



BSI Standards Publication

Space Engineering — Thermal design handbook

Part 15: Existing Satellites

National foreword

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Space Engineering - Thermal design handbook - Part 15: Existing Satellites

Ingénierie spatiale - Manuel de conception thermique -
Partie 15 : Véhicules spatiaux existants

Raumfahrttechnik - Handbuch für thermisches Design -
Teil 15: Existierende Satelliten

This Technical Report was approved by CEN on 28 June 2021. It has been drawn up by the Technical Committee CEN/CLC/JTC 5.

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European Foreword

This document (CEN/CLC/TR 17603-31-15:2021) has been prepared by Technical Committee CEN/CLC/JTC 5 "Space", the secretariat of which is held by DIN.

It is highlighted that this technical report does not contain any requirement but only collection of data or descriptions and guidelines about how to organize and perform the work in support of EN 16603-31.

This Technical report (TR 17603-31-15:2021) originates from ECSS-E-HB-31-01 Part 15A.

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This document has been developed to cover specifically space systems and has therefore precedence over any TR covering the same scope but with a wider domain of applicability (e.g.: aerospace).

1

Scope

In this Part 15, existing satellites are described and examined from a thermal control and design view. The thermal control requirements are given and an assessment is made of the thermal control systems used against performance for each satellite.

The Thermal design handbook is published in 16 Parts

TR 17603-31-01	Thermal design handbook – Part 1: View factors
TR 17603-31-02	Thermal design handbook – Part 2: Holes, Grooves and Cavities
TR 17603-31-03	Thermal design handbook – Part 3: Spacecraft Surface Temperature
TR 17603-31-04	Thermal design handbook – Part 4: Conductive Heat Transfer
TR 17603-31-05	Thermal design handbook – Part 5: Structural Materials: Metallic and Composite
TR 17603-31-06	Thermal design handbook – Part 6: Thermal Control Surfaces
TR 17603-31-07	Thermal design handbook – Part 7: Insulations
TR 17603-31-08	Thermal design handbook – Part 8: Heat Pipes
TR 17603-31-09	Thermal design handbook – Part 9: Radiators
TR 17603-31-10	Thermal design handbook – Part 10: Phase – Change Capacitors
TR 17603-31-11	Thermal design handbook – Part 11: Electrical Heating
TR 17603-31-12	Thermal design handbook – Part 12: Louvers
TR 17603-31-13	Thermal design handbook – Part 13: Fluid Loops
TR 17603-31-14	Thermal design handbook – Part 14: Cryogenic Cooling
TR 17603-31-15	Thermal design handbook – Part 15: Existing Satellites
TR 17603-31-16	Thermal design handbook – Part 16: Thermal Protection System

2 References

EN Reference	Reference in text	Title
EN 16601-00-01	ECSS-S-ST-00-01	ECSS System - Glossary of terms
TR 17603-31-03	ECSS-E-HB-31-01 Part 3	Thermal design handbook – Part 3: Spacecraft Surface Temperature
TR 17603-31-05	ECSS-E-HB-31-01 Part 5	Thermal design handbook – Part 5: Structural Materials: Metallic and Composite
TR 17603-31-06	ECSS-E-HB-31-01 Part 6	Thermal design handbook – Part 6: Thermal Control Surfaces
TR 17603-31-07	ECSS-E-HB-31-01 Part 7	Thermal design handbook – Part 7: Insulations
TR 17603-31-08	ECSS-E-HB-31-01 Part 8	Thermal design handbook – Part 8: Heat Pipes
TR 17603-31-09	ECSS-E-HB-31-01 Part 9	Thermal design handbook – Part 9: Radiators
TR 17603-31-11	ECSS-E-HB-31-01 Part 11	Thermal design handbook – Part 11: Electrical Heating
TR 17603-31-12	ECSS-E-HB-31-01 Part 12	Thermal design handbook – Part 12: Louvers
TR 17603-31-13	ECSS-E-HB-31-01 Part 13	Thermal design handbook – Part 13: Fluid Loops
TR 17603-31-14	ECSS-E-HB-31-01 Part 14	Thermal design handbook – Part 14: Cryogenic Cooling

All other references made to publications in this Part are listed, alphabetically, in the **Bibliography**.

Terms, definitions and symbols

3.1 Terms and definitions

For the purpose of this Standard, the terms and definitions given in ECSS-S-ST-00-01 apply.

3.2 Abbreviated terms

The following abbreviated terms are defined and used within this Standard.

ABM	apogee boost motor
ACM	acquisition camera module
ACS	attitude control system Clause 8: attitude control sensors
AFNOR	(association Francaise de normalisation), French standards association.
AIT	assembly, integration and testing
AMI	active microwave instrument
AOCS	attitude and orbit control system
AOP	advanced on-board processor
ATSR	along track scanning radiometer
BAPTA	bearing and power transfer assembly
BASD	ball aerospace system division
BCU	bus coupling unit
BD	Clause 8: burst disc Clause 9: (boitier de détection), detector housing
BOL	beginning of life
BSR	back side reflection cell
CCD	coupled charge device

CCHP	constant conductance heat pipe
CEU	control electronic unit
CM	communication module
CNES	(centre national d'etudes spatiales), French space agency
CRU	command relay unit
CSS	coarse sun sensor
CU	(charge utile) payload
d.o.d.	depth of discharge
DAX	Dutch additional experiment
DBS	direct broadcast service
DC	direct current
DET	direct energy transfer
E	equinox
EAIM	attitude measurement and control electronics
ECS	European communication satellite
EGSE	electrical ground support equipment
EIRP	equivalent isotropic radiated power
EM	engineering model
EMA	(electronique de mesure d'attitude), gyro electronics
EOL	end of life
EPC	electronic power conditioner
ERS	European remote sensing satellite
ETU	engineering test unit
FCV	fuel control valve
FDMA	frequency-division multiple access
FES	fine error sensor
FMW	fixed momentum wheel

FPA	focal plane assembly
FSSS	fine sun sensor
GEO	geostationary
GPS	global positioning system
GRM	global reference mission
GSE	ground support equipment.
GSFC	goddard space flight center
GYRS	gyro sensor
HAPS	hydrazine auxiliary propulsion system.
HP	heat pipe
HPA	high power amplifier
HRV	high resolution visible
HV	high voltage
IDTH	instrument data handling and transmission system
IM	instrument module
IR	infrared
IRA	inertial reference assembly
IRAS	infrared astronomical satellite
IRES	infrared earth sensor
IRX	infrared experiment
IUE	international ultraviolet explorer
LRR	laser retro-reflector
LSS	largest space simulation
LTV	low thrust vents
LV	low voltage
MAM	mission adapter module
MCL	magnetic coil
MCS	maritime communication satellite

MCT	main cryogenic tank
MEGS	(mécanisme d'entraînement du générateur solaire) solar array driver system
MLI	multilayer insulation.
MMS	multimission modular spacecraft
MP	(module de propulsion) propulsion module
MS	(module de servitude) service module
MSS	multispectral scanner
MSTH	(modèle structural et thermique), structural and thermal model
OBC	on board computer
OCOE	overall check-out equipment
OSR	optical solar reflector
OTS	orbital test satellite
Pan	panchromatic
PCU	power control unit
PDU	power distribution unit
PEM	payload electronics module
PF	platform
PFM	plateforme multimission
PL	payload
PP	porous plug
PPE	(plateau porte équipements), equipment base plate
PRARE	precision range and range rate equipment
PROPOS	programmable power supply
PSE	power supply electronics
RA	radar altimeter
RCS	reaction control system
RDU	repeater drive unit

RF	radio frequency
RIG	rate integrating gyro
RIU/EU	remote interface unit/expander unit
S	solstice
SAR	synthetic aperture radiation
SCOT	spacecraft component on telescope
SES	solar environment simulator
SHF	super high frequency
SM	service module
SPOT	(satellite probatoire d'observation de la Terre), trial Earth observation satellite (in the beginning) (satellite Pour l'observation de la Terre), Earth observation satellite (current usage) (Dyson (1986) [18])
SS	summer solstice
SSM	second surface mirror
SS-TDMA	satellite-switched time-division multiple access
STD	(senseur terrestre digital), digital Earth sensor
TCA	thrust chamber assembly
TCS	thermal control subsystem
TDH	test data handling
TDRS	tracking and data relay satellite
TIMM	thermal interface mathematical model
TM	thematic mapper
TMCU	(télémessure de la charge utile), payload telemetry system
TT&C	telemetry, tracking and command
TU	tape unit
TVBS	television broadcast system
TWT	travelling wave tube

TWTA	travelling wave tube amplifier
UV	ultra-violet
VCS	vapor cooled shield
VHF	very high frequency
WBM	wide band module
XS	multispectral

3.3 Symbols

Q	heat transfer rate, [W]
T	temperature, [K]
T_m	characteristic temperature, [K]
l_{eff}	effective heat pipe length, [m]
m	mass, [kg]
q	louver heat rejection capability, [W.m ⁻²]
t	time, [d]
t_0	reference time, [d]
α_s	solar absorptance
β	attitude angle of the satellite with respect to Sun line, [angular degrees] it is not defined in a unified way Clauses 9 and 12: angle between the line to the Sun and the longitudinal axis of the satellite Clause 10: angle between the line to the Sun and the normal to orbit plane Clause 11: 90° minus the angle between the line to the Sun and the normal to orbit plane
ε	hemispherical total emittance
σ	standard deviation
τ	characteristic time

International ultraviolet explorer (IUE)

4.1 Mission

Observation of the ultraviolet spectra of astronomical sources in the spectral region between $1,15 \times 10^{-7}$ m and $3,2 \times 10^{-7}$ m.

Launching date: Jan. 26, 1978.

Mission life: 3 years design life with a 5 years design goal. After six year in orbit, a detailed study by NASA shown that the solar panels would support the operation till 1988.

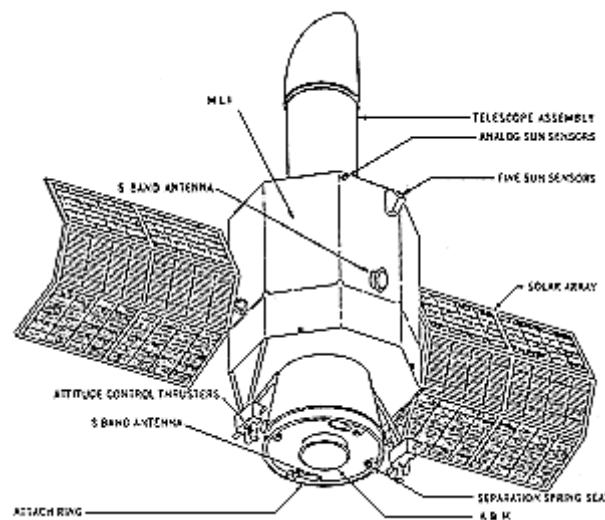


Figure 4-1: IUE spacecraft in orbital flight.

4.2 Main subsystems

IUE carries a 0,45 m diameter Cassegrain telescope, which uses an echelle spectrograph for UV astronomy in the spectral region between $1,15 \times 10^{-7}$ m and $3,2 \times 10^{-7}$ m. Aimed resolution was 10^{-11} m in the mentioned region of the spectrum for stars and planets brighter than 7th visual magnitude, and lower resolution (6×10^{-11} m) for stellar and extended objects as faint as 12th magnitude. Spectroscopy on stars as faint as 18th magnitude has been performed with this instrument.

Relevant characteristics of the different subsystems are summarized in Table 4-1.

Table 4-1: Characteristics of the IUE Main Subsystems

Subsystem	Purpose	Components
Power Direct Energy transfer (DET)	Providing regulated (28 V DC $\pm$ 2%) power which is transferred from the solar array to the spacecraft bus.	<ul style="list-style-type: none"> Two deployable solar panels furnished by ESA. Manufactured by Aérospatiale (France) and AEG-Telefunken (Germany), see Bulloch (1978) [11]. Power supply electronics (PSE) composed of redundant power modules. A power module consists of battery charger, shunt driver, boost regulator, control unit and 12 dump resistors. Mission Adapter Module (MAM) interfacing modules to one another, to solar arrays and to batteries. Two 21,6x10³ C (3,6 A.h) Ni-Cd batteries.
Communication	Transmitting data to ground. Receiving ground-generated commands. Providing range and range rate signals for metric tracking.	<ul style="list-style-type: none"> VHF System used during transfer orbit and for tracking during mission orbit. It consists of two redundant transponders, antenna distribution and a turnstile antenna system. S band downlink system for mission orbit. Two redundant transmitters and four S-band antennae. Active antenna is selectable on ground.
Commend & Data Handling	Accepting and decoding commands from ground or from onboard computer. Encoding spacecraft and scientific instrument telemetry. Performing attitude control computations. Monitoring critical subsystems and controlling exposure times of spectral images.	<ul style="list-style-type: none"> Two redundant command decoders processing messages either from VHF receiver analog signal or from digital information generated by the onboard computer. A Command Relay Unit (CRU). Two data multiplex units. Each one consists of dataplexer, analog subplexer and digital subplexer. Advanced On-board Processor (AOP) computer.
Stabilization & Control	Precessing the spacecraft spin axis 180° in preparation for the apogee engine burn. Attaining the correct orbit. Despinning once the orbit has been achieved.	<ul style="list-style-type: none"> Earth and Sun sensors for ground computer attitude determination. Rate gyros and analog Sun sensors for initial spacecraft acquisition. Inertial Reference Assembly (IRA). Six gas-bearing, pulse rebalanced, rate-integrating gyros.

Subsystem	Purpose	Components
	Controlled pointing and slew of the telescope according to specifications.	<ul style="list-style-type: none"> Redundant two-axis digital Sun sensor system. Redundant set of star trackers within the scientific instruments and utilizing the telescope optics. Redundant reacting wheel. Nutation reaction accelerometers. Apogee Boost Motor (ABM) and Hydrazine Auxiliary Propulsion System (HAPS)

4.3 Main characteristics of the satellite

The main body is octagonal in shape. The telescope extends from the upper end of the main body. It is attached to the spacecraft structure by means of a strong ring resting on three columns which carry the load to the lower spacecraft structure. The columns are supported laterally by truss members of the main body.

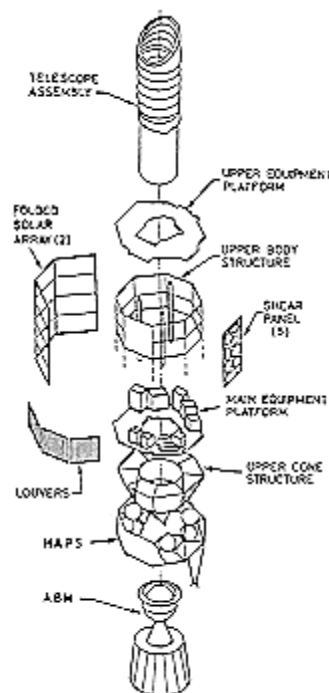


Figure 4-2: Exploded view of the IUE spacecraft.

Two fixed solar arrays extend from opposite sides. Spacecraft orientation is kept in mission orbit such that the front of the solar arrays always faces toward the Sun whereas the thermal control Louvers (see [ECSS-E-HB-31-01 Part 12](#)) face away from the Sun. Spacecraft subsystems are located as follows:

Most of the higher power electronic equipment is located on the main equipment platform, within the main spacecraft body and adjacent to the louvers.

Experiment electronics, attitude control reaction wheels, gyro electronics and Sun sensor electronics are located on the upper equipment platform.

Hydrazine auxiliary propulsion system is located below the main equipment platform.

Apogee boost motor, in the lower cone assembly.

Spacecraft length (in the roll axis direction) is 4,1 m.

IUE was launched on January 26, 1978, by a Delta 2914 booster.

The mass of the IUE Flight Segment is given in Table 4-2.

Table 4-2: IUE Flight Segment Mass Summary

Component	Mass [kg]
Spacecraft	312
Scientific Instruments	122
Apogee motor (218 kg of propellant)	237
Launch Vehicle Adapter	29
Launch Mass	700

Mass reduction was a major problem. More than 50 items were analyzed for possible mass saving during the course of the study, and 21 changes were adopted. Among them:

Orbit redesigned from initially circular synchronous to a lower energy elliptical. 59 kg savings.

Two Acquisition Camera Modules (ACM) plus a Fine Error Sensor (FES) were replaced by two FESs. 7,6 kg savings.

Battery capacity halved.

Electronics of the Inertial Reference Assembly (IRA) lightened through used of hybrids.

4.4 Orbit

Elliptical Geosynchronous (eccentricity: 0,21).

Inclination: 28,6°

Longitude: 71° W

Apogee: 44000 km

Perigee: 27000 km

These data shift with time and are daily issued by the Goddard Space Flight Center (GSFC).

Spacecraft in within continuous view of Goddard and within direct view, for at least 10 h per day, of the European Ground Observatory near Madrid (Spain).

The attitude of the spacecraft is such that

β : between 0° and 135° (Mission orbit)

between 45° and 135° (Transfer orbit)

Here β is the angle between the line from the Sun and the spacecraft roll or telescope axes, with $\beta = 0^\circ$ when the Sun looks up to the bottom of the spacecraft, Figure 4-3.

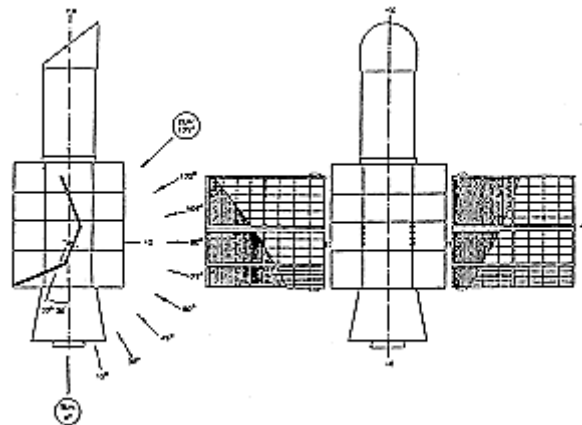


Figure 4-3: IUE orientation to the Sun and reference axes.

4.5 Thermal design requirements

The spacecraft will operate over 135° β angle range and can survive an 85 min zero power eclipse. Total power dissipation is around 186 W of which 130 W are dissipated in the main spacecraft compartment.

Thermal design requirements of the various component and subsystems are summarized in Table 4-3.

Table 4-3: Thermal Design Requirements

Component (Location in Figure 4-4)	Operating Temperature Range [K]	
Auxiliary Propulsion Area (Sta 0-45,5) ABM HAPS	258-311 278-338	Before ignition (50 h after launch). Throughout mission.
Main Spacecraft Compartment (Sta 45,5-87,5) Batteries Gyros	273-313 < 293 $330,5 \pm 1$	Contain own thermal control system.
Spectrograph	273-293	Thermal transients and thermal gradients minimized.
Telescope (Sta 87,5-164,5) Secondary mirror/focus mechanism Primary mirror	273-303 273	
Solar array		No specific requirement.

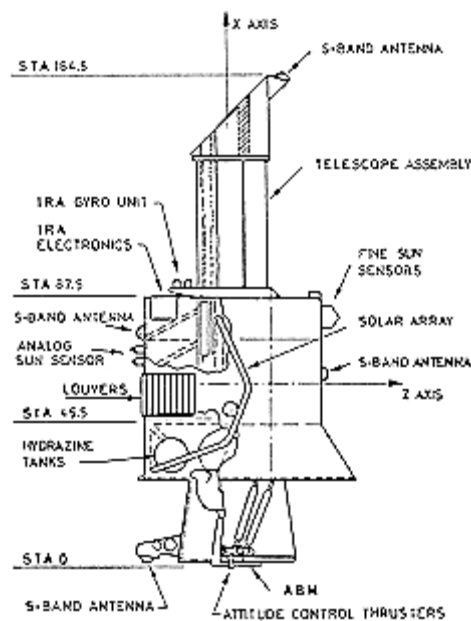


Figure 4-4: Assembled IUE Spacecraft. From Skladany & Seivold (1976) [42]. Notice that this figure, which corresponds to an earlier development, differs from Figure 4-1 in minor details.

4.6 Design tradeoffs

- **Modular vs. Integrated:** Modular design was selected because of the following reasons:
The five major components listed in Table 4-3 are thermally decoupled, each with its own specific thermal problems, and may use separate thermal control.
Majority of electronics can be controlled by a common radiator through a high thermal conductance support.
Since there is not heat transfer among modules, significant in orbit thermal distortions should not result.
- **Passive vs. Semi Active.** All components are passively controlled. Nevertheless, since the power dissipation is not constant and the spacecraft can operate over a wide range of Sun angles, the heat rejection capability does change and, thence, thermal louvers are used for thermal control of the Main Spacecraft Compartment.

4.7 Thermal control of various components

The Main Spacecraft Compartment encloses the spectrograph and the electronic equipment. The spectrograph is thermally decoupled from the compartment. The majority of the electronic equipment is mounted on a honeycomb platform (Figure 4-5) and the remainder on an upper platform.

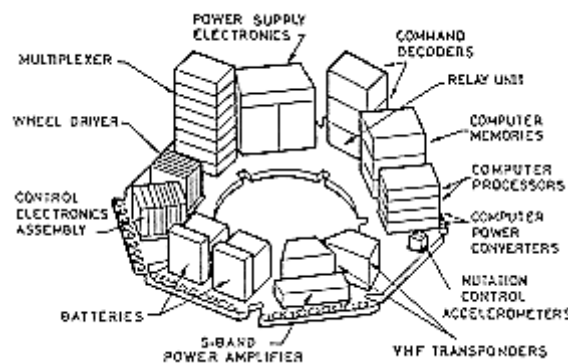


Figure 4-5: IUE main equipment platform. From Skladany & Seivold (1976) [42].

The compartment is covered with an MLI (see [ECSS-E-HB-31-01 Part 7, clause 6](#)) with a silvered Teflon outer layer.

Radiation to outer space is achieved through three sets of nine-bladed louvers located in the Anti-Sun side. Each blade is individually controlled by its own bimetallic spring located within the honeycomb platform as indicated in Figure 4-5 by three groups of nine holes each. When fully open the louvers provide 0,6 m² of radiating area, $q \sim 220 \text{ W.m}^2$ Heat Rejection Capability (see [ECSS-E-HB-31-01 Part 12, clause 6.1.2](#)). The corresponding baseplate temperature is not given (although a rough estimate can be based on data in) Table 4-4). The actuators are calibrated to move from fully closed at 273K to fully open at 283 K. Similar louvers were flown on the SAS-C satellite (Explorer 53, launched on May, 7, 1975).

Two circular, ammonia-filled, grooved Heat Pipes (see [ECSS-E-HB-31-01 Part 8](#)) are attached to the underside of the main equipment platform. The integral Heat Transport Factor of each Heat Pipe is $[Q_{eff}]_{max} = 75 \text{ W.m}$. Temperature differences in the platform are kept below 5 K. These heat pipes are of the type flown on the ATS-6 spacecraft (see Kirkpatrick & Brennan (1975) [32]).

The telescope external surface is covered with an MLI, the outer layer of which is silvered Teflon. The primary mirror is conductively decoupled from the telescope structure and held at 273 K with 3 to 8 W Heaters (see [ECSS-E-HB-31-01 Part 11](#)). Another heater (1,5 W) is used to maintain the secondary mirror at 273 K. The secondary mirror support is not insulated although it is conductively decoupled from the Sun shade.

Radiative decoupling of the spectrograph from the main compartment is achieved by utilizing low emittance surfaces. The spectrograph temperatures are maintained within satisfactory limits by controlling the primary mirror temperature.

The thermal control of the HAPS is achieved by radiation only. The canted side of HAPS bay is covered by an SSM consisting of vapor-deposited aluminium on a Kapton layer with SiO-SiO_x overcoating. This surface ($\alpha_s = 0,25$, $\varepsilon = 0,23$) provides a solar input of 20 W at $\beta = 135^\circ$. The sides of the HAPS bay are covered with an MLI with a vapor-deposited aluminium outer layer. The remainder of the propulsion area is covered with MLI with a black outer layer. Radiative decoupling from the Main Spacecraft Compartment is achieved by an MLI.

ABM is covered with an MLI to prevent heat transfer to the Main Spacecraft Compartment during firing, and to the outer space during orbital flight. A heat shield covers the nozzle exit before firing.

4.8 Estimated on orbit performance

Three different thermal models were used to predict on orbit performance. One of them (165 nodes) for the scientific instruments, other (186 nodes) for the remainder of the spacecraft, and a third more detailed model (250 nodes) for the Propulsion Area.

Nominal values of α_s were taken for all β angles except in the case $\beta = 67,5^\circ$ where a degraded, conservative, value was used. Reflections and shadowing were taken into account.

An Engineering Test Unit (ETU) was constructed for a thermal balance test in the Solar Environment Simulator, (SES) located at Goddard Space Flight Center.

ETU was thermally similar to the flight unit except for the following, nor correctly modelled, items:

Sensors and antennas penetrations through MLIs, except for the S-band antenna bracket on the telescope Sun shade.

HAPS, not truly modelled, and HAPS thrusters, not included.

Burned-out apogee motor, discarded because it was a source of contamination.

Solar radiation was simulated by electric heaters supported on $1,27 \times 10^{-6}$ m thick Kapton sheets (see [ECSS-E-HB-31-01 Part 11, clause 4.2](#)) which were attached to the irradiated surfaces. Solar arrays, apogee motor nozzle and electronic equipment were simulated in a similar way. The exception was a live VHF transponder assembly on the main platform.

Heat loss to the mounting fixture was minimized by use of an insulated zero heat transfer unit automatically monitored to maintain less than 1 K temperature between the spacecraft interface and the adapter ring.

Outer space conditions were simulated with an LN₂ cold wall sink and a vacuum of less than $1,33 \times 10^{-3}$ Pa.

350 thermocouples and auxiliary equipment were used.

Several operating cases at each β angle were run. In addition, 85 min shadow runs were accomplished after reaching equilibrium at $\beta = 0^\circ$ and $\beta = 135^\circ$.

A summary of the results are given in Table 4-4. A more complete set of data can be found in Skladany & Seivold (1976) [42].

Table 4-4: Estimated and Measured Performance of Spacecraft Components and Scientific Instrument Components with Nominal Power Dissipation.

COMPONENT	$\beta = 0^\circ$		$\beta = 90^\circ$		$\beta = 135^\circ$		$\beta = 67,5^\circ$	
	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]
Command Subsystem	288,3	286,6	291,3	290,7	289,6	289,8	295,8	293,1
Multiplexer Subsystem	288,8	286,2	291,2	289,8	289,4	288,3	295,5	292,0
Computer (Processor)	294,8	296,3	300,3	300,0	299,5	299,3	304,5	302,3
Computer (Memories)	290,1	285,7	292,4	289,7	291,4	288,8	296,8	292,0
S-band Power Amp. (Base)	301,6	301,2	303,6	301,8	301,2	302,2	307,3	303,3
Batteries	285,2	285,9	287,4	287,4	281,6	286,8	291,3	289,0
Control Electronics	283,5	283,2	285,6	285,1	280,6	284,3	289,6	286,5
Power Supply Electronics	290,6	288,6	293,3	292,5	292,1	291,3	297,6	294,8
Camera Electronics	289,1	285,0	291,7	288,7	289,8	288,4	295,7	290,4
Sun Sensor Electronics	283,8	281,2	286,7	284,5	284,3	284,2	290,8	286,3

COMPONENT	$\beta = 0^\circ$		$\beta = 90^\circ$		$\beta = 135^\circ$		$\beta = 67,5^\circ$	
	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]
Experiment Electronics	291,9	290,2	294,9	295,3	293,0	295,3	299,2	297,6
Pitch Reaction Wheel	285,2	284,5	288,0	289,2	284,0	288,7	292,2	290,8
Roll Reaction Wheel	285,4	282,6	288,4	287,6	284,3	287,2	292,9	289,4
Yaw Reaction Wheel	285,4	284,8	288,4	288,9	286,0	288,5	292,8	290,6
Redundant Wheel	285,2	278,9	288,3	284,5	284,0	283,9	292,7	286,6
Upper Platform	283,2	~ 280,0	287,2	285,2	283,2	285,2	293,2	288,2
Main Platform	288,2	~ 285,7	292,2	289,2	287,2	288,2	295,2	291,2
HAPS Tank (Facet D)	284,8	289,6	276,6	275,4	268,8	273,5	288,9	286,9
HAPS Tank (Facet B)	289,9	289,0	286,4	286,7	279,3	285,7	297,7	296,5
Telescope Tube (+z STA 133)	202,2	209,6	206,2	216,3	(a)	215,8	(a)	216,7
Telescope Tube (-z STA 133)	205,2	212,3	209,2	218,6	(a)	218,3	(a)	218,6
Primary Mirror	274,2	276,8	277,2	277,3	(a)	275,7	(a)	274,3

COMPONENT	$\beta = 0^\circ$		$\beta = 90^\circ$		$\beta = 135^\circ$		$\beta = 67,5^\circ$	
	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]	Estimated Temp [K]	Measured Temp [K]
Secondary Mirror	(b)	292,9	(b)	294,7	(a)	296,7	(a)	296,9
L.W. Camera 1	281,2 (c)	281,2	283,2 (c)	284,5	(a)	285,4	(a)	284,4
S.W. Camera 1	279,2 (c)	280,6	281,2 (c)	283,9	(a)	284,4	(a)	283,7

^a Test conditions were not analyzed.

^b Test conditions differed from analyzed design.

^c Analytical Model gives component average temperature. thermocouples located to give specific information.

References: Kirkpatrick & Brennan (1975) [32], Skladany & Seivold (1976) [42], Bulloch (1978) [11], Freeman & Longanecker (1979) [22]. All the figures, unless otherwise stated, are from Freeman & Longanecker (1979) [22].

5

Orbital test satellite (OTS)

5.1 Mission

Validating the telecommunication technologies and spacecraft hardware to be used in the European Communication Satellite (ECS) program.

Launching date: May 11, 1978.

Mission life: 3 yr, expandable to 7 for operational variants. End of Service tests were conducted early in 1984, followed by an operation in "hibernation" mode for 18 mo.

First experimental telecommunication spacecraft of ESA.

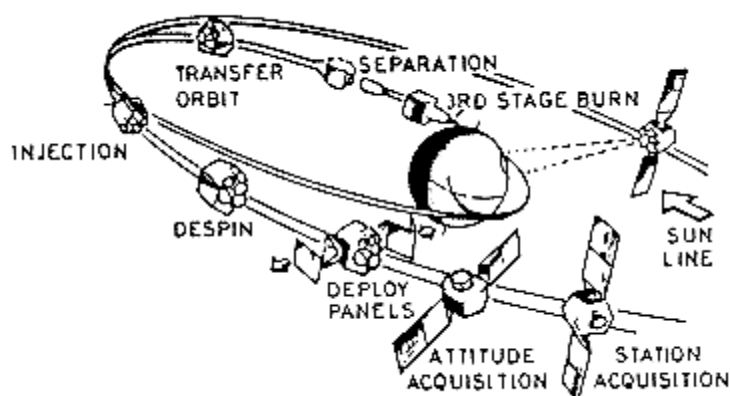


Figure 5-1: OTS mission event sequence. From Collette & Stockwell (1976) [14].

5.2 Main subsystems

OTS consists of a service module (SM) providing all the basic service functions and a communication module (CM) carrying mainly the payload.

Relevant characteristics of the different subsystems are summarized in Table 5-1.

Table 5-1: Characteristics of the OTS main Subsystems

Subsystem	Purpose	Components
Structure	To provide the means of support for the payload and service equipment. To provide the mechanical interface with the launch vehicle.	It forms a hexagonal shaped body with adequate volume on North and South faces to stow the solar arrays in the launch configuration. It consists of two separable Modules. <ul style="list-style-type: none"> The CM, with antennae and repeater units. The SM, with the subsystem equipments, providing functional services to the payload.
Power Generation and Electrical Distribution	To provide for power requirements and electrical interconnection of the various equipment and for protection of the main power bus against failures.	<ul style="list-style-type: none"> Two independently steerable rigid solar arrays. They provide 105 W at transfer orbit. 750 W at Beginning of life (BOL) on station. 600 W at 5 years on station. Two Bearing and Power Transfer Assemblies (BAPTA's) to align continuously the solar arrays to the Sun. A digital shunt regulator system to regulate the main power bus voltage (50 V DC $\pm$ 1%). A 28 cell 21 Ah Ni-Cd battery, that provides 122 W at end of life of the OTS. It is charged via redundant series charges.
Telemetry, Tracking and Telecommands (TT & C)	To receive incoming signals from a circular polarized antenna array used during transfer orbit and on station.	<ul style="list-style-type: none"> It operates at VHF prior to normal mode operation and in back-up modes. It operates at SHF, 14,125 GHz for the uplink and 11,575 GHz for the downlink, in normal-mode operation. It is divided into the SM part with VHF antenna and branching unit. VHF transponder. priority select and interface unit. SM decoder and encoder. the CM part with VHF receiver SHF transmitter payload decoder and encoder.
Attitude and	To maintain the electrical	<ul style="list-style-type: none"> Attitude sensors

Subsystem	Purpose	Components
Orbit Control System (AOCS)	<p>bore-sight of the antennae within a half-cone angle of $0,2^\circ$ (3σ) and the satellite yaw error within $\pm 5^\circ$ (3σ) under all orbital conditions.</p> <p>To provide the means of controlling the spacecraft attitude during transfer orbit, drift orbit and synchronous orbit phases.</p>	<ul style="list-style-type: none"> • V-beam Sun sensor. • Infrared pencil-beam Earth Sensors (IRES). • Two-axis infrared sensors. • Measurement and signal processing electronics • Passive nutation damper. • Control electronics. • Rate Integrating Gyro (RIG). • Actuators • Fixed Momentum Wheel (FMW). • Hydrazine thrusters.
Reaction Control System (RCS)	<p>To provide thrust required for:</p> <ul style="list-style-type: none"> • Transfer orbit. • Spin axis orientation. • Drift Orbit. • Earth and spin acquisition, initial stabilization in three axis mode, balance torque during wheel spin up, drift orbit and initial station localization. • Operations on station. • Momentum wheel unloading, disturbance torque removal. 	<ul style="list-style-type: none"> • Storage system 4 spherical pressure vessels. • Distribution Via launching valves, in a fully redundant manner. • Thruster Two groups of thrusters containing eight high level and two low level thrusters, each.
Apogee Boost Motor (ABM)	To impact sufficient velocity increment to the satellite at the apogee of the transfer orbit to inject it into a near-synchronous circular orbit with approximately zero inclination.	Solid-propellant motor is mounted within the central tube of the spacecraft structure.

The communication payload is split into Module A, Module B and the antennae.

Module A contains two chains of 40 MHz nominal bandwidth and two of 120 MHz, the telecommand receivers and the telemetry transmitters. There are two basic sections in this module: the wide-band section and the channelized section. Both utilize linear polarization.

The uplink and downlink frequencies planned for this module are, respectively:

Uplink

Telecommand receiver: 14 125 MHz

40 MHz chains: 14 125 MHz to 14 192,5 MHz

120 MHz chains: 14 242,5 MHz to 14 362,5 MHz

Downlink

40 MHz chains: 11 490 MHz to 11 530 MHz

Telemetry transmitter: 11 575 MHz

120 MHz chains: 11 580 MHz to 11 700 MHz

Module B includes repeater chains of 5 MHz nominal bandwidth with a gain substantially higher than the chains of module A. This module also contains an onboard beacon transmitter. It utilizes circular polarization.

Uplink

5 MHz chains: 14 455 MHz to 14 460 MHz

Downlink

Beacon transmitter: 11 786 MHz

5 MHz chains: 11 792,5 MHz to 11 797,5 MHz

There are six circular reflector antennae with frontal feedhorn supported by four struts, namely: Three Eurobeam A antennae with 26,5 db peak gains

5.3 Main characteristics of the satellite

The power subsystem capability allows to operate a maximum of six Travelling Wave Tubes (TWT's) during Sun illumination and only two during eclipse.

The battery is located as two separate units on opposite side of the spacecraft (for mass distribution) although it is considered electrically as one.

Two platforms supported by a central tube carry the electronic components.

Shunt electronics, which control the solar array output, are mounted on a dedicated radiator area on the North face of the spacecraft.

In orbital flight, a farm of white painted antennae are on the Earth viewing panel, however, the launcher adapter, the Apogee Boost Motor (ABM), the VHF antennae and 12 hydrazine thrusters are positioned facing away from the Earth.

OTS configuration can be seen in Figure 5-2.

OTS Mass Summary

Subsystem	m (kg)
Repeater	42.0
Antenna	14.5
TT & C	21.7
Power	56.5
Solar array	29.7
BAPTA	8.3
Electrical Distribution	18.5
AOCS	38.2
Reaction Control Assembly + Residuals	27.0
Structure	60.5
Thermal Control	10.7
Instrumentation	1.8
Pyrotechnic	3.0
Total dry mass	340.4
Balance mass	8.0
ABM pull	433.5
Hydrazine	45.2
Launch Vehicle Adaptor	35.3
Total at launch	861.4

Note: non-si units are used in this figure

Figure 5-2: Exploded view of the OTS spacecraft. From Bouchez, Howle & Stümpel (1978) [9].

Typical dimensions are:

2,18 m diametral envelope,

2,10 m height,

8,62 m deployed arrays.

OTS was launched on May 11, 1978 by a Delta 3914 booster. The mass before launching was 865 kg, and mass in orbit 444 kg. Mass distribution, before launching, is given in Table 5-2.

Table 5-2: OTS Mass Summary

Subsystem	m [kg]
Repeater	42,0
Antenna	14,5
TT & C	21,7
Power	56,5
Solar array	29,7
BAPTA	8,3
Electrical Distribution	18,5

Subsystem	m [kg]
AOCS	38,2
Reaction Control Assembly + Residuals	27,0
Structure	60,5
Thermal Control	18,7
Instrumentation	1,8
Pyrotechnic	3,0
Total dry mass	340,4
Balance mass	6,0
ABM pull	433,5
Hydrazine	45,2
Launch Vehicle Adaptor	36,3
Total at launch	861,4

NOTE From Wearnouth & McLaurin (1977) [57].

OTS was developed and manufactured by an industrial organization of firms grouped within the MESH consortium (Figure 5-3).

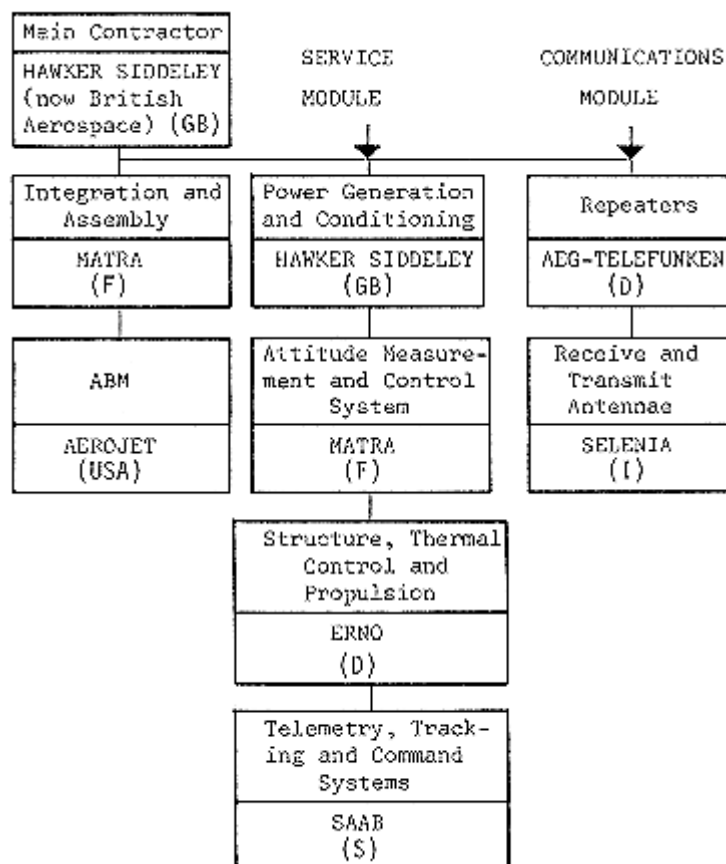


Figure 5-3: OTS main organic diagram. From Collette & Stockwell (1976) [14].

5.4 Orbit

Geostationary (over Gabon). Equatorial.

At first positioned at 10° E longitude. In April 1982 was moved to 5° E to make room for the ECS satellites.

Apogee: 35779 km.

Perigee: 35072 km.

β : the solar angle for the first 14 h after cut-off was about 20° from below (-z side of spacecraft).

During the second half of the transfer orbit, β was close to zero (x-y plane).

On station β varied between 0° and 23,5°.

5.5 Thermal design requirements

The temperature requirements of each unit area are represented in Figure 5-4.

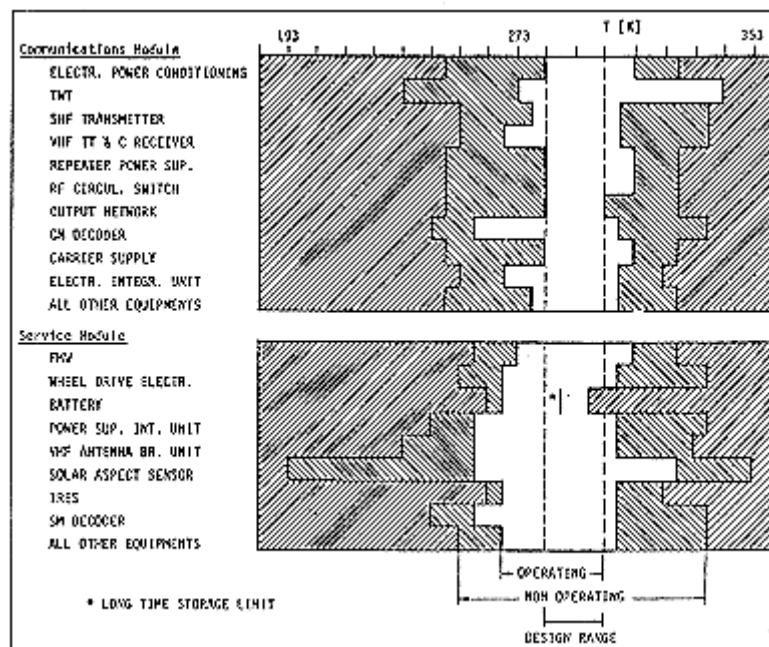


Figure 5-4: OTS Thermal Control Subsystem temperature limits. From Stümpel (1978)a [45].

The major constraints besides those generated by the temperature limits were:

- No active thermal control elements except for RCS heaters of very limited total heating power (≤ 25 W), controlled by telemetry command switches.
- No heating admitted in transfer orbit and on station eclipses.
- Large range between maximum and minimum total unit dissipation (approx. 100 W shunt power variation possible).
- Three-axis stabilization causing extreme sensitivity to polar input.
- Maximum eclipse of 72 minutes.

The RCS configuration imposed severe requirements on the thermal control, for example:

- The lowest temperature of any part in contact with hydrazine should be above 277 K.

- The highest temperature of the Fuel Control Valve (FCV) under all operational conditions of the thruster (preheated, firing, stand by) and under daily and seasonal extremes should be below 328 K (and 348 K for short time peaks).
- The Thrust Chamber Assembly (TCA) should be preheated for firing to ≥ 463 K, for the 0,5 N thrusters, and to 363 K, for the 2,0 N thrusters.
- FCV heater power minimized.
- Thermostats to automatically operate heaters were not permitted.
- Telecommand switches minimized (initially 4, finally 9).
- Operation of telecommand switches restricted to seasonal variations and operation of boost heaters prior to eclipses.
- FVC & TCA had to penetrate the insulation of the spacecraft and had to be located in remote corners, poorly coupled to the thermal mass of the spacecraft.

5.6 Design tradeoffs

- Modular vs. integrated. The thermal design of OTS was conceived to achieve the modularity of the spacecraft to the greatest extent. It was, however, constrained by the desire to embark previously developed European units, payload and electronic units (antennae, TWT's, FMW, etc.).
- Passive vs. active: ESA requested a passive thermal design. ESA didn't allow for this spacecraft the use of heat pipes, automatically controlled heaters, or other active or semiactive elements, to ease the thermal design task.

5.7 Thermal control of various components

The main characteristics of the OTS thermal control are summarized in Figure 5-5.

1. TWT RADIATORS

- Located on North-South panels of the CM conductively decoupled from the structure.
- Profiled honeycomb construction.
- Front surface coated with rigid Second Surface Mirrors (SSMs) (see [ECSS-E-HB-31-01 Part 6, clause 5.2.6](#)) and flexible Optical Solar Reflectors (OSRs), rear painted black. Power output of each TWT variable within a few W per tube. Two extra redundant TWT carried on the central radiator of each North and South panel, but only one energized) at any time.

2. CONTROL RADIATOR

- North-South panels of the SM.
- Front Surface coated ad TWT radiator.

3. SHUNT RADIATOR

- Located on South face of SM, conductively decoupled from the structure.
- Solid aluminium plate, profiled with respect to attachment points and heat distribution capability.
- Front surface coated as TWT radiator, rear painted black.

Shunt electronic power dissipation in the range 40 W to 145 W.

4. MULTILAYER INSULATION (MLI)

(see [ECSS-E-HB-31-01 Part 7, clause 6](#)).

4.1. High temperature resistant type to shield spacecraft against heat generated by the ABM and hydrazine thruster.

- Located at upper and lower dome of ABM, inner thrust cone and cylinder, -z plane of the spacecraft, in the vicinity of hydrazine thruster.
- Kapton insulation: 18 crinkled sheets of 0,5 mil (12×10^{-6} m) aluminized Kapton, sandwiched between two 3 mil ($76,2 \times 10^{-6}$ m) aluminized Kapton foils. Launcher adapter, ABM, VHF antennae and 12 hydrazine thrusters pierce the MLIs on the -z plane.

4.2. Mylar insulation.

- Located at $\pm x$ sides, antenna platform, antennae.
- 10 crinkled Mylar sheets of 0,25 mil ($6,35 \times 10^{-6}$ m) aluminized on one side, sandwiched between 2 mil ($50,8 \times 10^{-6}$ m) and 1 mil ($25,4 \times 10^{-6}$ m) Mylar foils, rear side aluminized.
- Where no VHF shield is applied (antennae, antenna platform), an outer layer of 3 mil ($76,2 \times 10^{-6}$ m) Kapton, aluminized rear side, is applied.

4.3. VHF shield.

- Covering all blanketed areas except inner thrust cone, cylinder, ABM rear dome, antennae, antenna platform.
- 30×10^{-6} m aluminium foil, 0,25 mil ($6,25 \times 10^{-6}$ m) clear Kapton foil and 3 mil ($76,2 \times 10^{-6}$ m) Kapton foil, aluminized rear side.
- Grounded to structure by equally spaced straps of 70×10^{-3} m length and 10^{-2} m width.

4.4. Blow-off insulation at ABM.

(see [ECSS-E-HB-31-01 Part 7, Clause 4](#)).

- Closing the exit plane.

Same type as used on antennae.

4.5. Insulation Supporting Grid.

- Pretensioned wires to support the -z insulation at the free areas of the -z framework.

5. HEATERS (see [ECSS-E-HB-31-01 Part 11](#)).

5.1. Travelling Wave Tube Amplifier (TWTA) Simulation Heaters.

- Operated to simulate dissipation of switch off TWT Electronic Power Conditioner (EPC) during on-station sunlight to keep constant the spacecraft internal temperature.
- Bonded onto the radiator or structure of CM adjacent to the TWTA units.

5.2. Line Heater (see Figure 5-7).

- Wrapped on spirals onto the lines.
- Fixed by aluminium tape.
- Prevent hydrazine from freezing.
- Operated by ground command.

5.3. Tank Heater (see Figure 5-8).

- Bonded onto tanks.
- Prevent hydrazine from freezing.
- Operated by ground command.

5.4. Valve Heaters.

- Bonded onto FCV.
- Prevent hydrazine from freezing within the valve.
- Operated by ground command.

5.5. Batteries Heaters.

- Bonded onto batteries.
- Operated by ground command.

6. BRACKETS OR FLANGES OF LIMITED CONDUCTANCE

- Fiber-glass polyimid prepreg stand-offs, brackets, etc., to provide minimum conductive heat exchange between:
 - Antenna dishes - platform.
 - Radiators - structure.
 - BAPTA. - yoke, structure.
 - Batteries - structure.
 - Tanks - structure.
 - Lines - structure.
 - RCA thrusters (see Figure 5-9) - thrusters brackets.
 - ABM - structure.
 - Antenna Platform - support structure.

7. THERMAL CONTROL PAINT- Black paint for spacecraft interior and units

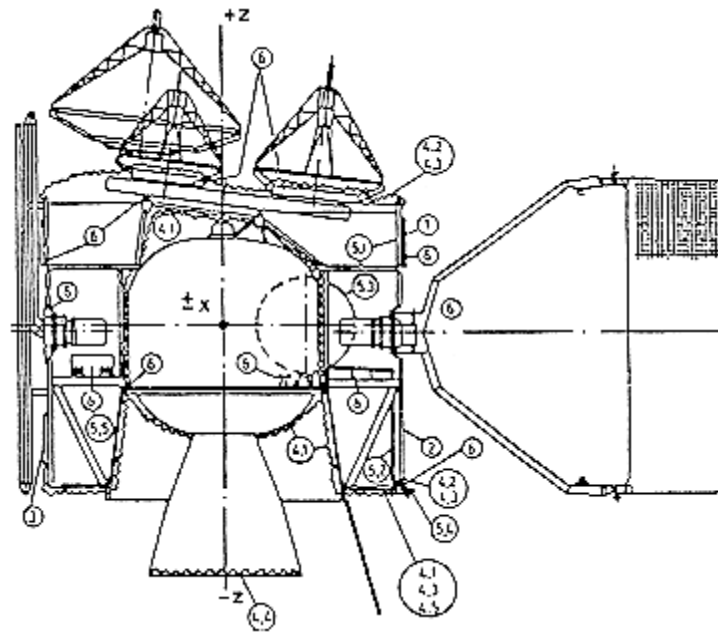
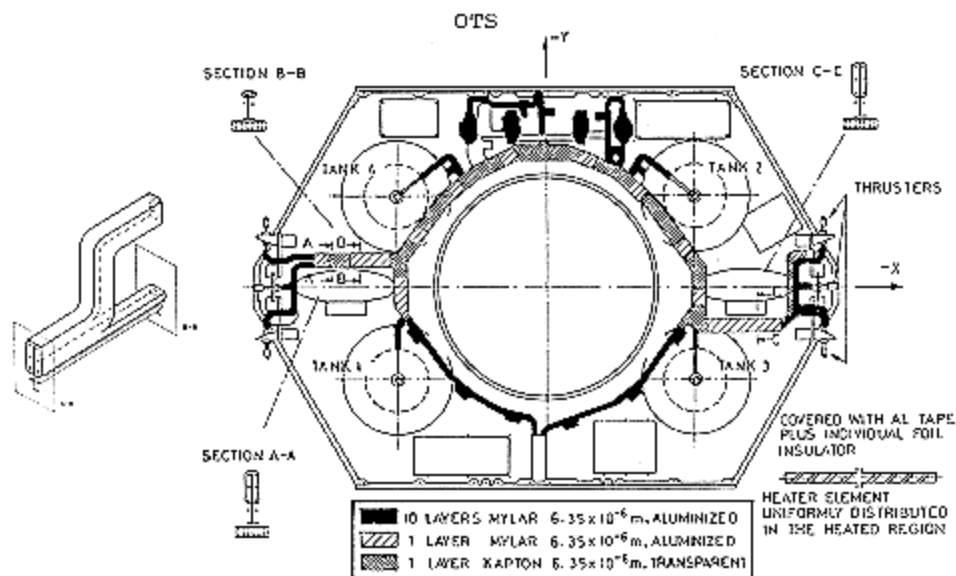


Figure 5-5: OTS thermal control layout summary. From Stümpel (1978)a [45].

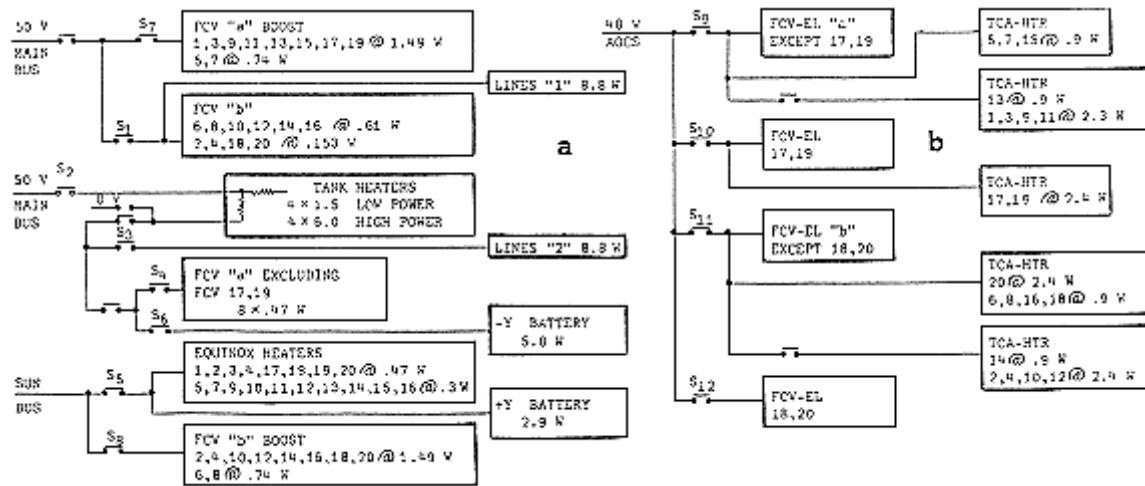
The AOCS presents its own thermal control problems which are not dealt with in Figure 5-5.

Figure 5-6 shows the insulation of the hydrazine line system. Heating to avoid freezing of hydrazine in tanks, lines and valves under widely varying environmental conditions demands for a complex electric circuitry operated by ground command, Figure 5-7. Figure 5-8 shows the heaters of the hydrazine tank and Figure 5-9 the thermal decoupling between the TCA and FCV.



Note: non-si units are used in this figure

Figure 5-6: Insulation in the OTS hydrazine line system. From Stümpel (1978)a [45].



Note: non-si units are used in this figure

Figure 5-7: OTS heater switching diagram.

a) Heating of lines tanks, valves and batteries.

The different heaters are grouped in 8 circuits. Five circuits (corresponding to telecommand switches S1 to S5) are implemented for seasonal operation, one of them (S5) also activates the heater of one battery half. The other battery half has its own separate heater (through switch S6) to limit the temperature differences between the two halves. Two heater circuits (S7 and S8) are reserved for boost heating of the hydrazine components prior to solar eclipse periods which have to be survived without heating.

The telecommand switch in the tank heater circuit (not labelled) allows either the normal heating or the boost heating of the tank.

b) Heating of valves and thrusters of the AOCS (telecommand switches S9 to S12) is also shown.

From Stümpel (1978)b [46] and Bouchez & Gülpen (1980) [5].

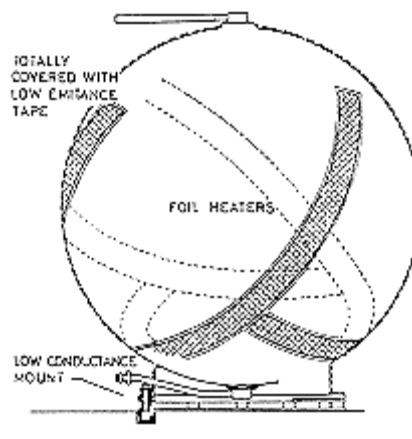


Figure 5-8: Thermal insulation of the hydrazine tank. The tank is totally covered with low emittance tape. Heaters are of the foil type (see [ECSS-E-HB-31-01 Part 11, clause 4.2](#)). The tank contacts the platform via a low conductance amount. From Stümpel (1978)b [46].