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FINAL REPORT

OPTIC

Orbiting Plutonian Topographic Image Craft

Proposal For An Unmanned MissionTo Pluto

APRIL 24, 1990

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INTRODUCTION

This report presents and describes the Orbiting Plutonian Topographic Image Craft (OPTIC). The vehicle's twenty year trip will culminate upon arrival at Pluto, the only major body in the solar system that has not been studied by an earth launched probe. After arrival OPTIC will begin its data collection which includes image and radar mapping, surface spectral analysis, and magnetospheric studies. This initial investigation into the remote study of Plutonian space utilizing an unmanned probe was conducted by AAE 241 Group 4 at the University of Illinois, Urbana-Champaign, Spring 1990.

This probe's design was developed based on the Request for Proposal requirements generated for the spacecraft design section of AAE 241, an Aeronautical & Astronautical Engineering Senior Design class affiliated with the Universities Space Research Association. The design work presented herein is the original work of the six members of Group 4. It has been produced and compiled based on individual research and knowledge acquired from class work, in addition to the annotated guidance and information received from outside sources.

Based upon the Request for Proposal emphasis on study of Plutonian space, and NASA's stress on the importance of not only photographic data, but also mapping, an orbiter seems to be the best solution. The problems which an orbiter presents are varied, but all appear solvable. The distinct problems which an orbiter causes for each subsystem are discussed in their respective sections throughout the report.

The final design formulation revolved around two important factors: (1) the ability to collect and return the maximum quantity of information on the Plutonian system and (2) the weight limitations which the choice of an orbiting craft implied. The velocity requirements of this type of mission severely limited the weight available for mission execution - owing to the large portion of overall weight required as fuel to fly the craft with present technology.

While the mission is not constrained to only arrive and examine Pluto, Plutonian space is its prime objective. This and other factors, describe within, lead to the choice of an orbiting craft. Since the science objectives are what directed this mission, the justification for what may appear to be an extravagant task is contained within the Science Instrumentation subsection.

OPTIC DESIGN GROUP

SCIENTIFIC INSTRUMENTATION

Jonathan E. Kelly [redacted] Jonathan Kelly

MISSION MANAGEMENT, PLANNING AND COSTING

Randall John Hein [redacted] Randall J Hein

POWER AND PROPULSION

David Lee Meyer [redacted] David Lee Meyer

ATTITUDE AND ARTICULATION CONTROL

David Mark Robinson [redacted] David M. Robinson

STRUCTURES

Mark James Endre [redacted] Mark James Endre

COMMAND, CONTROL AND COMMUNICATIONS

Eric W. Summers [redacted] Eric Summers

1.0 SCIENCE INSTRUMENTATION

INTRODUCTION

This section presents and describes the four major components of the OPTIC science subsystem: an electronic imaging unit, a magnetometer assembly, an ultraviolet spectrometer, and a radar mapping system.

Also included is the explanation of design choice based upon defined mission objectives.

This subsection was investigated and prepared by Jonathan E. Kelly.

1.1 DIRECTING FACTORS

Pluto is the only remaining planet in the solar system which has not been studied with the help of an earth-launched spacecraft. What little information that is known about this planet has been extrapolated from the blurred views which earthbound telescopes provide. The quantity and quality of the information that is possessed is excellent when the limitations of the collection process and the distance separating Earth and Pluto are considered.

The successful launch and deployment of the Hubble Space Telescope will increase the knowledge of this tiny world many fold. An actual physical encounter with Pluto and its satellite, Charon, utilizing a probe, would increase even that knowledge by many orders of magnitude. With this goal - to investigate these bodies intensively - the Orbiting Plutonian Topographic Image Craft (OPTIC) has been designed.

Since OPTIC's primary objective is to gather information about Plutonian space, the driving factor behind its mission design is its scientific instrumentation.

SCIENCE DIRECTIVES

OPTIC, as the acronym's words imply, is an orbiting probe which will gather topographic and image data about Pluto. This craft will also collect data concerning any significant existing magnetic fields that are located around the Pluto system and additionally perform spectral analysis of the system's bodies' surfaces.

After extensive investigation, which included interviews with a limited number of the members of the scientific community, and analysis of the recommendations of past NASA and scientific conferences, an orbiter was chosen as the best means of obtaining the maximum quantity of useful data and fulfilling the mission objectives.

During the preliminary investigations for the science subsystem members of the University of Illinois Department of Geology, all who had previous experience with planetary studies, were contacted and briefly interviewed. Professor Albert T. Hsui from this department emphasized that the major questions which a mission to Pluto should attempt to answer might include: surface makeup, existence of magnetic fields, gravity, and overall planet size. It was also stressed that photographic data would most likely help in answering these questions. (4. Hsui)

In the subsequent study of published documents the importance of images also became evident. NASA published planetary exploration goals which are in agreement with Dr. Hsui's recommendations. These primary goals, as listed by the participants in an Arizona State University (ASU) sponsored Planetary Geology Workshop, emphasize preliminary acquisition of planet surface characteristics. Following in the exploration phase should be studies of planets' topography, gravity magnetic fields, surface chemistry, and mineralogy. This conference emphasized mapping as the best method for obtaining the maximum information concerning planetary surfaces. (7. NASA, pgs. 5-7)

The present information available about the Plutonian system, coupled with the round trip signal time, makes a lander unfeasible both technologically and conceptually. Technologically because the artificial intelligence (AI) necessary for independent control of a mobile lander at that distance does not presently exist. That level of AI will probably not be developed sufficiently by the year 2000.

Even more importantly, with regards to a lander, at the present time, knowledge of the Pluto system is not extensive enough to provide design criteria for safely putting a lander (mobile or otherwise) down on the planet. Foremost, there exists basically zero knowledge concerning surface makeup - a vitally important piece of information for choosing possible landing sites. Additionally the knowledge of the physical makeup and conditions of the plutonian environment does not allow for producing a lander for which survival could be guaranteed with any confidence.

The flyby option is technologically feasible. It would also be less costly. But since flyby encounter times are extremely brief (the Neptune encounter of 1989 was only five days) and one of the main requirements of the proposal is to study Plutonian space, a probe going solely to Pluto would spend years in transit for an encounter that would last only days.

Based on the limitations, with regards to a thorough plutonian study, that a flyby mission has, the logistics and technical problems included in the lander option, and the recommendations from NASA and the scientific community, an orbiter was chosen as the best configuration for fulfilling the RFP guidelines.

1.2 SCIENCE INSTRUMENTS

CHOSEN SYSTEMS

The most important constraint which an orbiter presents is weight limitations. Mass is a premium, and, as discussed in the propulsion section, one kilogram of payload makes necessary an additional 8.4 kilograms of propellant. Because of this major design limitation, the chosen scientific instruments have been limited to four. These include: an imaging system consisting of two cameras, an ultraviolet spectrometer, a magnetometer, and a radar mapper. Table 1.1 lists the four science instrumentation systems, their weights, and power requirements.

All four of these science instruments will have been flight proven by the launch of OPTIC. The camera imaging system is modeled after the Cassini/CRAF imaging systems. The design of the radar mapper is derived from the small radar mapper used with Pioneer Venus in 1978. Finally the magnetometer and the spectrometer designs were developed for and placed on Galileo.

TABLE 1.1

SCIENCE INSTRUMENTS

Instrument	Weight (kg)	Power (W)
Imaging System	36.6	44.75
Radar Mapper	9.7	18.0
Ultraviolet Spectrometer	5.33	5.33
Magnetometer	5.30	6.0
Total	56.93	74.08

Since mapping and photographic date are of highest priority for NASA in planetary studies, both the camera system and the radar mapper were first priority scientific instruments. The camera system is by far the largest and most massive science instrument. It and the other science equipment will be described separately below.

In the initial studies, a multitude of other possible instruments were examined. This was limited to five prior to choosing the magnetometer and UV spectrometer. The eliminated instruments are discussed in the following section.

In the end the choices were made based on science data desired rather than weight tradeoffs. The weight limitation merely eliminated the number of instruments, not type.

The magnetometer was chosen for two reasons. The first of those reasons is the scientific interest in the existence of a magnetic field about Pluto and its satellite. Information about magnetic fields about a planet was given as much priority by the ASU conference participants as topography and gravity.

The second important reason for the magnetometer links it to the imaging system in terms of mission success. This linking occurs when the usefulness of the data collected is considered. Data from magnetometer readings and camera images is more familiar to more members of the scientific world than other specialized data acquisition devices. (7. NASA, pg. 13) This means that data collected using these instruments will be of greater interest to more scientists and, therefore, may spawn more studies and analysis than other forms of data which may be relevant to only a few experts.

In order to attempt to determine the makeup of the Plutonian surface and its atmosphere, the UV spectrometer was added as the fourth instrument. The UV spectrometer fulfills important mission requirements based on the ASU conference conclusions recommending the investigation of mineralogical and elemental makeup of planets.

These four instruments are at present the only proposed science systems for OPTIC. As emphasized previously, mass limitations called for a compromise which would result in minimum weight with maximum useful scientific data collection. It is felt that these instruments will provide a wealth of information about Pluto. Their capabilities should allow OPTIC to fulfill the NASA/scientific request for visual images, topographical data, magnetic field information, and surface chemistry/mineralogy of Pluto and to a lesser degree its satellite Charon.

OTHER INSTRUMENTS CONSIDERED

Following are the major science instruments which were cut from the OPTIC science package: a laser altimeter, a gravity analyzing microprobe, and an infrared spectrometer. These three instruments were ruled out primarily because of the uncertainty of either the need for them, or the ability to develop them. Thus, they were eliminated due to these constraints, combined with the concern for mass, .

The most promising and valuable of these is probably the gravity analyzing microprobe. This concept envisions a microprobe mounted transponder, ejected from the main probe, and tracked using OPTIC's secondary antenna. Data collected concerning the microprobe's motion could be used for gravity field calculations for the Pluto - Charon system. Since this is an untested/developed concept it was shelved to conserve weight and save development costs. If significant advances in propulsion, trajectory, or budget, are made during the following development stage, more investigation in this instrument is recommended.

The laser altimeter is an exciting topographical data collection concept planned for the Mars Observer. (6. Mars Observer, pg. 79) Unfortunately, this instrument requires an extensive framework of information concerning the target body's gravitational field to function. The necessary data concerning Pluto is not available. Without this data, preflight calibration of an altimeter of this type would be nearly impossible.

Finally, the infrared spectrometer is an instrument which has travelled on numerous interplanetary missions before. Its main task has been the study of appreciable atmospheres. Since it is not believed that Pluto has much atmosphere in existence, the trade off for saving mass seemed a better proposition.

1.3 INSTRUMENT DESCRIPTIONS

IMAGING SYSTEM

The imaging system for OPTIC utilizes the Imaging Science Subsystem (ISS), originally developed for the Comet Rendezvous Asteroid Flyby (CRAF) and Cassini missions. This two camera system provides OPTIC with a reliable, flight tested (assuming both missions are executed prior to 2004), system that will not incur the extravagant costs which new system design implies.

The ISS has been designed for use on several missions. It has been developed in such a manner that it can easily be adapted to this mission to Pluto. The systems design provides for different data output rates, distinct data compaction options, and the ability to be used for navigation purposes.

ISS employs a wide angle camera and a narrow angle camera. Both of these draw upon a common electronics module. The relevant data for both cameras is outlined in Table 1.2. The two focal lengths provide for two. distinct scales of image resolution. Each contains filters which allow for varied spectral study of their focused target.

TABLE 1.2

CAMERA DATA
(Adapted from 5. JPL, pgs. 8, 9)

Camera	Narrow Angle	Wide Angle
Optics Type	Ritchey Chretien	Refractor
Focal Length	2000 mm	250 mm
Spectral Range	200-1100 nm	350-1100 nm
Filters	22	14
Field of View	0.35° square	2.8° square

The imaging system cameras can operated simultaneously and they can be calibrated and ready for use in less than an hour. The system is equipped with automatic exposure control and the frame time rates can be varied from 9 to 1479 seconds. The focusing ability for the Cassini/CRAF versions allow for 25 and 3.8 km passes, respectively. (5. JPL, pg. 8) While present planning for OPTIC place these altitudes well below the mission orbits (~ 2 Pluto radii from the surface) the 25 km value allows for great flexibility in imaging mission modification. (The 3.8 km value for CRAF is achieved with extra lenses). (5. JPL, pg. 8)

The frame time ranges are the total system process time. This includes exposure time and the time needed by the system to prepare for the next exposure.(filter rotation, etc.) The automatic exposure control is accomplished by taking two photographs. The first is used to supply irradiance information for the control system. The system then recalibrates and takes the second photo, which is the science image data. (5. JPL, pg.10)

The imaging subsystem weighs 36.6 kilograms and consumes a maximum of 44.75 watts of power. (1. Advanced Projects) They are both mounted on the three degree of freedom science platform. This provides for nearly unconstrained aiming possibilities. In this mannerboth cameras can be used to photograph Charon and any other targets of opportunity. The cameras can be activated in route for asteroid study, Jupiter analysis, and navigation backup. Figures 1.1 and 1.2 show the side view representations of both cameras, including their dimensions.

FIGURE 1.1

NARROW ANGLE CAMERA
(Adapted from 5. JPL fig 3-2)

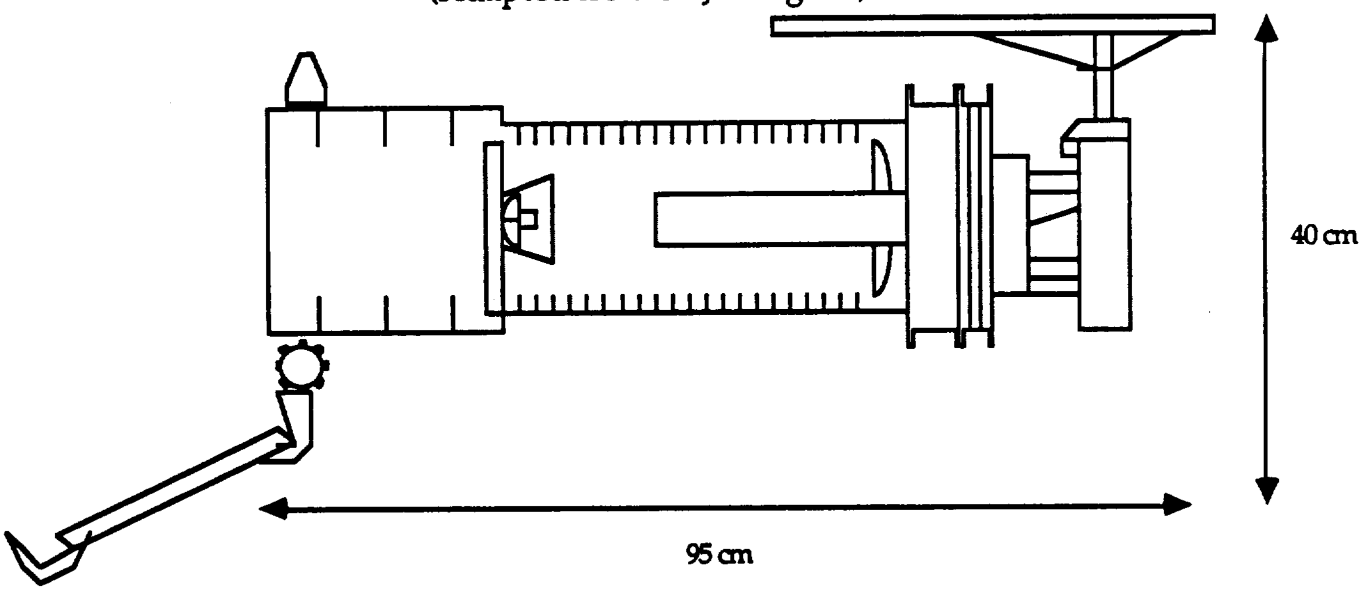
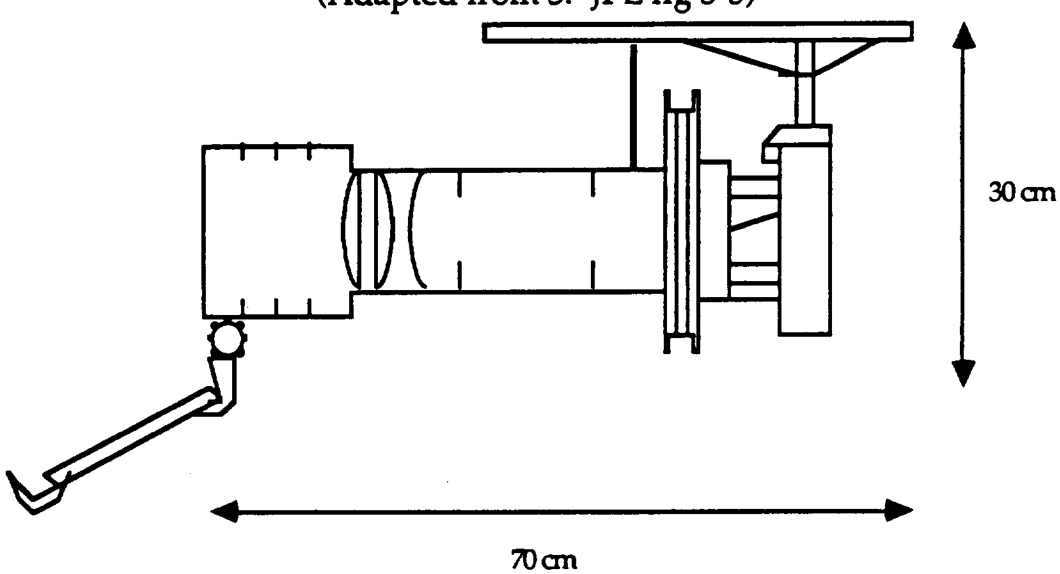


FIGURE 1.2

WIDE ANGLE CAMERA
(Adapted from 5. JPL fig 3-3)



IMAGING PROCESS

While the mass of Pluto is known, there remains some uncertainty concerning the value radius. Most recent information indicates a radius of 1150 km. (8. SAIC) The program used for this mission design trajectory calculations, MULIMP (see section 2), assumes a radius of 1500 km. This discrepancy presents problems for developing exact mapping times.

While the final mapping orbit is stated as an ellipse with a periapse of 2 Plutonian radii from the surface, this is a 700 km difference in altitude depending on the assumed radius. Since all trajectory mechanics are based on the 1500 km radius, this value was used for preliminary mapping calculations.

Using the narrow angle camera, and assuming 100% coverage with minimal overlap, photographs at orbit periapse would cover 336 square kilometers (see Appendix 1.1). Mapping is assumed to occur while the true anomaly ranges from 270° to 90°. this provides a pass time of 5.4973 hours. With this period of mapping time per orbit the total mapping duration is calculated to be 482.2 days. 17.46%, or 84.2 days, of this time is actually spent imaging the planet. These numbers are obtained using a photographic rate of 68.6 seconds per photo.

The ISS provides variable data rates to which the photo rate can be fitted. Using the available compression rate of 2 to 1, and assuming the automatic exposure control is activated (requiring two photos for every one science image), the frame time is 60 seconds. This mapping rate generates 175 kilobits per minutes of output while providing the highest resolution.(5. JPL, pg. 12) Exact resolution in units of size depends on altitude, but the narrow angle camera has a resolution of 6 microradians square and the wide angle 48. (5. JPL, pg. 8)

If upon arrival, fuel stores are sufficient orbit altitude can be decreased, and a slower rate (frame time) could be utilized. The slower output rate means lower resolution, but combined with the lower altitude, can provide near equal image resolution. The flexibility of the SSI allows OPTIC to proceed towards Pluto even without full knowledge of final science mission conditions. If, after arriving at Pluto, initial data collection can be used by scientists and controllers to adjust the final imaging course to an optimal route.

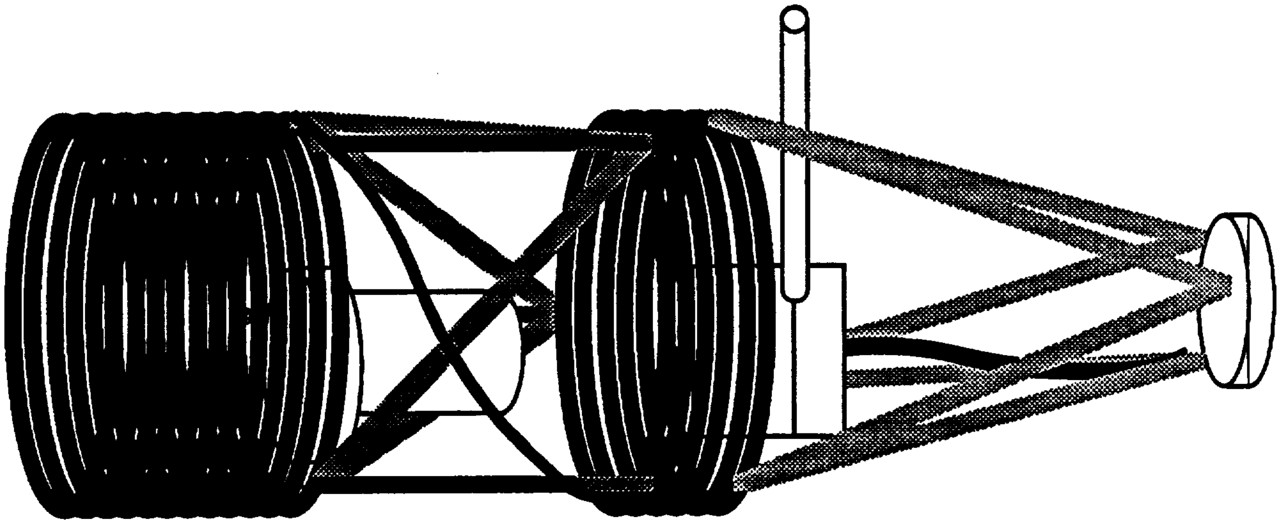
MAGNETOMETER

OPTIC's magnetometer is based upon the Galileo magnetometer design. When magnetometer data is to be collected the craft is put into a rotation rate of 3.15 rpm around its Z-axis. This spin is induced to obtain more totally encompassing field data on any existing magnetosphere.(9. Yeates, pg. 105) Because of this spinning motion, magnetometer data acquisition cannot occur simultaneously with the other instruments' data collection. This instrument is to be used in an attempt to answer questions pertaining to the existence of a magnetosphere about the planet.

The instrument's sensors, (see Figure 1.3) separated into two sets of three, are located at the end of the 10 meter boom. Separated slightly to provide correction for any magnetic fields generated by OPTIC's other electronics, the two sets measure magnetic fields of distinct intensities. The farthest set is sensitive over the span of ±32 to ±512 nT, and the second set from ±512 to ±16384 nT.(9. Yeates, pg. 131)

FIGURE 1.3

MAGNETOMETER SENSORS
(Adapted from 9. Yeates, pg. fig. 93 a.)



As the sensors move through space, both forward along the orbit trajectory, and about with respect to OPTIC's Z-axis, an analog voltage is generated proportional to the magnetic field. This is converted to a 16 bit digital signal by the magnetometers data system. The data system samples, averages, and stores the measured data prior to its transfer to the main OPTIC computer. (9. Yeates, pg. 133) From the main computer the data is transmitted to earth.

The data can either be processed and stored in "packets" that are partitioned by equal increments of time over the duration of the measurement period, or in a form of X,Y, and Z location coordinates separated data. There are also two lesser used processing modes involving the extremes of long duration measurements and short high speed data acquisition. When not operating a final mode merely places the system on hold for command changes. (9. Yeates, pg. 133)

The magnetometer weighs 5.3 kg and utilizes approximately 6.0 W of power.(3. Giampeoli) Data acquisition will be performed during each distinct orbit (arrival and mapping) in order to analyze the broadest reaches of the Plutonian magnetosphere. To conserve fuel in the control thrusters and increase coverage, the instrument will be activated during the last orbits of the arrival orbit path, remain on during the orbit change, and run during the initial passes on the mapping orbit. The system can also be activated to collect more Jovian magnetic data as OPTIC nears the planet for its gravity assist during the trip out.

ULTRAVIOLET SPECTROMETER

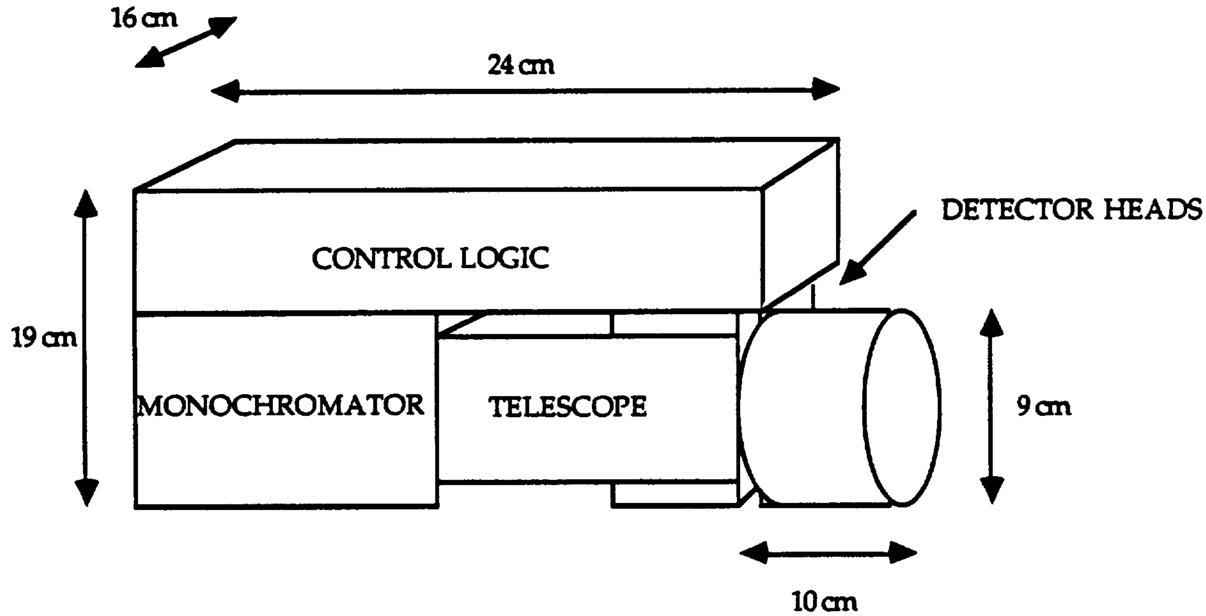
This instrument was also originally designed for the Galileo probe. For Galileo's mission it was used to analyze the Jovian atmosphere and its satellites' surface makeup. Without the highly significant atmosphere of Jupiter's system to analyze, OPTIC's UV spectrometer will be utilized to perform detailed analysis of the Plutonian surface. It will, additionally, if arrival orbit orientation permits, focus upon Charon's surface.

The spectrometer extends OPTIC's spectral range from the 1100 angstroms covered by the ISS to include the span between 1150 angstroms to 4300 angstroms. Its observations will provide ultraviolet reflectivity readings. This data is pertinent to the physical state in which the materials on the surface exist: i.e. ice, frost and grain size. It will also attempt to detect the presence of the elements hydrogen, oxygen, and nitrogen, in their atomic states.(9. Yeates, pg. 130)

The system has four major components which are housed within the instrument: a telescope, a monochromator, three detectors, and the system control logic unit. The components are identified in Figure 1.4.

FIGURE 1.4

SPECTROMETER ASSEMBLY
(Adapted from 9. Yeates, fig. 91)



The telescope has a 250 mm aperture which creates a field of view of 0.1° by 1.4° for the 1100 to 1900 and 2800 to 4300 angstrom detectors. The field of view for the 1600 to 3000 angstrom range is 0.1° by 0.4°. These field of views provide for spectral analysis of small selected regions upon the Plutonian surface.(9. Yeates, pg. 130)

The monochromator, with a reflecting diffraction grating of 125 mm focal length, disperses the ultraviolet light. Grating position (and, therefore, wavelength measured), is regulated by the control logic of the system which instructs the grating drive. The resolution is 13 angstroms in the first order spectrum, and 7 angstroms in the second order spectrum.(9. Yeates, pg. 131)

Photons which hit the 3 detectors produce pulses which are counted and read every 7E-4 seconds. In turn, these pulses are sent to the main computer for transmission back to earth.(9. Yeates, pg. 131)

The system processor can instruct the detectors to measure for only one given wavelength or view the entire UV spectrum approximately every 4 1/3 seconds. These, and the variations available in between allow investigations that range from single wavelength intensity changes across a large planetary swath, or broader general analysis.(9. Yeates, pg. 131)

The system weighs 5.33 kg and is run with 5.33 W of power.(9. Yeates, pg. 131) The spectrometer is mounted on the scan platform and runs simultaneously with the ISS providing a wide range of spectral coverage for all imaged targets.

RADAR MAPPER

The radar mapper of the OPTIC science subsystem is the most modified instrument to be utilized. This small mapper uses the secondary antenna to obtain topographic data of the planets surface. The radio science possible with this mapper includes the search for rings (all the outer planets have been found to have rings, with the exception of Pluto) and precise radii measurements. These are accomplished using occulation measurements of the received signals on earth. The extreme distance to Pluto make the success of this type of test improbable.(9. Yeates, pg. 55)

In upgrading the Radar mapper, the basic design from the Pioneer Venus craft will be modified to transmit on two bands rather than one. The addition of the 3 cm X-band, to supplement the 12 cm S-band, will provide better, more complete radar images of what is assumed to be a mostly rock surface.

The second main modification involves the use of the 1 m diameter secondary antenna dish. The Pioneer Venus version used a 0.38 m diameter dish. This increase in size will increase the overall topographic resolution of the data collected by the mapper.

The final difference in the OPTIC version is in its use. This model will be operated continuously during the photo mapping passes, rather than on a rotating basis as on the Pioneer Venus.

Utilizing the collected radar data, in conjunction with the data that will be obtained concerning Pluto's exact dimensions, it is hoped that absolute surface elevations can be calculated.

The Pioneer Venus system weighed 9.7 kg and used 18 W of electrical power.(2. Fimmel, pg. 58) It is assumed that the present technology may lower these values, even with the upgrading being planned,but the Pioneer Venus numbers have been used for all OPTIC system calculations.

APPENDIX 1.1

MAPPING CALCULATIONS

Assummptions from MULIMP (8. SAIC):

r_{pl} = 1500 km μ_{pl} = 663.5622 km^3/s^2 A_{pl} = 2.8274E7 km^2

Calculations and Data:

mapping orbit:	$r_{periapse} = 3r_{pl}$	$T_{map\ orbit}$	$= 2\pi(a^3/\mu_{pl})^{1/2}$
	$r_{apoapse} = 5r_{pl}$		= 113361.589 s
	$a = 4r_{pl}$		= 31.4893 hr
	$e = .25$		

Partial orbit period given by: $t=(a^3/\mu_{pl})^{1/2}[E-e\sin E]$ where E, eccentric anomaly, is defined as: $E=2\operatorname{atan}[((1-e)/(1+e))^{1/2}\tan(f/2)]$ f = true anomaly

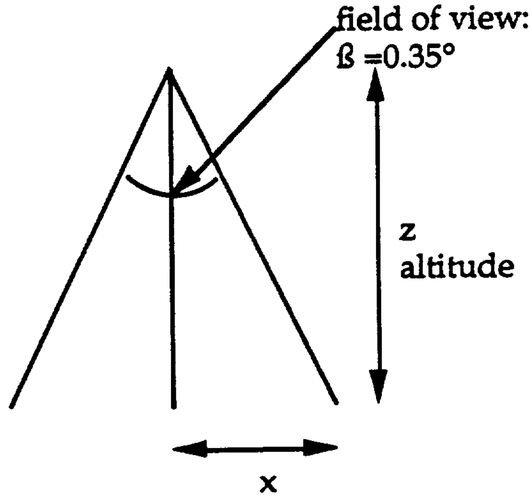
Mapping occurs from f = 270° and f = 90°

These equations yield a mapping pass time of 5.4973 hours.

APPENDIX 1.1 (cont)

Figure 1.5

MAPPING GEOMETRY



$x = z\tan ß$

$=9.163$ km

Narrow Angle Camera field of view: 0.35° square 0.35/2=ß (see figure 1.5)
Minimum coverage occurs at periapse (3000 km). One mapping pass covers one half of Pluto's circumference (4712.3890 km)

This yields a total area of 335.8428 $km^{\wedge 2}$. This value is the minimum coverage per photo. Dividing A_{pl} by this value yields the number of photos necessary for complete coverage: 84188 photos. Using the distance covered in one pass, 4712.389 km, and the coverage of each photo the value of 257 photos per mapping pass this reduces to one photo every 77 seconds. Actual mapping time becomes 75 days of actual photographing, or a total time to map of 430 days (including orbit time from f = 90° to f = 270°)

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2.0 MISSION MANAGEMENT, PLANNING AND COSTING

INTRODUCTION

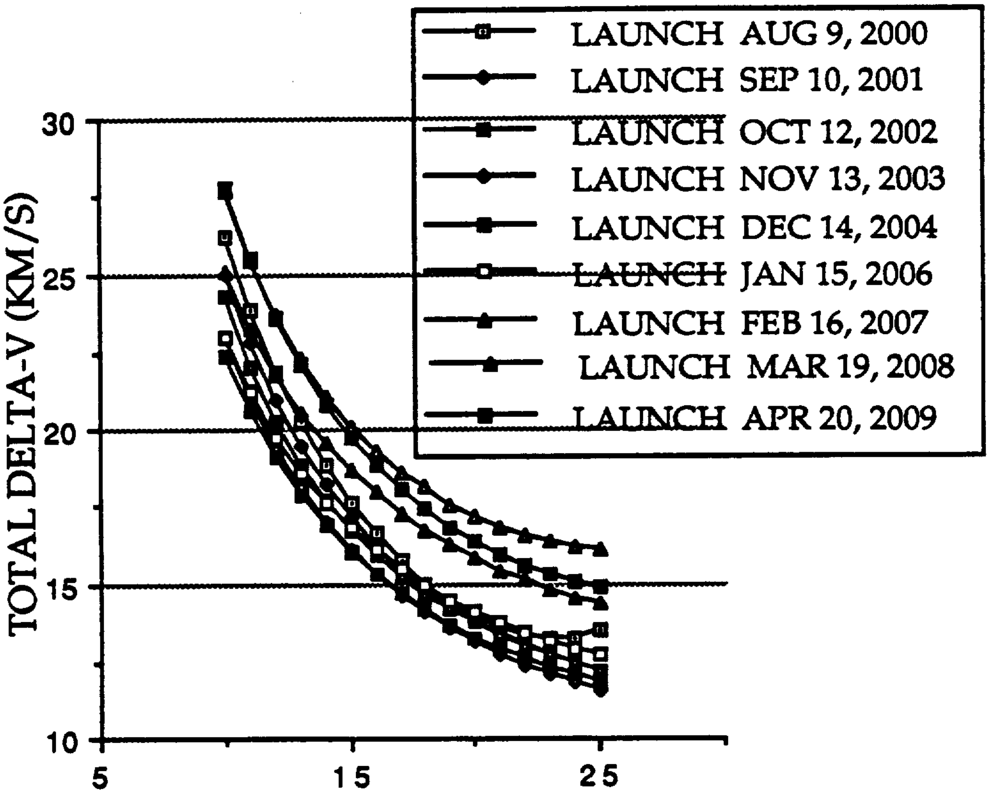
The primary requirement and problem for an orbiter is the need to keep the velocity at Pluto a minimum. If the velocity is too high, the amount of fuel needed for orbit insertion becomes unrealistic. Also of concern is the need to keep the delta-v at Earth departure at a minimum while reducing mission time as much as possible.

This section was prepared and written by Randall John Hein.

2.1 METHOD OF ATTACK

To find the optimum orbits a technique using synodic periods was used. Each pair of planets in a trajectory (excluding Pluto) were examined. The optimum transfer point was propagated through our 10 year launch window . When only 2 planets (excluding Pluto), such as Earth and Jupiter, are involved in a trajectory, a mission time frame of 10 to 24 years was examined in each synodic period. We see from Figure 2.1 that the optimal launch date will be about December 14, 2004 or November 13, 2003, depending on mission length. When 3 or more planets are involved , such an in a Earth-Mars-Jupiter-Pluto trajectory, the optimum points in each pair of synodic periods were compared until one ore more viable launch dates could be found. Each of these was then examined for a 10 to 24 year mission.

FIGURE 2.1
FLIGHT TIME VS. TOTAL DELTA-V
FOR JUPITER PLUTO TRAJECTORY
WITH ORBIT INSERTION AT PLUTO



FLIGHT TIME (YEARS)

Many possible mission plans were examined. Trajectories similar to that used in the Galileo mission(1. D'Amario) were tried first . The Venus-Earth-Jupiter-Pluto trajectory proved to yield a better initial delta-v than the a Jupiter-Pluto trajectory (fig 2.2), but the final delta-v was too high (fig. 2.3) . The same problems accrued with other trajectories that use inner planets for a gravity assist as opposed to strictly using outer planets. A Mars-Jupiter-Pluto trajectory was also considered. Though the total and initial delta-v's were the best (fig. 2.4 and 2.2), the Jupiter-Pluto trajectory gave the better final delta-v (fig 2.3).

FIGURE 2.2
COMPARISON OF INITIAL DELTA-V'S
FOR PLUTO MISSION WITH ORBIT INSERTION

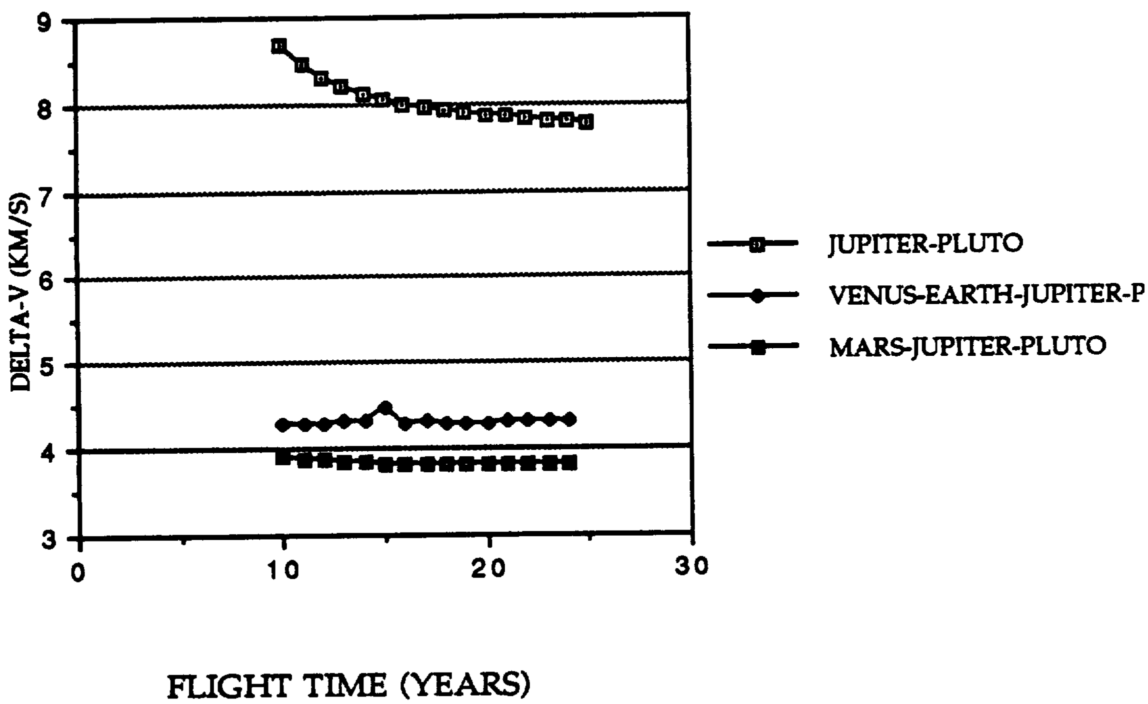


FIGURE 2.3
COMPARISON OF FINAL DELTA V's
FOR PLUTO MISSIONS WITH ORBIT INSERTION

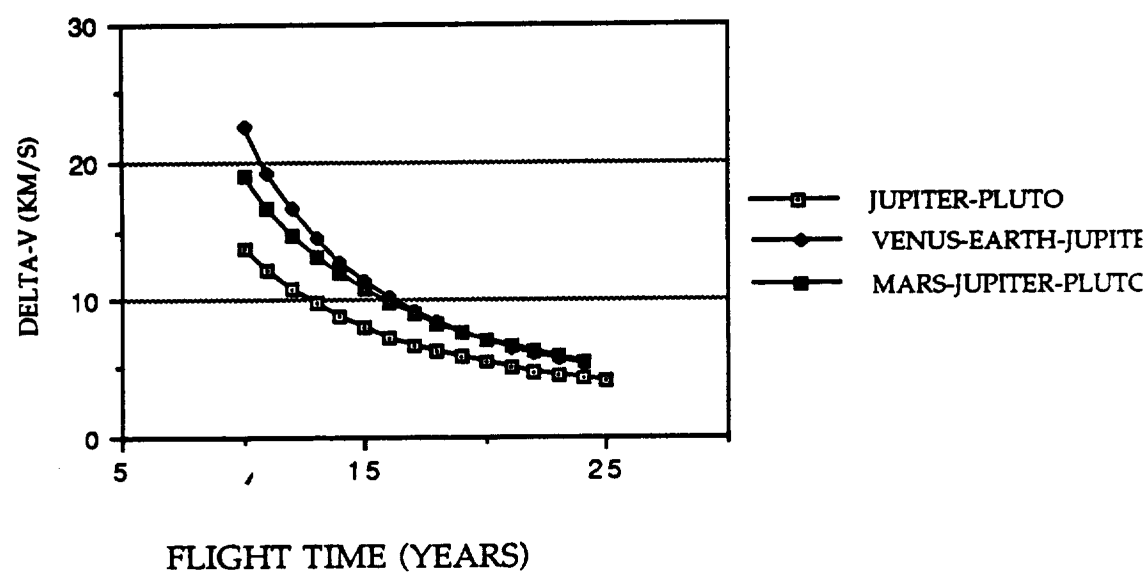
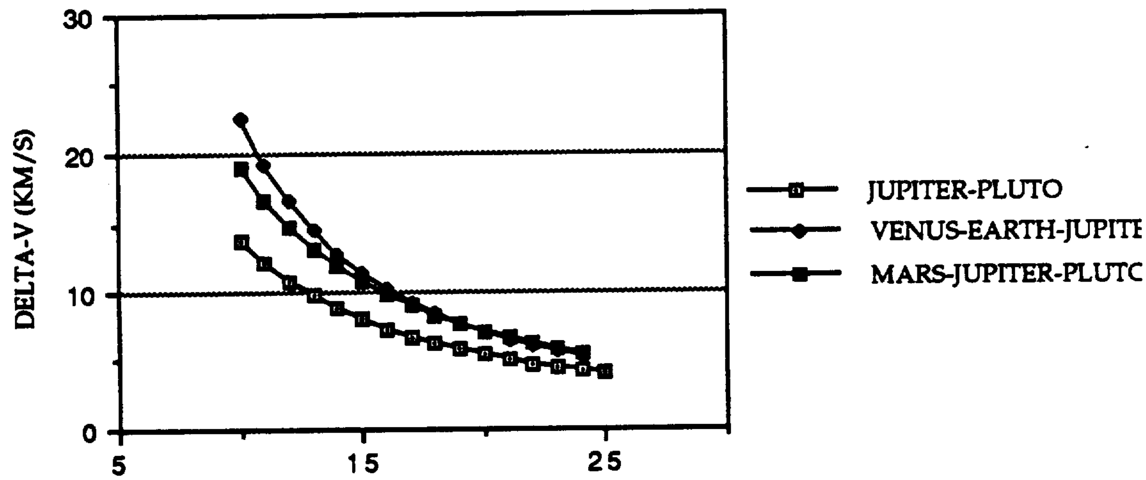


FIGURE 2.4
COMPARISON OF TOTAL DELTA V's
FOR PLUTO MISSIONS
WITH ORBIT INSERTION

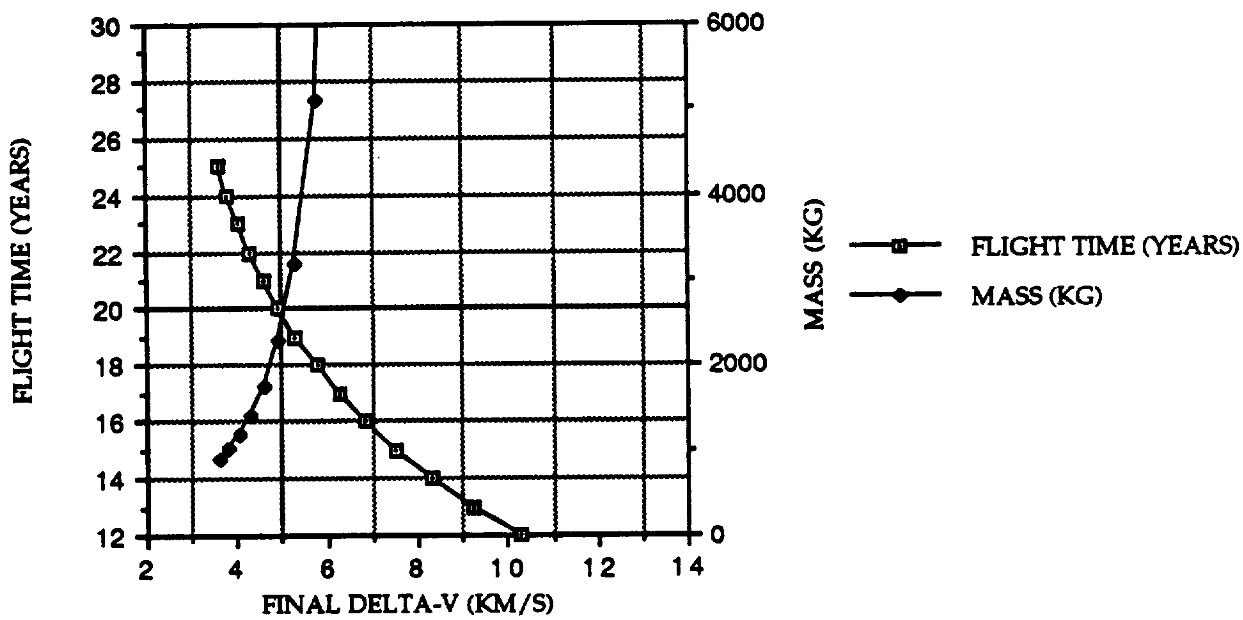


FLIGHT TIME (YEARS)

Saturn alone was not a useful gravity assist body. The only possible advantage to using Saturn would be in a Saturn-Jupiter trajectory. Unfortunately, Saturn lags Jupiter's orbit during our prescribed launch window. A similar problem occurs when using Neptune or Uranus for a gravity assisted deceleration. Pluto's orbit lags Neptune's and Uranus', thus eliminating the option.

The Jupiter-Pluto trajectory was finally chosen because of the need for a low final delta-v. The decision for a 20 year mission came from the need to keep the fuel weight for the final delta-v under 3000 kg. By looking at figure 2.5 one can see the that a mission time of 20 years or more is required to keep the final delta-v under the needed 5 km/s.

FIGURE 2.5
FLIGHT TIME VS. FINAL DELTA-V
COMPARED WITH
FUEL MASS VS FINAL DELTA-V



DELTA-V REQUIREMENTS

A Jupiter-Pluto trajectory was chosen with a gravity assist at Jupiter. Once in Low Earth Orbit (LEO), 300 KM above the Earth's surface a delta-v of 7.116 km/s would be needed for orbit departure. At 576.5 days into the mission OPTIC would reach its close approach point of Jupiter at 2,196,911 km from the planet's surface. At the 20 year mark, an orbit insertion burn (delta-v = 4.927) will be needed. The orbit radius at perhaps will be 4500 km and 15000 km at periapse. Upon completion of required data acquisition (approximately 17 days) a second burn will take place (delta-v = .0470 km/s) at periapse. This will put OPTIC in an orbit with a radius of 7500 km at apoapse and 4500 km at periapse. The reason for the non circular orbits is to reduce the delta-v needed for orbit insertion.

TIMELINE

DECEMBER 13, 2004
Launch OPTIC aboard Shuttle-C, using an initial Space Transfer Vehicle (ISTV) for departure from LEO.

JULY 12, 2006
Jupiter Encounter: OPTIC will study Jupiter for 5 days while using the planet for a Gravity assist.

December 13, 2024
Pluto Encounter: Optic will do an orbit insertion burn to get into a eccentric orbit about Pluto. The time in this orbit will be used for scientific study focussed at Charon.

JANUARY 1, 2025 [This date is dependent on scientific needs.]
Orbit Maneuver: OPTIC will move into a lower orbit about Pluto to allow for mapping and further scientific study of Pluto.

DECEMBER 13, 2026 [This date is dependent on scientific needs.]
Mission ends.

2.2 COSTING

The cost of OPTIC comes to $999.78 Million in fiscal 1989 dollars. An additional charge for the ISTV booster (cost unavailable) and Shuttle-C must be included. The estimated cost for the Shuttle-C is about $2000 per pound which comes to about $69.4 Million(2. Kolcum, p.134). To keep the cost down, we used as many unmodified components as possible. Due to the high delta-v required for orbit insertion, a propulsion system requiring major modifications had to be developed. Several other subsystems were forced to do major modifications or design new components due to the duration of the mission and the distance from Earth.

APPENDIX 2.1

<u>FORMULAS USED FOR COSTING</u>(costs are given in fiscal 1977 dollars)

DEVELOPMENT PROJECT - FLIGHT HARDWARE

STRUCTURES & DEVICES	Labor hrs. to labor cost = 34.52
DLH = 1.626 (N*M)^0.9046	RLH = 1.399(N*M)^0.7445
Thermal Control, Cabling & Pyrotechnics	Labor hrs. to labor cost = 34.03
DLH = EXP(4.2702+.00608*N*M)	RLH = 3.731(N*M)^0.6082
Propulsion	Labor hrs. to labor cost = 38.11
DLH = 56.1878(N*M)^0.4166	RLH = (N*M)^0.9011
Attitude & Articulation Control	Labor hrs. to labor cost = 35.58
DLH = 21.328(N*M)^0.7230	RLH = 1.932(N*M)
Telecommunications	Labor hrs. to labor cost = 33.49
DLH = 4.471(n*m)^1.1360	RLH = 1.626(N*M)^1.1885
Antennas	Labor hrs. to labor cost = 34.52
DLH = 6.093(N*M)^1.1348	RLH = 3.339(N*M)
Command & Data Handling	Labor hrs. to labor cost = 30.61
DLH = exp(4.2605+0.02414*N*M)	RLH = EXP(2.8679+.02726*N*M)
RTG Power	Labor hrs. to labor cost = 30.21
DLH = 65.3(N*M)^0.3554	RLH = 7.88(N*M)^0.7150
Power	Labor hrs. to labor cost = 32.77
DLH = EXP(3.9633+0.00911*N*M)	RLH = EXP(2.5183+.01204*N*M)
Radar	Labor hrs. to labor cost = 31.83
DLD = 11.409(N*M)^0.9579	RLH = 1.2227(N*M)^1.2367
Imaging	Labor hrs. to labor cost = 34.14
DLH = 4.463(N*M)^1.0369	RLH = (N*M)^1.1520
Particle & Field Instruments	Labor hrs. to labor cost = 36.05

DLH = 25.948(N*M)^.7215 RLH = 0.790(N*M)^1.3976

Remote Sensing Instruments Labor hrs. to labor cost = 35.0

DLH = 25.948(N*M)^.5990 RLH = .790(N*M)0.8393

DEVELOPMENT PROJECT - SUPPORT FUNCTION

System Support & Ground Equipment

DLH = .36172(SUM DLD Hardware) Labor hrs. to labor cost =32.45

Launch +30 days Operations & Ground Software

DLH = 0.09808(SUN DLH Hardware) Labor hrs. to labor cost = 34.42

Imaging Data Development

DLH = .00124(Pixles per line)^1.629 Labor hrs. to labor cost = 35.87

Science Data development Labor hrs. to labor cost = 50.87

DLH = 27.836(non-imaging science mass)^0.3389

Program Management Labor hrs. to labor cost = 31.07

DLH = 0.10097(SUM DLH all categories)^0.9670

FLIGHT PROJECT

Flight Operations Labor hrs. to labor cost = 33.90

DLH = ((SUM DLH Hardware/3100)^.6)*(10.7MD + ED)

Data Analysis Labor hrs. to labor cost = 35.76

DLH = 0.425(DLH Flight operations

COST REDUCTION ALGORITHM BY INHERITANCE CLASS

X1 = % of subsystem off- the-shelf

X2 = % of subsystem exact repeat

X3 = % of subsystem minor modifications

X4 = % of subsystem major modifications

X5 = % of subsystem new design

Z = 1(X1) + .8(X2) + .25(X3) +.05(X4) + 0.0(X5)

Total costs = (100%-z)NRC + RC

NRC = Non-recurring cost = (DLH-RLH) * Labor hrs. to total cost

RC = Recurring costs = RLH * Labor hrs. to total cost

CONVERSION FROM 1977 DOLLARS TO 1989 DOLLARS

Cost in 1989 = Cost in 1977 * (894.7/505)

TRAJECTORY PLANNING

All figures were derived using MULIMP (2. Frielander)

2.3 BIBLIOGRAPHY

1 D'Amario,Louis A "Galileo 1989 VEEGA Mission Description" AAS/AIAA Astrodynamics Specialist Conference, August 7-10, 1989

2. Frielander, Alan L. "MULTI-IMPULSE TRAJECTORY AND MASS OPTIMIZATION PROGRAM" , Science Applications, Inc 1974

3. Kolcum, Edward H. "Decision on Shuttle-C Expected After Internal NASA Review," Aviation Week and Space Technology. June 19, 1989, pp 123-125.

3.0 POWER AND PROPULSION

INTRODUCTION

The following section will explain the design process of both the power and propulsion subsystems of OPTIC. This section was made especially difficult because OPTIC's mission plan calls for an orbit insertion about Pluto after a 20 year voyage. These two criteria put an incredible demand on both subsystems. This section was prepared by David L. Meyer during the spring of 1990.

3.1 POWER SUBSYSTEM

The first problem that needed to be addressed was the selection of the power system. This process was simplified because of the type of mission. Solar cells were ruled out immediately because of the distance from the sun that OPTIC will be travelling. The power that solar cells can produce greatly diminishes past a distance of 2 A.U.'s (~3E+8 km) from the sun. The fact that OPTIC will be orbiting Pluto at a distance of approximately 40 A.U.'s makes the use of solar cells impossible because the power at just 6 A.U.'s is reduced to about 5-10% of the power available at 1 A.U. (5. Koepke, p 11) With solar cells an impossibility, the only alternatives are nuclear power systems.

Incorporating a nuclear power system into OPTIC will most likely bring out some political opposition. The groups that presently oppose the use of nuclear power in space (SANE/Freeze, Citizens to Stop Plutonium in Space, and the National Mobilization for Survival) will probably still be active well into the twenty-first century. Their argument is based around two possible disasters: an explosion at launch and a possible reentry during a fly-by of Earth. The possibility of an explosion during launch has thoroughly tested for in existing Radioisotope Thermoelectric Generators (RTGs). The RTGs have been subjected to both loads of up to 2000 psi and projectiles traveling at

TABLE 3.1

PREDICTED RTG ADVANCEMENTS

Source	Specific power (W/kg)	Date
Koepke	11	1990
Schock	~10.5	"future"
Mondt	10-13	2000

speeds of 360 m/s. The Challenger accident resulted in loads of only 10 psi and in the case of a solid rocket booster exploding, the shrapnel would be traveling at approximately 90 m/s (9. Nichols, pp 8-15). This data suggests that RTGs should be 100% safe during a launch. This leaves the case of a possible reentry into Earth's atmosphere during a fly-by. Our trajectory calls for OPTIC to leave low-Earth orbit (LEO) and to never return near Earth-space. Given existing test results and the trajectory that OPTIC will take, nuclear power will be a very safe option.

Now that the power source will be nuclear power, the issue is whether to use a nuclear reactor or RTGs. The problem encountered in the past with nuclear reactors has been their extremely high mass and need for shielding. Recently, the mass of these reactors has been rapidly decreasing with technological advancements. It has been forecasted that the specific power of nuclear reactors could reach as high as 55 W/kg by 1991 (5. Koepke, p 17). However, at this date, no nuclear reactors have been flight tested.

Presently, RTGs have a relatively low specific power of about 5 W/kg (5. Koepke, p 17). According to a number of sources, specific power of RTGs could reach as high as 13 W/kg (see Table 3.1).

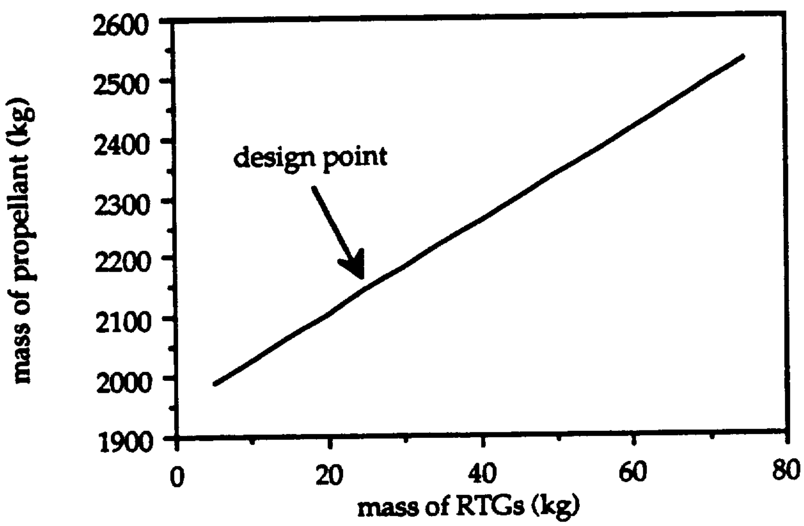
Although nuclear reactors will have superior specific power, other factors must also be weighed. Our mission duration is over 21 years. This figure does not include any time in storage or grounded because of various reasons. The time at which the demand on the power supply will be the greatest is when it will be needed the most. This time will be over 20 years after the launch date while OPTIC is in orbit about Pluto. The mission will be a complete waste if the power system is not working properly after 20 years. Knowing this, design lifetime, past results and reliability play a much larger part in the power system design. For this mission, those criteria outweigh specific power.

By the year 2000, design lifetime for nuclear reactor thermoelectric conversion will be from 10-14 years with 10 years being the most "probable" value (8. Mondt, p 40). This is less than half of our mission duration and there is no flight-tested data to suggest a longer lifetime.

The current design lifetime for RTGs is about 5 years. Lifetime predictions for the year 1990 are roughly the same as for nuclear power generators (~10 years). This figure will most likely keep demonstrating an upward trend towards 1999 (8. Mondt, p 47). However, what separates the 10 year design life of RTGs from the 10 year design life of nuclear reactors is the flight-proven data from various missions where RTGs were used. RTGs that have been used in the past have provided both longer lifetimes and power that was well above the predicted values (1. Bennett, p 327). Some good examples of this are the Voyager probes that were launched in the early 1970's and are still operating. Conditions that RTGs must undergo can also affect the lifetime. For example, reducing the hot junction operating temperature of the RTG assembly can prolong the life of the RTGs substantially (8. Mondt, p 48).

Since solar cells are not feasible, the decision of which power system was decided by the trade of mass for dependability and longer life. The fact that an orbiter is being attempted requires that weight be minimized as much as possible. However, a light power system that does not work upon arrival cannot compare to a heavier power system that is dependable.

FIGURE 3.1
RELATIONSHIP BETWEEN
PROPELLANT MASS AND RTG MASS



Now that a power source has been chosen, it has to be designed around the power draws of the other subsystems. The specific power of the RTGs, for design purposes, was set at 12 W/kg. As stated earlier, predicted values for specific power by the end of the 20th century are 10-13 W/kg. Although 12 W/kg is not the most conservative estimate, it is just above the middle of the predicted range, so it should not be very far off (if any) from the actual values. As a result of this high specific power, no batteries were required because the various subsystems will be able to draw power off of the main power source without an appreciable weight loss. The lack of batteries takes a lot of the complexity out of the power system. With a redundant power distribution from the RTGs, this should be a very simple and reliable power system while, at the same time, not being overly massive.

The members of the design team were told to keep power draws down to a minimum. Figure 3.1 shows the relationship of the mass of the RTGs to the mass of propellant needed to achieve the orbit insertion into Plutonian space. Although the relationship is linear, it is hardly a 1:1 ratio. Every

TABLE 3.2

SUBSYSTEM POWER DRAWS AND MASSES

Substystem	Max. Power (W)	Mass (kg)
Scientific instrumentation	61.38	56.93
Articulation & control	20.0	68.1 (inculdes 40 kg of Hydrazine)
Command & communications	10.0	67.0
Structural	145.0	80.0

kilogram of RTGs added to the power system adds 7.73 kg of propellant which, inturn, adds .67 kg of tank weight. Therefore, for every kg added to the power system, 8.4 kg of extra mass is added to OPTIC. Relating these figures to Watts, every Watt of power needed for a subsystem adds .78 kg of mass to OPTIC. The above information illustrates how important rationing of power is. The final values of subsystem power draws, along with masses, are shown in table 3.2.

The degradation rate of the RTGs had to be determined in order for OPTIC to be operational at Pluto. A graph relating the ratio of power output to original power to time was available in Bennett et al. on page 327. This contained the following data: LES pre-launch prediction, LES actual data, Voyager pre-launch data and actual Voyager data. The actual data of both cases was much better than the predicted values. There was also a substantial improvement from the LES results to Voyager's results. The plot covered a time period of 44000 hours. The curves were very conservatively extended to 240000 hours (27.38 years). Included in these 27.38 years are the 20 year mission time and an allowance of 5 years in storage or grounded preceding the launch. This leaves 2.38 years at Pluto. Information from the Scientific Instrumentation subsystem states that it will take 1.18 years to completely map Pluto. Excluding the pre-launch allowance of 5 years, this is a cushion of about 1 year. After 240000 hours, the ratio of power output to original power is at .770 for actual Voyager data and .722 for predicted Voyager data. For design purposes, the ratio of .77 will be used. This is *highly* conservative because the curve was extrapolated conservatively and it is based on out-dated technology. However, because of the long mission duration, conservatism is best.

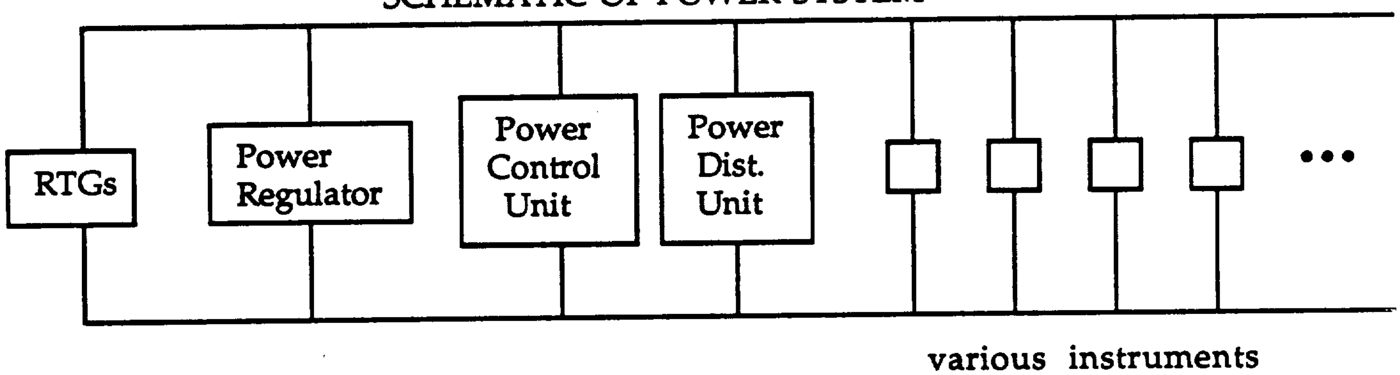
Using the total peak power of 236.38 W, dividing by 12 W/kg and then dividing by .77, the mass of the RTGs that need to be installed initially into OPTIC is 25.582 kg. This is a starting power of about 307 W. The power that will be left on OPTIC, once in Plutonian space, will be able to keep every instrument running simultaneously for 2.38 years. This substantial time cushion, along with the conservative estimate of the available power, should keep OPTIC operational for many years after orbit insertion.

Additional components for the power system are as follows: a power regulator, a power control unit and a power distribution unit. It was assumed that each subsystem would provide their own inverters. The three components mentioned above were taken straight from the equipment list of the current Galileo probe. The reasoning behind this was that these devices had to be of the highest quality and dependability because of the myriad of electronics and experiments on board Galileo. Although OPTIC's equipment list is minimal, dependable, heavy-duty electronic regulators and distributors are needed to maintain a steady power signal throughout the full duration of the mission. Another positive factor of the Galileo equipment is that there will be some flight data which could point out potential problem areas before installation into OPTIC.

The need to minimize weight was explained above. Since the electronic subsystem is the heart of any mission, any failures could severely jeopardize the mission's success. The one problem area that the power system might encounter are breakdowns because of the long mission duration. So, when selecting components and a power supply, reliability was stressed more than weight savings. Even after these precautions to guard against any failures, a redundant system was attempted (as shown in figure 3.2). In sum, barring any catastrophic failures, the power subsystem should provide reliable power and it will most likely outlast OPTIC's mission time.

FIGURE 3.2

SCHEMATIC OF POWER SYSTEM



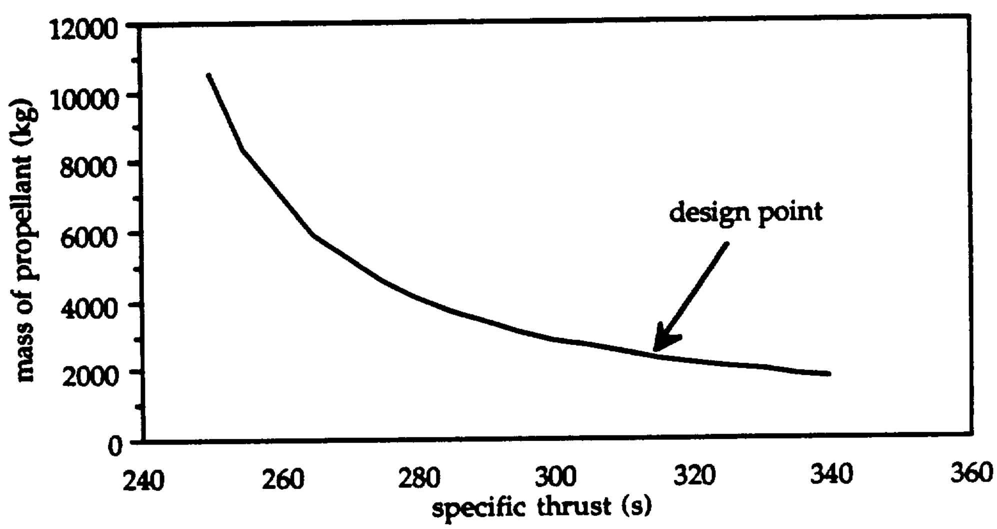
3.2 PROPULSION SUBSYSTEM

Similarly to the power subsystem, the first problem encountered with the propulsion subsystem was the selection of a propellant. Because our mission calls for an orbit insertion about Pluto, there will be a great demand on the propulsion system. The change in velocity (Δv) needed at Pluto is 4927 m/s. The mission duration is about 20 years. This combination of a high Δv and the need for long-term storage presents quite a problem.

If the high Δv was the main focus, a propellant with a high specific thrust (I_{sp}) would be the best choice. Similarly, if the mission duration was the only aspect to be analyzed, a highly storable propellant would be chosen.

The combination of these two parameters in an existing propellant is hard to find. Most stable propellants have relatively low I_{sp} values and the high performance fuels are not storable for long periods of time (12. Sutton, pp 168-182).

FIGURE 3.3
PROPELLANT MASS NEEDED FOR
Δv OF 4927 m/s vs. I_{sp}



More preference was given to stability because if the propellant has boiled off or eaten away the tanks before arrival at Pluto, there would be no orbiter. However, the storable fuel must have a good performance value so the tank size does not become infeasible.

The question of whether to use a monopropellant or a bipropellant also needed to be addressed. Monopropellants have the advantage of simplicity while giving up some performance values, as compared to bipropellants. One of the most common monopropellants, hydrazine, has a theoretical I_{sp} of ~300 s. Certain storable bipropellants can have actual I_{sp} values of 310-320 s. A comparison of propellant needed to execute the Pluto orbit insertion, as related to I_{sp}, is shown in Figure 3.3. Although bipropellants usually have better performance values, firing them is a much more complicated process because of the need for exact mixing. This possibility for error will result in more unused propellant than one would get with a monopropellant (6. Koepke, p 20).

Other desirable qualities of propellants include high specific gravity and a low freezing point. A relatively high specific gravity will result in smaller tanks (12. Sutton, pp 168-182). Our mission requires minimizing mass wherever possible, so bulky tanks brought on by a low density propellant could jeopardize the success of the mission. At the distance OPTIC will be from the sun, the temperature will be extremely cold. Frozen propellant prior to the orbit insertion could also threaten the mission's chances.

The final choice was the bipropellant consisting of nitrogen tetroxide and hydrazine (N_2O_4/N_2H_4) in use with a LEROS 1 engine. The mixture of N_2O_4/N_2H_4 combines stability with performance when coupled with a LEROS 1 engine. The LEROS 1 is capable of 500 N of thrust and an actual I_{sp} of 316 s. The current LEROS is configured to perform with MON3 as the oxidizer and either hydrazine or monomethyl hydrazine as the fuel. It is a very small engine with a thrust chamber length of 12.7 cm and a mass flow of only .162 kg/s (4. Gray, pp 2,15). Because of the oxidizer adaption, an I_{sp} decrease to 315 s will be used for the design.

Most of the criteria that is desired in a propellant are satisfied by the N_2O_4/N_2H_4 combination. Hydrazine is storable for long periods and has been used repeatedly for deep space probes. It is compatible with four types of stainless steels: 303, 304, 321 and 347. It has one of the higher densities of liquid fuels at 1008 kg/m^3. The major drawback of hydrazine is its high freezing point of 274.3 °K (12. Sutton, pp 170-181). Nitrogen tetroxide also displays some great properties as an oxidizer. It is compatible with all stainless steels so this eliminates the need for different tank materials (10. Parcel, p 508). Like hydrazine, nitrogen tetroxide has a very high density of 1447 kg/m^3 and a high freezing point of 361.5 °K. However, nitrogen tetroxide makes up for its high freezing point by being able to "...be stored indefinitely in sealed containers made of compatible materials." (12. Sutton, pp 171,178) The high freezing point can be overcome by strategic placement of heaters (see Structures subsystem). Since hydrazine and nitrogen tetroxide need to be kept at approximately the same temperature, regulation should not be a problem. This combination is also hypergolic, so no igniters will be needed; just mixing (12. Sutton, pp 170-181). This propellant combination is dependable and should perform at Pluto when it is needed the most.

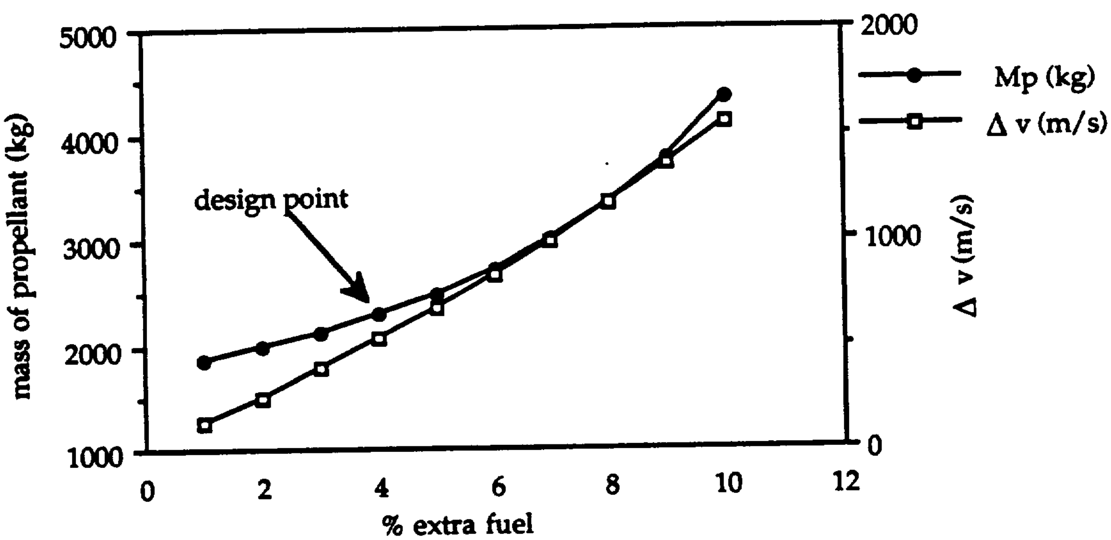
Because of its size, one would assume that the LEROS engine is a poor option. However, since reliability has been the main factor stressed throughout the design process, this makes the LEROS an excellent selection. In order to make the Δv at Pluto of 4927 m/s, it will require a long burn. There will also be trajectory control maneuvers (TCMs) throughout the mission. At its present configuration, the LEROS has a firing time of about 12000 s (4. Gray, p 7). If more than one engine is used, this is more than enough time to complete the mission's Δv requirements. Along with its proven reliability, its performance values are virtually unmatched by any other engines with storable propellants.

The pressurization technique that will be used will be used is basically none. For propellant feed, a technique called liquid reorientation/settling will be attempted. This technique depends on the bond numbers (B_o) of the propellants. As long as $B_o > 1$, settling of the liquid will occur and the engines will be providing the pressure. This has been used successfully in both the Saturn V and the Centaur upper stage (2. Cramer, pp 1-5). As shown in Appendix 3.1, the minimum thrust needed for settling is 1.45 N. A diaphragm that only lets fluid out will be at the bottom of both tanks. There should always be enough propellant in the lines to provide this small thrust. However, in a case where there is not enough fluid in the lines, there will be two 15 N thrusters that are pressurized with small tanks that are to be used for attitude control (see that section). These thrusters will be pointed in the same direction as the engines so settling should still be able to be achieved.

Now that propellants, engines and tank materials have been chosen, they must be sized according to the Δv at Pluto and any TCMs expected throughout the mission. Every subsystem was told to minimize mass wherever possible. The results were excellent and are in Table 3.2. Because of my present level of education, all Δv's will be calculated using the impulsive burn approximation (see Appendix 3.1). It was assumed that 20 out of the 40 kg of hydrazine alloted to the attitude control system would be used prior to the burn at Pluto. Another assumption was a structural efficiency (ε) of .08 (Buckmaster, 1989). This is a very conservative estimate because there will surely be advances in materials within the next decade. However, it is probably accurate for this problem because of the need for extra insulation for storage of the propellants.

The next constant that needed to be set was the percentage of extra fuel needed for last minute corrections and maneuvers once in orbit about Pluto. Figure 3.4 shows the amount of propellant needed against the percentage of extra fuel included. Figure 3.4 also shows the approximate Δv capability once orbit about Pluto is achieved. This was the deciding factor. The percentage was set at 4 for the initial calculation and the amount of propellant added to make the orbit insertion was not too burdensome. The fact that Charon, Pluto's moon, is relatively close, there will probably be a need to execute some orbit corrections during the mapping. Shortly after orbit insertion, an orbit correction that requires a Δv of 47 m/s will be executed. This leaves OPTIC with about 400 m/s of additional Δv's.

FIGURE 3.4
Mass of Propellant needed for Plutonian Orbit
Insertion and Δv Capability After Insertion as Related to % Extra Fuel



A problem encountered with these iterations is their circularity. In order to overcome this, after iterations were made, certain values had to be fixed. Please refer to Appendix 3.1 for details on these calculations. After the first propellant mass iterations, the mass of the extra fuel was fixed at 95 kg. Then, more propellant needed to be added for TCMs. A TCM capacity of around 400 m/s was incorporated into OPTIC. Because of the duration of the mission and the need for a precise approach to Pluto, this high TCM ability is worth the added mass.

The small mass flow of the LEROS engine makes it necessary to have more than one engine to perform the Δv at Pluto. Multiple engines are also needed to keep the burn time under LEROS' 13000 s rating. The shortening of time reduces the distance over which the burn will be made and, in turn, make the impulsive burn approximation more accurate. Figure 3.5 illustrates

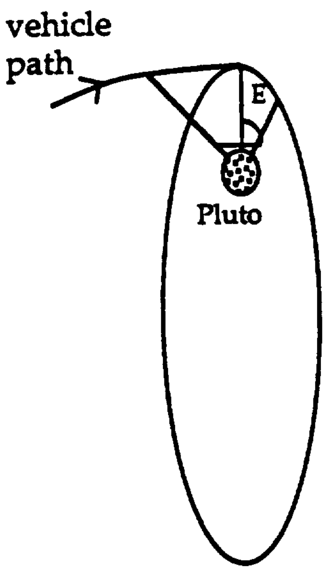
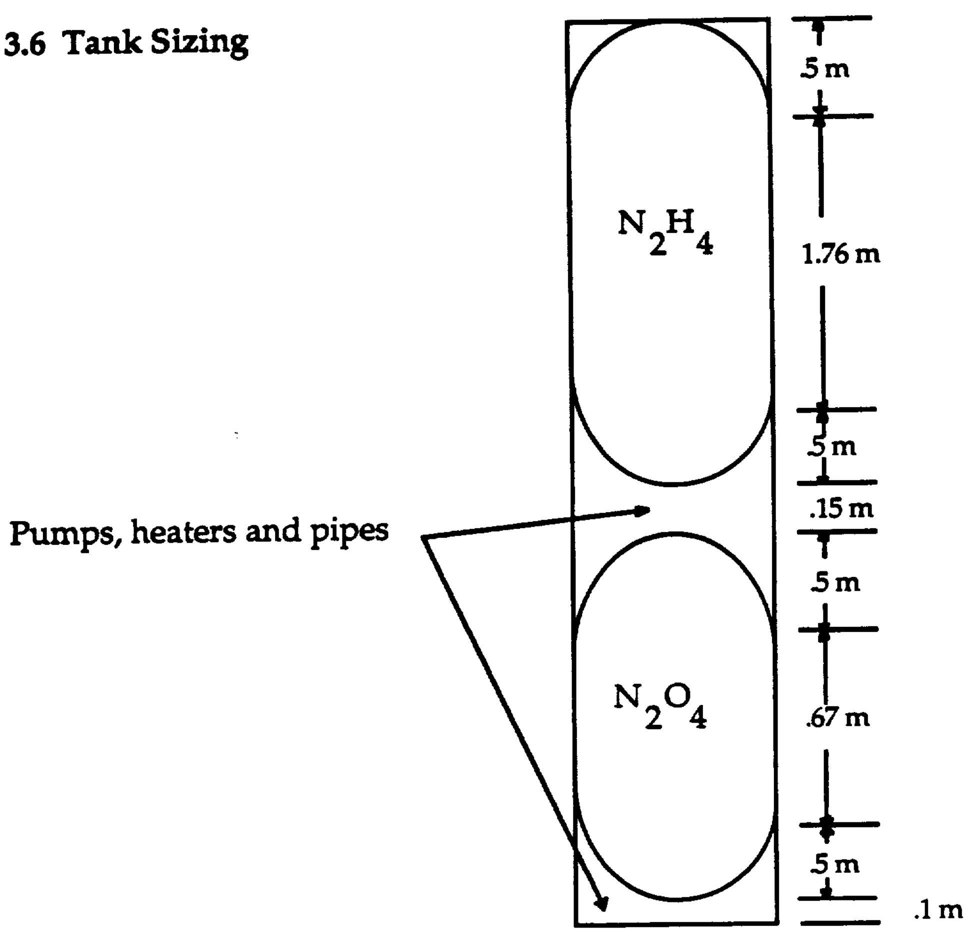


TABLE 3.3 Engine Configurations

Engines	Burn time (s)	Load (g's)	E (deg)	Total angle (deg)
5	3168.57	.16	5.28	15.84
4	3960.71	.13	6.56	19.68
3	5280.95	.10	8.73	26.19
2	7921.43	.06	13.02	39.06
1	15482.85	.03	25.36	76.08

FIGURE 3.5 Physical Length of Burn Time

FIGURE 3.6 Tank Sizing

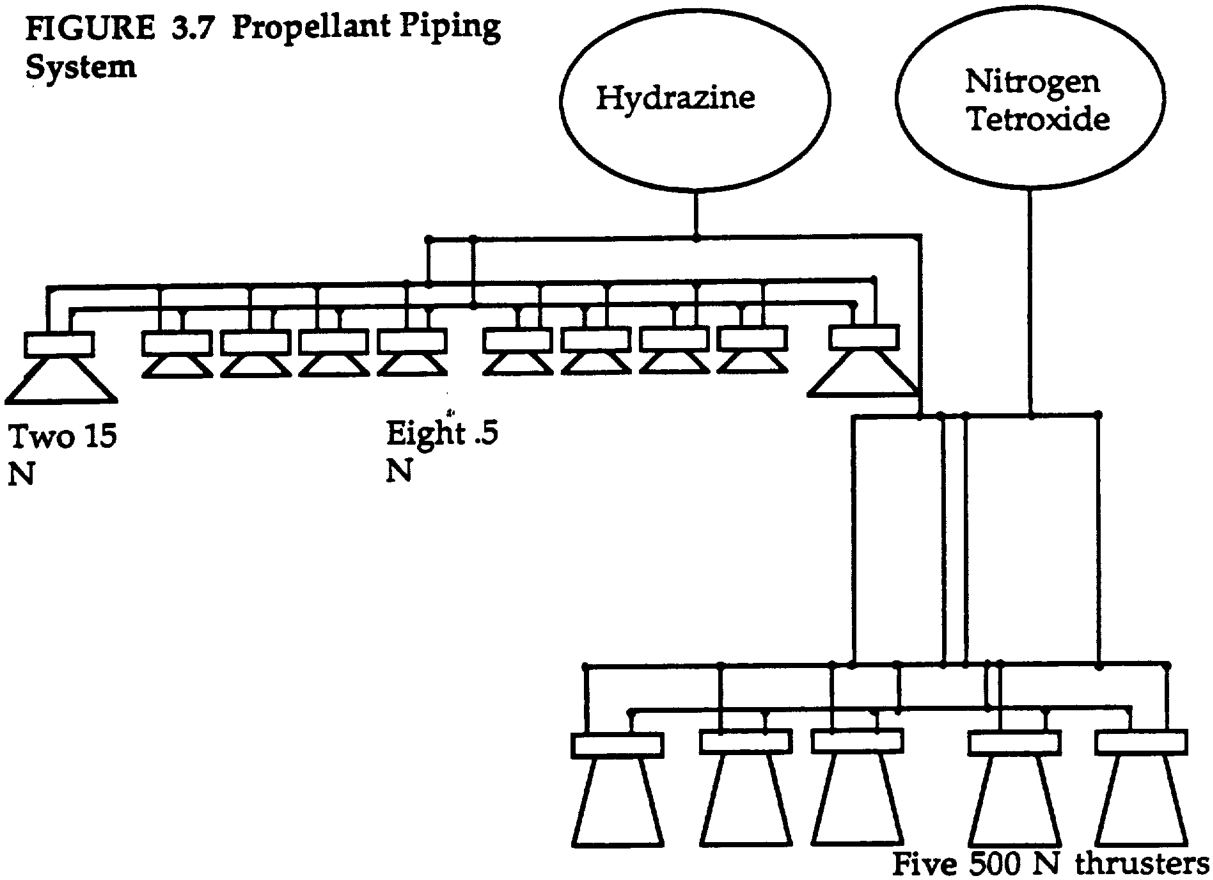


the proposed path and angle (E) that OPTIC will travel past periapse of its orbit about Pluto. Table 3.3 displays results for different engine configurations. Using the impulsive burn approximation it is assumed that the burn will take place at periapse. In this case, the burn will be spread to both sides of the periapse point with a total angle roughly 3 times E. Five engines will be used and will impart a load of only .16 g's on OPTIC. The 5 engine configuration leaves room for failure. Three to four days prior to the orbit insertion, the engines will be test-fired and if one or more fails, the firing system will be reprogrammed for the number of engines that work (see Command and Control sec.).

The sizing of the tanks had a limit of a 1 m diameter cylinder. The tanks are half-spheres connected by cylinders that are part of the outer cylinder (see figure 3.6). The half-spheres, along with a .15 m spacing between the two tanks, leave plenty of room for wiring, heaters and piping. A thickness of 1 cm was assumed and this thickness will guard against any micro-meteorites (10. Parcel, p 504). A redundant piping system will be used in case of any failures. The hydrazine tank will supply the attitude control thrusters with propellant. Figure 3.7 illustrates the redundancy and the different supply routes. Both the attitude control thrusters and the main engines will have 2 feeds. The final dimensions of the tank, excluding engines, is a cylinder 1 m in diameter and 4.68 m in length.

OPTIC's final configuration is set. OPTIC's propulsion system was only designed to handle the Δv at Pluto, TCMs throughout the mission and supply the attitude control thrusters with propellant. An expendable stage is needed to supply the Δv of 7871 m/s out of LEO in order to send OPTIC on the correct

FIGURE 3.7 Propellant Piping System



trajectory to Pluto. The stage that can provide this level of performance is the initial Space Transfer Vehicle (ISTV). The ISTV is a liquid oxygen/liquid hydrogen system that will be available by the late 1990's. It will carry a propellant load of 28182 kg and provide an I_{sp} of 482 s (2. Cramer, pp 1-3). The ISTV is 11.81 m long and has a diameter of 4.27 m. It will be designed for compatibility with the current Space Shuttle or the proposed Shuttle-C and will give OPTIC a Δv of up to 7993.35 m/s out of LEO (see Appendix 3.1) (2. Cramer, pp 1-3).

The total mass of the ISTV and OPTIC is 34544.937 kg with a total length of 19.39 m. These values are well within both the Shuttle-C's lift capability of 45359 kg and length of 24.7 m. Shuttle-C will be ready for launch capability in the mid-1990's and it will be the launch vehicle that will deliver OPTIC and the ISTV to LEO (7. Kolcum, pp 123-125).

3.3 CONCLUSION

OPTIC's mission criteria presented a number of problems to be addressed. Reliability was the main feature in selecting the various components from the RTGs to the LEROS engine. Even with the amount that reliability was stressed, there will still be some potential problem areas. One of these areas is the potential for the failure of components in the power subsystem. Twenty years is a very long time to not have an electrical component failure. The redundant, parallel circuit will hopefully prevent a small failure from becoming catastrophic. Another problem that could be encountered is propellant feed once OPTIC is orbiting Pluto. There will only be a small amount of propellant floating in the tanks and settling that small amount to feed the attitude control thrusters will be difficult. This design aspect should be analyzed more if the design advances beyond the preliminary stage. The final mass budget for the power and propulsion subsystem is in Table 3.4. Refer to Table 3.2 for the various subsystems' mass budget. Overall, the power and propulsion subsystem should provide a reliable support for OPTIC's other systems and successfully execute an orbit insertion into Plutonian space.

TABLE 3.4
Final Mass Budget for
Power and Propulsion Subsystem

Component	Mass (kg)
Power regulator	4.213
Power control unit	6.24
Power distribution	4.86
RTGs	25.582
Tanks and pumps	271.361
Hydrazine	1751.473
Nitrogen Tetroxide	1369.178
ISTV stage	30870.0

APPENDIX 3.1
EQUATIONS AND CALCULATIONS

Calculation of bond numbers:

$$B_o = \frac{a r_t^2 \rho}{\sigma} \qquad r_t = .48\text{ m}$$

$$\text{for } N_2H_4: \quad \rho = 1008\frac{kg}{m^3}, \quad \sigma = .0915\frac{kg}{s^2}$$

$$B_o = 2538.18a \qquad \text{for } B_o > 1 \rightarrow a > 3.94\times10^{-4}$$

To calculate the minimum T required: $T > aM_{SC} = (3.94\times10^{-4})(3684.937) = 1.45$ N

$$\text{for } N_2O_4: \quad \rho = 1447\frac{kg}{m^3}, \quad \sigma = .0275\frac{kg}{s^2}$$

$$B_o = 12123.23a \qquad \text{for } B_o > 1 \rightarrow a > 8.25\times10^{-5}$$

Following same procedure, $T > .30$ N. $\therefore$ min. T required for settling = 1.45 N

Initial propellant mass iterations:

$$\text{impulsive burn approx.} \equiv \Delta v = g_o I_{SP} \times \ln\left(\frac{M_{SI}+M_{CC}+M_{AC}+M_{OS}+M_{PS}+M_{TP}+M_{EF}+M_P}{M_{SI}+M_{CC}+M_{AC}+M_{OS}+M_{PS}+M_{TP}+M_{EF}}\right)$$

where: $g_o = 9.81\frac{m}{s^2}$, $I_{SP} = 315$ s, $M_{SI} = 56.93$ kg, $M_{CC} = 80.0$ kg, $M_{OS} = 67.0$ kg, $\Delta v = 4927\frac{m}{s}$

$M_{AC} = 48.1$ kg (20 kg of N_2H_4 used before Pluto burn), $M_{PS} = 40.895$ kg

$$\text{assume structural efficiency} \equiv \varepsilon = .08 = \frac{M_{TP}}{M_{TP}+M_P} \quad \rightarrow \quad M_{TP} = \frac{M_P}{11.5}$$

$$\text{set } M_{EF} = .04M_P$$

solving for $M_P = 2292.244$ kg $\rightarrow$ $M_{EF} = 91.69$ kg, set $M_{EF} = 95.0$ kg

APPENDIX 3.1 (cont)

TCM determination:

want $\Delta v = 100\,\frac{m}{s}$ $\rightarrow$

use impulsive: $$\Delta v = g_o I_{SP} \times \ln\left(\frac{M_{P_{tot}}+M_{SI}+M_{CC}+M_{AC}+M_{OS}+M_{PS}+M_{TP}+M_{P_{TCM}}}{M_{P_{tot}}+M_{SI}+M_{CC}+M_{AC}+M_{OS}+M_{PS}+M_{TP}}\right)$$

where $M_{P_{tot}}$ = Mp(from above) + 95 + 20(attitude control) kg

$$M_{TP} = \frac{M_{P_{tot}}+M_{P_{TCM}}}{11.5}$$

solve for $M_{P_{TCM}} = 95.968$ kg. add this to find new Mp at Pluto

new $M_{P@Pluto} = 2377.251$ kg

repeat above steps up to $\Delta v = 400\,\frac{m}{s}$ using previous Mp's

Final values: $M_{P_{TCM}} = 439.109$ kg, $M_{TP} = 271.361$ kg

$M_{P@Pluto} = 2566.542$ kg, $M_{EF} = 95.0$ kg

$$M_{SC} = M_{SI}+M_{CC}+M_{AC}+M_{OS}+M_{PS}+M_{TP}+M_{P_{TCM}}+M_{P@Pluto}+M_{EF} = 3684.937 \text{ kg}$$

Because of the circularity of the problem, the Δv was only a figure to base calculations on. Now that masses are final, calculate final TCM ability.

impulsive approx. $$\Delta v_{TCM} = g_o I_{SP} \times \ln\left(\frac{M_{SC}}{M_{SC} - M_{P_{TCM}}}\right) = 392.09\,\frac{m}{s}$$

Δv capability once around Pluto:

$M_{SC@Pluto} = M_{SC} - M_{P_{TCM}} - M_{P@Pluto}$ - 20 kg N_2H_4 (used for attitude control)

= 659.286 kg (includes another 20 kg N_2H_4 for attitude control)

M_P after burn $= M_{EF}$ $\rightarrow$ $$\Delta v = g_o I_P \times \ln\left(\frac{M_{SC@Pluto}}{M_{SC@Pluto} - M_{EF}}\right) = 480.82\,\frac{m}{s}$$

$\Delta v_{req.} = 47\,\frac{m}{s}$ $\therefore$ we have an extra Δv capability of $\approx 433\,\frac{m}{s}$

APPENIX 3.1 (cont)

Burn time at Pluto:

for 5 engines: $t_b = \frac{M_{P@Pluto}}{\#eng.(\dot{m})} = \frac{2566.542}{5(.162)} = 3168.57$ s

acceleration(g's): $a = \frac{\Delta v}{t_b} = \frac{4927}{3168.57} = 1.55 \frac{m}{s^2} \rightarrow g's = \frac{a}{9.81} = .16$

angle traveled past periapse: $t_b = \sqrt{(\frac{a^3}{\mu})} \times (E - e\sin E)$

where $a = 9750$ km, $\mu = 663.5622 \frac{km^3}{s^2}$

solve for E using above values: $E = 5.28°$

total angle traveled during burn $\approx 3E = 15.84°$

repeat down to 1 engine, results in Table 3.4

Tank sizing:

Assumptions: cylindrical shell with diam. = 1 m, 1 cm thick, elongated spheres for tanks.

$r_t = .48$ m $\quad V_{sphere} = \frac{4}{3}\pi r_t^3 = .463\ m^3$

vol. $N_2O_4 = \frac{M_{N_2O_4}}{\rho_{N_2O_4}} = \frac{1369.178}{1447} = .946\ m^3 \rightarrow$ vol. $N_2O_4 - V_{sphere} = V_{cyl.} = .483 = \pi r_t^2 h$

solve for h $\rightarrow h_{N_2O_4} = .67$ m

vol. $N_2H_4 = \frac{M_{N_2H_4}}{\rho_{N_2H_4}} = 1.738\ m^3$, same calc. as above: $h_{N_2H_4} = 1.76$ m

refer to Figure 3.6

Δv out of LEO:

impulsive approx. $\Delta v = g_o I_{SP} \times \ln(\frac{M_{ISTV}+M_P+M_{SC}}{M_{ISTV}+M_{SC}})$

where: $M_{ISTV} = 2688$ kg, $M_P = 28182$ kg, $M_{SC} = 3684.937$ kg, $I_{SP} = 482$ s

$\Delta v = 7993.35 \frac{m}{s}$

$\Delta v_{possible} > \Delta v_{needed}$ by $\approx 122 \frac{m}{s}$

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4.0 ATTITUDE AND ARTICULATION CONTROL

INTRODUCTION

The primary objectives of the AACS are discussed below. They include the stabilization of flight during all phases of the mission, orientation of the craft for communications, trajectory control maneuvers (TCMs), data collection, and determination of the relative position of the spacecraft.

This section was researched and prepared by David Mark Robinson.

4.1 SYSTEM OBJECTIVES

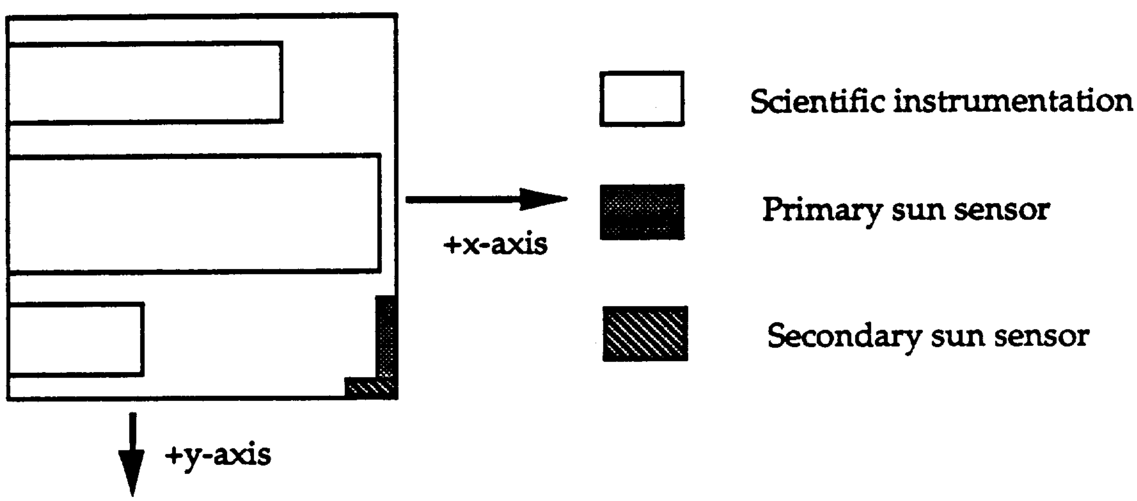
As stated in the request for proposal, the objective of this project is to develop a conceptual design for a spacecraft to study Plutonian space. The optimization of performance, weight and cost are very important. Our choice of an orbiting spacecraft has made the optimization of weight substantially more demanding with every kilogram of additional dry mass adding 8.4 kg of fuel. The attitude and articulation control system (AACS) designed for OPTIC is believed to provide versatility and reliability at a relatively low weight.

4.2 ATTITUDE DETERMINATION

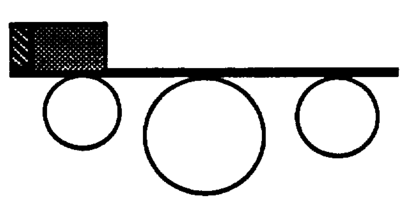
The attitude of this spacecraft will be determined with the use of a fixed head star tracker, a two-axis sun sensor/horizon sensor, two secondary sun sensors, and a state of the art fiber optic rotation sensor. This combination provides reliability, simplicity, and again relatively low weight.

The primary sun sensor incorporated for use of this craft is a two axis mask sun detector providing attitude determination about two axis and horizon sensing capability for the Plutonian orbit. A sensor of this type has low power requirements and its light weight makes it an excellent choice for this mission. This device will operate as a sun sensor in the cruise mode and when near enough to Pluto it will switch to Plutonian acquired cruise. When OPTIC begins to orbit Pluto this device will operate as a horizon sensor. The primary sun sensor and one of the secondary sensors will be located on the scan platform with the primary sensor pointing the same direction as the cameras and the secondary on an adjacent side (figure 4.1).

Figure 4.1 Sun sensor location on scan platform



View of science platform from x-axis

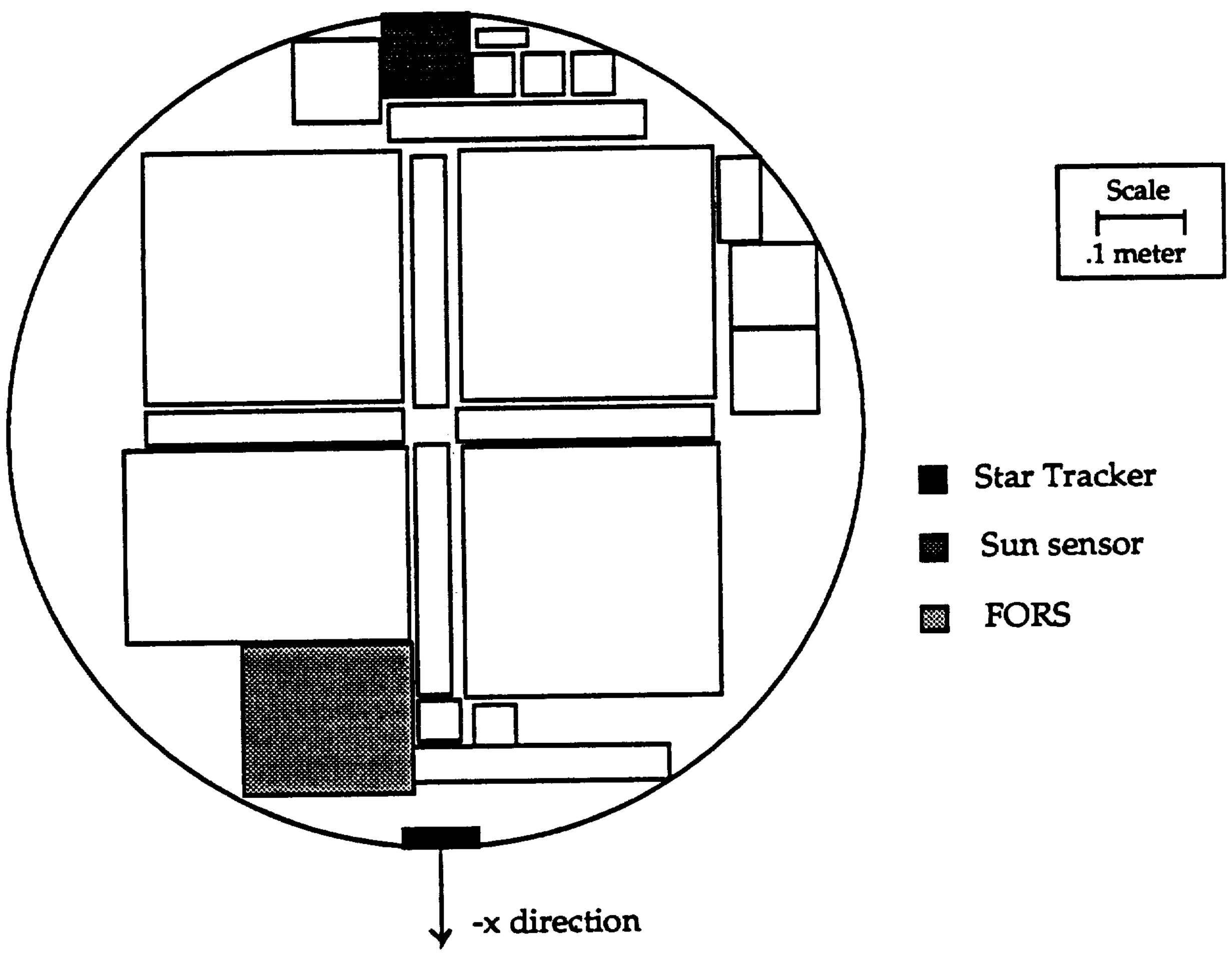


The additional sun sensor will be located on the electronics bus opposite the boom of the scan platform (figure 4.2). The secondary sun sensors will primarily be used during sun acquisition and as a backup for the primary sun sensor. Mass estimates for these sun sensors are based on sensor found in NASA documentation (4. Giampeoli).

A fiber optic rotation sensing (FORS) system has been chosen for inertial reference (3. Draper, p.14). This technology is not yet available but is expected to be by the mission dead line of 1999. The characteristics of the FORS that make it advantageous are many. First of all the system provides three axis rotation sensing with full redundancy which is very important in an attitude determination system. Its residual drift rate, power requirement, and weight are low relative to present inertial reference hardware. This system also has an extended lifetime expectancy due to its lack of moving parts and the fact that it contains no short term wear-our

FIGURE 4.2

Location of AACS equipment on electronics bus.



In order to conserve energy during the long cruise to Pluto the FORS will be turned off when the craft is in cruise mode. It will be reactivated, however, before any type of maneuver or if sun acquisition is lost for any reason. During data collection and transmission at Pluto the FORS will be activated to ensure pointing accuracy. With the FORS, pointing accuracies as low as 0.1 degrees are realistic (5. Hansen, p.111). This satisfies the requirements for antenna pointing during data transmission to Earth. The FORS will be located on the electronics bus as near to the roll axis as possible (figure 4.2). Table 4.1 list some of the specifications of the FORS system.

Fiber Optic Rotation Sensor (FORS)

Residual Drift Rate deg/hr	Rate Noise deg/sec	Angular resolution arc-sec	Availability yrs	Power watts	Mass kg	Volume in^3
2E-4	1E-5	0.005	10	<10	10	1000

TABLE 4.1
(Draper, p.14)

The star tracker chosen for our spacecraft, the Canopus/Tracker, was chosen primarily because of its proven performance, light weight (4.3kg), and low power requirements (4.5 w). It will be located on the electronics bus where it will have a relatively unobstructed view of space (figure 4.2). This tracker is significantly lighter than the newer advanced star trackers (ASTROS), 23.7 kg lighter, and with a fuel addition of approximately 8.4 kg for every 1 kg of dry mass this is a savings of about 200 kg of fuel. The CBS tube used by this tracker is no longer made so slight design changes may be necessary if one can not be located. It is felt that the favorable attributes of this particular device warrant the modifications.

As further backup for the attitude control system the camera chosen for this mission by the science officer can image star fields for navigation.

MODES OF OPERATION

All data gathered by the attitude determination hardware will be channelled to the spacecrafts command computer where it will be processed and the appropriate steps will be executed depending on the mode of operation currently in effect. There will be several modes of operation which are discussed below.

After OPTIC has been released from the bay of Shuttle C an autonomous control system will stabilize and orient the entire system for the initial delta-V. The booms and primary antenna will not be deployed until after this maneuver considering the excessive g-loading involved. After separation from the launch vehicle, sun acquisition will follow utilizing the three sun sensors. The FORS will be enabled to ensure that rotation rates are not excessive for boom deployment. Once OPTIC has been stabilized, equipment will be deployed and sun-acquired cruise will follow.

Since the secondary antenna is powerful enough for communication with Earth until OPTIC reaches Jupiter the spacecraft will travel in sun acquired cruise with the main dish pointing in the direction of flight until after the necessary TCMs are made on the approach to Jupiter. After the Jupiter flyby it will be necessary to turn the spacecraft 180 degrees and point the larger antenna back to Earth for communication purposes. For any TCMs thereafter the craft will have to be rotated back and forth to insure communication with Earth for the majority of the flight. In the sun-acquired cruise mode the FORS will be turned off to conserve power leaving the responsibility of attitude control to the sun sensors and the star tracker. FORS will only be used in the cruise mode when TCMs and for a short period every 24 hours to monitor the roll rate. No rotation about the pitch or yaw axes will be allowed but the craft will be allowed the roll within deadband constraints. The deadband roll rate will be 5 rev/min or 0.5236 rad/sec. This is to assure that the craft does not spin out of control on its roll axis. If FORS senses a greater rotation rate than the deadband the roll axis thrusters will be activated to reduce the roll rate. To reduce the role rate from 0.5236 rad/sec to a full stop when the craft is fully loaded with fuel it will take a steady burn for 2655.2 sec of the four thrusters opposing the rotation (see Appendix 4.1). If anything is encountered during the cruise mode that warrants investigation the FORS will be turned on for enhanced pointing accuracy and control. When the data has been taken and transmitted back to Earth the cruise mode will resume (FORS off).

The scan mode designed for this spacecraft will be initiated on the approach to any entity that the craft has been instructed to observe or that is determined to worthy of investigation. The system will confirm approach parameters and make necessary corrections in the trajectory. When within scanning distance the system will respond to preprogrammed commands and/or to stimuli imposed by unforeseen phenomena. Upon approach to Pluto the scan mode will measure approach parameters and any necessary TCMs will be executed. Once of the correct trajectory the spacecraft will be tuned 180 degrees for its deceleration burn (FORS on). At Pluto the scan mode will include orbit station keeping, stability, and orientation requirements imposed by scientific instrumentation and communications. Specifically, this mode will include the mapping of the planet which includes the collection of the data during closest approach to Pluto and data transmission through the apogee of the orbit. Data transmission imposes the most demanding pointing requirements and this is where the FORS will be very effective.

The magnetometer experiment also imposes requirements on the attitude control system. A rotation rate of 3.15 rev/min or 0.32987 rad/sec about the roll axis is desirable for the most accurate data collection (see Appendix 4.1). To avoid possible problems of data transmission to Earth while the craft is spinning, all magnetometer data will be collected and then OPTIC will be despun before data transmission commences. The magnetometer experiment will be executed during the last 3x10 orbit and the first 3x5 orbit. Despinning the craft assures pointing accuracy.

THREE AXIS HYDRAZINE JET CONTROL SYSTEM

The final decision to use a gas jet system resulted primarily from the choice of an orbiter mission since this significantly increased the size and weight of the spacecraft. The amount of fuel required for this mission has made it necessary to virtually send up a drum of fuel with the science instrumentation, power supply and additional equipment on booms with the exception of the electronics bus. Much consideration was involved in the final choice of a hydrazine jet system. Also, the longevity of this mission puts the restraint of bearing lifetime on some hardware. Hydrazine jets provide the torques necessary to maneuver a craft the size of OPTIC with more agility than momentum wheels or control moment gyros and utilization of an already present fuel source helped to reduce the weight of the system significantly. There has been concern in the past over the affects of the exhaust produced by hydrazine thrusters on scientific instrumentation but it was not found to present a problem (7. Wertz, p.208).

The configuration of three-axis hydrazine jet attitude control system for the proposed spacecraft is believed to provide ample maneuverability and stability with sufficient redundancy. The hydrazine jet control system consists of eight 0.5N jets for roll axis maneuvers and two 15N jets for pitch and yaw maneuvers. Thruster locations are shown in figure 4.3. The four roll thrusters not visible in the diagram are symetrically located at the opposite end of the main body of the craft near the electronics bus (Figure 4.2). With the fuel source of the primary propulsion system being hydrazine and nitrogen tetroxide, a monopropellant control system using hydrazine seemed to be a good opportunity to save weight since this eliminates the need for additional fuel tanks if the primary hydrazine fuel source can be utilized. This also adds a margin of safety considering that if the control system fuel budget is exceeded there should be more than enough in the primary systems fuel budget to compensate. The concern of fuel supply is addressed in a later section. The fuel budget of 40kg is based on estimates using a similar system described by Wertz. (7. Wertz, p.209)

As mentioned earlier, the 0.5N jets will provide roll torques to the space craft. The torque will be produced by four gas jets at a lever arm of 0.5 m providing a 2 N-m of torque. The system designed provides complete redundancy allowing up to four of the eight thrusters to be disabled and retaining control of the craft. The 15N jets are mounted at the rear of the craft, the line between them being parallel to the y-axis. They will be used for applying pitch and yaw torques(figure 4.3). By orienting these jets such that the line between them is parallel to the yaw axis and firing one jet a pitch torque is produced. In this position firing the top jet will produce a negative torque or firing the bottom jet will produce a positive torque. Yaw maneuvers are accomplished using a similar procedure aligning the line between the two jets parallel to the pitch axis (figure 4.3).

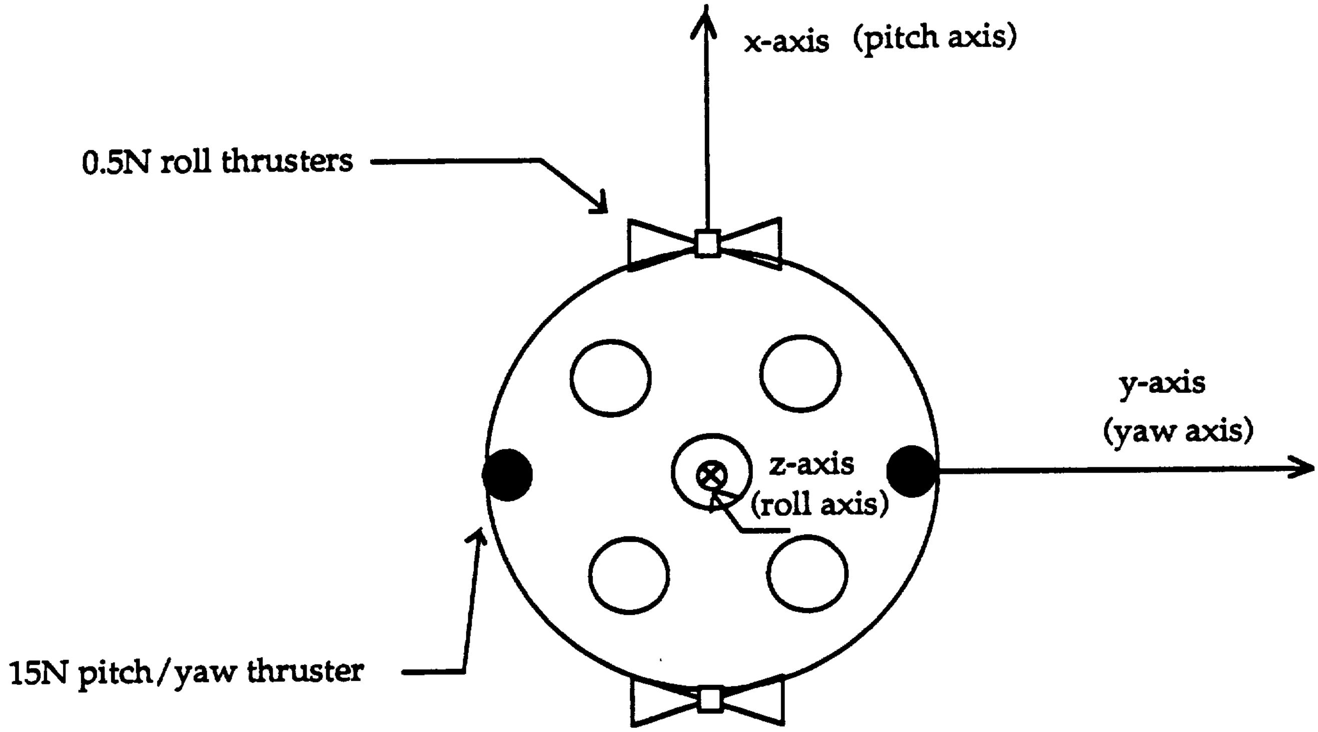
The two 15N jets will serve the additional role on this mission of settling the fuels in their respective tanks before major burns of the primary propulsion system. To assure fuel supply to the 15N thrusters a 1kg capacity nitrogen pressurized hydrazine fuel source has been provided for each jet. Once the settling burn has been completed these jets will then resume feeding off of the primary hydrazine fuel source.

The pulse duration of the burns will range form 0.1 seconds to up to several minutes depending on the desired maneuver. Pulse burns a more efficient however the longer the burn (6. Sutton). Optimal pulse durations are a trade between fuel consumption and the require speed at which the maneuver must be executed. This optimization is beyond the scope of this proposal. Mass estimates are based on similar hardware described in a NASA document (4. Giampeoli)

For trajectory control during burns of the primary propulsion system a thrust vectoring device will be utilized. This should facilitate excellent control during long duration burns and take the burden of some trajectory control maneuvers off of the smaller hydrazine jet system, hopefully resulting in lower fuel consumption.

FIGURE 4.3

Location of attitude control thrusters



4.2 ARTICULATION CONTROL

For scan platform articulation control a micro step actuator will be used (3. Draper, p.14). This is a new device which incorporates momentum compensation technology and will give increased pointing accuracies for the scientific instrumentation. It will also be used during sun acquisition to give the sun sensors on the scan platform a greater field of view.

The secondary antenna also requires articulation control. An actuator with three axis control is needed to give it the pointing freedom required for its optimal use. The more freedom this antenna has to move the less the spacecraft is required to move for it.

The masses of these devices were not located, but estimates were made using NASA documentation (4. Giampeoli).

POSSIBLE PROBLEMS

Without the use of a bladder or a pressurization system for the hydrazine fuel tank, fuel supply could pose a problem for the attitude control system. The addition of the nitrogen pressurized fuel sources for the 15N thrusters assures them of fuel, however, the 0.5N roll jets are not equipped with these pressurized tanks. This was primarily to reduce weight. A remedy for this problem could be a short duration pulse, approximately 0.1 sec, of the 15N jets supplied by the pressurized tanks to settle the fuel in the main tank. After this pulse all thrusters would then resume feeding off of the primary hydrazine source. Also, to remedy this problem a one way valve has been used as the feed valve for all propulsion systems. This locks fuel in the lines for assured short duration burns.

Another consideration that was overlooked until lat in the design was the need for multiple sun sensors and their placement. Originally only one primary and one secondary sensor were thought necessary. However, further research revealed the possible necessity of more sun sensors to assure sun acquisition. The final configuration decided upon consists of two sensors on the scan platform that will be able to scan space with the sweeping motion of the platform and a third located on the opposite side of the spacecraft in the electronics bus out of the field of view of the others (figure 4.2). This configuration may be inconsistent with other information in this proposal but it is the recommended configuration.

As can be noted in Appendix 4.2 the roll axis moment of inertia (Izz) changes from the smallest to the largest of the three axial inertias sometime during the deceleration burn at Pluto. It would be likely that sometime during the burn that Izz will be the intermediate value rendering it the unstable axis. Thrust vectoring and the control system should retain stability but this is still an area of concern.

4.4 AACS MASS SUMMARY

COMPONENT	MASS (kg)
Fuel Budget	42
Roll Thrusters (4 @ 0.5 kg each)	4
Pitch/Yaw Thrusters (2 @ 1 kg each)	2
Pressurized Fuel Tanks (2 @ 0.1 kg each)	0.2
Star Tracker (Canopus/Tracker)	4.3
Sun Sensors	~ 0.2 (Possible underestimation)
FORS	10
Thrust Vectoring Assembly	3
Secondary Antenna Actuator	3.1
Scan Platform Actuator	1.5

Total: 70.3 kg

APPENDIX 4.1

Calculation of pulse duration for specific roll axis maneuvers.

Torque = Force x Lever arm = Moment of inertia x Angular acceleration

$$\tau = FL = I\alpha$$

$$\alpha = FL/I$$

integrate once $d\alpha/dt = \omega = FLt/I + \text{const.}$ (const. $= \omega_o$)

integrate again $d\omega/dt = \text{theta} = FLt^2/2I + \omega_o t + \text{Const.}$

Assume all constants to be zero for these calculations.

F = Sum of forces of all thrusters acting (N)

L = Lever arm (m)

I = Moment of Inertia (kg m^2)

ω = angular velocity (rad/sec)

α = angular acceleration (rad/sec^2)

For worst case of deceleration from deadband (5 rev/min) to full stop with full tanks.

$$t = \omega I/FL = (.5236 \text{ rad/sec})(5071.05 \text{ kg m}^2)/4(0.5\text{N})(0.5\text{m})$$

$$t = 2655.202 \text{ sec} = 44.25 \text{ min.}$$

For spin requirement of the magnetometer experiment of 3.15 rev/min

$$t = (0.329867 \text{ rad/sec})(4378.506 \text{ kg m}^2)/4(0.5\text{N})(0.5\text{m})$$

$$t = 1444.325 \text{ sec} = 24.072 \text{ min.}$$

Pitch and yaw burn duration for 180 degree maneuver

Before deceleration burn: $t = (2\pi I/FL)^{0.5} = ((2\pi)(9859.6)/(15\text{N})(0.5\text{m}))^{0.5}$

$t = 90.88 \text{ sec} = 1.515 \text{ min}$

After deceleration burn: $t = ((2\pi)(3835.062)/(15\text{N})(0.5\text{m}))^{0.5}$

$t = 56.682 \text{ sec} = 0.9447 \text{ min.}$

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5.0 STRUCTURES

INTRODUCTION

This section is a discussion of the overall OPTIC structure, the materials used, and the final configuration of the spacecraft.

This data presented in this subsection was written and investigated by Mark James Endre.

5.1 STRUCTURES SUBSYSTEM OVERVIEW

As outlined in the Request for Proposal (RFP), once the OPTIC mission was decided upon, each subsystem must meet certain requirements. Since the launch date is between the year 2000 and 2010, the spacecraft must use materials and techniques available prior to the year 2000. These materials and subsystem components should be "off the shelf" whenever available, be reliable, easy to operate, and be relatively inexpensive.

An important factor in the design of the spacecraft configuration was the fact that there were two semi-conflicting requirements in the RFP. They were that once the orbiter was decided upon, nothing in the design could preclude the spacecraft from performing several different missions.

The main concern of the structure subsystem was to make sure that the craft could survive the launch sequence, the space environment, and be able to complete the outlined mission. Of the three requirements, the last was the most difficult. Since the mission type was an orbiter, several limitations were placed upon the craft itself. The most important of these was the fact that all subsystem masses had to be minimized. The design also required further optimization to meet the low mass requirements.

5.2 STRUCTURAL CONSIDERATIONS

There are many different structural considerations for any interplanetary mission. It was decided that for any mission in Plutonian space, an orbiter , if feasible, would be the most cost effective. However, the decision of using an orbiter created many technical hurdles that had to be overcome to prove feasibility. The main problem facing the structure subsystem was the reduction of the spacecraft mass.

MASS REDUCTION

The problem of reducing the mass as much as possible was attacked in various ways. Three prominent methods of reducing mass were considered. They were: the use of new extremely strong materials; using necessary components in a dual purpose capacity; and optimizing design techniques.

The advances in material sciences tends to suggest that the newly improved materials technique would be used. More specifically the use of a "smart" material that knows when to change properties could be very useful (4. Of Material, pg. 22). These smart materials are materials with an electrorheological fluid embedded in them. These fluids change viscosity almost instantly when a low-amperage, high voltage current is passed through them. These materials could be used to increase the strength of other materials and be used as a low weight, active dampening system. Although smart magnesium and aluminum metals have been fabricated, most research has been dealing with composites. Unfortunately, these materials along with the other advanced materials, have not had rigorous long term testing that would prove their space readiness. Since these new materials have not been space proven, it is possible that they could fail over the course of the mission. Consequently it was decided that this option conflicted with the RFP requirements of reliability.

The choice of a structural material was based on material properties such as strength to density ratios, operating temperatures, and properties of the material in a space environment. Aluminum is inexpensive, does not suffer large radiation effects, does not sublime, and would be able to be used far below temperatures at which it deteriorates in vacuum (5. Parcel, pg. 498, 2. Ashby, pg. 14). Since aluminum has been repeatedly proven to be spaceworthy, it was chosen as the material to be used in the electronics bus, the science platform, and in the micrometeorite shield.

STRUCTURAL LAYOUT

The fact that the mission type was an orbiter created a necessity for a large amount of fuel, and subsequently a large fuel tank. To reduce the overall spacecraft mass the fuel tank was used as the main structural component. This option not only reduces mass, but makes the spacecraft simpler and structurally more reliable.

The four extendable booms of the spacecraft are directly attached to the top end of the fuel tank. These booms, once deployed, will take on the configuration illustrated in Figure 5.1.

The main high gain antenna will also connect directly to the fuel tank. The mounting will pass directly through the electronics bus coincidentally with the axis of revolution of the fuel tank, hereafter the Z axis, as an Figure 5.2.

FIGURE 5.1: Top View of the OPTIC Spacecraft

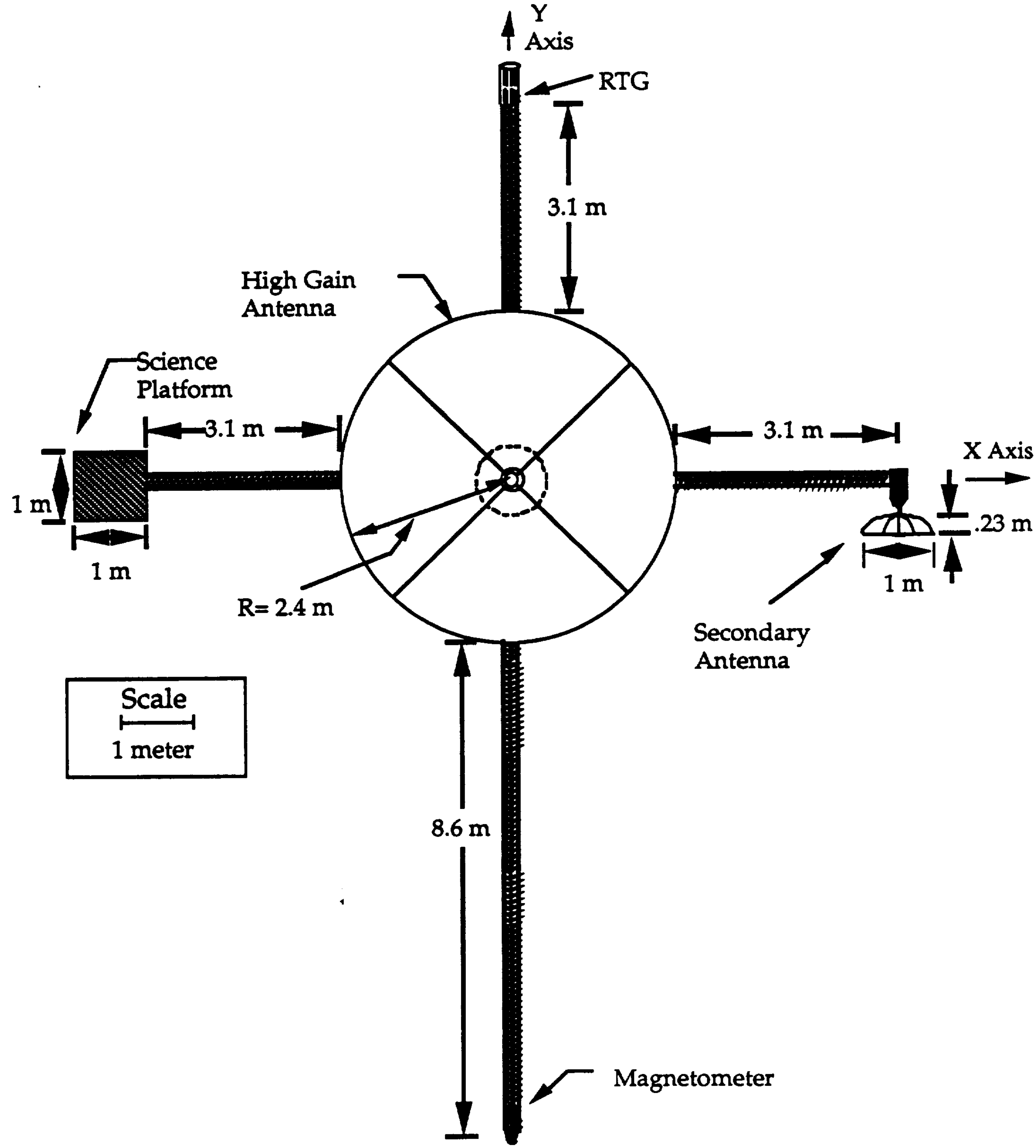
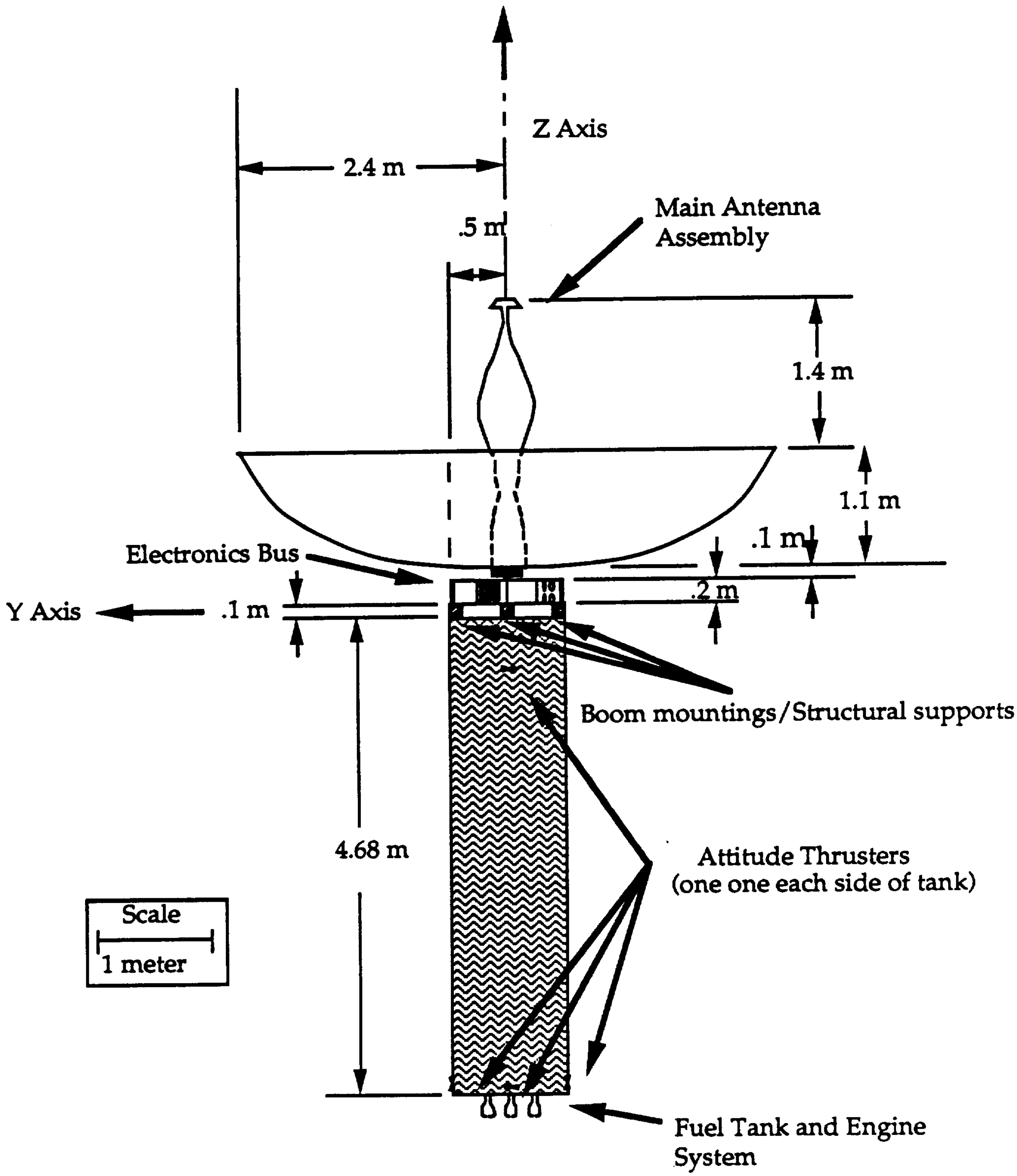


FIGURE 5.2: Side View of the OPTIC Spacecraft



Also centered around the antenna mounting is the electronics bus. This component is stationed on top of the extendable boom mountings along with other additional structural supports, and several vibrational dampeners. These dampeners are mounted on top of the structural supports and protect the electronics from vibrational damage.

Since the power source for the mission emits radiation, it must be stationed far enough away from the rest of the instruments so that it will not interfere with their performance. Using the Galileo probe as a guide, the RTG was mounted on a five meter extendable boom. Once extended this boom was designated as the positive Y direction in the spacecraft internal coordinate system. See Figure 5.1.

The magnetometer is also highly sensitive to the emissions of other various components on the spacecraft, particularly the RTG. For this reason the magnetometer is also mounted on an extendable boom, again mimicking the Galileo probe. The magnetometer is mounted on a ten meter boom orientated 180 degrees away from the RTG boom.

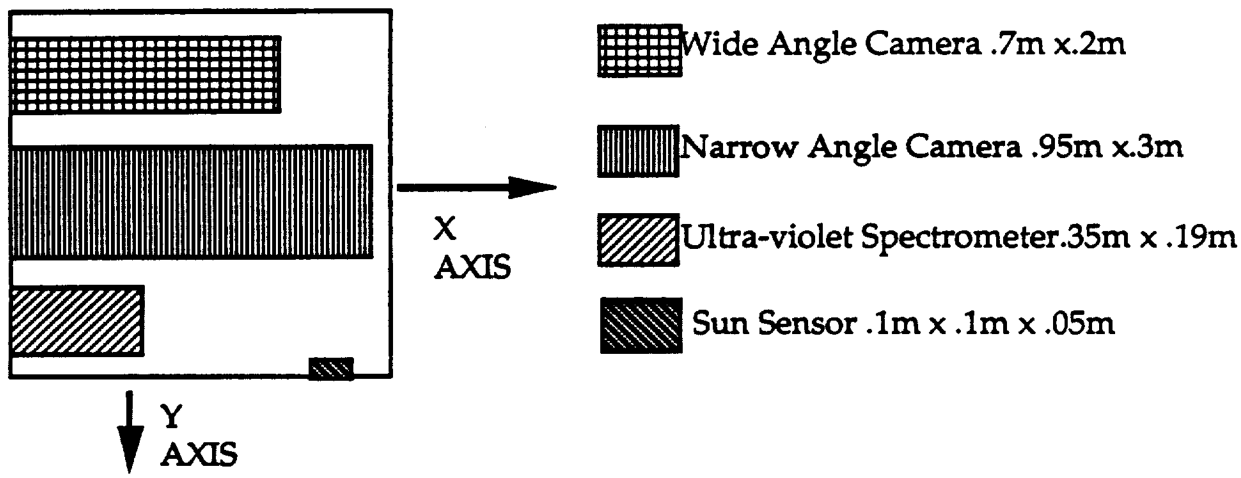
This left the science platform and secondary antenna mounted on five meter booms that are perpendicular to the Y axis. The direction toward the secondary antenna was arbitrarily designated as the positive X direction. See Figure 5.1.

The science platform itself has four instruments mounted on it. The only requirement of these instruments is that they have unimpeded fields of view. All four components are mounted on the underside of a meter square Al 2024 plate. The three optical scanners are mounted such that they are flush with the negative X edge of the plate. In this configuration they point away from the spacecraft when the science platform is in the neutral position, as depicted in Figure 5.1. The last instrument is a sun sensor. It is mounted flush with the positive Y side of the platform, as pictured in Figure 5.3.

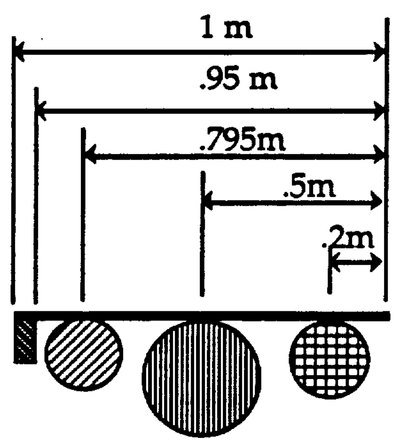
The science platform itself is mounted with actuators that allow it to rotate independently of the spacecraft. This reduces the amount of spacecraft pointing, and therefor reduces the amount of required fuel and total mass of the mission.

FIGURE 5.3 The Science Platform

Science Platform Viewed from Below



Science Platform Viewed from the Side



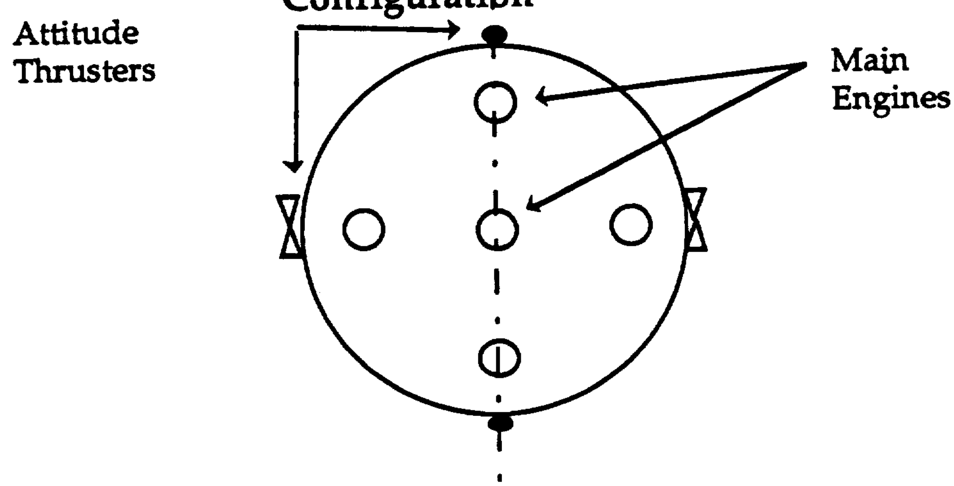
The secondary antenna is also mounted on actuators that allow independent pointing. This function will be used in the mapping portion of the mission.

The fact that these two independently movable systems are on opposite ends of the spacecraft allows a reduction of spacecraft maneuvering, both alone and in conjunction. If for instance when one system is not being used, that system can move in such a way that it can negate or greatly reduce the torques applied to the spacecraft by the system in use.

In the event of a failure of the pointing system, the pointing of the instruments can be obtained by spacecraft orientation. If only one system fails, spacecraft orientation will be used for pointing that system, while the operational system will point itself. These pointing functions would be controlled by the on board artificial intelligence.

The five main engines of the spacecraft are placed as in Figure 5.4. The placement of the attitude control thrusters are also depicted in Figures 5.2 and 5.4.

FIGURE 5.4: Engine Configuration



MICROMETEORITE PROTECTION

An important factor of the space environment is the presence of micrometeorites. Micrometeorite damage over the course of the mission could conceivably cause enough damage to a spacecraft to stop its operation. To prevent this damage to the sensitive electronics controlling the spacecraft and communications, a three stage defense was designed.

The first stage defense is the electronics bus itself. The electronics of the spacecraft are mounted inside the electronics bus, and the bus protects the equipment from the space environment. From Figure 5.5 an electronics bus with a 2.2 m^2 area on a 25 year mission has approximately a 95% chance of not being punctured by a meteoroid if it is 2 mm thick (5. Parcel, pg. 503).

FIGURE 5.5

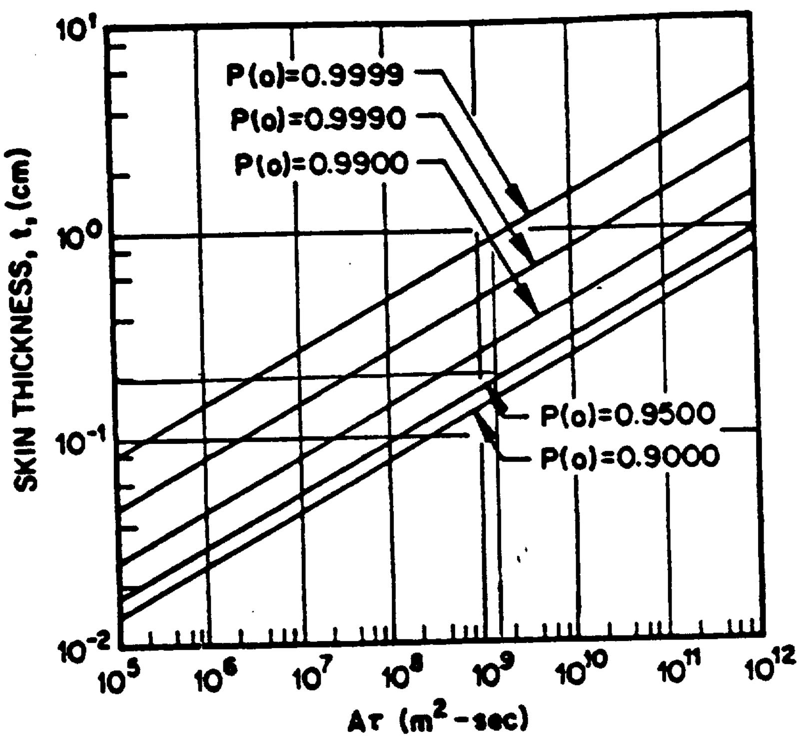


FIGURE 5.5: Sheet Thickness of Al as a Function of the Surface Area-Lifetime Product Required for Various Probabilities of No Meteoroid Puncture (Ref. 5, p. 503).

The second stage is a buffer system, or more commonly a micrometeorite shield. This shield is 1 mm thick and for maximum effectiveness is mounted 1.2 mm outside of the electronics bus. This shield covers the side and top of the electronics bus (5. Parcel, pg. 505).

The last stage of defense is a thermal, electrostatic and micrometeorite protection blanket similar to the one used in the Galileo probe (3. JPL). This blanket is inside the electronics bus and covers the electronic components.

THERMAL CONTROL

There were two available options for the thermal control of the spacecraft. The first option, passive control, requires a complete knowledge of the environment. Although passive control is generally a lower mass solution, the knowledge of the environment was not complete enough for this option to be used. For this reason active control is used for thermal control of the spacecraft (6. Vajta, p 99). A full thermal analysis however is beyond the scope of this course. Therefore thermal control is regulated with six single watt heaters in the electronics bus and four single watt heaters in the fuel tank (7. Yeates, pg. 112).

ELECTRONICS PLACEMENT

The placement of the electronics components in the electronics bus is based on the size of the component , heater placement, and an attempt to balance the mass of the science platform. See Figure 5.7 for an illustration of the electronics bus layout.

RADIATION EFFECTS

The planned trajectory at first raised a possibility of radiation contamination. However, since the gravity assist maneuver is a great distance from Jupiter, radiation exposure and contamination of the spacecraft is not a problem. Therefore no additional radiation protection is necessary (1. Andrew, pg. 52).

FIGURE 5.7 The Electronics Bus and Electronics Layout

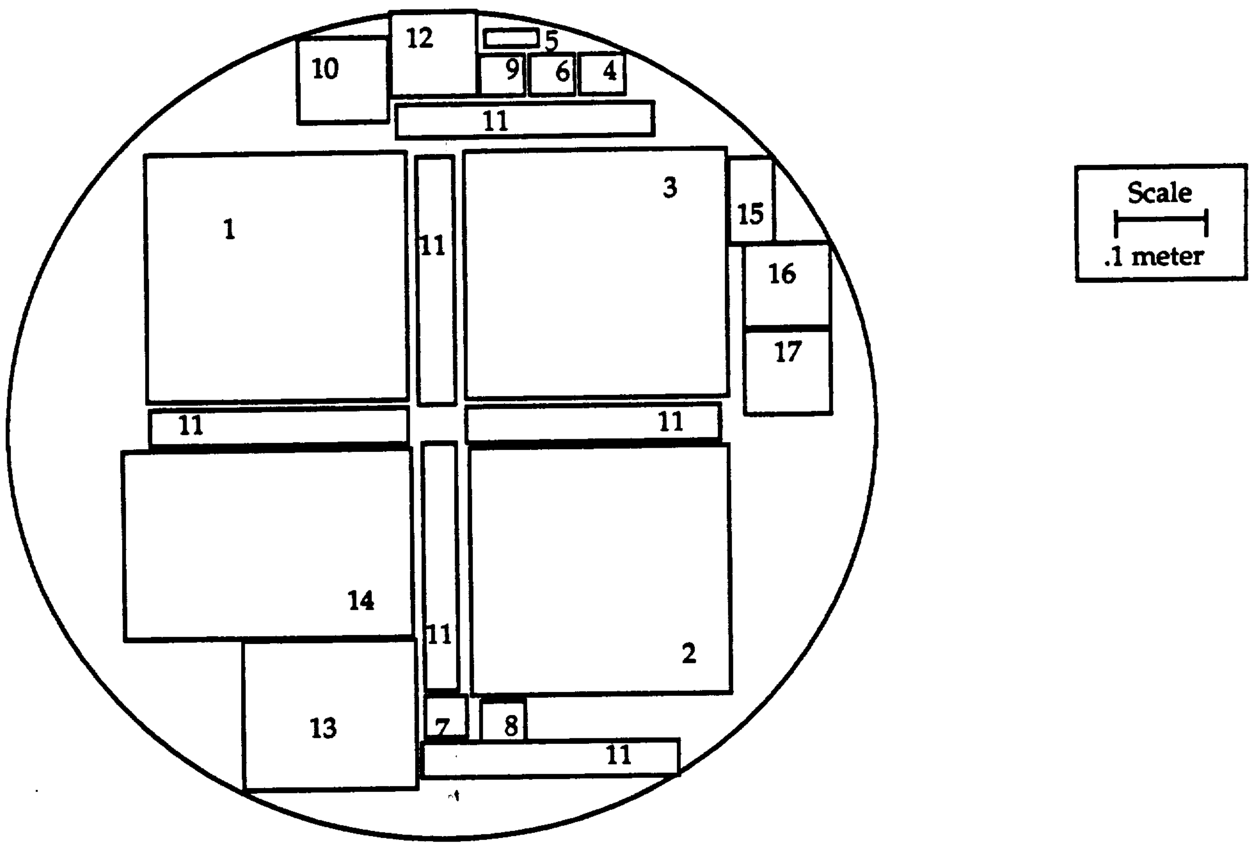


TABLE 5.3: Electronics Compononent Masses

Unit #	Component Name	Mass (Kg)
1	Computer	6
2	Computer	4
3	Reciever System	20
4	Down Converter	2.4
5	Convulution Coder	.6
6	Main Telemetry Modulation Unit	2.4
7	MTMU Backup	2.4
8	Command Detector Unit	2.5
9	CDU Backup	2.4
10	TWTA Subsystem Assembly	5.9
11	Heaters	6 @ 1
12	Star Tracker	4.3
13	Inertial Reference System	10
14	Radar Imaging Electronics	9.7
15	Power Regulator	4.21
16	Power Control	6.24
17	Power Distribution Unit	4.86

MASS AND INERTIA CONFIGURATION

Using masses, physical locations, generalizations, idealizations, and dimensions of the other subsystem components, an approximate center of mass and approximate inertia matrix were calculated for the spacecraft. The idealizations and generalizations used were that components had uniform densities and conformed to simple symmetric shapes. For this preliminary design these methods should have resulted in values reasonably close to the actual values. Structure subsystem components and masses are listed in Table 5.1, while all subsystem and component masses are in Table 5.2. The spacecraft inertia matrix at the start of the mission and upon arrival at Pluto are listed in Figure 5.6. These are based on a coordinate system at the center of mass with the same orientation of the coordinate system in Figure 5.1 and 5.2.

TABLE 5.1: Structure Subsystem Masses

Component Name	Mass (kg)
Magnetometer Boom	8.81
Science Platform Boom	4.4
Secondary Antenna Boom	4.4
RTG Boom	2.8
Electronics Bus	12.17
Micrometeorite Shield	3.87
Protection Blanket	1.55
Science Platform	10
Main Antenna Mounting	5
Heaters	10
Vibration Dampeners	2
Structural Supports	2
Total Mass	67

FIGURE 5.6: Inertias

Inertia Matrix With Full Fuel Load

$$\begin{bmatrix} 9859.6 & 41.35 & 607.82 \\ 41.35 & 10268.34 & -122.72 \\ 607.82 & -122.72 & 5071.06 \end{bmatrix}$$

Inertia Matrix at Pluto Arrival

$$\begin{bmatrix} 3835.06 & 21.90 & 397.61 \\ 21.90 & 4146.59 & -82.22 \\ 397.61 & -82.22 & 4378.50 \end{bmatrix}$$

APPENDIX 5.1: Equations

Skin Thickness Equation:
(Ref. 5, pp. 501-502)

$$t = \frac{1.94\,\alpha\,(A\tau)^{\frac{1}{3\beta}}}{(-\ln p(o))^{\frac{1}{3\beta}}}$$

t = vehicle skin thickness (cm)
A = exposed surface area (m^2)
τ = exposure time (sec)
$\alpha = 3.3 \times 10^{-15}$
$\beta = 1.34$
$p(o)$ = probability of no punctures

Shield Spacing Equation:

$$s = 2\,t_t \left(\frac{t_t}{t_b\;\rho_b} \right)^{\frac{1}{2}}$$

t_t = thickness of main plate (bus) (cm)

t_b = thickness of shield (cm)

ρ_b = density of shield (gm/cm^3)

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6.0 COMMAND, CONTROL AND COMMUNICATION

INTRODUCTION

This section describes and explains the command, control, and communication of OPTIC. This subsystem involves data management, transmission, and onboard computing.

This section was compiled and written by Eric W. Summers.

6.1 SYSTEM OBJECTIVES

Command, control, and communication is responsible for the well being and good management of the spacecraft. It is also responsible for the proper management of all data generated by the spacecraft. This data can be science data, engineering data, or imaging data.

Science data comes from measurements made by the science instrumentation. Engineering data is data about the spacecraft's health. Imaging data is pictures or radar images of the planetary surface. All of these types of data are called telemetry. This is the most important type of spacecraft communication. There are two other types of communication: command and tracking. Command is communication from Earth that tells the spacecraft what to do. It is generally a rather high quality signal so that the spacecraft doesn't get an incorrect instruction. There are numerous safeguards, such as parity checks and checkwords, to insure that this doesn't happen. Tracking yields information about spacecraft velocity, position, and the interstellar medium. If the signal happens to pass through an atmosphere then atmospheric composition can also be determined.

The command, control, and communication subsystem must aid and assist the mission at hand, which is to reach Pluto and to learn the most that it can;

however, there is also a set of guidelines which must be followed and, whenever possible, adhered to.

6.2 RFP REQUIREMENTS

The RFP lists several general requirements that are applicable to the command, control, and communication (C^3) subsystem. These requirements are as follows. There should be nothing that will limit the spacecraft to only a Pluto mission. The spacecraft must last, at least, to complete the mission. All materials and techniques must be available or be projected to be available by 1999. The spacecraft must be as simple, reliable, practical, original, feasible, and as low cost as possible. Off the shelf hardware should be utilized when possible. Components should be optimized for weight and cost. Significant intermediate events, potential problem areas, and concerns should be indentified. The only requirement that is specifically directed at C^3 is that the latest advances in AI (Artificial Intelligence) should be used to decrease spacecraft cost and increase spacecraft autonomy. This requirement is somewhat at odds with the simplicity, low cost, and off the shelf hardware requirements. A compromise will have to be reached.

6.3 COMMUNICATIONS

The communication system is broken down into four basic parts: antennas, frequencies, telemetry, and receiving stations. All of these parts are interdependent and require optimization.

There will be two antennas onboard the spacecraft: a main high gain antenna (HGA) and a secondary low gain antenna (LGA). Both antennas are of the new collapsible mesh design. This type of antenna will be used on both the Galileo and

Cassini missions; therefore, the design should be adequately field tested. The HGA will be the one used on the Galileo mission. This was chosen because it is the largest of the new mesh antennas to be designed, and it is off the shelf hardware. This antenna is 4.8m in diameter. The main difference is that Ka band will used as opposed to X or S band. This is a relatively small adjustment. The feed cone will need to be replaced to make this antenna Ka capable. The antenna surface is optimized for X band; however, there is only a small decrease in efficiency when Ka band is used.

The LGA will be used for spacecraft to Earth communication en route to Jupiter. This is done to facilitate communications without having to rotate the spacecraft 180°. This antenna will use Ka band for communicating with Earth; however, once Pluto orbit is achieved this antenna will be used to do radar mapping of Pluto's surface. Radar mapping will require X and S bands. This means that the LGA will have to be X, S, and Ka band capable. Cassini uses all three bands so this shouldn't present any problems. The LGA will be 1.0m in diameter and three axis gimballed so that it can be pointed in any direction in its field of view.

Ka band is the frequency of choice for a Pluto mission. The higher the frequency the lower the transmitted power required for communication with Earth. Ka is a higher frequency than X; therefore, less power is needed. The mass of a Ka system is comparable, or will be by 1999, to that of an X system. Ka is currently unproven, since Cassini is going to use Ka for communications, it will be proven in the near future.

Ka is the newest of the frequencies to be allotted for deep space communications, which is why it hasn't been proven yet. Ka band is 32 GHz for spacecraft to Earth (downlink) and 34.5 GHz for Earth to spacecraft (uplink). X band is 7.161 GHz for uplink and 8.414 GHz for downlink. Ka was chosen because it was the last good window in the microwave spectrum. There is some concern about the effects of atmospheric attenuation of a Ka signal due to water in the atmosphere. However, there is only an 8% loss due to the atmosphere in the worst case average. This is not a large loss. The advantage of Ka over X band far outweighs a small atmospheric loss.

The last frequency to examine is an optical one. Optics promises a great deal: higher bit rate, lower power, and smaller size. Currently, optics is not feasible within the 1999 development deadline. Optical communications requires pointing 1000 times more accurately than the best X band pointing accuracies and the development of an entirely new receiving network (9. Metscher, p.110). Such a network is not realistic before 2010 and may have to be in orbit as opposed to being ground based (12. Smith p.98). Considering the demands and uncertainty of an optical communications system (8. Layland, p.123), optics is currently an unacceptable option for the Pluto mission.

Telemetry is the most important part of the communication system. If no data is returned then the mission is a failure. There are many things to consider when working telemetry in smoothly with the other systems: bit rate, redundancy, coding, data compression, when to send, how much to send, and in what order should the data be sent.

The bit rate is based on the amount of data there is, transmission time and computer memory available, background noise temperature, bandwidth, signal to noise ratio (SNR), and the redundancy. The science subsystem sends data to the computer for storage, coding, and then transmission. If the amount of data received is greater than the amount transmitted then memory space will start to fill up. This is not a problem until the memory reaches capacity and then data will be lost. The amount of power available, antenna size, and the background noise is fixed. Therefore the bandwidth and SNR must be manipulated to provide an adequate bit rate. The amount of data that is sent can be compressed. This means that a smaller bit rate can be used. The amount of time that communication with Earth is possible is dictated by science needs and the orbit. The HGA must be pointed towards the Earth; however, when science is taking measurements the spacecraft must be pointed to where science wants to look. Transmission is only possible when science does not require any particular orientation, and the spacecraft is in the right place in its orbit. The redundancy also requires memory space. If transmitted data is stored until confirmation from Earth is received, then a large part of the memory could be taken up by this data. The best way around this is to have a large SNR, so that most of the data arrives at Earth, and a bit rate larger than that of science input. The order that the data is sent will conform to current NASA standards.

It is evident that all of the factors are dependant on each other. This means that optimization in an iterative process. Fortunately, most of the factors are bounded or in some way fixed. The SNR must be greater than 10 for good communication, transmission power has an upper bound, the maximum Ka bandwidth is .5 GHz, and the DSN receiver is 70m. This means that the bit rate is within a narrow range, so that optimization is easier.

The receiver system will be the Deep Space Network (DSN). There are three major DSN sights around the globe: Goldstone California, Madrid Spain, and Canaberra Australia. Because these sights are spread as they are there is no time that the spacecraft cannot communicate due to the rotation of the Earth. Each of these sights has a 70m and a 34m dish. The 70m dish is preferable for a Pluto probe because of its larger size. DSN will be modified for Ka band by 1995 (5. Imbriale, p.127). This modification will make the DSN more efficient, in Ka band, over a wider range of elevation angles.

6.4 COMMAND AND CONTROL

Command and control is comprised of three basic areas: command hardware and software, the computer, and interactions with other subsystems. The command hardware consists of a command detector unit and a convolution coder. Currently these are pieces of equipment separate from the computer; however, in the future they may become part of the software. The heart and soul of the command and control system is the computer. A computer is basically a central processing unit (CPU) and memory. There will be four CPUs that will be used, 3 for processing and 1 for a backup. In case of multiple failures the computer could operate in a reduced mode on only 1 CPU. The CPU that will be used is a 32 bit microprocessor being developed by the Department of Energy (DoE) (2. GAO, p.24). The use of a microprocessor that is currently being developed will save the program money by utilizing off the shelf hardware. A 32 bit processor is more powerful than any processor currently in use, or planned for use in space. This extra power will allow the spacecraft to use higher level programming languages than are currently in use in space. Current spacecraft are programmed in assembly language. Assembly is difficult to program in because it is a low level language. The advantage of a higher level language is two fold. It is cheaper and easier to program in. Low cost is an important point in the RFP. The relative ease of programming in a higher level language will also make the spacecraft more flexible. If the spacecraft should experience any long delays then it could be reprogrammed for newer technology and techniques. This would have done for Galileo had it been programmed in a higher level language (2. GAO, p.35).

There are two choices in the selection of a higher level programming language. The first choice is "C". C is widely used in industry and thus well known. The other choice is ADA. ADA is a government standard language and it has been proven in space by the Europeans. ADA also has a prioritizing function. This means that when something more important comes to its attention it will stop what it is doing and do that which is more important. These factors make ADA the language of choice. A higher level language is required for artificial intelligence (AI) and expert systems. However, higher level languages are not all pros and no cons. On the flipside, ADA will require more memory than assembly. Both ADA and AIs will require more memory than is currently in use on spacecraft.

A computer has two different types of memory: internal and external. There will be 8 megabytes (MB) of internal random access memory (RAM), a 660 MB hard drive, and a 256 MB external optical drive. The 8 MB of RAM is somewhat of a standard on commercial high end personal computers, the Macintosh™ IIcx by Apple Inc. is a prime example. Both the 660 MB hard drive and the 256 MB optical drive are new products of NeXT™ Inc. The selection of an optical drive over a more conventional magnetic tape drive is easy when it is realized that an optical system is much smaller than a magnetic system of comparable memory, and the fact that optical disks can be written to as many times as desired without any data dropouts or degradation. Magnetic drives cannot come close to this kind of performance. The size of computer memory is rapidly increasing while the cost is decreasing. This trend is what will allow the spacecraft to have an unprecedented amount of memory. This large amount of memory will allow the spacecraft to use the latest in AI technology, entirely backup its internal programming, and save all transmitted data until confirmation from Earth is received. Therefore if the internal memory is lost, the spacecraft can continue to operate from its other memory sources. If either of the external drives should be lost the other can also act as a backup. There should also be almost no data loss, as all data can be retransmitted until it is correctly received. This should make the computer system extremely reliable.

There will be heavy use of AI technology. The onboard AIs will monitor and control all of the spacecraft's functions. There will be an AI to monitor the health and status, make minor course adjustments, regulate and prioritize the onboard systems, and control the communications system. The AI will be able to be overridden from Earth if such an occasion should arise. This large degree of autonomy will be necessary once Plutonian orbit insertion is achieved. There will be a great deal to do and many new and unexpected situations will arise. Assistance from Earth will not be quick enough as the round trip signal lag time will be about 11 hours. This is why the use of AIs will be needed. The AI technology for this is not available to do all of these things at this time. Currently all of the above functions (health and status, course adjustments, etc...) can be done, but the AIs cannot yet deal with the unknown. This may not be necessary if enough data about the size of the Pluto/Charon system can be learned before orbital insertion. This information could be learned by the spacecraft as it gets close to the system or perhaps Hubble will find out the size of the system.

Command and control must also interact with the other onboard systems. Attitude and articulation control is going to run itself off of the command computer. This was done to cut down on mass. Science will send its data to the command computer. Structures decides when the booms should be deployed and the computer will do it. Should there ever be any sort of power shortage or equipment failure then the computer will have to prioritize and sequence accordingly, this would be an interaction with the power and propulsion subsystem. The command and control subsystem is responsible for the well being and good management of the spacecraft.

6.5 SPECIFICATIONS

The HGA will be 4.8m in diameter and transmit with 6.3 watts of power. The LGA will be 1.0m in diameter and transmit with 2.1 watts of power. There is a DC to RF power conversion factor of .21 (4. Hansen, p.111). This means that for the HGA to transmit with 6.3 watts of power there must be 30 watts of power coming in.

The NeXT™ optical drive uses a 5.25" disk suspended in a polycarbonate medium. The disk spins at 3000 to 3600 revolutions per minute (rpm). It requires 18 watts to read and 40 watts to write. There is a 92 millisecond (ms) search time with a 18 ms search time if the information is within a 3 MB range. The drive can read/write bursts at rate of 4.6 MB per second, or there is a sustained read/write of .8 MB per second. The drive must be kept between 10° and 30° Celsius.

Data will be sent at about 388 kilobits per second with a signal to noise ratio (SNR) of 25. A SNR greater than 10 is required for good communication. A SNR of 25 was chosen so that a minimal amount of data would be lost on its way to Earth. Data compression at 4:1 will be used (4. Hansen, p.113). The background noise is assumed to be 8°K. The signal losses due to the atmosphere are 8%, transmitter pointing losses are 11%, receiver losses are also 11%. For calculations and equations see appendix 6.A.

6.6 CONCLUSIONS

There are a few significant current developments which were not included in the design of OPTIC, but bear further investigation. The subreflector system aboard Cassini utilizes a frequency selective surface (FSS). This would allow an antenna to adjust itself to the desired frequency in order to optimize the transmission (5. Imbriale, p.128). There is also the possibility that a new DSN array will be built, this would give a greater effective area received; therefore, utilizing available power better. There are also the hope that a greater DC to RF efficiency can be achieved. This would also better utilize available power.

There is a slight problem in the communication bandwidth that must be resolved. The current bandwidth is about 20 kHz. Typical bandwidths are no less than 1% of the operating frequency, or 320,000 kHz. The 20 kHz cannot be made to be 320,000 kHz without decreasing the SNR, an unacceptable choice, or increasing transmitted power to 106,000 watts, also unacceptable not to mention impossible. Since any space mission would encounter similar problems, there must be a solution to this dilemma.