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VESTA PROJECT - PHASE Ø REPORT

Perret, A.

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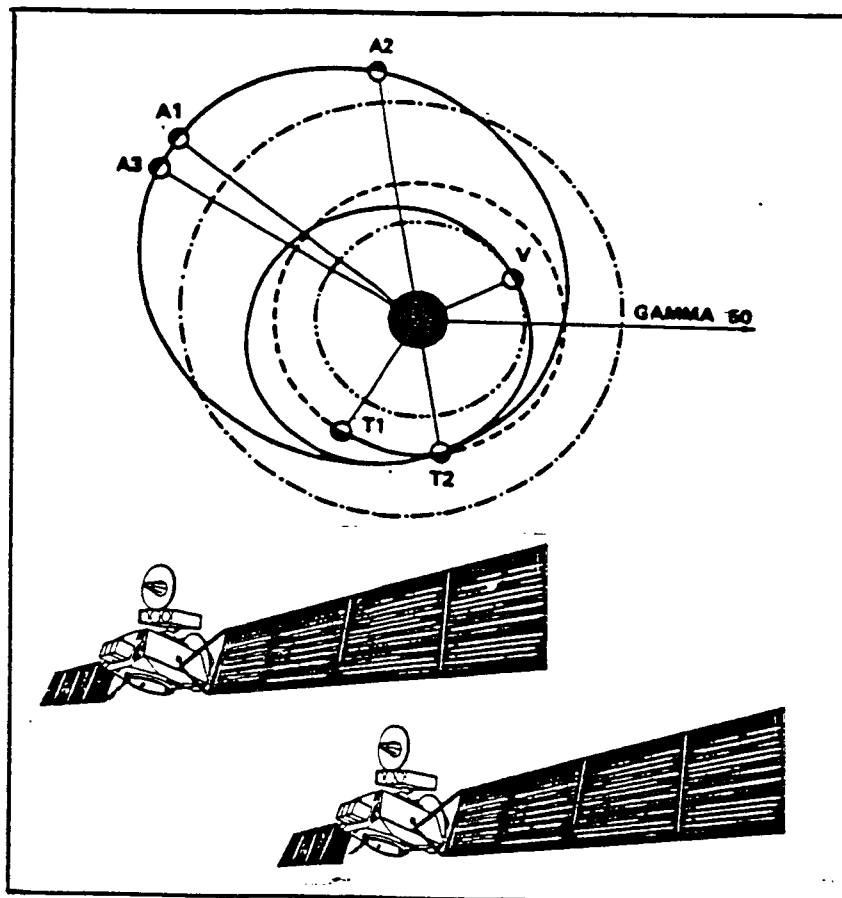
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16. Abstract This report presents the results of technical studies conducted at CNES/PMF/APS between October 1984 and September 1985 on the VESTA project. These preliminary studies were conducted to establish mission feasibility in terms of the trajectory and the scientific objectives, and to determine the type of interfaces which will be required with the Soviets and to measure the magnitude of French participation. A joint French-Soviet report recommends that the project enter phase A, and was approved by CNES and INTERCOSMOS in September, 1985. The mission analysis is made for a mid-1991 launch, and a development schedule for an end-1992 launch is suggested. The decision to postpone the mission was made during the course of the study.			
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CENTRE NATIONAL D'ETUDES SPATIALES  
(CNES)

VESTA PROJECT  
REPORT ON PHASE Ø

by  
A. Perret  
(Project Chief)



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VESTA PROJECT - REPORT ON PHASE Ø

by

A. Perret\*

VESTA Project Chief

INTRODUCTION

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This report presents the results of technical studies conducted at CNES/PMF/APS between October 1984 and September 1985 on the VESTA project.

These preliminary studies were conducted to establish mission feasibility in terms of the trajectory and the scientific objectives, to determine the type of interfaces which will be required with the Soviets and to measure the magnitude of French participation.

During the same period, the Soviets conducted a similar study on their part of the mission, i.e. launching and study of Venus.

A joint report was written to suggest that the project go into phase A, and was approved by the presidents of CNES and INTERCOSMOS in September 1985.

In this study a mission analysis is made for a mid-91 launching whereas a development schedule is suggested for a launching toward the end of 92. The decision to postpone the mission was made during the course of the study. We did not have enough time to calculate new trajectories and to introduce them in this report.

General Characteristics of the Mission

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The VESTA Mission for the study of Venus and small bodies will be launched in December 1992 - January 1993 with a Proton

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\*\*Numbers in the margin indicate pagination in the original text.

launcher over a transfer trajectory to Venus. About 5 tons will be released from the pull of the Earth in a joint launching. It will be divided into three parts: a French satellite dedicated to flying over asteroids or comets of the main belt, a Soviet sphere of re-entry into the atmosphere of Venus and a Soviet spacecraft which after dropping the sphere during the flight over Venus, continues its way to fly over an Appolo-Amor asteroid or to conduct astrophysics experiments. A supplemental spacecraft is under study: a lander or penetrator over one of the asteroids (see figure 1).

The mission will be doubled, i.e. that two launchings will be made a few days apart with identical spacecraft and scientific objectives, but the targets (asteroids or landing sites on Venus) may be different.

Venus will be flown over in mid-1993, the Appolo-Amor asteroid in the middle of 1993 and the asteroids of the main belt in 1994-1996.

#### The VENUS Mission

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In mid-1993, the descent module will separate from the Soviet fly-over vehicle and penetrate into the atmosphere at a speed of 11.6 km/s.

It carries several subvehicles for studying the atmosphere and the surface of Venus (planetary probe, floating probe, landing probe).

#### The APPOLO-AMOR Mission

The Soviet vehicles will continue their trajectory to fly over an asteroid (typically Anteros, diameter of 2.5 km) at a speed of 7 to 8 km/s and at a distance of 1,000 km. An adjustable instrument-carrying platform makes it possible to aim at the asteroid during the fly-over.

Two French satellites for small bodies, which are autonomous as of their launch, rejoin the main asteroid belt (3 UA) after a fly-over with gravitational assistance from Venus then from the Earth. Their objective is to fly over each of the three large asteroids (diam. >60 km) or two asteroids and a cometary nucleus. Accurate targets are not yet identified.

The fly-overs took place at a minimum distance of 500 to 1,000 km for speeds between 4 and 12 km/s. The flight over a cometary nucleus is likely to entail higher values. The scientific payload, with a total mass of about 80 kg, is distributed between an orientable platform (60 kg) and the satellite body. A measurement will be done of the asteroid mass by determining how far a test mass wanders off course.

A payload of the autonomous spacecraft suggested by the Soviet /6 team should be looked at closely. It is an asteroid penetrator. Owing to its characteristics (mass, volume) and the constraints it has on the trajectory and operations, it plays a considerable role in the mission definition. It was not actually included in this report.

## 1 - SCIENTIFIC MISSION

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### 1.1 - Objectives

#### 1.1.1 - Scientific Objectives

-Determining duration and spin axis.

The observation distance must be such that:

- .the ground resolution be 1% of the asteroid diameter;
- .the image of any asteroid be in the CCD matrix;
- .the total observation time covers complete asteroid spin; images are made at regular intervals.



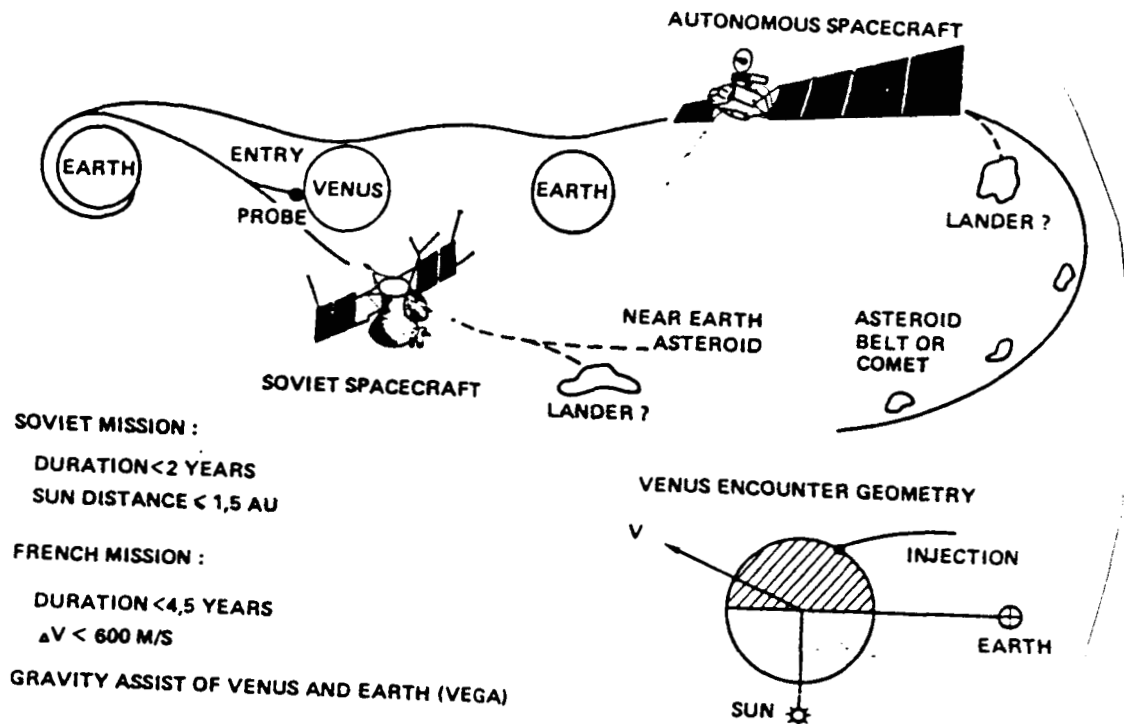


Fig.1 : Mission VESTA

These measurements will be accomplished during the APPROACH and WITHDRAWAL phases.

-Determination of asteroid volume and form.

The observation distance should be such that the ground resolution is  $\sim 1 \text{ km}$ .

These measurements will be accomplished during the APPROACH and WITHDRAWAL phases.

-Mass Measurement:

.either by determining how far a ball ejected from the autonomous spacecraft is off course because of the asteroid;

These measurements will be accomplished during the TEST MASS phase.

.either with an IN SITU measurement of the asteroid gravity using a gravimeter installed on a penetrator;

.or by determining how far the autonomous spacecraft is off course.

-Measurement of the magnetic field:

Based on the assumption of a low magnetic field, these measurements may only be accomplished IN SITU using a magnetometer placed on a penetrator.

-Characteristics and Morphology of the Surface using imagery (possibly stereoscopic imagery) by means of:

.an optical system with a low ( $\approx 10$  m) and average ( $\sim 100$  m) resolution,

.a radioelectric system.

These measurements will be accomplished during the FLY-OVER phase closest to the asteroid.

-Chemical and mineralogical properties

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.either with IN SITU measurements = Instruments located on a landing probe;

.or from an autonomous spacecraft with VIS and I.R. spectrometric measurements.

These measurements will be accomplished during the FLY-OVER phase.

#### 1.1.2 - Technical Objectives

The optical imagery system should provide:

.A trajectory correction prior to the fly-over = asteroid detection in a field of known stars and transmission of the image to the ground.

→NAVIGATION phase

.orienting adjustable platform ≠ possibly orientation correction to asteroid during the APPROACH and WITHDRAWAL phases.

## 1.2 - Scientific Payload Model

### 1.2.1 Payload Organization

One part of the payload is installed on the orientable platform, the other on the autonomous spacecraft:

-Orientable platform:

- .high resolution camera (CHR)
- .large angle camera (CGA)
- .infra-red camera (CIR)
- .ultra-violet spectrometry (SUV)
- .altimeter/radiometer radar (RAR)

-Body of autonomous spacecraft:

- .cosmic dusts,
- .zodiacal light,
- .test masses.

The system is managed by a payload management module (MGCU).

### 1.2.2 High Resolution Camera (CHR)

OBJECTIVE:

This imagery system makes it possible to determine the dimension, form and volume and period of the inclination of the small body spin axis.

During the fly-over phase = surface properties.

During the test mass phase = asteroid mass by monitoring the test mass.

During the navigation phase = besides the scientific objective /8  
tive, the high resolution camera should permit the acquisition  
of small bodies for the trajectory correction phases of the au-  
tonomous spacecraft.

EXPERIMENTAL FEATURES:

- .focus = 1200 mm
- .field of view =  $6^{\circ}8$
- .type of detector = CCD (512 X 512 pixel)
- .angular resolution (altitude 500 km) = 125 m/pixel
- .ground resolution (altitude 500 km) = 125 m/pixel
- .spectral range = 0.3 to 1  $\mu\text{m}$
- .detection of small bodies (type S) =  $D - 1.6 \cdot 10^5 \times \emptyset$  km  
 $\emptyset$  diameter of asteroid

TECHNICAL FEATURES:

- .dimensions 600 X 150 X 150
- .mass 7 kg
- .telemetry 1 MB/image (with compression of data stored in buffer  
memory)
- .power consumption: 25 W
- .CCD cooling: via passive radiator + Peltier effect.

During the navigation phase = Besides scientific objective /9  
the high resolution camera should permit the acquisition of  
small bodies for trajectory correction phases of the autonomous  
spacecraft.

EXPERIMENTAL FEATURES:

- .focal distance = 1200 mm
- .field of view =  $0.6^{\circ}$
- .detector type = CCD (512 X 512 pixel)
- .angular resolution = 2.5 arc sec./pixel)
- .spectral range: 0.3 to 1  $\mu\text{m}$
- .small body detection (type S) =  $D - 6 \cdot 10^5 \times \emptyset$  km
- .ground resolution (altitude 500 km) = 6.25 m/pixel.

Remark: Possible addition of a second defocalized channel for monitoring the test mass.

TECHNICAL FEATURES:

- .dimensions = 1000 X 300
- .mass = 15 kg
- .telemetry = 1 MB/image (with data compression) stored in buffer memory
- .power consumption = 25 W
- .CCD cooling = via passive radiator + Peltier effect.

LOCATION:

- .On orientable platform (PFO)

1.2.3 Wide Angle Camera (CGA)

OBJECTIVE:

During the fly-over phase, it measures the surface characteristics of small bodies.

During the Approach/Withdrawal phase, it corrects the orientation of the adjustable platform.

EXPERIMENTAL FEATURES

- .focal = 105 mm
- .field of view  $6^{\circ}8$
- .detector type = CCD (512 X 512 pixel)
- .angular resolution = 36.25 ar/pixel
- .ground resolution (altitude 500 km) = 125 m/pixel
- .spectral range = 0.3 to 1  $\mu$  = 0.3 to 1  $\mu$ m
- .small body detection (type S) =  $D = 1.6 \cdot 10^5 \lambda \theta^{-1}$  km.  
     $\theta$  asteroid diameter

TECHNICAL FEATURES:

- .dimensions 600 X 150 X 150
- .mass 7 kg
- .telemetry 1 MB/image (with compression of data stored in 1 buffer memory)
- .power consumption: 25 K
- .CCD cooling: passive radiator + Peltier effect.

LOCATION: .On orientable platform (PFO)

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1.2.4 Infra-Red Camera (CIR)

OBJECTIVE

During the fly-over phase, the IR camera is used to determine the surface features = mineralogical and chemical composition ratios.

EXPERIMENTAL FEATURES:

- .Field of view =  $6^{\circ}5$
- .spectral resolution =  $\frac{\lambda}{\Delta\lambda} = 50$
- .angular resolution = 12 arc min/pixel
- .spectral resolution = 1 km/pixel
- .spectral ranges -800 to 1800 nm
  - 1800 to 3600 nm
  - 3600 to 6000 nm
- .detector types = mosaic = 32 X 32 pixel.

TECHNICAL FEATURES

- .dimensions 1000 X 250 X 250
- .weight = 17 kg
- .telemetry = 0.82 MB/image stored in buffer memory
- .mosaic cooling: 80°K = cryogenic helium machine
- .power consumption: .70 W when placed on cold (5 min.)
  - .55 W during observation time.

LOCATION: .On orientable platform (PFO)

#### 1.2.5 - UV Spectrometer (SUV)

OBJECTIVE: the instrument is used

- .to determine the abundance of the coma components,
- .to monitor activity of the nucleus,
- .to measure the albedo of the nucleus and dusts;

when a comet passes close by;

- .to measure the surface albedo to determine an UV cartography and the nonhomogeneities during a flight over asteroids.

#### EXPERIMENTAL FEATURES:

- .Field of view = 5 arc min
- .ground resolution (altitude 500 km) = 100 m
- .spectral resolution = 5 to 10 Å
- .spectral range = 1150 to 3500 Å.

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#### TECHNICAL FEATURES:

- .weight < 10 kg.

LOCATION: .Over orientable platform (PFO).

#### 1.2.6 Altimeter/Radiometer Radar (RAR)

OBJECTIVE:

.ALTIMETER RADAR enables:

- information to be obtained on physical properties, and therefore on the composition of materials on the surface;

- the variation of the thermal flux to be measured of regions located on either side of the terminator;

- a topographical study to be conducted.

.MICROWAVE RADIOMETER:

-study of the surface emissivity and subsurface temperature gradients.

EXPERIMENTAL AND TECHNICAL FEATURES

The conclusions of the Radar group which performed a preliminary assessment of the scientific and technical aspects of the altimetric and radiometric methods are the following:

1 -the altimeter:radar is considered to be the principal instrument

-the microwave radiometer is considered to be the auxiliary instrument

2 - recommendation to following during phase A:

- .a detailed technical appraisal
- .simulation experiments

In order to define:

- .the advantages and drawbacks of the 2 techniques,
- .the frequency selection,
- .the sweeping mode,
- .the technical features of the instrument.

1.2.7 Cosmic Dust Detections

OBJECTIVE

To obtain a more accurate representation of the particle population in the asteroidal belt, and of its vibration in the vicinity of the asteroids and comets.

EXPERIMENTAL FEATURES

.Method: to measure the capacitative impacts with a detector; to record capacitor discharges toward a plane produced by the impact of a particle moving at high speed;



.Mass of dust detected =  $10^{-12} < m < 10^{-8}$  g.

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.detectors with a different dielectric thickness (capton)  
(between 5 and 100  $\mu$ );

.Flight sequence: .during the Earth-Venus path,  
.from a distance  $> u_a$ , steady state operation.

#### TECHNICAL FEATURES

.dimensions = 500 X 200 X 100  
.weight ~ 1 kg  
.electrical power consumption = 0.5 W  
.telemetry = 300 B/day  
.buffer memory for storing information during 2 months (~20 KB).

#### LOCATION

On the body of the autonomous spacecraft, the collection surface should be perpendicular to the speed vector.

#### 1.2.8 Zodiacal Light

#### OBJECTIVES

.Optical study of interplanetary dusts, and particularly  
at the level of the asteroid belt;

.Optical study of cometary dusts;

.Monitoring of asteroid polarization rates with phase angle.

#### EXPERIMENTAL FEATURES

.Field of view =  $2^\circ$ .

Spectral ranges: 300, 440, 550 and 740 nm obtained with a  
mosaic of interferential filters.

. $\Delta\lambda$  = nm.

.Detector = deck of micro channels with 5 anodes; 1 detection chain (photon counting) per anode.

.Orientation =  $\pm 90^\circ$  in 2 directions.

.Operating modes:

- interplanetary mode = 1 cycle of 100 measurements every 5 days;
- asteroid mode - m measurement every 2 seconds.
- calibration mode - a few measurements on a brilliant star.

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#### TECHNICAL FEATURES

.Dimension =

-Detector unit = 450 X 250 X 250;

-Electronics = 200 X 160 X 140.

.Total weight = 7 kg.

.Telemetry:

-interplanetary mode = 662 KB for 2 months;

-asteroid mode = 280 BS.

This information is stored in a buffer memory.

.Electrical power consumption = 8 W;

.Detector cooling - PLETIER Effect.

#### LOCATION

.On spacecraft body; during the flight over an asteroid or a comet, the tracking system will aim at small bodies based on guidance information provided on the orientable platform.

### 1.2.9 Test Mass

#### OBJECTIVE

.To determine the mass of an asteroid by measuring changes in the trajectory of a test mass dropped from the autonomous spacecraft.

#### METHOD

.The trajectory modification may be estimated in several ways:

- either by measuring direction changes with an optical TV instrument;
- or by measuring speed variations = SINGLE PATH DOPPLER;
- or by altimetric measurements of distance variations.

These three methods were the subject of a feasibility study, the preliminary conclusions of which are the following:

.it is possible to measure the asteroid mass with the trajectory of a test mass =

- either using a radioelectric system;
- or using a CHR camera in conjunction with a defocalized observation channel.

.To conclude about the measuring accuracy, it will be necessary to perform observation simulations and to derive from these the feasible accuracy, after the preliminary dimensioning of the instruments.

### 1.2.10 Payload Management Module

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The MGCU will provide the interface between the probe and payload.

- .Airborne power,
- .Ground links = TM and TC

and manage the payload

.Electrical power supply of the payload

.operation sequence of instruments

.storage of information in memory the features of which are the following:

-max capacity = 256 Mbit

-access speed = 600 Kbit/sec.

-consumption = .Watch = 15 W

.in operation (memory access) 50 W

-weight = 36 kg.

.data compression.

### Constraint Associated with Access Speed in Bubble Memory

If we let  $T = 2$  s be the sum of the shooting time, matrix reading, airborne processing, storage in a buffer memory then in a bubble memory,

the time between 2 shots may be less than 2 sec.

for high resolution, wide angle and infra-red cameras.

### 1.3 Experimental Sequence

The experimental sequence is made up of 2 types of sessions =

-monitoring sessions during the Earth-Venus-Earth-Small Body cruise phases;

-scientific sessions during flights over small bodies.

These sessions will be adapted to the mission profile.

#### 1.3.1 Mission Profile

To prepare this experimental sequence, we selected a mission with 2 asteroids described below:

	FLY-BY DATE	ARRIVAL SPEED (KM/S)	DEPARTURE SPEED (KM/S)	FLY-BY ALTITUDE (KM)	TYPE OF DIAMETER (KM)	V. (M/SEC)
EARTH	03/05/91		3,99	/	/	
VÉNUS	20/11/91	5,52	5,50	$8.10^3$	/	36
EARTH	01/10/92	8,93	9,13	$4,5.10^3$	/	3
DÉMOULINA 1335	28/01/94	3,99	4,32	500	TYPE : A.D Ø (ESTIMÉ)= 6	417
LAMBERTA 187	15/08/95	11,90	/	500	TYPE : C Ø 143	

The flight over the DEMOULINA asteroid will enable the /15 PENETRATEUR (PENETRATOR) mission to be accomplished ( $V < 4$  l./sec).

### 1.3.2 Monitoring Sessions

#### OBJECTIVE

.To make sure payload equipment are operating properly throughout the mission: scientific instruments and management moduel;

.Repetition of all fly-over phases during the cruise phase: flight sequence, ground processing.

#### DEFINITION:

#### Instruments Concerned

All instruments are placed in service = the sequence to follow will be defined as a function of the designated objective.

#### Schedule

1 monitoring session approximately every 2 months, i.e. about 15 sessions for this mission.

## .Objectives

EARTH: a few days after launch

→ simulation of withdrawal phase with CHR + A.D. Sequence for other instruments;

VENUS → simulation of fly-over and withdrawal phases (thermal constraints permitting);

EARTH → simulation of approach and fly-over phases.

The other objectives will be defined after analysis of the final mission.

### 1.3.2.a EARTH Session No. 1

.Functional monitoring of all instruments after launch = TO + 6 days (TO = launch), i.e. at a distance of  $2 \cdot 10^6$  km all instruments are powered up and are oriented toward the earth. The high resolution, wide angle and infra-red cameras each produce 6 images at regular time intervals (10 s for example). The other instruments function continuously for 1 mn.

Information is stored on board.

#### .Simulation of spacecraft withdrawal phase

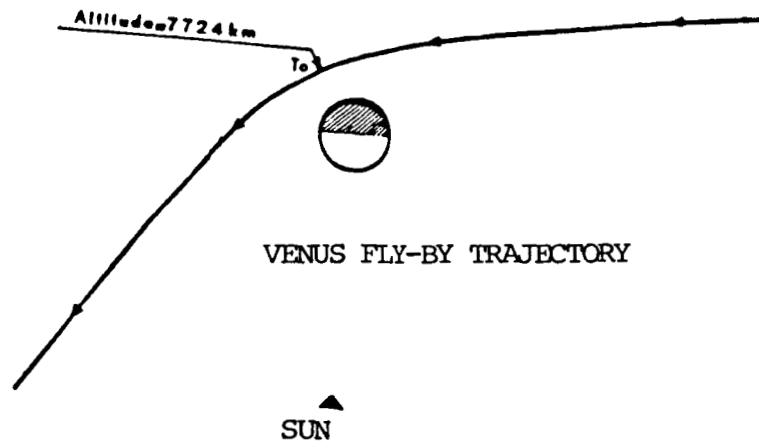
The high resolution camera produces 5 images of the Earth in 20 mn intervals (corresponding to  $5^\circ$  Earth rotation):

Information is stored on board.

.Summary of information to be stored on board, then to be transmitted to the Earth =  $\sim 25$  MB.

### 1.3.2.b VENUS Session

.Simulation of a fly-over phase = at TO +  $\sim 30$  mn, i.e. at a distance of about  $1.5 \cdot 10^4$  km, for an observation period from VENUS of 500 s:



-the high resolution camera produces 10 images (1 image every 50 s); the ground resolution is about 200 m per pixel;

-the wide angle and infra-red cameras produce 3 images each;

-the other instruments operate continuously during this phase;

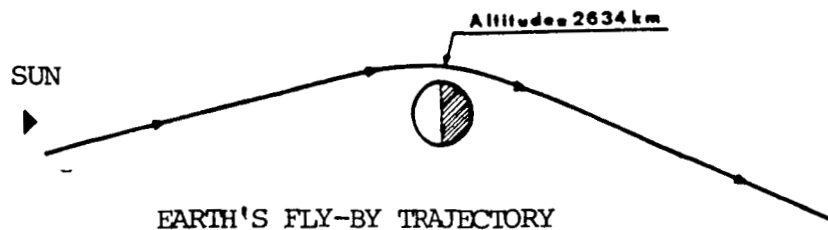
-the information is stored on board.

.Simulation of a spacecraft withdrawal phase =

at  $T_0 + 4$  days, i.e.  $2 \cdot 10^6$  km, the high resolution camera produces 5 images of VENUS at regular intervals of about 10 mn.

.Summary of information to be stored, then to be transmitted to the Earth = ~25 NB

### 1.3.2.c EARTH Session No. 2



.Simulation of an Approach phase =  
at TO - 2.5 days, i.e. at  $2.10^6$  km; the high resolution camera  
produces 5 images of the Earth within 20 mn intervals.

The information is stored on board.

.Simulation of a fly-over phase =  
as of TO - 10 mn, i.e. at  $5.10^3$  km,

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-the high resolution camera produces 10 images (1 image  
every 2 s) of the Earth, with a ground resolution of about 50 m  
per pixel;

-the wide angle and infra-red cameras each produce 3 images  
(1 image every 15 s);

-the other instruments operate continuously during this phase;

-the information is stored on board.

.Summary of information to be stored, then to be transmitted to  
the Earth = ~25 MB.

#### 1.3.2.d. Other Monitoring Sessions

Each session will include:

.Aiming of orientable platform at a celestial object:  
Simulation of a Navigation phase with a high resolution camera;

.A series of taking aim off this initial point = 1 image  
will be produced for each step in this series using high resolu-  
tion, wide angle and infra-red cameras; the other instruments  
will operate continuously.

.Summary of information to be stored, then to be transmitted  
to the Earth = ~ 25 NB per monitoring session.



### 1.3.3 Scientific Sessions

#### 1.3.3.a. Base Data

##### -NAVIGATION PHASES

.The orientable platform pre-oriens the high resolution camera toward the asteroid. The images taken by the camera are then sent to the ground for porcessing for trajectory correction.

##### -TEST MASS PHASE

.The orientable platform pre-oriens the optical axis of the high resolution camera concurrently with the ejection axis of the test mass.

.During this pahse there is no link with the Earth; the information (1 image every hour) is stored on board. /18

##### -APPROACH AND WITHDRAWAL PHASES

.The orientable platform pre-oriens the high resolution camera toward the asteroid; the aim is then corrected by a wide angle camera (or other system).

.The observation distance should be such that =

-to determine the rotation period and axis =  $0.5\% < \text{ground resolution} < \text{asteroid diameter}$ ,

-to determine the volume and form =  $\text{ground resolution} \sim 1 \text{ km}$ .

-the observation time should cover one rotation period of the asteroid,

-no link with ground: information (1 image every 5° asteroid rotation) is stored on board.

-FLY-OVER PHASE

.All instruments are operating and are oriented toward the asteroid. The aim is corrected using an asteroid probe of the orientable platform.

.The high resolution camera produces a maximum number of images, with a 50% overlapping between 2 images.

.The wide angle and infra-red cameras produce 1 asteroid image every 15° of relative rotation.

.Information is stored on board.

-PENETRATOR PHASE

.After the autonomous spacecraft has transmitted the required ballistics, the penetrator is dropped and autonomously orients itself toward the asteroid.

.After, or during, the development of the scientific mission on the asteroid, information is sent and store on board the autonomous spacecraft.

1.3.3.b - The Flight Over the Asteroid DEMOULINA

During this session, there will be no TEST MASS phase; measurements will be performed IN SITU using a gravimeter installed on the penetrator.

-NAVIGATION PHASES

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Two navigation sessions will take place:  
.one at TO - 6 days (i.e. at  $2 \cdot 10^6$  km);  
.the other at TO - 2 days (i.e.  $7 \cdot 10^5$  km).

-PENETRATOR PHASE

The penetrator will be dropped at TO - 2 hours, i.e. at  $3.5 \cdot 10^5$  km.

.At TO + 2 hours, information transmitted by the penetrator will be received and stored on board the autonomous spacecraft.

-APPROACH PHASE

Measurements will be performed at a distance between  $2.4 \cdot 10^3$  and  $7.2 \cdot 10^4$  km. The observation time is about 5 hours. This allows the entire asteroid rotation period to be covered with a ground resolution between 30 M and 900 M.

.During this period 72 images were produced with a high resolution camera, i.e. an information volume of 72 MB to be stored.

-FLY-OVER PHASE

.The observation period lasts 500 sec. (from TO - 250 sec. to TO +250 sec.), i.e. a distance between 1000 km and 500 km.

.The high resolution camera will produce in all 25 images which, with the aiming errors of about  $\pm 15'$ , will enable the entire visible surface of the asteroid to be analyzed with a ground resolution between 6.25 m and 12.5 m per pixel.

.The wide angle and infra-red cameras will each produce 9 images which will enable the entire visible surface of the asteroid to be analyzed.

.The other instruments will operate continuously during this phase.

.The information volume to be stored will be 44 MB.

-WITHDRAWAL PHASE

.Identical to the APPROACH phase.

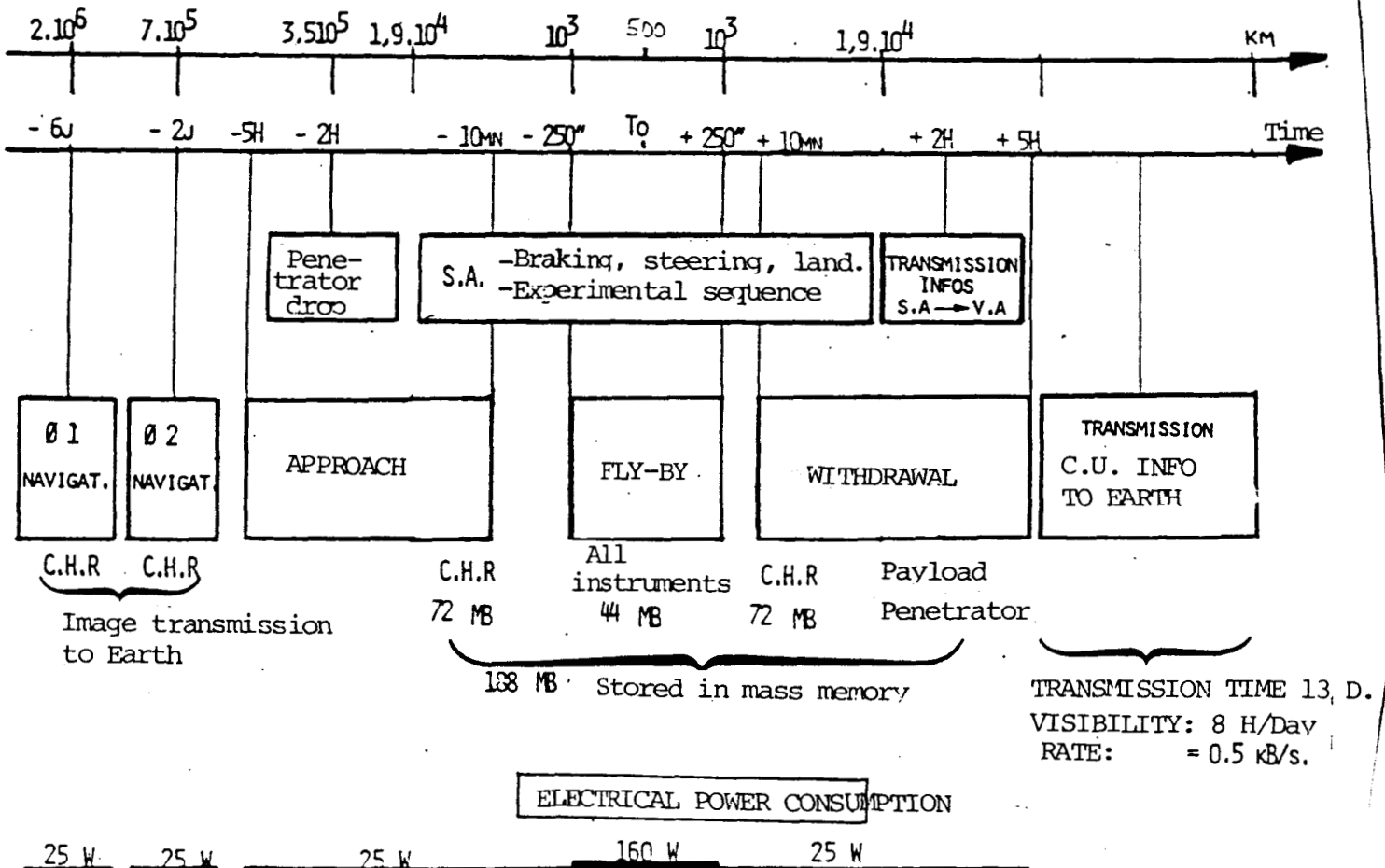
-TRANSMISSION OF INFORMATION TO THE EARTH

/20

During the flight over the asteroid DEMOULINA, 188 MB in all were stored in the bubble memory.

After correcting the trajectory as needed to guide itself to the asteroid LAMBERTA, the high gain antenna will be oriented toward the Earth for data transmission at a rate of 0.5 kB/s, i.e. a transmission duration of 13 days for station coverage of 8 H per day.

-SUMMARY OF FLIGHT OVER THE ASTEROID DEMOULINA



-NAVIGATION PHASES

Two navigation sessions will take place:

- .one at TO - 10 days (i.e. at  $10^7$  km);
- .the other at TO - 4 days (i.e. at  $4 \cdot 10^6$  km).

-TEST MASS PHASE

-Prior to the fly-over, at TO - 48 H, the test mass is ejected and followed by the high resolution camera up to TO - 5H. During this period, the avionics system of the camera calculates the real trajectory of the mass.

-After the fly-over, from TO + 5H to TO + 48H, the high resolution camera is again reoriented to the test mass based on the preceding calculations.

-The information volume is 1680 KB (168 images at the rate of 1 image per hour), based on an airborne calculation allowing only the X, Y coordinates and their range to be stored of images on the CCD of the test mass and celestial objects.

-APPROACH PHASE

-The measurements will be performed at a distance between  $2.4 \cdot 10^5$  and  $2.3 \cdot 10^4$  km. The observation time is about 5H thus allowing coverage of the entire asteroid rotation, with a ground resolution between 290 m and 3000 m.

-During this period 72 images were produced with the high resolution camera, i.e. an information volume of 72 MB to be stored.

-FLY-OVER PHASE

The observation period lasts 360 sec., i.e. a distance between 2000 and 500 km.

The high resolution camera will produce in all 68 images which will allow the continuous band of the asteroid to be analyzed (190 km long and 3.2 to 12.8 km wide, with a ground resolution between 6.25 m and 25 m per pixel.

-The wide angle and infra-red cameras will each make 11 images allowing virtually all of the visible surface of the asteroid to be analyzed.

Other instruments will operate continuously during this phase.

-The total information volume to be stored during this phase /22 will be about 90 MB.

-WITHDRAWAL PHASE = identical to the APPROACH phase.

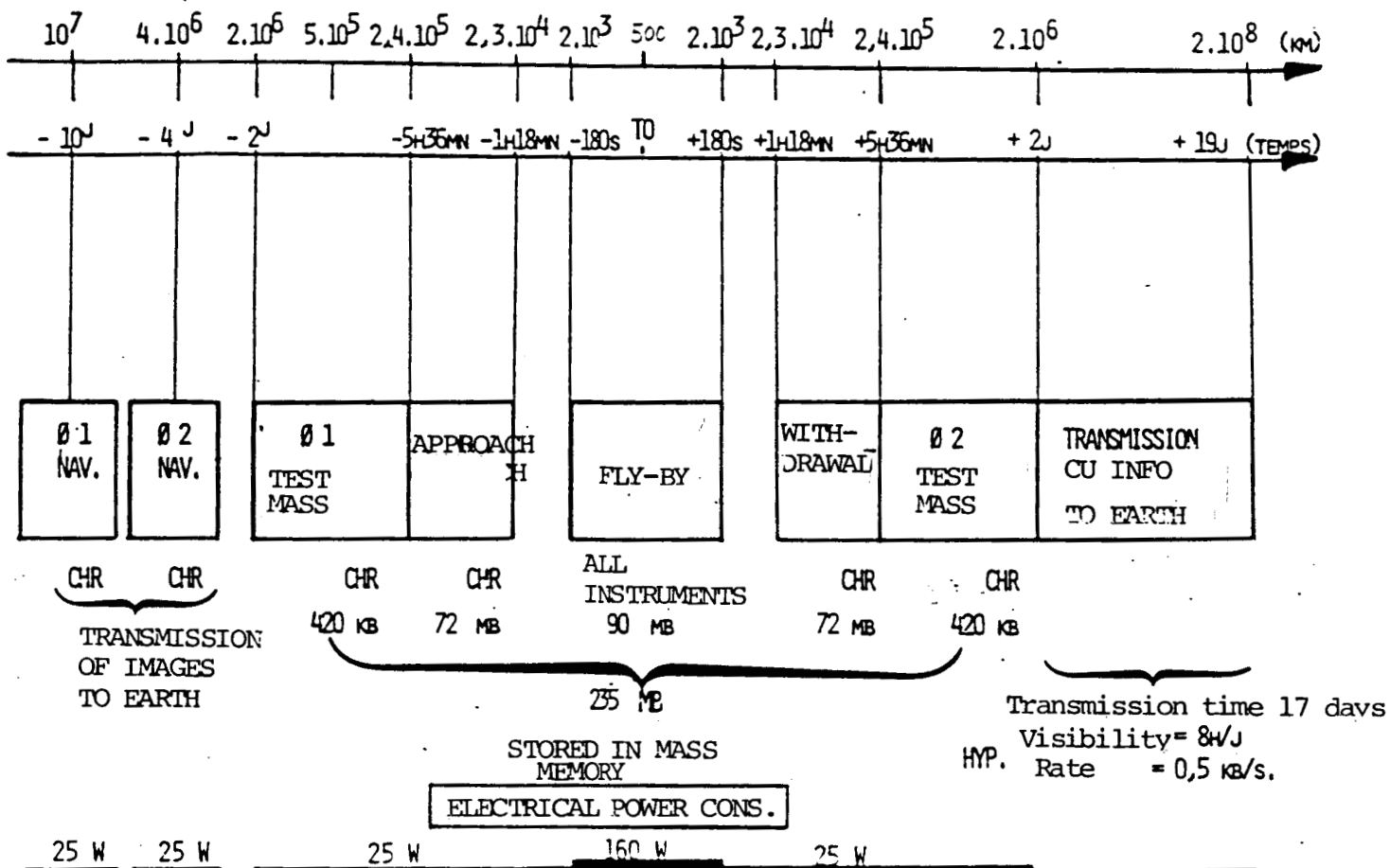
-DATA TRANSMISSION TO THE EARTH

During the flight over the asteroid LAMBERTA, 235 MB (i.e. virtually the entire storage capacity of the bubble memory) were stored.

At the end of the phase of monitoring the test mass, the high gain antenna will be oriented toward the Earth for data transmission at a baud of 0.5 KB/sec., i.e. a transmission time of about 17 days for a station coverage of 8H per day.

-SUMMARY OF FLIGHT OVER THE ASTEROID LAMBERTA

The summary is shown on the next page.



### 1.3.3.d - Experimental Constraints Associated With Bubble Memory Characteristics

#### .Summary of Characteristics

- Access speed = 0.6 MB/sec.
- Capacity = 256 MB.

If we consider the following as constant:

-the information volume required during the Approach and Withdrawal phases =  $\Sigma = 144 \text{ MB}$ ;

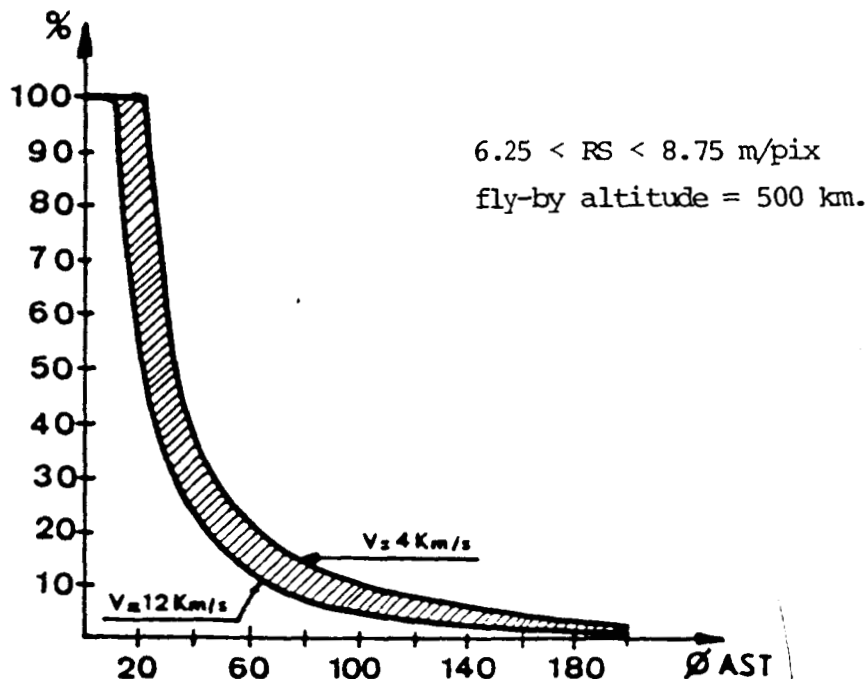
-the information volume required during the Fly-over phase for all instruments except for the high resolution camera =  $\Sigma 30 \text{ MB}$ .  
(Case where all corresponding scientific objectives are reached by

100%), the surface analyzed by the high resolution camera will be limited by:

i - the memory access speed when the observation distance is such that  $6.25 \text{ M} < \text{Ground resolution} < 8.75 \text{ m per pixel}$ .

For example, in the observation conditions defined above, for an asteroid with a 100 km diameter and a fly-over altitude of 500 km, the high resolution camera analyzes a continuous band of 80 km long and 3.2 km (at the nearest) to 4.5 km wide.

The table below shows the percentage of surface analyzed by the high resolution camera as a function of the diameter and of the relative fly-over speed.



ii - The access speed and storage capacity of the memory if one wanted to analyze the surface with a ground resolution  $> 8.75 \text{ m/pixel}$ , therefore a larger surface.

→ This is especially true because the speed is low.



In these conditions there will be a compromise to be found between the scientific objectives of the Fly-over phase, and those of the APPROACH and WITHDRAWAL phases.

## 2. TECHNICAL DESCRIPTION

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### 2.1 Analysis of the Basic Constraints

The basic scientific objective is to fly over several small bodies (Asteroids of the main belt, or Asteroid and Comet) located at a distance of about 3 UA from the sun.

The launching is two-fold; the mission profiles will differ for the 2 vehicles.

The Asteroids will be flown over at an altitude between 500 and 1000 km; the relative speed of the autonomous spacecraft will be between 4 km/s and 12 km/s.

The instruments will be aimed at the small bodies during the Approach, Withdrawal and fly-over phases of an orientable platform whose characteristics will be the following:

- .precision  $\pm 10'$
- .aiming stability
  - for observation =  $3''/s$  (100 Hz to 1 KHz)
  - for navigation =  $10''/s$  ( 1 Hz to 10 Hz)
- .maximum angular speed  $2^\circ/s$
- .maximum angular acceleration  $3^\circ/s^2$
- .fields of view:
  - azimuth =  $360^\circ$
  - site =  $180^\circ$

During the fly-over phase, the aiming should be such that the relative spacecraft speed when brought to the surface of the asteroid is as slow as possible.

Considering the length of the mission, payload monitoring

sessions will be performed during the cruise phase in addition to the scientific sessions during the flight over small bodies.

-Resource allocation for payload:

MASS = .60 kg on orientable platform;  
.25 kg on body of spacecraft.

#### SCIENTIFIC INFORMATION

.scientific sessions = 256 MB at the most;  
.monitoring sessions = 25 MB at the most.

#### ELECTRICAL POWER

.25 W for the Approach/Withdrawal phases and test mass  
.160 W during fly-over phases.

-Thermal stresses on the instruments:

.during operation = - 10°C to + 40°C;  
.during storage = - 30°C to + 50°C.

#### 2.1.2 - Soviet Interface Constraints

/25

These constraints, due to the use of the PROTON launcher, are of two types:

##### Technical Constraints

In addition to cases of static and dynamic mechanical loads which are more severe than those of MUA ARIANE IV, the volume under the nose cone has a cylindrical part ( $\emptyset$  3000) of only 400 (see figure 2.1.2.1).

The separation is accounted for by the Soviet party and may be accomplished with any diameter less than  $\emptyset$  1600.

The maximum mass of the asteroid probe shall not exceed

870 kg for a distance from the center of gravity of less than 1 meter from the separation plane.

#### .Mission Constraints

The launching of the PROTON rocket per se is followed by a ballistic phase lasting at least two hours, which precedes the injection into interplanetary orbit. After separation of the Franco-Soviet composite, the probe is separated in any attitude, with residual speeds of less than  $3^\circ/\text{s}$ .

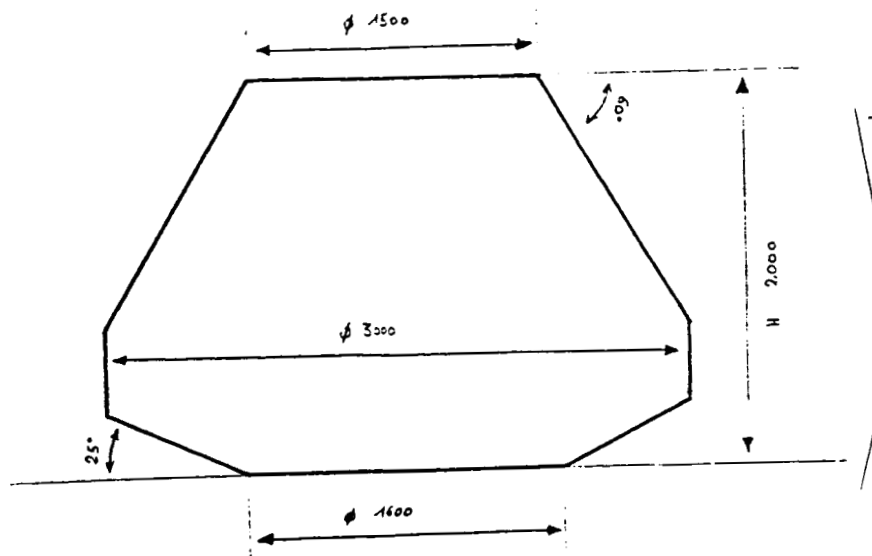
The trajectory upon which the probe is injected is a transfer orbit toward the planet VENUS, which is the first objective of the Soviet spacecraft injected concurrently. The mission's objective is to observe asteroids, and will begin by a long trip to Venus, whose gravitational effect will be used to modify the trajectory. An impulse of about 100 m/s will be given to the periastron so as to be injected into a return orbit to the Earth, where a similar maneuver will occur, allowing the probe to move toward the asteroids.

The Earth-Venus-Earth phase will last about 14 months, and involves specific constraints affecting the following functions:

.Satellite aiming system, because the angle (probe-Earth, probe-sun) varies during this phase between about  $30^\circ$  and  $162^\circ$ , whereas it remains in a much smaller fork during the rest of the mission.

.Thermal control system, because the solar flux received near VENUS is twice as large as on the Earth's orbit.

Orbital maneuvers to be accomplished at the periastrum, which to be efficient must not exceed 30 mn in duration, and should be achievable in any direction.



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Fig 2.1.2.1. Permissible Volume under Nose Cone

## 2.2 Mission Analysis

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### 2.2.1 Description of Constraints

The objective of the Franco-Soviet project is to study Venus and the Asteroids. Two probes will be separated after the launch. The Soviet spacecraft shall land on a specific site of Venus (Aphrodite). The French spacecraft, will use planetary gravitational pull (VEGA principle; Venus - Earth - Gravity assist) enabling multiple rendezvous with the asteroids (or comets).

The constraints are therefore quite different for the two missions.

#### a) Constraints

These are launching constraints.

- Circular parking orbit at an altitude of 170 km, inclination =  $51^{\circ}5$ .

- Greenwich longitude of descending node =  $180^{\circ}$  (on date of first passage at this node).

- Speed of hyperbola at infinity =  $V_{\infty} < 4$  km/s.

b) Specific French Constraints

Venus pericenter radius  $> 6850$  km.

Earth pericenter radius  $> 7400$  km.

Mission duration  $< 5$  years.

Total increment  $\leq 600$  m/s.

c) Specific Soviet Constraints

Entry angle of lander VE  $< 20^\circ$

Landing site =  $.120^\circ \leq \text{longitude} \leq 140^\circ$

$.-20^\circ \leq \text{latitude} \leq +20^\circ$

Speed at infinity  $\leq 6.2$  km/s.

(i.e. entry speed into the atmosphere  $\leq 12$  km/s).

2.2.2 Trajectory Research

This research took place in 3 steps.

a) Earth-Venus-Earth Ballistic Trajectory Research

Sweeping over dates

.from 26.03.91 to 02.07.91 for Earth: 2-day step.

.from 12.09.91 to 21.02.92 for Venus: 1-day step.

.from 04.06.92 to 17.12.92 for Earth: 1-day step.

The three dates selected correspond to:

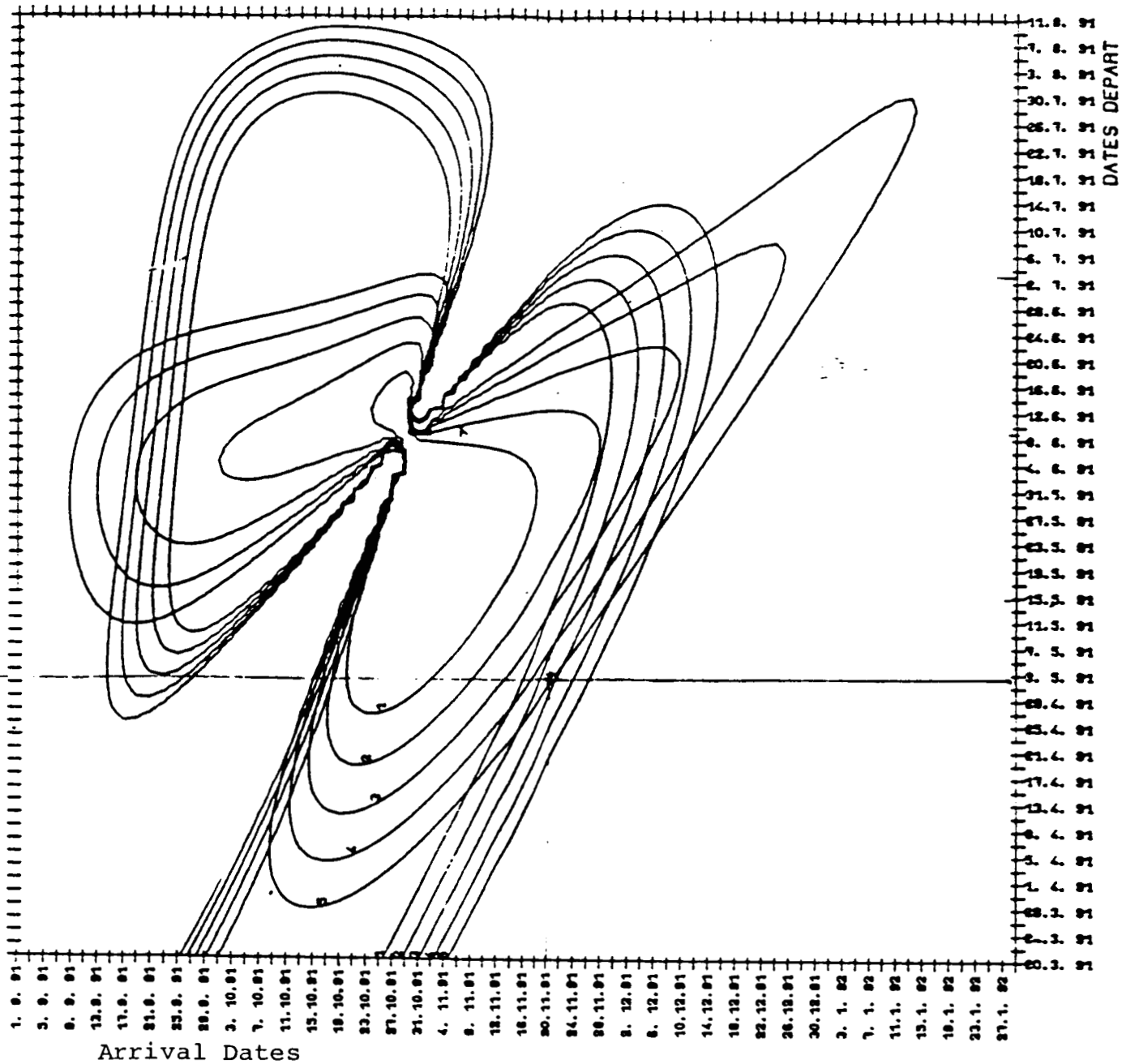
.Infinite speed moduli at Venus of less than 5 km/s

.Maximum aphelion radius (after S.B. from the Earth)  $> 2$  UA.

After processing and analyzing the results it was revealed /31 that a pair of dates (departure from the Earth, arrival at Venus) gave two possible dates for returning to the Earth.

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VESTA 1991 • EARTH-VENUS TRANSER

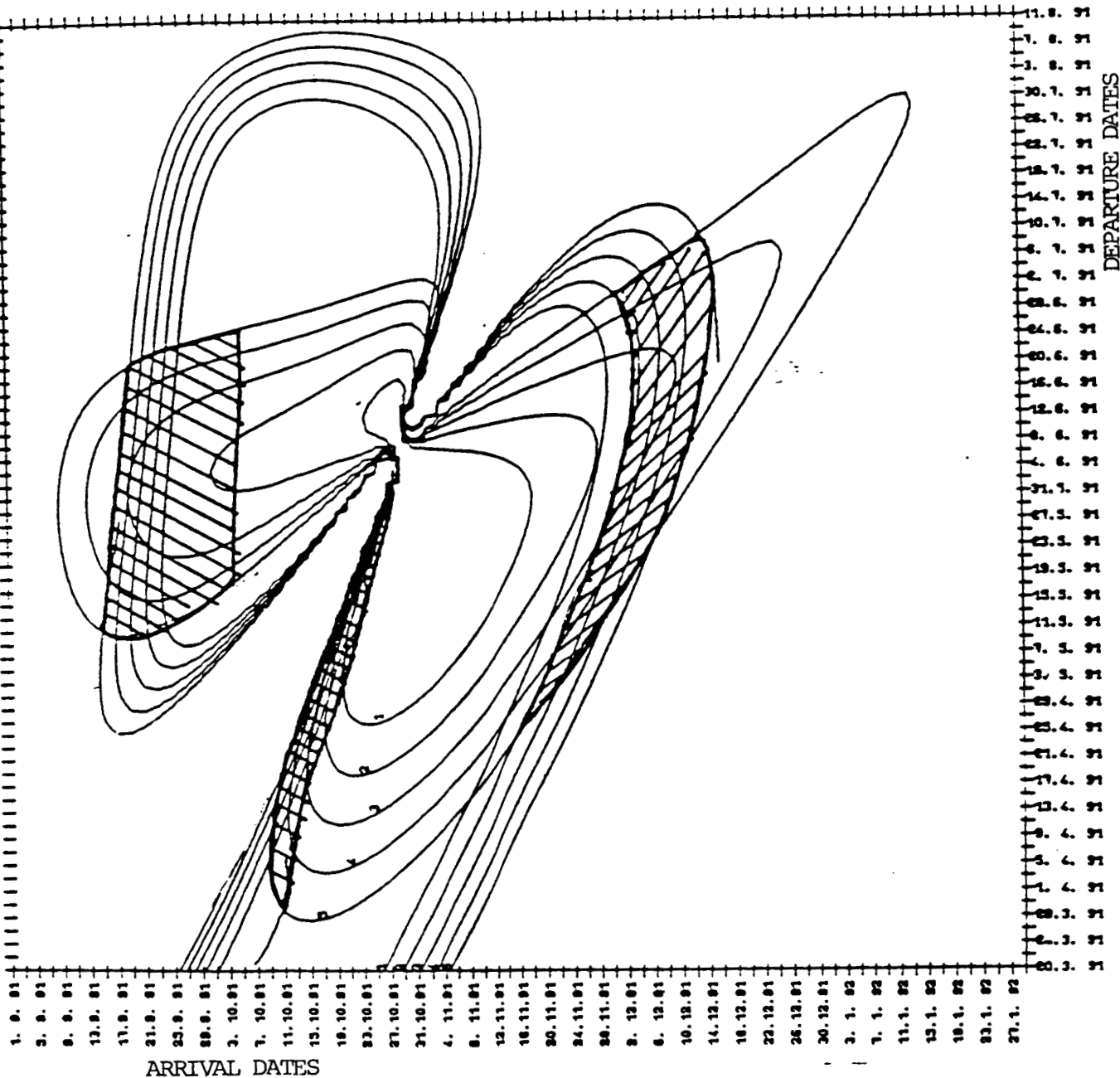


VIT INF DP  
 level 1= 3.0  
 level 2= 3.2  
 level 3= 3.5  
 level 4= 3.8  
 level 5= 4.0

VIT INF AR  
 level 1= 5.0  
 level 2= 5.3  
 level 3= 5.6  
 level 4= 5.9  
 level 5= 6.2

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VESTA 1991 : EARTH-VENUS, EARTH-ASTEROIDS WINDOW TRANSFER

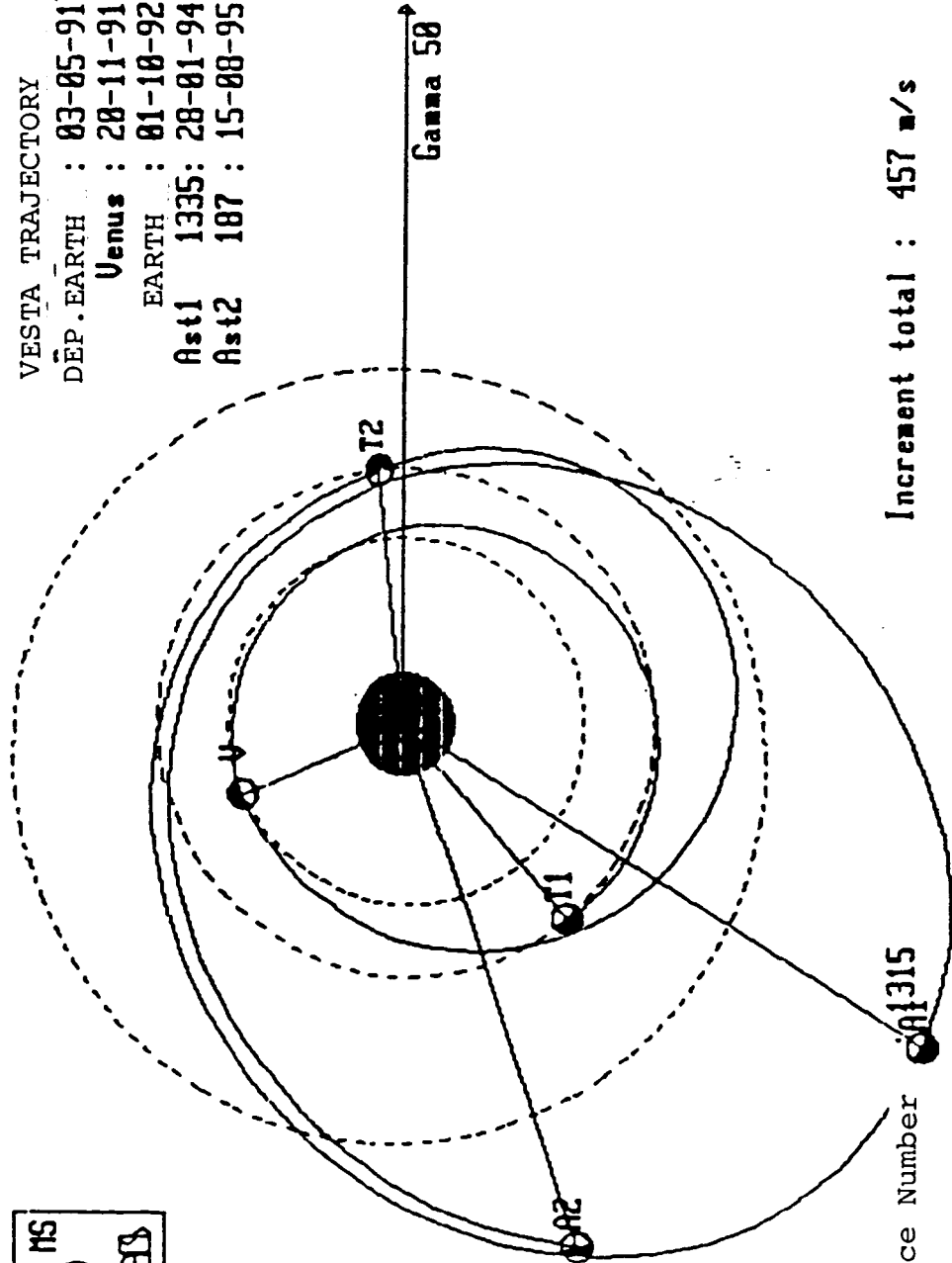


VIT INF DP  
 level 1= 3.0  
 level 2= 3.2  
 level 3= 3.5  
 level 4= 3.8  
 level 5= 4.0

VIT INF AR  
 level 1= 5.0  
 level 2= 5.3  
 level 3= 5.6  
 level 4= 5.9  
 level 5= 6.2



VESTA TRAJECTORY  
 DEP. EARTH : 03-05-91  
 Venus : 28-11-91  
 EARTH : 01-10-92  
 Ast1 1335: 28-01-94  
 Ast2 187 : 15-08-95



Increment total : 457 m/s

Sequence Number 01315



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Distributions of velocity increments  
expressed in m/sec.

Characteristics of Asteroids visited

N° SEQUENCE	ASTEROID 1	ASTEROID 2	ASTEROID 3	ASTEROID 4	EARTH	VENUS	EARTH	AST.1	AST.2	AST.3	TOTAL
350	Name Type- * N° 1274 DELPORTIA S 6	N° 654 ZELINDA C 60	N° 5 ASTRAE S 60	N°	0	3	35	325	154		517
538	Name Type- * N° 1657 BOPERA 14	N° 453 TEA S 10	N° 364 ISARA SMRU 16	N°	0	3	153	354	46		556
675	Name Type- * N° 1330 SPIRIDONIA P 28	N° 594 MIREILLE S-5	N° 5 ASTRAE S 60	N°	0	21	1	114	425		561
1284	Name Type- * N° 813 BAUREIA 7	N° 873 MECHTHILD EXP 12	N° 187 LAMBERTA C 72	N°	0	10	348	303	52		715
1316	Name Type- * N° 1335 DEMOULINA 3	N° 1948 KAMPALA 6			0	29	6	237			272
1315	Name Type- * N° 1335 DEMOULINA 3	N° 187 LAMBERTA C 72			0	36	3	417			457
1058	Name Type- * N° 1281 JEANNE 9	N° 2061 ANZA C 1, 2			0	0	9	595			604
2586	Name Type- * N° 1858 LOBACHEVSKIJ 8-3	N° 969 LEOCADIA 9-4	N° 1982 CLINE 6	N° 652 ANTIKLEIA S 15	0	49	4	97	258	190	598
1285	Name Type- * N° 813 BAUREIA 7	N° 873 MECHTHILD EXP 12	N° 187 LAMBERTA C 72	N° 1193 AFRICA 7	14	64	54	347	117	27	623
2583	Name Type- * N° 1858 LOBACHEVSKIJ 8-3	N° 969 LEUCADIA 9-4	N° 1843 BROWN 6	N° 652 ANTIKLEIA S 15	0	111	95	182	322	7	717
2704	Name Type- * ENCKE	N° 966 MUSCRI S 15			0	32	7	315			354
2705	Name Type- * ENCKE	N° 440 THEODORA 7			0	56	8	516			580

\*Radius

Furthermore, the same date of return to the Earth could come from several pairs, but in this case the various return speeds were similar.

b) Earth-Asteroid Trajectory Research

Based on the Earth return dates and speeds, we quantify the swing-by cone, the opening of which is given by the minimum fly-over altitude.

This quantification principle led to 70 conditions of departure from the Earth.

Each heliocentric trajectory thus created is thus processed and deformed to assure a multiple asteroid fly-over, and is then optimized.

We thus obtained 2705 multiple fly-over trajectories.

c) Overall Trajectories

As the results of 2.2. were stored in a data base, the latter is queried to have different required satisfactory trajectories, namely:

- in all cases, acquisition notice > days;
- total Earth-Asteroid increment < 800 m/s prior to optimization.

Then we divided into several trajectory classes:

- trajectory class providing flights over 3 asteroids, two of which are known;
- those which provide a fly-over of at least four asteroids, without standard constraint;
- those which provide a flight over a comet;

-those which provide a landing on an asteroid, namely, verifying:  
fly-over speed  $< 4$  km/s.

The asteroid band used is TRIAD 70.

The trajectories thus preselected were then optimized and the final trajectories retained are given below. The strategy The strategy is one maneuver for each fly-over.

#### d) Landing Site on Venus

As the Soviet party is interested in using a lander on Venus, for each overall French trajectory we studied the possible landing sites on this planet, for various entry angle value into the Ven-  
usian atmosphere.

### 2.3 - Spacecraft Design and Description of sub-Systems

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#### 2.3.1 General Spacecraft Configuration

The general spacecraft configuration is based on the selection of the following systems and constraints:

.type of stabilizer

Data obtained from the first mission analyses and conditions stemming from scientific instruments led to the selection of a 3-axis stabilization mode with airborne kinetic moment.

.type of aiming

Preliminary studies showed that solar aiming made it possible to offer both a reliable aiming and convenient links with the Earth during most of the Earth/Asteroid cruise orbits (skewed solar aiming).

.orientation of the optical instruments and of the radar

As a result of this requirement these instruments are installed in an orientable turret. Its main spin axis is parallel to the airborne kinetic moment.

.power generation

Since the sun moves farther away during the fly-over phases, the use of a nuclear reactor seems to be an interesting power source. However, this equipment is available only in the United States and its use was ruled out for the VESTA spacecraft which will be taken from the Soviet Union. A rigid solar generator, associated with batteries was therefore be selected as primary power source.

.TM/TC links

The links are critical because of the withdrawal of the vehicle during fly-over missions. Furthermore, for cost and planning reasons, the use of an "umbrella" or "petal" antenna was excluded. The spacecraft configuration should therefore accommodate a conventional antenna with maximum diameter ( $\emptyset$  1.4 m nominally).

.volume under the nose-cone

This volume (see section 2.1.2) is penalizing because of its large conical part. This constraint along with the location of /35 the rigid solar generator makes it necessary to use a pyramidal structure.

Figure 2.3.1 below shows the configuration of the VESTA spacecraft in orbit and figures 2.3.2 show the launching configuration.

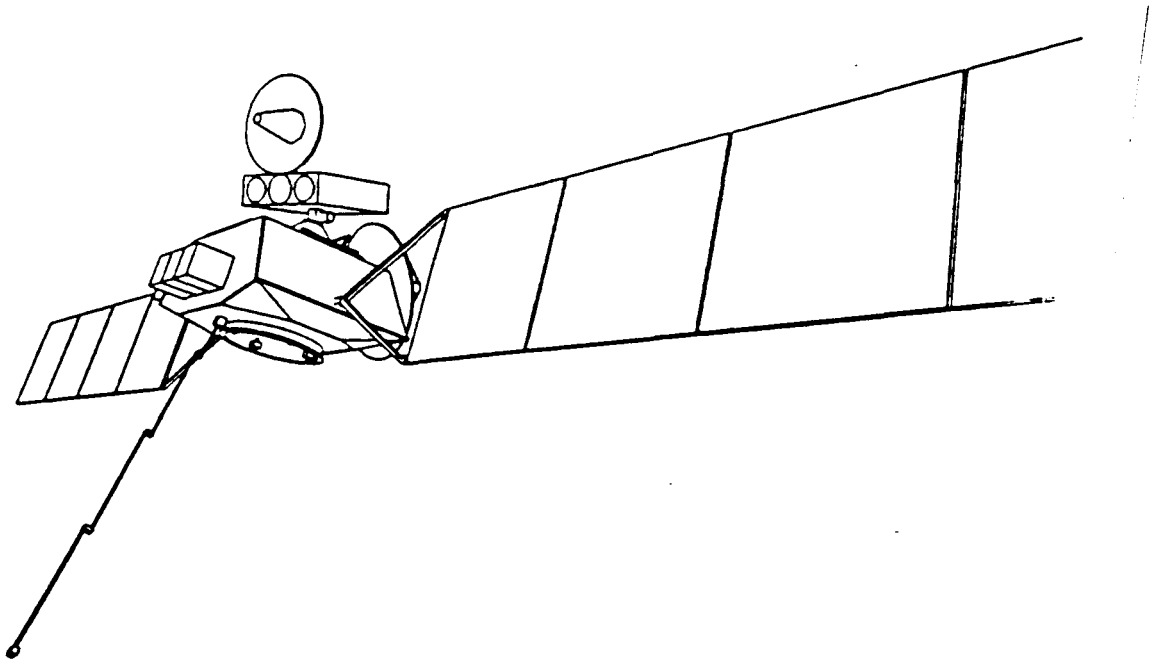


Fig 2.3.1 - Orbit Configuration

### 2.3.3 Description of Subsystems

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#### 2.3.2.1 - Structure and Mechanisms

The primary structure is made up of a central tube  $\varnothing$  940 in which a hydrazine tank is located and which has three inside shearing walls and outside walls (see figure 2.3.2.). The -X side is slightly concave for lodging a high gain antenna.

The solar generator, which has two wings with four flaps, is stored on the  $\pm$  Y faces and is deployed with its main symmetry axis in the X, Y plane.

The high gain antenna is installed on the -X wall, whereas the orientable turret is installed on a conical extension of the central tube. Retractable rods are located upon the latter so as to lock it during the launch. Two motors like those currently developed for DRS enable it to be aimed during the fly-over phase. Test weights are installed on the +X wall.

Finally the arm carrying the magnetometer is mounted in folded position over the -Z wall.

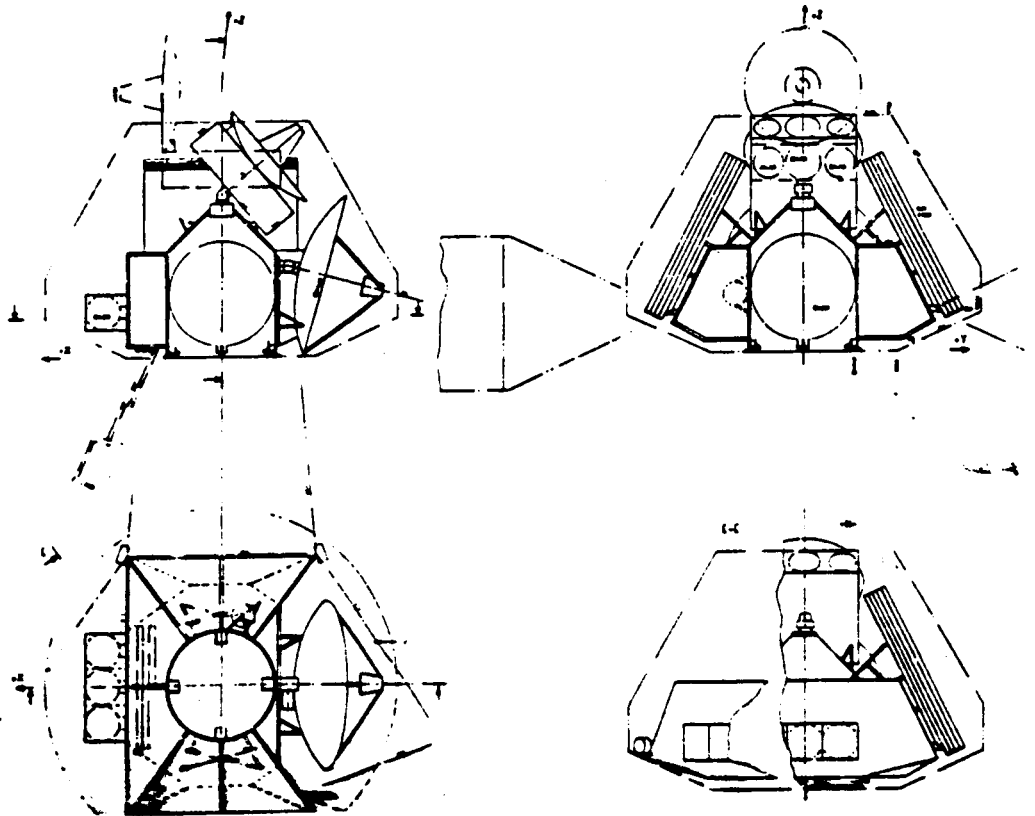


Fig 2.3.2 - Launching Configuration. Mechanical Architecture

#### 2.3.2.2 - Heat Regulation

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The heat control of the spacecraft should be designed to adapt to large solar flux variations (Venus:  $2,600 \text{ W/m}^2$  - Earth:  $1,350 \text{ W/m}^2$  - Asteroids:  $150 \text{ W/m}^2$ ) and to require minimum electrical power consumption.

The scientific payload installed in the orientable turret is disconnected conductively from the spacecraft by the aiming motors. Its thermal control during the various environmental and operating phases is provided by a coupling which varies with the cold space via mobile flaps and radiative surfaces and, via inactive heating. The inactive faces are covered with super-insulation.

The thermal control of the spacecraft sub-systems also requires radiators and mobile flaps whereas thermal control of the tube with progressive waves and its supply requires only a radiative surface and a heater owing to the substantial thermal power to be dissipated. The inactive faces are also covered with super-insulation.

Figure 2.3.3 shows the main features of this sub-system.

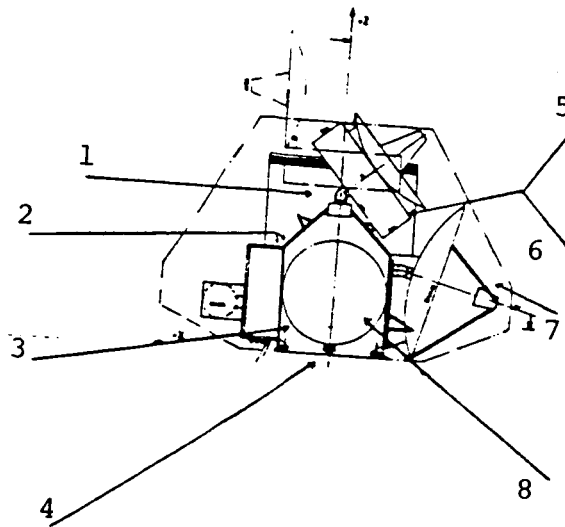


Fig 2.3.3 - Sub-System - Thermal Control

Key: 1-PF/CU conductive and radiative decoupling;  
 2-General superinsulation of outside surfaces;  
 3-General passive control by radiators and thermal flaps;  
 TOP radiators and battery, surfaces not receiving sunlight + heaters  
 4-Active control of RCS;  
 5-Control of CU electronic system by thermal flaps and radiators compensated by heaters;  
 6-CU partitioned into 3 uncoupled pockets (to be verified).

#### 2.3.2.3 - Propulsion

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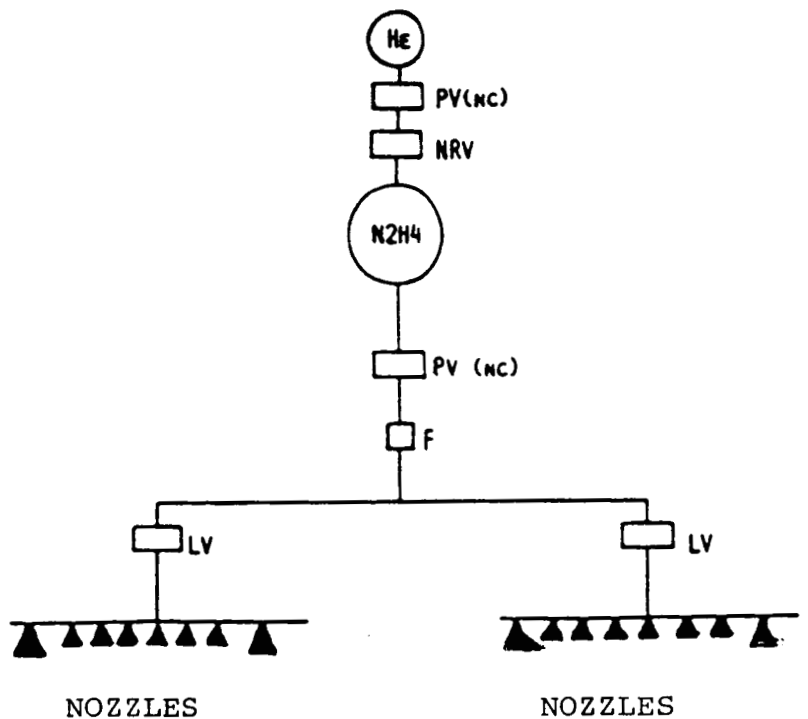
The propulsion system has the following two-fold function:

- orbit correction (~800 m/2)
- attitude control.

A conventional hydrazine propulsion was selected for its implementation and installation ease on the spacecraft and despite its weight handicap which is not critical in this project.

This sub-system is therefore made up of an already existing large diameter tank lodged inside the central tube, a helium tank (providing repressurization of the system in a mission environment), 12 low thrust nozzles and 4 average thrust nozzles (see figure 2.3.4).

This solution offers a simple, reliable and economical configuration and does not require any development.



#### 2.3.2.4 Attitude and Orbit Control

As we have seen in section 2.3.1, a three axis stabilizer and solar aiming system were retained. During a large portion of the operational phase, this type of skewed aiming enables terrestrial aiming to be accomplished indirectly and thus facilitates the links.



In normal mode, the probe is controlled by a solar sensor and a stellar sensor controls the inertia wheel and low thrust nozzles.

Finally, during the fly-over phases, any use of the nozzle is excluded so as not to disturb the weight measurement of the asteroid) and the orientable platform spins. The stellar sensor and gyrometric detector maintain the attitude of the spacecraft and control the nutation along a sighting axis of the sensor, They control the inertia wheel system with one degree of freedom. The turret oriented by an asteroid sensor, diverting from a developing stellar sensor with the distance permitting, and by scientific optical instruments in approach and withdrawal phase.

Finally, in all cases, the standby mode is a solar aiming system.

Figure 2.3.5 shows the general configuration of the A.O.C.S.

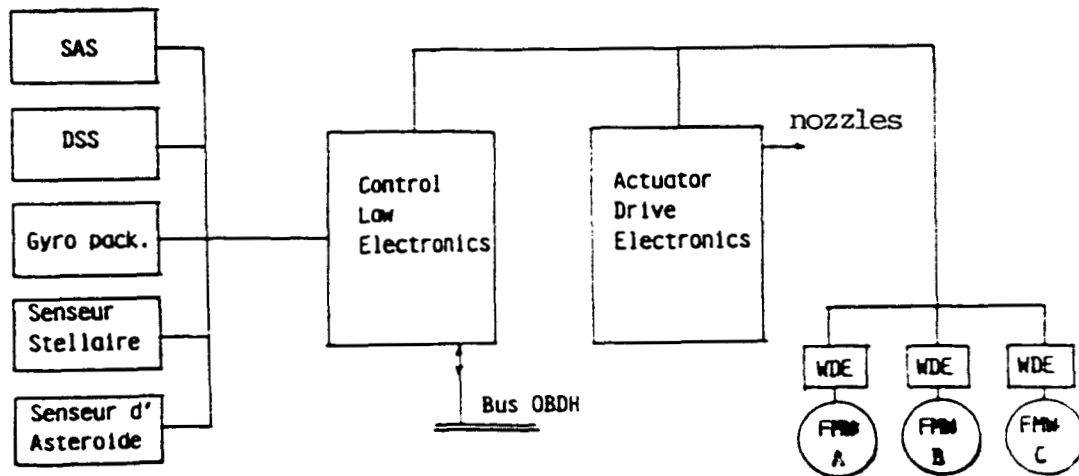


Fig 2.3.5 - AOCs Subsystem

### 2.3.5. Electrical Power Supply

/40

Since use of a nuclear reactor is excluded, the only power source available is sunlight. Accordingly, the architecture of

conventional and simple (see figure 2.3.6.) and utilizes a bar concept regulated at 28 V.

Its elements are the following:

- a solar generator of about 16 m<sup>2</sup> supplying 180 W to 3 UA,
- an NiCd battery of 16 Ah for phases when solar aiming is not possible and when the solar generator is inadequate,
- A direct-direct power regulator,
- A battery discharge regulator,
- A distribution and interface unit.

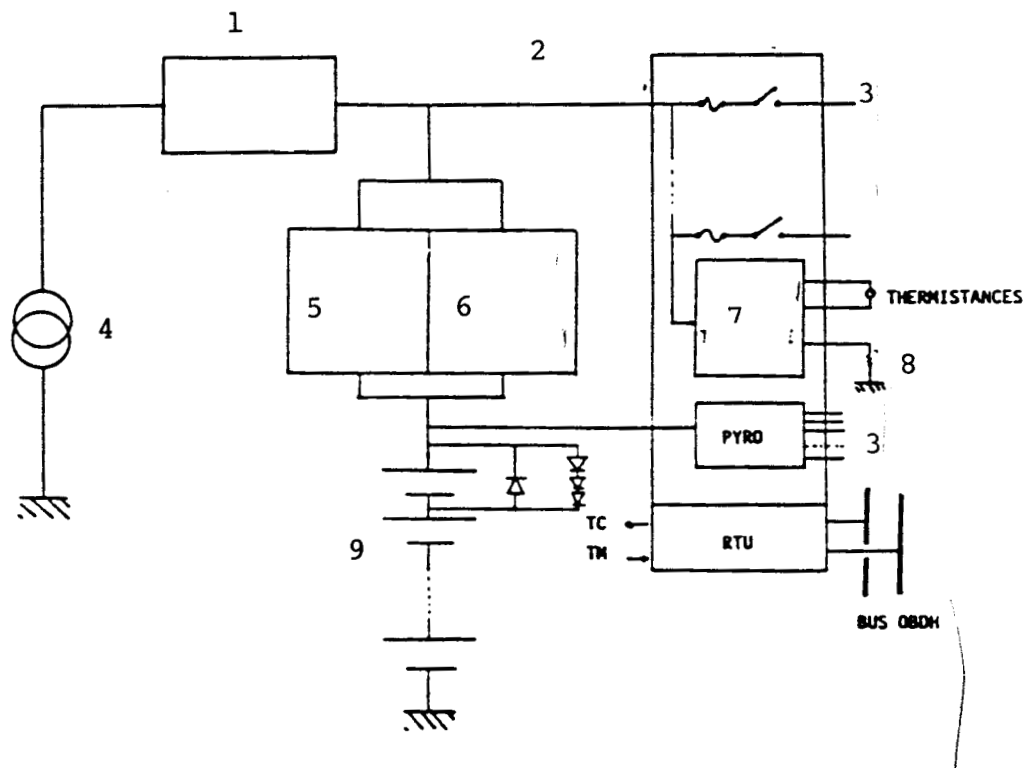


Figure 2.3.6: Subsystem - Electrical Power Supply

- 1-Day regulator; 2-27 DC V  $\pm$  1%; 3-To users;
- 4-Solar generator; 5-Load regulator; 6-Discharge regulator;
- 7-Thermal control; 8-Heaters; 9-Battery;

#### 2.3.2.6. - Communications System

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The ground stations provided are either the WEILHEM station or the D.S.N. stations used in S band (rising or descending links) and in an X band (descending links).

### 2.3.2.6.1 - Radio-Electric Links

The radio-electric links between the spacecraft and the Earth are established according to the following strategy:

.Earth/Venus/Earth phase: activating spacecraft aimed at sun through one of the two omnidirectional antennas and remote control of the spin to accomplish about the Z axis to aim at the Earth (rhythm \* TC=125 b/s TM=500 b/s).

.Earth/Asteroid phase: as long as the probe is at a distance of less than 2 UA from the sun, this phase is identical to the one above.

As of 2 UA when the satellite is aimed at the Earth, access to the satellite is continuous when it is not in fly-over phase. The standby mode allows the satellite to be accessed via a wide coverage helix antenna (rate ~8 b/s with DSN network).

The table below shows a summary of the descending link in X band (8.4 GHz) over the WEILHEM station (telemetry + remote measurement).

---

	Descending Link	
Tube power	43	dBm
Output losses	-0.5	dB
Antenna gain	38.8	dB
Aiming loss	-1	dB
Atmospheric losses	-0.2	dB
Space attenuation (4 UA)	-286.5	dB
G/T WEILHEM	48	dB/K
BOLTZMAN constant	-198.6	dB.KHz
Descending S/N	40.2	dBm.Hz

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	Carrier Attachment	
Modulation loss	- 3.5 dB	
Loop band (48 Hz)	16.8 dBHz	
Required S/N	10 dB	
	9.9 dB	
Margin		

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	Telemetry (With M.D.)	
Modulation loss	- 4 dB	
Binary rate (500 b/s)	27 dBHz	
Demodulation loss	- 1.5 dB	
E/N <sub>o</sub> required	4.6 dB	
	3.1 dB	

---

By retaining a daily visibility of 8 hours and a scientific information volume of 256 Mb, the memory dump will take about 20 days.

#### 2.3.2.6.2 Architecture of the Subsystem (See figure 2.3.7) /43

As we have just seen, the communications subsystem includes 4 antennas (2 omnidirectional antennas, one wide opening antenna ( $\pm 20^\circ$ ) and a high gain antenna  $\emptyset$  1.4 m).

The reception of rising signals is accomplished using two S-band receivers in hot redundancy which points the remote controls toward the airborne management and the remote measuring signals either toward the S band transmitter with 5 W (redundant) or toward the X-band transmitter of 20 W (redundant).

These two transmitters will also be used for transmitting remote measurements.

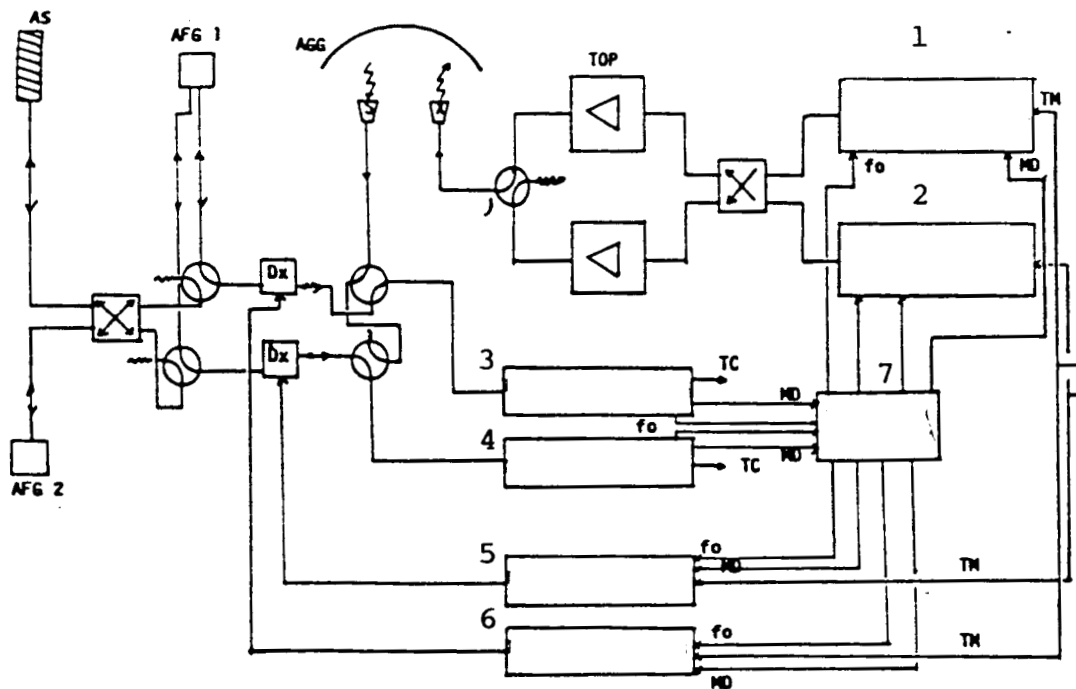


Fig. 2.3.7 - Communications Subsystem

Key: 1-X-band transmitter; 2-X-band transmitter;  
 3-S-band receiver; 4-S-band receiver;  
 5-S-band transmitter; 6-S-band transmitter;  
 7-Relay.

#### 2.3.2.7. Airborne Management

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In addition to remote control distribution and telemetry formatting, this subsystem provides the following functions:

- processing of scientific telemetry (compression if required) and its storage.

- autonomous management of the spacecraft and of the payload during all phases, including the standby phase, based on sequences and application software stored on board.

The subsystem is organized around a data processing structure including a central management unit linked to the various sub-systems of the spacecraft by a OBDH series bus, a bubble mass storage of 256 Mb. The acquisition of the scientific telemetry is accomplished either with the OBDH bus, or with a rapid

interface for optical instruments (see figure 2.3.8)..

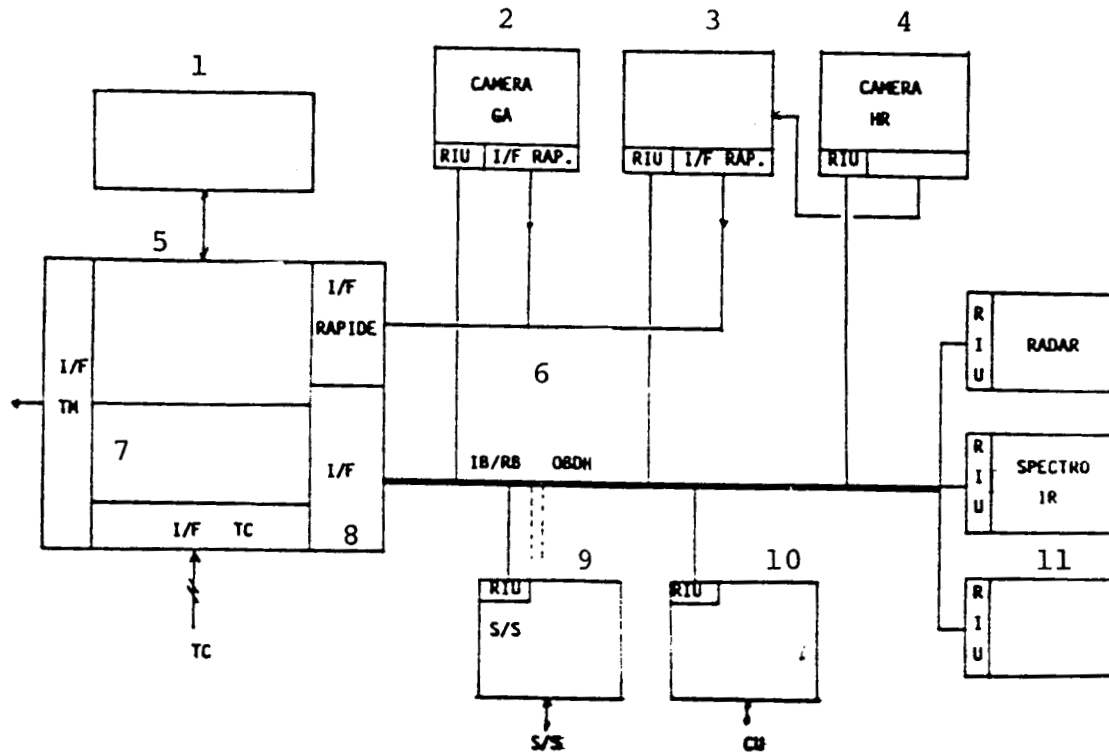


Fig 2.3.8 - Subsystem - Airborne Management

Key: 1-Bubble memory (256 MBITS); 2-GA Camera;  
 3-Compressor computer; 4-HR Camera;  
 5-Rapid telemetry processor; 6-High-speed line;  
 7-Management processor; 8-Artery I/F;  
 9-Platform S/S; 10-CU interface unit;  
 11-Magnetometer;.

## 2.4 - System Analysis

/45

The preliminary weight analysis shown in table 2.4.1 reveals a margin of 130 kg and makes it possible to conclude that the mission is feasible with the proposed concept.

The power analysis shown in table 2.4.2 corresponds to the fly-over mode which is considered to be dimensioning in a first analysis.

Table 2.4.1 Weight Analysis

PAYLOAD		PLATFORM	
CGA	5.0	STRUCTURE-MECHANISMS	100.00
CHR	13.0	THERMAL	20.0
RADAR	6.0	PROPULSION	28.1
+ ELECTRONIC	19.0	TM/TC AIRBORNE MANAGEMENT	92.5
IR SPECTROMETER	17.0	ATT. AND ORB. CONTROL	45.0
MAGNETOMETER	1.0	AIRBORNE POWER SUPPLY	67.8
+ ELECTRONIC	3.0	(incl. GS and battery)	67.8
TEST WEIGHT	7.5	HARNAIS	15.0
UNIDENTIFIED TESTS	<u>20.5</u>		
S/TOTAL	9.15	S/TOTAL	<u>368.4</u>

DRY TOTAL	460
DRY MARGIN	<u>130</u>
DRY AVAILABLE WEIGHT	590
PROPELLENTS	279
LAUNCH MASS	<u>870 Kg</u>

Table 2.4.2 Power Analysis

	Average P	Peak P	GS P	Battery
Payload	53	179	34	115
Data management	30	75	30	ε
TM/TC	6	6	6	0
Att. and Orb. control	103	132	52	100
Propulsion	0	0	0	0
Thermal	50	50	50	0
Airborne power	<u>10</u>	<u>10</u>	<u>10</u>	<u>0</u>
	252 W	452 W	182 W	215 Wh

## 2.5 Operations and Ground Segment

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### 2.5.1 Summary of Operational Requirements for the VESTA Mission

#### a) Launching and Parking Orbit

The probe is inert until the time of separation and injection into hyperbolic orbit.

This phase does not require special operational means on the French part. The launcher will be monitored by the Soviets (boats).

#### b) Injection and Close Orbit

The orbit parameters of about 15 minutes after separation will be provided to the ground segment.

This phase is relatively critical because the probe was just activated with ground antennas with a small diameter (9-15 m) in the first hours then we will move on to WEILHEIM type 30 m antennas.

#### c) Earth/Venus and Venus/Earth

In this phase complete control sessions are made of the spacecraft and of the payload every 2 months and over a 15 day period.

Simplified control sessions are performed every week for 8 hours.

#### d) Passage near the Earth

Control sequence of the payload: simulation of approach and fly-over phases.



e) Cruise Toward an Asteroid

The ground segment functions and the operations performed are identical to the EARTH/VENUS and VENUS/EARTH Phase.

f) Prior to the Flight Over an Asteroid

Critical phase of the mission. Station coverage 24H/24. The trajectory corrections are made after analysis of images from the navigation camera. The communication means of stations at the control center will be dimensioned. The same will be true for data processing means.

g) Flight Over an Asteroid

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No airborne/ground link during the flight over an asteroid. The airborne memory of 256 Mbits is loaded with payload data.

h) After Flighing over an Asteroid

After flying over an asteroid a trajectory correction is made, then the contents of the airborne memory is dumped on ground stations adapted for this operation (X-band, large diameter antenna).

This phase is the most limiting today because of the dump time.

Memory to dump of 256 Mbits.

Digital rate 500 B/s → 1 memory dump in 18 eight hour sessions.

If the memory is dumped twice for a flight over an asteroid, and if 6 asteroids are insepcted, we obtain:

216 EIGHT HOURS SESSIONS WITH LARGE DIAMETER ANTENNAS

This gives virtually 1 YEAR of support for DSN type stations.

This condition is virtually impossible to accomplish. It is therefore indispensable to review the digital rate to obtain acceptable dump times.

For comparison:

AGORA MISSION: Digital rate < 1.6 KB/s X-Band

GIOTTO MISSION: Digital rate of 10 KB/s to 40 KB/s X-Band

ISPM MISSION: Digital rate of 8 KB/s X-Band, Distance 1 UA.

## 2.5.2 Preliminary Architecture of the Ground Segment

The preliminary architecture which was retained to study phase 0 was established on the following assumptions:

- Mission employing Deep-Space type stations in certain phases.
- 24H/24 station visibility in the critical phases of the mission.
- Use of a small diameter antenna (S & X Band) in close orbit.
- Lack of information on departing orbit (station visibility)
- Installation of a control center and mission entity on the C.S.T..

### 2.5.2.a. Stations

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The stations which will be retained for the VESTA mission shall fulfill a certain number of criteria;

- transmission and reception in S-Band (2 GHz).
- reception in X-Band (8 GHz, for airborne memory dump).
- availability for mission.
- cost of minimum support.

The use of large diameter antennas (64 m) poses problems with DSN (availability). However because of improvements provided on the DSN 34 m antennas there is possibly a solution.

SUMMARY OF OPERATIONAL REQUIREMENTS

/48

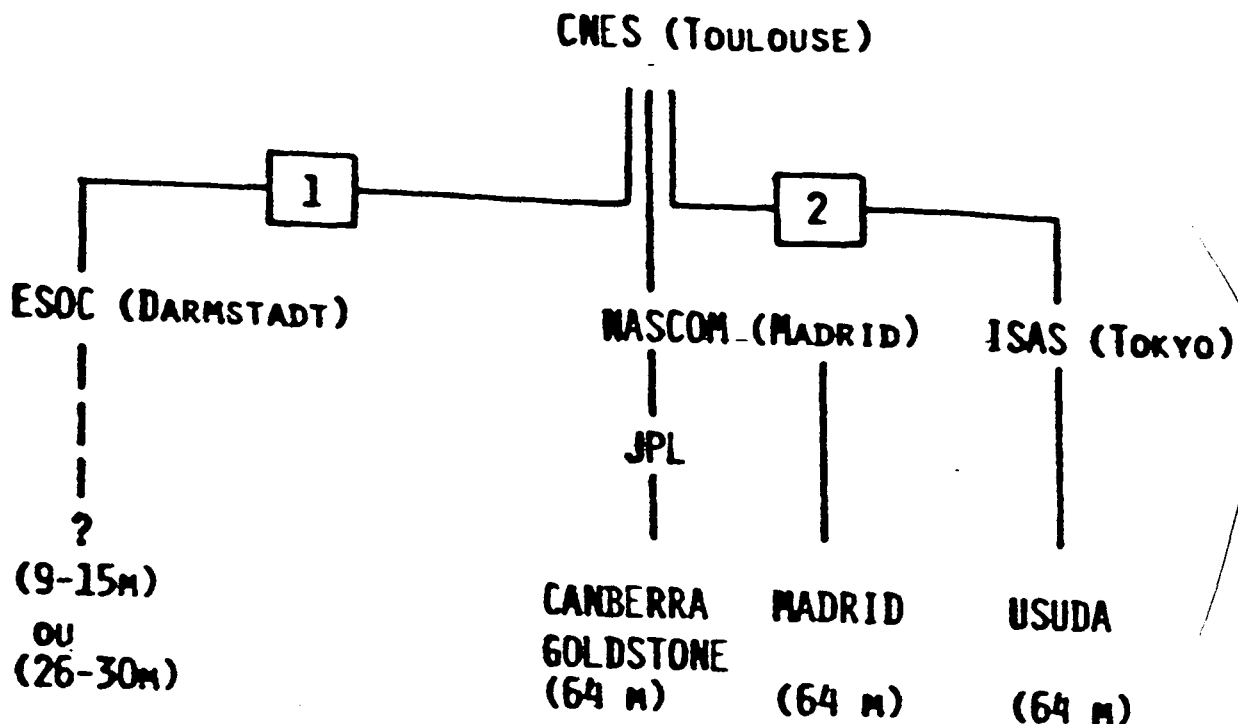
PHASES	NB	STATIONS ANTENNA	FREQUENCY TM/TC	TIME	DIGITAL RATE & OPERATIONS	CRITICAL PHASE
LAUNCH		INERT PROBE Soviet Tracking		<1.5H		NO
INJECTION & CLOSE ORBIT	>3	9-15 M or 30 M	S X&S	2-3 H >3 H	TM~160 to 500 BPS TC~15 to 100 BPS LOC: Doppler+MDI PF-CU VERIFICATION TRAJE. CORRECTIONS	YES
CRUISE	1	34 M	X	1 D/WEEK	TM~200 BPS (HSK) LOC: Doppler-MDI	NO
	2	34-64 M	X & S	15 D/ 2 MONTHS	TM~500 BPS TC~15 to 100 BPS LOC: Doppler-MDI CU CONTROL	NO
PRIOR TO FLY-OVER	>3	64 M	X& S	15 D/ 24 H/24	TM > 500 BPS TC > 15 to 100 BPS LOC: Doppler MDI Image processing Nav Camera	YES
FLIGHT OVER ASTEROID		NO SPACECRAFT/GROUND LINK		few H to few D	RECORDING OF AIRBORNE MEMORY	NO
	>	64 M	X & S	few D	LOC: Doppler-MDI TRAJE. CORRECTIONS	NO
AFTER FLY-OVER	1+1	64 M	X & S	2.5 MONTHS	MEMORY DUMP (360 MB) TM > 500 BPS to 1 KBPS TC~15 BPS	NO

This RF analysis will be conducted in phase A.

Other USUDA type antennas (I.S.A.S.) or PARKES antennas (E.S.A. GIOTTO mission) are compatibel on the RF plan, but their compatibility remains to be studied.

POSSIBLE NETWORK OF STATIONS TECHNICALLY COMPATIBLE WITH  
THE VEHICLE

/49



The most limiting aspect of choosing stations is the duration /50 of the dump operations (1 year approximately at 500 B/s). These operations should necessarily be reduced if one wants to find a support for the mission.

A solution to this problem consists of increasing the digital rate of the dump and to perform this operation as close as possible to the Earth.

MISSION	DISTANCE	SCIENTIFIC TEM RATE	STATION
.VESTA	4 UA	500 B/s to 1 KB/s	DSN 64 M
.GIOTTO	1 UA	20 KB/s to 40 KB/s	PARKES 64 M

Only the mission analysis may determine a trajectory upon which the selection of stations and supports required will be optimized.

This "pilot" trajectory is indispensable in phase A if an

accurate RF analysis is to be made as well as a finer estimate of costs and of difficulties for later phases.

#### 2.5.2.b Control and Mission Center

The control and mission center is made up of three entities:

- CCM: Control and Mission Center
- COM: Orbitography and Maneuver Center.
- COR: Network Operations Center.

This architecture was built on the assumption that the control centers were over CNES.

In phase A, we will study the feasibility of a control center which is partially or totally outside of C.S.T. (E.S.O.C.).

### 2.6 Special Study Points

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#### 2.6.1 Mission Analysis

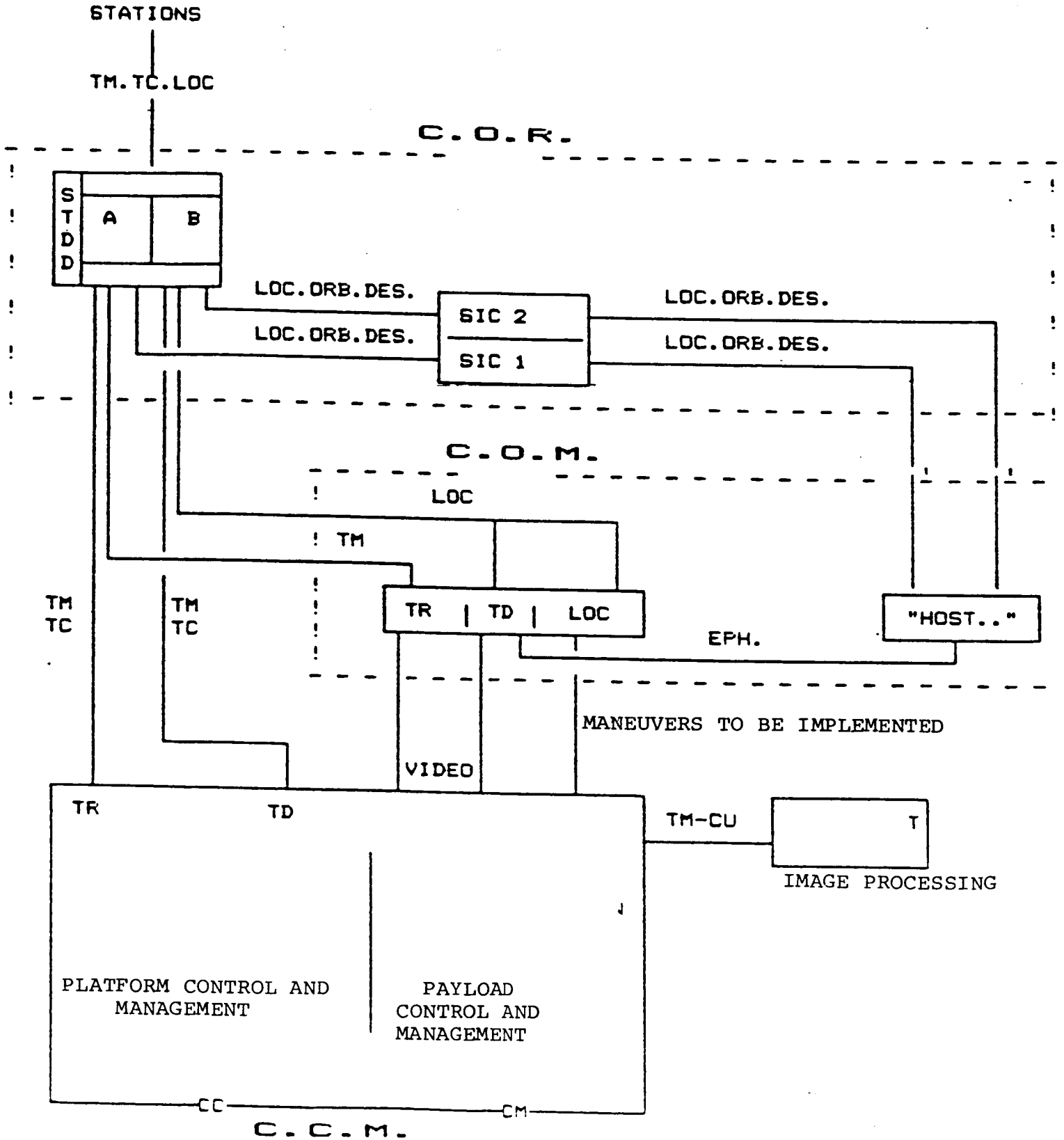
It is now acquired that the VESTA mission will not be ready for the 1991 launch window.

An analysis of mission phase A should therefore account for the following window (November 15 1992 - February 10 1993).

The study of phase 0 has shown certain flaws in the determination of potential targets which will have to be corrected in phase A.

An accurate and full description of the trajectories is necessary for spacecraft studies. Considerable effort shall therefore be accomplished in this direction.

The problem of tracking and navigation has not been investigated in detail until now. It is essential to quickly know



(December 1985) the tracking and navigation performances accessible by an airborne optical system.

For other navigation phases, an analysis of the tracking requirements should be conducted (including operational needs) to be able to discuss with other members on a solid technical basis.

#### 2.6.2 Payload

.Complementary feasibility study of the Altimeter Radar/Radiometer - detailed technical inspection and simulation tests.

.Complementary feasibility study of determining the mass of an asteroid by measuring trajectory changes of a test mass dropped from an autonomous spacecraft.

This study will be conducted on the following 2 methods:  
Estimate of trajectory changes by measuring =

- direction changes using an optical television instrument;
- distance changes at high altitudes.

.Determining the type of aiming system for the platform which may be oriented over the asteroid so that the relative velocity at the surface is as slow as possible.

.Defining the imagery system and its performances, the high resolution camera required for trajectory correction maneuvers during the navigation phase.

.Definition of a Quality Assurance plan applicable to the high resolution camera which fulfills technical functions essential for mission success.

.Preliminary definition of the management module of the payload during which focus will be given to airborne data

requirements linked to the operating time constraint of the ground segment.

.Preliminary definition of image processing on the ground.

.For research on instruments of phase A, it will be necessary to give laboratory support in the form of recommendations and advice on the use of certain hardware and software and to perform detailed appraisals of the feasibility studies conducted by these laboratories on the hard points identified.

.For the use of instruments at the beginning of phase B, it will be necessary to establish the payload specifications.

### 2.6.3 Spacecraft

During phase A and based on a more accurate definition of the various phases and of the mission, more detailed studies shall be undertaken on the following points:

.Study of the impact of increasing the volume under the nose cone allowable on the configuration and systems selections.

.Thermal studies for worst cases (hot and cold) for dimensioning the thermal control of the payload and of the spacecraft; evaluation of the possibility of operating the payload in an intermediate EARTH/VENUS/EARTH orbit.

.Study for the flight control of the spacecraft and of the turret during the fly-over phases. Feasibility analysis and performance of an asteroid sensor.

.Behavior tests of solar cells under low lighting.

.Performance of solar generator in extreme environments (hot and cold).



.Study of day regulator.

.Impact of the spacecraft autonomy constraint on airborne management.

#### 2.6.4 Ground Segment/Operations

##### a) Airborne/Ground Links: Effects on Airborne Architecture

.Use of frequencies proposed by Svoiets for LOC and airborne memory dump (see N3 protocol).

→Effects on airborne architecture, and on operations.

.Use of a single frequency band (2 GHz) for any mission /54  
(dump during passage near Earth or at 1 UA).

→Regulations, Airborne Architecture, Operations.

##### b) Stations

.To establish the list of "Deep-Space" stations with the following characteristics:

- reception in X-Band of scientific TM;
- digital rate from 5 to 10 kB/s;
- G/T allowing this TM to be received at 2.5 UA;
- diameter of aerial less than 34 meters.

.Study of EUSUDA (I.S.A.S.) and PARKES (E.S.A.) stations within VESTA mission.

##### c) Operations

.To establish an operations scenario based on two "pilot" missions with agencies involved and any constraints at operations level.

.Based on a Control Center located at ESOC, for example, to establish an operations and ground segment scheme to be implemented.

d) Ground Segment

Analysis of processings to be accomplished.

Analysis of hardware and software requirements.

Architecture of the ground segment.

e) Definition of "image" products and of related processing operations.

3. PROJECT MANAGEMENT

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3.1. Program Schedule

3.1.1 Spacecraft Development Program

The development philosophy retained should minimize technical risks and costs. Given the originality of the mission and the related new technical constraints, the model chaining is the following:

- qualification model of the structure and volume under the nose cone and thermal test model (MSTh)
- identification model (MI)
- qualification and flight model (MOV)
- second flight model (MV)
- replacements for most critical equipment (MR).

The MSTh structural model will verify the resistance of the primary structure to static loads and will allow a dynamic analysis to be accomplished. Static, random vibrations and acoustics tests will therefore be conducted. On this same model, we will then perform vacuum/sun tests to verify the thermal modelisations.

The MI model will allow the operation of the integrated satellite to be validated from the electrical, electromagnetic - and electrical continuity standpoint.

Finally, on qualification and flight models, tests will be performed on vibrations, vacuum/sun and on electromagnetic compatibility.

MSTH is manufactured concurrently with that of a structural model fitted with inert equipment delivered to the Soviet party at the end of 88 for conducting mechanical tests on the composite.

### 3.1.2 Payload Development plan

In compliance with the development plan for the autonomous spacecraft 7 models shall be developed per experiment:

.Volume Model	= M.E. (VM)
.Structural Model	= M.S. (SM)
.Structure and Thermal Model	= MSTH (STM)
.Identification and Qualification Model	= M.I.Q. (EM)
.Protoflight Model	= P.V. (PFM)
.Flight Model	= M.V. (FM)
.Spare Model	= M.R.

Table 3.1.2 shows:

- .The definition of these models;
- .their use;
- .the delivery date to CST of these models for pre-integration.

The general planning of the VESTA Project shown in figure /58 3.1.1 allows for a 6 month margin which may be doubled if the PFM and MV2 are integrated concurrently.

TABLE 3.1.2

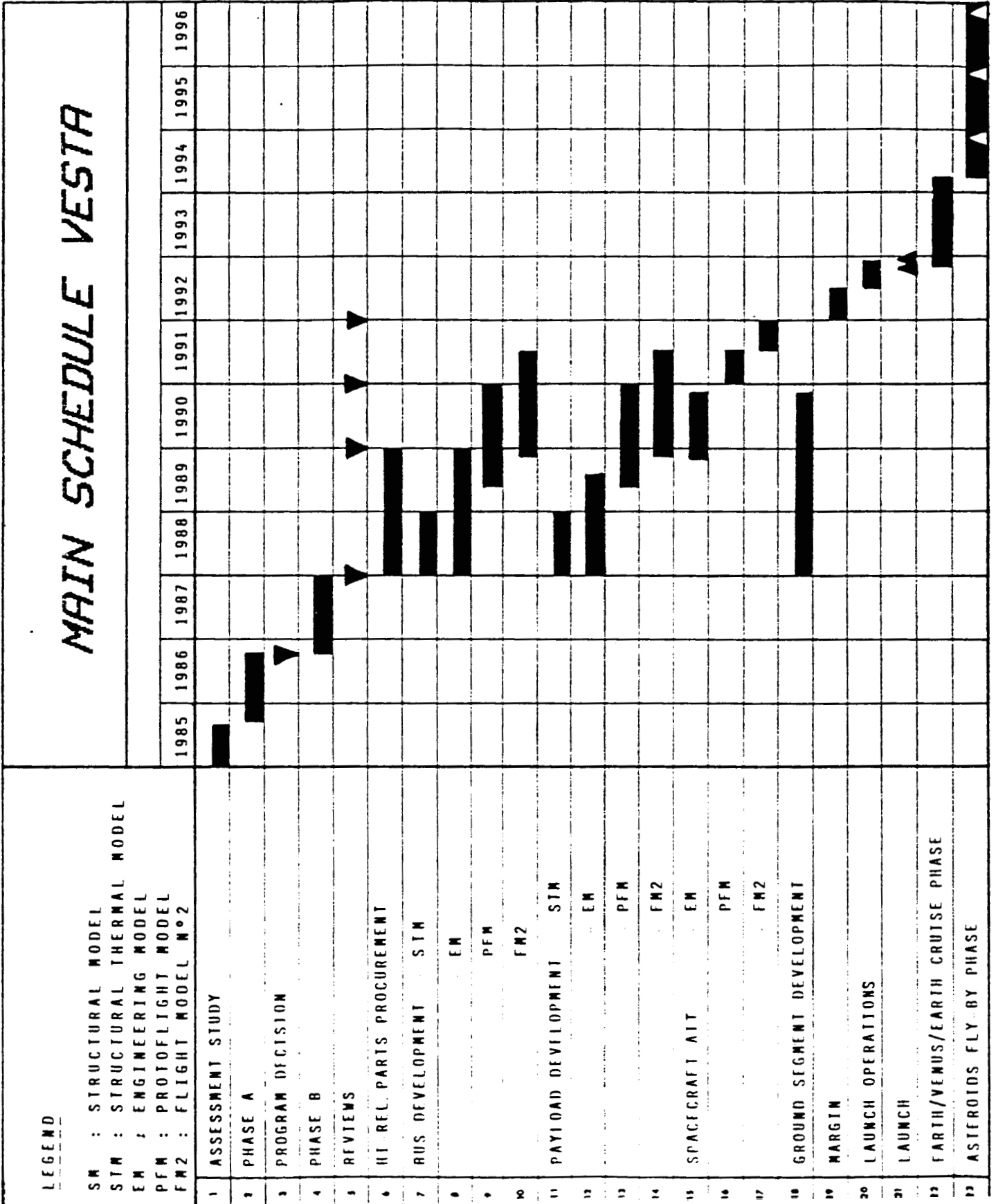
VESTA - INTEGRATED EXPERIMENTS ON THE FRENCH SPACECRAFT

MODELS	DEFINITION	UTILIZATION	DELIVERY DATE TO CST
ME	-Volume -Weight, CdG -connector position	-Acceptance at C.S.T. -Integration to ME/VA -shipment to USSR	15.10.88
MS	-Weight, CdG - Inertia -rigidity-mechanical strength	-Acceptance to CST - qualification of structural vibrations 1. integration to MS/VA 2. vibrations = acoustic tests 3. shipment to USSR	15.11.88
	Ditto MS + -thermal dissipation -linings in compliance (superinsulation, $\alpha$ , D)	-Acceptance at C.S.T. - possible complementary vibration 1. integration to MSH/VA 2. vacuum sun 3. acoustics	15.12.88
MIQ	-electrical conformity (standard composites) -structure = MV -linings - MV	-Checks on integrated instrument 1. inlet checks (mechanical and electrical interfaces) 2. functional tests with management module 3. weight, CdG, inertia 4. vibrations 5. EMI, MEC - After integration on VA/MIQ 1. Overall functional tests 2. EMI, EMC	01.10.89
P V	-model in compliance with flight specifications	-Checks on integrated instrument 1. inlet checks (mechanical interfaces and electrical interfaces CdG mass) 2. functional tests with management module 3. sun vacuum (integration level to be defined) - After integration on VA/PV 1. vibration 2. EMI, EMC 3. sun vacuum - Final integration and launch at BAYKONUR	01.12.90

MODELS	DEFINITION	UTILIZATION	DELIVERY DATE TO CST
MV	-MVI ditto -	-Checks on integrated instruments: MVI ditto -after integration on VA/MV; ditto MVI except VS replaced by VT -final integration and launch at BAYKONUR	01.15.91
MR	-MIQ upgraded with space components	-Inlet check -functional tests -vibrations, integration on VA/MIQ	01.01.91

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Figure 3.1.1 Planning général VESTA



### 3.2. Planning

CNES will be the project contractor and contractor for its three main components (see figure 3.2.1):

- system
- space segment
- ground segment.

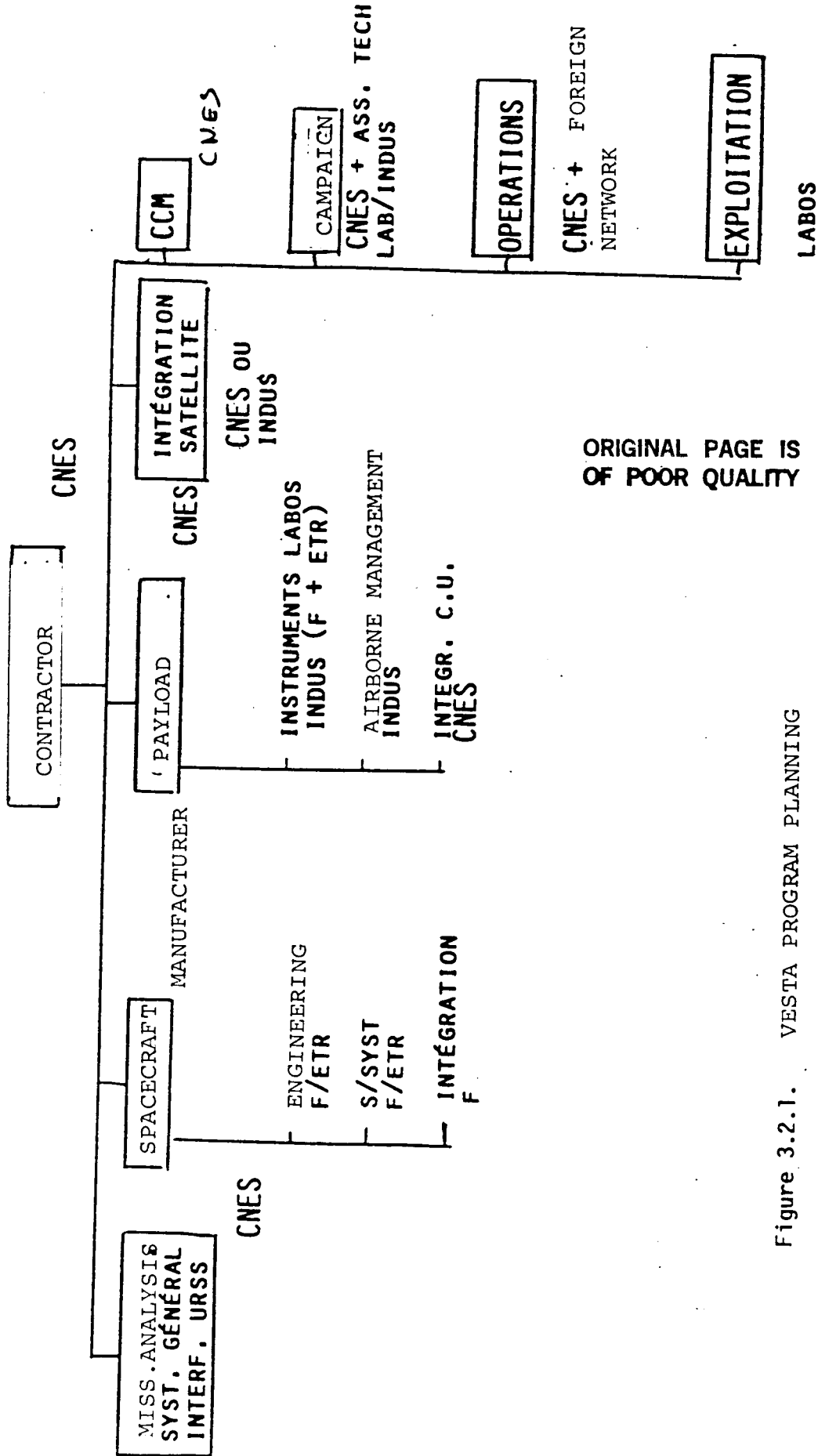
However the spacecraft part of the space segment will be under an industrial contractor (nominally French) with European participants for the subsystems.

The instruments will be developed under the responsibility of scientific laboratories whereas their integration will be accomplished by CNES.

The final integration of the spacecraft and payloads as well as the launch campaigns will be accomplished nominally by a combined Industrial/CNES team.

Finally, the ground segment will call on ESA and on NASA for network aspects.

PROGRAM PLANNING



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Figure 3.2.1. VESTA PROGRAM PLANNING



#### 4. REFERENCES

/60

##### 4.1. General Interest

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- .VESTA Project technical report (USSR)
- .Basic data on the construction of an autonomous spacecraft to be used for the study of asteroids (USSR)
- .Technical report no. 2 (USSR).
- .Technical report - Mission analysis for an overall study of Venus and of small bodies of the solar system 1991.
- .PMF Report/APS/292 - Copy of guidelines of the meeting for the presentation of phase A of 04/10/85.

##### 4.2. Asteroid Spacecraft

- .VESTA: Feasibility study - MATRA Technical Report 85/CNES/5087/00/843 and copy of guidelines of the infal presentation of 03/07/85.
- .Assessment of MATRA spacecraft costs: 092/DM/DR/104/85.
- .Record of special technical caluses of contract 85/CNES/5087/00/843.
- .VESTA Project: Solar Generator - CNES Technical Report CT/DRT/AQC/EB/353.

##### 4.3 Payload

/61

- .Conclusion of Radar work group by Ph. MASSON (not referenced)
- .Preliminary feasibility study of the measurement of the mass of an asteroid using a test mass (Report CT/DRT/CI/VS No. 201).
- .Scientific experiments - general philosophy of acceptance and integration activiteis No. CT/DRT/DI/SE No. 227.
- .Preliminary feasibility studies of a separation and test mass ejection system Report CT/DRT/SST/SE/1401.

.Preliminary study of a radioelectric system for tracking the test mass (not referenced) by N. LANNELONGUE (15.07.85).

.Assessment of the mass of an asteroid - mission Analysis; Error analysis (not referenced) by D. SPAUTE (July 85).

#### 4.4 Mission Analysis - Ground Segment

.Analysis of operational requirements for the VESTA mission and data processing for the ground segment - Report CNES DTI/PSA/RS/85-269.

.Preliminary analysis of telecommunications in the remote probe project. Report CNES/DTI/PSA/RS (interim report).

.Assessment of the cost of the ground segment for the VESTA mission. Note CNES/DTI/PSA/RS/85-274 - Edition 1.

.Documents on the ground segment issued to the Soviets (02/09 to 05/03/85) edited by DTI/PSA/RS.

.Report on phase 0 of the VESTA ground segment. Report CNES/DTI/PSA/RS/85/354.

.VESTA 1991 - Investigation of trajectories. Report CNES/CT/DTI/MS/NM/296.