.1.3.3- ACQUISITION PHASE

This phase is divided in two sub-phases:

a) Orbit Acquisition Sub-phase

During this sub-phase the orbit of the satellite will be determined such that when it comes into visibility of the Cuiaba tracking station after the sixth orbit, the station antenna can be pointed to an accuracy better than $\pm 0.5^{o}$ in azimuth and elevation. This sub-phase will last for the complete duration of the first seven consecutive orbits and will end when the satellite enters the 5^{o} elevation visibility circle of the Cuiabá tracking station. The receivers, the TC decoder, the telemetry transmitter and the telemetry encoder will be powered for the whole duration of the orbit acquisition sub-phase. The satellite will provide range and range rate data when operated accordingly by the network of tracking stations which will support the orbit acquisition sub-phase.

b) Attitude Acquisition Sub-phase

During this sub-phase the satellite attitude will be determined based on its magnetic and solar sensors data telemetered to the Cuiabá tracking station. The sub-phase starts when the satellite enters the 5^{o} elevation angle visibility circle of the Cuiabá station in the eighth orbit when its sensors are telecommanded on, and ends when the satellite exits the 5^{o} elevation angle visibility circle of the Cuiabá station at the end of the second complete set of 8 passes visible from Cuiabá, approximately 49,5 hours after lift-off. The launch window shall be such as to maximize the number of sunlit satellite passes during the atitude determination sub-phase, to allow useful sun sensor data. During the atittude determination sub-phase the satellite telemetry transmitter, telemetry encoder and sensors are telecommanded off whenever the satellite exits the visibility circle of the Cuiabá station. The use of range and range rate data shall be made only when the satellite is operated accordingly by the Cuiabá station.

3.1.3.4- ACCEPTANCE PHASE

The acceptance phase starts immediately after the attitude acquisition sub-phase and lasts one month. During this phase the initial operation verification and the in-orbit tests are effected. The relevant satellite equipments will be telecommánded on and off by the Cuiabá station whenever the satellite enters or exists the station's visibility circle, respectively. This phase is completed when it becomes possible to make a flight verification of the quality of PCD data provided by the mission center. The onboard computer and data collecting payload will be operated for the first time during the acceptance phase.

3.1.3.5- ROUTINE PHASE

The routine phase starts immediately after the acceptance phase and lasts eleven months. During this phase the satellite will fulfil its mission objectives. The Cuiabá station will provide the telecommands for the required operating mode changes.

This phase includes an attitude change sub-phase to be accomplished whenever the satellite attitude is unfavorable to the thermal control and the power supply subsystems.

3.1.3.6- ATTITUDE CHANGE SUB-PHASE

During this sub-phase the satellite magnetic torque coil will be energized to create the necessary torque to change the satellite spin axis attitude. The torque coil will remain energized for a time corresponding to several orbits, as determined by the control center. When the manoeuver is complete, one set of passes will be utilized for determining the satellite attitude. During this sub-phase telemetries and telecommands will be through the Cuiabá station only.

3.1.4- OPERATING MODES

At any instant in time the power status of the satellite equipments defines its operating mode. For each mission phase or sub-phase there are a number of possible operating modes of the satellite as shown in Table 3.1.4.1, together with the function of each mode.

Table 3.1.4.2 describes the powered equipments during each operating mode.

TABLE 3.1.4.1

MISSION PHASES AND RELATED OPERATING MODES

PHASE	SUB-PHASE	MODES	MODE FUNCTION
Launch	-	Launch	Avoid sending telecommands for powering vital equipments and guarantee the reception of commands to change the operating mode
Acquisition	Orbit acquisition	Orbit determination	Determine initial satellite orbit parameters
		Stand by	Stand-by in case of malfunction
	Attitude acquisition	Attitude determination	Determine initial satellite attitude parameters
		Manoeuver	Change of attitude in emergency
		Manoeuver stand-by	Stand-by of manoeuver mode
		Stand by	Stand-by when out of visibility or in malfunction
Acceptance	-	Normal operation	Initial operation verification
		Stand by	Stand-by when out of visibility or in malfunction
		Experimental operation	In-orbit tests of the onboard computer
		Experimental stand-by	Stand-by when out of visibility in the experimental mode
Routine	-	Normal operation	Fulfilment of the data collecting mission objectives
		Stand by	Stand-by when out of visibility or in malfunction
		Experimental operation	Long term test of the onboard computer or malfunction of direct encoder
		Experimental stand-by	Stand-by when out of visibility in the experimental mode
	Attitude change	Manoeuver	Change of attitude after nominal life
		Attitude determination	Attitude determination after or between manoeuvers
		Manoeuver stand-by	Manoeuver when out of visibility
		Stand by	Stand-by in case of malfunction

TABLE 3.1.4.2

MODE	RC	TC	TX	SENSORS	TM ENC	PAYLOAD	COMPUTER	COIL
Launch	x	x	–	–	x	–	–	–
Stand-by	x	x	–	–	–	–	–	–
Orbit Determination	x	x	x	–	x	–	–	–
Attitude Determination	x	x	x	x	x	–	–	–
Normal Operation	x	x	x	x	x	x	–	–
Experimental Operation	x	x	x	x	–	x	x	–
Manoeuver	x	x	x	x	x	–	–	x
Experimental Stand-by	x	x	–	A.N.	–	–	x	–
Manoeuver Stand-by	x	x	–	–	–	–	–	x
Degraded	x	x	TBD	TBD	TBD	TBD	TBD	TBD

OPERATING MODES

3.1.4.1 - LAUNCH MODE

The launch mode will be used during launch and is characterized by having the TM receivers, the TC decoder and the TM encoder powered. This mode is set on ground during the count-down phase.

3.1.4.2 - STAND BY MODE

This mode shall be employed in the following cases:

a - As stand-by mode of the normal operation of the satellite when it is not in visibility of the Cuiabá tracking station.

b - As a transition mode when changing the operating mode of the satellite

The mode is characterized by the satellite having only the TMTC receivers and the TC decoder powered.

3.1.4.3 - ORBIT DETERMINATION MODE

This mode shall be utilized only during the orbit acquisition sub-phase. It is characterized by the active power state of telemetry encoder, transmitter, and temperature sensors, together with the TMTC receivers and the TC decoder.

3.1.4.4 - ATTITUDE DETERMINATION MODE

The satellite shall be commanded to this mode during the attitude acquisition sub-phase and whenever an attitude redefinition is necessary. Transition to this operating mode shall be done only through the Cuiabá tracking station. The powered equipments during this operating mode are the same as in the orbit determination mode with the inclusion of the solar and magnetic sensors.

3.1.4.5 - NORMAL OPERATION MODE

This is the operating mode in which the satellite will accomplish its data collecting mission. The mode will be used only when the satellite is in visibility of the Cuiabá tracking station. It is characterized by the active power state of the same equipments as in the attitude determination mode, with the inclusion of the data collecting transponder payload. This operating mode shall be utilized for the first time during the acceptance phase.

3.1.4.6 - EXPERIMENTAL OPERATION MODE

The experimental operation mode shall substitute for the normal operation mode in case the telemetry encoder malfunctions, or it shall be utilized to test the onboard computer. Operation in this mode shall be only when in visibility of the Cuiabá tracking station, and all the satellite equipments but the TM encoder and torque coil are powered.

3.1.4.7 - MANOEUVER MODE

This operating mode is to be used emergencialy during the attitude acquisition sub-phase or to reposition the satellite attitude during the attitude change sub-phase. The same equipments as in the attitude determination mode are active, together with the torque coil. This mode shall be commanded only through the Cuiabá tracking station, and will be active only during the visible passes.

3.1.4.8 - EXPERIMENTAL STAND-BY MODE

This mode corresponds to the stand-by of the experimental operation mode, when the satellite is out of visibility of the Cuiabá tracking station. During the experimental stand-by mode the telemetry transmitter and the payload are turned off and the sensors and thermistors are commanded on or off by the onboard computer, as needed.

3.1.4.9 - MANOEUVER STAND-BY MODE

This mode corresponds to the stand-by of the manoeuver mode, when the satellite is out of visibility of the Cuiabá tracking station. The equipments powered during this operating mode are the same as during the survie mode, with the addition of the torque coil.

3.1.4.10- DEGRADED MODE

A degraded mode is foreseen for emergency situations and the equipment power status during this operating mode is to be determined by proper analysis of the emergency developed.

3.1.5 - ORBIT DEFINITION

The MECB S1 spacecraft will be launched by the IAE's launch vehicle VLS from the Alcântara, Maranhão (longitude $=-44.2^{\circ}$, Latitude $=-2.4^{\circ}$, altitude =60m) launch base CLA.

The calculated satellite injection point is:

- Latitude (O) $O = 5.202^{\circ}$
- Longitude (λ) $\lambda = -28.558^{\circ}$.

and its orbit definition shall take into account the following constraints:

a) The injection point will have an altitude of 750km ± 50km.

b) The injection angle error in the plane of the orbit will have an uncertainty of $\pm 2.5^{\circ}$.

c) The orbit inclination will be $25^{\circ} \pm 1^{\circ}$.

d) The orbital lifetime under the worst possible solar condition will be greater than 1 year.

The above constraints are summarized in Table 3.1.5.1 which indicates the nominal and the maximaly dispersed orbital elements.

TABLE 3.1.5.1

SUMMARY OF ORBITAL ELEMENTS

	Nominal Value	Value with dispersion
Inclination (i)	25^{o}	$\pm 1^{o}$
Semi-major axis (a)	7128Km	$\pm$ 200Km
Eccentricity (e)	0	<.0424
Geographic ascending node (λ)	-39.8^{o}	0^{o}
Argument of perigree (ω)	12.27	NA
Mean anomaly (μ)	0	NA
Eccentric anomaly (E)	0	NA
True anomaly (u)	0	NA

The spacecraft shall have no orbit control capability.

For calculation purposes the following limits shall be considered:

Apogee : 700Km to 1100Km Perigee: 400Km to 800Km

3.2 - SYSTEM REQUIREMENTS

3.2.1 - SATELLITE ATTITUDE STABILIZATION AND DETERMINATION

The sapcecraft shall be stabilized by the launcher system during the phases from launch through spacecraft separation from the launch vehicle. The spacecraft shall be stabilized by its own passive means for all subsequent mission phases. A limited attitude manoeuvering capability shall be provided through an onboard torque coil, but the system design shall be such as to minimize the probability of attitude control manoeuver being required during the design life of the satellite.

The spacecraft attitude shall be spin stabilized through the rotation imparted by the launch vehicle. Immediately after injection the spacecraft spin rate shall be 160 ± 20 rpm and at the end of the satellite lifetime, the spin rate shall not be lower than 20rpm.

The satellite attitude is defined as the orientation of its spin axis (z-axis) in accordance with the satellite coordinate system described in A-EAB-0001.

The satellite shall be provided with an attitude control subsystem with the following functions:

. To obtain a predictable and defined attitude behaviour in order to allow proper planning for satellite use and normal thermal operation during the satellite nominal life.

. To provide the sensor signals which will be transmitted to ground for the satellite attitude determination.

. To conduct experiments on satellite attitude control through the use of a magnetic torque coil for the spin axis orientation.

In order to attain the above functions, the following requirements shall be met:

a) the spacecraft shall be equipped with a nutation damper which shall be able to produce a nutation cone angle decay from 5^{o} to 1^{o} in less than 5 minutes;

b) the spacecraft shall be equipped with magnetometer and sun sensors to allow its attitude determination through data telemetered to the ground. The accuracy of the satellite attitude prediction shall be better than 1^{o};

c) the spacecraft shall be equipped with a torque coil capable of providing a positive or a negative magnetic moment along the spacecraft z-axis.

3.2.2 - SATELLITE ATTITUDE DURING LIFE

The satellite attitude during its nominal lifetime shall be such that the solar aspect angle with respect to its spin axis shall not be greater than 90^{o}. This attitude behavior shall be attained by the choice of the launch time, by the constraints imposed on the satellites residual magnetic moment, and by manoeuver through an onboard torque coil.

The torque coil dimensioning shall be such as to allow an attitude manoeuver of at least $\pm 4^{o}$/day in declination or right ascention of the spin axis. The torque coil actuation time and duration shall be determined prior to the manoeuver based on the desired resultant satellite attitude.

3.2.3 - PAYLOAD RELATED REQUIREMENTS

The spacecraft shall include a communication payload subsystem to relay to earth the PCD data transmitted from the ground.

The subsystem shall perform the following functions:

a) Receive signals from PCD's within the frequency band of 401.635 MHz ± 30 kHz.

b) Down-convert each PCD signal into frequency of 95 ± 30 kHz.

c) Phase modulate one carrier with the converted 95 kHz band signals, mantaining the total modulation index constant.

d) Transmit the modulated signal in the 2267.52 MHz band to the ground station

The payload shall be capable of simultaneous operation with the service telecommunication subsystem.

The payload shall operate only during the satellite passes which are visible from the Cuiabá data receiving station.

The payload interface and performance requirements shall be as in A-EIF-0002.

3.2.4 - SERVICE TELECOMMUNICATIONS REQUIREMENTS

The spacecraft shall include service telecommunications (TMTC) and onboard supervision (OBS) subsystems to provide telemetry, command and ranging functions during all mission phases. The ESA standard for these communications shall be adopted.

All active elements performing the real time TM, TC and ranging functions shall be redundant. A failure of the onboard computer shall not cause loss of real time TM and TC functions.

Telemetry and telecommand functions shall be provided during spacecraft assembly, integration and test, ground check-out and during launch preparation via hard-line connection.

3.2.4.1 - TELEMETRY REQUIREMENTS

The following spacecraft information shall be available on the ground by selection of appropriate telemetry parameters:

- verification of command execution
- identification of the spacecraft operating mode
- data of onboard sensors
- house-keeping data necessary to ascertion the spacecraft operation
- data for failure analysis for remedial actions from ground.

The acquisition of telemetry data shall be done in accordance with the interface requirements of A-ERC-0001 for the four types of telemetries to be provided:

- analog(AN)
- digital bi-level (BL)
- digital serial (DS)
- thermistors (TH).

The OBS and TMTC subsystems shall be capable of processing as a minimum 64 AN, 64 BL, 1 DS and 32 TH telemetries.

The telemetry data format shall be suitable for the transmission of all TM data and shall include spacecraft identification, in accordance with the spacecraft to ground interface definition A-EIF-0002.

Different types of TM formats will be used for messages generated by the TM encoder (real time telemetry format) or by the onboard computer (OBC telemetry format), but both TM formats shall use the frame structure presented in A-EIF-0002.

The format generated by the TM encoder shall consist of only one 128 byte frame containing real time telemetry data. The format generated by the onboard computer will consist of six frames. The first frame of the OBC telemetry format shall contain real time data and shall use the same data structure as the real time telemetry format. The remaining frames of the OBC format contains stored TM data, acquired along the satellite orbits and OBS house keeping data.

The telemetry format to be transmitted to ground shall be selected by direct command.

The telemetry transmission shall be done in accordance with A-EIF-0002.

3.2.4.2 - TELECOMMAND REQUIREMENTS

The TMTC and OBS subsystems shall provide all necessary commands to control or change the operating mode of the satellite in all mission phases. The spacecraft shall not depend on the on-board computer for the routing of commands in an operational basis.

The essential command actuations on the satellite shall be done by the TC decoder (direct commands) using up to 64 high-level pulses, in accordance with the interface specifications of A-ERC-0001. Some non-essential command actuations in a maximum number or eight are done by the on-board computer (normal commands).

The command reception shall be done in accordance with the specifications of A-EIF-0002.

The TC decoder shall accept telecommand frames coded in accordance with A-EIF-0002. Two addresses and sychronization words (05D9 and 05E2) shall be used, to address each redundant section of the TC decoder.

3.2.4.3 - RANGING REQUIREMENTS

The TMTC subsystems shall be able to relay to the ground ranging tones in accordance with the requirements A-EIF-0002.

The main TMTC subsystem performance requirements for the ranging function are:

- the transponder shall be of coherent type with a turn around frequency ratio of 240/221;
- the transponder shall contain an AGC in the ranging channel to maintain the rms sum of the major tone, any minor tone, residual TC and any noise, constant;
- the phase delay of the 100kHz major tone shall be known and shall be constant to within ± 70ns over the full range of doppler, input level, temperature, voltage and lifetime;
- the phase distortion introduced by the transponder between the highest and lowest range tones shall not exceed $\pm 5^{\circ}$ of the highest tone;
- a positive phase shift of the range tones on the uplink shall give rise to a positive phase shift on the downlink;
- the relaying of range tones shall be possible in both non-coherent and coherent modes.

3.2.5 - ELECTRICAL ARCHITECTURE

The electrical architecture of the spacecraft shall take into account:

- the electrical power supply for all equipments
- the necessary connections of command, telemetry and control lines between equipments
- the layout of the satellite to favor sensitive units and to minimize cabling
- the general EMC requirements imposed upon the equipment interfaces, power supply, wire harness and the consequent grounding scheme of the satellite.

3.2.5.1 - ELECTRICAL POWER SUPPLY REQUIREMENTS

The spacecraft solar arrays and battery shall be sized to provide the spacecraft power needs of all operating modes listed in Table 3.2.5.1.1 during the illuminated and shadow periods of each mission phase listed in Table 3.1.4.1.

TABLE 3.2.5.1.1

OPERATING MODES AND POWER CONSUMPTIONS

MODE	MAXIMUM TOTAL POWER CONSUMPTION (W)
Stand by	13
Orbit determination	24.4
Attitude determination	21
Normal operation	27.9
Experimental operation	32.9
Manoeuver	21.3
Experimental stand-by	23.4
Manoeuver stand-by	14
Degraded	34.9

The power supply subsystem shall be specified to cover the following conditions:

- Orbital period: 100 min
- Maximum eclipse time: 33 min
- Maximum number of passes in shadow: 5
- Maximum number of orbits without communication: 6
- Transmission average time per orbit: 11.4 min (after acquisition phase)
- Percent maximum depth of discharge: 30%
- Minimum solar angle aspect: 0^o
- Maximum solar angle aspect: 90^o
- Maximum area available for solar cells: top panel = $0.4352m^2$
  each lateral panel = $0.2176m^2$
- Maximum time in survie mode before separation from the VLS = 15 min

The main bus voltage shall be 26.5V and the other voltages (+15V, -15V and +5V) shall be derived from it through a DC/DC converter. The distribution of the power lines starts at the power distribution unit (PDU) where the telecommands to turn the equipments on or off shall be received.

3.2.5.2 - ELECTRICAL HARNESS REQUIREMENTS

The electrical integration of the spacecraft shall electrically interconnect all spacecraft equipments and shall provide the interactions between the spacecraft and the ground support equipments during the integration and launch phases.

The electrical integration shall comply with the following requirements:

- to distribute the power lines for each equipment
- to interconnect the signal lines between equipments
- to provide the required connections to the umbilical connectors
- to provide the required interconnections between the test connectors and the integration connector
- to provide the interconnections between the thermistors and the TM encoder
- to give an indication of the separation between the spacecraft and the launch vehicle, turning on the S-band transmitter.

Manufacturing of the electrical harness shall be in accordance with A-ERC-0001.

3.2.6 - MECHANICAL ARCHITECTURE

The maximum weight allowable for the integrated spacecraft is 115kg, including 4kg for balance masses. Construction and design requirements are specified in A-ERC-0001.

3.2.6.1 - STRUCTURE REQUIREMENTS

The structure of the MECB-S1 satellite shall be able to provide mechanical support and mounting for all onboard equipments, maintaining all the specified dimensions, alignments and tolerances under all operational conditions. It shall provide protection against the space environment and resist to all handling, testing, launching and orbital loads, as detailed in A-EAB-0001.

Figure 3.2.6.1.1 shows the structure shape to be detailed by the Structure Subsystem.

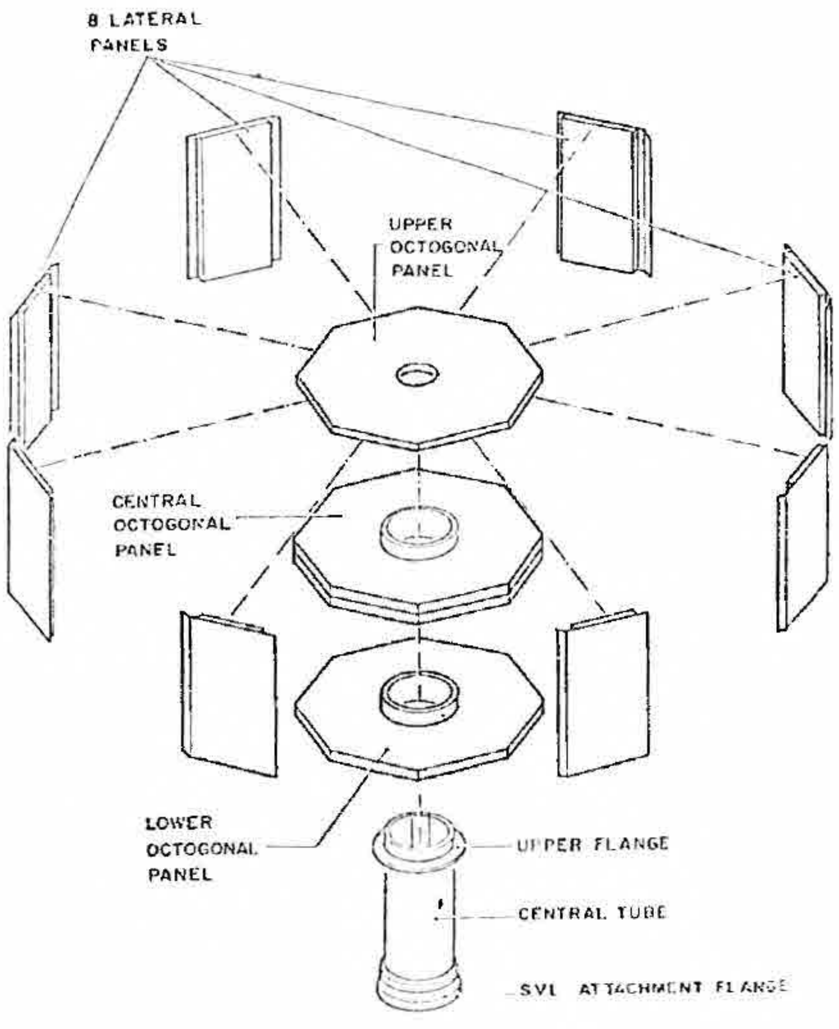

Fig. 3.2.6.1.1 - Structure Exploded View.

3.2.6.2 - LAYOUT REQUIREMENTS

With respect to Figure 3.2.6.1.1, the following surfaces shall be available as mounting surfaces for the spacecraft equipments:

- internal face of the superior panel
- upper and lower faces of the central panel
- internal face of the inferior panel.

The external surfaces of the lateral panels and the superior panel shall supply mounting areas for solar cells. On the external faces of the lateral panels, solar sensor attachments shall be provided; on the internal faces, only the diode plates of the solar arrays may be mounted. On the external faces of the superior and inferior panels, attachments for the antennas shall be supplied.

No equipment may be attached to any face of the central cylinder (external and/or internal). The lower UHF antenna shall be attached internally to the VLS interface flange.

The mechanical layout distribution shall take into account attitude control requirements due to the satellite spin stabilization. These requirements are:

- the C.M. location shall be along the spacecraft spin axis (z-axis) with a maximum dispersion of 10mm
- moments of inertia shall be such that

$$\frac{Iz}{Ix} > 1.08$$

$$\frac{Iz}{Iy} > 1.08$$

where x, y and z are the principal directions of inertia

- the z principal direction of inertia shall be within a 1^{o} cone with respect to the geometric z axis.

The reference system is defined in A-EAB-0001.

All final dimensions shall comply with the available volume for the spacecraft inside the VLS fairing as given in A-EIF-0001.

3.2.7 - THERMAL ARCHITECTURE

The spacecraft thermal control shall provide a temperature distribution in the satellite so that all onboard equipments operate within their design operational temperature range under all spacecraft attitudes of paragraph 3.2.2.

Thermal analysis shall take into account the selected equipment layout.

Passive thermal control systematics shall be used. No active thermal control devices are to be used.

Considering the external loads (sun, earth radiation and albedo) and the internal dissipation profile during all possible orbits, attitudes and operating modes, the following properties shall be established: solar absorptance and infrared emittance of coatings inside and outside the spacecraft, conductive couplings between electronic boxes and panels, and conductive coupling between adjacent panels. The selected properties shall establish a thermal model with low sensitivity to alterations during coatings application, bonding compounds film tickness and layout changes.

The equipment temperature shall be defined as the average value which may occur on the equipment external case, when the equipment is represented by a single node. Temperature distribution within the electronic boxes and hot spots on the external cases shall be analysed by the equipment suppliers.

3.2.8 - EXTERNAL INTERFACES

The spacecraft shall interface mechanicaly with the launch vehicle through an interface flange riveted to the inferior part of the structure central tube. The flange configuration is as defined in A-EIF-0001.

The spacecraft shall interface with the check-out station during integration and tests through an integration connector and two umbilical connectors, and during the launch phase via only the umbilical connectors. The integration connector shall include accesses for magnetometer calibration and for battery disconnection. The umbilical connectors shall have the configuration depicted in A-EIF-0001.

3.2.9 - RELIABILITY

The spacecraft shall be designed to meet the requirements of this specification over the periods indicated in 3.3.1.

The subsystem designs and redundancy configurations shall be such as to meet the twelve months numerical reliability figures of table 3.2.9.1 following ground operations, test activities and storage periods.

TABLE 3.2.9.1

ALLOCATED RELIABILITY

SUBSYSTEM	RELIABILITY (6 MONTHS)	RELIABILITY (12 MONTHS)
Payload	0.970	0.941
TMTC	0.990	0.980
Power Supply	0.970	0.941
Attitude Control	0.980	0.960
Structure	0.995	0.990
Thermal Control	0.995	0.990
Electrical Integration	0.980	0.960
OBS	0.980	0.960
Spacecraft	0.867	0.752

Subsystem designers shall provide subsystem reliability analysis comprising reliability block diagrams, reliability prediction, parts stress analysis and single point failure list in accordance with the PA plan A-CQL-0006.

3.2.9.1 - SINGLE POINT FALURES

A Single Point Failure (SPF) is defined as any piece, part, equipment, assembly or element of construction, the failure of which would result in subsystem performance below the specified level. All other SPF will be designated as a non-permitted SPF if it has failure rate greater than 20 FITS. In such later cases effort shall be given to removing the SPF (e.q. by redundancy or new design). A waiver will be raised against this requirement only as a last resort after a through systematic and exhaustive investigation has been performed.

3.3 - DESIGN AND CONSTRUCTION REQUIREMENTS

Satellite equipment design and construction requirements shall be in accordance with A-ERC-0001.

3.3.1 - LIFE

Spacecraft equipment shall be designed to survive and function within specification over the following durations:

- Assembly, integration and test (AIT) 1 year
- Storage 1 year
- In orbit 1 year

3.3.2 - REDUNDANCY

Redundancy shall be used as necessary to achieve the required reliability figure. Single point failures shall be avoided where technically feasible. The remaining SPFs shall be submitted to contractors approval and its effects on spacecraft and probability of occurrence evaluated.

3.3.3 - INTERCHANGEABILITY

Each equipment of the same configuration status shall be directly interchangeable without mechanical and thermal modifications such as hole drilling or planarity adjustment. It must be of the same qualification status and reliability when considered for interchange.

3.3.4 - MAINTAINABILITY

Each equipment shall be designed so that calibration can be maintained, adjustments performed and faults identified with standard tooling and test equipment. When this requirement cannot be fulfilled, the equipment contractor shall supply the contractor in charge of integration with the necessary special tooling, together with a written procedure for using it.

3.3.5 - TEST POINTS

Complete bench checkout of the equipment and isolation of failures at the circuit (not part) level is required. To this end, test points shall be provided if necessary.

The test points shall be identified in the equipment specifications. Preferably, all the test points shall be grouped on a dedicated connector. Test points on multi-pin connectors shall be designed to withstand, without causing damage, voltages above the highest voltage on that connector as well as short circuits, unless a higher voltage is specified in a particular subsystem. This voltage may be the output voltage from that equipment or the expected maximum test input voltage - whichever is the largest. Shorting these test points to ground shall not cause damage.

All test connectors shall be provided with covers to ensure protection against mechanical damage when the equipment is operational. Unless otherwise specified, the protective covers shall be easily removable.

3.3.6 - SAFETY

The equipment shall be designed and manufactured with compatible materials in such a manner that all hazards associated with the equipment are eliminated or minimized and controlled.

The design of the equipment and of the subsystem shall be capable of sustaining a failure in flight and retaining the property not to cause damage to the launch vehicle and satellite.

Safety requirements of the launch vehicle shall be applicable.

3.3.7 - WORKMANSHIP

Workmanship shall be such that the design standard is not degraded or changed. At all points during manufacturing, integration, test, handling, storage and transportation, the design standard shall be maintained. The skill level of personnel is to be such that all aspects of workmanship will ensure retention of the high reliability standards.

Manufacturing and process standards shall be in accordance with A-GQL-0006.

3.3.8 - IDENTIFICATION OF PRODUCT

Each equipment, excluding interconnecting cables, shall be permanently labelled with an identification plate. The label includes as a minimum, the following:

- Equipment name
- Manufacturer's name
- Manufacturer's part number
- Serial number
- Contractor's part number
- Manufacturing date (year, month, day)

3.3.9 - PRODUCT ASSURANCE

The requirements of Product Assurance plan (A-GQL-0006) shall apply. Equipment subcontractors shall develop a PA plan in accordance with the requirements of subcontractors PA requirements for satellite equipments (AD7).

3.3.10 - MATERIALS, COMPONENTS AND PROCESSES

Components, materials and processes used must comply with A-GQL-0006 PA plan.

a) Fungus Resistance

Materials that are nutrients to fungi shall not be used when their use can be avoided. Where used, and not hermetically sealed, they shall be treated with a suitable fungicidal agent. If they are used in a hermetically sealed enclosure or if they are stored in a continuously controlled environment, fungicidal treatment will not be necessary.

b) Dissimilar Metals

Contact of dissimilar metals with each other shall be avoided wherever possible. Protection against electrolytic corrosion which can result from such contact shall be provided by surface treatment of the metals.

c) Magnetic Characteristics

Magnetic materials shall be included only when required for proper operation of the equipament. Magnetic requirements, shall be as specified in A-ERC-0001. If necessary equipments shall be depermed in the de-energised state, prior to integration.

d). Coatings and Adhesives

Coatings and adhesives shall be capable of withstanding the environmental requirement of Environmental Specification (A-EAB-0001) without undue performance degradation. Product Assurance Requirements (A-GQL-0006) shall be used for the selections of space adhesive materials.

e) Standard Parts

The selection, evaluation, qualification, and screening of high reliability parts for qualification and flight hardware shall conform to the requirements of Product Assurance Requirements. The engineering model may use the equivalent design status parts but of a lesser reliability level. Parts derating requirements of A-GQL-0006 shall apply.

f) Processes

Documented controlled processes and trained certified personnel shall be maintained for all operations which may impact the reliability or quality of the equipment. All processes, such as heat treating, polymerization curves, soldering, protective coatings, thermal coatings, potting, conformal coating, welding, etc., shall be identified.

g) Corrosion

Selected materials shall be of the corrosion-resistant type of suitably treated to resist corrosive conditions likely to be met in storage and/or normal service.

h) Organic Materials

Organic materials selected for use in the equipments shall not outgas more than 0.1% volatile condensable materials nor more than 1% of the total weight. This requirement shall not apply where less than 0.5 grams of a given material is used.

i) Flammability, ToxiCity, Degradation

Materials characterized by flammability, toxidity and degradation shall not be used.

3.3.11 - ELECTROMAGNETIC COMPATIBILITY

The equipments shall meet all requirements of A-ECE-0001 for radiated and conducted emission and susceptibility when tested as an equipment, and as part of the system.

3.3.12 - ENVIRONMENTAL REQUIREMENTS

The equipments shall be designed to withstand all mechanical, thermal, climatic and other in-orbit (pressure and radiation) environmental conditions defined in A-EAB-0001.

3.3.13 - PREPARATION FOR DELIVERY

All deliverable equipments shall be prepared in a manner which will afford protection against corrosion, deterioration and damage, from the supply source until its requirement for use. The requirements of PA plan (A-GQL-0006) for packing, handling, identification and shipping shall apply.

a) Handling

Storage and shipping containers for deliverable equipment requiring special handling, loading or unpacking instructions shall be fully identified and documented by appropriate procedures. Special carts and containers shall be used to transport the equipment to different areas within the facility to minimize risk of mishandling.

b) Packing

Equipment subject to deterioration, corrosion or damage while being transported shall be packed in a manner and with materials necessary to prevent or minimize damage. Packing methods and instructions shall be prepared and provided on the exterior of the package or container for maintenance of specific internal or external environments.

c) Identification

The contractor's complete name and address or destination of the equipment shall be clearly identified and marked on the shipping container. Additional markings on the shipping container shall include the following information:

- Gross weight loaded (in lb and kg)
- Grosss weight empty (in lb and kg)
- Freight width and length (in ft., in. and cm, meters)
- Caution or warning markings (i.e., to be opened under PA supervision).

d) Storage

Equipment requiring storage shall adequately be protected against deterioration and damage to ensure life and utility for the intended period of time. Means shall be provided for adequate safety, preventive maintenance and periodic inspection. Appropriate special requirements or instructions dates shall be provided by the equipment contractor.

e) Shipping

The equipment container can be transported by car, closed truck or by aircraft. Minimum/Maximum temperatures and g-sensors shall be used for shipping flight equipments. Transportation by railway is not allowed.

4. VERIFICATION

4.1 - INTEGRATION AND TEST REQUIREMENTS

The activities of integration and test of the S/C shall be sequenced and planned such as to provide:

- the mechanical assembly and electrical interconnections of the various subsystem equipments aimed at molding them into an electrically compatible, mechanically rigid and reliable system,
- an interface check of each subsystem after individual integration,
- successive evaluations after each major integration event to allow an evaluation of the performance evolution,
- the environmental test reports of the S/C,
- the necessary means for a gradual implantation and validation of the EGSE (equipment and software), integration and test plan, MGSE and of the interfaces with the test facilities, launcher, ground segment, etc...

The specification of the assembly, integration and tests shall be in accordance to the following requirements:

a) The equipments shall be sequentially integrated to validate their mutual interfaces and their compatibility with the mechanical, electrical, physical and operational environment of the S/C.

b) The test definitions shall be based on the input of known data to the S/C through simulations or stimulations, on the analysis of those results which allow an evaluation of the behavior of the S/C, and on the comparison of the test results with the intended performances.

c) Once an equipment is integrated, a test is performed and its results shall be compared with those supplied by the subsystem responsible. The test conditions should be as similar as possible to those used in the subsystem tests performed prior to integration. The results of this test shall serve as a reference to the respective integrated equipment.

d) After all equipments are integrated, the so called reference tests (S/C REF) shall be performed to establish the results at the given instant and S/C configuration. A selection of these tests shall be repeated, on the same conditions, after each additional tests applied to the S/C.

e) The S/C REF tests are representative of all phases of the S/C lifetime. To achieve their objective these tests shall explore all the S/C modes of operations and redundancies, and shall be performed in repeateable conditions and with the least duration possible. These tests shall be chosen and optimized such as to allow a maximum degree of their automatic control which is compatible with the existing test facilities. The S/C REF tests serve to detect the mutual interference between equipments and the degradation of the critical S/C parameters. Therefore, they shall necessarily include at least:

1) Performance tests of each subsystem (referred to the subsystem test report) in all operating modes and in all redundancies, with access to the test points. During these tests all other subsystems shall be in their most susceptible configurations.

2) System performance tests (referred to the system tests) in operational, stand-by and degraded configurations, intended to measure the essential mission characteristics and making the maximum possible use of the natural links with the satellite in its flight configuration.

3) Analysis of all particular critical configuration recognized during the integration and the previous tests, or which results from studies or of limited points in EMC-EMI.

f) The functional tests shall be performed through the longest possible equipment chain and in two distinct ways:

1) Qualitative tests by excitation of the detectores (yes or no) since quantitative tests at this level are not possible.

2) Quantitative tests through the sensors (if possible) or through the sensor electronics utilizing the longest possible chain and preferably performed in closed loop via the telemetry.

g) All operations carried out on the spacecraft shall be performed according to previously approved documentation.

h) The total operating time and the number of on off cycles that each equipment is subjected to during the tests will be recorded in the equipment log-book.

i) The test equipment design shall minimize the risk of damaging the S/C or S/C equipment under test. For this purpose it is required that:

1) The test equipment have appropriate overvoltage and overcurrent protection.

2) Whenever possible and without degrading performances, all qualification and flight equipment be tested through connector savers until they are integrated into the satellite.

3) The connectors of the test equipment be such that they prevent inadvertent misconnections.

j) All test equipment with any associated harness and software shall be fully checked and validated prior to use with the satellite. This applies also for mechanical handling equipment used in testing the S/C.

k) Only a limited number of personnel, knowledgeable in the procedures will be permitted to work on the S/C at one time.

4.2 - SYSTEM TESTS

The system tests shall be effected on the S/C models described in the following section.

4.2.1 - SATELLITE MODELS

- Identification Model (MI): Composed of the sybsystems MI according to the hardware matrix on A-GRC-0010.

- Proto-Flight Model (PF): Composed of the subsystems flight model according to the hardware matrix on A-GRC-0010.

- Structural and Thermal Models (SM,TM): Composed of representative models of the S/C structure and of structural and thermal equipment mock-ups, according to the hardware matrix on A-GRC-0010.

- Radio-elctric Mock-up (RFM): Representative of the external geometrical configuration of the S/C and equipped with performing antennas, according to the hardware matrix on A-GRC-0010.

4.2.2 - REQUIRED TESTS

The required system tests shall be in the cathegories: performance tests, compatibility tests, radio coverage tests, mechanical tests and thermal tests. A summary of the required system tests is given in section 4.3.

4.2.2.1 - PERFORMANCE TESTS

These tests shall be of a repetitive characteristic and shall be executed through the electrical chains of the S/C utilizing the S/C check-out station. These tests shall be performed on MI and PF.

4.2.2.2 - COMPATIBILITY TESTS

4.2.2.2.1 - ELECTROMAGNETIC COMPATIBILITY TESTS (EMC)

To be performed on MI according to A-ECE-0001 and shall include emitted and conducted tests. The EMC tests shall be repeated on PF only if important configuration changes occur.

4.2.2.2.2 - LAUNCHER INTERFACE COMPATIBILITY TESTS

A fit check with the launcher interface shall be performed on PF.

4.2.2.2.3 - GROUND SEGMENT COMPATIBILITY TESTS

A compatibility test with the ground segment shall be performed during the S/C performance tests.

4.2.2.2.4 - MAGNETIC BALANCE TEST

The PF model of the S/C shall be magneticaly balanced to make its residual magnetic moment value compatible with the S/C requirements.

4.2.2.3 - RADIO COVERAGE TEST

Radio coverage tests shall be performed on RFM.

4.2.2.4 - MECHANICAL TESTS

4.2.2.4.1 - WEIGHT AND CM MEASUREMENTS

Weight and CM measurements shall be performed on all subsystem equipments prior to integration and on MI and PF. Balance masses and moments of inertia will be inferred from the above measurements. CM location shall be corrected according to the margin allowance.

4.2.2.4.2 - BALANCE

Balance shall be performed on PF to obtain the necessary inertias margin allowances.

4.2.2.4.3 - ALIGNMENT

Alignment measurements shall be performed before and after the environment tests performed on PF.

4.2.2.4.4 - VIBRATION TESTS

Sinussoidal and random vibration tests shall be performed on PF and SM according to the levels indicated in A-EAB-0001.

4.2.2.4.5 - SEPARATION TEST

A separation test shall be performed on SM according to A-EIF-0001.

4.2.2.5 - THERMAL TESTS

Thermal tests under vacuum condition and IR (skin heater) shall be performed on TM according to the philosophy and levels indicated in A-EAB-0001.

4.2.3 - DESIGN VERIFICATION MATRIX (DVM)

All tests shall be performed in accordance with comprehensive test procedures delivered in advance of the relevant tests. Rigid change control will be maintained for all test procedures and related computer programs.

Satisfactory completion of the integration and test program shall be confirmed by a formal review of the test data which shall be in conformity to the spacecraft DVM.

4.2.3.1 - MANDATORY TESTS

The satellite and subsystem design verification matrices shall necessarily include the required verifications listed in the Appendix.

4.3 - SUMMARY OF SYSTEM TESTS

	RFM	SM/TM	MI	PF
Performance: S/C REF and Subsystem			X	X
Compatibility				
- EMC-EMI			X	X*
- Launcher Interface				X
- Ground segment				X
- Magnetic balance				X
Radio Coverage	X			
Mechanical				
- Mass and CM			X	X
- Balance				X
- Alignment				X
- Vibration		X		X
- Separation		X		
Thermal		X		X

* Eventually

APPENDIX

Required Verifications

The following pages list for each subsystem and for the satellite, the minimum mandatory requirements to be verified in their respective DVM, together with the verification method and the test descriptions.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN FLIGHT TEST
	STRUCTURE					
A1	Vibration	X	X	X	X	

A1 - The vibration test shall be performed on the SM and PF by mounting the model to an electrodynamic shaker using an adapter and flight type separation clamp. The test levels for the SM and PF will be qualification and acceptance, respectively, as outlined in A-EAB-0001.

Test accelerometers and strain gages will be installed in the spacecraft at relevant locations, in order to be able to compare test results with structural mathematical model predictions and to measure the levels of vibrations applied to the equipments.

Random vibration and sine vibration shall be applied consectively in three directions: thrust axis (z-z) and two orthogonal lateral axes (x-x and y-y).

The sequence of events for each sinus vibration axis will be as follows:

- low level run
- determination of notching levels according to measured qualification factors
- high level sweep
- low level verification sweep for comparison to initial satellite signature.

Notching on levels applied to the satellite during sinusoidal testing will be made at the resonant frequencies for the main structure (and of secondary structure or equipments) to prevent overstressing the structure.

During the PF test, the satellite will be in the electrical launch configuration and verified through TM. Test batteries are installed and used to power the spacecraft.

After the 3 runs the satellite will be inspected; no disconnections will be made until system performance test is performed, the non acessible points being inspected at that time.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	THERMAL CONTROL					
B1	Thermal Balance	X	X	X		
B2	Thermal Cycling			X	X	

B1 - Infra-red test technique will be utilized for the thermal balance test on the TM. The purpose of the test on the TM is to check the thermal mathematical model employed in the design and to validate the hardware solution adopted for the thermal control. This test applied to the PF or to an up dated version of the TM will give an ultimate indication of the satellite thermal control design.

For the TM test, the structure is outfitted with thermal mock-ups of all the spacecraft equipments. Every aspect of the thermal control subsystem (e.g., coatings, insulation and shields) is included on the test model. Resistance heaters are used in the mock-up of all power dissipating equipment. No functional checks are made during the tests.

Thermal balance test is performed in a space simulation chamber. The chamber must feature ultra-clean vacuum pumping facilities to mantain the required low pressure during the test, as well as liquid nitrogen cooled shrouds to simulate the heat sink of space. Simulation of the solar heating on the spacecraft is accomplished by the skin heater method. The test conditions imposed will include the critical cold an warms orbits as well as a number of eclipses.

B2 - The thermal cycling test on PF is intended at verifying the satellite performance under extreme vacuum, temperature and thermal gradient conditions. The satellite is driven to alternate hot and cold soaks during which performance tests are effected. In the hot and cold soaks the satellite is placed in a temperature profile corresponding to the maximum and minimum flight conditions, respectively, and a certain margin is added on to these values. It may be necessary to reinforce the temperature levels of same equipments to ensure that they are requested up to their specified temperature limits.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	POWER SUPPLY SUBSYSTEM					
C1	Solar Array Verification		X	X	X	
C2	Solar Array Output	X	X			
C3	Bus Operating Range		X		X	
C4	Eclipse Mode Operation		X		X	
C5	Power Consumption		X		X	
C6	Battery Charge		X		X	
C7	Power Distribution		X		X	
C8	Battery Discharge		X		X	
C9	Bus Stability		X			
C10	Power Generation	X				
C11	Shunt Dissipation		X		X	
C12	Orbit Condition Simulation				X	

C1 - Electrical continuity from the solar array to the satellite will be verified by inspection. Each panel of the solar array will be partially illuminated after assembly onto the S/C. Proper integration of the panels will be verified by monitoring the main bus voltage during partial illumination.

C2 - Each solar panel will be tested in a non-reflective area, prior to integration, using the solar panel test set. A xenon light pulse simulating the solar illumination will be used and by simulating various loads on the array, the I-V curve will be obtained.

C3 - The solar array simulator (SAS) will be set to simulate full power condition. The SAS output will be decreased to zero. Verification that the bus voltages and regulations remain in their ranges irrespective of the loads will be made via telemetry. The output of the SAS will be increased from zero to full voltage while monitoring the bus voltages and regulations. It will be verified that they remain in their operating ranges as the SAS output is increased. During this test the SAS will simulate the solar array ripple. The Main Error Amplifier (MEA) redundancies will be verified by stimuli signals at the maximum and minimum SAS outputs.

C4 - This test is aimed at showing that the battery has sufficient capacity to power the S/C through a maximum eclipse period. The battery will have been fully charged prior to the test. With the battery powering the S/C and the eclipse operating mode loads on, it will be verified that the main bus voltages remain in the specified range for the eclipse period.
The battery DOD will be verified bu telemetry return.

C5 - This test will verify that all subsystem power requirements are within specification and demonstrate that the total integrated primary power bus loads are within the capabilities of the solar arrays and the battery. As each subsystem is commanded on, the main bus currents will be measured by telemetry and the power consumption of each subsystem element will be evaluated. At subsystem level, equipment simulation loads will be used. These tests shall be performed in the absence of shunt operation and battery charge the PSS power consumption shall be inferred by analysis.

C6 - The purpose of this test is to show proper battery charge capability and charge regulator performance. The solar array simulator (SAS) will power the S/C electrical power subsystem with the maximum battery charge current. Maximum battery charge capability will be verified by its voltage and temperature telemetries. The performance of the battery charge regulator will also be monitored by its telemetries, and its redundancy tested.

C7 - This test is aimed at verifying the switching functions of the DC/DC converters (including times) and the proper operation of the power distribution unit (PDU). With power supplied by the SAS, without shunt operation and battery charge, the times operation and DC/DC converter redundancy will be checked. All possible operations of the PDU will be explored by sending in the appropriate telecommands and verifying the telemetry returns.

C8 - This test serves to verify the operation of the battery discharge regulator (BDR) and of its redundancy. With an initially charged battery the BDR must maintain its output voltage within the specification up to the average battery voltage. This test must be effected using a dynamic load simulator (varying from minimum to maximum load) at subsystem level. Redundancy operation during the test is checked by telemetry. At system level, only the BDR redundancy operation shall be checked.

C9 - A solar array simulator (SAS) shall be used to operate the S/C under normal sunlight conditions. Adjustable loads shall be connected across the secondary bus lines. An oscilloscope can be used to measure bus transient response as a pulsed load is used. A test battery shall be used to operate the S/C under eclipse condition, and the same procedure shall be repeated as above.

C10 - The capability of the spacecraft solar array and battery to provide the spacecraft power needs in the orbit environment during all mission phases shall be demonstrated by analysis for confirmation by in-orbit tests.

C11 - The shunt power dissipation and power balance shall be measured when the spacecraft power is supplied by the SAS. The measurements shall be performed as the SAS power is varied from maximum to minimum. The test shall be performed for maximum and minimum spacecraft loads.

C12 - In-orbit operation condition shall be simulated (minimum of 7 orbits) to verify subsystem operation. For this test the spacecraft loads shall be those equivalent to the experimental and experimental stand-by modes. The test will be repeated for the SAS simulating maximum and minimum power from the solar array. The battery charge and discharge shall be monitored throughout the test and its DOD inferred by analysis.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	ATTITUDE CONTROL					
D1	Nutation damper performance	X				
D2	Nutation decay time	X	X			
D3	Torque coil effect	X				
D4	Torque coil moment	X	X		X	
D5	Magnetic balance				X	
D6	Magnetometer performance		X		X	
D7	Sun sensor performance		X		X	
D8	Alignment				X	

D1 - The nutation damper functional performance shall be assessed by analysis. A simplified nutation damper model shall be used for the damping constant calculation. Estimates of the nutation damper constant shall include two special dyanamic conditions:

a) large initial nutation angle,

b) environmental transverse torque effects.

D2 - An experiment shall be performed to verify the nutation decay time. An air-bearing supported table which is a representative dynamical mock-up of the S/C will be properly balanced by the instalation of weights. The table will be driven to a certain angular speed about an axis making a desired angle with respect to the symmetry axis. A light beam fixed above the table will be used to obtain pictures of the axis of symmetry motion.

D3 - The effect of the torque coil on the S/C attitude shall be assessed by analysis through the knowledge of its magnetic moment and of the residual magnetic moment of the S/C.

D4 - At the equipment level the torque coil moment shall be obtained by the direct measurement of the magnetic field created at its center when energized. At system level, this moment shall be inferred to an accuracy better than 1 Am^2, by measuring the perturbation caused in the local earth magnetic field by the presence of the S/C with energized torque coil. This measurement shall be performed outdoors with a three axes magnetometer located at a distance no greater than 70cm from the S/C center and having the S/C fixed to a non-magnetic table which is capable of rotating.

D5 - The S/C shall be magneticaly balanced by the inclusion of suitable permanent magnets, to an accuracy batter than 0.15 Am^2. The S/C magnetic moment with de-energized equipments shall be inferred by a measurement similar to the one described in D4.

D6 - At the equipment level the magnetometer performance shall be verified through the measurement of a calibrated field (to an accuracy of 5 mG) created by a Helmholtz coil system. At system level, the calibration current imput of the magnetometer shall be activated and the magnetometer telemetry verified.

D7 - The sun sensor performance shall be verified using a sun simulator. A rotating mirror placed in front of the sun simulator will create the necessary spin effect. The test shall be aimed at verifying:

a) Sensor field of view.

b) Sensor resolution and accuracy around the field of view.

c) Sensor output stability and sensitivity to the spin rate.

D8 - Alignments of the integrated sun and magnetic sensors shall be optically verified to an accuracy of $\pm 0.25^{\circ}$ relative to all spacecraft axes.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	DATA HANDLING					
E1	Command operation		X		X	
E2	Frame acceptance		X			
E3	Command sensitivity				X	
E4	Telemetry operation	X				
E5	Real-time telemetry		X		X	
E6	Storage telemetry		X		X	
E7	Software validation		X			
E8	Self-test				X	

E1 - The command operation will be tested by sending Execution Data Packages to generate he commands and verifying the existence of pulses at proper command outputs at subsystem level and checking the switch status in the received telemetry frames at system level. All commands in both modes of operation will be checked.

E2 - Wrong patterns will be generated in the Memory Load Command to test the capability of the Command Analizer Routine to detect them. The result of the verification will be sent in the telemetry frames.

E3 - Command sensitivity will be tested at system level as a function of the receiver input power to show the threshold of the complete chain. Threshold will be defined as the power level at which at least 50% of the commands are verified.

E4 - The telemetry operation will be verified testing the real-time telemetry and the storage telemetry.

E5 - The real-time telemetry will be tested, at subsystem level, by aplying predetermined signals at the telemetry channels of the DH and verifying the received telemetry frames using the on Board Test Checkout System (MTSB). At system level, the values of the telemetries present in the received frames will be compared with the expected values.

E6 - The storage telemetry will be tested, at subsystem level, by applying a predetermined time-varying signal at each telemetry channel input of the DH and verifying whether the correct value will be present at the storage telemetry frames. At the system level, the values of the acquired telemetries present in the storage frames will be compared with the expected values.

E7 - The software will be validated using the Lab. Prototype supported by the software Development Station (HP 69000) and the MTSB following the procedures defined by the IEEE std 829-1983.

E8 - The DH will have diagnosis routines for self-testing. These routines will be initiated by an Execution Data Package which provides the initial values for the tests. The results of the diagnosis routines will be sent in the telemetry frames.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	SERVICE TELECOMMUNICATION					
F1	Receive G/T	X				
F2	Antenna receive characteristics		X			
F3	Antenna noise temperature	X				
F4	EIRP	X				
F5	Antenna transmit characteristics		X			
F6	Output power		X		X	
F7	Receiver rest frequency		X		X	
F8	Acquisition threshold		X			
F9	Carrier acquisition		X		X	
F10	Tracking range		X			
F11	Selectivity	X	X			
F12	AGC response		X			
F13	PM demodulation	X				
F14	Command video sensitivity		X			
F15	Ranging output sensitivity		X			
F16	Output frequency		X		X	
F17	Spurious emission		X		X	
F18	Frequency stability		X		X	
F19	Modulated spectrum		X		X	
F20	Output phase noise		X			
F21	Telemetry modulation		X			
F22	Ranging modulation		X			
F23	Phase delay		X		X	
F24	Residual AM		X			
F25	False Command Probability	X				
F26	Bit error rate		X			

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
F27	Squelch operation		X			
F28	Command Operation	X				
F29	Direct commands		X		X	
F30	24-bit serial interface		X			
F31	Frame acceptance	X				
F32	Command word acceptance		X			
F33	Mode word acceptance		X			
F34	Address and sync word acceptance		X			
F35	Command sensitivity				X	
F36	Telemetry monitoring		X			

F1 - The receive G/T will be evaluated by analysis using the measured antenna characteristics, the antenna noise temperature and the value of the transponder noise figure.

F2 - The measured antenna receive characteristics (gain, diagram and axial ratio) shall be obtained through the RFM test results. This test will follow the outline detailed in the subsystem test procedure.

F3 - The antenna noise temperature is obtained by analysis using the measured antenna characteristics and the equivalent noise temperature of earth and space. This calculation will follow the outline detailed in the subsystem test procedure.

F4 - The EIRP coverage is calculated using the antenna transmission characteristics and the measured RF output power.

F5 - The measured antenna transmit characteristics (gain, diagram and axial ratio) shall be obtained through the RFM test results. This test will follow the outline detailed in the subsystem test procedure.

F6 - Measure the output power of an unmodulated transmit carrier at the antenna port.

F7 - An unmodulated carrier is manually swept ±50kHz about the nominal frequency at the input of the transponder and the static phase error (SPE) is monitored at the test connector. The frequency at which the SPE is zero is measured. This test is performed for input powers of -115dBm and -50dBm. At system level this test is performed at the antenna port for input levels 3dB higher and the SPE is monitored at the telemetry of the decoded frames.

F8 - Measure the minimum RF input power to the transponder at which the receiver will acquire an unmodulated carrier at the rest frequency. This measurement is performed by varying the input power and monitoring the lock status at the test connector.

F9 - An unmodulated carrier is swept by ±50kHz about the nominal frequency at a rate of 32kHz/s at the transponder input. The carrier acquisition is verified by the lock status at the test connector. This test shall be performed for input powers of -50dBm and -115dBm and shall be repeated a number of times to estimate the probability of carrier acquisition. At system level the test is performed at the antenna port increasing the power levels by 3dB and verifying the lock status in the decoded telemetry frames.

F10 - An unmodulated carrier with power level of -115dBm is applied to the transponder input. The maintenance of lock is verified at the test connector while the carrier is swept by:
±60kHz about the nominal frequency at a rate of 32kHz/s
±80kHz about the nominal frequency at a rate of 3kHz/s
±115kHz about the rest frequency at a rate of 5kHz/s
The test is repeated for an input power level of -50dBm. At system level this test is performed at the antenna port for an input level 3dB higher and the lock verified at the decoded telemetry frames.

F11 - An unmodulated carrier at -35dBm is manually swept by ±100MHz about the nominal frequency at the transponder input port. The AGC level is monitored at the test connector. The frequency selectivity is derived from the AGC calibration curve and the measured AGC response.

F12 - The power of an unmodulated carrier at nominal receiver frequency is varied from -115dBm to -50dBm and the AGC response is measured at the test connector. The AGC response time is measured with an oscilloscope by applying a 0.25Hz square wave modulation to the carrier.

F13 - The PM demodulation performance is evaluated from the command video and ranging output sensitivity measurements.

F14 - The TC video sensitivity is measured with an input PM signal of -115dBm and S/N ratio equal to 10dBHz, modulated by an 8kHz tone with modulation index of 1 rad. The S+No is measured at the demodulated TC subcarrier output using a spectrum analyser.

F15 - The ranging output sensitivity is measured with an input PM signal of -115dBm level and S/No ratio equal to 10dBHz, modulated by the tones at 16kHz and 100kHz. The S/N ratio of the tones are measured at the ranging demodulated output.

F16 - The transponder output frequency shall be measured in the coherent and non-coherent modes. In the coherent mode the unmodulated output carrier frequency is compared with the input carrier frequency and the frequency conversion rate of 240/221 is verified.

F17 - Feed the transponder input with an unmodulated carrier of -90dBm at nominal frequency and measure the relative amplitude of spurius emissions with respect to the transmit carrier in a spectrum analyser, up to 8GHz. The test will be repeated with the transmit carrier modulated by TC and ranging tones at nominal modulation indexes. The test is also performed in the non-coherent mode with and without telemetry subcarrier.

F18 - The output frequency setting is measured for transponder operation in the coherent and non-coherent mode. The frequency stability is measured over the specified temperature and power supply variations according to the outline described in the subsystem test procedure. The long term stability is measured by recording the reading of a frequency counter with 3s gate period during 15 minutes.

F19 - Modulate a -90dBm carrier with test tones of 20kHz and 100kHz with nominal peak modulation indexes and feed the transponder input port. Feed the telemetry subcarrier input with a 65.535 kHz tone of 3 V peak to peak. Measure and record the spectrum of the output modulated carrier with an spectrum analyser in a 1.5MHz frequency span.

F20 - Feed the transponder input port with an unmodulated carrier at nominal frequency and -115dBm. Measure the output phase noise integrating the noise from 10Hz to 100kHz in a Phase Noise Measurement System.

F21 - Feed the telemetry subcarrier input with a test tone at 65.536kHz and voltage varying from 0.1 Vpp to 3 Vpp. Measure the modulation index of the transponder output signal using a modulation analyser at the IF output of the telemetry receiver, or at the IF output of a down converter. For a fixed voltage level of 3 Vpp sweep the test tone from 3kHz to 1.5MHz and record the modulation index values.

F22 - Feed the range video input with test tones of 16kHz and 10kHz and voltage varying from 0.1 Vpp to 3 Vpp. Measure the modulation index of the transponder output signal using a modulation analyser at the IF output of the telemetry receiver, or at the IF output of a down converter. For a fixed voltage of 2 Vpp sweep the test tone from 1,2kHz to1° 5MHz and record the modulation index values.

F23 - For ranging tones of 16.20 and 100kHz (sequentially), measure the tone delay by comparing the phase difference between the generated and received (at the check-out station receiver) tones. For calibration of the test equipment, the check-out station transmitter will generate a carrier at the down-link frequency which will be fed directly to the RF input of the check-out station receiver. The ambiguity resolution is obtained by analysis of the tone delay test results.

F24 - For the same input as in test F19 measure the AM modulation index of the output signal using a modulation analyser at the IF Output of the telemetry receiver or at the IF output of a down-converter.

F25 - Estimate the false command probability from the bit error rate measurement and the squelch test result.

F26 - Apply to the TC decoder input a PSK/NRZ-L subcarrier with nominal frequency and bit rate, modulated by a TBD bit pattern plus gaussian noise with spectral density such that E/No=16dB. Measure the bit error rate of the demodulated NRZ-L data stream at the test connector using a BER meter.

F27 - Change the E/No ratio at the TC decoder input from 16dB to 6dB and verify at the test connector the squelch operation. The input E/No value at which the squelch operates is measured.

F28 - The command operation is assessed from measurements of each of the direct command output pulses and of the 24-bit serial output.

F29 - Apply to the TC decoder input a PSK/NRZ-L subcarrier at the nominal frequency and bit rate, modulated by on/off command messages. Verify for each H/L output the command pulse characteristics. At system level this test is performed at the antenna port (modulating the input carrier by the command subcarrier with a modulation index of 1 rad) and verifying the command actuations in the decoded telemetry frames.

F30 - With the same conditions of as in F29, but modulated by memory load serial messages, verify at the 24 bit serial interface output, the transmission of clock, data and sampling signals.

F31 - Verify the frame acceptance using test results of address and sinchronization word, mode word and command word acceptance.

F32 - Apply to the TC decoder input a PSK/NRZ-L subcarrier with nominal frequency and bit rate modulated by an ON/OFF command message. First verify at each of the ON/OFF command outputs the actuation of the corresponding command, then, introduce errors (1 or 2) in each of data words and repeated data words and verify the command outputs. For memory load command the same procedure is repeated and the 24 bit serial interface output verified. This test will follow the outline detailed in the subystem test procedure.

F33 - With the same initial conditions as in F32 but introducing errors (0 or 1) in each of the mode word and repeat mode word, verify the command actuatiors. This test will follow the outline detailed in the subsystem test procedure.

F34 - With the same initial conditions as in F32 but introducing errors (0,1 or 2) in each of the initial an final ASW'S, verify the comand actuations. This test will follow the outline detailed in the subsystem test procedure.

F35 - Command sensitivity is tested at system level as a function of antenna port input power to show the threshold of the command system. The input signal is modulated by the TC subcarrier with a 1 rad modulation index. Threshold will be defined as the power level for which at least 50% of the commands are executed.

F36 - Verify the operation of the telemetries provided by the subsystem, monitoring the telemetry outputs at the test connector.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	PAYLOAD SUBSYSTEM					
G1	Turn-on time		X			
G2	Receive G/T	X				
G3	Noise temperature		X			
G4	UHF antena characteristic		X			
G5	Antenna noise temperature	X				
G6	In band frequency response		X			
G7	EIRP	X				
G8	Output power		X		X	
G9	S-band antenna characteristic		X			
G10	Modulation index		X		X	
G11	Amplitude linearity		X			
G12	Frequency stability		X		X	
G13	Incidental AM		X			
G14	Group delay slope		X			
G15	Output frequency		X		X	
G16	Spurious output		X		X	
G17	Selectivity		X			

G1 - The equipment will be commanded on having an input carrier at 401.635MHz of -100dBm and the time elapsed to obtain the correct AGC and output RF power telemetry lecture will be measured.

G2 - The receive G/T will be evaluated by analysis using the measured antenna characteristics, the antenna noise temperature and the transponder noise figure.

G3 - The equipment noise temperature will be measured through the test connector IF output using a spectrum analyser and a power meter to evaluate the signal and noise powers.

G4 - The measured antenna characteristics obtained through the RFM will be utilized.This test will follow the outline detailed in the subsystem test procedure.

G5 - The antenna noise temperature is obtained by analysis using the measured antena characteristics and the equivalent noise temperature of the earth and space. This calculation will follow the outline detailed in the subsystem test procedure.

G6 - An input carrier at 401,635MHz with -100dBm will be swept in frequency by ±30KHz at 100Kz and the frequency response will be verified with an spectrum analyser at the output.

G7 - Compute the EIRP coverage using the antenna characteristics and the RF output power.

G8 - The RF output power will be measured with no input signal and checked against the respective telemetry indication.

G9 - Same as G4 for the S band antenna.

G10 - An input carrier at 401.635MHz will be varied from -125dBm to -95dBm and the PM modulation index will be measured in a modulation analyser at the IF of the check-out station receiver or at the IF port of a down-converter.

G11 - With two input carriers at 401,62MHz and 401,635MHz with -95dBm each, the intermodulation products sh 11 be verified in the output with an spectrum analyser.

G12 - The short term frequency stability shall be measured by DIT's HP system.

G13 - With an input carrier at 401,635MHz and -100dBm the AM modulation index shall be measured with a modulation analyser at the IF output of the check-out station receiver or at the IF port of a down-converter.

G14 - The group delay slope shall be measured with a 401,635MHz input signal AM modulated by a 25Hz signal. The AGC telemetry signal shall be compared in a phase detector with the 25Hz modulating signal.

G15 - Measure output frequency with no input signal.

G16 - With a carrier in 401.665MHz and -95dBm register the RF output signal using a spectrum analyser up to 8GHz.

G17 - An input carrier with -50dBm will be manually swept from 200MHz to 600MHz and the IF output of the test connector and the PM modulation index verified.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	ELECTRICAL INTEGRATION					
H1	Harness loss	X				
H2	Harness shakedown		X	X		

H1 - To provide verification by analysis that the harness losses are within the specifications.

H2 - The objectives are to verify the following:

. End-to-end continuity on all power and return wires.

. Integrity of all wire paths including switch and thermistors.

. Open circuit voltage on all power connectors.

. Unit contact resistence.

. Certain redundant wiring paths.

. Grounding.

. Insulation.

. Bonding.

. Upon completion of the harness shakedown, all connectors may be safely mated so subsequent tests and calibrations can commence.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	IN-ORBIT ACCEPTANCE TESTES					
I1	Electrical power	X				X
I2	Eclipse/penumbre mode	X				X
I3	Battery conditioning	X				X
I4	Service telecommunications	X				X
I5	RF threshold	X				X
I6	TM EIRP	X				X
I7	Command operation	X				X
I8	Thermal Control	X				X
I9	TM temperature monitoring	X				X
I10	Attitude Control	X				X
I11	Flight dynamics	X				X
I12	Torque coil operation	X				X
I13	Data Handling	X				X
I14	Telemetry operation	X				X
I15	Command operation	X				X
I16	Data Collection Payload	X				X
I17	EIRP	X				X
I18	Frequency	X				X
I19	Modulation Index	X				X
I20	Payload link performance	X				X

I1 - The spacecraft available electrical power is assessed by analysis of telemetred data and correlation with pre-launch in-plant test data. The telemetred data to be analysed pertains to:

i) bus voltage,

ii) bus current into the equipment units of the S/C vehicle subsystems,

iii) bus control current from the shunt regulator.

After data analysis and reduction, this test allows derivation of:

i) the power required by spacecraft equipment units and subsystems in operation,

ii) the power supplied by the solar generator,

iii) the available power margin over demand.

It also allows the verification of the orientation of the solar array after comparison between computed and expected values of the available S.A. power. This test is to be performed in sunny periods for selected operating modes.

I2 - This test is to cover the change from solar-panel power supply to battery operation and vice versa, under eclipse/penumbra conditions. This test is to take into account the changes over duration (about 2 minutes). Bus voltage is telemetred in order to plot the bus voltage transients during the change over duration.

I3 - For any selected battery charging mode (trickle and normal charging current and battery voltage are measured using telemetry data. Battery voltage is verified and compared with prelaunch measured characteristics. Likewise, the battery current is verified on Telemetry data. One full complete battery discharge, charge cycle, occurring during the longest eclipse duration will be evaluated to determine the battery capacity.

I4 - The satellite service telecommunication subsystem performance shall be evaluated through in orbit tests of the RF threshold, EIRP and command operation.

I5 - The command RF threshold figure is obtained from the two receivers by decreasing the TT and C uplink density until commands are no longer properly received in the associated command unit (verified by Telemetry).

I6 - The telemetry EIRP is deduced from downlink power level measurements, taking into account the satellite attitude and the associated antenna pattern.

I7 - Commands will be transmitted to the satellite and their operation will be verified by telemetry.

I8 - The thermal behaviour of the spacecraft is assessed using telemetred temperature data.

I9 - Verification that telemetred spacecraft temperatures remain within the predicted limits. When the launch date is known it will be performed by analysis, temperature predictions which will be compared with flight telemetred temperatures during in orbit, tests.

I10 - The attitude control subsystem performance shall be assessed by the analysis of the flight dynamics in orbit tests and of the torque coil operation experiments.

I11 - Flight dynamics information will be gained by recording the attitude of the satellite (calculated through the sensors telemetered data, for a period of at least one month, and comparing it with predictions that will be made by analysis once the launch date and the initial satellite attitude are known.

I12 - The torque coil operation experiment will be performed after at least three months from the launch date. The test will consist of sending an on + command to activate the torque coil, and after one day an on - command. After one more day an off command to the torque coil will be sent. The attitude record of the satellite during all the test will be compared with analytical results.

I13 - The data handling subsystem in orbit test shall consist of telemetry and command operation tests.

I14 - Telemetry through the data handling subsystem shall be tested in all its modes of operation. A test of stored telemetry shall be programmed.

I15 - The telecommands through the data handling subsystem shall be sent and their actuation checked by normal telemetry return.

I16 - The in orbit data collection subsystem evaluation shall consist of power, frequency and modulation index measurements.

I17 - The payload EIRP is calculated from a downlink power level measurement, taking into account the satellite attitude and the associated antenna pattern reference measurement.

I18 - The received frequency from the satellite payload shall be measured and correlated with prelaunch test and calibration data, mainly the frequency stability over temperature. The frequency of a reference PCD in the ground station shall be measured too this measurements shall taking due account for the doffer shift.

I19 - The same reference PCD used in test I18 shall be utilized for the modulation index test. The same kind of test method made for determining the modulation index at subsystem level shall be employed. This test shall be repeated for various orbits to take into account the most unfavorable satellite ranges.

I20 - A reference PCD installed near the ground station shall be used as a continuous transmitter and the BER calculate as a overage of the errors measured in 6 good fasses.

TEST Nº	REQUIREMENT	VERIFICATION METHOD				
		ANALYSIS	S/S EQUIPMENT TEST	INSPECTION	SYSTEM TEST	IN ORBIT TEST
	SATELLITE					
S1	Storage	X		X		
S2	Handling-transport			X		
S3	Reliability	X				
S4	Service life	X				
S5	Pre-launch test	X			X	
S6	Launcher compatibility	X				
S7	Mass Properties	X	X		X	
S8	Size	X	X		X	
S9	Mechanical Interface	X		X		
S10	Electrical Interface	X			X	
S11	Separation	X			X	
S12	Loads	X				
S13	Thermal environment	X				
S14	Launch window	X				
S15	Ground segment compatibility	X			X	
S16	Electromagnetic compatibility	X			X	

S1 - Verify by inspection and analysis that the S/C is manufactured with compatible materials in such a manner that it is capable of sustaining storage without degrading or changing its performance.

S2 - Verify by inspection that the S/C attachment points are properly positioned to allow the necessary functions during integration, testing, handling, and transportation. Attachments specificaly designed to receive the solar panel covers shall be inspected.

S3 - The overal S/C reliability shall be verified by analysis of the individual equipments reliability figures.

S4 - Service life prediction shall be effected by dynamic analysis and reliability study results.

S5 - Pre-launch tests shall be performed in the Alcântara Launching Center (CLA) to ensure the readiness of the S/C before launch. These tests are defined in A-ETC -0104.

S6 - Satellite/Launcher compatibility shall be assertained by analysis based on tests S7 through S16.

S7 - Mass properties will be measured at equipment and satellite levels. Prior to the satellite weight, it will be mounted on a balance schenck machine and the unbalance will be measured. If required the S/C will be dynamically balanced by placing weights at each of the two planes perpendicular to the z (spin) axis so that the imbalance is below the specified value. A c.g. machine will be used to locate the S/C center of gravity. An oscillating table will be used to measure the S/C moments of inertia around the three axes.

S8 - Size measurements shall be performed with the S/C assembled in the integration dolly by using measuring devices.

S9 - A fit check will be performed with the flight hardware of the launcher mechanical adapter and clamp-band to verify the dimensional compatibility. The requirements of A-EIF-0001 must be verified.

S10 - The verification of the satellite/VLS electrical interface compatibility shall be done concomitantily with S9. This test shall consist in verifying the continuity and the proper identification of all umbilical connections. Micro-switch operation will also be verified.

S11 - A separation test shall be performed by activation of the separation system using a clamp-band in flight configuration. The S/C is suspended under a crane by its lifting points to allow the adapter to fall free after separation.

S12 - The S/C vibration test results shall be used to complement the satellite/VLS coupled load analysis in order to determine the S/C compliance to the limit loads specified to occur during the launch.

S13 - The S/C thermal test results shall be used to assertain the S/C readiness to withstand the pre-launch and launch phases thermal environments.

S14 - All possible launch windows of the S/C shall be calculated and communicated to the launcher agency.

S15 - The ground segment compatibily tests are depicted in the document TBD.

S16 - Electromagnetic compatibility and interference tests shall be performed in the S/C identification model according with the procedures of A-ECE-0001.