P-98

N91-16051

AAE 241 Spacecraft Design Project

Group #5

Spring 1990

Marty
Stephan
Julie
Dave
Tim
Kim

J. Martin McLain 326-52-0656
Kimberly J. Korkolewski 329-72-9299
[illegible] 246-43-1439
[illegible] 349669241
[illegible] 329-74-453
[illegible] 354 46 2864

INTRODUCTION

PULSE, Pluto Unmanned Long-Range Scientific Explorer, is an unmanned probe that will do a flyby of Pluto. It is a low weight, relatively low costing vehicle which utilizes mostly off-the-shelf hardware, but not materials or techniques that will be available after 1999.

PULSE will be launched within the first decade of the twenty-first century.

MISSION MANAGEMENT, PLANNING, AND COST

1.1 INTRODUCTION

In the subsystem of mission management, planning, and cost, many selections were made. The mission type, trajectory, and launch date were selected. The optimum delta-v and cost of the project were also calculated.

1.2 TYPE OF MISSION

A flyby was the type of mission selected. This selection was made due to its low delta-v, short mission duration, and simplicity, all of which are directly related to this mission's low cost.

Simplicity was a main issue in selecting this mission class. Since there have been no missions to Pluto and Pluto's distance from the Earth is very far, very little is known about Pluto and Charon. Therefore, before a high-cost, elaborate mission can be sent, scientists need more accurate information. A flyby mission is the most efficient way to get the information that is needed.

1.3 TRAJECTORY

The trajectory selected for this mission is a direct Earth to Pluto path. Again, simplicity was an important issue in the selection process. The more complex a mission, the greater the opportunity for something to fail. So by using a direct path, simplicity is optimized.

1.4 MISSION DELTA-V REQUIRED

The delta-v required for the PULSE mission is 8.606 kilometers per second from a parking orbit around Earth.

1.5 MISSION TIMELINE

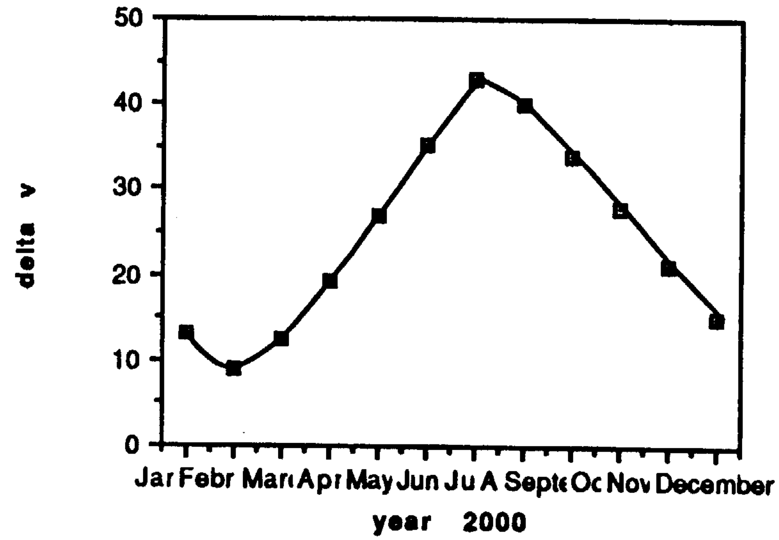
The launch date was determined to be January 30, 2003. The arrival at Pluto was determined to be February 1, 2019. The mission length is 16.005 years. The launch date was chosen by selecting the date with the optimum delta-v. To obtain a selection of dates, data was input for the first of every month of every year from the year 2000 to the year 2010. (Graph 1.1)

1.6 COSTING

The costing process of this mission was done in several steps. First, for each subsystem, the direct labor hours and the recurring labor hours were calculated. This was done by several different formulas that used the mass of each subsystem and the number of spacecraft. The number of spacecraft costed were four, three of which are flight ready and one which is used in an integrated ground test system.

Next, for each subsystem, an inheritance class had to be defined. Class One is an off-the-shelf buy. Class Two is an exact repeat of a subsystem. A Class Three inheritance is the use of a previous subsystem with minor modifications. A Class Four inheritance is also a use of a previous subsystem, but with

Graph 1.1 Illustrates ΔV for 1st of every month in the year 2000.



major modifications. Finally, a Class Five inheritance is an entirely new subsystem. (Table 1.1)

The next step was to convert labor hours into labor cost. Then the labor costs were converted into total costs. The conversion factors were given in Fiscal Year 1977 which needed to be converted to Fiscal Year 1988. This was done by using a consumer price index. The consumer price index for all items in 1977, with a base of 1967=100, was 181.5. The consumer price index for all items in 1988, with a base of 1967=100, was 354.3. (Appendix 1).

Finally, these conversions were made for each subsystem and then added to obtain the total cost of the project. (Table 1.2). The total cost of the PULSE project is about 1.7 billion dollars.

1.7 EFFECTS ON SUBSYSTEMS

Many of the selections made affected the selections of the other subsystems. The selecting of a flyby affected the science instrument selection. Because the mission is a flyby, only instruments which can be used quickly and at a distance could be used. The power and propulsion subsystem was also affected. By utilizing a flyby instead of an orbiter or a lander, less fuel was needed. These factors also affect the design of the structure.

The length of the mission and the trajectory selected also affected the other subsystems. Due to the length of the mission, 16.005 years, science instruments and other materials which lifetimes exceed 16.005 years had to be selected. These

Table 1.1

SUBSYSTEM INHERITANCE CLASS

Category	Inheritance
Structure	5
Thermal Control	1
Propulsion	1
Attitude & Articulation	3
Telecommunications	2
Antennas	2
Command & Data Handling	1
RTG Power	2
Line-Scan Imaging	2
Particle & Field Instruments	1
Remote Sensing Instruments	1

Table 1.2

Costing for PULSE

Category	Cost (FY 88 Dollars)
Structure	59,988,162.98
Thermal Control	11,037,938.33
Propulsion	412,927,670.50
Attitude & Articulation	62,614,609.37
Telecommunications	64,098,191.33
Antennas	13,043,018.66
Command & Data Handling	24,500,108.53
RTG Power	37,386,446.55
Line-Scan Imaging	170,454,335.10
Particle & Field Instruments	71,222,537.72
Remote Sensing Instruments	29,154,302.64
System Support & Ground Equipment	280,062,535.20
Launch + 30 Days Ops & Ground S/W	57,185,698.78
Image Data Development	6,957,007.47
Science Data Development	11,487,733.40
Program Management	17,365,267.83
Flight Operations	258,722,216.60
Data Analysis	115,984,760.70
TOTAL	1,704,192,542.00

selections affect the amount of fuel needed and the design of the structure.

1.8 CONCLUSION

Within the mission management, planning, and cost subsystem, many important selections were made. The PULSE mission is a flyby with a mission duration of 16.005 years. The launch date is January 30, 2003. PULSE is scheduled to arrive at Pluto on February 1, 2019. This mission requires an 8.606 delta-v from a parking orbit.

APPENDIX I

Fiscal Year '77 to Fiscal Year '88 Conversion:

(Total Cost)(FY88 dollars)/FY77 dollars = Total Cost for the Fiscal Year 1988

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PULSE ATTITUDE AND CONTROL SYSTEMS (AACS)

1. INTRODUCTION

Pulse is a three-axis stabilized spacecraft utilizing solid state sensors and reaction jets to provide control moments. The control hardware utilizes advances in microprocessor accuracy capability, reliability and efficiency.

2. AACS FUNCTIONS

For the purposes of identifying AACS requirements, three main mission phases are distinguished. These phases and their associated AACS tasks are listed below.

GEOSTATIONARY EARTH ORBIT (GEO)

The launch vehicle and upper stage will insert PULSE into GEO. During this phase the deployment, of the booms, the spacecraft attitude, and it's insertion into it's inter planetary trajectory will all be controlled from the ground via the low gain antenna.

CRUISE PHASE

During the cruise phase of the mission, both the determination and control of the spacecraft attitude will be autonomous. The main spacecraft control requirement is that of maintaining the antenna pointing within one degree of earth as the spacecraft progresses along it's trajectory. This task can be viewed as a continuous maneuver of low angular rate or as stabilization of the spacecraft in a non-inertial reference frame.

ENCOUNTER PHASE

The accuracy required of the AACS is much greater as it now must control the scanning of the scientific instruments. The antenna pointing requirement must be maintained both during and after the encounter while stored data from the science instruments is transmitted to earth.

3. DESIGN OF AACS

The primary movers in design of attitude determination and control systems are reliability and low cost. The emphasis of current research in spacecraft attitude determination and control is in the area of control systems, where much of the fundamental work remains incomplete (Ref. p 714-715). Therefore, in the area of attitude determination, use of off the shelf components that have been flight tested on interplanetary missions of long duration, is maximized. Some of the components, such as rate integrating gyros and servomotors will be directly implemented. In other cases, such as that of optical sensors, hardware that is already under development will be utilized. This use of developing technology is justified where it makes use of advance in solid state technology to improve performance yet can still be integrated into flight tested attitude determination systems

(Ref. 2). The rapid advances in microprocessor technology that have taken place since the design of the last interplanetary probes will also be made use of. Modern microprocessors once space hardened, will permit the implementation of control laws which greatly improve performance parameters of the AACS (Ref. 3). The computing power and memory capability available will permit utilization of artificial intelligence (AI) applications such as expert systems. While their low processing power precludes their use in low level control loops they will be useful in the areas of system checkouts and trouble shooting (Ref. 4). Previous missions have employed a fault recovery ability which monitors the system and placed the spacecraft in a safe mode in the event of failure. However, ground control was necessary to reconfigure and reprogram the system before the mission could resume. An expert system would be able to not only diagnose the fault, but to make and implement decisions to rectify the failure.

ATTITUDE DETERMINATION

Figure 1 is an overview of sensor types (Ref. 4). The relevant criteria are that the sensors chosen must be applicable to three-axis stabilized spacecraft in eccentric orbits and have at least medium accuracy. The sensors to be utilized on PULSE are the Yaw Sun Sensor (YSS) and the Solid State Detector (SSD) star tracker.

The Yaw Sun Sensor under development utilizes a charge coupled device (CCD) detector. This sensor is easily integrated

Paper Reference	Sensor	Targets				Orbits				Mode		Accuracy			Notes
		Earth	Sun	Stars	Planets	Low Orbit	Transfer	Geostationary	Eccentric	Spinning S/C	3-Axis Stabiliz	Low	Medium	High	
21	HORIZON CROSSING INDICATOR	*				*	*			*		*			FLOWN
22	DUAL BEAM IR EARTH SENSOR	*					*	*		*			*		FLOWN
23	IR FAN BEAM SENSOR	*	*		*		*		*	*			*		QUALIFIED
24	IR PITCH / ROLL SENSOR (STATIC)	*						*			*		*	*	FLOWN
25	IR PITCH / ROLL SENSOR (SCANNING)	*						*			*		*	*	FLOWN
26	LOW ORBIT IR EARTH SENSOR	*				*					*	*	*		DEVELOPMENT
27	YAW EARTH SENSOR	*				*					*		1) *		DEVELOPMENT
28	ALBEDO SENSOR	*	*		*		*		*	*			*	*	QUALIFIED
31	FAN BEAM SENSOR		*			*	*	*	*	*			*		FLOWN
32	HIGH ACCURACY SUN SENSOR		*			*		*	*		*			*	DEVELOPMENT
33	YAW SUN SENSOR		*			*		*	*		*		1) *		STUDY
41	STAR MAPPER			*		*		*	*	*			*	*	QUALIFIED
42	STAR TRACKER (IDT)		*	*	*	*		*	*		*			*	QUALIFIED
43	STAR TRACKER (SOLID STATE DET)			*	*	*		*	*		*		*	*	DEVELOPMENT
44	RATE SENSOR			*		*		*	*	*			*		STUDY

1) TARGET TO BE VERIFIED

Fig. 1 OVERVIEW OF SENSORS PROPERTIES

REF 5

into optico-inertial systems. In addition sensors being developed on this baseline can be radiation hardened. and can utilize hybrid electronics to minimize weight and reduce dimensions. Finally it may be employed as a high sensitivity sun sensor to aim at sources of light much fainter than the sun (Ref. 6). In this capacity as a planet sensor it may be used to generate error signals to drive the servomechanism which controls the instrument scanning platform.

The Sun Sensor provides only the orientation of a sun pointing vector to the spacecraft. A star tracker which tracks the star Canopus, near the south ecliptic pole provides additional input which uniquely fixes the spacecraft attitude. Such sun-canopus systems have been flown on the mariner, surveyor and lunar orbiter missions (Ref. 1 pp.189). The CCD star tracker to be used features inherent geometric stability, low voltage operation and high reliability (Ref. 5). Because the angular displacements between the earth, sun and canopus are small and the high gain antenna must be earth pointed. The optical sensors must be place on the antenna rim to avoid blocking their field of view.

Rate integrating gyros can be used off the shelf and be integrated with the optical sensors into an optico-inertial attitude measurement system. The gyros will be placed on the body of the spacecraft and on the scan platform to measure pointing of the science instruments.

The gyros will be used for short term attitude measurement and the optical sensors will be used for long term measurement and calibration of the gyros.

CONTROL HARDWARE

A high precision microprocessor implemented control system accepts the angular displacement, rate and disturbing torque from the sensors above. The control law produces time optimal recovery from large angle errors and can obtain stable control with disturbing accelerations approaching the control torque. The control law also incorporates fuel optimal slewing through unlimited angles. Steady state limit cycles in the arc-second region are attainable for precise control during the encounter phase (Ref. 3).

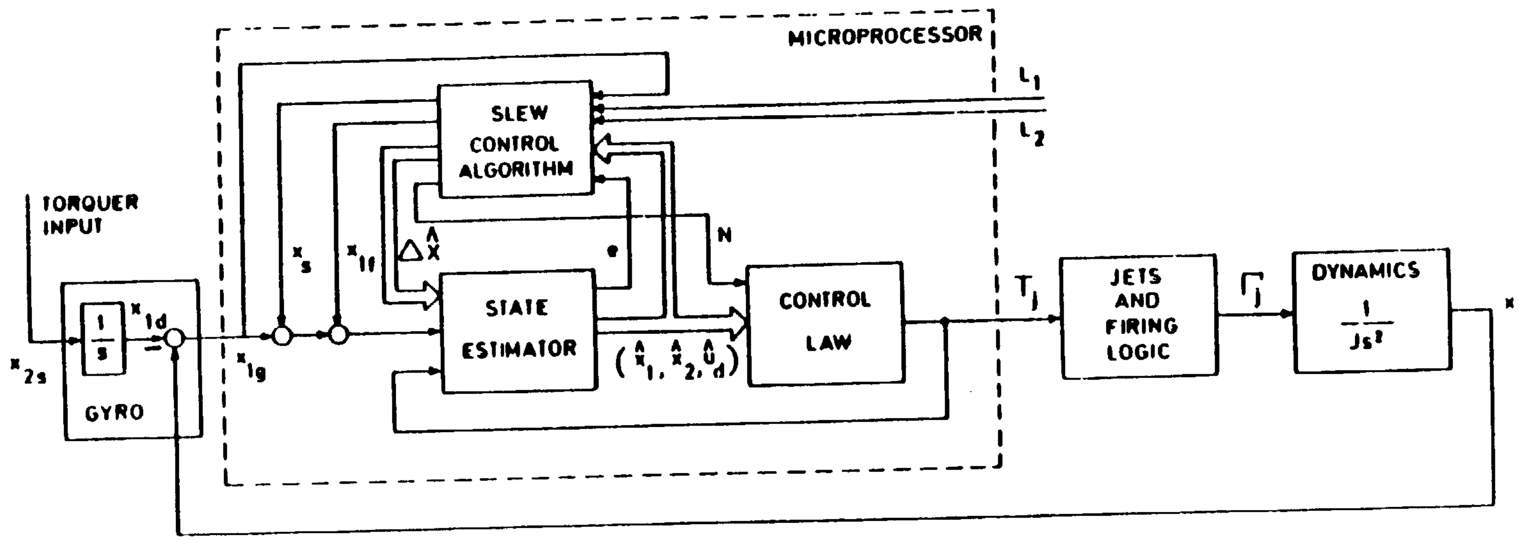
Fig 2 shows a block diagram of the control loop. The state estimator generates a state vector consisting of angular rate, displacement and disturbance torque. The slew algorithm optimizes fuel consumption. The control law controls timing of jet firing.

For the PULSE mission it is required that the microprocessor also generate the command input. This requires on board calculation of the proper earth pointing angle at all stages of the mission. Another difficulty may arise in controlling. The scanning of the science from integrated gyro and accelerometer data. A separate planet sensor on the scan platform may be required to provide an error signal to the servomotor which controls the platform.

Torquer Selection

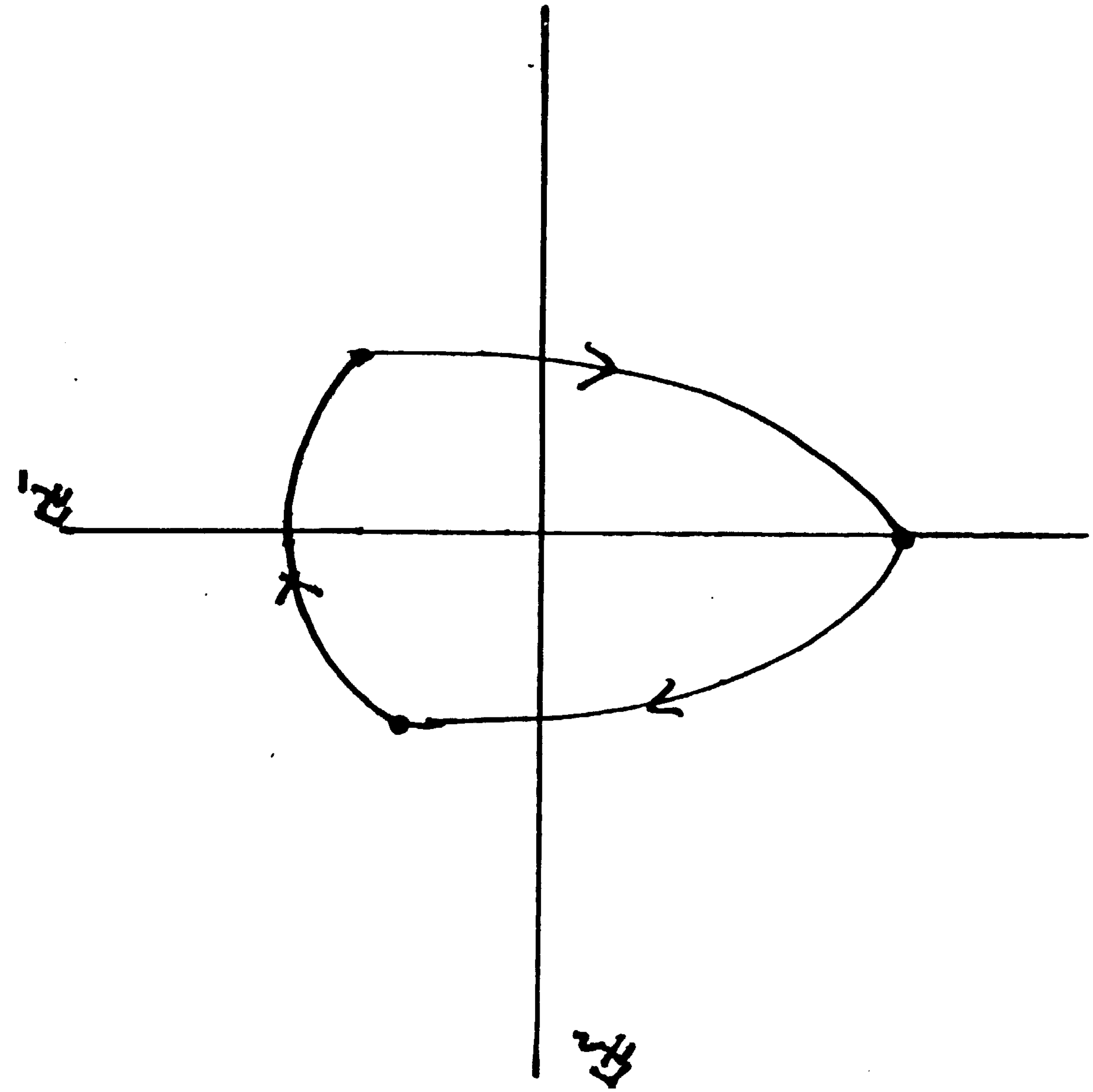
There are two types of torquers available for a field free

Attitude control system for microprocessor implementation



(Fig. 2) Control loop

REF 3



(Fig. 3) Ideal limit cycle

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environment: momentum exchange and mass expulsion. Gas jets are the only viable alternative for missions of this duration (Ref.7) estimates of spacecraft moment of inertia and an assumed impulse bit of .005 s and a limit cycle deadband of 1 degree were used to estimate total impulse required for maintaining antenna pointing during cruise. This assumes that any maneuvering requirements are negligible compared to the essentially continuous limit cycle (Ref. 8)(Appendix A). The total impulse led to a trade study among possible propellants. Cold gas, hydrazine and bipropellants were the candidates. Bipropellants and augmented hydrazine were eliminated because of the required complexity. Fig 4 shows a trade analysis for the propellants. This shows the optimum propellant is hydrazine.

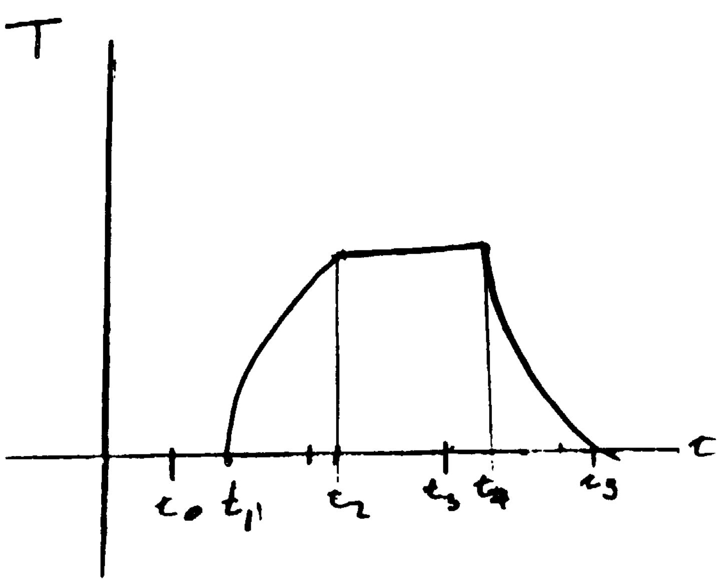
This analysis assumes a torque free environment. To check the validity of this assumption an estimate of the maximum solar torque was made. This torque was shown to be negligible when compared to the control torque thus justifying the assumption (Appendix B).

Other possible errors are introduced into the analysis by changes in thruster performance over time, propellant sloshing in the tank, and inaccurate modeling of thrust profile.

	cold gas	hydrazine	Aug. Hydrazine	Bi propellant
Isp	~60	~100	~200	~300
Minimum Imp. bit	~.0001	~.001	~.01	.015
Corrosive Exhaust	~ no	yes	no	no
Added Complexity	no	no	yes	yes
Mp				

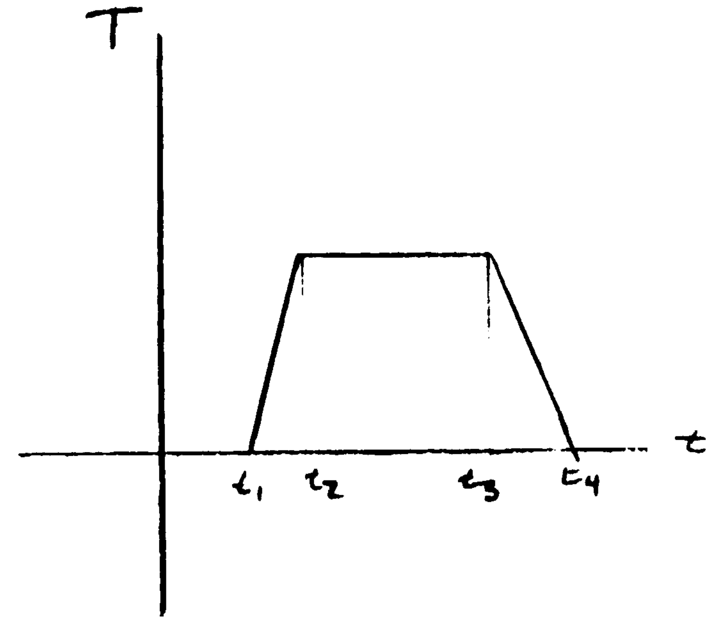
using L=2m Theta= 1 degree I=2000 N*s T = 5*E8

Figure 4



Actual Thrust Profile

(Fig. 5)

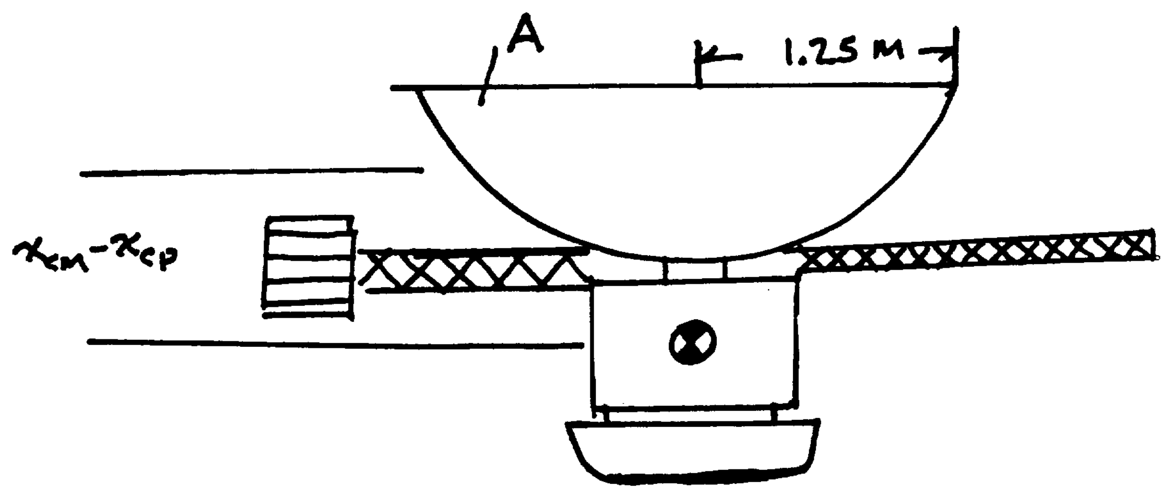


Trapezoidal approximation for thrust profile

REF 1.

APPENDIX B

$$T_{SR} = C_A P_S A (x_{cm} - x_{cp})/2$$



$x_{cm} - x_{cp} = .4M$

$A \cong \dfrac{\pi (1.25M)^2}{3} \cong 1.6 M^2$

$P_s = 4.62 \times 10^{-9} \frac{N}{M^2}$

$C_A = .7$

$$T_{SR} \cong 10^{-8} \ N \cdot M$$

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3.0 Science

3.1 Mission Objectives

The primary objective for this unmanned, scientific study of Plutonian space is to expand upon our current knowledge of the Pluto-Charon system. This will be accomplished by obtaining and returning information concerning our three scientific objectives which are listed and prioritized in Table 3.1. Each of these objectives will be investigated through the use of the PULSE Experimental Package and the radio science equipment aboard the probe.

Table 3.1
Scientific Objectives of the PULSE probe
1. Investigate Plutonian Characteristics
2. Investigate Satellite Characteristics
3. Investigate Planetary and Interplanetary Particles and fields.

The investigation of each of these scientific objectives is the major concern of this mission. Since no probe has visited Plutonian Space, little is known about the planet Pluto or its satellite Charon. However the scientific community has conducted recent studies concerning the Pluto-Charon system. The knowledge gained from these studies was one of the determining factors for instrument selection aboard the PULSE probe. Although these studies have given us some new information, none of the information can be considered conclusive until a closer investigation is conducted.

3.2 Science Objectives

3.2.1 Plutonian Characteristics

One characteristic of Pluto which must be investigated is the atmosphere. Astronomers have found that Pluto does have a dilute atmosphere which extends several hundred kilometers above the planet's surface(Ref.2, p.45). This complex atmosphere is believed to contain heavier molecules than methane which was previously believed to make up the entire atmosphere(Ref.7, p.326). Other atmospheric properties which must be investigated include, measurements of temperatures and pressures at various altitudes and cloud characteristics (if present).

A second characteristic which needs investigation is the surface characteristics of the planet. Earth observations have shown the existence of polar ice caps at the poles of Pluto which are believed to be composed of methane ice (Ref.13, p29). This possibility along with other surface features need investigation.

Other areas of interest include, mass, shape, density, orbit characteristics and composition. By investigating these areas, we hope to gain improved knowledge of the planet Pluto.

3.2.2 Charon Characteristics

Pluto is believed to have only one orbiting natural satellite named Charon. The characteristics which need to be studied are relatively the same ones found in the previous section. One difference is that the amount of methane on Charon is believed to be much less than on Pluto. Charon is believed to be composed of water ice and not methane ice.

3.2.3 Planetary and Interplanetary Particles and Fields

One interesting area which falls under this category is the gravitational and magnetospheric interactions of the Pluto-Charon system. Charon is relatively large compared to Pluto. It is because of this that the Pluto-Charon system was thought to be one planet which led to incorrect measurements. There is no other planet-satellite system known so it seems very important to study these interactions.

Other areas shall include investigation in; charged particle environments, wave particle interaction, solar wind and cosmic rays.

The instrumentation used in most of these measurements is located on the probe's scientific boom which allows for measurements in the interplanetary environment as well as the planetary environment.

3.3 Pulse Experimental Package

The Pulse Experimental Package(PEP) will consist of five remote sensing instruments and four particle and field instruments and radio science. Each of these instruments is listed in Table 3.2. Also listed in this table are mass and power specifications. The total PEP weight is approximately 94.9 kg and the approximate power they consume is 80 W. The selection of these instruments was based on their ability to investigate the scientific objectives.

3.3.1 REMOTE SENSING INSTRUMENTS

IMAGING SCIENCE SUBSYSTEM

The Imaging Science Subsystem(ISS) was selected because it has a much higher resolution (1024 x 1024 pixels) than any of its predecessors(Ref.5, p.9). Many of the instrument's components are just improvements upon the camera systems of its ancestors. This instrument also offers data compression and storage which will be necessary because of the large amount of data that will be obtained during our flyby of the Pluto-Charon system since most of the investigation will be carried out at this time. The data rates of the ISS are selectable. They range from 6.2 kbps to 350 kbps(Ref.5, p.10).

The ISS offers the opportunity to view the Pluto-Charon system. The characteristics of Pluto and Charon will be investigated with the ISS. We also will be able to investigate

Table 3.3

NAC and WAC Optics

<u>NAC</u>

Type:	Ritchey Chretien with three field correctors
Focal Length:	2000 millimeters
Focal Ratio:	f/10.5
Spectral Range:	200-1100 nanometers
Resolution:	The resolution per pixel will be six microradians square.
Coverage:	The field of view will be 0.35 degrees square.

<u>WAC</u>

Type:	Refractor
Focal Length:	250 millimeters
Focal Ratio:	f/4.0
Spectral Range:	350-1100 nanometers
Resolution:	The resolution per pixel will be 48 microradians square.
Coverage:	The field of view will be 2.8 degrees square.

-Ref. 3, p. 8

the Pluto-Charon interactions and determine other areas that may be of interest in Plutonian Space.

This instrument, which is essentially the same as the ISS that will be flown on the Cassini and CRAF missions scheduled to be launched in 1995 and 1996, is composed of two cameras, a Narrow Angle Camera(NAC) and a Wide Angle Camera(WAC). The cameras will have a spectral range which is extended visible and they well operate at a temperature slightly below room temperature. The components of these two cameras include a dust cover, hood, optics, filter mechanism, shutter detector head and radiator. The dust covers are a method of protection for the optics which will be motor activated. The hood is designed to also protect the optics and reduce the glare. The optical parameters for both the NAC and the WAC are listed in Table 3.3(Ref.5, p.9).

The filter mechanism of the cameras was derived from the Hubble Space Telescope. Unlike Galileo's filter mechanism that had a maximum of seven positions, Pulse's filter mechanism has a maximum of 36 positions. The two filter wheels of the NAC and the WAC contain 22 filters and 14 filters respectively(Ref.5, p.10).

The shutter technology oriented from shutters on Voyager and Galileo. It consists of a dual blade focal plane which may operate in either direction. The lower limit on exposure time is .005 seconds and no limitation on the upper limit. One advantage of this system is that both shutters may be activated simultaneously (Ref.5, p.9).

The detector head of the ISS contains the Charge Coupled Device(CCD), driver, thermal control unit and signal chain circuits. This electronic module is common to both the NAC and the WAC. Other components of this module include: 1) a microcomputer 2) memory 3)power supplies 4) engineering sensors 5) image data multiplexer 6) square root processor 7) image memory 8) image data compression 9) bus interface unit(Ref.5, p.10).

The radiator of the ISS is responsible for cooling the CCD to temperatures approximately -80 degrees Celsius(Ref.5, p9).

NEAR INFRARED MAPPING SPECTROMETER

The Near Infrared Spectrometer(NIMS) is one of the instruments that is aboard the spacecraft Galileo. This instruments unique ability of combining spectroscopy and imagery in one instrument makes it a prime candidate for PEP. Another reason for its selection is that it can monitor both methane and water vapor which are believed to be present on Pluto and Charon respectively (Ref.8, p.207).

The objectives of NIMS fall into the first two scientific objectives. NIMS will be used for both the investigation of geological properties of both Pluto and Charon. NIMS will accomplish this objective by investigating surface features and surface composition through surface mapping and infrared spectral investigations.

NIMS will also investigate atmospherical properties. Goals of this investigation include information about atmospheric structure and composition. Also investigations about the existence of clouds, cloud properties and temperatures at various altitudes will also be conducted. Table 3.4 lists a summary of specifications for this instrument.

The NIMS will be placed on the scan platform. It is protected against contamination by covers and heaters. It also has a passive radioactive cooler which will keep the instrument at is operation temperature of 80 K(Ref.1, p.201).

PHOTOPOLARIMETER-RADIOMETER

Photopolarimeter-Radiometer(PPR) was also an instrument flown on the Galileo spacecraft. It was selected primarily because of ability to measure its intensity and linear polarization of scattered sunlight in the spectral region where methane strongly absorbs radiation(Ref.19, p.128). It is also unique because of the combination of three separate experiments it may conduct; photometry, polarimetry and radiometry.

The objectives of this instrument is as described above to measure the intensity and linear polarization of scattered sunlight in the narrow spectral bands.

Another objective of the PPS is the measurement of thermal infrared radiation. This may only be investigated if clouds do exist in the Plutonian atmosphere since the radiation is believed to be emitted primarily from cloud particles.

Some atmospheric properties well also be investigated. This experiment is mostly concerned with the particles in the atmosphere and their distribution.

Table 3.4

NIMS Instrument Characteristics

Angular Resolution:	0.5 mrad x 0.5 mrad
Angular Field:	10 mrad (20 pixels) x 0.5 mrad (1 pixel)
Spectral Range:	0.7 - 5.2 micrometers
Spectral Scan Time:	4-1/3 seconds (20 pixels, 204 wavelengths)
Telescope:	23 cm diameter f/3.5 Ritchey - Chretien wobbling secondary for spatial scan, 800 mm equivalent focal length
Spectrometer:	40 lines/mm plane-grating spectrometer, f/3.5 Dall Kirkham collimator f = 400 mm, f/1.86 wide-angle flat-field camera f = 210 mm
Detectors:	InSb (15), Si (2), discrete elements, quantum efficiencies = 70-80%, noise equivalent power = 10^{-14} watt, $D^* = 3 \times 10^{13}$ cm $\sqrt{Hz}$ $watt^{-1}$
Signal-to-Noise:	100:1 (0.075 albedo surface at 3 micrometers)
Mass:	18.0 kg
Power:	12 W (average), 13 W (peak)
Date Rate:	11.52 kbps
Data Encoding:	10 bits

-Ref. 1, p. 201

There are several different channels for the PPS the "polarimetry channels are centered at 4100, 6780, and 9450 and the photometry channels are centered at 6180, 6330, 6460, 7980, 8300, 8410, and 8920 angstroms. When the instrument is used for radiometry the infrared channels are centered below 4 micrometers at 17, 21, 27.5, and 37.5 micrometers, and above 42 micrometers." (Ref.19, p.129)

There are two operational modes, a cycle mode and a radiometry mode. The cycle mode rotates the filter wheel allowing each channel to transmit at least once every 18 seconds. The radiometry mode rotates the infrared filter wheel back and forth.

The PPS weighs 4.8 kg and has both a replacement heater and a sunshade as safety features(Ref.19, p.129).

ULTRAVIOLET SPECTROMETER

The ultraviolet spectrometer was selected for determining the composition and structure of the planet Pluto and its satellite Charon.

A secondary objective of this instrument is to determine the properties of the upper atmosphere. Although Pluto's atmosphere may not be as large as that of Jupiter, there is a possibility of molecular absorption features and auroral zone emissions that are believed to be common among planets with large atmospheres. Through airglow and occultation modes we hope to determine both the atmospheric structure and the atmospheric composition.

This Galilean successor will consist of a 250 mm-aperture

Cassegrain telescope, a 125 mm focal length Ebert-Fastie monochromator, three detectors and control logic. The UVS weighs approximately 4 kg and consumes 5.33 W. The wavelengths covered by the UVS range from 1100 to 1400 angstroms(Ref.19, pp.130-131).

The UVS also has flexibility. It may take data at a fixed wavelength or it may change the wavelength every 0.0007 second. It is not limited to these two modes, however. Other variations may be programmed into the microprocessor of the UVS (Ref.19, p.131).

3.3.2 PARTICLE AND FIELD INSTRUMENTS

MAGNETOMETERS

The magnetometers that were selected for PEP are actually the same magnetometers used aboard the Voyagers. They were selected because of their ability to measure fields ranging from 0.006 gamma to 20 G(Ref.4, p235). This wide range of field measurements will be needed to measure the fields in both the Plutonian and interplanetary environments. The fact that the PULSE probe is three-axis stabilized, like Voyager, also gives reason for this selection.

The magnetometers that have been selected are two Low Field Magnetometers(LFM) and two High Field Magnetometers(HFM). This redundancy makes the system reliable in the event that one of the magnetometers does not function properly. The magnetometers purpose is to study the planetary and interplanetary particles and fields. These objectives are described as follows:

1) Investigate Pluto-Charon magnetospheric interactions.
2) Measure the magnetic field of Pluto and Charon.
3) Measure interplanetary magnetic fields
4) Determine magnetospheric interactions with solar wind,
cosmic rays and plasma waves.
5) Use observations to make further observations.
6) Search for interaction between interplanetary and interstellar media.

The LFM and the HFM are located on the particle and field boom. The placement of these magnetometers will be proportionately the same as the ones on the Voyager missions. There will be one LFM located at the outboard end of the boom and the other LFM will be placed approximately at the center of the boom. The two HFM will be located near the inboard end of the boom approximately one meter apart. This placement allow for some measurement correction factors due to the spacecraft's magnetic field(Ref.4, p.247).

The range of the measurements as state earlier is fairly large. The LFM range is ±8.8 gamma to ±0.50 G and the HFM range is ±0.50 G to ±20 G with uncertainties of ±2.2 milligamma to ±12.2 gamma and ±12.2 gamma to ±488 gamma respectively. This total ±20 G range has a 12 bit digital resolution(Ref.4, p.236).

As the probe increases its distance from the sun, the data rate will not vary greatly because of the data compaction modes of the instrument(Ref.4, p254).

COSMIC RAY DETECTOR SYSTEM

Like the magnetometers of the PEP, The Cosmic Ray Detector

System(CRS) selected for PEP has also flown on the Voyager missions. This instrument was selected because Earth-based observations show that something is blocking the light during Pluto's occultation. There are beliefs that this "extinction layer" is produced by particles which originated from cosmic rays(Ref.13, p.29). Therefore the CRS investigation may enhance our knowledge of both cosmic rays and the components of the Plutonian atmosphere.

The CRS objectives fall in the category of planetary and interplanetary particles and fields. These objectives may be almost exactly compared to those of the Voyager CRS objectives. There only difference is the planet that is being targeted. Below is a list of the objectives of the Voyager mission from the Flight Science Office Science and Systems Handbook with the appropriate modifications for the Pluto mission.

1) Measure the energy spectrum of electrons 3-110 MeV.
2) Measure the energy spectra and elemental composition of all cosmic ray nuclei from H through Fe over an energy range from approximately 1-500 MeV/nuc.
3) Provide information on the energy content, origin, acceleration process, life history and dynamics of cosmic rays in the galaxy and contribute to an understanding of the nucleosynthesis of elements in cosmic ray sources.
4) To provide information on the transport of cosmic rays, Plutonian electrons and low energy particles over an extended region of interplanetary space.
5) Measure the three-dimensional streaming patterns of the nuclei from H through Fe and electrons over an extended range.
6) Measure particle charge composition of the magnetosphere of Pluto and Charon(Ref.17, p4.1)

One may say that these objectives, inherited from the Voyagers, are still of great importance to the scientific community.

The CRS is composed of three systems; the High Energy Telescope System, the Low Energy Telescope System and the Electron Telescope System. These three systems share some common electronics and are responsible for the above objectives. The nuclei charge and energy spectra may be determined by these instruments for elements with atomic numbers from 1 to 30 and energy ranges of 1 MeV to 500 MeV for H and 2.5 MeV to 500 MeV for Fe. For isotopes the range of atomic numbers is 1 to 8 with an energy range of 2 MeV/nuc. to 75 MeV/nuc. Finally, the range of atomic numbers of anisotropies is 1 to 26 with an energy range of 1 MeV to 150 MeV for H, 2.7 MeV to 500 MeV for Fe and 3 to 10 MeV for electrons (Ref.4, p.365).

PLASMA INSTRUMENT

The Plasma instrument(PLS) that has been selected was flown aboard the Galileo Spacecraft. It was selected because of its energy/unit charge and the decreased temporal resolutions for obtaining electron and positive ion spectra. The plasma instruments of the Voyagers and the Pioneers don't even approach the values of the PLS.

The objectives of this mission are also of the particle and field type. These objectives include measurements of the plasma properties in solar wind, assessments of composition, energy, intensities and three-dimensional distribution of low energy particles.

The PLS is composed of the following:

1) Two electrostatic analyzers that measure the energy/unit charge of electrons and positive ions.

2) Seven sensors that determine electron intensities.
3) Seven sensors that determine positive ion intensities.
4) Three mass spectrometers that determine the composition of ions(Ref.19, p.133).

The PLS capabilities range from 1 Vto 50,000 V in 64 different passbands. The PLS also contains software which permits ground command alterations to the instruments commands. The instrument weighs 12 kg and will be mounted on the science boom of the PULSE probe(Ref.19, pp.133-135).

ENERGETIC PARTICLE DETECTOR

Another instrument selected from the Galilean payload is the Energetic Particle Detector(EPD). It was selected because of the need for measurement of high energy particles in the magnetospheres of Pluto, Charon and interplanetary space. Although the PULSE probe is three-axis stabilized, we should still be able to obtain a great deal of data about the high energy electrons, protons and heavy ions even without sweeping motions.

The EPD is made up of two subsystems, a Low Energy Magnetospheric Measuring System(LEMMS) and a Composition Measuring System(CMS), formed by two separate telescopes(Ref.19, p.136).

The LEMMS consists of two components. The first component is an ion telescope with two solid-state detectors. One detector, the low field detector covers an energy range of 0.02 MeV to 3.4 MeV. The other detector will be used for the definition of additional electron, proton, and alpha particle channels. The second component of the LEMMS is a magnetic electron spectrometer with two detector pairs. These detector pairs span a range of 0.015 MeV to 0.20 MeV and 0.10 MeV to 1.0 MeV(Ref.19, p.136).

The CMS components will be used for the measurement of composition, energy spectra and pitch angle distributions of the high energy ions. These components are the CMS telescope and nine detectors(Ref.19, p.136).

The EPD weighs 9 kg and will also be located on the science boom(Ref.19, p.6).

PLASMA WAVE SUBSYSTEM

The last particle and field instrument is the Plasma Wave Subsystem(PWS). The PWS was selected because of the importance of plasma wave investigations.

These investigations include wave particle interactions and their effects on the Pluto-Charon system and measurements of spectral characteristics of electric and magnetic fields in the range of 5 Hz to 5.65 MHz. We will also be able to distinguish the difference between electrostatic and electromagnetic waves(Ref19, p.137).

There are two sensors of the PWS. The first is a 6.6 meter electric dipcle antenna which has two tapered graphite epoxy elements mounted at the end of the magnetometer boom. The other sensor is a search coil magnetic antenna. This antenna consists of two high-permeability rods, 26.6 and 27.5 cm long. The low frequency search coil has a winding of 50,000 turns of 0,07 mm diameter copper wire and a frequency range of 10Hz to 3.5 kHz. This search coil must be mounted parallel to the electric antenna. The high frequency antenna has a winding of 2,000 turns of 0.14 mm copper wire and a frequency range of 1 Hz to 50 kHz. This search coil must be mounted perpendicular to the electric antenna. There will also be a preamplifier mounted near the search coil to provide a low impedance to the electronics(Ref.19, p.136).

The processing of the signal received from the sensors may be processed by a low-frequency spectrum analyzer, a medium-frequency spectrum analyzer, a high-frequency spectrum analyzer and a wideband waveform receiver. The fastest measurements are provide by the wide band waveform receiver(Ref.19, pp.136-137).

3.4 CONCLUDING REMARKS

The Objectives in this subsystem report are by no means the only investigations that will be conducted. There are indeed some that were not mentioned and some that will not materialize until a probe visits Plutonian space. The purpose of this mission is to observe as much as possible so as to enhance our knowledge for further scientific investigations.

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4.0 Introduction to command, control, and communication

The command, control, and communication subsystem has several design requirements which include:

1) minimization of cost and weight
2) maximization of performance of reliability, performance, and simplicity
3) use of off-the-shelf hardware
4) use of technology before 2000
5) application of AI, if applicable
6) sufficient life time to carry out the mission

The priority that overshadows all of them is cutting the cost of the mission. As far as incorporating new technology into PULSE, we are taking a conservative approach. Proven designs will be chosen over new technology, except in the case where it would be more cost effective to use the latter. When possible, past deep space probes will be used as a prototype due to reliability and cost requirements.

4.1 Antenna System

Reliability is the dominating factor when discussing antennas. Voyager 2 and Galileo will be used as the prototype for this subsystem due to the fact that proven techniques enhance reliability and lower the overall cost of the vehicle. A high-gain circular parabolic antenna will be used because this shape optimizes the gain. A low-gain antenna will be included mostly for communication when near earth for attitude articulation and control reasons, since the high gain antenna can not be used these ranges.

4.1.1 High-gain antenna

The high-gain antenna (HGA) meets all of the requirements stated in the RFP. HGA's are the most cost efficient antennas because they use off- the- shelf hardware. They are reliable and their performance is well known because they were used in many previous spacecraft and are based on already proven technology. This antenna was chosen because it meets all of the applicable requirements.

4.1.2 HGA trade- offs

The most important trade- off in HGA's is the power-gain tradeoff. Gain is increased as the antenna size is increased, this also result in a higher weight. If more power is needed the weight also increases because the weight of the RTG's must be greater. This is accompanied by the requirement of minimizing the weight of the antenna. The maximum of power- gain trade- off occurs when the product results in minimum weight.

4.1.3 A Look at Laser Communication

Optical communication could result in 47 bps from 50 AU from a mass of one kilogram. There are many reasons that this technology cannot be justified given the requirements from the RFP. Optical communication is in the high- risk department as of now because it has not been deep space tested yet. Plans for testing are planned but it is doubtful optical communication will be ready for deep space missions before the year 2000. This antenna would also require that a 20 m receiving antenna be put in orbit, since optical communications have a severe limiting factor of weather dependence.

4.1.4 Size of High-gain Antenna

The size of the high-gain antenna is going to be 2.5 meters in diameter. This is the maximum size that the launch vehicle will allow. This is smaller than either Voyager or Galileo, which are 3.66 and 4.8 meters in diameter consecutively. This decrease in size can be accounted for in several different ways including increase of gain in the antenna, improvements in the Deep Space Network (DSN), and improvements in the encoding and decoding of data.

4.1.4.1 DSN

The DSN applies the technique of antenna arraying. It includes many large antennas from all over the world.

LOCATION	DISH SIZE	X-BAND REC'V
GOLDSTONE	34m	YES
	70m	YES
	34m	YES
V.L.A.	27x 52m	YES
CANBERRA	34m	YES
	70m	YES
	34m	YES
USUDA	64m	NO
PARKES	64m	YES
MADRID	34m	YES
	70m	YES
	34m	YES

Possible improvements to this network include changing the Usuda antenna so it is capable of X- band reception. Increasing the size of the 64 m antennas to 70 m. Adding a 34 m antenna at the Parkes and Usuda location would add 1.1 db each. General Electric has suggested that the masers be replaced by high- electron- mobility transistors, which would cost a third as much to operate and a quarter of the implimentation cost. These improvements could led to 3-4 db increase in gain.

4.1.4.2 Encoders and Modulators

The effectiveness of digital satillite communications systems (DSCS) will increase when well chosen modulation and noise- immune encoding methods are used. The PSK-4-CC was found to to be a good method. Both the frequency effectiveness and energy can be increased. Power gains may reach 5 db and specific rates can increase by a factor of 1.5. From a costing side, increasing the efficiency of the encoder is less expensive than increasing antenna size or transmitted power, or increasing the receiver noise sensitivity.

4.1.5 Amplifier

The amplifier used will very from the one in Voyager 2, but will be similar to the one used for the generic Mariner Mark 2 (MM2) design. This design includes the use of gallium arsenide field-effect transistors in the amplifier to produce an output of 5.6 W. This value could be raised to about 10 W with only minor modifications. This application of solid state electronics would cost less than half that of the system used in the Voyagers which featured traveling-wave-tube-based amplifiers.

4.1.6 Radio-frequency Subsystem

PULSE's high-gain antenna will maintain communication with Earth in only X- band, as in the case of CRAF. S- band communication was used in the Voyagers because not all ground stations could not handle X- band when they were launched. Now, all stations except the Japan based antenna are capable of X- band communication. X- band offers better range and range- rate measurements, and greater immunity to charged particle interference. Using only one band simpifies the ground system and lowers the operational costs.

4.2 On- board Computers

Radiation- hardened versions of widely available microprocessors and integrated- circuit chips supported by well-known software development tools. Handling of scientific data during and after the mission must make use of the latest technology.

4.2.1 Lag in Technology

The computer industry is one of the most rapidly developing industries. There has been a problem with computer systems in past spacecraft due to the lag in technology because of this rapid development. This is difficult to avoid because of the time delay between deciding on a system and the actual launch date.

4.2.2 Performance Characteristics

The PULSE probe will be outdated by the time it is launched, as in the case of all spacecraft, but on- board computers need to be selected about five years in advance to develop, test, and integrate the spacecraft subsystems. A schedule and summary of major features of the PULSE computer system are listed below.

Launch date	2003
Year computer selection made	1993
Year commercially available	1990
Difference in launch and avalable	13
Microprocessor	32 bit
Performance	4 MIPS
RAM	4000 kbytes possible

4.2.3 Space Qualification of Computers

The problem with spacecraft computers is that they must be able to withstand radiation and the bombardment of high-energy particles, and operate in a highly reliable manner. NASA, Defense, and the Department of Energy are working to develop and deploy space qualified computers.

There are several space qualified computers. Sandia National Laboratory is developing a set of advanced 32- bit and 16- bit microprocessors called the SA 3300 family. The microprocessor and its associated computer hardware should be available in about four years. There is also a generic version of the 32- bit processor RH32 which will be fully developed soon.

4.2.4 Computer trade- offs

Because of size, weight, and power limitations on-board computers must be small in size, lightweight, and have low power requirements. Selecting more advanced computers for the spacecraft can result in higher development costs, but the overall result is lower overall life- cycle costs of space missions through lower software development and maintenance costs. This can be further decreased when a universal higher level languages are approved for space programs. The Department of Defense approved Ada recently. The advantage for this standardization is lower cost, lower development risks, shorter delivery schedules and ease of maintenance. To date, assembly language source coding has been used for spacecraft data processing. Sufficient support software should be available by the time PULSE is launched. The emphasis will turn from hardware to software to control the spacecraft. By putting all the sophisticated logic in software, much less hardware is needed and designers have the flexibility of reprogrammability.

4.2.5 Problem with Galileo

NASA used a RCA 1802 8- bit microprocessor which caused problems due to the limited capabilities. Its relative low speed and its limited memory increased cost because of problems with writing efficiency and maintainable software. The 32- bit processor in PULSE will allow expanded mission objectives such as acquiring and relaying more pictures faster, and allowing more autonomous operations. While scientific objectives could be reached with a less modern computer, lower cost and risks encourage its use.

4.2.6 Data Management Systems (DMS)

The DMS must regulate power management, command and telemetry, thermal regulation, and antenna control. centralization of the DMS ensures command prioritization and synchronization of resources. Using separate microprocessors and spares can result in power, weight, and code complexity to provide the necessary redundancy. The DMS may make use of a internally redundant Intel 80386 for data processing and automatic control purposes. The only problem is that it is not radiation hardened yet and may not be by the year 2000. If it is not a back-up option would be a 32- bit radiation hardened microprocessor combined with a direct memory access chip that simplifies software which is being developed by JPL.

The DMS will be similar to the ESA probe ISPM include a Central Terminal Unit (CTU), Remote Terminal Unit (RTU), Command Decoder, and data storage (a tape recorder or hard drive). The CTU controls the automatic functions and operations. The main tasks will be performed on the Intel 80386 microcomputer. The software governing articulation and control is based on the Ada language. The CTU contains a fault detector which will switch to redundant units when problems arise. The command detector that will be used is the NASA standard which is upgraded from the one used in Galileo.

4.3 Conclusion

The most important features of this subsystem is the 2.5 m high- gain antenna which will communicate with the Deep Space Network at a distance of around 33 AUs with x- band uplink and downlink and the centralized Data Management System which utilizes the Intel 80386 computer, and the Ada language for software applications.

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5. STRUCTURE

5.1 Requirements to be met by the structure:

The structure has the objective to support all other subsystems and carry them out to Pluto safely. It has to protect them from destruction or damage and also from influences which might affect the performance of those subsystems. In this context the following requirements were derived from the RFP.

use no materials available after 1990

lifetime long enough, with a safety margin

weight and cost optimization

stress reliability

stress simplicity

stress low cost

nothing should preclude other missions

interface to the launch vehicle

if necessary, on orbit assembly should be minimized

5.2 Shape and Configuration:

5.2.1 Grouping:

The structure of PULSE has to support all subsystems and meet all the different requirements from those systems. In order to comply with conflicting requirements, groups of subsystems with similar requirements have to be placed together. This subsystem grouping yielded 4 major areas with different necessary attributes:

The main body :

Requirements: provide thermal environment
support mass
radiation shielding
micrometeoroid protection
withstand launch forces

Subsystems: Communication electronics
Control electronics
Data storage
Gyroscopes
Power conditioning equipment
Fuel pumps and lines

To meet these requirements the subsystems have to be encased in a shell which will protect the inside from micrometeoroids, radiation, will not yield due to the launch forces and provide a sufficient insulation against heat loss. Conflicting requirements are here low cost and low weight against high protection and strength. Desirable is also good damping of vibrations during take off to protect the electronics from mechanical damage.

The science boom :

Requirements:	negligible magnetic and electric interference support mass provide thermal environment micrometeoroid protection
Subsystems:	magnetic field instruments particle detectors

The predominant point in this group is, that the science instruments have to be able to measure an as much as possible undisturbed environment. To keep disturbance by the electronics on board the probe as low as possible, those instruments have to be away from the spacecraft. Even though micrometeoroid protection is necessary, shielding is not feasible since that would shield off the fields to be measured also. The same applies for the heating. On one hand the electronics needs to be kept at an operating temperature, but on the other hand, heaters would create a disturbance. For these reasons, the instruments have to provide these measures themselves.

The science platform:

Requirements:	Pointability and good field of view support mass micrometeoroid protection provide thermal environment pointability
Subsystems:	Science instruments (cameras,infrared spectrometer)

Other science instruments require less shielding than the field and particle instruments. For this reason they can be mounted on the main body and micrometeoroid protection and heating can be supplied by the structure. In addition to the control electronic housed in the main body these instruments needs to be pointable and they have to have a good field of vision. This is accomplished by separating them from the main body and mounting them on a movable platform on top of the main body. To ensure the micrometeoroid protection, a steel canopy is placed over the platform. Steel has been chosen to maximize the protection since the science instruments are the essential parts of this mission. During the cruise phase it will be closed and only when PULSE approaches Pluto it tilts open. The platform will be turnable by 360 degrees and tiltable by +- 15 degrees. These values ensure that a large area can be scanned by the mounted instruments.

The power boom:

Requirements: micrometeoroid protection
allow heat radiation
support mass

Subsystem: RTG

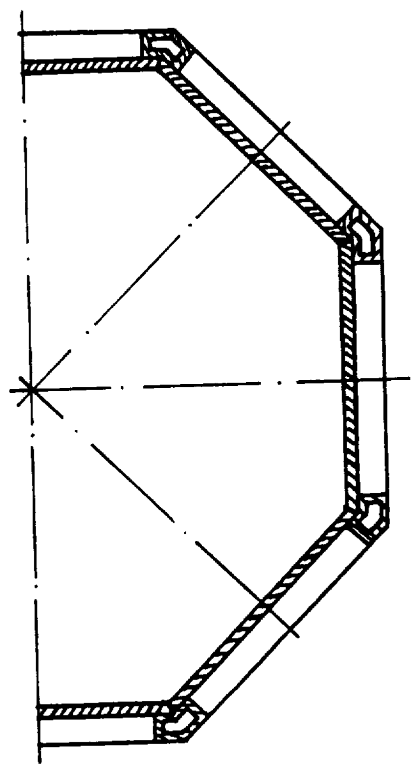
RTG's radiate a large amount of unwanted radiation which would have a negative influence on the performance of electronic equipment, this radiation has to be kept away from those instruments. It would require heavy shielding to protect the computers which would interfere with the requirement of low weight. It also would affect the necessary heat radiation of the RTG's. Thus the RTG's have to be moved away from the main body. This yields now two booms which can be spaced by 180 degrees to enhance symmetry and maximize the distance between the sensitive science instrumentation and the high radiation of the RTG's. The spacecraft body also functions as a shield. The science platform will not be operational during the cruise phase. During the flyby, the open steel canopy will be tilted in the direction to the RTG's to provide shielding.

Other subsystems:

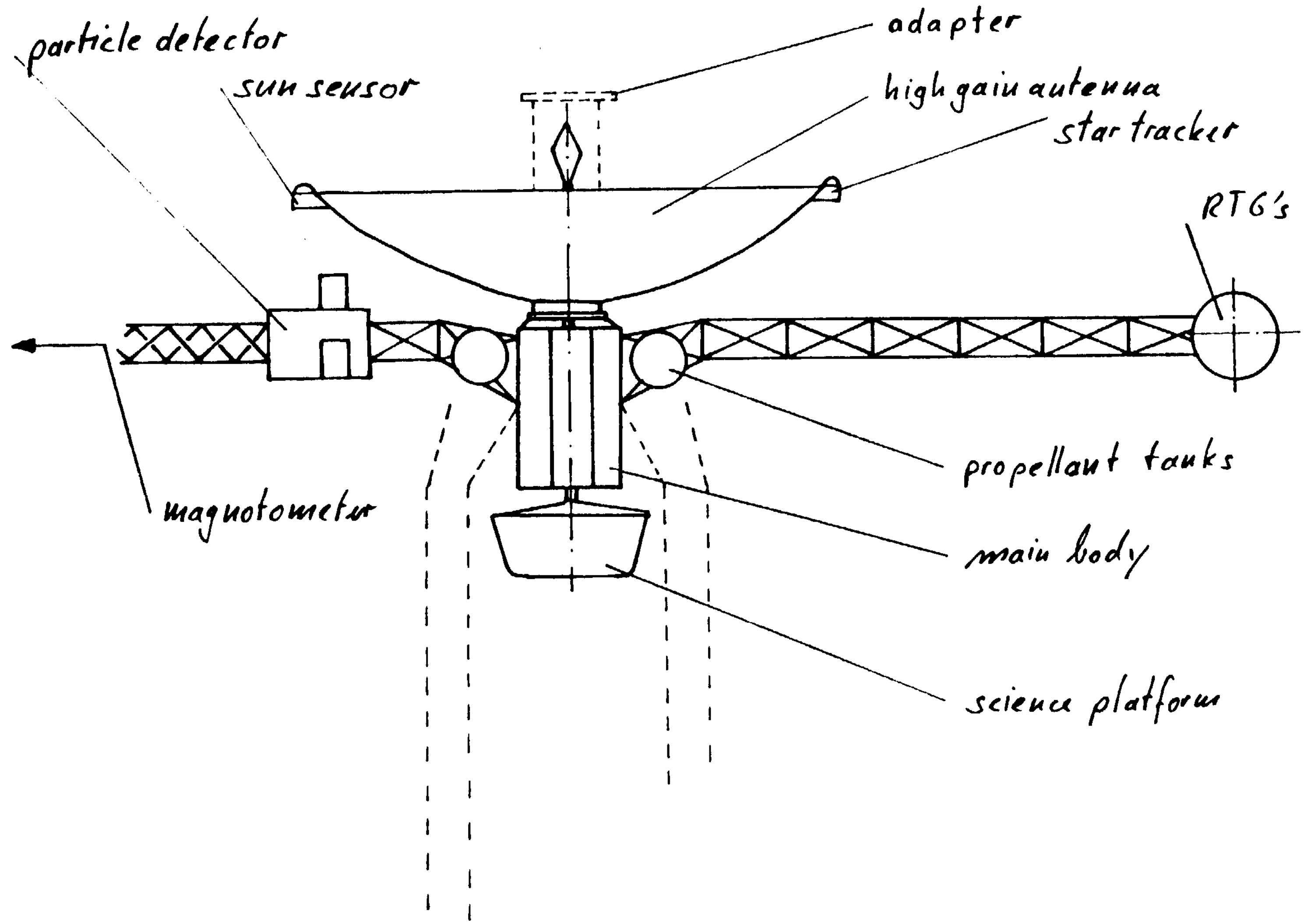
The remaining subsystems are the antenna, the propulsion tanks and the startracker and sun sensor. The predominant requirement for the antenna is, that it has to be pointed to Earth at all times. Additionally the antenna is required to function as an adapter interface with the launch vehicle. This yields, that the antenna is firmly mounted on the main body to provide the necessary support. Thus the whole body of the spacecraft will be pointed at earth.

The propellant tanks will be bought from stock and placed next to the main body on both sides of the boom structure. This will limit the volume needed for the main body and thus decrease the weight. There will be four propellant tanks and the their steel body will provide a sufficient protection against micrometeoroids.
The startracker and the sun sensor need a good field of vision to be able to scan a large area. This is accomplished by placing them on the rim of the parabolic antenna. Both have similar pointing requirements, and since the difference in angles to the sun and the earth is maximal 12 degrees in the periphery of our sun system the instruments have to provide only a small correction to their pointing. Here they also have a large angle available where no obstacles block their field of vision.

Cross Section of main body



The PULSE Space Probe



5.2.2 Shape determination

The main driver when determining the shape of the main body, is the prevention of heat loss to space. An important variable there is the surface. The smaller the surface, the smaller the heat loss. Therefore I considered shapes which allow me to have a large volume but also have a small surface area. Obviously the sphere has the highest volume to surface ratio (V/S ratio) but production and interface problems make the sphere less desirable to be used on PULSE. I then considered the cylinder. It has a smaller V/S ratio, but provides two flat interface surfaces. Looking at the amount of equipment to be mounted inside the hull it is apparent, that this is not enough. Adapters need to be installed to fit the instruments to the curved surfaces. This would increase the weight of the structure and complicate the manufacturing. From these considerations I propose a regular octagon as the shape of the main body. It has still a high V/S ratio but has flat sides so the instruments can easily be mounted.
From the volume required I derived the design sizes. This yielded a diameter of 0.5 m and a height of 0.8 m.

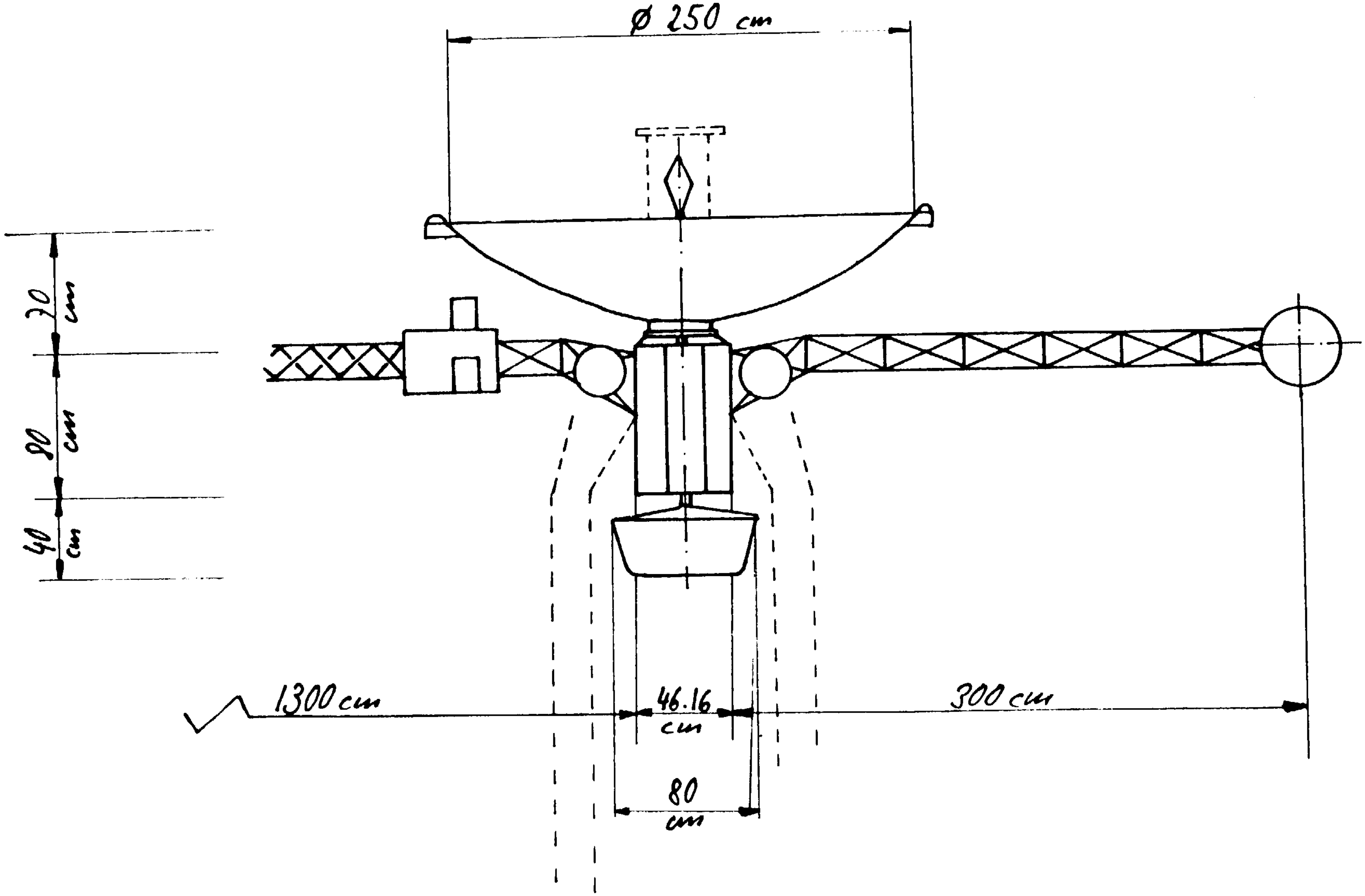
5.2.3. Configuration:

Due to the requirements of having both RTG's and highly sensitive particle and field instruments on the same craft, it is necessary to separate them as far as possible. For this reason booms need to be employed. I propose two booms, one carrying the two RTG's and the other all the particle and field sensors. This enables a 180 degrees separation which gives the maximum separation distance. This way the main body also acts as a shield in between. Since even the on board electronics interfere with those sensors, the science boom needs to be considerably longer than the power boom. Only 3 m are necessary for the power boom this allows the downward folded boom to fit in the launch vehicle in it full length. The science boom, which requires a length of 10.6 m needs to be partially retractable. This retraction technique can be directly inheritated from the Galileo spacecraft.

The antenna will be firmly mounted on top of the main body so that its center section can support the adapter to the launch vehicle. I also considered making the antenna pointable. This would decrease the attitude correction maneuvers and thus reduce the necessary amount of propellant. Added weight and complexity due to the pointing mechanism and compatibility problems with the launch vehicle discard this option. A pointing mechanism would not be able to provide a stiff support when placing the adapter on the antenna. A complex design is necessary to comply with both, the pointability and the stiffness during launch. Placing the adapter on the other side of the craft requires a very large adapter because it has to give room to the booms and using the booms is not feasible because they, as the pointing mechanism are not stiff enough to firmly support the probe during launch.

Since the remote sensing instruments need to be pointed at the object of interest and the antenna needs to be pointed at earth, a pointing mechanism is necessary for the science platform which will house the remote sensing equipment. These can than be

The PULSE Space Probe



pointed independently from the main body. During the cruise phase these instruments are not used and to protect them a steel canopy is placed over them. This canopy will tilt open when the instruments are operational.

5.3. Material selection:

To perform the material selection I gathered as much information from different sources as possible and incorporated them into the following table.

PROPERTIES:

Property	Al	Be	Mg	Ti	Kevlar	Steel	Unit
Density	2.8	1.85	1.74	4.5	1.9	7.87	g/cm^3
Yield str.	500	415	103	830	1600	1800	MPa
machinability	ex.	poor	ex.	good	poor	good	
weldability	good	poor	ex.	good	none	ok	
handling	ex.	poor	ok	ex.	poor	ex.	
cost	low	high	low	mod.	high	low	
corrosion resistance	ex.	ok	poor	ex.	ok	ex.	

I then awarded points for their properties on the scale of 0 through 100 according to the desirability of the properties.

POINTS:

Property	Al	Be	Mg	Ti	Kevlar	Steel	weight
Density	72	81.5	82.6	55	81	21.3	0.55
Yield str.	25	20.75	5.15	41.5	80	90	0.1
machinability	100	40	100	80	40	80	0.1
weldability	80	40	100	80	0	60	0.075
handling	100	40	60	100	40	100	0.05
cost	100	0	100	60	0	100	0.1
corrosion resistance	100	60	40	100	60	100	0.025
Sum :	577	282.2	487.7	516.5	301	551.3	1

The final evaluation is based on the points received and a weighing factor which allows to stress more important properties over less important ones.

EVALUATION:

Property	Al	Be	Mg	Ti	Kevlar	Steel	weight
Density	39.6	44.82	**45.43**	30.25	44.55	11.715	0.55
Yield str.	2.5	2.075	**0.515**	4.15	8	9	0.1
machinability	10	4	**10**	8	4	8	0.1
weldability	6	3	**7.5**	6	0	4.5	0.075
handling	5	2	**3**	5	2	5	0.05
cost	10	0	**10**	6	0	10	0.1
corrosion resistance	2.5	1.5	**1**	2.5	1.5	2.5	0.025
Sum :	75.6	57.4	**77.44**	61.9	60.05	50.715	1

Selection made: Magnesium

Legend:

Points	synonym
100	ex. or low
80	good
60	ok or mod.
40	poor
20	bad
0	none or high

Formulas used: For density : Points = 100 - density/10

=> density = 0 -> 100 Points
=> density =10 -> 0 Points

For yield strength : Points = Ys / 20

=> Ys = 2000 -> 100 Points
=> Ys = 0 -> 0 Points

5.4. Calculation of required wall thickness for micrometeoroid protection.

Material proposed:

Magnesium

Constants:

meteoroid mass,M :	0.1 g
meteoroid velocity,V :	25 km/s
meteoroid density,roh :	0.5 g/cm^3
mat. constant for Al :	0.06 (from reference)
mat. constant for Mg,K :	0.08 (estimated)
Density of Mg,RMG :	1.74 g/cm^3
Yield strength,YS :	22000 lbf/in^2

Derived Values:

meteoroid diameter,D : 0.725566 cm
(spherical meteoroid shape assumed)
first sheet thickness,T1 : 0.072556 cm
(T1/D=0.1 requ. by Formula)

Variable:

spacing,S : 2 cm

Formula : (for double sheet penetration)

t = K*roh^0.15*M^.35*V/S^0.*(70000/YS)
t = 1.015542 cm

Summary :

First sheet thickness,T1 :	0.072556 cm
Second sheet thickness,t :	1.015542 cm
Spacing,S :	2 cm

Protects from 0.1 g micrometeoroid at average speed.

Design sizes :

First sheet thickness,T1 :	0.2 cm
Second sheet thickness,t :	0.9 cm
Spacing,S :	2 cm

5.5. Mass estimation from design and sheet thickness:

Constants:

First sheet thickness, $t1$:	0.2 cm
Second sheet thickness, t :	0.9 cm
Lid thickness, tl :	1 cm
Density of Mg, roh :	1.74 g/cm^3
Area of spar, Asp :	4.1 cm^2

Variables:

Height, h :	80 cm
Diameter, d :	50 cm

Formulas:

Panel length, s :	$s = d/2 * (2-2^{0.5})^{0.5}$	
	$s =$	19.13417 cm
Panel area, Ap :	$Ap = 8 * s * (t1+t)$	
	$Ap =$	168.3807 cm^2
Spar area, As :	$As = 8 * Asp$	
	$As =$	32.8 cm^2
tot. cross sect. area, Ac :	$Ac = As + Ap$	
	$Ac =$	201.1807 cm^2
Lid area, Al :	$Al = D^2 * 2^{.5} / 2$	
	$Al =$	1767.766 cm^2
Lid volume, Vl :	$Vl = 2 * Al * tl$	
	$Vl =$	3535.533 cm^3
Trunk volume, Vt :	$Vt = Ac * h$	
	$Vt =$	16094.45 cm^3
total Volume, V :	$V = Vt + Vl$	
	$V =$	19629.99 cm^3

Total weight of the main body structure:

M = 34.16 kg

5.6. Production techniques required:

The magnesium side panels can be bought from stock, cut and welded to the spars. The magnesium spars need to be extruded. The main body lids and the base of the science platform have to be casted. The steel canopy has to be produced by deep drawing and then weld the second sheet onto it to enhance the micrometeoroid protection. The boom struts can be bought from stock and then assembled.

All these techiques are well known and readily avalible today. Any new developements can be incorporated at a later point to improve the performance of the craft.

5.7 References:

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Propulsion

Numerous factors must be considered in selecting propellants and propulsion systems for space missions. One of the more general characteristics is performance, in terms of both specific impulse and hardware mass. Final selection must depend on tradeoffs between several of the major competing selection criteria: for example performance, reliability and cost.

The first decision to make was what launch vehicle the Pulse probe would be launched on. After evaluation of all of the United States vehicles and some International launch vehicles, it was found that the four best choices for this mission were the U.S. Space Shuttle, the Ariane IV, the Titan IV Centaur G Prime, and the Titan IV IUS. This primary trade study was based on the mass that each vehicle could be place into a geostationary transfer orbit. The United States Space shuttle was ruled out because of the higher cost for a non-expendable launch vehicle.

After this preliminary study a more in depth study was performed on the Ariane IV and the Titan IV configurations. Using the equations from Conway (Ref. 4), a comparison was made between the three launch vehicles on the basis of payload ratio, propellant mass and total mass, given a delta-v and a payload mass (Figures 6.1, 6.2, 6.3, 6.4, 6.5). The conclusion reached was that the Ariane IV launch vehicle was the best selection in all comparisons. The Launch Specifications for the Ariane IV are given in the appendix.

The fuel used for each stage of the Ariane vehicle will be the specified fuel in the launch specifications in the appendix. In these specifications one will find that the diameter of the upper stage is 2.59 meters in diameter which is sufficient for the largest diameter of our spacecraft which allow the antenna to fit in uncollapsed.

Fig.6.1

Subsystem Masses

System and components	Number of components (1+redundancy)	Weight (kg)
Science	1	94.9
Telecommunications		75.46
Control	2	22.73
Receiver	4	14.55
Amplifier	4	3.64
Data handling	2	16.36
Data storage	2	18.18
Spacecraft control		38.17
Computer and sequencer	2	10.91
Sun sensors	2	5.45
Canopus tracker	2	5.45
Gyros	2	5.45
Scan control and planet sensor	1	10.91
Electrical power		121.41
RTG's	1	44.4
Conditioning and control	2	45.45
Cabling	1	31.82
Structure and mechanical		290.46
Bus	1	150
Parabolic antenna	1	9.1
Temperature control	1	11.36
Trajectory correction propulsion		120
Total spacecraft weight		620.4
Launch vehicle adapter		50
Total injected weight		670.4

Launch Specifications

Fig.6.2

Variables	Ariane IV	Titan IV Centaur G Prime	Titan IV IUS
thrust1 [N]	204318.20	72715000.00	72715000.00
thrust2 [N]	40227.30	23636.40	23626.40
thrust3 [N]	3181.80	7500.00	13840.90
thrust (total) [N]	247727.30	72746136.40	72752467.30
c1 [km/s]	3038.00	2989.00	2989.00
c2 [km/s]	3136.00	3136.00	3136.00
c3 [km/s]	3528.00	3528.00	2842.00
c (total) [km/s]	9702.00	9653.00	8967.00
R1	2.56	2.46	3.45
R2	2.13	2.19	2.88
R3	2.90	2.95	1.99
R (total)	7.58	7.61	8.32
Ms1 [kg]	786.02	779.84	1210.00
Ms2 [kg]	334.52	366.74	372.91
Ms3 [kg]	125.11	129.82	57.84
Ms (total) [kg]	1245.64	1276.39	1640.75
Mp1 [kg]	10510.00	10420.00	16180.00
Mp2 [kg]	3161.00	3465.00	3524.00
Mp3 [kg]	1672.00	1735.00	773.20
Mp (total) [kg]	15343.00	15620.00	20477.20
Mo [kg]	17260.00	17570.00	22790.00
lambda 1	0.53	0.57	0.31
lambda 2	0.71	0.66	0.39
lambda 3	0.37	0.36	0.81
lambda (total)	1.61	1.59	1.50

Fig.6.3

Payload Mass Ratio

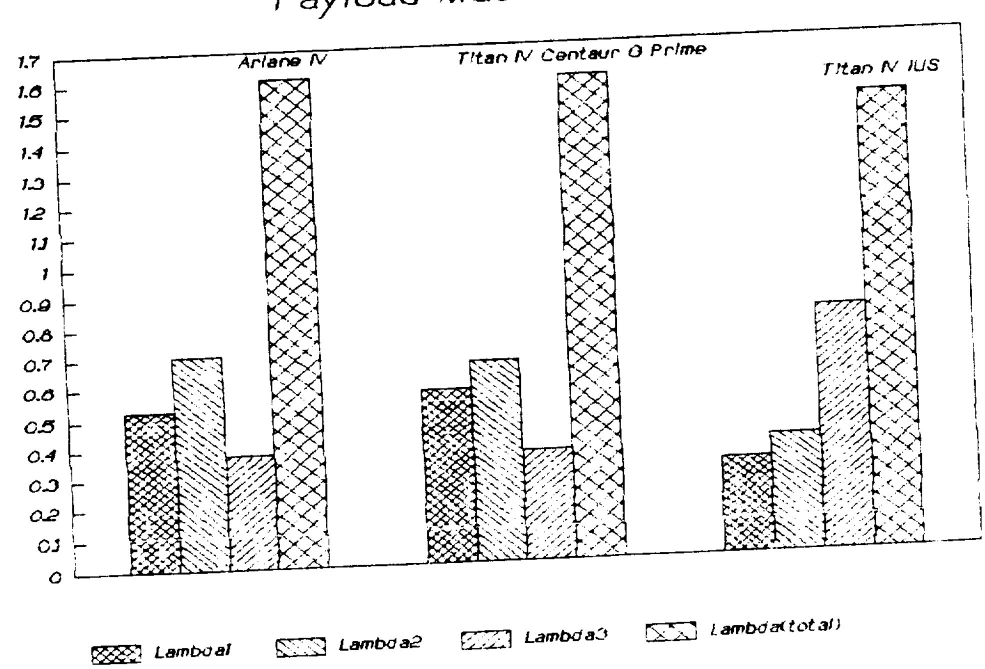


Fig.6.4

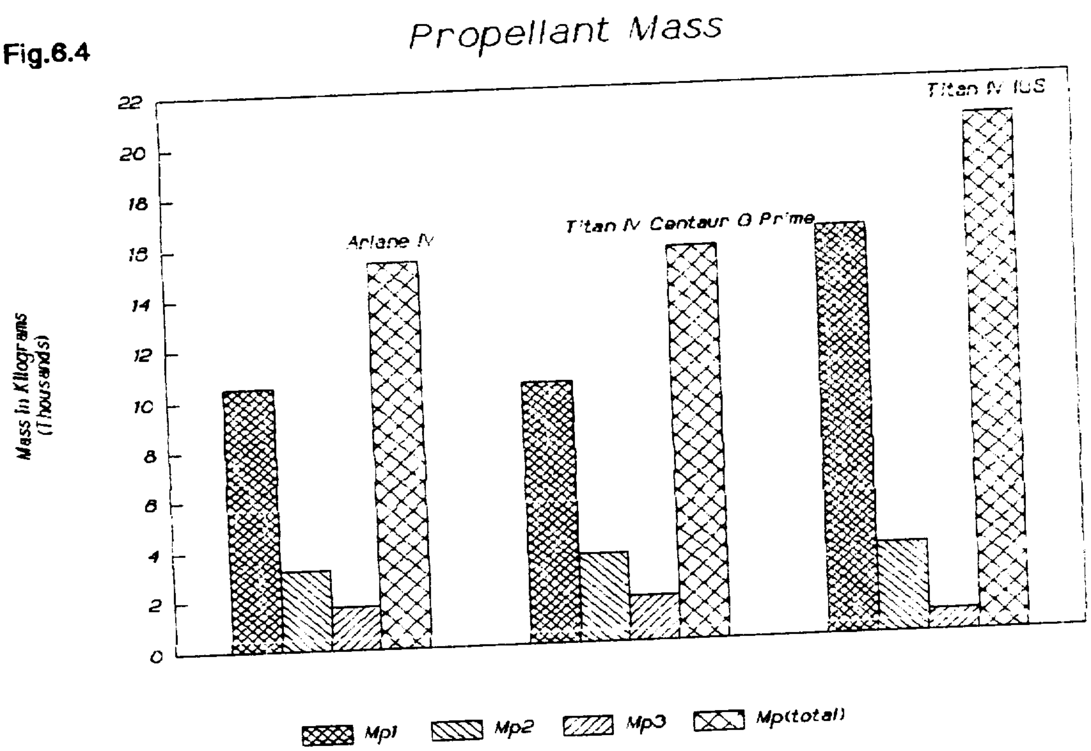
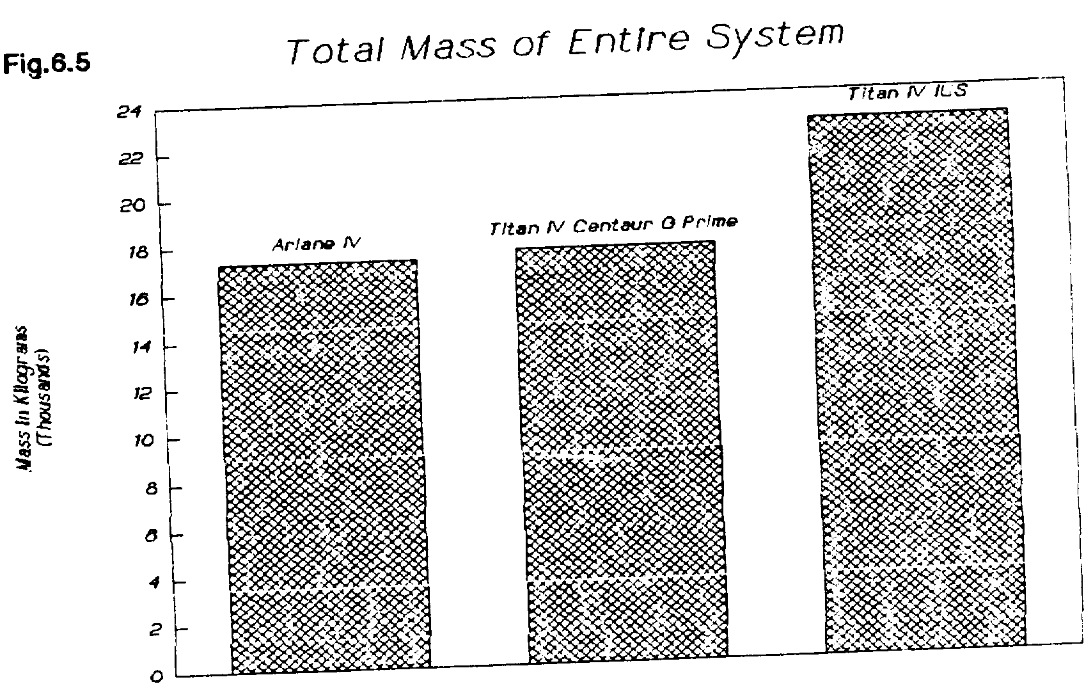

Propellant Mass
Mass in Kilograms (Thousands)
22
20
18
16
14
12
10
8
6
4
2
0
Ariane IV
Titan IV Centaur G Prime
Titan IV IUS
Mp1
Mp2
Mp3
Mp(total)

Fig.6.5


Total Mass of Entire System
Mass in Kilograms (Thousands)
24
22
20
18
16
14
12
10
8
6
4
2
0
Ariane IV
Titan IV Centaur G Prime
Titan IV IUS

Power System

The operational capabilities of a space vehicle is dependent upon an adequate supply of power. This power is necessary for communications, guidance, control, and operation of sensors or scientific instrumentation.

When trying to select a power source for the PULSE probe there were 12 factors which I took into consideration: 1)Duration 2)Mission 3)Availability 4)Reliability 5)Weight 6)Compatibility 7)Environment 8)Power level 9)Area 10)Cost 11)Volume 12)Hazard. Since the mission duration of our probe is about 16 years the selection of power source was limited to nuclear power, either from decay of an isotope or a nuclear reactor. Batteries were also considered for storing the electrical energy provided by the power source. The approach taken consisted of listing the 12 factors and rating the sources from 1 to 10(highest) on the quality of performance related to each of the 12 factors as shown in figure 6.5.

The results from this trade study eliminated the nuclear reactor as a power source but showed that batteries should be further considered as energy storage devices for the RTGs. But when looking at the predicted power to weight ratio of both the RTG(12 W/kg) and the Ni-Cd battery (10 W-Hr/kg) in the year 2000 the choice was that the RTGs were the only power source that was going to be used on the PULSE probe (Ref. 10, pp.1-45).

The next step in developing the power system was finding out how much power the power system would have to put out at peak operating loads. Figure 6.6 shows a list of the subsystems and the power that each subsystem requires at peak level. Figure 6.7 shows the percentage of power each subsystem requires of the total power. A total power system requirement of 372.94 W is needed upon arrival at Pluto.

The isotope selected for this mission is Pu 238, with a half life of 87 years. This isotope has been proven by earlier space missions and often exceeded its original design life requirements. Some studies have used a design lifetime of 10 years for the RTG and found that the RTG has a 20% reduction in power at the end of the projected 10 year life (Ref. 10, pp.1-48).

The PULSE probe's RTGs will have to supply power for at least 16 years. This results in a 70% reduction in 16 years which shows that at launch the PULSE probe will have 529.7 W of power that would diminish to the amount needed at Pluto (See appendix for these calculations). No safety margin is needed with these figures because the Pu 238 RTG "has operated considerably longer than their original design life requirements" (Ref. 10, pp.1-44). From the total power needed at launch a calculation was made to determine the mass of RTG needed. The mass of RTG needed is 44.40 kg, which would require 23 slices of fuel cells in the Modular Isotopic Thermoelectric Generator (Ref. 12, pp.340) (See appendix for calculations). The RTG fuel capsule is designed to withstand intact reentry should there be a mission failure or abort.

The electrical power from the RTG will go to the Power Conditioning Unit which will regulate the voltage and convert the DC power into whatever form it needs to be in for the applied loads. This will depend upon the voltages needed by the instruments and if they are powered by AC or DC voltage (Figure 6.8).

Fig.6.6

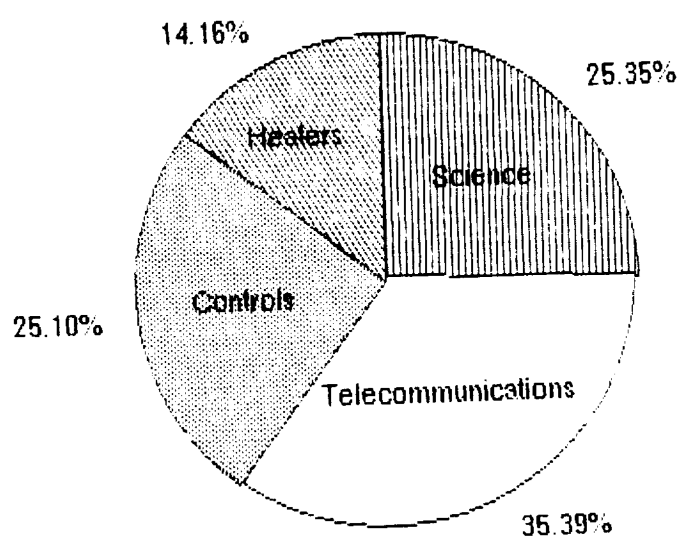
Power Supply Determination

	Reactor	RTG	Battery
Duration	6	8	4
Mission	2	10	6
Availability	6	8	8
Reliability	6	10	10
Weight	4	8	8
Compatibility	6	10	8
Environment	8	8	8
Power level	10	8	6
Area	4	8	6
Cost	4	6	10
Volume	2	8	8
Hazard	6	8	10
Total	64	100	92

Fig.6.7 **Power Systems**

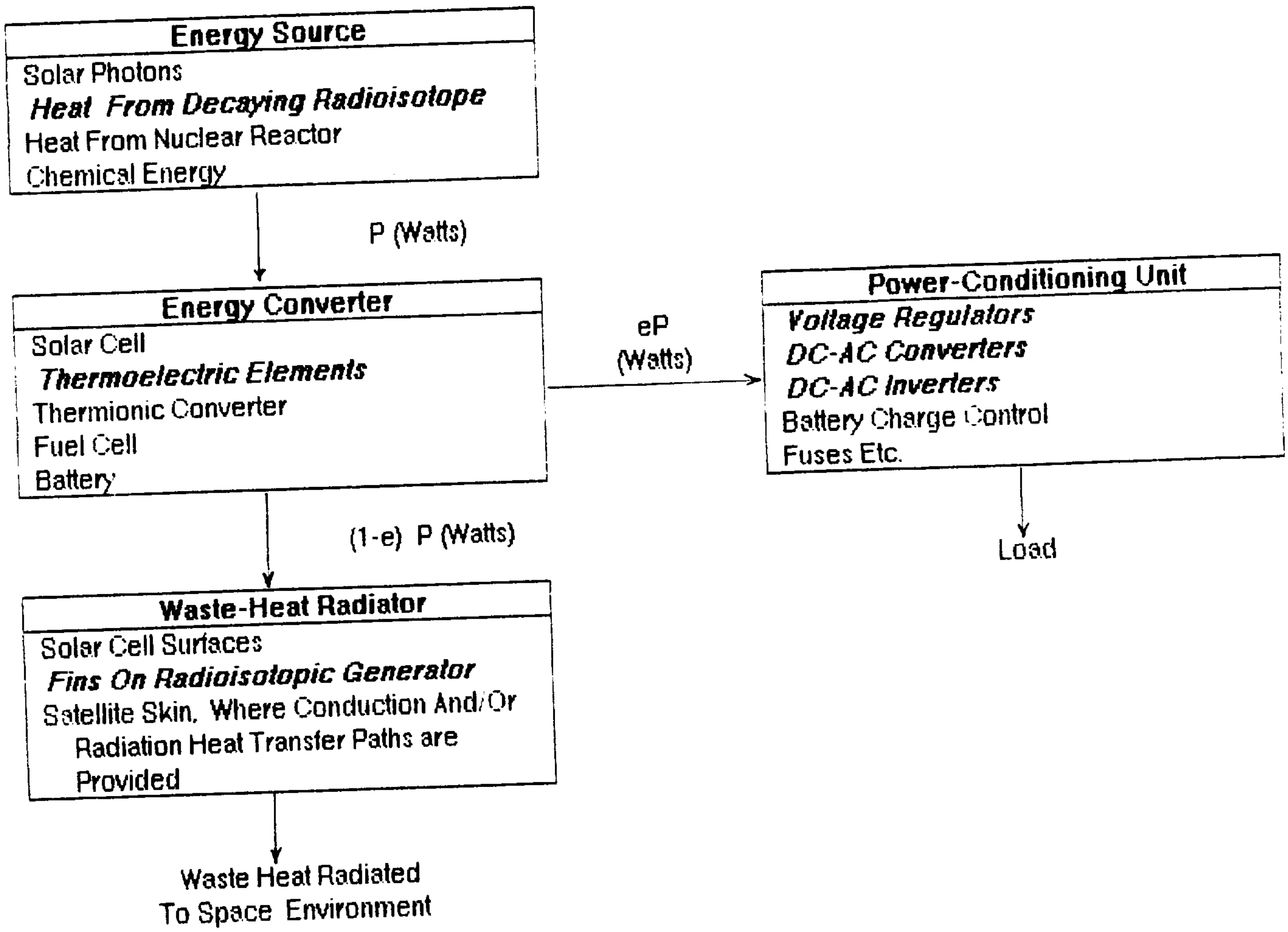
System function	Power required at peak levels (Watts)
Science	78.78
Telecommunications	110
Control	5
Receiver	10
Amplifier	70
Data samplin, encoding, and decoding	20
Data storage	5
Spacecraft control	78
Sequencing and command	10
Sun sensors	3
Canopus tracker	10
Gyros	15
Electronics	40
Heaters	44
Total system requirements	310.78
Conversion loss (20%)	62.16
Total power requirement	372.94

Fig.6.8 **Power Subsystems**



Power System

Fig.6.9



Appendix I

Propulsion

Isp = specific impulse
V = delta V needed
ε = structural coefficient
ML = mass of the payload (spacecraft)
Thrust = thrust given by each of the stages
c = exhaust velocities
f(α) = function used for Newton's Approximation
fprime(α) = derivative of f(α)
α = Lagrange multiplier
R = mass ratio
MSP = mass of structure and propellant of that stage
M = mass of that stage plus payload weight
Ms = mass of the structure of that stage
Mp = mass of the propellant on that stage
Mo = total mass of the launch vehicle and spacecraft
λ = payload ratio
Massflow = massflow of that stage
Burntime = burntime of that stage

Base units:

$sec \equiv 1T$ $\qquad$ $kg \equiv 1M$ $\qquad$ $m \equiv 1L$

Normal units:

$N \equiv kg \cdot \dfrac{m}{sec^2}$ $\qquad$ $km \equiv 1000 \cdot m$ $\qquad$ $lb \equiv \dfrac{kg}{2.2}$ $\qquad$ $lbf \equiv 4.4 \cdot N$

Constants:

This shows only one launch vehicle. This process was done 3 times
A chart with all the values is in the text

$Isp_1 := 310 \cdot sec$ $\qquad$ $Isp_2 := 320 \cdot sec$ $\qquad$ $Isp_3 := 360 \cdot sec$

$g := 9.8 \cdot \dfrac{m}{sec^2}$ $\qquad$ $V := 8.974 \cdot \dfrac{km}{sec}$ $\qquad$ $ML := 670.40 \cdot kg$

Assuming structural coefficients to be the same for Titan and Ariane
(Actual Ariane values)

$\varepsilon_1 := .0696$ $\qquad$ $\varepsilon_2 := .0957$ $\qquad$ $\varepsilon_3 := .1008$

$thrust_1 := 899000 \cdot lbf$ $\qquad$ $thrust_2 := 177000 \cdot lbf$ $\qquad$ $thrust_3 := 14000 \cdot lbf$

Equations:

$i := 1 \,..\, 3$

$c_i := Isp_i \cdot g$

Iteration using Newton's approximation

$$f(\alpha) := V - \sum_i c_i \cdot \ln\left[\frac{\alpha \cdot c_i - 1 \cdot \frac{km}{sec}}{\alpha \cdot c_i \cdot \varepsilon_i}\right]$$

c_i
$3.038 \cdot 10^{3}$ length^{1} time^{-1}
$3.136 \cdot 10^{3}$ length^{1} time^{-1}
$3.528 \cdot 10^{3}$ length^{1} time^{-1}

$$fprime(\alpha) := \sum_i \left[c_i \cdot \left[\left[\frac{c_i}{\alpha \cdot c_i - 1 \cdot \frac{km}{sec}}\right] - \frac{1}{\alpha \cdot .99999999999}\right]\right] \cdot -1$$

$j := 0 \,..\, 20$ $\qquad x_0 := .43$

$$x_{j+1} := \mathrm{until}\left[\left|f\left[x_j\right]\right| - .0001 \cdot \frac{km}{sec}, \left[x_j - \left[\frac{f\left[x_j\right]}{fprime\left[x_j\right]}\right]\right]\right]$$

$n := \mathrm{size}(x)$ $\qquad \alpha := x_n$ $\qquad \alpha = 0.4$

$$R1 := \frac{\alpha \cdot c_1 - 1 \cdot \frac{km}{sec}}{\alpha \cdot c_1 \cdot \varepsilon_1} \qquad R2 := \frac{\alpha \cdot c_2 - 1 \cdot \frac{km}{sec}}{\alpha \cdot c_2 \cdot \varepsilon_2} \qquad R3 := \frac{\alpha \cdot c_3 - 1 \cdot \frac{km}{sec}}{\alpha \cdot c_3 \cdot \varepsilon_3}$$

$R1 = 2.557$ $\qquad R2 = 2.128$ $\qquad R3 = 2.898$

$$MSP3 := \frac{ML - R3 \cdot ML}{R3 \cdot \varepsilon_3 - 1} \qquad MSP3 = 1.798 \cdot 10^{3}$$

$$MSP2 := \frac{MSP3 + ML - R2 \cdot MSP3 - R2 \cdot ML}{R2 \cdot \varepsilon_2 - 1} \qquad MSP2 = 3.495 \cdot 10^{3}$$

$$MSP1 := \frac{MSP2 + MSP3 + ML - R1 \cdot MSP2 - R1 \cdot MSP3 - R1 \cdot ML}{R1 \cdot \varepsilon_1 - 1} \qquad MSP1 = 1.129 \cdot 10^{4}$$

$$M_{03} := MSP3 + ML \qquad M_{02} := MSP3 + MSP2 + ML \qquad M_{01} := MSP3 + MSP2 + MSP1 + M$$

$$M_{03} = 2.468 \cdot 10^{3} \cdot mass \qquad M_{02} = 5.963 \cdot 10^{3} \cdot mass \qquad M_{01} = 1.726 \cdot 10^{4} \cdot mass$$

$$Ms_1 := \varepsilon_1 \cdot MSP1 \qquad Ms_2 := \varepsilon_2 \cdot MSP2 \qquad Ms_3 := \varepsilon_1 \cdot MSP3$$

$$Ms_1 = 786.017 \cdot mass \qquad Ms_2 = 334.515 \cdot mass \qquad Ms_3 = 125.111 \cdot mass$$

$$Mp_1 := MSP1 - Ms_1 \qquad Mp_2 := MSP2 - Ms_2 \qquad Mp_3 := MSP3 - Ms_3$$

$$Mp_1 = 1.051 \cdot 10^{4} \cdot mass \qquad Mp_2 = 3.161 \cdot 10^{3} \cdot mass \qquad Mp_3 = 1.672 \cdot 10^{3} \cdot mass$$

$$M_0 := MSP1 + MSP2 + MSP3 + ML \qquad M_0 = 1.726 \cdot 10^{4} \cdot mass$$

$$\lambda_1 := \frac{M_{02}}{M_0 - M_{02}} \qquad \lambda_2 := \frac{M_{03}}{M_{02} - M_{03}} \qquad \lambda_3 := \frac{ML}{M_{03} - ML}$$

$$\lambda_1 = 0.528 \qquad \lambda_2 = 0.706 \qquad \lambda_3 = 0.373$$

$$\mathrm{Massflow}_i := \frac{\mathrm{thrust}_{i,1}}{c_i} \cdot \frac{1}{1000} \qquad \mathrm{Burntime}_i := \frac{\mathrm{Mp}_i}{\mathrm{Massflow}_i}$$

$\mathrm{Massflow}_i$
$1.302 \; \mathrm{mass}^{1} \; \mathrm{time}^{-1}$
$0.248 \; \mathrm{mass}^{1} \; \mathrm{time}^{-1}$
$0.017 \; \mathrm{mass}^{1} \; \mathrm{time}^{-1}$

$\mathrm{Burntime}_i$
$8.07 \cdot 10^{3} \; \mathrm{time}^{1}$
$1.273 \cdot 10^{4} \; \mathrm{time}^{1}$
$9.579 \cdot 10^{4} \; \mathrm{time}^{1}$

FRANCE

ESA/Arianespace

Ariane 2	CNES/Arianespace	1	4 x Viking 5 liquid	Aerospatiale/SEP	L-140	N_2O_4/UH25	601,000	12.5	59.8	490 000	4,796	3,200
		2	1 x Viking 4 liquid	ERNO/SEP	L-33	N_2O_4/UH25	177 600	8.5	37.6	(total)	(geostationary	
		3	1 x HM-7B liquid	Aerospatiale/SEP	H-10	LOX/LH_2	14 000	8.5	34.2		transfer)	
Ariane 3	CNES/Arianespace	1	4 x Viking 5 liquid	Aerospatiale/SEP	L-140	N_2O_4/UH25	601,000	12.5	59.8	530 000	5,590	3,790
		1	2 x P7.3 solid	BPD	PAP	Solid	250,000	3.5	26.2	(total)	(geostationary	
		2	1 x Viking 4 liquid	ERNO/SEP	L-33	N_2O_4/UH25	177,500	8.5	37.6		transfer)	
		3	1 x HM-7B liquid	Aerospatiale/SEP	H-10	LOX/LH_2	14 000	8.5	34.2			
Ariane 4[38]	CNES/Arianespace Arianespace	1	4 x Viking 5 liquid	Aerospatiale/SEP	L-220	N_2O_4/UH25	601,000	12.5	82.5	5[illegible]3 000	4 [illegible]0	—
		1[a]	2-4 x Viking 6 liquid	ERNO/SEP	L-36	N_2O_4/UH25	152,000	7.1	62.3	to	to	
		1[a]	2-4 x P9.5 solid	BPD	P9.5	Solid	146 000	3.5	29.2	1 233 000	9,250	
		2	1 x Viking 4 liquid	ERNO/SEP	L-34	N_2O_4/UH25	177 000	8.5	37.6	(AR44L)	geostationary	
		3	1 x HM-7S liquid	Aerospatiale/SEP	H-10	LOX/LH_2	14 000	8.5	34.2		transfer, [illegible] incl	

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Power

20% decrease in power over 10 years (Ref. 10, pp.1-48)

$N(t)$ = percentage of power after t years

N_o = percentage of power at launch

k = decay constant

t = time

$$N(t) = No\ e^{-kt}$$

$$.80 = 1\ e^{-k(10)}$$

$$k = -\ln(.80)/10$$

$$k = 0.022314$$

$$N(t) = 1\ e^{-(0.022314)(16.005)}$$

$$N(t) = 0.69967$$

This is a 30% decrease over 16 years

Total power needed/70% = Power at launch/100%

372.94/70% = Power at launch/100%

Power at Launch = 529.69 W

Assuming (12W/kg) power to weight ratio predicted for the year 2000 (Ref. 10, pp.1-45)

529.69 W/12W/kg= 44.40 kg of RTG at launch

MITG Generator give 23.5W/slice (Ref. 12, pp.340)

529.69 W / 23.5W/slice = 22.54 slices   approximately 23 slices

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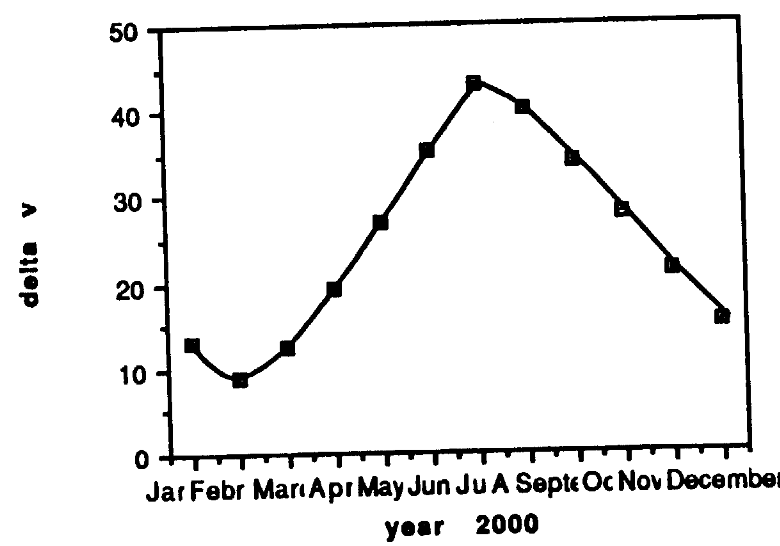
18. Unmanned Exploration of the Solar System, Volume 19, Advances in the Astronautical Sciences, Proceedings of the American Astronautical Society, Symposium on Unmanned Exploration of the Solar System, Feb 8-10, 1965, (TL787.A6A2).

19. U.S. and International Launch Vehicles, Aviation Week & Space Technology, March 20, 1989, (629.105AU).

The spacecraft PULSE uses much off-the-shelf hardware from Voyager and other planned probes. New technology is only applied if it would include a more reliable and less costly trade-offs, as in the case of onboard computers. PULSE willyield quality science at low cost by using incorporation of off-the-shelf products, choosing radiation-hardened version of widely available microprocessor and integrated-circuit chips supported by efficient software. In general, proven techniques were used throughout the entire design.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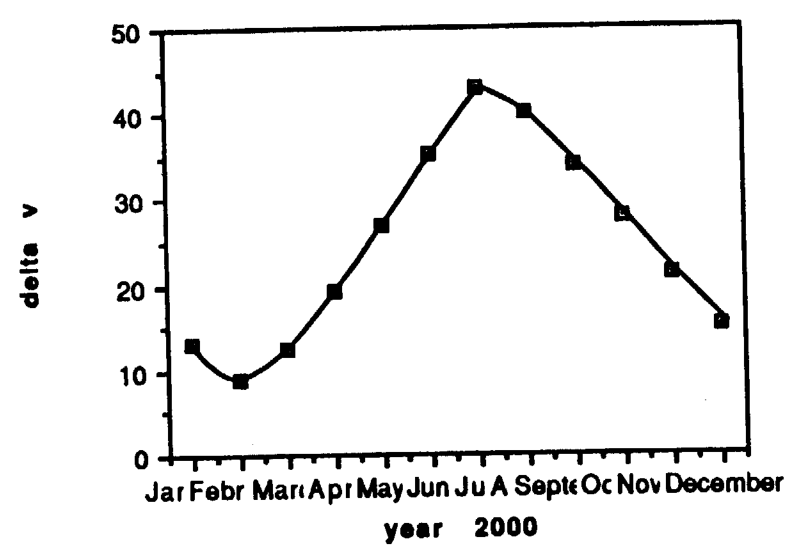
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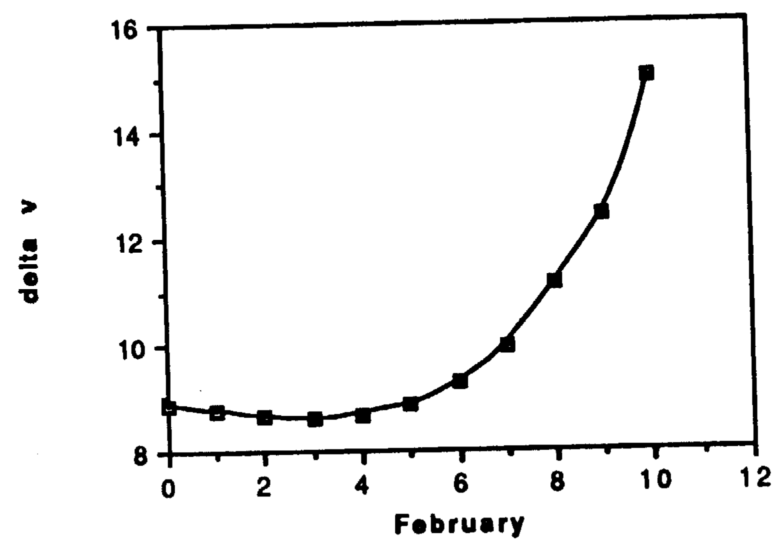
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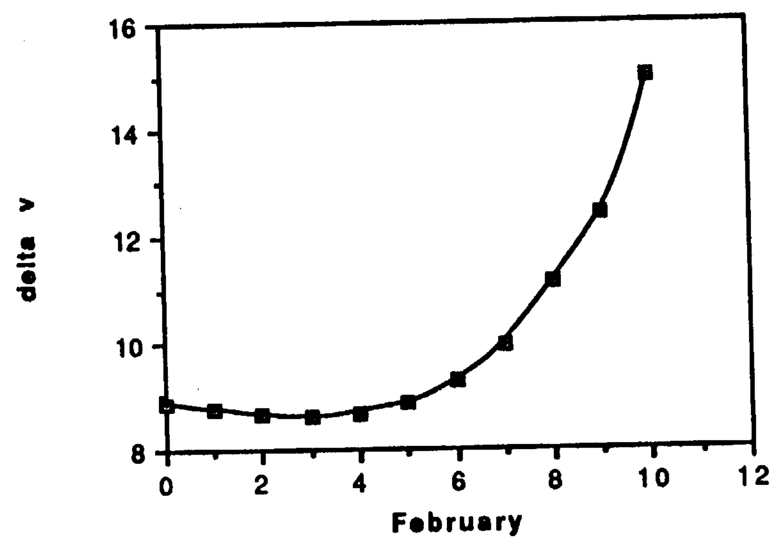
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delta v
February

Costing for PULSE

Category	Cost (FY 88 Dollars)
Structure	59,988,162.98
Thermal Control	11,037,938.33
Propulsion	412,927,670.50
Attitude & Articulation	62,614,609.37
Telecommunications	64,098,191.33
Antennas	13,043,018.66
Command & Data Handling	24,500,108.53
RTG Power	37,386,446.55
Line-Scan Imaging	170,454,335.10
Particle & Field Instruments	71,222,537.72
Remote Sensing Instruments	29,154,302.64
System Support & Ground Equipment	280,062,535.20
Launch + 30 Days Ops & Ground S/W	57,185,698.78
Image Data Development	6,957,007.47
Science Data Development	11,487,733.40
Program Management	17,365,267.83
Flight Operations	258,722,216.60
Data Analysis	115,984,760.70
TOTAL	**1,704,192,542.00**

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DELTA V REQUIRED: 8.606 KM/SEC (FROM PARKING ORBIT)

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Table 3.2

Weights and Power for PEP Instrumentation

Instrument	Power (W)	Mass (kg)
ISS	20	28*
MAG	2.2	5.6
NIMS	13	18
PPR	4.5	4.8
VVS	5.33	4
PLS	10*	12
EDP	10*	9
PWS	8.4*	6
CRS	5.35	7.5

* Values are estimates

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1

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D.D. Group 5