University of Illinois
at
Urbana-Champaign
Aeronautical and Astronautical Engineering

AAE 241
Aerospace Vehicle Design, Spacecraft Section
Final Project Reports

Volume I
Project Groups 1 through 4

May 1990

P-71

B 635159

Group 1
AAE Final Design Report
April 24th, 1990

PFERD Mission

Pluto Flyby Exploration/Research Design

Abstract

The PFERD mission will consist of a flyby spacecraft to the planet Pluto and its satellite, Charon. The mission lifetime is expected to be 18 years. The Titan IV with a Centaur upper stage will be utilized to launch the craft into the transfer orbit. Each subsystem of the craft was designed by a different individual and is presented in a seperate section of the report. The group did tradeoff studies to optimize all factors of design, including survivability, performance, cost, and weight. Problems encountered in the design were also presented.

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Introduction

The PFERD mission will be one of immense scientific interest. Since its discovery, not much knowledge has been gained about this far away planet. It is the purpose of this mission to change this.

Our mission has been dubbed PFERD, which stands for Pluto Flyby Exploration/Research Design. It will consist of a Pluto flyby spacecraft and all of the components needed to send it to Pluto.

Our proposal has been divided into six main subsystems. They are, in order of appearance in this paper, Scientific Instrumentation; Command, Communications, and Control; Attitude and Articulation Control; Power and Propulsion; Structures and Thermal Control; and finally, Mission Management and Costing.

SCIENTIFIC INSTRUMENTATION

by KEVIN L. SUTTON

The ultimate goal of this mission is the return of new scientific information about Pluto and its satellite Charon. The discovery of Pluto occurred in 1930 while Charon was not discovered until 1978. During the six decades following the discovery of Pluto, determining the characteristics of the planet has been a difficult endeavor. Although Pluto was at perihelion in 1989 (29.6 AU), Pluto's mean distance from the sun is 39.5 AU. Pluto's orbital period is 248 years. The physical parameters of the Pluto-Charon system have been derived from mutual event observations of the system from 1985 through 1988. Table SI-1 lists the values of the most extensive analysis of the mutual events to date.

TABLE SI-1: Pluto-Charon Physical Parameters (Binzel, 1989)

Semimajor Axis	19640	+/-	320	km
Eccentricity	0.0001	+/-	0.001	
Period	6.387245	+/-	0.000012	days
Pluto's Radius	1150	+/-	7.0	km
Charon's Radius	593	+/-	10.0	km
Mean Density	2.030	+/-	0.035	gm/cm^3

The individual densities of Pluto and Charon cannot be determined because the mutual event observations cannot predict individual densities. There are many other uncertainties about Pluto and Charon, including the following questions:

- Does methane frost cover the surface of Pluto ?
- What composition and structure does the atmosphere have ? What is the haze layer composed of ?
- What is the composition and structure of the bodies ?
- Are there color variations over the surface of Pluto ?
- What is the origin of Charon ?
- Is an atmosphere refreezing to the surface of Pluto as it moves away from the sun and at what rate ?
- Are there any other satellites or rings ?
- What is covering Charon's surface ? Water frost ?
- What is the nature of the magnetic field and the interaction with the solar wind ?
- What is the population of the proposed Kuiper Comet Belt (30-50 AU from the sun)

SUBSYSTEM REQUIREMENTS

The Request For Proposal lists the general requirements of the overall spacecraft design. Some additional requirements for the scientific instrumentation subsystem include 1) describing and justifying the science objectives, 2) selecting and optimizing the instruments, and 3) determining the location, mass, power requirements, and data rate of the selected instruments. These requirements must be met while also stressing reliability, simplicity, and low cost. Performance must be optimized while minimizing the mass of the subsystem. Materials or techniques expected to be available after 1999 cannot be used.

SCIENCE OBJECTIVES

This mission to Pluto should answer all of the questions about the planet, in addition, many unprecedented discoveries should be made. The science objectives have been determined so that all of the true values for the many uncertainties will be revealed. The science objectives of the mission are:

1) Determine the composition and structure of Pluto's atmosphere
2) Determine the mass, composition, and structure of Pluto and Charon
3) Determine the dynamics of the Pluto-Charon system
4) Determine the color variation over the surface of the planet
5) Determine the nature of the magnetic field
6) Determine the origin of Charon
7) Study the impacts and impact rates to estimate the population and mass of the proposed Kuiper Comet Belt
8) Determine the interaction with the solar wind
9) Search for any satellites or rings

INSTRUMENT SELECTION

The instruments have been selected to accomplish the science objectives of the mission. The first step of the selection process was to examine existing or planned spacecraft to determine what off-the-shelf instruments were available to help minimize costs. The space vehicles researched include Voyagers 1 and 2, Galileo, Magellan, Pioneer 10 & 11, Giotto, Mars Observer, microspacecraft, and the Mariner Mark II program (CRAF and CASSINI). To meet the science objectives, the following instruments are desired:

Solid state imaging system (SSI) - take pictures to help investigate the surfaces and atmospheres of the two bodies, the magnetospheric interactions, the system dynamics, and conduct other visual searches.

Photopolarimeter (PPO) - determines the distribution and character of atmospheric particles (determines the nature of the haze layer).

Ultraviolet Spectrometer (UVS) - measures gases in the atmosphere to determine its composition and structure.

Infrared Spectrometer (IRS) - determines the composition and structure of the surface of the planet and satellite.

Magnetometer (MAG) - monitors the magnetic field for strength and changes.

Plasma Analyzer (PLA)- determines the interaction with the solar wind.

Radio Science (RSC) - determines the dynamics of the system, using the high gain antenna and the communications equipment.

Gamma Ray Spectrometer (GRS) - measures the elemental composition of the surface of the planet.

Laser Altimeter (LAT) - determines the global topography of the planet.

Once it became clear that this mission would involve the flyby of a spacecraft instead of an orbiting spacecraft, the laser altimeter and the gamma ray spectrometer were eliminated because they were designed for an orbiting spacecraft (Komro, 1989).

The choice of a specific imaging system involves many decisions. The Voyager imaging system has been proven to be reliable over long periods of time in space, although it uses outdated technology. The Galileo imaging system uses charged-coupled devices allowing for advanced solid state imaging. The imaging system designed for a microspacecraft is very light weight, but it does not give good resolution. Table SI-2 gives a comparison of the three imaging systems.

TABLE SI-2: Imaging System Comparison
(Flight 1987, Galileo 1985, Jones 1989)

	Voyager	Galileo	"micro"
Mass (kg)	30	28	0.8
Power (W)	29	17	3.8
Resolution	0.07 m/pixel @ 1000 km	0.07 m/pixel @ 1000 km	7.0 m/pixel @ 100 km

At first, the microspacecraft camera seems to be the best. It is very light weight and consumes much less power than the other two systems, but its resolution is much worse than the other two systems and it has not been proven in space. Due to these negative factors of the microspacecraft camera, it was not given any further consideration for use. Between the Galileo and Voyager imaging systems, the Galileo system represents the best choice, because it uses the latest imaging technology, has the least mass, and consumes the least amount of power, so it will be included on the Pluto probe.

To achieve reliability and low costs all of the instruments to be included on the probe are existing instruments from other spacecraft systems. Table SI-3 gives the mass, power requirements, data rate, and spacecraft of origin for each of the instruments to be included on the spacecraft.

TABLE SI-3 : Instrument Characteristics
(Flight 1987, Galileo 1985, Report 1985)

INSTRUMENT	MASS (kg)	DATA RATE (kbps)	POWER (W)	ORIGIN
SSI	28	115.2	17	GALILEO
PPO	5.0	10.0	10	VOYAGER
UVS	4.0	0.1-2.0	5.3	GALILEO
IRS	18	0.5-10.0	12	CRAF
MAG	3	0.01-0.4	4	CRAF
PLA	5	0.01-115.2	4	VOYAGER
RSC	---	---	---	GALILEO
TOTALS	63.0	144.6-271.6	52.3	

INSTRUMENT LOCATION

The instruments have specific requirements that specify where they can be located. Some of the instruments like the imaging system, need and unobstructed field of view. The high gain antenna is the main source of obstruction. To give a good field of view the instruments will be mounted on a high precision scan platform. This scan platform will be located on a boom and have two degrees of freedom so that it gives the instruments located on it an almost unobstructed field of view in any direction. The scan platform requires a pointing accuracy of 0.0034 rad and a slew rate of .00576 rad/sec to accommodate the instruments. The magnetometer needs to be located far away from the electronics bus because if the electronics bus generates a magnetic field, it will interfere with the magnetometer's sensors. To minimize this problem, the magnetometers will be located on an extendable boom of their own. The boom, when extended, is eleven meters long. One magnetometer sensor is located at the end of the boom while another is located is located approximately five meters from the end of the boom. The magnetometer electronics are located in the electronics bus in order to isolate the magnetometer sensors. The location of each of the instruments is given in Table SI-4. Figure SI-1 is a scale drawing of the high precision scan platform and the instruments that are located on it and Figure SI-2 is a scale drawing of the extendable boom and the magnetometer sensors.

TABLE SI-4 : Instrument Location

High Precision Scan Platform	Imaging System Ultra Violet Spectrometer Infrared Spectrometer Photopolarimeter Plasma Analyzer
Extendable Boom	Magnetometer Sensors

FIGURE SI-1 : High Precision Scan Platform

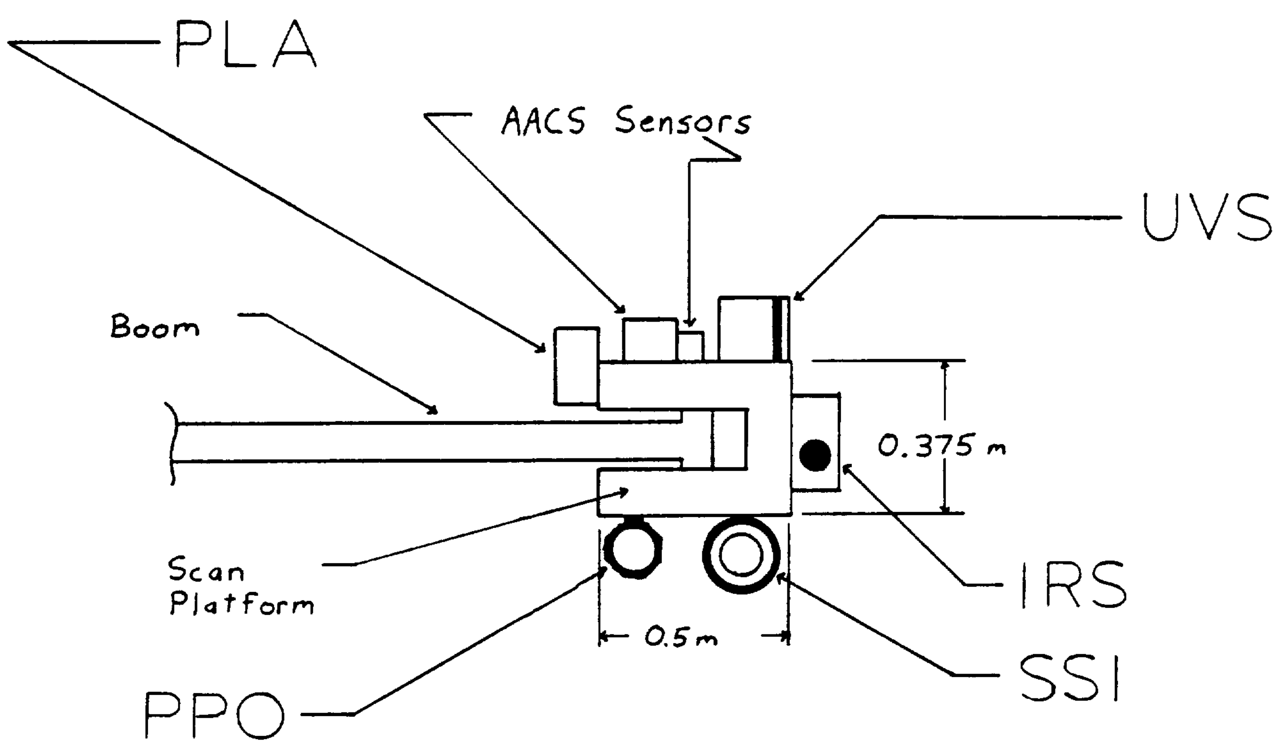
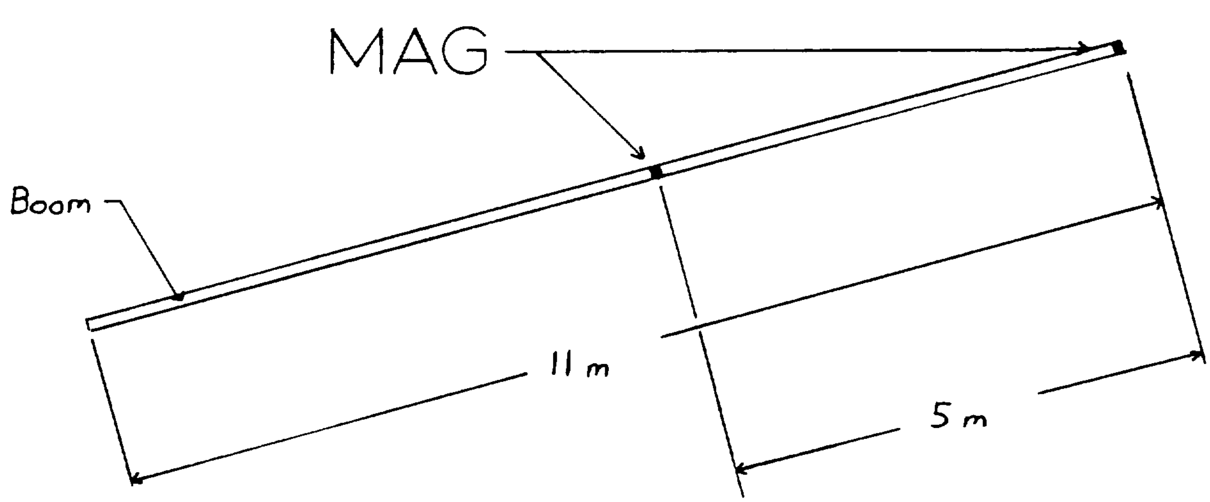


FIGURE SI-2 : Magnetometer Boom



CONCLUSION

The science instruments have been selected to maximize the scientific return for a flyby mission to Pluto, while also minimizing weight and cost. Off-the-shelf instruments have been incorporated into a scientific package that is simple, yet reliable. The instruments will meet or exceed all of the objectives of the mission.

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Command, Control and Communication

In order to determine the essential requirements for the Command, Control and Communication Subsystem, the Request For Proposal document must be examined. These requirements were found to be as follows: optimize performance, minimize weight and cost, use off-the-shelf and reliable hardware, materials and techniques must be developed before 1999, the design lifetime has to be sufficient for the mission with a safety margin, the spacecraft must communicate a distance of 38 A.U.'s and the subsystem can not conflict with the other subsystems. What separates this mission from all previous missions is the great distance the spacecraft must travel. The challenge presented to the C.C.C. Subsystem is the ability to communicate with Earth at this distance. Therefore this report focuses mainly on communication. The other aspects of C.C.C. are covered, but to a lessor extent due to the fact that they will be more standard and similar in design to previous missions.

The main part of communication is the choice of antenna to be used. In comparing the parabolic vs. the isotropic antenna, it can be shown that the parabolic antenna produces almost thirty thousand times the power received back at Earth of that produced by the isotropic antenna ((Yuen, p. 6) eqn.#1). This is because the isotropic antenna radiates in all directions while the parabolic antenna concentrates its waves in a cone configuration. Therefore a parabolic antenna is selected over an isotropic antenna.

Now that the parabolic antenna has been selected, "to achieve best possible performance, we must design the telecommunications system which gives the highest signal power, lowest amount of noise, and most efficient use of signal-to-noise ratio, within constraints such as spacecraft weight, size and cost (Yuen, p.3)." We want to optimize the power received back at Earth. Looking at the equation for the power received (Yuen, p.6), there are several ways to increase the power received. These are: increase the transmitting power, increase the diameter of the receiver, increase the diameter of the transmitter, decrease the wavelength used and decrease the transmitting distance. Decreasing the transmitting distance might entail putting some sort of transmitter half-way between Pluto and Earth. But this would be another mission in itself and is not considered an option. Next we can look at increasing the power transmitted, but this will be a set amount depending on how much power is available from the Power Subsystem. "Spacecraft-transmitted power is typically only 20 watts (Yuen, p.4)." There are a couple of ways to increase the diameter of the receiver. The most obvious being to make a larger and larger receiver. But this is too costly. A second method involves arraying already built receivers electronically to increase the effective area of reception. "This network is being upgraded to nine antennas: six 34-meter antennas and three 64-meter ones...the DSN 64-meter antennas will be enlarged to 70 meters (Posner, Horttor and Grant, p.62)." By arraying these antenna receivers, the power received will be increased by more than 5 times that of just one 64-meter receiver (eqn. #2). Arraying the receivers into a network is a good option. By doubling the diameter of the transmitter on the spacecraft, this will increase the power received by 4 times (eqn. #3). But the weight of the antenna will be doubled, and that does not sit well with the Mission Management Subsystem who is trying to minimize the weight. Also "the largest planetary spacecraft antenna yet is 4.8 meters (Posner and Stevens, p.20)," meaning a bigger antenna would not meet the R.F.P. requirements of off-the-shelf reliable material developed before 1999. Therefore increasing the transmitter diameter antenna greater than 4.8 meters is a bad option. Lastly, we can decrease the wavelength used in transmitting. To do this we must increase the frequency used. There are assigned frequencies used for space communications so that outside interference is minimized. The X-band (8.4 GHz) is now the standard down-link frequency used (Posner and Stevens, p.8). But by 1995, "the down-link could well be at 32 GHz (Posner, Horttor and Grant, p.62)." By using this Ka-band, it can increase the power received by almost 15 times of that using the X-band (eqn. #4). Therefore using the Ka-band (32 GHz) frequency is an excellent way to increase the power received.

Another method for transmitting data to Earth is through laser technology. In comparison with the 20 watts needed for the parabolic antenna, the laser only needs .5-2 watts of power and is only 10 cm. in diameter (Lesh and Rayman, p.81). This gives the laser a great weight reduction advantage. Also because the wavelength of a laser is only .5 micro-meters, this increases the efficiency of the signal at Earth one million fold compared to the parabolic antenna (Lesh, p.106). And the laser can provide all sorts of new "light sciences (Lesh and Rayman, p.84)." But the laser also has disadvantages at this time. The laser must be pointed with extreme accuracy. "With the long propagation time, you only have one shot at beam acquisition (Lesh, p.106)". Also little deep space testing with lasers has been done, which means its reliability is unknown. We do not know if the laser technology will be complete for deep space use by 1999. Therefore, laser technology is in direct conflict with the R.F.P. and can not be used.

After analyzing all the options for communication with Earth, we selected the parabolic antenna to be the best. The diameter will be 4.8 meters (the largest spacecraft antenna available) to optimize the power received. The network of the nine receiving antennas arrayed together will be used also to optimize the power received. The wavelength will be decreased by using the Ka-band (32 GHz) frequency to increase the power received. The 4.8 meter diameter parabolic antenna, along with the arrayed receivers and Ka-band frequency, will give the spacecraft the best possible power received at Earth while staying within the constraints of the R.F.P.

The spacecraft C.C.C. Subsystem must provide the Scientific Instrument Subsystem with a maximum data rate estimate so that the S.I. Subsystem can know what amount of data he will be able to send to Earth. The data rate is mainly dependent on the signal-to-noise ratio and the power received. Assuming a signal-to-noise ratio of 20, the power received can be calculated using the parabolic antenna and options chosen earlier (eqn. #5). The power received equals 1.593*E-16 watts. Therefore a data rate estimate for the S.I.

Subsystem equals 316891 bits/sec (eqn. #6). If this data rate is not large enough for the S.I. Subsystem, then storage considerations for the data are necessary. This can best be achieved through use of an optical storage disk. Another possibility is to compress the data being transmitted. "With Galileo, this can raise the partly compressed imaging output rate...400 times...Such data must be received with a very low bit error probability; because of the compression, a single error can destroy a large amount of data (Posner, Horttor and Grant, p.63)."

In order to communicate with Earth, the spacecraft antenna must be directed toward Earth. "The sun is still the primary attitude reference (J.P.L., p.19)," which is used to point the antenna toward Earth. The antenna will be mounted on the front of the spacecraft to avoid the delta-v burns used to go to Pluto. Therefore, the Structure Subsystem must provide a shield for the antenna to combat environmental and atmospheric hazards. After the initial delta-v burn, the Attitude and Articulation Subsystem must rotate the spacecraft 180 degrees so that the antenna is facing the Earth. If another delta-v burn is necessary, the spacecraft must first be rotated 180 degrees back into its proper position. Then after the burn is complete the spacecraft can be rotated 180 degrees once again to face the Earth. This process will need to be repeated as many times as the number of delta-v's necessary for the mission.

For the Command and Control part of C.C.C. Subsystem, we must look at the use of computers on-board the spacecraft. One problem with the distance that must be traveled for this mission is that it takes over five hours for a signal sent from Earth to reach Pluto (eqn. #7). In a time of crisis aboard the spacecraft, five hours may be too long of a time to wait for a command. Therefore, it is necessary that Artificial Intelligence be available to the spacecraft so that it can analyze a situation and make a decision on its own to correct the problem. As far as the type of computer system to be used, one similar to that used for Galileo or Voyager would be a good choice because of their proven reliability. The only problem with these computer systems is that they are ancient. They are very slow and their memory capabilities are limited. Therefore, we selected the advanced High Performance Micro Computer. This computer contains a 2 million Byte memory, uses 20 watts of power, and only weighs .1 kg (Jones, p.11). Also, the Command and Control Subsystem "can survive any single internal fault, because each of its functional units has a duplicate elsewhere in the subsystem (J.P.L., p.21)." Therefore with the Command and Control Subsystem completed, this concludes the design for the Command, Control and Communication Subsystem.

Appendix 1: Equations

#1 Parabolic vs. Isotropic Antenna

Parabolic Power Received (Pr)

Pr = Pt*Lt*Gt*Ls*Lr*Gr

Gt = .55*SQR(3.14159*Dt/Wavelength)

Ls = SQR(Wavelength/12.56*r)

Gr = .55*SQR(3.14159*Dr/Wavelength)

Isotropic Power Received (Pri)

Pri = .5*Ar*Pt/12.56*SQR(r)

Assume: Pt = 20 watts, Lt = Lr = .5, Dr = 64 meters

r = 38.5 A.U., Dt = 4.8 meters,

Wavelength = 3*E8(m/s)/8.4(GHz)

W = .0357 meters

Pr (para) = 2.082*E-18 W Pri(iso) = 7.716*E-23 W

Pr(para) = 26983*Pri(iso)

#2 Increase the diameter of Receiver

Dr = SqrRoot(6*SQR(34) +3*SQR(70)) = 147 meters

Pr = (SQR(147)/SQR(64))*Pr(original)

Pr = 5.28*Pro

#3 Increase the diameter of Transmitter

Pr = (SQR(2*4.8)/SQR(4.8))*Pro = 4*Pro

#4 Decrease the Wavelength

Wavelength = 3*E8/Frequency

Freq. = 32GHz

Pr = (SQR(32)/SQR(8.4))*Pro = 14.5 Pro

#5 Power Received for Parabolic Antenna with Options

Pr = Pt*Lt*Gt*Ls*Lr*Gr

Pt = 20 W

Lt = Lr = .5

Gt = .55*SQR(3.14159*4.8/(3*E8/32GHz))

Ls = SQR((3*E8/32GHz)/12.56*5.76*E12)

Gr = .55*SQR(3.14159*147/(3*E8/32GHz))

Pr = 1.593*E-16 W

#6 Data Rate (B)

B = w*log(SNR + 1)/log(2)

w = Pr/k*T*SNR

k = 1.38*E-23(J/K)

T = 8K

SNR = 20

B = 316891 (bits/sec)

#7 Time of Transmission to Pluto

Time = distance /Velocity

dist. = 5.76*E12 meters

vel. = 3*E8 meters/sec

Time = 5.33 Hours

Reference Page

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Attitude and Articulation Control Subsystem.

We can now examine the Attitude and Articulation Control System (AACS) for our spacecraft. The Request for Proposal (RFP) requires that the AACS design should 1) optimize weight, cost and performance, 2) be reliable and easy to operate, 3) use off-the-shelf hardware when possible, 4) be able to have a lifetime sufficient to carry out the mission plus a safety margin and 5) be able to perform several possible missions. The mission itself required that the system should guarantee communications with Earth, maintain the spacecraft's trajectory and be highly autonomous due to the mission length and the distance the spacecraft must travel away from Earth. These requirements served as a guideline in the design of our spacecraft's AACS. The primary AACS hardware selected consists of a star tracker, a gyroscope, a sun sensor, a computer and an assembly of thrusters for the attitude and trajectory correction maneuvers as well as the corresponding electronics and actuators to complete the system.

High Precision Scan Platform Sensors.

The star tracker selected for our mission is the Advanced Star and Target Tracking Optical Sensor (ASTROS II), and the gyroscope selected is the Fiber Optic Rotation Sensor (FORS). Both of these sensors were selected from the Mariner Mark II: Comet Rendezvous and Asteroid Flyby (CRAF) mission (Bell and Lehman). The ASTROS II was selected because it enables closed loop target tracking which allows for autonomous science data gathering (Bell and Lehman). It is also relatively lightweight, with a mass of 11 kg, and has a relatively low power requirement of 15 Watts (Bell and Lehman). The ASTROS II also provides very accurate star tracking to 20 arcsec for up to three stars simultaneously and allows for the autonomous calibration of the gyroscopes based on the star tracker data (Bell and Lehman). FORS provides low mass, solid state inertial angular rate and position sensing and is designed to meet or exceed NASA DRIRU II performance specifications (Bell and Lehman). These two instruments were also selected because they exceed the the requirements imposed by the Science Instrumentation subsystem. This subsystem required that the camera have a pointing accuracy of .0034 radians and that the scan platform have a slew rate of .00576 radians per second. The pointing requirement is met by ASTROS II which provides a target dependent accuracy from 1 to 10 arcsec and the slew rate requirement is met by FORS which provides a slew rate range from .00523 radians per second to .06981 radians per second (Bell and Lehman). The High Precision Scan Platform (HPSP) was selected for the placement of these sensors for several reasons. It provides an adequate separation distance from the contamination of the attitude thruster exhaust and from the radiation generated by the Radioactive Isotope Thermoelectric Generators (RTG). The HPSP was also selected because it minimizes any translational and rotational errors between the sensors and the science instruments (Bell and Lehman).

Bus Sensors and Hardware.

The inertial attitude as determined by ASTROS II and FORS is transferred to the basebody of the spacecraft to provide bus inertial rate, position knowledge for High Gain Antenna (HGA) pointing and thrust vector control (Bell and Lehman). This attitude determination is backed up by the a Fine Sun Sensor Assembly (FSSA) to provide redundancy. The FSSA was selected because it was used on the GRO satellite and is thus flight proven and reliable and also because it provides lightweight, low power redundancy with a mass of 1.75 kg and a power input of approximately 3.5 Watts (Wertz). It was also selected because it meets the pointing accuracy requirement imposed by the Communications subsystem of approximately .15 degrees to .50 degrees by providing an accuracy of .022 degrees (Jerkorsky, Keranen, Koehler, Tung and Ward).

An onboard computer (OBC) was needed to handle the autonomy required by the mission and the storage of science and communications data as well as the implementation of the attitude correction maneuvers (ACM) and trajectory correction maneuvers (TCM) determined by the sensors. To accomplish this task a high performance micro computer, developed by the Lawrence Livermore National Laboratory, will be placed in the bus of the spacecraft. This computer was selected because it is extremely lightweight, has relatively low power requirements and also because its storage capability of 2 million bytes is over 50 times more powerful than the computers used in the Galileo and Voyager missions (Koepke). A total of 2 computers will be used to provide full redundancy even though only one computer will operate at a given time.

AACS Propulsion System.

To perform all ACM's, TCM's and gravity assist maneuvers (GAM) required, thrusters were chosen over reaction wheels. Thrusters were selected because they 1) provide easier and quicker rotation of the spacecraft due to its large dry weight of 500 kg, 2) have been used on many other missions and are therefore reliable and 3) provide enough accuracy in combination with the attitude sensors for all science instrumentation. I studied several possible AACS propulsion systems that would handle the requirements imposed by the mission subsystems. A description of each is given below along with with the reasons for disqualification or acceptance.

I). 12, 10 Newton thrusters shielded and mounted in sets of 6 on booms protruding from opposite sides of the bus for all ACM's and TCM's, in combination with a 400 Newton engine used for all GAM's (Yates, Johnson, Colin, Fanale, Frank and Hunten). This system is identical to the system used on the Galileo spacecraft and is therefore reliable, but the problem is that the system is fueled by a bipropellant which will not last the duration of our mission of 18-22 years.

II). 12, 10 Newton hydrazine fueled thrusters mounted and positioned as described above in combination with a bipropellant fueled extra complete stage used for all GAM's. This option was disqualified because of its weight and high cost.

III). 24, 10 Newton thrusters in combination with a 400 Newton engine. This system will be divided up into 2 sets. The first set will contain 12 thrusters mounted and positioned as described above and fueled by hydrazine and the second set will consist of the other 12 thrusters and the 400 Newton engine which will be fueled by a bipropellant. The first set will be used for all ACM's and TCM's that are required after all GAM's are completed. In the second set the thrusters will be mounted on the 400 Newton engine. These along with the engine will perform all ACM's, TCM's and GAM's needed from the time of launch until the completion of the last GAM. Upon completion of the last GAM the engine and its thrusters will be jettisoned. This AACS propulsion system was chosen because it uses the same thrusters and engine that were used in the Galileo mission and because it uses bipropellant in the second set which has a better specific impulse than hydrazine. The total delta V needed to be generated by this system is approximately 3.3 km/s to 3.5 km/s and has been estimated from the requirements imposed by the Mission Management Planning and Costing subsystem (MMPC). The breakdown of the delta V is estimated as follows. A delta V of 1.7 km/s is needed for all GAM's to insure that the spacecraft will make it to Pluto. This was determined by the MMPC subsystem. The delta V required for all ACM's and TCM's is approximately 1.6 km/s and was calculated assuming that the spacecraft needed a delta V of .12 km/s every 1.5 years for 20 years.

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POWER AND PROPULSION SUBSYSTEM

POWER SYSTEM

The main requirement for the power source is that it must deliver enough reliable and uninterrupted power throughout the lifetime of the mission with a sufficient safety margin. The other requirements for the power subsystem are as follows. It must be designed for reliability, simplicity and low cost. It must be easy to operate and use off the shelf equipment where possible. All technology must be available on or before 1999. The power delivery system must protect the circuits, protect the load, and be able to control and distribute the power.

When selecting from the possible power sources, the most common space power source, solar, was eliminated from consideration because it would not be able to produce enough power at the distances that our mission would cover (over 40 AU). Batteries and fuel cells would not have a useful lifetime sufficient for our mission, so they were also eliminated from consideration. The two sources of power that could supply power at 40 AU and beyond for the duration of our mission are a space nuclear reactor and a Radioactive Isotope Thermoelectric Generator (RTG). RTG's were selected to provide the power for our mission because the have been proven to be reliable, safe, and easy to operate on several deep space missions. A space nuclear reactor was eliminated from consideration because there are no current space qualified reactors, and the only one that is currently being designed is designed to produce 100 mW, which is approximately 300 times larger than our required power of 305.5 Watts. The RTG's are also much safer from an environmental aspect, and they will require much less shielding to protect the scientific instrumentation. This analysis is summarized in table PP-1.

POSSIBLE POWER SOURCES

ADVANTAGES	RTG	NUCLEAR	SOLAR	BATTERY	FUEL CELLS
sufficient life	X	X	X		
operate at 40AU	X	X		X	X
fully developed technology	X	X	X	X	X
flown in space	X		X	X	X
proven reliable on long duration spacecraft	X		X	X (if recharged)	

Table PP-1. TRADE FOR POWER SOURCE

The type of RTG that we will utilize for our mission will be similar to the design proposed in a study by Fairchild (Schock). This design has many advantages over previous designs. This study was done to optimize the design of current RTG's by incorporating the latest developed power source, the newest materials, and utilizing a modular design so that it would be able to be used for many missions. The power system for our mission will be capable of 305.5 watts. The breakdown of the power requirements for the various subsystems is shown in table PP-2. The RTG thermal power source consists of 13 modular slices of the General Purpose Heat Source (GPHS). Each thermal power source slice delivers 250 Watts of thermal power, which will be converted into 23.5 Watts of electric power. The Modular Isotopic Thermoelectric Generator (MITG) design was selected over the two previously used RTG's these possible RTG designs are compared in Table PP-3 The Multi Hundred Watt system is the one that was flown on the voyager missions, so it has been proven in flight, but it does not take advantage of any of the recent improvements in RTG designs. The GPHS/RTG takes advantage of the new modular General Purpose Heat Source, but it does not utilize a fully modular design. It also does not make use of the new thermoelectric materials (SiGe+GaP instead of just SiGe). The only disadvantage to the MITG design is that it has not flown or even been produced, but developing a new RTG based on this design will more than double the power to weight ration of current RTG's (Schock, p342). This RTG is Shown in Figure PP-1.

POWER REQUIREMENT BREAKDOWN

Scientific Instrumentation	75 Watts
Attitude and Articulation	52 Watts
Mission Planning	0 Watts
Command, Control, and Communication	40 Watts
Propulsion	20 Watts
Structures	70 Watts
15 % for lifetime losses	40 Watts
Total power required	297 Watts
Total power delivered (nearest modular power)	305.5 Watts

Table PP-2. Power Requirements

TYPES OF RTG'S

ADVANTAGES	MITG	GPHS/RTG	MHW
apx. specific power	4.7 W/lb	2.3 W/lb	1.8 W/lb
able to provide required power	X	X	X
uses modular heat source	X	X	
fully modular design	X		
uses latest thermoelectric materials	X		
flown in space		X	X

Table PP-3. TRADE FOR RTG TYPE

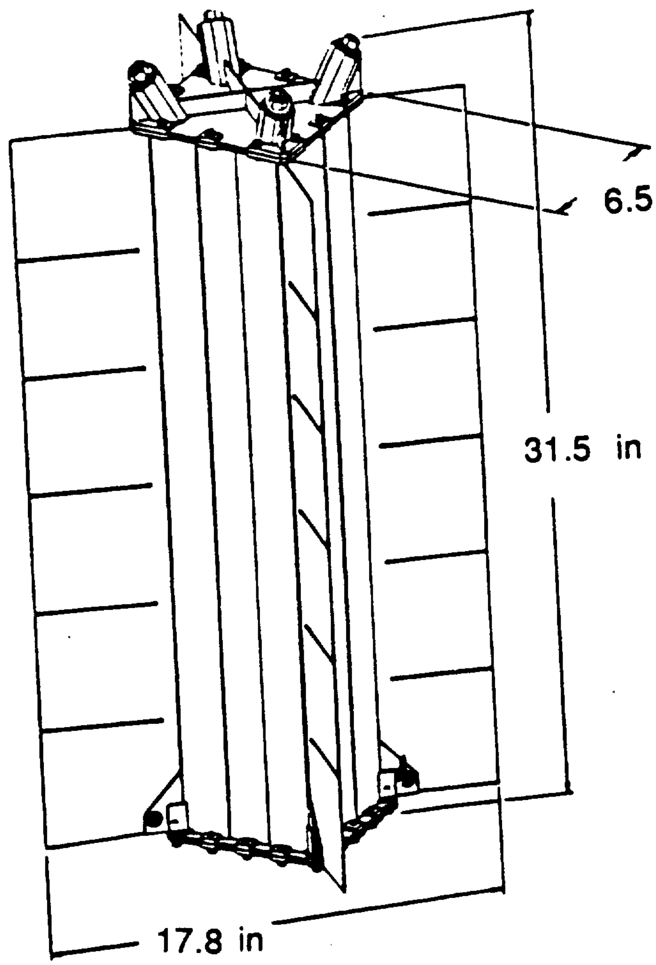

Figure PP-1

Power Supply sizing

The size of the RTG power supply is based on the power estimates listed in table PP-2. The 15% additional is to account for the degradation of the thermal power source over the lifetime of the mission. The weight breakdown for the total power supply system is given in table PP-4. These weights were calculated by scaling the weights of all of the components that will either be larger or that there will be more of in a larger RTG design (such as the number of modular heat sources), and then adding in the weights that would be constant for any size RTG (such as the end plates and their associated mounting hardware).

WEIGHTS

Housing		
outer shell	(1)	6.5
fins	(4)	2.5
emissive coating	-	0.27
aux cooling manifold	(1)	0.325
nuts	(4)	0.07
Converter		
T/E module	(128)	4.12
T/E C-seals	(128)	0.0325
nuts	(4)	0.038
foil ins.	(1)	2.275
foil ins ends	(2)	0.20
power converter	(1)	0.35
gas management assembly	(1)	0.29
electrical straps	-	0.40
PRD	(1)	0.90
C-seal - ends	(2)	0.06
other insulation	-	0.10
end caps	(2)	1.47
screws - end caps	(32)	0.17
pads and bushings	-	1.00
Heat Source Support System		
load spreaders	(8)	0.35
PG buttons	(8)	0.04
Bushings	(8)	0.09
Belleville Springs	(16)	0.14
Pistons	(8)	0.32
Compression plates	(40)	0.06
Preload Screws	(8)	0.31
Heat Source (3250 Watts Thermal)		
Heat source module	(13)	41.62
Total Weight		
lbs	-	64.0
kg	-	29.1

Table 4. Power Subsystem Weight Breakdown

Possible Use of Batteries

Batteries were considered for use in conjunction with the RTG's for power. They were looked at to be utilized when there was a peek demand for power. When analyzing the mission, this would be when transmitting from the Vicinity of Pluto, (or other planetary encounters). They could be charged on the way there and then utilized when transmitting. This did not turn out to be such a good idea. The batteries do not have that great of a weight advantage over the RTG source, and it would interrupt the mission when the batteries had to be recharged. Therefor, the added complexity for the power subsystem and the interference with the mission objectives ruled out the use of this type of hybrid power system.

Power conditioning and regulation

Because there are no batteries in the power system and because the RTG's supply a fairly constant voltage, power regulation does not seem to be a problem with RTG sources. The breakdown of the thermal source will show up as a loss of current, and the voltage will be basically constant. The wiring of the RTG will be redundant to increase the reliability and to ensure that no catastrophic loss of power will occur.

PROPULSION

The propulsion system for our spacecraft consists of two main subsections. The first will consist of the thrusters and fuel for the gravity assisted delta V at Jupiter and the thrusters and fuel for the AAC maneuvers for the portion of the mission from Earth to Jupiter. The second will consist of the thrusters and fuel for the AAC maneuvers from Jupiter to Pluto and beyond.

FUEL SELECTION

Trade studies have been performed to select the optimum fuel for the specific requirements of each stage. The Fuels considered for the various stages are: monopropellant hydrazine, bipropellant N2O4/MMH, bipropellant liquid oxygen and liquid hydrogen, and solid propellant. The advantages of hydrazine are that it is a monopropellant so it is simple and very reliable, and would therefor costs less than other systems, but it has very low performance. Bipropellant N2O4/MMH has much better performance, but it adds the complex valving necessary for bipropellant use, and reliability past 2 years has not been proven. LOX-LH has the highest performance of any propellant combination, but it is not storable for long periods of time, so it is ruled out for all but the earth departure stage. Solid fuels are storable for long periods of time and have intermediate performance. Their main disadvantage is they must be burned to completion. If they are selected, the AAC thrusters should be increased to make up for this loss of flexibility. These possible propellants are compared in Table PP-5.

POSSIBLE PROPELLANTS

ADVANTAGE/ PERFORMANCE	HYDRAZINE	LOX-LH	N2O4/MMH	SOLID
Isp (apx)	235	450	340	300
Storable	>12 years	continuous losses	2-10 years	>20 y.
complexity	medium	high	high	low
flexibility	high	high	high	low
average specific gravity Sutton p.206-9	1.008	0.28	1.20	1.174
used in deep space	yes	no	yes	yes
used for deep space missions	yes	yes	yes	yes

Table PP-5. PROPELLANT COMPARISONS

Thruster Selection

The selection of the thruster type and position is discussed in the AAC section of this report.

Fuel Selection for Each Stage

The thrusters selected to be mounted to the spacecraft will use hydrazine propellant because of a combination of the length of the mission (so the high reliability of hydrazine systems is preferred), and the low delta V required (so the low performance is not that large of a penalty). These thrusters will only be utilized in the segment of the mission from Jupiter to Pluto.

The AAC trusters mounted on the Jupiter assist stage as well as the main thruster for the gravity assisted delta V will utilize bipropellant (N2O4/MMH). The increased performance needed for this stage drove the decision for this propellant selection. This segment of the mission will only last apx. 4 years, to the thrusters on this stage will have a sufficient lifetime.

Tank sizing for Each Stage

The estimated delta V for the AAC system from Jupiter to Pluto is apx. 1.2 Km/sec. A safety factor of 0.1 km/sec has been added to this. This will give a total delta V for this stage of 1.3 Km/sec. From the rocket equation this gives a fuel mass of 412.5 Kg. Using the density of hydrazine, this fuel will require a spherical tank that is 0.922 m in diameter.

The estimated delta V for the AAC system from Earth to Pluto is apx. 0.4 Km/sec. The delta V required at Jupiter, with a safety margin is 1.7 Km/sec, for a total delta V of 2.1 Km/sec. Again using the rocket equation, a total propellant mass of 1075.8 Kg has been determined. Using the densities of the fuel and oxidizer and their mass mixture ration, the oxidizer of this stage requires a tank that is 0.935 m in diameter, and the fuel requires a tank that is 0.956 m in diameter.

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Structures Subsystem -- Group 1

The structural subsystem of the PFERD mission presented an interesting challenge. The constraints of a mission to Pluto are formidable. The spacecraft must survive the journey while keeping everything on the craft in working order. To this end, several different designs were looked at .

The basic structure of the craft is very important to the craft's survivability. The nature of the mission to Pluto requires that certain instruments are used; these in turn require that three booms are needed. For instance, the science platform can not be near the Radioisotope Generators (RTG's). Neither of these can be near the magnetometer, and these should all be kept away from the high gain antenna. This necessary configuration leads to a three boom arrangement with the fourth component in the center.

There were three basic arrangements that were looked at, all having a general Y-shape. The first consisted of a craft with the science instrumentation platform at its center. The three booms held the communication equipment, the magnetometer, and the RTG's. This arrangement has several benefits. Since the antenna is on a boom, the propulsion module can be attached to the center part facing the Earth. This would eliminate the need for the craft to turn 180° to perform a burn, and then to turn back to reestablish communications. This version also has its drawbacks, however. The antenna is very large, and placing it on a boom presents stabilization problems.

The next arrangement considered moved the antenna to the center of the Y-shape along with an electronics bus. The three booms hold the scientific scan platform, the RTG's, and the magnetometer. This craft has the stability of a large structure at its center, but it still has problems. Since the antenna is now at the center and pointing at the Earth, the propulsion module can not be there and is placed on the opposite side. This necessitates a 180° flip before a burn is performed. This presents a serious attitude control problem, but not an unsurmountable one. The main problem is getting the spacecraft to turn itself around without instructions from the Earth. The next craft attempts to eliminate the need for this extra maneuver.

The third arrangement is basically the same as the second one, with one important change. The craft's propulsion module would be pointing through the center of the antenna, thereby pointing in the correct direction. This eliminates the need for an orientation reversal, but presents numerous other problems. Since the nozzle will be pointing directly at the Earth, it will also be pointing at the instruments located at the focus of the antenna. These must be protected from the hot exhaust of the engine. Also, problems in communication due to reduction of antenna area and in sizing of the engine need to be addressed.

Due to constraints and tradeoffs mentioned above, the second of these arrangements was selected for use in the PFERD mission. It is Group One's belief that the problem of turning the craft 180° will be handled by existing technology in redundant computers and artificial intelligence. Therefore, the second arrangement will be the safest and most efficient way to reach Pluto.

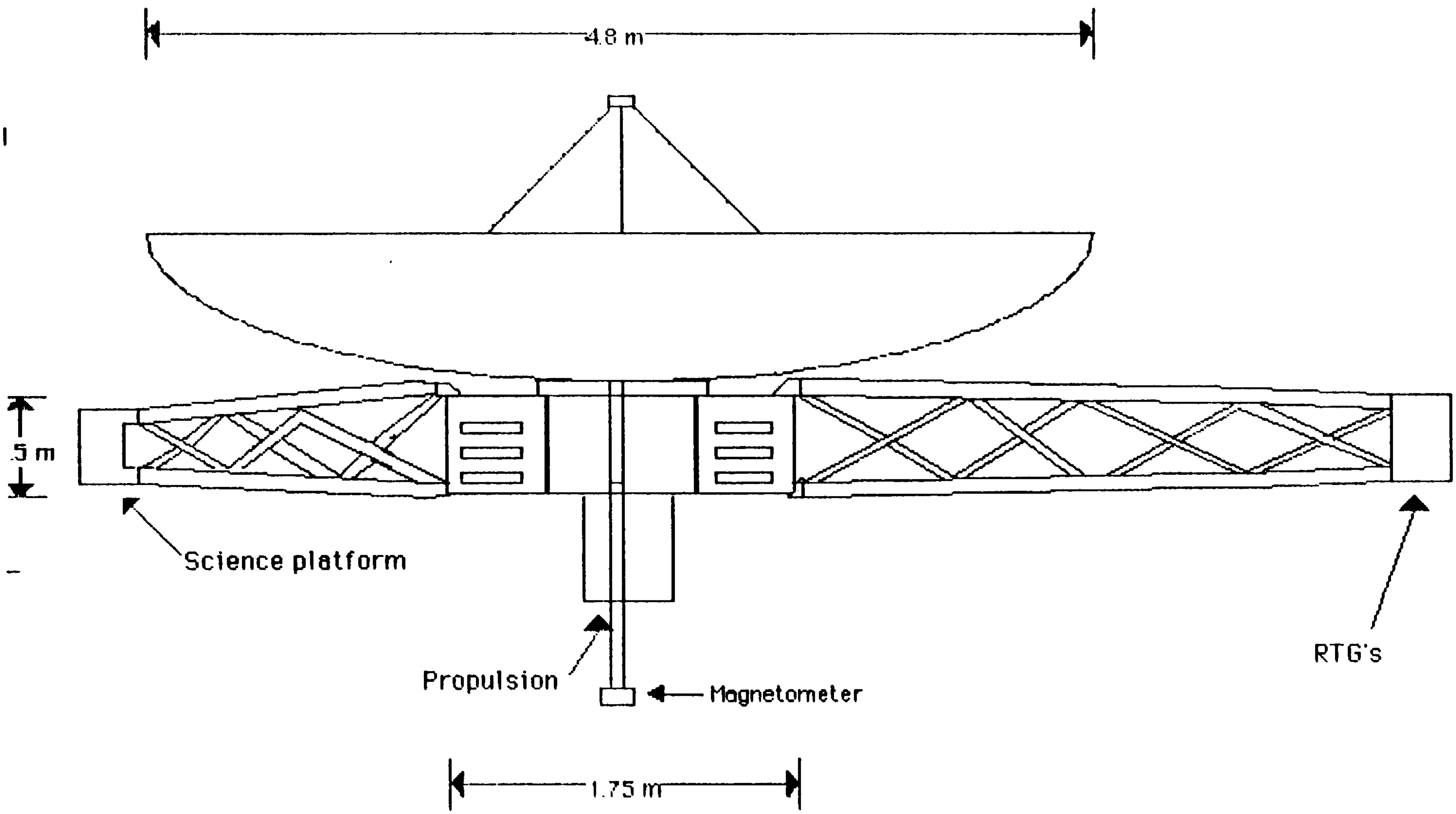
A sample drawing of this craft has been included on the next page. The observant reader will note the similarities in design among Group One's choice and previous craft flown by NASA, such as Voyager, Mariner, and Galileo (Dumas, p. 535). This is not just coincidence. Not only do these designs make good sense, but they also use many current technologies and manufactured parts that would be easily available to the PFERD mission. This will reduce cost and time required to complete the project, both of which are benefits to the PFERD program.

In addition to the components mentioned above, the craft will have micrometeorite shields outside the bus, protecting the antenna, and protecting the RTG's. The U-shaped scan platform will provide much of its own protection.

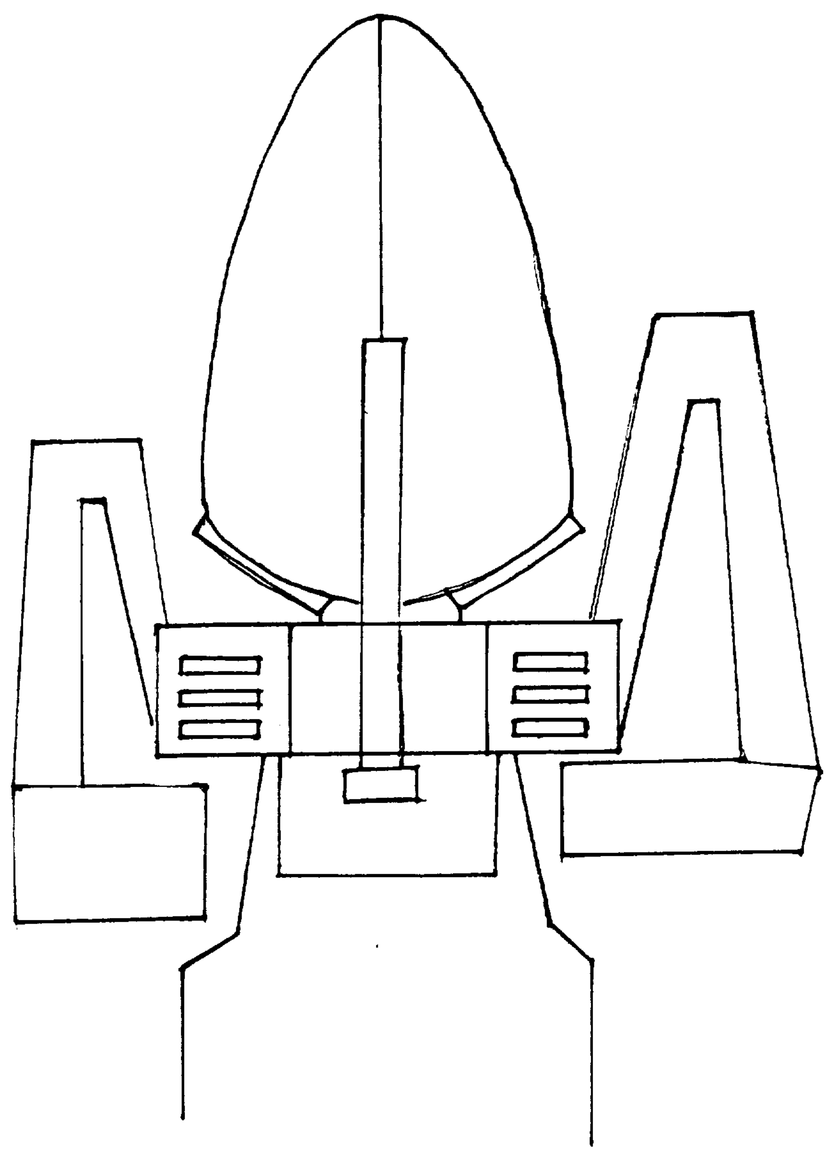
The spacecraft will also have a "fourth boom". The midcourse booster will be attached to the bus on the opposite side of the high gain antenna with explosive bolts to allow it to be jettisoned after firing. This will provide the boost at an intermediate planet to attain the required velocity to make it to Pluto.

The craft will have a pair of spherical fuel tanks slung under the boom above the midcourse booster. These will provide fuel for the attitude control thrusters and an additional maneuvering engine.

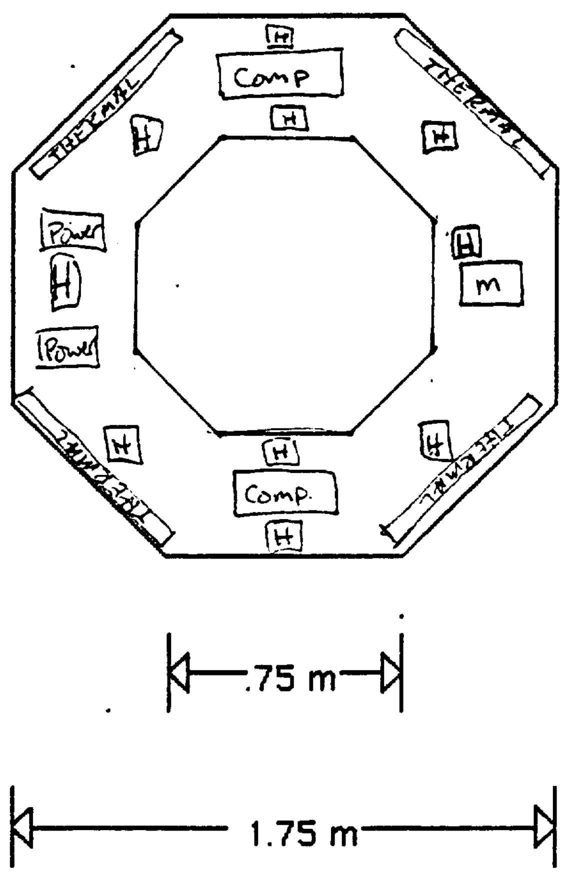
Finally, we must consider the layout of the craft before it reaches orbit. The PFERD mission will utilize the Titan IV to


4.8 m
.5 m
Science platform
Propulsion
Magnetometer
RTG's
1.75 m

PFERD LAUNCH CONFIGURATION



Electronic Bus Layout



put the probe into Earth orbit, and therefore the craft must fit in the payload bay of Titan IV. Utilizing a folding antenna and retractable booms, the probe will be able to fit into a cylinder 3.5 meters in diameter and 5 meters tall, well within the range of the Titan IV's capabilities. (see launch configuration diagram)

Once the spacecraft's general shape has been determined, the materials used to construct it must be examined. The driving factors in material selection are low weight, high strength, good temperature ranges, and resistance to radiation.

High strength and low weight are the initial considerations for the materials. Composite materials have the best strength to weight ratio, but are very expensive and are resistant to loading in only one direction. Titanium and Aluminum are both very strong and very light, with Titanium being the better of the two. Aluminum is available at a much lower cost, however.

Almost all metals are resistant to radiation, so this is not a factor in their tradeoff studies. However, composite materials have been known to suffer degradation due to radiation exposure in the space environment. (AAE 241 notes, Set #9) This makes them a poor choice for external structural components of the probe.

Temperature is also a factor in material selection. Designers must worry about metal evaporation at high temperatures, which is demonstrated in the following chart. The metal will lose 0.040 inches in one year at the corresponding temperatures.

Table STR 1

Metal	Temperature (°C)
Cd	120
Mg	240
Al	810
Fe	1050
Ti	1250
Mo	1900
Ta	2300
MgO	1090

From Space Materials Handbook, page 498

Titanium looks to be the better choice for high temperatures. The PFERD probe will spend most of its time in the outer solar system where temperatures are very close to absolute zero, but the high temperatures will be present while the probe is in the vicinity of the earth and inner planets. However, in choosing metals for use in spacecraft structures, one does not usually worry about the effect of the environment because most metals have very similar properties in space. (Space Materials, p. 640) It is only a design using composites that must take these factors into careful account.

Due to the above constraints, Titanium was selected as the main structural material for the booms, platforms, and structure supporting the instruments on the high gain antenna dish. These all will have exposure to the sun for the greatest time period. Titanium also can be used to construct the fuel tanks. Aluminum will be used to construct the bus and micrometeorite shields protecting the craft on its journey. Since the craft has been modeled after several existing spacecraft, the choice of materials and launch configuration shown will allow the craft to withstand the loads during launch.

The following chart presents the approximate masses of the general components of the spacecraft. The structural masses are all estimated from various figures found in JPL Div. 35 Mass Estimation Reference for the Galileo and the Voyager probes modified to fit the PFERD mission needs. The subsystem masses are from the individuals responsible for that subsystem. At the bottom of the chart is the final estimate for the cruise craft, excluding the propulsion modules.

Table STR 2

Component	Mass (in kg)	Totals
Electronic Bus		164
AAC	72	
CCC	5	
Science	1	
Thermal	15	
Structure	61	
Power	10	
Antenna Structure		50
Antenna	30	
Support	20	
RTG Boom		45
RTG's	30	
Boom	15	
Science Platform		131
Boom	20	
Scan Platform Struc.	26	
Science Instr.	60	
AAC	22	
Thermal	3	
Magnetometer Boom		13
Instrument	3	
Boom	10	
Fuel tank (empty)		46
fuel tank thermal control		4
Total Non-propulsion mass		~453 kg

A complete breakdown of each subsystem is available at the end of this section.

Thermal Analysis

In order for a spacecraft to survive the journey to Pluto and be in working condition at the time of arrival, the temperature must be strictly controlled. The temperature of the space environment is near absolute zero, and instruments must be kept within a certain temperature range. There are two ways of controlling temperature: passive thermal controls and active thermal controls. (Space Materials, p. 99)

Passive controls use no power or moving parts. They consist of components such as multilayer blankets, reflective and absorptive panels, and coatings to control the temperatures of the interior of the spacecraft. These controls are very reliable, but are less precise. Also, very accurate knowledge of the thermal conditions in space is necessary to use passive controls. A passive control also generally results in a minimum weight and minimum cost system.

Active controls utilize power and/or moving parts to regulate temperature. Some examples of active controls are heaters, selective exposure disks, and fluid transport refrigeration/radiation devices. These controls give very precise temperature control, and they do not require accurate knowledge of the environment. As with all mechanical devices, the potential for failure exists. (Space Materials, p. 99)

The PFERD spacecraft will utilize a combination of these two methods. Since the majority of the cruise will be far away from the sun, the primary concern must be keeping the craft warm.

Since the instruments must be kept in a relatively narrow range of temperatures, the active controls will be used more than the passive ones. The active controls will provide continuous heat, while the passive will keep high temperatures near the sun at bay and improve heat retention far away from the sun.

During the cruise, the craft will be shielded from the sun by the high gain antenna. It is this design that renders the craft insensitive to changes in solar intensity. It is also the reason so many craft (such as Mariner and Voyager) use a similar structural design. (Dumas, p. 536)

The PFERD mission applies knowledge gained from previous space missions to control temperature. Thermal control is divided into certain areas: Bus, scan platform, and other boom control.

Bus thermal control consists of the isolation from solar heating provided by the high gain antenna, multilayered insulation on all sides to prevent thermal gradients, radioisotope heating units (RHU's) strategically placed in the bus, and thermostatically controlled louvers in the panels. (Dumas, p. 538) These measures provide a stable thermal environment inside the bus to allow operation of the instruments.

The PFERD scan platform will be shaped like a large three dimensional U. This will provide thermal protection for the instruments while allowing certain instruments to be able to view the environment (such as the Astros 2).

The RTG's will require no heating, as they are actually producers of waste heat. An attempt was made to harness and utilize this excess heat, but sufficient transport media was not discovered. Heat pipes were investigated, and it seems that an osmotic system might have the capability to transfer the heat, but the added weight and complexity of an osmotic pumped heat pipe did not fit with the requirements in the Request for Proposal.(Tanzer, p.184-5) The magnetometer will require no heating units, also. The RHU's used will interfere with the instrument's performance.

The thermal subsystem approximate mass is presented below. The masses are based primarily upon data from similar spacecraft, such as the Mariner. The reader will note large differences in the figures for the Galileo and those for the Mariner. This is probably due to the vastly different missions the two flew. Although both were scheduled to end up at Jupiter, the Galileo flew first inward towards the sun, requiring more heat protection.

Table STR 3

Estimated Thermal Subsystem Masses

Galileo probe	80 kg	(JPL doc)
Mariner Jupiter-Saturn scheduled for 1977	11 kg	(Dumas, p. 542)
PFERD mission	20 kg	(see mass section)

As the reader may note, the mass of the thermal subsystem can vary greatly according to mission plans.

Conclusion of Structural Section

This report presents Group One's view of the best structural subsystem for the Pluto mission, which has been dubbed PFERD. It has followed the Request for Proposal in its tradeoff studies and design. The structural subsystem has optimized weight and cost as much as possible in the choice of materials and thermal equipment. All of the materials considered in this report exist at the present time, so that requirement is taken care of. The Titan IV is being utilized, and a diagram is shown of how the craft will fit inside of the payload bay, and the Titan should easily be able to lift the entire mass of the craft. Simplicity has been stressed throughout, and as stated, most of the components have been flight tested on previous missions and can be relied upon. The structural system will exceed the mission life, because there is no practical limit on the materials, and the RTG heaters will last easily out to Pluto (in excess of 20 years). Off the shelf design is being utilized across the board, as shown by the craft being modeled after several craft that have already flown.

In summary, the PFERD mission will deliver the "spacehorse" (the actual craft) to Pluto safely and will return valuable scientific data to Earth.

MASS TABLE

SUBSYSTEM, Part	Location	Weight (kg)
AAC		
1 fine sun sensor	bus	1.75
1 microcomputer	bus	.1
electronics, support	bus	10
Star tracker, FORS	scan platform	22
control thrusters	bus	60
fuel needed for cruise	tank	n/c
CCC		
1 microcomputer	bus	.1
1 high gain antenna	antenna struct.	30
electronics	bus	5
SCIENCE		
platform instruments	scan platform	60
magnetometer	own boom	3
magnetometer electronics	bus	1
Power/Propulsion		
RTG's	own boom	30
fuel tank	below bus	46
electronics/cables/power	bus/booms	10
Mission Management		
none		

Structures

bus structure		
panels	bus	30
shielding	bus	26
connectors	bus	5
science boom	-	20
scan platform/machinery	-	26
magnetometer boom	-	10
RTG boom	-	15
antenna support	antenna struct.	20

Thermal

bus thermal protection		
blankets	bus	3
heaters (10)	bus	2
louvers and machinery	bus	10
scan platform heaters	-	1
scan platform blankets	-	2
fuel tank heaters	-	2
fuel tank blankets	-	2
coatings	-	1

TOTAL 453.9 (-FUEL)

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Structural Subsystem
Group 1
PFERD.

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Space Materials Handbook; Lockheed Aircraft Corporation. (Lockheed Missiles and Space, Sunnyvale, CA; 1965): pp. 95-177, 493-509, 639-640.

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Set No. 9: Last Page, structural materials data.

Set No. 13: Inertia data; pp. 2-6.

MISSION MANAGEMENT, PLANNING AND COSTING SUBSYSTEM

MISSION CONFIGURATION

The first consideration was that of the type of mission to be flown to the Pluto-Charon system. Depending upon which approach is taken, the requirements placed upon the probe design, trajectory type and the propulsion system vary greatly. Because of the preliminary nature of the exploration of Plutionian space, a lander was considered extravagent as a first mission and was ruled out immediately. Thus, the selection for the PFERD spacecraft was a decision between two possible configurations: orbiter and flyby.

Initially, both missions fulfilled the RFP requirements in the scientific domain. The orbiter would provide a greater amount of data return as opposed to the flyby mission, albeit at a greater cost due to the increased complexity of the probe and delivery system. Once the trajectories were examined, however, the delta-V required for the orbiter at Pluto became prohibitively large. Thus, by default, the configuration chosen for the PFERD mission was that of a flyby. Upon close examination, the flyby probe could easily accomplish the most compelling scientific objectives at Pluto, and do so at a cost that would make it a very viable mission.

TRAJECTORY DETERMINATION

The trajectory for PFERD had to fulfill several requirements, both stated and implied in the RFP. Specifically required by the RFP was a launch date between the year 2000 and the year 2010. Other considerations in the RFP imposed additional limitations. By stating that components available on or before the year 1999 must be utilized for the mission design, the launch vehicle selection was limited and thus the range of trajectories was narrowed further. Minimization of cost would suggest a short mission duration and a minimum delta-V requirement as well.

In determining the optimum trajectory for PFERD, three types of transfers were considered. The first of those was the direct trajectory, with no intermediate encounters en route to Pluto-Charon. This type of transfer has the benefit of a short flight time and simple navigation, but requires a large delta-V at Earth.

The next two types of transfers examined both involve gravity-assist maneuvers at planetary encounters along the way. The first of these uses an encounter with one of the outer planets to increase the probe's velocity and modify its flight path. By far the most influential body is Jupiter with a gravitational constant of 1.267×10^{8} km^3/sec^2, over three times that of the next most massive planet, Saturn. Thus, any trip to the outer planets would inheiret a large gravity assist if an encounter with Jupiter were possible. This is the type of trajectory that was used by Voyager I and II during their "grand tour" of the outer solar system.

With the increasing size of payloads and the unavailability of a heavy-lift vehicle in the U.S. arsenal of expendible launch vehicles, a new type of trajectory has been determined that uses a swingby of Venus. This type of trajectory costs less in terms of delta-V to deliver a payload to the outer solar system where additional gravity-assist maneuvers can be performed. These are of two types: Venus-Earth Gravity Assist (VEGA) and Venus-Earth-Earth Gravity Assist (VEEGA). These trajectories are currently in use by Galileo (VEEGA) and the upcoming Cassini mission to Saturn (VEGA) [1].

The tool used to evaluate the applicibility of various trajectories to the PFERD mission was MULIMP, a trajectory optimizing software [2]. This program, while somewhat clumsy and producing occaisionally conflicting results, provided a database of various trajectories from which the final trajectory was decided upon. Listed in Table 1 are the trajectories investigated and the criteria they met. After consideration of the results of MULIMP, a decision was made in favor of the Jupiter Gravity Assist (JGA) trajectory over the only other viable candidate, a Mars-Jupiter Gravity Assist (MJGA). This was primarily because of the fact that a large amount of the delta-V of the latter mission had to be executed at Jupiter (Table 2). This would entail transporting a large amount of propellant to Jovian space and thus drastically increase the fuel-to-payload ratio of the spacecraft.

It can be noted here that Jupiter, having a synodic period of about 12 years and providing gravity-assist to the Voyagers in the late1970s, could be expected to be in a position to do so again in the first few years of the twenty first century.

TRAJECTORY COMPARISON

	Criteria	
Trajectory	Low Delta-V	Trip time <20 years
Direct	-	X
NGA	-	X
MNGA	-	-
VNGA	X	-
VJNGA	X	-
VEJNGA	X	-
VJGA	X	-
VEJGA	X	-
VSGA	-	-
VESGA	-	-
JGA	X	X
JNGA	X	-
MJGA	X	X
JENGA	X	-
SNGA	-	-
JUGA	-	X
MJUGA	-	-

Table 1

Note: Planet names are represented by the first letter of their name;

GA denotes "gravity assist"

JUPITER GRAVITY ASSIST TRAJECTORY

Date	Location	Velocity (km/sec)	delta-V (km/sec)
2000 AUG	Earth	9.77	9.77
2004 MAY	Jupiter	6.237	0.91
2018 MAY	Pluto	8.582	0.00
-------------			-----
17.68 Years		Total DV =	10.68

MARS-JUPITER GRAVITY ASSIST TRAJECTORY

Date	Location	Velocity (km/sec)	delta-V
(km/sec)			
2000 MAR	Earth	9.774	9.774
2007 DEC	Mars	19.079	0.455
2010 MAR	Jupiter	7.150	7.960
2019 JAN	Pluto	16.581	0.00
---------------			------
18.85 Years		Total DV =	18.189

Table 2

LAUNCH VEHICLE SELECTION

As previously stated, the United States variety of launch vehicles for a mission of this scale is at present limited to two: the Space Shuttle and the Titan expendible launch vehicle (ELV). With the shuttle accident of 1986 and the subsequent banning of the Centaur upper stage from future shuttle flights, the only capable vehicle currently available is the Titan IV. With the introduction of the solid rocket motor upgrade (SRMU) and an improvement of around 30% mass to low Earth orbit [3], the Titan IV/Centaur G' was selected as the ideal launch vehicle for a flyby mission to Pluto. This increase in performance will enable the Titan IV to deliver 13,600 pounds to geosynchronous orbit, and approximately xxxx pounds to escape velocity.

If development leads to production of the shuttle-C, this could aslo be employed as an alternative to the Titan ELV. The shuttle-C, with its proposed payload of upwards of 80,000 pounds to low Earth orbit, could provide for a whole new range of trajectory options. With the possibility of greater delivered payload, trajectories such as the Mars-Jupiter gravity assist, which require a substancial delta-V at the intermediate encounter body, could be made accessable.

SEQUENCE OF EVENTS

Using the Jupiter gravity assist trajectory data derived from MULIMP, a timeline of mission events is presented.

2000 AUG 31	Launch by Titan IV/Centaur Nine day launch window
2000 SEP	PFERD spacecraft extends antenna, booms System tests/equipment checkout
2000 OCT	Except for course correction control, PFERD systems shut down
2004 MAR	PFERD system power-up, begin data taking for Jupiter flyby
2004 MAY 19	PFERD at Jupiter closest approach, 3.0 radii Engines fired to produce DV of .91 km/sec
2004 JUL	System shutdown
2018 MAR	System power-up
2018 MAY 07	Pluto closest approach, 3.0 radii
2018 JUL	Pluto encounter ended, extended mission Extended mission objectives commence

COSTING

Cost estimation for a project such as this has been found to conform to a simple algorithm. The estimate is divided up into two major categories: the development project, and the flight project. The development project is subdivided into two more groups, one comprising the actual flight hardware and the other the support functions.

Each of these can be estimated in the number of manhours required for each subsystem as either Recurring Labor Hours (RLH), or Development Labor Hours (DLH).

DEVELOPMENT PROJECT-FLIGHT HARDWARE

System	Mass (kg)	DLH	RLH
Structures	106.0	387	126.5
Thermal control	23.0	125.2	58.35
Propulsion	56.0	535.5	131.2
Attitude & Articulation	93.85	1550	725.3
Communications	20.0	634	155
Antenna	30.0	1394	400.7
Command & Data	0.1	71.5	17.65
RTG Power	30.0	358	242
Line-Scan Imaging	5.0	435	136.2
Vidicon Imaging	28.0	604	229
Particle/Field Inst.	8.0	314	101
Remote Sensing Inst.	22.0	380	33.85

DEVELOPMENT PROJECT-SUPPORT FUNCTIONS

	DLH
System Support & Ground Equipment	2085.6
Launch+30 days Ops	665.8
Imaging Data Development	.37
Science Data Development	88.1
Management	711

FLIGHT PROJECT

	DLH
Flight Operations	11014
Data Analysis	4681

TOTAL LABOR HOURS	21,599.62
TOTAL LABOR COST(FY77)	226,796.01

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P-97

N91-16047

Intrepid: A Mission to Pluto

Aerospace Design Class
AAE 241
April 24, 1990

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Introduction

A proposal for an exploratory spacecraft mission to the Pluto/Charon system has been written in response to the "Request for Proposal for an Unmanned Probe to Pluto," or RFP. The RFP lists many design requirements that must be satisfied by the proposed spacecraft. They are as follows:

1. design an unmanned scientific study to Pluto and Charon
2. mission science objectives must be described and justified
3. optimize performance, weight, and cost of spacecraft in design tradeoffs
4. launch time: between the years 2000 and 2010
5. design should stress reliability, simplicity, and low cost
6. spacecraft must be able to adapt to whatever environment it may encounter
7. lifetime of spacecraft must include mission time and a safety margin
8. nothing in spacecraft's design should preclude it from performing several missions
9. use existing hardware when possible in design
10. use materials and techniques available by 1999
11. use latest advances in artificial intelligence
12. amount of on-orbit assembly should be identified and minimized
13. identify use of space shuttle if applicable
14. if space shuttle is used, it must apply to NASA standards
15. for cost estimates: assume four spacecraft built, three flight ready and one for integrated ground test system

Under the guidance of the RFP requirements, the spacecraft *Intrepid* was designed. The RFP requirement which was of primary importance is that to keep costs at a minimum. The less expensive the design is, the more attractive it would be to those ultimately in control of funding this project: the United States Congress.

Also, the reduction of flight time is of extreme importance because the atmosphere of Pluto is expected to collaspe close to the year 2020. If *Intrepid* should arrive after the collapse, the mission would be a failure; for Pluto would be only a solid rock of ice.

SCIENTIFIC INSTRUMENTATION (SI)

Mission Science Objectives

Our desire to discover more about the Pluto/Charon system is the driving reason this project was conceptualized. Due to the vast distances involved, there is an extremely limited amount of knowledge pertaining to the Plutoian system. Information that is presently available is subject to a relatively high degree of error. Our Pluto/Charon mission will answer questions regarding the system's age and origin, its classification as planet and moon, internal dynamic interactions, and gather such planetary characteristics as surface and atmospheric composition, magnetic field, rotation rates, et cetera.

In addition to obtaining data from the Plutoian system, *Intrepid* will observe Jupiter and examine the interplanetary space through which it passes. The Jovian system will provide a gravity assist to the spacecraft. During this time, *Intrepid* will point its far-scanning instruments towards the Jupiter in order to gain more information about its complex planetary system. The vast majority of the mission, however, will be spent in deep, interplanetary space. *Intrepid* will gather data concerning the solar winds, the interstellar particle medium, and the solar and interplanetary magnetic fields. This information will help answer questions such as how our solar system interacts with the rest of the galaxy and where the boundary of the solar system is. In addition, these investigations are necessary for monitoring the calibration and performance of field instruments.

Science Objectives at Pluto

Upon arrival at Pluto and Charon, the majority of science objectives will be examined. The far-sensing instruments will begin collecting data on the Plutoian system up to three months before the encounter date. These instruments will study the transition between the interplanetary media and solar wind/interstellar media, the interaction of Pluto and Charon's planetary magnetic fields, the structure of the system's magnetospheres, the ionosphere, plasma density profiles, rotation rates, and dynamic interaction between Pluto and Charon.

As the encounter date nears, the remaining scientific experiments will also be in operation. The atmospheric character will be closely scrutinized. This includes any unusual features or haze, temperature and pressure profiles, and composition. The surfaces of Pluto and Charon will also be a major item of focus. The scientific instrumentation will carry out a mapping of the surface geology, examine the polar icecaps, study the color variations and albedo of the surface, determine cratering rates, and composition of the atmosphere and surface. In addition, the imaging equipment will take numerous photographic pictures of Pluto and Charon. Accurate densities, masses, and radii of the planetary figures will be determined.

All this information will be used to understand what type of planetary system Pluto and Charon comprise. Whether Pluto is actually a moon of Neptune which escaped, an oversized asteroid, or truly a planet may be ascertained. Because it is the farthest planet from the Sun and has never been examined by a spacecraft, there is very little known about it.

This mission will fill this void and perhaps provide more insight into the formation of our solar system.

Limitations and Requirements of the Mission .

Because of Pluto's extreme distance from the Sun, it is of prime importance to arrive before Pluto's thin atmosphere collapses. This is predicted to occur between 2020 and 2025. Should *Intrepid* arrive too late, it will encounter only a large ball of ice. Thus in order to gain as much valuable information as possible, it is of the utmost importance that the spacecraft reach Pluto as soon as practical.

In order to beat the atmospheric collapse deadline, the *Intrepid* spacecraft must be assembled and launched before 2005. Therefore, it must be fully designed, built, and launched within twenty-five years. Plus, the spacecraft must be assembled with reliable parts so it will survive the fifteen year voyage.

The RFP has several explicit requirements that directly relate to the SI subsystem. They are as follows:

1. design an unmanned science study to Pluto and Charon
2. mission science objectives must be stated and justified
3. design should stress reliability, simplicity, and low cost
4. spacecraft must be able to adapt to whatever environment it may encounter
5. lifetime of spacecraft must include mission time and a safety margin
6. nothing in the spacecraft's design should preclude it from performing several missions
7. use existing hardware when possible in the design
8. use material and techniques available by 1999

In order to fulfill these inherent and explicit requirements, the scientific hardware has been selected from previous deep space missions. This decision to use existing hardware greatly reduces the development costs involved, guarantees that they can survive the deep space environment, and that they are indeed reliable. Fewer experiments have been included than in previous missions in order to keep the spacecraft design simple, low in mass and power requirements, and as inexpensive as possible and while carrying out the science objectives.

The spacecraft missions from which the instrumentation originates from should be: recent enough to have relatively current technology, not be dependent upon much visible light, and have long component lifetimes. The missions which fit these requirements best are the *Voyager*, *Galileo*, and *Mariner Mark II* series. Although the *Mariner Mark II* series (MMII) has not been launched, considerable research has been invested to ensure that they are deep space worthy. Also, the MMII will be completed well before the technology deadline of 1999.

The total component lifetimes is the largest stumbling block that must be overcome with respect to science instrumentation. The scheduled mission lifetimes of the *Voyager*, *Galileo*, and *Mariner Mark II* series are approximately four, four and one half, and eight years respectively. Since *Intrepid's* scheduled mission time is fifteen years, the scientific instrumentation portion of the development costs will be spent upon modifying the existing equipment so it will be able to surpass the lifetime requirement. The mission lifetime can be a deceiving measure of the actual lifetime performance. The Voyager's instruments, for example, have been proven to be extremely reliable and are expected to exceed the original mission lifetime many times. Therefore, the development costs related to ensuring instrument lifetime should prove to be rather small.

Also, the science objectives may be slightly altered once the Hubble Space Telescope examines the Plutoian system. It may provide knowledge of large craters or surface anomalies that ought to be more closely examined, a better approximation of the depth of the atmosphere, and more accurate planetary characteristics. Once this data is analyzed, there may be additional scientific equipment chosen in order to better study the system. In addition, the existing equipment may be further modified to incorporate additional objectives.

Science Instruments

The following science instruments have been selected:

Cosmic-Ray Detector System (CRS), from *Voyager*
Plasma Detector System (PLS), from *Mariner Mark II* (MMII)
Magnetometer (MAG), from MMII
Ultrastable Oscillator (USO), from *Voyager*
Solid-State Imaging (SSI), from MMII
Ultraviolet Spectrometer (UVS), from MMII

Figure 1.1 shows the scientific instrumentation layout. Appendix B lists the masses and power requirements for these instruments.

The equipment selected from the MMII mission is not yet fully developed. *Mariner Mark II* is also using available hardware designs and upgrading them with more recent technology. The designs that they are using, judging from the approximate masses and power demand estimates, are from the *Galileo* spacecraft. (Draper, 10) Also, the MMII has the longest lifetime of the missions considered. Therefore, when the

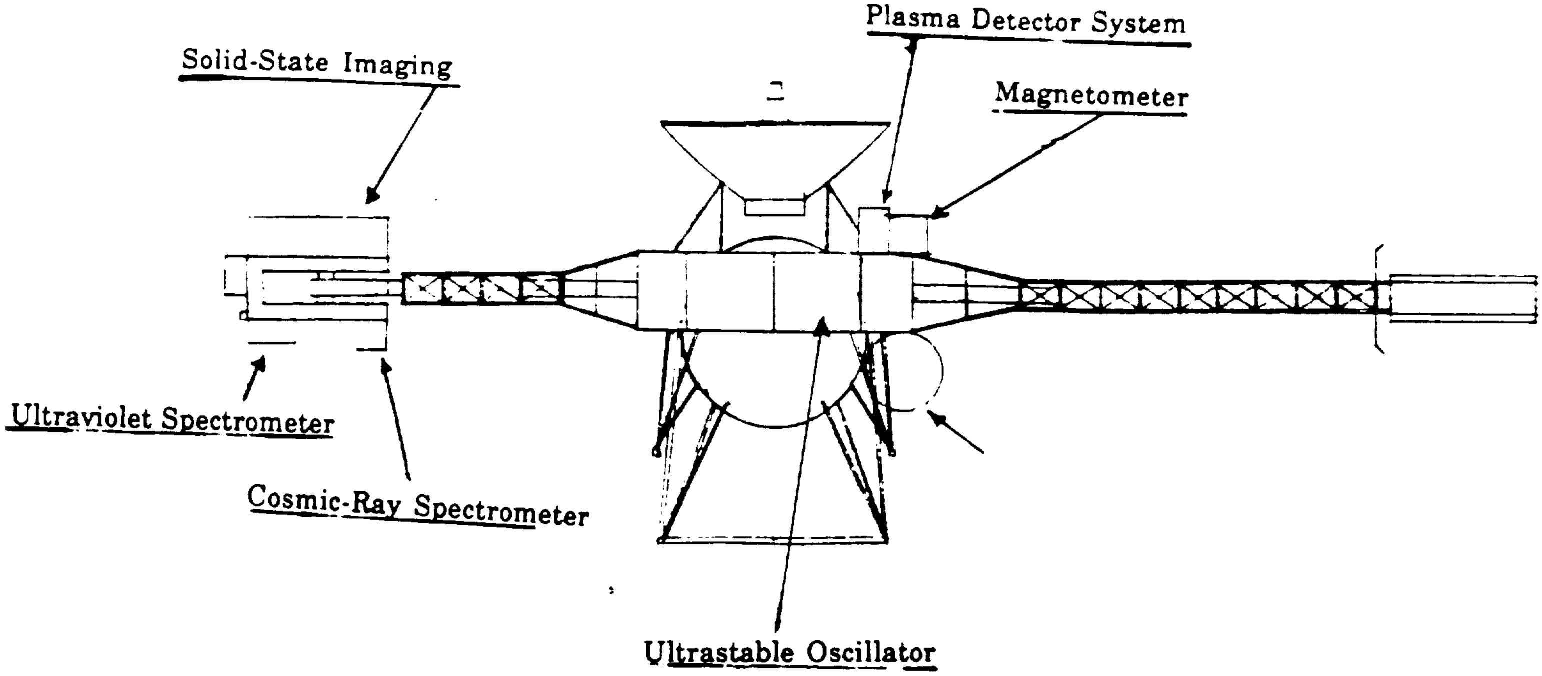

Fig. 1.1 Layout of Scientific Instrumentation

Galilean instrument is better than that of the *Voyager* for the Pluto mission, the MMII instrument with the more recent technology already incorporated is the final selection. The *Pioneer* is not considered because of the age of the technology incorporated in it.

The first three instruments listed above, the CRS, PLS, and MAG, are particle or field scanning devices. These do not require accurate pointing at a particular target body. The precise orientation of the spacecraft needs to be incorporated with the instrument data for an accurate record of the medium under study. These instruments are turned on intermittently throughout the mission, and when *Intrepid* approaches Pluto. The other three instruments are concerned with planetary science and are turned on only when nearing a planetary target. The USO is a supplement to the radio equipment; it is used in conjunction with the radio instruments to conduct many radio science experiments. The USO does not require any particular pointing or unobstructed view of space. The last two instruments, the SSI and UVS, are best placed on a high precision pointing platform because of their accurate pointing requirements.

The Cosmic-Ray Detector System

Cosmic-rays originate from planets and from stellar sources. Their energies vary accordingly: (Flight Science Office, 4.1)

Energy	Origin
< 100 MeV/nucleus	Interstellar (from our Galaxy)
1-100 MeV/nucleus	Nearby interstellar, or outer solar system
> 30 MeV/ nucleus	Jovian magnetosphere

Cosmic-ray composition past the giant planets is currently unknown. The study of cosmic-rays in this region may provide insight on galaxial composition and formation. Also, it will help scientists decide where the boundary of the solar system is located.

The CRS has three separate particle telescopes for examining the different energy ranges listed above. For the study of Pluto and Charon, the telescope of the lowest energy range would be used. These individual telescopes measure the charge, energies, and particle directions in their respective energy range. In order to provide an unobstructed view for the telescopes, the CRS has been mounted on the Science Scan Platform. Also studied will be the cavities caused by the Plutoian system in the stellar radiation. Because the CRS measures the charge composition of the planetary magnetosphere, it provides a certain redundancy of data for planetary magnetic fields. Therefore, not all the information concerning the magnetic fields would be lost should the magnetometer become damaged.

The reason for the selection of Voyager's CRS is, primarily, that it fulfills the scientific objectives of cosmic-ray investigation. The *Galileo* spacecraft instrumentation studied particles of less than 60 MeV. (Colin et al., 6) Their primary purpose was to investigate the Jovian system, and therefore did not have as great of a range as *Intrepid's* mission requires. Thus, Voyager's design is the best choice.

<u>The Plasma Detector System</u>

Plasma is composed of mostly low-energy electrons and ions. The plasma found in the solar system originates from stellar sources and from the magnetospheres of planets. Because of the limited exploration of the area, little is currently known about the composition of interplanetary plasma beyond the giants.

The PLS measures this plasma and records its energy levels, ionic composition, and velocities. Of particular interest for this mission is the measurement of the magnetospheric plasma of Pluto and Charon. The PLS will record the interactions between Pluto's and Charon's magnetospheric plasmas with one another, and with interstellar and solar winds. In addition, the PLS will identify the species of interstellar ions. Also, the interaction of the interstellar and solar winds in the heliopause will be studied.

Galileo's plasma experiment is clearly the better choice. It is able to measure approximately ten times the energy ranges that *Voyager* could. Its temporal resolution is twenty times faster, which is extremely important for a flyby mission. Also, the PLS can produce a three-dimensional vector velocity profile distribution of the plasma particles every twenty seconds and identify species of interplanetary ions; which are tasks the *Voyager* could not perform. (Colin et al., 133)

Magnetometer

Magnetic fields, studied by the magnetometer, are present everywhere in the solar system. They are present on most planets, and accompany the streams of charged particles which comprise the solar wind. There is a great difference in strength between the magnetic fields of planetary and stellar origin. Because of this variance, there are two sets of sensors which are sensitive to the differing levels of intensity.

The MAG studies the magnetic fields from all origins: planetary, solar, and interstellar. It measures the strength, fluctuations, and structure of these fields. Of special interest is where these fields meet and influence one another. The MAG also measures the heliopause accurately. Because of the wide-spread presence of the magnetic fields, the magnetometer is an important instrument for studying the large-scale characteristics of the solar system. It provides much information on how the solar system interacts with itself and the rest of the galaxy. For this mission, the MAG will also determine if Pluto and Charon have magnetic fields. If so, it will study the structure and characteristics of these fields.

The MAG's sensors are located on a separate boom of the spacecraft. This location has been designed in order to minimize the detection of the spacecraft's magnetic fields by the sensors.

Each magnetometer from the separate spacecraft has the same purpose: to provide an in depth study of the magnetic fields. Therefore, since the MMII incorporates the more current technology and has the longest design lifetime, it is the one that *Intrepid* will use.

Ultrastable Oscillator (for radio science experiments)

Unlike the other instruments, the USO is only a supporter of a larger experimental system. The telecommunication equipment is used to preform a number of important experiments. The USO reduces the transmitter frequency fluctuations to 1 to 4 x 10^{-12}. (Anderson et al., 228) It drastically improves the accuracy of results from the radio science experiments. The radio science is able to deduce the following information by use of occultations and scintillations: temperature, pressure, and density profiles of the upper atmospheres; electron density profiles and irregularities in the ionosphere; magnetic field direction; gravity fields; mean densities; bulk composition of the planets; and plasma density and dynamics. Relativistic effects may be also be investigated by means of comparing how signals of different frequencies from the spacecraft are affected by the solar wind and corona. (Anderson et al., 224)

As *Intrepid* passes Pluto, the path will be such that Pluto is positioned between Earth and the spacecraft. This configuration provides the occultation of the radio signals required for many of the experiments listed above.

In this case, the MMII used the same USO as did *Voyager*. (JPL Mission Group) Because there is little available information about Galileo's USO and the technology used for the *Voyager* is identical as for the MMII, *Intrepid* will use the USO of *Voyager*.

<u>Solid-State Imaging</u>

The SSI device is a combination telescope-camera which observes and determines much about planetary atmosphere as well as its visual physical characteristics. The SSI will be focused on the Plutoian system for 72 days before and after the encounter date. This prolonged viewing window, using long exposure times, will allow an in depth study of the internal dynamic interactions of the Pluto/Charon system. Since Charon rotates about Pluto once every 6.4 days, numerous rotations will be observed. In addition, the SSI system will perform geographical mappings of both planetary figures when closer to the system, using shorter exposure lengths. Between these two modes of viewing, the following scientfic objectives will be completed: locate the spin axes; record the dynamic interaction of the system; obtain accurate measurements of the planetary figure and size; study the surfaces' morphology, color, albedo, and surface textures; measure the atmospheric energies via wave propagation modes; help determine atmospheric radiative properties; search for possible auroral interactions caused by magnetospheric interactions; and obtain optical images of Pluto and Charon.

The *Galileo* design is a modification of the *Voyager* and *Mariner 10* designs with the major upgrade being the use of charge-coupled devices (CCDs). This along with other improvements yields an increase in sensitivity of a factor of 100. (Hunten et al., 226) Also because of the smearing problems, software was written to allow careful spacecraft maneuvering while the camera is in use. (Fisk et al., 11) These modifications will eliminate the problems *Voyager* had with image smear. Even though there is very little light at Pluto, *Intrepid* should obtain accurate optical data due to the CCD modification and that *Intrepid* is passing within approximately 20,000 km of the system. Since this instrument is extremely sensitive to movement, it is mounted on the high precision pointing Science Scan Platform (SSP) in order to maximize pointing accuracy. Another benefit of mounting the SSI on the SSP is to guarantee an unobstructed view for the optical equipment.

Ultraviolet Spectrometer

The Ultraviolet Spectrometer is used in the investigation of atmospheric conditions. It is designed to study the atmospheric composition, upper atmospheric atomic and molecular hydrogen, search for ultraviolet emissions from the dark side of the planet to indicate any auroral activity, and examine the cloud and haze structure. (Hunten et al., 233-234) The UVS is a telescope-spectrometer with three detectors attached. It will be located on the SSP so that the spacecraft need not perform any special maneuvers for the pointing requirements of the UVS. It is not a long range scanning device, thus it will begin scanning approximately 24 hours before the encounter date and will remain scanning for an additional 24 hours afterwards. The range of Pluto's atmosphere is unknown and Charon's atmosphere, if it does exist, has not been confirmed. Therefore to ensure that no features will be missed, the UVS will scan the entire distance between the planets.

Although this distance is only 19,400 km, the scanning field will be 20,000 km long. This extra scanning range will reduce the uncertainty of the Plutoian's system's measurements. To provide an accurate reading, the UVS will remain fixed at an angle and allow the spacecraft drift to move the field of vision. At Pluto *Intrepid* will be traveling 11.6 km/sec (see MMPC subsystem), therefore, each scan of 20,000 km will last one half hour. Using this method, almost one hundred scans will be recorded. The Hubble Space Telescope will provide more data about the atmosphere before launching. This would reduce the scanning length involved.

The reasons for selecting the *Mariner Mark II* over the *Voyager* design are the improved wavelength ranges, longer expected lifetime, and the more recent technology incorporated in the instrumentation.

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MISSION MANAGEMENT, PLANNING AND COSTING (MMPC)

Requirements of the Mission

The RFP for this mission has several design limitations that directly affect the decisions and recommendations of the MMPC Subsystem. They are as follows:

1. design an unmanned scientific study to Pluto and Charon
2. optimize cost and simplicity in design tradeoffs
3. launch time between 2000 and 2010
4. design should stress reliability, simplicity, and low cost
5. for cost estimates: assume four S/C built, three flight ready and one for the integrated ground test system

The decision to develop a flyby mission, for example, was a direct result of the consideration of the following RFP requirements: optimize cost and simplicity in design tradeoffs; and stress reliability, simplicity and low cost of the overall mission.

Type of Mission

The first step in the design of an exploratory spacecraft is to determine the type of mission that will best satisfy the mission requirements. The three types of missions which can be flown are: 1) lander, 2) orbiter, and 3) flyby. Although each type of mission has both advantages and disadvantages, it is the responsibility of the MMPC Subsystem to determine which type will best satisfy the objectives and requirements of the mission.

The main advantage of a lander mission is the amount of time the spacecraft is exposed to the planet. As a result of this large encounter time, the largest (and most accurate) quantity of data is obtained. The drawbacks of a lander mission, however, tend to outweigh the advantages. The large spacecraft weight that accompanies a lander mission directly affects several key mission requirements. Both the flight time and the change of velocity (ΔV) are significantly increased as a result of the increased weight. The mission driving factor (low cost), however, ultimately rules out the lander as the spacecraft's mission type.

The orbiter mission is similar to the lander in that the spacecraft is exposed to the planet for a great deal of time. Once again, however, the drawbacks tend to outweigh the advantages. Although an orbiter mission weighs less than a lander mission, the weight of the spacecraft (mainly due to the amount of propellant needed to inject the spacecraft into a planetary orbit) continues to affect the mission requirements. The mission driving factor of maintaining low cost, therefore, rules out an orbiter mission as well.

The flyby mission differs from the other mission types in that the encounter time with the planet is greatly reduced. Although the amount (and accuracy) of information obtained is less than the other mission types, the advantages of using this type of mission tend to outweigh the disadvantages. The small spacecraft weight that is characteristic of flyby missions directly translates into a decrease in total ΔV, flight time, and spacecraft cost. This reduction in spacecraft cost (along with the reduction in ΔV and flight time) is the reason the flyby mission type was ultimately selected over the lander and orbiter missions.

Trajectory Determination

One of the many requirements the MMPC Subsystem is the determination of a trajectory that will best fulfill the mission requirements. This was best carried out using a program provided by SAIC (Scientific Applications International Corporation) entitled MULIMP. Given certain trajectory parameters (such as total mission time, arrival boundary condition, and gravity assist bodies), the MULIMP program would find the minimum ΔV trajectory. Comparing the outputs from several different types of trajectories, one is able to decide upon the trajectory which satisfies the mission requirements best.

The MMPC Subsystem considered numerous trajectories. The most promising of these are: 1) Earth-Mars-Jupiter-Pluto (E-M-J-P), 2) Earth-Jupiter-Pluto (E-J-P), and 3) Earth-Earth-Jupiter-Pluto (E-E-J-P). The E-M-J-P trajectory was considered because of the Mars Initiative currently under developedment. The E-J-P trajectory, as well as the E-E-J-P trajectory, was considered because of Jupiter's large gravity assist potential.

In the selection of the best trajectory, several parameters were placed under consideration. The best combination of launch energy (C_3), flight time, nonlaunch ΔV (which determines the majority of the propellant needed), and launch vehicle compatability (including its cost) will determine the trajectory that will be used. Table 2.1 summarizes the three trajectories under consideration (values were taken from MULIMP outputs for each trajectory). From table 2.1, the trajectory that best fulfilled the mission requirements (especially low cost) was the Earth-Earth-Jupiter-Pluto trajectory.

Table 2.1: Parameters for trajectory consideration

Trajectory	Trajectory Parameters: C3	Flight time	Nonlaunch ΔV	Launch Vehicle Compatible*
E-E-J-P	68.4	15.0 years	1536 m/s	Titan IIID/Centaur ($130-140 million)
E-J-P	91.7	14.3 years	8353 m/s	Titan IV/Centaur ($230 million)
E-M-J-P	27.6	13.2 years	5799 m/s	Titan IV/Centaur ($230 million)

* Least expensive compatible launch vehicle was selected

Trajectory Analysis

The *Intrepid* spacecraft will depart on its Earth-Earth-Jupiter-Pluto trajectory in early February 2002. The parameters of the launch are given in table 2.2 below (values retrieved from the MULIMP program & the Propulsion Subsystem).

Table 2.2: Launch parameters for the *Intrepid* spacecraft

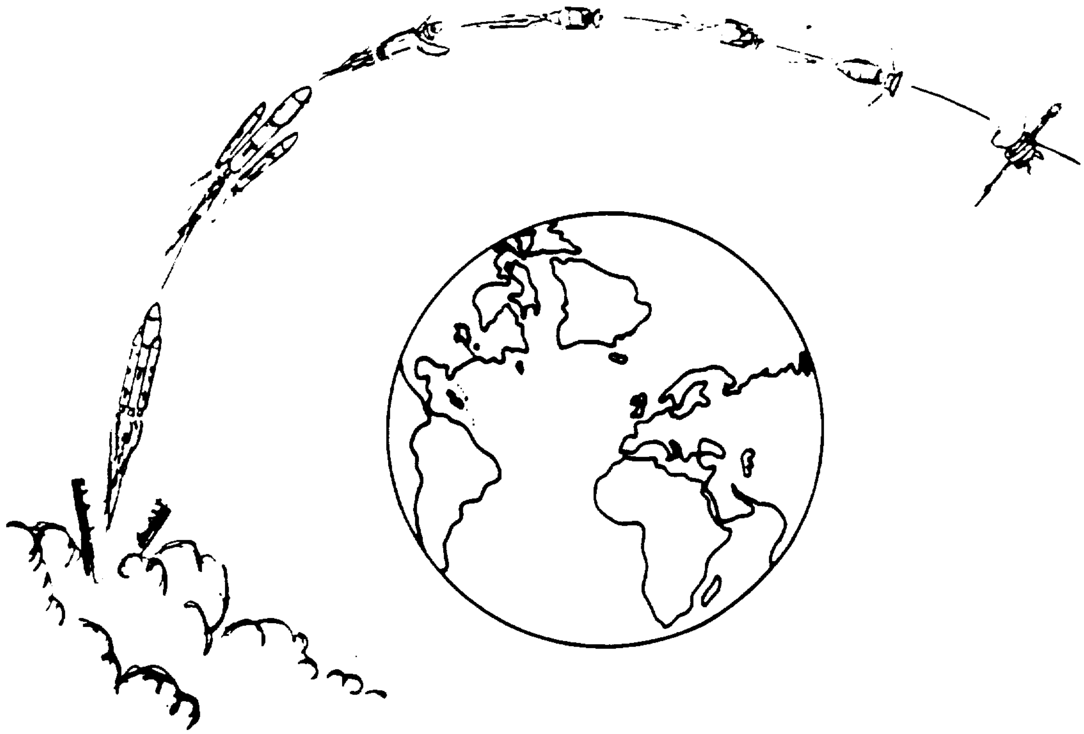
Launch Parameter	Value
Launch Energy	68.4
Launch ΔV	5980 m/s
Launch (wet) Mass	1386 kg
Propellant at Launch	800 kg
Launch Vehicle	Titan IIID/Centaur
Launch Vehicle Cost	$130-140 Million

The spacecraft is mounted in launch configuration on a Titan IIID/Centaur launch vehicle (see Launch Vehicle Subsystem for launch vehicle selection). The launch sequence begins with the firing of Titan's solid rocket motors and first stage. Upon completion of the burn, the rocket motors and first stage will fall off and the second stage will commence firing. The second stage will fall off after the completion of its burn and the Titan's payload fairing will separate, releasing the spacecraft and Centaur upper stage.

The spacecraft and Centaur combination will coast for a length of time for safety and proper trajectory insertion purposes. During this coasting period, the Centaur will be spun in order to stabilize itself and the spacecraft. After coasting for a sufficient period of time, the Centaur upper stage is ignited. After completion of this burn the spacecraft will deploy its booms. The booms are deployed before Centaur separation in order to reduce the possibility of damage to the RTGs and science platform. The Centaur upper stage will then be separated from the spacecraft with pyrotechnics. The launch vehicle adapter will also be separated from the spacecraft at this time (also with pyrotechnics). Finally the spacecraft is stabilized and proper trajectory corrections are performed to place the spacecraft on the desired trajectory. The launch sequence is summarized in figure 2.1.

Approximately 2.9 years after launch, the spacecraft will return to Earth and use it as a gravity assist body. It will perform a ΔV maneuver of 1536 km/s at Earth to place it on the proper trajectory toward Jupiter. The ΔV maneuver along with the gravity assist will hurl the spacecraft toward Jupiter by an additional 3650 m/s. Its closest approach to Earth will be approximately 1.22 Earth radii or 1400 km from the surface. Nearly 1.45

Figure 2.1: *Intrepid* launch sequence



years later, the spacecraft will encounter the Jovian system. However, the spacecraft will rely entirely upon the gravity assist of Jupiter to aid in its velocity since a ΔV maneuver is not needed. The spacecraft will gain an additional 6700 m/s as a result of the assist. The closest approach to Jupiter is approximately 19.8 Jupiter radii or 1,400,000 km from the surface. This distance is maintained in order to minimize the effect of Jupiter's intense radiation and magnetosphere.

Approximately 10.7 years after the flyby of Jupiter (15 years since the launch from Earth), the spacecraft will encounter the Plutoian system. The spacecraft will be traveling approximately 11.6 km/s and it will be close to 34 AU (5.1 billion km) away from Earth. About a week before encounter the spacecraft will direct itself to a point 20,000 km from Pluto's surface (determined by the Scientific Instrumentation Subsystem to be a good distance for studying the system). The thrusters will fire until the spacecraft is on a direct path toward this point. The mission will end approximately 72 days after the Pluto flyby, although extra fuel will allow for additional burns if so desired. A possible post-mission of the spacecraft may include the search for the heliopause (the point at which the Sun's influence ends). The trajectory encounters are summarized in table 2.3.

Table 2.3: Summary of trajectory encounters

	Earth	Earth Assist	Jupiter	Pluto
Time[1] (years)	0	2.9	4.35	15
ΔV^2 (km/s)	5.978	1.536	0	0
ΔV^3 (km/s)	N/A	3.65	6.70	N/A
Dist.[4] (AU)	0	0	4.49	34.01
Dist.[5] (km)	N/A	1400	1,400,000	20,000

1 -- time since launch
2 -- ΔV performed by the spacecraft
3 -- ΔV provided by the gravity assist bodies
4 -- distance from the Earth
5 -- closest approach distance

Trajectory correction maneuvers (TCM) will be required to keep the spacecraft on its trajectory. The TCMs will occur every 2-3 weeks on the average and immediately before and after planetary encounters. As a vehicle travels through space the trajectory is degraded (mainly due to gravity gradients and solar flux). Correction against these degradations is essential to ensure that the spacecraft will remain as close to its trajectory as possible. This is especially important during planetary flybys due to the severe trajectory changes the vehicle endures.

Mission Costing

The process of examining cost allocations is a necessary ingredient in the development of a new spacecraft. By breaking down major subsystems into standard cost categories, MMPC Subsystems can determine top-level cost estimates for these categories as well as for the spacecraft as a whole.

The process of determining system cost estimates begins with the calculation of direct and recurring labor hours (DLH and RLH respectively). A summary of the equations involved are presented in Appendix A2.1. Using these results and the conversion factors from Appendix A2.2, the recurring (RC) and non-recurring (NRC) cost estimates for the hardware-related categories can be calculated using the following equations:

RC = RLH (labor hours to labor cost)(labor cost to total cost)

NRC = (DLH - RLH)(labor hours to labor cost)(labor cost to total cost)

These values are then used to determine a total cost estimate (TC) for each hardware-related category based on the SAI Planetary Program Cost Model given in Appendix A2.3 (the X values used to determine Z were supplied by the subsystems). The total cost estimates for the functional support-related categories, on the other hand, are calculated using the following equation:

TC = DLH(labor hours to labor cost)(labor cost to total cost)

The total cost estimate can now be calculated by summing the following category cost estimates:

- Development Project - Flight Hardware
- Development Project - Support Functions
- Flight Project

This total added to the cost of the launch vehicle used will result in the spacecraft's top-level cost estimate.

Intrepid's top-level cost estimate using the process described is $1032 million (based on the cost of building 4 spacecrafts). The category cost breakdown is summarized in Appendix A2.4.

Appendix A2.1: Summary of Cost Model Algorithms (Koepke)

<u>Development Project - Flight Hardware</u>

<u>Structure & Devices</u>

$$DLH = 1.626\,(N{*}M)^{0.9046}$$

$$RLH = 1.399\,(N{*}M)^{0.7445}$$

<u>Thermal Control, Cabling & Pyrotechnics</u>

$$DLH = \exp\{4.2702 + 0.00608\,N{*}M)$$

$$RLH = 3.731\,(N{*}M)^{0.6082}$$

<u>Propulsion</u>

$$DLH = 56.1878\,(N{*}M)^{0.4166}$$

$$RLH = 1.0\,(N{*}M)^{0.9011}$$

<u>Attitude & Articulation Control</u>

$$DLH = 21.328\,(N{*}M)^{0.7230}$$

$$RLH = 1.932\,(N{*}M)$$

<u>Telecommunications</u>

$$DLH = 4.471\,(N{*}M)^{1.1306}$$

$$RLH = 1.626\,(N{*}M)^{1.1885}$$

<u>Antennas</u>

$$DLH = 6.093\,(N{*}M)^{1.1348}$$

$$RLH = 3.339\,(N{*}M)$$

<u>Command & Data Handling</u>

$$DLH = \exp\{4.2605 + 0.02414\,N{*}M)$$

$$RLH = \exp\{2.8679 + 0.02726\,N{*}M)$$

RTG Power

$$DLH = 65.300 (N*M)^{0.3554}$$

$$RLH = 7.88 (N*M)^{0.7150}$$

Landing Radar/Altimeter

$$DLH = 11.409 (N*M)^{0.9579}$$

$$RLH = 1.2227 (N*M)^{1.2367}$$

Line-Scan Imaging

$$DLH = 10.069 (N*M)^{1.2570}$$

$$RLH = 1.989 (N*M)^{1.4089}$$

Particle & Field Instruments

$$DLH = 25.948 (N*M)^{0.7215}$$

$$RLH = 0.790 (N*M)^{1.3976}$$

Remote Sensing Instruments

$$DLH = 25.948 (N*M)^{0.5990}$$

$$RLH = 0.790 (N*M)^{0.8393}$$

Development Project - Support Functions

System Support & Ground Equipment

$$DLH = 0.36172 (\Sigma DLH_{\mathrm{hardware}})^{0.9815}$$

Launch + 30 Days Operations & Ground Software

$$DLH = 0.09808 (\Sigma DLH_{\mathrm{hardware}})$$

Imaging Data Development

$$DLH = 0.00124 (\text{Pixels-Per-Line})^{1.629}$$

Science Data Development

$$DLH = 27.836 (\text{non-imaging science mass})^{0.3389}$$

Program Management/MA&E

$$DLH = 0.10097\ (\Sigma DLH_{\text{all categories}})^{0.9670}$$

Flight Project

Flight Operations

$$DLH = (\Sigma DLH_{\text{hardware}}/3100)^{0.6}(10.7\ MD + 27.0\ ED)$$

Data Analysis

$$DLH = 0.425\ (\text{DLH Flight Operations})$$

N	-	Number of spacecrafts
M	-	Mass in kg
DLH	-	Direct labor hours in 1000 hours
RLH	-	Recurring labor hours in 1000 hours
MD	-	Mission duration in months
ED	-	Encounter duration in months

Appendix A2.2: Labor/Cost Conversion Factors (Koepke)

Cost Category	Labor Hours to Labor Cost[1] (Feb'90 dollars/manhour)	Labor Cost to Total Cost
Development Project		
Structure & Devices	22.04	3.303
Thermal Control, Cabling, & Pyrotechnics	21.64	3.317
Propulsion	22.23	3.616
Att. & Artic. Control	22.42	3.347
Telecommunications	21.07	3.352
Antennas	21.01	3.466
Command & Data Handling	20.41	3.163
RTG Power	20.06	3.177
Landing Radar/Altimeter	21.26	3.158
Line-Scan Imaging	22.29	3.604
Particle & Field Instruments	22.40	3.395
Remote Sensing Instruments	22.46	3.286
System Support & Ground Eq.	22.25	3.076
Launch + 30 Days Ops & Gr. S/W	22.59	3.214
Image Data Development	24.17	3.130
Science Data Development	26.91	3.987
Program Management/MA&E	24.40	2.685
Flight Project		
Flight Operations	22.02	3.247
Data Analysis	22.02	3.425

1 – Feb'90 dollars = 2.109(FY77 dollars) (US Dept of Commerce 462 & US Dept of Commerce 6)

Appendix A2.3: SAI Planetary Cost Model (Koepke)

Inheritance Class Categories

- Class One: Off-the-Shelf/Block Buy
- Class Two: Exact Repeat of Subsystem
- Class Three: Minor Modifications of Subsystem
- Class Four: Major Modifications of Subsystem
- Class Five: New Subsystem

Cost Reduction Algorithm by Inheritance Classes

Let X_1 = Percent of Subsystem Off-the-Shelf
X_2 = Percent of Subsystem Exact Repeat
X_3 = Percent of Subsystem Minor Modifications
X_4 = Percent of Subsystem Major Modifications
X_5 = Percent of Subsystem New Design

Thus $X_1 + X_2 + X_3 + X_4 + X_4 + X_5 = 100\%$ of Subsystem Mass

NRC = Non-recurring cost estimate (without inheritance)
RC = Recurring cost estimate
TC = Total cost estimate (including inheritance effects)
Z = Percent cost reduction

If $Z = 1.0X_1 + 0.8X_2 + 0.25X_3 + 0.05X_4 + 0.0X_5$

Then $TC = (100\% - Z)\, NRC + RC$

Appendix A2.4: *Intrepid* cost estimates

Cost Category	Cost (in millions of dollars)
Structure & Devices	$22.85
Thermal Control, Cabling & Pyro.	$10.13
Propulsion	$24.53
Attitude & Articulation Control	$32.55
Telecommunications	$35.68
Antennas	$11.18
Command & Data Handling	$56.40
RTG Power	$25.92
Landing Radar/Altimeter	$0.82
Line-Scan Imaging	$142.21
Particle & Field Instruments	$33.36
Remote Sensing Instruments	$0.72
System Support & Ground Equip.	$215.29
Launch + 30 Days Op & Ground S/W	$73.46
Image Data Development	$5.03
Science Data Development	$9.41
Program Management/MA&E	$70.52
Flight Operations	$83.92
Data Analysis	$37.62
Subtotal	$891.6
Launch Vehicle	$140.0
Top-level cost estimate	**$1031.6**

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POWER AND PROPULSION SUBSYSTEM (PPS)

Propulsion Subsystem

The propulsion subsystem must provide the spacecraft with the ability to make ΔV as well as course correction maneuvers and attitude control adjustments. This subsystem may be divided further into three categories: propellant and pressurant, propulsion feed system, and thrusters. The breakdown of masses for the entire propulsion subsystem can be found in Appendix B.

Propellant and Pressurant

For the spacecraft, two types of chemical propellants are considered. These are monopropellants and bipropellants. Solid fuels are immediately ruled out because of their nonexistent stop-restart capabilities which are essential for a Plutoian mission. Furthermore, cryogenic fuels are neglected because of their poor storage qualities over long durations, again a requirement for a mission to Pluto.

A monopropellant system will be used on the spacecraft because of the advantages it has over a bipropellant system. First, monopropellants require less complicated propellant feed systems. This results in mass savings, and consequently cost savings, due to the reduced tankage and valving required. Also, many monopropellants are storable for long durations; bipropellants are not. Bipropellants do however have a distinct advantage over monopropellants in specific impulse. This advantage will reduce the amount of propellant required for a given mission. However, a driving factor in choosing a propulsion system is simplicity and reliability. Overall, a monopropellant system better fits the RFP requirements than a bipropellant system for a long duration mission.

The spacecraft fuel selected is monopropellant hydrazine (N_2H_2), with a specific impulse of 225 seconds. Not only does hydrazine provide the lowest cost propulsion system (Koepke), it is simple, reliable, and has been used on a number of previous spacecraft. Additionally, hydrazine has been shown to be space storable for extended periods of time (greater than 12 years) and has stop-restart capabilities (Koepke). Furthermore, thrust levels from as low as 0.2 N to moderate levels of approximately 2500 N have been demonstrated using hydrazine (Koepke).

Despite the advantages of using hydrazine for the spacecraft's propellant, disadvantages also exist. Drawbacks to hydrazine include its moderate plume contamination of the spacecraft and its toxicity. The problem of plume contamination is easily remedied by the use of shielding and strategic placement of sensitive instruments. The dangers of handling hydrazine have been greatly reduced because of the familiarity gained with the fuel from previous use in other spacecraft.

Estimates based on trajectory analysis performed by the MMPC (Mission Management, Planning and Costing) subsection indicate that the total ΔV propellant required for the mission is 677 kg, while the total trajectory correction and attitude correction propellant is 65 kg. A contingency of 58 kg of extra propellant was included in order to account for 2.0% unusable fuel (14.84 kg) and an error margin for propellant consumption. Thus the resulting total propellant mass is 800 kg.

The pressurant for the propulsion system is used to force the propellant into the propellant feed lines. Two types of pressurants were considered for the spacecraft: helium and gaseous nitrogen. Although each is reliable, cheap, and proven, helium was selected due to its mass savings: 2.24 kg as opposed to 15.63 kg for GN_2. Additionally its lower liquifying temperature is better because fewer heaters are required.

Propulsion Feed System

The prime requirement for the propulsion feed system is to supply the thrusters with propellant. This system can be further reduced into: tankage, valving, filtering, and tubing. A schematic of the propulsion feed system is given in figure 3.1.

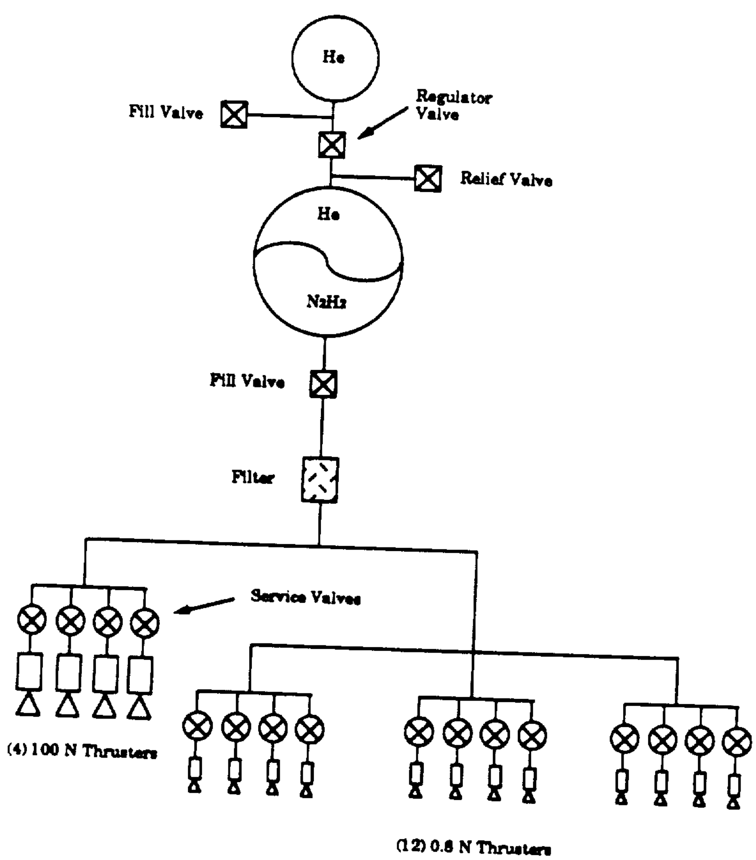

Figure 3.1. Propulsion System Configuration

Both the main trajectory and attitude correction fuel will be contained within one main tank. This provides for a simpler and lighter system although redundancy is sacrificed. However, research found no evidence of spacecraft failures due to faulty propellant feed systems thus the design is considered adequate.

A regulated pressurant feed system was selected over a blowdown system. For a blowdown system, the tank inlet pressure decreases over the lifetime of the mission resulting in a reduction of thruster performance. A reduction in performance decreases the specific impulse of the system (figure 3.2).

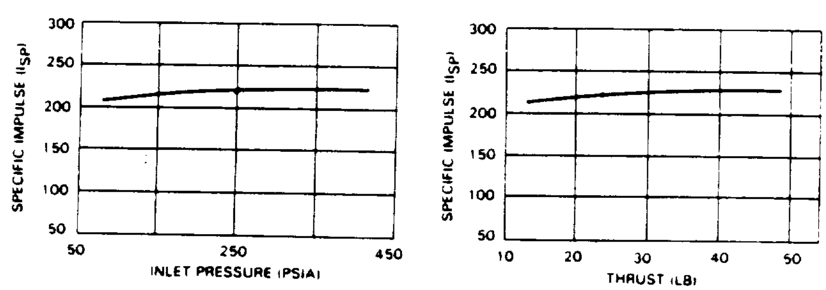

Figure 3.2. Inlet Pressure versus Specific Impulse (TRW)

This loss in performance would require the use of more propellant which would negate any mass savings achieved by using less valving and tubing.

Two spherical, titanium tanks were designed for propellant and pressurant storage. A spherical design was chosen because it provides the greatest volume to surface area ratio, it can withstand large stresses using a minimal amount of material, aids in the stability of the spacecraft, has been flight tested, and may be an off-the-shelf item which is currently available.

The propellant tank is approximately 1.2 meters in diameter and is designed for a pressure of 250 psi (1.7236 MPa) with a safety factor of 1.5. The design pressure was chosen based on desired thruster performance plots (figure 3.2). The tank can hold 800 kg of hydrazine and will use a spherical bladder system. This will help keep the center of mass of the tank from moving as fuel is drained. A titanium alloy, Ti-6Al4V, is chosen for the tank material because of its resistance to damage from the hydrazine fuel, its moderate density (4400 kg/m^3), its high yield strength (850 MPa), and its proven reliability (Ashby, 10). Although research has not indicated that a tank as previously defined already exists, there is confidence that such a tank may already be available.

The pressurant tank has a diameter of 0.50 meters and is designed for a pressure of 3000 psi (2.068 MPa) again with a safety factor of 1.5. This tank will hold 2.24 kg of helium pressurant. The same titanium alloy as used for the propellant tank will be used here resulting in a tank mass of 16.15 kg. Again, this type of tank may already be available.

Aluminum alloys, particularly the Al 7000 series, were considered for the pressurant tank material giving a mass of approximately 18.39 kg, however material costs would be reduced by $95.54 (1988 dollars) over the titanium tank (Ashby, 10). This dollar savings was not considered sufficient in order to account for the much larger specific cost expected at launch.

The valves, filter, and tubing mass required for the system was estimated based on the *Voyager* spacecraft (Mangano). All the material used here will easily be found as off-the-shelf hardware which has been previously flight tested.

Thrusters

Both the main propulsion system and attitude control system require the selection of thrusters. While the attitude system uses low-thrust thrusters in order to make minor attitude adjustments, the primary propulsion system will use larger thrusters to provide the necessary ΔV maneuvers.

For the main thrusters, four 100 N thrusters were selected providing a total of 400 N. A four thruster configuration provides redundancy whereas one thruster does not. The MRE-50 monopropellant thruster manufactured by TRW (TRW) is an off-the-shelf, flight proven thruster capable of meeting the design requirements. Care was taken sizing the thrusters in order not to perturb the spacecraft structure during thrusting (maximum accelerations of approximately 0.06g are expected). Additionally, the duration of the ΔV burns must be within the design limits of the thrusters. The ΔV maneuver requiring the longest duration burn will occur at the spacecraft's earth gravity assist. This burn will last approximately 3736 seconds which can be accomplished by pulsing the thruster nine and one-thirds times with each pulse lasting 400 seconds. Since the selection of thrusters is not final, other types may be substituted to better fit the requirements. However, the basic design should remain the same.

For the attitude correction maneuvers, the AAC (Attitude and Articulation Control) subsystem has determined that twelve 0.2 N thrusters similar to those used on *Voyager*, will be adequate. Again this is an off-the-shelf item which has been flight tested and proven reliable on earlier spacecraft.

Further Comments

An additional feature to the propulsion subsystem design includes autonomous control. Since the time for one way communication with the spacecraft at Pluto will be on the order of four hours, pre-programmed ΔV maneuvers based on the spacecraft's location (determined from attitude control) must be included. Furthermore, the spacecraft will be passing Pluto at a relative speed of approximately 11.6 km/sec so manual trajectory corrections will be impractical.

Placement of the thrusters is also of importance. Four 0.2 N thrusters will be placed on the outer rim of the bus ninety degrees apart. Orientation has been determined by the AAC subsystem. The remaining eight 0.2 N thrusters and four 100 N thrusters are located on four separate pods underneath the spacecraft (opposite the high gain antenna). Each pod has one 100 N thruster and two 0.2 N thrusters. These pods will be arranged in a square pattern 1.5 meters apart. This location and orientation was chosen in order to reduce plume contamination to sensitive instruments, provide control in the event of a thruster failure, and avoid interference with the adapter structure. Shielding will also be used to help reduce plume contamination.

Since the overall design of the propulsion subsystem is general and simple, nothing should preclude it from performing other missions. However, this design may be better suited for long duration missions in order to help justify the use of a monopropellant. Propellant requirements for other missions will dictate whether this spacecraft design would be feasible.

Any changes to the propulsion subsystem would affect the structure and AAC subsystems primarily. Specifically, changing the type of propellant, thruster system, or tankage would have the largest impact. Care should be taken if such changes are necessary.

The lifetime of the spacecraft is indirectly affected by the propulsion subsystem. If a major failure occurs, the spacecraft would be unable to correctly orient itself and would essentially become "paralyzed". Of key importance is the thruster lifetime. It is particularly dependant on the frequency and duration of burns so these must be kept to a minimum to ensure an appropriate spacecraft lifetime.

Power Subsystem

The power subsystem may be divided into the following categories: power generation, energy storage, and power conditioning. A schematic of the power subsystem is shown in figure 3.3 while a breakdown of power requirements for each subsystem is given in table 3.1. Furthermore, Appendix B provides a listing of the masses for the power subsystem.

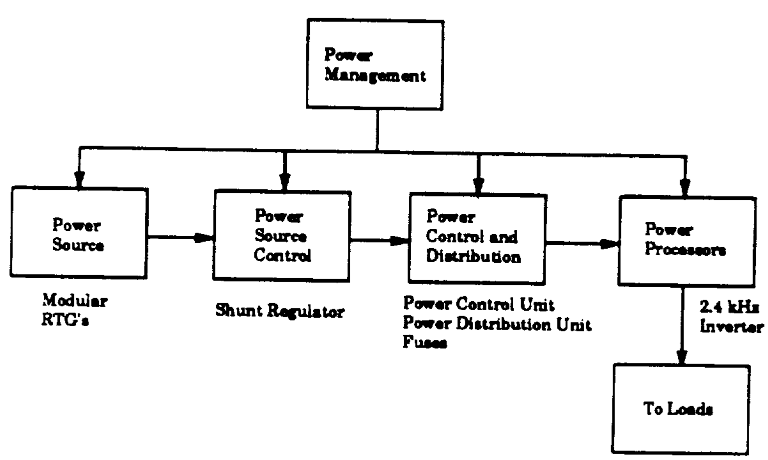

Figure 3.3. Schematic of Power Subsystem

Subsystem	Cruise	Pre-Fire	Maneuver	Science	Comm.
Telecom	10	10	10	65	65
CCC	15	15	20	25	25
TCS	20	20	20	20	20
AACS	90	90	90	90	90
SI	19	19	19	56	56
TOTALS	154	154	159	256	256

Table 3.1. Breakdown of Power Requirements [watts]

Power Generation

Power generation is achieved using a modular radioisotope thermoelectric generator (MRTG) design. RTG's provide the only feasible source of power for the distances and trip times that the spacecraft will encounter on a mission to Pluto. In addition to providing essential redundancy, a modular design also offers a weight savings over a non-modular design because of the ability to tailor the power source to the power requirements. Furthermore, if the power requirements are changed, only adding or subtracting RTG "slices" is required for meeting the new power demands. This is especially important if the spacecraft is to have multi-mission capability. Using RTG's do however have serious drawbacks. These include thermocouple degradation and public concern over the RTG's radioactivity.

For the spacecraft's mission to Pluto, it is determined that the maximum required power is 256 W (required at Pluto). In order to account for the degradation of the thermocouples and the half-life of the plutonium-238 fuel, an extra 75 W will be required at launch (degradation of 5 W/yr (Fisk, 6) over a trip time of 15 years). Thus, the total power required at launch is 357 W.

Specifically, the MRTG design here is comprised of fifteen individual "slices" (figure 3.4) each producing 24 W of power at 28 V with a specific power of 10.57 W/kg (Schock, 341). This produces a total output of 360 W and a MRTG mass of 34 kg. It should be noted that although the power contingency margin appears to be narrow, each subsystem's requirements includes an additional contingency.

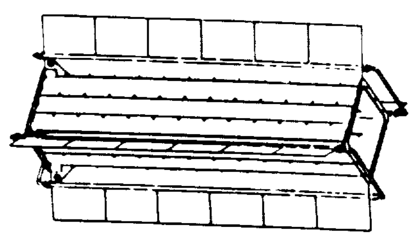

Figure 3.4. Modular RTG Design (Schock, 338)

Concerns on the safety of the RTG's during launch is also of prime importance in the design. However, since there is extensive testing done on RTG containment in the event of a launch failure, problems are not likely to occur.

Energy Storage

The selection of an energy storage system is an important consideration. Two devices are considered: batteries and a discharge controller. Although both provide excellent energy storage qualities, it was decided that an energy storage device would not be incorporated into the spacecraft design. Because the spacecraft will generate more power than is required during all phases of the mission, any power shortage may draw on the contingency power. Furthermore, the power management system will reduce power to those components which are considered less essential in the event that more power is required than that available from the contingency. This allows a weight savings by not including extra components and keeps the design simple.

Power Conditioning

Power conditioning entails the following: power source control, power control and distribution, power processing, and power management. All the components used in this subsection are off-the-shelf items that are taken from the *Voyager* design (Mangano).

Power source control is achieved through the use of a regulated power system. The primary component for this system is a shunt regulator. This design was chosen because it provides a constant voltage supply and has been proven on previous spacecraft. A disadvantage to a regulated system is that it requires more mass than an unregulated system. However, this added mass can be justified because a regulated system is a proven system.

The power control and distribution component consists of a power control unit, a power distribution unit, and fuses. All three prevent power overloads which may damage the spacecraft's electronics. Additionally, the power control and distribution units control the power to various subsystems during maximum power periods.

Power processing consists of inverters for changing the RTG's DC generated power to AC power. The primary unit used here is a 2.4 kHz inverter.

The power management system has control over the entire power system. Power cycling, which helps reduce the chance of power shortages and extends the lifetime of the various electronic components, is controlled through this system. Also included in the power management system are parallel connections and redundancy which reduce the chance of single point failures. Lastly, autonomous control of the power management system will be provided by the command subsystem in case of a loss of communications with the ground stations, or when communications are too long to permit appropriate control.

Further Comments

The lifetime of the entire spacecraft is dependant primarily on the lifetime of the RTG's thermocouples. As these degrade, the spacecraft's power supply is reduced. The power management system is designed to cycle the available power to the various subsystems in order to alleviate the problem of reduced power and extend the useful life of the spacecraft.

Changing the power subsystem primarily affects the scientific instrumentation and communications subsystems. Of particular concern are changes to the RTG's. For this design, tailoring of the power supply is easily accomplished by using the modular RTG's. This helps the spacecraft adapt to various mission designs.

Launch Vehicle Subsystem

The launch vehicle has the responsibility of giving the spacecraft enough energy to begin its mission on the proper trajectory. The following are requirements for choosing the spacecraft's launch vehicle:

1) it must provide an adequate C_3 (launch energy) for the given spacecraft launch mass
2) it must adequately fit the spacecraft and any upper stages within the launch vehicle
3) it must be as inexpensive as possible
4) it must be reliable.

Various launch vehicles were considered. The two that best fit the above requirements are the Titan IIID / Centaur and the Titan IV / Centaur. The corresponding payload dimensions and cost are given in table 3.2 while launch energies are given in figure 3.5. Keeping the previous requirements in mind, the Titan IIID/Centaur was selected as the launch vehicle (figure 3.6).

Launch Vehicle	Payload Dimensions	Cost (1989 dollars)
Titan IIID/Centaur	3.65 m diam, 10.7 m length	$130-140 million
Titan IV/Centaur	4.33 m diam, 8.93 m length	$240 million

Table 3.2. Launch Vehicle Information+

+ The Titan IIID/Centaur payload dimensions are in fact Titan III Commercial specifications (Gizinski, 4). The Titan IIID/Centaur information was not available. The Titan III Commercial dimensions (for a dedicated launch) are expected to be smaller than those for the Titan IIID. Titan IV information (Aviation Week, 163).

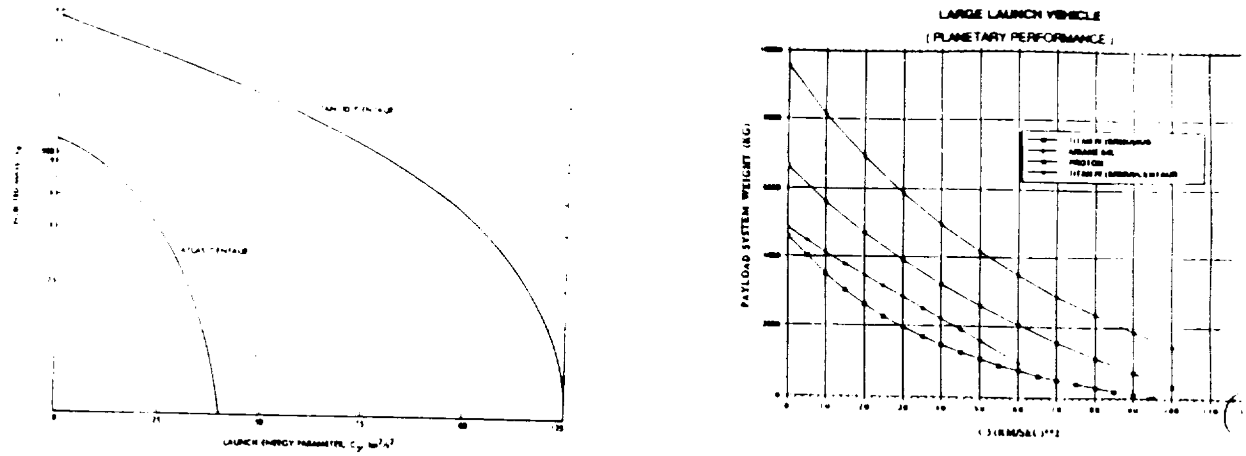

Figure 3.5. Titan IIID/Centaur and Titan IV/Centaur Launch Energies versus Launch Mass (Atkins, 43 and JPL Mission Group)

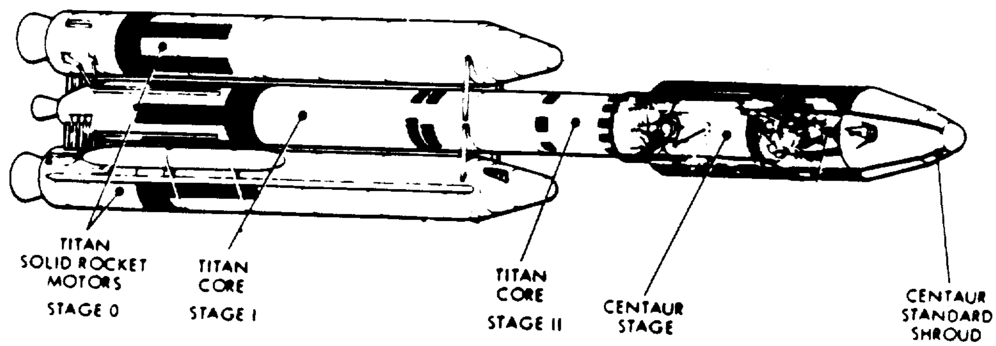

Figure 3.6. Titan IIID/Centaur*

Both launch vehicles provide the spacecraft (mass at launch = 1386 kg) with the required launch energy of 68.4 km^2/sec^2. Additionally, the payload fairings for both adequately fit the spacecraft in its launch configuration with the Centaur upper stage. Other features similar for both launch vehicles include their outstanding success rate (94% reliability) and launch site (Cape Canaveral) (Gunn). The cost of the Titan IV / Centaur however was nearly twice that of the Titan IIID / Centaur. This factor is the deciding reason for choosing the Titan IIID / Centaur.

* Figure 3.6 is actually that of a Titan IIIE/Centaur (Heacock, 214). The Titan IIID/Centaur looks similar to the Titan IIIE/Centaur.

Further Comments

The choice of using an expendable over a reusable launch vehicle is driven by the requirements of low cost and simplicity. Any increase in the spacecraft's mass, however, will not allow the use of the Titan IIID due to launch energy constraints. Some other type of launch vehicle, such as the Titan IV / Centaur must be used.

Any changes to the launch vehicle selection should not greatly affect the design of the spacecraft as long as payload fairing and launch load requirements are met. However, if a more powerful launch vehicle (higher C_3 capacity) is selected, changes in the planetary trajectory may be allowed perhaps reducing the flight time of the mission.

Appendix A.3 - Equations

Propulsion Subsystem

- Equations for propellant mass and volume required:

$$m_{prop}|_{\Delta VJ} = (m_{dryS/C} + 1/3m|_{\Delta VTCM})\,(\exp(^{\Delta VJ}/_{Isp}) - 1)$$

$$m_{prop}|_{\Delta VE} = (m_{dryS/C} + 2/3m|_{\Delta VTCM} + m_{prop}|_{\Delta VJ})\,(\exp(^{\Delta VE}/_{Isp}) - 1)$$

$$m_{proptotal} = m_{prop}|_{\Delta VJ} + m_{prop}|_{\Delta VE} + m|_{\Delta VTCM}$$

Volume of propellant = mass propellant / density of propellant

For a spherical tank: $V_{tank} = 4/3*PI*r^3$

- Equation for pressurant mass and volume required:

m_{press} = (atomic weight of pressurant) * (number of moles of pressurant)

# of moles = $(P_{tank}V_{tank})/(R_{press}T_{tank})$

- Propellant and pressurant tank sizing equations

$m_{spherical\ tank}$ = (safety factor)*(2PI)*$P_{tank}r^3$(tank material density/yield stress)

- Thruster sizing

acceleration = $^{m}{}_{S/C}/_{thrust}$

$t_{burn} = ^{(DmIspg)}/_{thrust}$

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STRUCTURE SUBSYSTEM (SS)

The structure subsystem (SS) is further divided into three smaller subsections: structural design (SDS), thermal control (TCS), and materials (MS). The driving requirements of the SS are low cost (i.e. low weight), simplicity, and reliability.

Structural Design Subsection

The SDS consists of the layout of all components, launch vehicle compatibility, launch load requirements, and on-orbit assembly. Adapting previously flight tested designs offers greater reliability and less redesign work. *Pioneer 10 & 11* provided invaluable data to the *Voyager* program in eliminating concerns with the asteroid belt hazard, defining radiation environments at Jupiter, and demonstrating that a spacecraft (S/C) can survive in space for extended trip times. Similarly *Pioneer*, *Voyager*, *Galileo*, and the *Mariner Mark II* (MMII) program give *Intrepid* invaluable data. Therefore to reduce cost, the *Intrepid* design uses existing technology whenever possible.

Figure 4.1 shows the primary subsystem components. Numerous iterations were performed to satisfy the structural requirements of the complex payload against competing requirements of the entire system.

Science	Power/Propulsion	Communication	Attitude Control
Imaging camera	RTG	High Gain Antenna	Platform
Magnetometer	Propulsion Tank	Low Gain Antenna	Computer
Plasma Detector	Various Thrusters	Computer	Various Thrusters
UV-Spectrometer			
Ultra Stable Oscillator			
Cosmic Ray Detector			
Particle Detector			

Figure 4.1 Subsystem Necessities

Item	Problem	Solution
All Items	Keep Cost Low	Off Shelf Hardware Use Existing Technology
Magnetometer	Interference	Maximize Distance From S/C Bus
RTG	Radiation/Thermal Inertial	Mount On Boom, Shield Balance Science
Science	Provide Unobstructed View Inertial	Mount On Boom Balance RTG

Figure 4.2 Structural Design Considerations

Figure 4.2 shows the structural design considerations of various components. During the structural design phase, these considerations were taken into account, as well as the placement of the center of mass (CM) and the component inertial contributions to the spacecraft (S/C) . The

CM placement and inertial contributions are also very important considerations for the attitude/articulation control subsystem (AACS). These values were calculated and optimized using the INERT program (INERT, AAE 241).

Intrepid *Spacecraft Design*

The configuration of the *Intrepid* S/C (without thermal blankets) is shown in Figures 4.3 - 4.5. The basic structural component of the S/C is a ten bay metallic bus which houses the electronics. A trade study for the MMII program was performed examining different bus structure configurations and the number of electronic bays. The study concluded that the bay approach is more mass efficient(Draper, 7). Also, for the bay packaging approach, all required testing fixtures, procedures, and experience presently exist, reducing risk of developing and cost . Thus, the bay approach was chosen.

Inside the bays, the electronics are packaged on flat plate (sandwiched aluminum honeycomb) sub-chassis which, when installed vertically, become integral structural elements providing strength to the spacecraft (Heacock, 217). This packaging allows for more desirable CM properties. Another consideration is the use of a rectangular group of bays or a toroid design. Rectangular groups have the advantage of having a small end so thermal shielding on one face of the mission can be quite small (Draper, 7). However, the MMII program found that the use of rectangular bays leads to a larger structure than when using a toroid. A reduced mass plus experience with a toroidal bus design are the driving factors for selecting the toriod design.

Figure 4.3 *Intrepid* Spacecraft Flight Configuration (x-z plane)

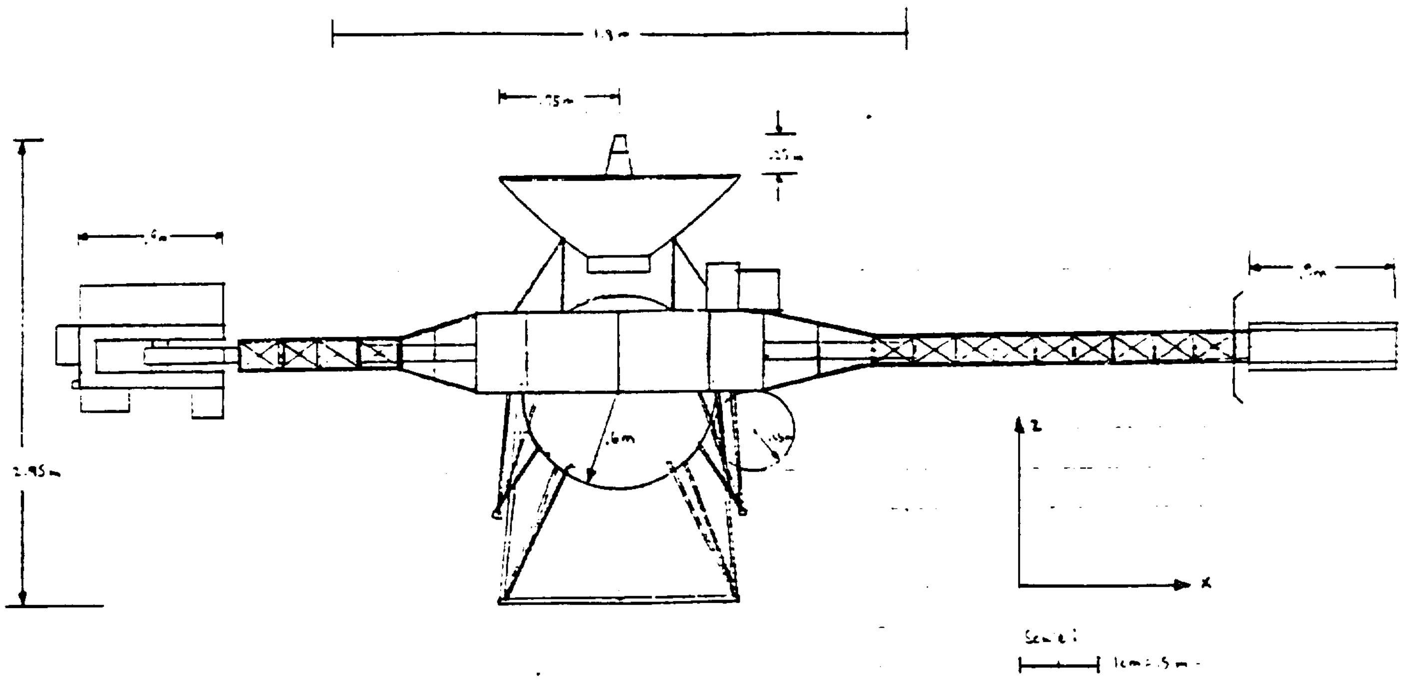


Figure 4.4 *Intrepid* Spacecraft Flight Configuration (x-y plane)

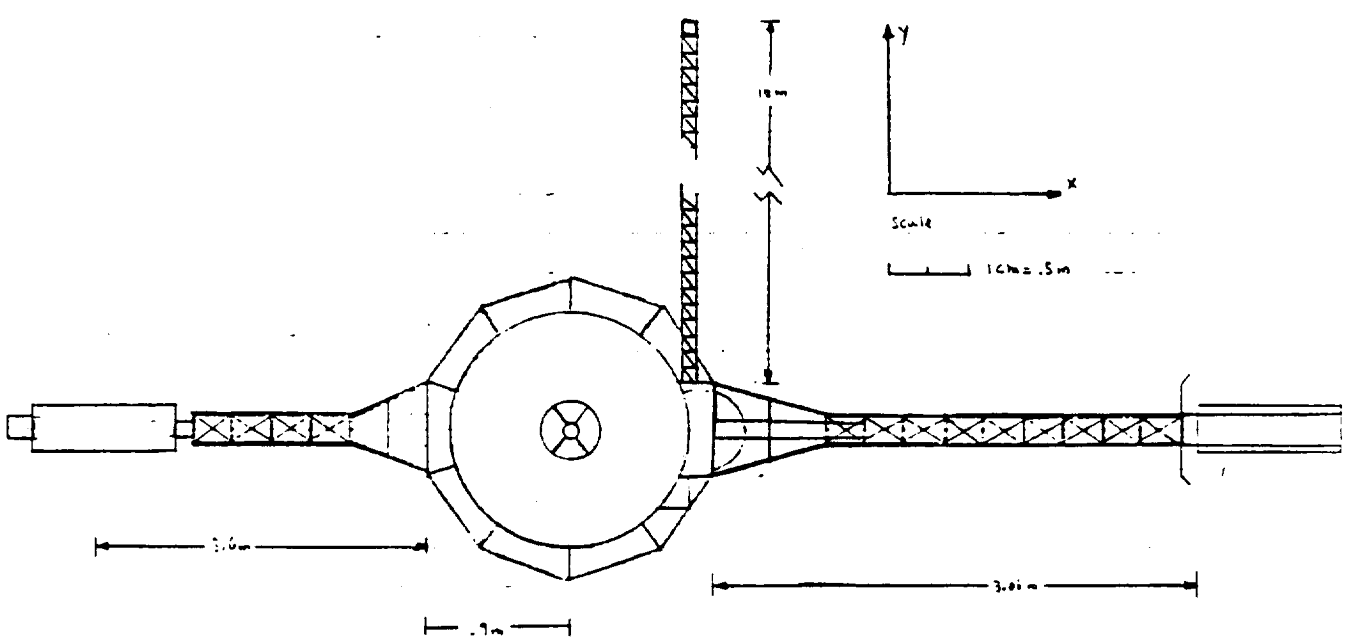
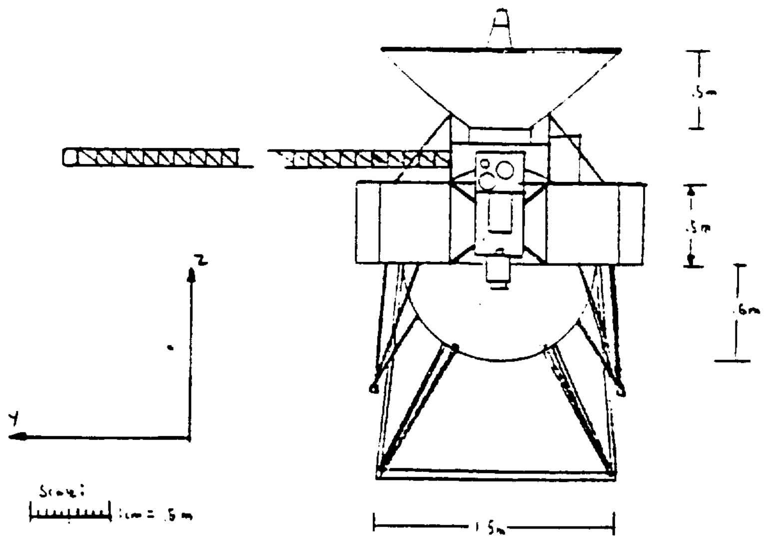


Figure 4.5 Intrepid Spacecraft Flight Configuartion (y-z plane).



The last bus design consideration is the number of bays required. *Galileo* used eight bays, but had several packages of electronics outside the bus. Due to the trip duration (15 years), a design which houses electronics on the outside of the bus is not prefered. Therefore, a ten bay bus design is selected, similar to the Voyager design.

Power/propulsion subsystem (PPS) selected the use of modular Radioisotope Thermal Generators (RTG)s. The RTG placement is governed by their gamma ray and neutron radiation and the need to balance the mass of the science platform. Science instrumentation subsystem (SI) selected various instruments which are mounted on a scan platform, providing mass balancing of the RTGs about the S/C CM. Furthermore, the field of view for the sensors is maximized. To reduce the effects of the RTG radiation, the science instruments are mounted on the science boom 180° from the RTG boom. An added benefit of the 180° design is that the main structure of the S/C will help shield the science from exposure to radiation. Another device reducing the radiation exposure is a radiation shield on the RTG boom. These techniques help reduce the length of the booms needed to adequately separate the RTGs from the science.

Reducing the length of the booms also helps alleviate vibrational problems. The science and RTG booms are 2.1 and 3.5 meters long, respectively. The use of dampers will also reduce vibration problems.

Any electrical current flowing within a S/C can produce a magnetic field sufficient to distort measurements of weak interplanetary fields. Similar to the *Voyager* S/C, the low field magnetometers are mounted on a 10 meter Astro-Mast boom in order to minimize interference from the S/C's magnetic fields. The magnetometer is directed 90° from the other booms. Additional strategies for reducing magnetic interference include proper selection of materials and strategic placement of all necessary electrical apparatus.

The remaining components: high gain antenna (HGA), low gain antenna (LGA), and propulsion tank are positioned based on previous S/C and CM placement requirements.

Launch Vehicle Compatibility

Once the total weight of the S/C was fixed (see Appendix B for calculations) the PPS selected the Titan III-D /Centaur launch vehicle for launch.

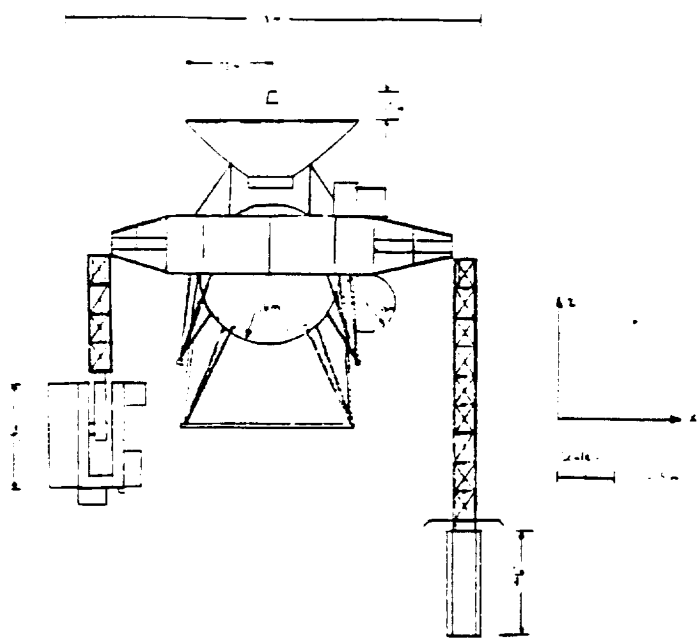


Figure 4.7 *Intrepid* Spacecraft in Launch Configuration

Figure 4.7 shows the *Intrepid* S/C in its launch configuration, excluding thermal blankets. The Titan III-D has a payload bay diameter of 3.65 m, a payload length of 10.7 m. The upper stage Centaur is 5.9 m long with a resulting maximum S/C length of 4.8 m (Gizinski III, 4). As can be seen in Figure 4.7 the *Intrepid* S/C (in launch configuration) fits within the allotted dimensions.

Launch Vehicle Survivability

In the launch configuration, the primary S/C structure (bus and propellant tank) are connected to an upper stage assembly (USA) adapter. The S/C and supporting adapter will experience maximum accelerations of 4g's during launch, producing a maximum of 52,960 N of force. Since the adapter's graphite/epoxy truss elements can withstand 59,270.4 N of axial compression, the S/C meets launch load requirements (see appendix A.4.2). During launch, the adapter structure will be used to support the large mass of the RTG and science platform booms. This is required to prevent damage from launch phase acceleration, vibration, shock, and acoustic environments (Heacock, 216).

On-Orbit assembly

The mission's on-orbit assembly consists of deploying the RTG, Science, and magnetometer booms. The RTG and science booms are deployed by actuators, the adapter is separated by pyrotechnics, and the magnetometer boom is deployed.

The magnetometer and boom are stored in a canister that is .23 m in diameter and .66 m long. The boom is a triangular truss made of fiberglass longitudinal members held in place by fiberglass triangles. These are stiffened with tensioned, collapsible diagonal filaments. The boom is stowed by twisting the entire structure so that the diagonal filaments interlace and the triangles are nearly in contact with each other. This puts a considerable elastic force on the assembly. When the canister is opened by a pyrotechnical device, the boom expands at a controlled speed to prevent the boom from popping out with destructive violence (NASA, 20-21).

Thermal Control Subsection

The TCS is responsible for the regulation of temperature on board the S/C. The temperature in deep space is -273.33 °C (Genovese, 2), while the operating temperature of most instruments is between -20 °C and 40 °C. In addition, TCS policy generally requires that components be operable at 25 °C above and below the predicted temperature extremes (Braun, 5). This difference creates a critical design problem. Some considerations for the TCS are gradual decreasing temperatures away from the sun, frigid periods (i.e. Earth shadow, Jupiter shadow), engine heat, and RTG heat.

Another important design consideration is the degradation of instrument and S/C optics and thermal control surfaces. Therefore, highly sensitive surfaces are oriented so they receive minimal ultraviolet radiation and thruster plume influence (Braun, 3). Also all lower thrusters will have plume shields to protect sensitive surfaces from damage.

Table 4.1 Method of Achieving Thermal Control

	Heating	Cooling
Passive	Multi Layer Insulation Surface Paints (Black) Placement of Heat Generating Equipment	Spring-actuated Louver Surface Paints (White) Placement of Heat Generating Equipment
Active	Electric Heaters Radio-isotope Heaters	Shunt Radiators

Table 4.1 lists the devices that will be used for temperature control in the *Intrepid* S/C. Passive means are employed whenever possible to reduce cost. Passive devices include bimetallic spring-actuated louvers, multi-layered insulation (MLI) blankets made of aluminized Mylar or Kapton, various surface paints, and placement of heat generating equipment. Active means include electrical heaters, radio-isotope heaters, and temperature sensors. Existing hardware elements from previous S/C have been used to provide a low-cost, low-risk, thermal control design.

The TCS was further subdivided into two smaller subsections: bus and science thermal control (BSTC), and the propulsion unit thermal control (PTC).

Bus & Science Thermal Control

The BSTC subsection controls the bus, science boom and the antenna thermal enviornment. The antenna is painted white to control its temperature in near-Earth (1 AU) solar environment. Additionally, an electrically lossy paint will be used to prevent electrostatic charging of the surface (Heacock, 217). The bus thermal control will be achieved using several strategically placed supplemental heaters, louvers to release internal heat, and MLI blankets. The science boom thermal control uses supplemental heaters and an MLI blanket taking into account operating temperature differences from instrument to instrument. The MLI blanket will be constructed with the aluminized side toward space. Thereby the solar input to the S/C increases by six fold in surface absorptivity (Braun, 4). The bus and science temperature will be controlled to be -10 °C to 30 °C, within the required operating temperature.

Propulsion Unit Thermal Control

The PTC system can be divided into two distinct zones: the interior hydrazine propulsion bay and the strut mounted outboard thruster modules. A minimum requirement of 8.33 °C is applied to all propellant delivery items, propellant tankage, and thruster valves to provide an adequate margin and therefore preventing hydrazine freezing. Maximum temperature limits were established to ensure structural and functional integrity during the mission. These limits are 65 °C (fluid), 148.88 °C (thrusters), to 37.77 °C (propellant tank) (Genovese, 3). The propulsion bay temperature will be controlled by multiple heater/thermostat circuits and thermal isolation by three MLIs (Genovese, 2). The thruster modules are thermally controlled according to specific sections. The fluid distribution and control elements such as propellant feed lines and valves will be conductively insulated, actively controlled with heaters and thermostats, and radiatively isolated with MLI blankets, while the thrust chamber will be equipped with heaters and sensors (Genovese, 3).

Materials Subsection

The MS selects the materials to be used for different subsystems. The MS is also in charge of micrometeorite protection. Table 4.2 mentions different environments which will dictate the choice of material for the S/C. These environments include: radiation; cyclic temperature changes; high vacuum; vacuum outgassing; contamination; and particle debris/micrometeorites (Pope, 760).

Table 4.2 Environmental Considerations

Hostile Environments
Radiation
Cyclic Temperature Changes
High Vacuum
Vacuum Outgassing
Contamination
Micrometeorites

Desirable material properties to satisfy these demanding conditions include: high specific modulus and strength; good radiation resistance; high vibrational damping characteristics; low thermal expansion; and low density (Pope, 760). Low density is important as it is directly related to weight and therefore cost. Other considerations include low development and overall cost, previous space worthiness, and space environment lifetime. All the materials examined displayed good overall material properties. All materials were selected on the basis of cost and that they were flight proven. Table 4.3 gives the materials considered versus requirements.

Table 4.3 Materials Considered

	Space Tested	Overall Cost	Development Cost
Aluminum	yes	low	N/A
Beryllium	yes	high/medium	N/A
Titanium	yes	medium	N/A
Composites			
-Graphite/epoxy	yes	low	N/A
-Carbon-carbon	little	medium	medium
-Metal Matrix	little	medium	medium
-New fiber resin	no	high	high
-Sol-gel	no	high	high
-triphasic	no	high	high

Graphite/epoxy was selected as our primary material. Graphite/epoxy is flight proven and requires little developmental cost. Composite materials provide high payoff in systems which require high thermostructural stability because of their low coefficient of thermal expansion and high thermal conductivity ("NASA", 2-12). Composites consisting of high-temperature thermoplastics with graphite-fiber reinforcement provide light weight, good dimensional stability, and excellent resistance to corrosion, chemicals, and wear (Dreger, 52). Graphite/epoxy will be used for the bus bays, the adapter trusses, the science boom, the RTG boom, and all supporting trusses.

The RTGs exterior will use beryllium. Beryllium offers a great combination of weight, strength, and thermal conductivity. Beryllium is one of the lightest structural metals and is used satellite structures. The drawback of using beryllium - toxicity, special design needs, handling requirements, and machining techniques - make it expensive to use. The unique RTG requirements of high temperature, radiation, and severe space environment dictates the use of this more expensive material.

The propulsion system's high temperature, and high stress require a special material. The propulsion system will be made mainly of 6Al4V titanium alloy. Titanium is used in both the tank and thruster strut network. Titanium is space tested, has low development cost and was selected for the propulsion system for its low thermal conductivity and high strength. The drawback of using titanium include slightly higher costs and higher weight density.

Other materials used include: fiberglass (magnetometer boom), aluminized Mylar (MLI thermal blankets), honeycomb aluminum material (antenna), Al_2O_3 coating (protective coating), and lossy surface paints (thermal control).

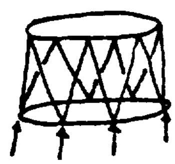
Fiberglass is selected for use in the magnetometer boom because of its unique launch configuration and on-orbit assembly. Fiberglass was also used for the *Voyager* magnetometer boom thus reducing development cost

Appendix A.4.2

Launch loads and axial loads carried by adapter truss.

Maximun acceleration felt during launch (from propulsion subsystem): 4g

(where g=9.8 m/s^2)



USA truss is able to carry 13,230 lb_f in axial compression (Stang, 1)

$$(13{,}230\ lb_f)*(4.48\ \text{Newtons} / 1\ lb_f) = 52970.4\ \text{Newtons}$$

Force on one truss: F_{tr} = (mass of S/C) * (acceleration) / (# of trusses)

S/C launch mass = 1351 kg
acceleration = 4g = 4*(9.8 m/s^2)

$$F_{tr} * (\#\ \text{of trusses}) = 52959.2\ \text{Newtons}$$

Since the force the USA can carry is greater than the force on one truss, the truss will be able to withstand launch loads.

$$59270.4\ N > 52959.2\ N$$

Note: during launch the adapter withstands most of the forces acting on the S/C. Reducing the stress on the actual S/C structure.

and ensuring reliability. Aluminized Mylar is the established material for MLI blankets and has been used on several deep space missions.

Micrometeorite Protection

An additional concern of the MS is micrometeorite protection system (MPS). Though guarding against a large asteroid would be futile, protection from penetration of small micrometeorites is very important. Previous S/C have proven flight worthy for extended flight times (*Voyager* 13 + years), therefore our MPS uses existing flight proven, reliable, technology. The *Intrepid* S/C will use micrometeorite protection shields while aluminized Mylar MLI adds additional protection. A similar system is now being used on *Galileo.*

Further Comments: Conclusion and Recommendations

Possible problem areas include the need to do more precise inertial calculations. The boom vibrational problem must be examined more in depth, and the problem of the electronics vibrating inside the bus must be studied.

The main concern of this proposal is to keep cost low and the design simple, yet reliable. This was achieved through the use of off-the-shelf hardware whenever possible. The SDS used previous S/C as their starting point in design. TCS used established passive means of thermal control where ever possible to reduce cost. MS avoided new untested composites which would bring development cost up. The simple, reliable, and low cost *Intrepid* is practical and has a long line of successful design predecessors.

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COMMAND,CONTROL, AND COMMUNICATIONS (CCC)

Communication Subsystem

The communication subsystem must fulfill the following functions: transmit information from the S/C to Earth (telemetry); receive information from mission management (command); and provide information on the spacecraft position, velocity, and radio propagation medium (tracking) (Yuen, 1-3).

The communication subsystem can be subdivided into the Radio Frequency Subsystem and the Antenna Subsystem. A detailed list of masses and power requirements appears in Appendix B.

Radio Frequency Subsystem

Intrepid's Radio Frequency Subsystem (RFS) will employ X-band (8.414 GHz downlink and 7.161 GHz uplink) to fulfill all communication and radio science functions. Although the *Voyager* and *Galileo* predecessors successfully communicated using X-band as well as S-band, the foremost reason for utilizing only X-band is the decreased cost and simplicity of calibrating one less frequency for operation through the Deep Space Network (DSN) (Draper, 15). In addition to the higher recurring DSN costs of other bands, the testing and fabrication of these bands is more expensive and may require increased power for transmission (JPL Mission Group).

Fundamental to the sole employment of X-band is the X-band transponder. The usage of this transponder as opposed to an S-band uplink possesses the following advantages: decreased ionization effects that accompany the utilization of a higher frequency will result in increased rate and range measurements; charged particle interference is reduced thereby facilitating command of the S/C when approaching Solar conjunctions; in the two-way coherent mode, doppler tracking accuracy and bit-error rate are improved. This last advantage is due to a higher SNR on uplink and virtually no continuation of the uplink frequency to the downlink frequency. Fewer sources of Radio Frequency Interference (RFI) exist on X-band than S-band and channel assignments are 50 MHz wide for X-band as opposed to 10 MHz wide for S-band. The above trade studies were conducted for the MMII program (Draper, 15).

The other major component of *Intrepid's* RFS is the X-band Solid State Amplifier (SSA). Due to its greater simplicity, the SSA was selected instead of the Traveling Wave Tube Amplifier as used on *Voyager* and *Galileo*. In high power mode, the output of *Intrepid's* SSA will be approximately 40 watts and have a mass of 5.5 kg. Details of mass and power requirements for the RFS are located in Appendix B.

Antenna Subsystem

Intrepid's 1.5m High Gain Antenna (HGA) and 0.5m Low Gain Antenna (LGA) will both operate on X-band during uplink and downlink. The technology for the redesign of a LGA to be compatible with the RFS is based on research conducted during the MMII Program. This LGA will operate near Earth until the Sun-S/C-Earth angle falls below 10 degrees. Then the LGA will function mainly during emergencies and when reorientation of the S/C is required (Draper, 21).

Technology for the HGA is also borrowed from the MMII Program. The driving reason for utilizing the LGA and HGA technology from MMII is to reduce cost. Although *Intrepid* will travel 14 AU farther than any MMII mission, the 1.5m HGA was selected. This decision was based on its low weight and launch vehicle compatibility. Even though greater power is generally required to transmit signals as the area of the HGA is reduced, this problem can be offset with the employment of the X-band transponder and the SSA in the RFS (Draper, 20).

Table 5.1 lists selected performance parameters for the *Intrepid* S/C radio systems (Andrew, 61&67) (Anderson et al, 228):

TABLE 5.1

Performance Parameters for *Intrepid* Radio System

Transmitting Parameters:	
Transmitting Frequency	8.414 GHz
Transmitter Powers	
X-band Low Power	10 watts
X-band High Power	13 watts
Transmitting Antenna Gain	
0.5 Meter Parabola	35 dB
1.5 Meter Parabola	41 dB
Receiving Parameters:	
Receiving Frequency	7.161 GHz
Receiving Antenna Gain	
0.5 Meter Parabola	34 dB
1.5 Meter Parabola	40 dB

Further Comments

Intrepid's antenna subsystem may need to be redesigned. Either a larger HGA or possibly the addition of one or two Mid-Gain Antennas (MGA) may be necessary to overcome the vast distances involved in order to successfully transmit and receive signals from Pluto.

Another problem area concerns the life-time expectancy. *Intrepid's* communication subsystem relies heavily on technology developed for the MMII Program. *Intrepid's* trip time is 15 years - 7 years longer than any MMII mission. However any developmental design costs to extend *Intrepid's* life-time should be minimal. In addition, life-time expectancies are generally underestimated as witnessed by the success of the *Voyager* mission. *Voyager* was designed for a life-time of 4 years - it is now entering its 13th year of travel.

Command and Control Subsystem

The Command and Control Subsystem (CCS) must monitor the health of the entire S/C, ensure semi-autonomy and provide stability and control for the S/C. This subsystem is further divided into the following classifications: Command and Data Subsystem; Computer Command Subsystem; Flight Data Subsystem; Data Storage Subsystem; Attitude and Articulation Control Subsystem; microprocessors; data rates; and data memory.

Command and Data Subsystem

Two single module computers will comprise *Intrepid's* Command and Data Subsystem (CDS). Each computer will be programmed identically to provide redundancy and will contain the Computer Command Subsystem (CCS), the Flight Data (FDS) and Data Storage Subsystems (DSS), and the Attitude and Articulation Control Subsystem (AACS). Detailed information regarding the AACS is outlined in the Attitude and Articulation Control section.

Although *Intrepid's* computer system is modelled after *Voyager's*, the following differences exist between them: *Intrepid* employs two computers instead of *Voyager's* six; and each *Intrepid* computer contains the CCS, FDS, and AACS whereas Voyager employed a separate computer for each subsystem (Adamski, 2-3) . Because of *Intrepid's* greater memory capabilities (1 Gigabyte each), only two computers are necessary.

Computer Command Subsystem

The CCS is the principal controller of the S/C: The CCS relays instructions to the AACS which controls the S/C attitude, sensors, gyros, scan action, and thrusters; instructs the FDS to record and compress images; and directs the DSS to record and/or playback the DTR. However, the CCS's foremost responsibilities are the following: collect science data and execute instructions commanded from the ground to operate the S/C; and respond to any problems that materialize with other subsystems (Kohlhase, 40-1).

This second CCS function is executed through Expert Systems (ES). The ES contains software routines called Fault Protection Algorithms

(FPAs). Approximately 20 percent of the CCS memory is comprised of FPAs to allow the S/C to be semi-autonomous. Therefore the S/C can respond to problems as they materialize (Kohlhase, 40). The implementation and reliable performance of ES is crucial to *Intrepid's* mission since such vast distances are involved; roundtrip light time may be as high as eight hours. Since ground stations will communicate very infrequently with *Intrepid*, once a month during cruise mode and then more frequently as Pluto draws near, further S/C autonomy is mandatory.

Flight Data Subsystem

Pluto science and engineering telemetry data are gathered and formatted in the FDS for transmission to Earth. The science data will be downlinked at 300 bps whereas the engineering telemetry data will be transmitted at approximately 40 bps. This engineering data is essentially a status report of the health of the different S/C subsystems. Other information provided are the S/C attitude and scan platform position (Kohlhase, 42).

The FDS will utilize the Reed-Soloman (RS) encoding process. This process, developed during the Voyager II mission, reduces overhead to about 20% by adding only 1200 bits to every 3600 bits of raw science data sent back to Earth. In addition, the number of bit errors is reduced from 5 in 100,000 to only 1 in 1 million (Kohlhase, 126-7).

The FDS includes the software routine of Image Data Compression (IDC) developed for Voyager II. By counting only the difference between the brightness of successive pixel grey levels rather than the entire picture, IDC can reduce by at least 60% the number of bits that characterize each image. This process reduces the time needed to transmit a complete image from Pluto to Earth (Kohlhase, 126).

The FDS operates in a "dual processor mode" whereby the principal mode formats the general science data and the secondary mode compresses the images. By functioning in parallel and executing different functions, the computer memory is effectively doubled. This allows for higher computing tasks (Kohlhase, 42).

The scientific instrumentation and the digital tape recorder are also controlled by the FDS. Such variables as filter choices, exposure times, and imaging shutter modes are determined by the FDS (Kohlhase, 42).

Data Storage Subsystem

The Digital Tape Recorder (DTR) is the integral component of the DSS. Since *Intrepid's* return rate is a low 300 bps, the DTR must contain sufficient memory to store all of Pluto's images. Memory storage of 880 Megabytes of memory will be required to fulfill *Intrepid's* Pluto science imaging objectives. At Pluto encounter, three speeds of the DTR are in use: 1500 bps (record only); 300 bps (playback only); and 100 bps (record and playback).

Microprocessors

As modelled after the CRAF S/C to be launched in 1995, *Intrepid's* CDS will incorporate a SA3300 16-bit solid state microprocessor with less than 1 MIPS and a RAM of 128 or 256 kilobytes (Bowlin, 14). According to a study conducted at Energy's Sandia National Laboratory in 1989, approximately 5 years will be required to design, build, and test this space-qualified microprocessor and its accompanying hardware (Bowlin, 15). Final selection of a microprocessor will not be mandatory until 1997 at design lock-in. This will ensure that the computer hardware can be developed, tested, and integrated into the S/C subsystems to create a fully operational S/C (Bowlin, 16).

The employment of a 16-bit microprocessor will allow for increased on-board processing of mission data thereby reducing the amount of necessary space-to-ground communications (Bowlin, 18). Improved on-board computer processing speed and memory allows for the use of a high-level programming language further reducing software development costs. Because of this, the S/C computers will be programmed in C. In addition, less rewriting of software will be required upon upgrade of the S/C computer subsystem (Bowlin, 19).

Data Rates

To fulfill our low cost mission objective, data rates will be severely reduced as compared to *VoyagerII*. Whereas *Voyager II* transmitted 115.2 kbps in high data rate mode and 266 bps in low data rate mode (Kohlhase, 124), *Intrepid* will transmit from Pluto at a low rate of 300 bps. Low rate return requires less power for transmission and therefore reduces costs (JPL Mission Group).

Upon destination, all science information will be gathered and temporarily stored for later low rate return. Picture data will then be returned over a period of two to three months. At closest approach, 1.5 to 2 passes will be made through the DSN per day. Because of this, base stations on Earth at Goldstone, Madrid, and Canberra will be able to provide 8 hours each of coverage. When only one pass is made through the DSN per day, one base will be utilized at a time, greatly reducing recurring DSN costs (JPL Mission Group).

As stated previously, reduced data rates will decrease costs and require less power for transmission. Estimated power requirements include 65 watts of raw power for a 13 watt transmission at 8.414 GHz. More than 110 watts of raw power for a 22 watt transmission at 8.4 GHz were required during the *Voyager* mission (Edelson et al, 921). These values assume a 20 percent efficient radio frequency transmitter.

Data Memory

The DTR will comprise the Data Memory Subsystem (DMS) and will utilize technology from the MMII Program (Draper, 20). Due to *Intrepid's* low data rate return, it is imperative that its tape recorder have ample storage capabilities. A memory of 880 Megabytes will be required to fulfill our Pluto science objectives (JPL Mission Group). Memory capabilites of each computer will consist of 1 Gigabyte.

Further Comments

The success of *Intrepid's* Command and Control Subsystem relies heavily on the SA3300 16-bit microprocessor. If the design, testing, and development of this microprocessor is delayed and therefore unavailable at design lock-in, an alternate microprocessor must be selected and further redesign of the computer system must be initiated.

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ATTITUDE AND ARTICULATION CONTROL SUBSYSTEM (AACS)

Introduction

The driving design factors for the AACS, in addition to the requirements stated in the request for proposal, are the strict pointing accuracy and stability requirements needed for the spacecraft to communicate back to Earth and to perform the mission's science experiments. The AACS provides spacecraft orientation from the time of separation from the launch vehicle through the end of the mission. The subsystem must be able to control the spacecraft during all mission modes and adapt to changing mission requirements. And possibly the most important factor in the design of the subsystem is that it must be able to perform autonomously during all mission modes. A system to meet these requirements has been developed using standard technology and off-the-shelf hardware.

Method of Stabilization

When selecting the method of stabilization to be used for *Intrepid*, the mission requirements had to be examined closely. The nature of the mission calls for stringent pointing accuracy and stability while the request for the mission calls for low cost, simplicity and reliability. These factors have been taken into consideration, and have led to the selection of a 3-axis stabilization technique.

The 3-axis technique has been selected for its excellent pointing accuracy, reliability and versatility (Koepke, 36). The pointing accuracy of a three-axis spacecraft is only limited by the sensors used. The quality of attitude sensors has increased substantially in the last few years, thus improving the accuracy of the 3-axis spacecraft. This improvement will be spelled out later when the attitude components are described. 3-axis systems have been flight tested for long duration missions (*Voyager*), thus the reliability of such a system is proven. The lifetime of the system is a factor of the on-board propellent and the lifetime of the components. There is going to be enough fuel on-board to carry out the mission plus contingency, and the components have a lifetime exceeding the length of the *Intrepid* mission, 15 years. The 3-axis method is much less complex, less costly, and more reliable than the dual-spin method. Complexity and the cost seem to go hand in hand. The more complex the system, the more costly and less reliable it is. Dual-spin is not as reliable as a 3-axis system because of the substantial increase of moving parts which are subject to wear over a very long mission such as *Intrepid's*. The versatility of the spacecraft is a very important aspect for a long duration mission. The 3-axis system is more adaptable to changing mission requirements than the spin-stabilized system. Furthermore, the 3-axis system has greater maneuverability than the spin-stabilized spacecraft which gives it better versatility when performing science procedures. The spin-stabilized spacecraft has imaging limited to the line of scan. This makes it more difficult to scan objects, which is a major objective of the mission. Also, the configuration of the spin stabilized spacecraft is constrained by its geometry, where as the 3-axis spacecraft is not. This gives greater versatility to the design and layout of the spacecraft.

The disadvantage of heavy, costly hardware for 3-axis spacecraft is not as apparent. The new generation of attitude hardware for 3-axis systems is coming down significantly in mass, power and cost. This makes the method of stabilization even more attractive.

High Precision Scan Platform (HPSP)

The nature of the mission requires very stringent pointing accuracy and stability. The mission also entails that the spacecraft have the capability to meet demanding and changing mission requirements. One of these is the stability required to do imaging in the low level of light that will be encountered at Pluto. The low level of light forces the camera to use longer exposure times for imaging. The longer exposure time dictates a need for increased stability to eliminate image smear. The increased pointing accuracy and stability of the spacecraft call for a better method of achieving these requirements. This increased pointing accuracy and stability can be achieved using new technology from the *Mariner Mark II* program, the high precision scan platform (Bell, 807).

The HPSP provides a rigid, 2 degree of freedom, momentum compensated platform for precision inertial and celestial sensors and science instruments with high pointing requirements (Bell, 807). The momentum compensation in the platform allows the activity of the scan platform to become dynamically isolated from the spacecraft bus. Thus, the bus is not disturbed by scan platform motion. This is advantageous because it allows scan activity without an impact on fuel consumption, thus creating a savings on mass and cost. The HPSP gives greater pointing accuracy and stability due to its rigid mount and decoupling. This eliminates many sources of error in instruments that are evident on existing platforms.

The HPSP can slew about two axes, azimuth and elevation relative to the boom (see figure 1). This motion is accomplished by using an internally redundant microstep actuator for each axis of motion (Bell, 807). Each actuator has a reaction wheel that is automatically activated to compensate for the torque created by the platform motion. This allows the dynamics of the platform to be isolated from the bus. For attitude determination, the HPSP contains an inertial reference unit and high accuracy star tracker. The attitude is determined on the HPSP and then is transferred to the spacecraft bus. The drawback of the HPSP is seen because it creates a complex coordinate transformation to provide the spacecraft bus attitude (Draper, 12). Thus, a sophisticated algorithm must developed to provide quick relay of data from the platform to the microprocessor to the attitude actuators. The total HPSP pointing accuracy is estimated to be 0.76 mrad/axis. This a vast improvement over that of the *Voyager* scan platform accuracy of 2.26 mrad/axis (Bell, 808). A layout of the HPSP with the attitude and science instruments is given in figure 6.1.

Inertial Reference Unit (IRU)

The inertial reference unit provides 3-axis rotation rate data for the spacecraft in all control modes. The conventional IRU's used on past spacecraft have employed mechanical gyroscopes. These, due to their mechanical nature, are subject to drift error and must be updated frequently. However, the length of the mission calls for maximum autonomy and accuracy. So a new type of IRU is going to be used which can give this autonomy and accuracy with a mass and cost savings. This IRU is called the Fiber Optic Rotation Sensor (FORS) (Draper, 801).

FORS is capable of a long life with high reliability (Stokes, 162). The reliability of the sensor is a function of the light source used. This uncertainty has been eliminated by using a semiconductor light source that requires no thermoelectric cooler (Stokes, 163). FORS contains no high power components and also uses standard, proven technology for the space environment.

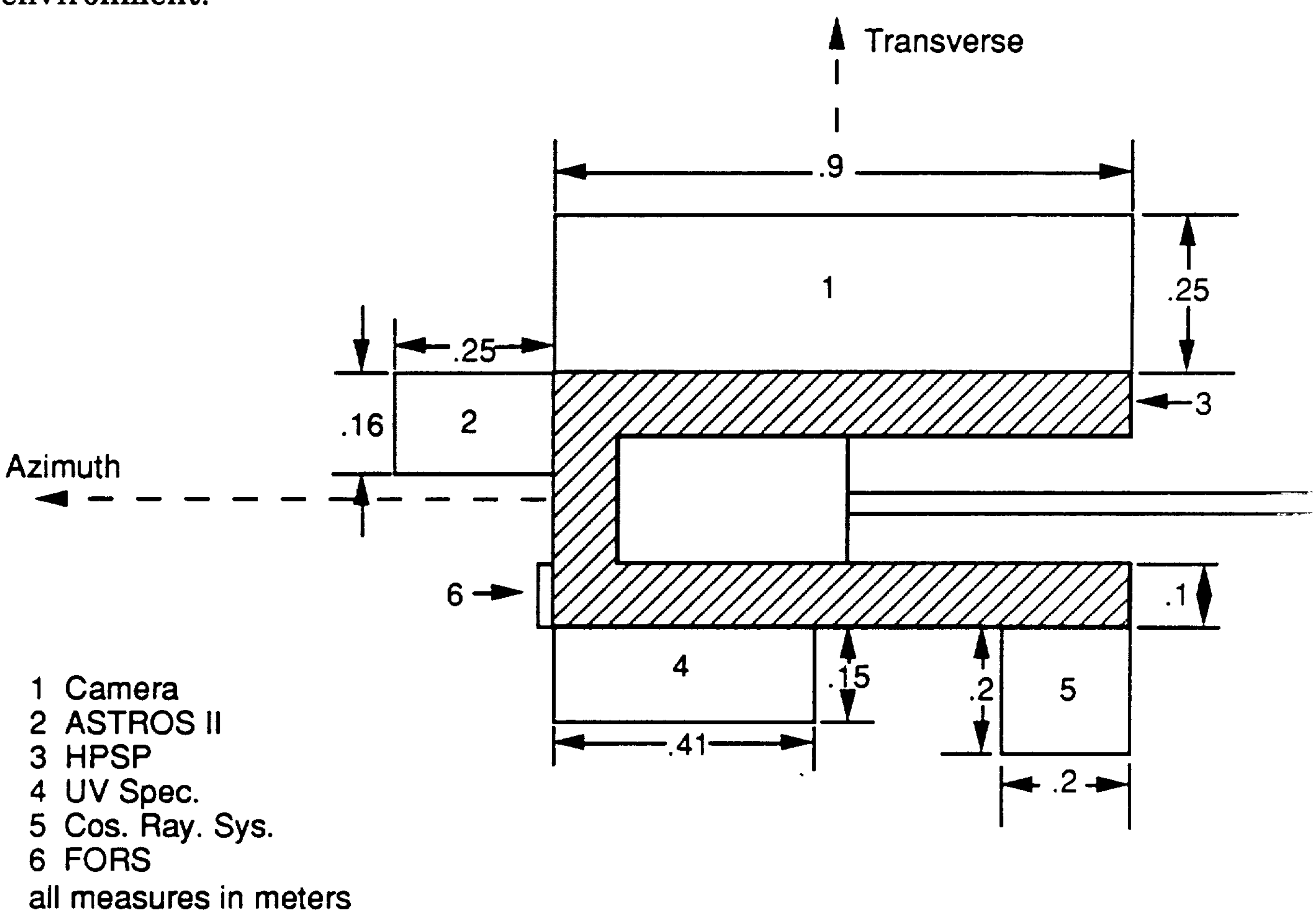

Figure 6.1 HPSP with attitude and science instruments

FORS has many advantages over other types of gyros available, namely mechanical and laser gyros (Stokes, 161). One advantage over the mechanical gyros is that FORS does not have any moving parts. This allows FORS to be active much longer without having to update for drift errors as with the mechanical gyros. This translates directly into a cost savings. Also with no moving parts, FORS is not subject to mechanical failure due to wear on parts, thus creating a longer life span. A second advantage over mechanical gyros is that there is no gravity sensitivity. This makes FORS absent from additional error while the spacecraft is in a high gravity region, such as near Earth or Jupiter. There are many advantages over the ring laser gyro which are very important for the *Intrepid* mission. A high voltage supply and mechanical dither are not needed in the FORS design. And in addition, the FORS is available at a much lighter weight. All of these advantages add up to a longer lifetime and lower cost for the FORS inertial reference unit. Table 6.1 shows the drift, weight, and power of the three types of sensors.

Rate Sensor	Residual Drift Rate (deg/hr)	Rate noise (deg/hr)	Power (W)	Mass (kg)
FORS	$2 * 10^{-4}$	$1 * 10^{-5}$	10	10
DRIRU II	$3 * 10^{-3}$	$1 * 10^{-5}$	22	11
Laser Gyro	$7 * 10^{-3}$	$4 * 10^{-4}$	18	18

Table 6.1 Comparison of rate sensors (Draper, 14)

FORS has been created with the same interface as the NASA standard gyro DRIRU II and is to be placed on the HPSP. Therefore the cost of creating a new interface is eliminated (Bell, 803). Since FORS is not flight tested, although it is expected to be flown on the *Mariner Mark II* missions, *Intrepid* can revert back to the standard gyro easily if any problems surface with the FORS unit.

FORS also measures body acceleration during thruster burns to sense body rates. This eliminates the need for accelerometers (Stokes, 161). The unit is 3-axis internally redundant, provides a long life, low mass and solid state inertial rate and position sensor, and is being used as the inertial reference unit for the *Intrepid* mission.

Star Sensor

The main requirements for attitude components has been reliability over a long life span and redundancy. The only star sensor that will be available that can meet these requirements while also giving optimum performance, low mass and power consumption, all for a low recurring cost is the Advanced Star and Target Reference Optical Sensor (ASTROS II) star tracker (Draper, 15).

ASTROS II has increased value in many ways to the existing types of trackers available. The main advantage is that it is fully internally redundant. None of the other available star trackers have this feature, which is very important for the length of our mission. Other advantages are its high accuracy, low recurring cost, low mass and low power consumption. It is expected to be flight tested on the *Mariner Mark II* missions thus becoming off-the-shelf hardware.

ASTROS II utilizes a Charge-Coupled Device (CCD) which enables closed loop pointing of small bodies such as Pluto or asteroids. This gives greater flexibility if there are changing mission requirements because it allows for autonomous science gathering operations (Bell, 803). The CCD also gives ASTROS II the ability to have a bright particle pass in front of it and not lose a star lock. This is very important for autonomy and cost since it cuts down on the number of maneuvers to regain star lock.

Sun Sensor

Sun sensors provide the pitch and yaw position of the spacecraft relative to the sun. It provides initial sun acquisition and backup or emergency attitude knowledge during cruise mode. A CCD sensor has been selected based on its redundancy, long life, reliability, high accuracy, low mass and low power consumption (Flamenbaum, 234). Two units are used for redundancy. One located on the scan platform boom and the other on the HGA. They are aligned so at least one of them has the Sun in its field of view at all times.

AACS Computer

The microprocessor is the heart of the AACS (Johnson, 4). All measurements from the attitude sensors are processed by the computer, which in turn gives the appropriate commands to the attitude control hardware. The AACS computer contains the flight software and fault protection algorithms to perform and maintain the AACS. The flight software contains the algorithms to maintain *Intrepid* attitude and control during all mission modes. The computer is also interfaced with the communications subsystem allowing for reprogramming of the flight software from Earth. The fault protection algorithms give the computer the power to reconfigure the AACS in case of system problems. There are four basic problem areas to be concerned; hardware failure, Single Event Upset (SEU), flight software problems and ground errors (Johnson, 7). The fault protection software continually monitors the health of subsystem, making sure that it is functioning properly to maintain *Intrepid* attitude.

The microprocessor used is going to allocate 32k of the main computer. It contains a bidirectional interface between it and the sensors and actuators and is internally redundant. These are unique and redundant sets of interfaces with each peripheral for optimum autonomy.

Reaction Control System

The reaction control system of the spacecraft produces the force necessary to maintain the attitude of Intrepid. There are many different ways of providing this necessary torque. The two different techniques considered for use on Intrepid are reaction wheels (RW) and reaction control thrusters (RCT).

When selecting which control system would be used, the basic overall requirements were the driving factors once again. The RCT provided the best maneuvering capability of the vehicle for emergency situations, delta-v corrections and in handling any changing mission requirements. It also provided the simplest system of operation.

The drawbacks of the RCT are the plume contamination created by the hydrazine thrusters, the additional torques created, and the moderate pointing capabilities in comparison with the reaction wheels. The moderate pointing accuracy is justified by the use of the HPSP for instruments with stringent pointing requirements, and the fact that the system is not used during fly-bys. Plume contamination is also created by the RW system since it also uses thrusters to unload the reaction wheels.

The RCS consists of twelve 0.2 N thrusters used on the *Mariner Mark II* spacecraft (Bell, 796). The system is broken down into three sets of four thrusters, with any set being able to perform the necessary attitude maneuvers for autonomy. Appendix A.6.1 calculates the minimum size of thrusters needed to perform a difficult maneuver.There is one set around the spacecraft bus, the thrusters are 90 degrees apart, and the two other sets are on pods around the truss of the propellant tank. The thrusters are angled so each set can perform the necessary maneuver for any axis. A layout of the thrusters around the bus is shown in figure 6.2.

Mission Modes

Intrepid is going to have to be able to perform in many different mission modes. It is not going to have to perform at peak performance for long time periods. There are going to be periods of maximum pointing accuracy and stability, and there are going to be times of relatively easy pointing requirements. The following are descriptions of all of the different modes.

Separation from launch vehicle

The AACS must stabilize the spacecraft after release from the launch vehicle and make corrections for the proper launch trajectory. The AACS should stabilize the vehicle's three inertial axes at a rate of 0.1 deg/sec (Dougherty, 19).

Cruise mode

The spacecraft is going to spend most of the mission in cruise mode. Cruise mode consists of keeping the high gain antenna (HGA) pointed at Earth and correcting for trajectory errors. The HGA has two different pointing modes. The first mode is for when the uplink/downlink is on. This mode is going to operate for 8 hours at a time once every week and maintain a pointing accuracy of 8.7 mrad (Bell, 799). The other mode of cruise is when the uplink/downlink is off. This mode is going to operate at all other times except during trajectory correction maneuvers. It has to maintain a pointing accuracy of 17.5 mrad within Earth (Bell, 798). The cruise mode corrects for trajectory errors every two to three weeks. This is commanded through the AACS microprocessor and the torque produced is going to vary in magnitude from correction to correction.

During cruise, the attitude reference is going to be performed primarily by the star tracker. The star trackers are also used to update the IRU. The IRU's primary responsibilities are during thruster activation and back-up attitude reference in case of a loss of any of the other sensors. The Sun sensors are used for back-up sensing and for reacquiring stars in case of a loss of star lock. The torque is produced by the 0.2N thrusters as commanded by the microprocessor.

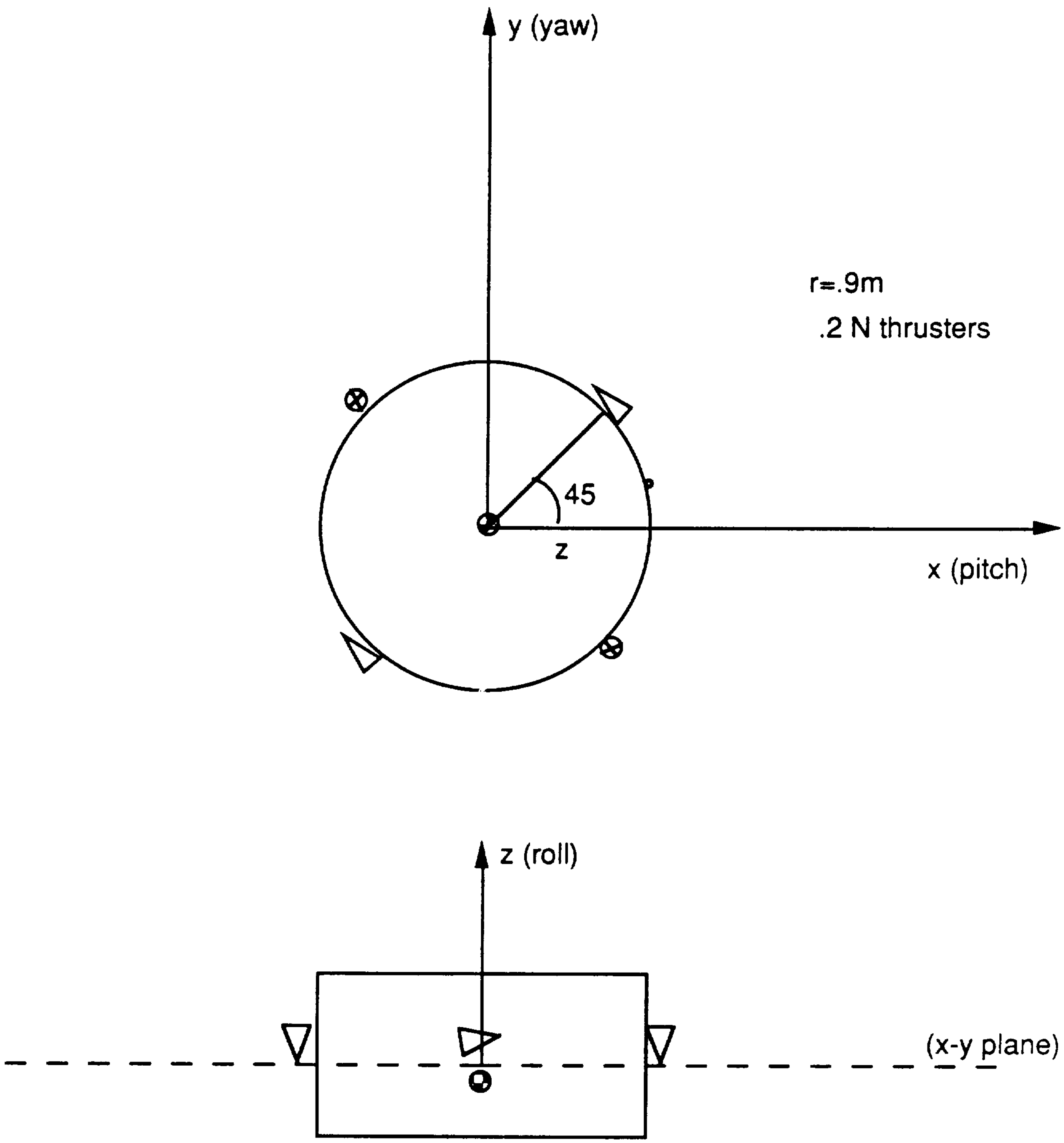

Figure 6.2 Bus thruster layout

Planetary encounter

The planetary encounter mode requires the most strict stability during the entire mission due to the imaging constraints. The far encounter phase pointing of the planetary fly-by consists of pointing the imaging camera at the planet. This is going to be accomplished through articulation of the HPSP. Thus the Intrepid-Earth communication link is not broken during the far encounter phase since the camera is actuated on the HPSP which is dynamically isolated from the bus. The near encounter phase pointing consists of performing the UV-spectrometer scanning in addition to maintaining the imaging. The planet is acquired using the ASTROS II in a closed loop pointing scheme.

Right before the near encounter, the IRU is to be updated by the celestial sensors and the RCT is shut off. The only attitude to be kept is on the HPSP. The RCS is shut off so not to contaminate any of the scientific equipment (Coupe', 30). The communication link is not kept during this phase. Once the phase is over (the UV-spectrometer scanning is complete), the celestial sensors reacquire the Sun and guide stars based on the attitude determined by the IRU. *Intrepid* now returns to the far encounter phase to complete the imaging of the planet.

During imaging, the AACS needs to create 2.43 (N m sec)/ sec for spacecraft stability during imaging. This is the momentum required per second of operation (see Appendix A.6.2) (Koepke, 37). A possible problem in this area is maintaining this stability of the spacecraft during the near encounter phase. The stability management is going to have to be studied further to see the feasibility of this approach. A possible solution at an extra mass and power cost would be to add reaction wheels to control the spacecraft during this short encounter phase.

Delta-v burns

The AACS aligns the spacecraft on the correct trajectory line for the delta-v burn, and must maintain thrust vector control (TVC) throughout the burn. The TVC keeps the spacecraft pointing on the correct trajectory. This is performed by pulsing the attitude thrusters as commanded by the computer using input from all of the attitude sensors. A TVC pointing error of 20 mrad is to be kept during the maneuver (Bell, 799). There is no communication link during the thrust. Earth is reacquired at the conclusion of the burn in the same manner as after a near planetary encounter. The different modes are shown in figure 6.3.

Departure from launch vehicle → Cruise → Earth delta-v → Cruise → Jupiter flyby → Cruise → Pluto flyby → Cruise

Figure 6.3 Mission modes

Conclusion

The AACS developed for the Intrepid mission stresses reliability, autonomy, low cost and simplicity. This has been accomplished while being able to meet extreme pointing and stability requirements. Thus the system is able to perform flawlessy while being able to also perform autonomously at a low cost.

Appendix A.6 .1

Turn spacecraft 90 degrees in 18 minutes, from Hubble Telescope (Dougherty, 17).

Equation:

$$\Theta_c = (2*F*L)*(t/2)^2/(I_v)$$

Solve for F, using:

$L = .65m$
$t = 1080$ sec
$I_v = 1808.8$ kg m^2
with full tank (Koepke, Inert)
$\Theta_c = \pi/2$ rad

then $F = .0054$ N

Thus 0.2 N thrusters are going to more than enough to handle the spacecraft.

Appendix A.6.2

The momentum per second required for maintaining stability during planetary encounter imaging (Koepke, 37) (Kohlase, 35).

$H_t=(F*t_{on})^2 * L^2 / (\Theta_t * I_v)$ where

$F = .2$ N
$t_{on}=15$ sec
$L = .65$ m
$\Theta_t = 1.75$ mrad
$I_v = 1719.016$ kg m^2
with 68 kg in tank (Koepke,Inert)

then $H_t=1.264$ N m sec/sec

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Conclusion

Final considerations for the Intrepid spacecraft include an implementation plan. Once the optimal design is finalized, a prototype must be built and thoroughly tested. Further trade studies, sensitivity analyses, and redesign may be required for increased optimization. Upon successful completion of testing, the remaining three spacecraft will be built. Since Intrepid is constructed mainly of off-the-shelf hardware, the development and manufacturing timeframes are greatly reduced.

Of utmost importance when determining a design freeze date is the incorporation of the latest computer and Plutoian system knowledge. Significant gains can be realized by integrating the most recent computer developments in the areas of self-autonomy, flexibility, and survivability. Any discoveries concerning the Plutoian system should affect the mission science objectives in order to retrieve the most meaningful and vital information.

Appendix B - Mass and Power Requirements

Subsection	Component	Mass [kg]	Power Req. [W]
SI	CRD	7.0	5.0
	PLS	10.0	8.0
	MAG	6.0	6.0
	USO	2.0	5.0
	SSI	31.0	26.0
	UVS	5.0	6.0
SUBTOTAL		**61.0**	**56.0**
MMPC	**N/A**	**N/A**	**N/A**
SS	HGA	35.72	N/A
	Bus structure	24.58	"
	Outboard shear plate	17.59	"
	Inboard shear plate	8.11	"
	Science boom	7.64	"
	RTG boom	6.69	"
	Fasteners	1.50	"
	Connector bracket	2.17	"
	Shunt radiators	6.88	"
	Tank support	4.24	"
	Truss Adapter	8.82	"
	Tank struts	0.66	"
	RTG launch support	3.73	"
	Misc. supports	8.67	"
	MLI - bus	22.52	"
	MLI - science	3.01	"
	MLI - propulsion	1.99	"
	Heaters - bus	0.75	6.0
	Heaters - science	1.21	7.0
	Heaters - propulsion	1.80	7.0
	Louvers - bus	2.94	"
	Louvers - science	0.78	"
SUBTOTAL		**170.00**	**20.0**
PPS	Service valves (16)	3.68	N/A
	Regulator valves (4)	4.44	"
	Filter	0.17	"
	Tubes, fittings, sensors, etc...	2.67	"
	100 N thrusters (4)	6.80	"

Subsection	Component	Mass [kg]	Power Req. [W]
	0.2 N thrusters (12)	7.20	"
	Propellant tank	16.10	"
	Pressurant tank	16.15	"
	He pressurant	2.24	"
	N_2H_2 propellant	800.00	"
	Power distribution unit	5.41	"
	2.4 kHz inverter	4.29	"
	Power control unit	5.65	"
	Shunt regulator	3.99	"
	Pyro switching unit	4.01	"
	Modular RTG's	34.06	"
	Pyrotechnics	5.00	"
SUBTOTAL		**921.86**	**N/A**
CCC	X-band transponder	5.00	
	Solid state amplifier	5.50	
	Input isolators (2)	0.35	
	Output isolators (2)	0.25	
	Duplexers (2)	2.94	
	Receiver RF switch	0.90	
	Transmitter RF switch	1.04	
	Low pass filter (2)	0.06	
	X-band attenuator	0.04	
	Command detector (2)	4.0	
	TLM Modulation (2)	4.0	
	Computers (2)	10.0	
	Solid State Mass Memory	10.0	
	Power Supply	1.0	
	Misc. electronics	4.0	
	Antenna subsystem	5.0	
	System assembly	5.0	
SUBTOTAL		**59.08**	**90.0**
AACS	FORS	10.0	10.0
	ASTROS II	8.0	11.0
	Sun sensors (2)	2.3	7.0
	HPSP	11.81	32.0
SUBTOTAL		**33.11**	**60.0**
Contingency			**30.0**
S/C TOTAL		**1245.05**	**256.0**

Appendix C - Trade Studies

RFP Requirements

Design Choices

	Avail. 1999	With-stand Envir	Exist. Hrdw	Fits Miss. Req.	Life time	Low cost	Re-cent tech.	Rel-iable	Sim-ple
CRS	V/G	V/G	V/G	V	V/G			V/G	
PLS	V/G	V/G	V/G	G	V/G		G	V/G	
SSI	V/G	V/G	V/G	G	V/G		G	V/G	
Al-7000 T.M.	X	X	X	X	X	X	X	X	X
Batteries	X	X	X	X		X	X	X	
Bipropellant	X	X	X	X			X	X	
Cryog. prop	X	X	X	X			X	X	
GN_2 press.	X	X	X	X	X	X		X	X
He press.	X	X	X	X	X	X		X	X
Modular RTG	X	X	X	X	X	X	X	X	
Monoprop.	X	X	X	X	X	X		X	X
Non mod RTG	X	X	X	X	X	X		X	X
One prop tank	X	X	X	X	X	X		X	X
Reg. Power	X	X	X	X	X	X			
Solid prop.	X	X	X	X	X	X		X	X
Ti-6Al4V tank	X	X	X	X	X			X	X
Titan IV	X	N/A	X	X	N/A		X	X	N/A
Titan IIID	X	N/A	X	X	N/A	X		X	N/A
Two prop tank	X	X	X	X	X			X	
Unreg. power	X	X	X	X	X	X		X	X
Beryllium	X	X	X	X	X		X	X	N/A
GraphiteEpoxy	X	X	X	X	X	X	X	X	N/A
Passive Therm	X	X	X	X	X	X	X	X	X
Titanium	X	X	X	X	X		X	X	X
Toroidal Bus	X	X	X	X	X	X	X	X	X
HGA	X	X	X	X		X	X	X	X
LGA	X	X	X	X		X	X	X	X
LowDataRate	X			X		X			X
Prog. in C	X			X		X		X	X
SA 3300	X	X		X			X		
SSA	X	X	X	X			X		X
X-band only	X			X		X			X
X-band transp.	X	X	X	X			X		X
ASTROS II	X	X	X	X	X	X	X		X
CCD Sun Sens	X	X		X	X	X	X	X	X
FORS	X	X	X	X	X	X	X	X	X
HPSP	X	X		X	X		X	X	
Thrusters	X	X	X	X	X	X		X	X

Appendix D - Acronyms

AACS	Attitude and Articulation Control Subsystem
AU	Astronomical Unit
BSTC	Bus and Science Thermal Control
CCD	Charge-Coupled Device
CCS	Computer Command Subsystem
CDS	Command and Data Subsystem
CM	Center of Mass
CRAF	Comet Rondezvous Asteroid Fly-by
CRS	Cosmic-Ray detector System
DSN	Deep Space Network
DSS	Data Storage Subsystem
DTR	Digital Tape Recorder
ES	Expert Systems
FDS	Flight Data Subsystem
FPA	Fault Protection Algorithm
HGA	High Gain Antenna
IDC	Image Data Compression
JPL	Jet Propulsion Laboratory
LGA	Low Gain Antenna
MAG	Magnetometer
MIPS	Millions of Instructions Per Second
MLI	Multi-Layer Insulation
MMII	*Mariner Mark II*
MPS	Micrometeorite Protection System
PLS	Plasma detector System
PPS	Power and Propulsion Subsystem
PTC	Propulsion unit Thermal Control
RAM	Random Access Memory
RFI	Radio Frequency Interference
RFP	Request For Proposal
RFS	Radio Frequency Subsystem
RS	Reed-Soloman
RTG	Radio-isotope Thermal Generator
SA3300	Sandia Application 3300 Family
S/C	spacecraft
SDS	Structural Design Subsection
SI	Scientific Instrumentation
SNR	Signal to Noise Ratio
SS	Stuctural Subsystem
SSA	Solid State Amplifier
SSI	Solid State Imaging
SSP	Science Scan Platform
USA	Upper Stage Assembly
USO	Ultrastable Oscillator

N91-16048

P-99

Project Cerberus: Flyby Mission to Pluto

AAE 241: Spacecraft Design Proposal

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Spring Semester 1990

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INTRODUCTION

Pluto, the ninth planet in the solar system, is named after the Greek god of the Underworld. The namesake of this project is Cerberus, Pluto's watchdog which faithfully stood guard at the gates of Hades.

Cerberus is designed to meet the requirements stated in the Request for Proposal (RFP). Those requirements that apply to all subsystems are summarized below:

1) Develop a conceptual design for an unmanned spacecraft to study Plutonian space.
2) Optimize performance, weight and cost.
3) Spacecraft should be simple, reliable and easy to operate.
4) Use off-the-shelf hardware and technology available by 1999.
5) Identify and minimize on-orbit assembly.
6) Should be able to perform several possible missions.
7) Sufficient design lifetime to carry out its mission plus a reasonable safety margin.
8) Use latest advances in Artificial Intelligence.
9) For costing and overall planning, assume that four spacecraft will be built.

The goal of the Cerberus Project is to design a feasible and cost-effective mission. The design stresses proven technology that will avoid "show stoppers," which could halt mission progress. Cerberus also utilizes the latest advances in the spacecraft industry to meet the stringent demands of a journey to the edge of the solar system. The result is Cerberus, a practical means to unlocking the mysteries of Pluto.

Section 1: MISSION MANAGEMENT, PLANNING, & COSTING (MMPC)

Section 1-1: MMPC REQUIREMENTS

MMPC entails several requirements from the Request For Proposal (RFP) specific to the subsystem. These are in addition to general requirements which pertain to the mission as a whole. The MMPC requirements, specific and derived, are listed below.

1) Spacecraft must travel to Plutonian space.
2) Spacecraft must travel to Plutonian space via an optimal trajectory.
    a. Trajectory should be optimized for Δv.
    b. Trajectory should be optimized for time of flight (TOF).
3) Individual burns must remain within limits of propulsion system.
4) Spacecraft trajectory should not subject spacecraft to conditions which will cause it undue damage.
5) MMPC analyst must perform cost estimates of individual subsystems.
6) MMPC analyst must perform cost estimate of entire mission.
7) Mission type must be one of three types: flyby, orbiter, or lander.
8) MMPC analyst must outline mission sequence of events.
9) Launch should take place sometime between 2000 - 2010 A.D.
10) Acronyms for Section 1: MMPC are listed in Appendix A-1.

Section 1-2: SELECTION OF MISSION TYPE

The flyby type of mission has been determined to be the best for the Cerberus project. This decision was based upon the results of preliminary trajectory studies using the multiple impulse optimizing program, MULIMP, as well as other considerations. These considerations include:

1) Existing technology does not facilitate the transportation of fuel mass necessary to burn into orbit capture at Pluto. This precludes the orbiter and lander class missions. MULIMP studies produced Δv values which would be required for orbit capture. There exists no trajectory which would allow orbit capture at Pluto, given the available technology, and the acceptable TOF for this mission.
2) Pluto and Charon are only separated by 19400 km.[3] An object placed in orbit about Pluto would likely have its orbit perturbed such that it left Plutonian space.
3) Other general solar system science can be emphasized on a flyby mission. Examples are a Jupiter study before arrival at Plutonian space, and a measurement of the heliopause after Pluto passage.
4) This is a preliminary mission; it is prudent for this to be of the flyby class given present knowledge of Plutonian space. This mission will assist in determining the benefit of another Pluto mission. This venture will yield information required for launch of an orbiter or lander class mission, when technology for such a mission becomes available. It is also prudent to keep this mission at the flyby class to keep from overshadowing the Lunar and Mars initiatives.

Section 1-3: TRAJECTORY DETERMINATION

Several paths were studied as a possible route to Pluto. These paths included different gravity assist flyby trajectories:

1) Earth - Pluto
2) Earth - Jupiter - Pluto
3) Earth - Mars - Jupiter - Pluto
4) Earth - Venus - Earth - Earth - Jupiter - Pluto
5) Earth - Earth - Jupiter - Pluto

Certain data were required to facilitate logical study of trajectories with MULIMP. Values of Δv and TOF required for Hohmann transfer between the various planets of interest were calculated by hand. These are listed in Table 1-1. Equations and parameters used in these calculations are listed in Appendix A-2.

Table 1-1: Hohmann Transfer Values

Planets of Interest	Δv(km/s)	TOF(days)
Earth & Pluto	11.8120	16581.5465
Earth & Mars	2.9458	258.9324
Earth & Jupiter	8.7920	997.5984
Mars & Jupiter	5.8968	1125.6354

The orbit periods and synodic periods of the planets of interest are given in Table 1-2.[2]

Table 1-2: Orbit Periods and Synodic Periods

Planet	Orbit Period	Synodic Period(days)
Venus	224 days	584
Mars	687 days	778
Jupiter	11.9 years	398
Pluto	247 years	367

The first path to be investigated was a direct transfer from Earth to Pluto. Observation of hand calculated data and study of available literature[3] lead to the following conclusions:

1) The mission TOF would be too long for practical purposes using existing technology.

2) The Δv at Earth parking orbit (1.0437 Earth radii) would be too large for available propulsion systems.
3) The launch energy (C3) would be in excess of the capabilities of existing launch vehicles.

The second path to be investigated was a transfer to Pluto with a gravity assist at Jupiter. The first step in studying this trajectory was to use MULIMP to find the optimal transfer from Earth to Jupiter, in terms of Δv, in the early part of the first decade of the twenty-first century. Knowing this locally optimal launch date, MULIMP optimized the trajectory to its completion at Pluto. The first launch date was then incremented by an amount equal to Jupiter's synodic period, and the Earth - Jupiter - Pluto transfer was again optimized by MULIMP, This process was repeated until the incremented launch date fell outside the ten year launch window prescribed by the RFP. MULIMP data also displayed position of each event in three dimensional Cartesian coordinates, with the sun as origin and its ecliptic as the X-Y plane. With this data, a path could be graphically plotted to ensure smooth flow of the trajectory in a counter-clockwise manner.

Upon examining the various data output by MULIMP, it was clear that although the Earth - Jupiter - Pluto trajectory was an improvement upon the Earth - Pluto trajectory, it was still unsatisfactory for these reasons:

1) The TOF was generally greater than 18 years.
2) The Δv at Earth parking orbit was larger than desired for cost effectiveness of the mission.
3) The launch energy approached or exceeded unacceptable levels (i.e., from 90 to several thousand).

The next path under consideration was Earth - Mars - Jupiter - Pluto. This trajectory was studied employing a similar method of attack to that used for the Earth - Jupiter - Pluto study. The best early launch date to Mars was determined using MULIMP, and this launch date was incremented by Mars' synodic period through 2010. At each launch date, MULIMP optimized the complete Earth - Mars - Jupiter - Pluto transfer. Several MULIMP studies were also performed in the region of time of two months preceding and following each of the aforementioned launch dates. Once again, this trajectory displayed improvement over the previous path studied, particularly in these areas:

1) Launch energy: C3 fell generally in range of 10 - 15.
2) Δv from Earth parking orbit was reduced to the range of 3 to 4 km/s
3) TOF was in the range of 15±2 years.

Prohibiting problems with this flyby path arose at Mars, where an impulse of at least 6.3 km/s was required.

The ensuing path under scrutiny was an extension of Galileo's Venus - Earth - Earth Gravity Assist (VEEGA) trajectory.[4] A flyby of Jupiter was added before the spacecraft continued on to Pluto. It became apparent upon reviewing MULIMP test output that the planets of the solar system are not in position conducive to this type of transfer during the prescribed ten year launch window. Although TOF was reduced significantly to roughly six years, flyby impulses unattainable.

The final trajectory to be considered was an Earth- Earth - Jupiter - Pluto path. The method of attack for studying this path was again similar to that described previously. This trajectory yielded the most satisfactory values. Table 1-3 displays values from each type of trajectory studied. These values were compared in a trade study manner to determine the best course of flight. Although these values are not likely to be the absolute optimum value for each case, they were determined to be sufficiently representative.

Table 1-3: Characteristic MULIMP Values for Different Trajectories

Path	Δv(km/s)	TOF(years)	C3	Comments
EP	N/A	N/A	N/A	Excluded from MULIMP study
EJP(typical)	11.195	26.935	94.560	
EJP(unusual)	8.194	15.383	90.523	Path falls through Jupiter at .8 planet radii

EMJP(typ.)	10.915	13.665	11.376	Δv at Mars: 7.201 km/s. Unattainable.
EMJP(unus.)	3.788	14.727	10.814	Path falls through center of Mars & .723 Jupiter radii.
EVEEJP	93.556	6.072	13.687	Combined Δv for the two E flyby's: 89.733 km/s.
EEJP	5.941	18.688	47.518	

E: Earth J: Jupiter M: Mars P: Pluto V: Venus

The Earth - Earth - Jupiter - Pluto trajectory was selected due to several considerations:

1) The Δv from Earth parking orbit was determined to be attainable for Cerberus' mass (see Section 3: Propulsion).
2) The launch energy was determined to be attainable for this mission using existing technology (see Section 3: Propulsion).
3) Midcourse and flyby Δv's were determined to be feasible(see Section 3: Propulsion).
4) Flyby of Jupiter occurs at a distance which does not require addition of radiation shielding to Cerberus' structure.
5) There is a launch window of eleven days.
6) Launch occurs early enough in the 2000 - 2010 period to allow postponement and still make that ten year window.

A later launch date would allow for more development time, better weather for launch, and a chance to find a better trajectory. However, Pluto is moving away from its perihelion distance of 29.6 AU, which it reached in 1989.

Therefore, an earlier launch has the potential to travel less distance to Plutonian space.

Table 1-4 displays information on Cerberus' trajectory, and outlines the mission sequence of events. The launch date shown falls in the middle of the eleven day launch window.

Table 1-4: Cerberus' Trajectory & Sequence of Events

Date	Event	Δv (km/s)	Radius of Passage
2002 Jan 4	Depart Earth parking orbit	5.195	1.0437 planet radii
2003 Jun 11	Midcourse burn	0.418	
2004 Nov 25	Earth flyby	0.008	1.3000 planet radii
2005 Jan 23	Midcourse burn, Plane change	0.320	
2006 May27	Begin Jovian science (50 days)		
2006 Jun 21	Jupiter flyby: closest approach	0.000	26.780 planet radii
2020 Aug 18	Begin Plutonian science (50 days)		
2020 Sep 12	Pluto flyby: closest approach		

The redeeming values of this course were considered advantageous enough to outweigh the disadvantage of the long TOF. The long time is a disadvantage since the spacecraft, its instruments, and components have not been tested and proven for such a length of time.

Figure 1-1 graphically displays Δv versus launch date in 30 day intervals for a span of 420 days surrounding the launch date. Figure 1.2 graphically displays Δv versus launch date in one day increments for a span of two weeks surrounding the launch date.

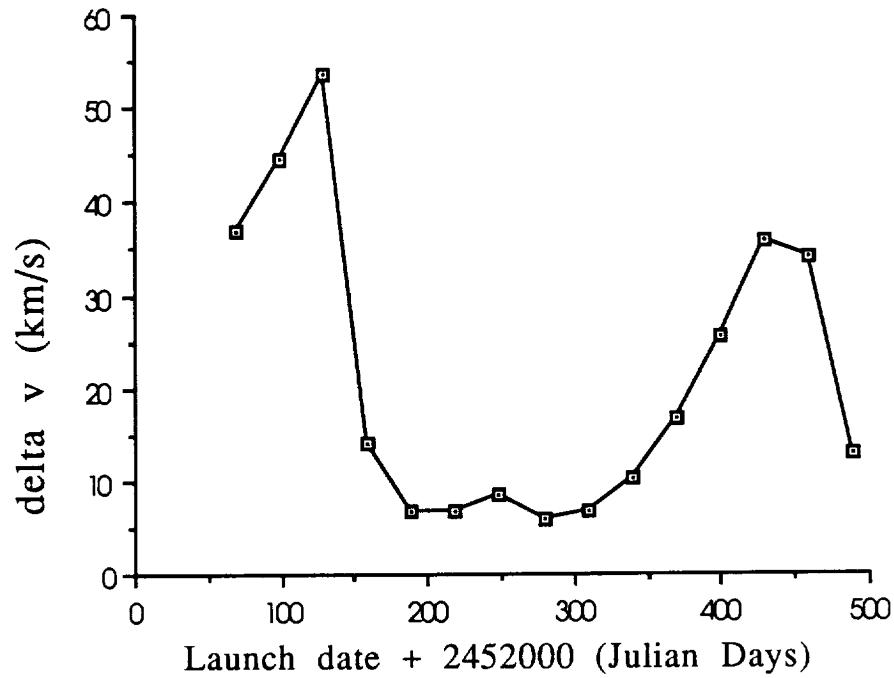

Figure 1-1: Comparison of Delta v versus Launch Date over 420 days

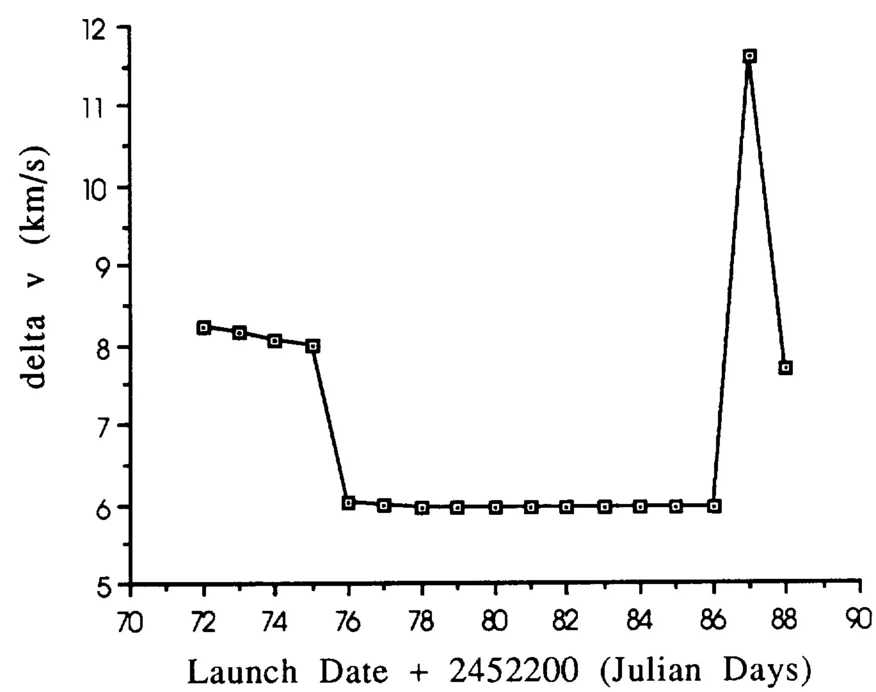

Figure 1-2: Comparison of Delta v versus Launch Date over 2 weeks

Section 1-4: MMPC EFFECTS ON OTHER SUBSYSTEMS

MMPC decisions naturally affect the other subsystems involved in the Cerberus mission, as all of these subsystems are integrated to accomplish one task.

The science analyst's decisions are influenced by the trajectory. Cerberus' course of flight determines which objects in space are available for study. Decisions are further affected by the specific proximity of the spacecraft to the aforementioned objects during flight, as experiments can be affected by closeness or distance to the object of scrutiny. The science analyst must determine from the trajectory the window of time available for study. In Cerberus' case, there are fifty days allotted for study at Pluto.

TOF is a concern of all functional subsystems: science, propulsion, attitude articulation & control, communication, and structures. If TOF exceeds the known service lifetime of a given unit, there may be concern for the ability of the unit to accomplish its ultimate task.

Closeness of passage to radiating bodies such as our sun and Jupiter is a structural concern. If flyby is too close to a radiating body, extra shielding must be added to the craft to protect it from undue damage. In Cerberus' case, the trajectory does not carry it close enough to the sun to merit unusual concern. Cerberus' path also falls far beyond the 'very safe' distance of ten planet radii when flying by Jupiter. The structure must also support the fuel mass determined by the Δv required.

The communication analyst must know when the craft will be in occultation behind a body. This knowledge is required in order to prepare autonomous control during this period without contact with the spacecraft.

The propulsion analyst must make decisions for that subsystem based upon data furnished by MMPC. Launch vehicle and propulsion system selection must reflect the needs stated for the trajectory. The spacecraft must carry with it the capability to perform midcourse burns when necessary.

Section 1-5: COSTING

The estimated cost of the Cerberus mission is $1,069,152,990.00 in February 1990 dollars. Costing data are displayed in Table 1-6 (Costing Spreadsheet). Costs are broken down under three headings:

1) Development Project - Flight Hardware
2) Development Project - Support Functions
3) Flight Project

The costs are evaluated for each unit and are totalled to produce the mission labor cost and the mission total cost. The sources of the equations and conversion factors used to determine these values are listed on the Reference page.[6,7] The conversion factor from FY77 producer dollars to February 1990 producer dollars is 1.815 (see Appendix A-2).[1,5] Mission costs by category are displayed as a bar graph in Figure 1-3.

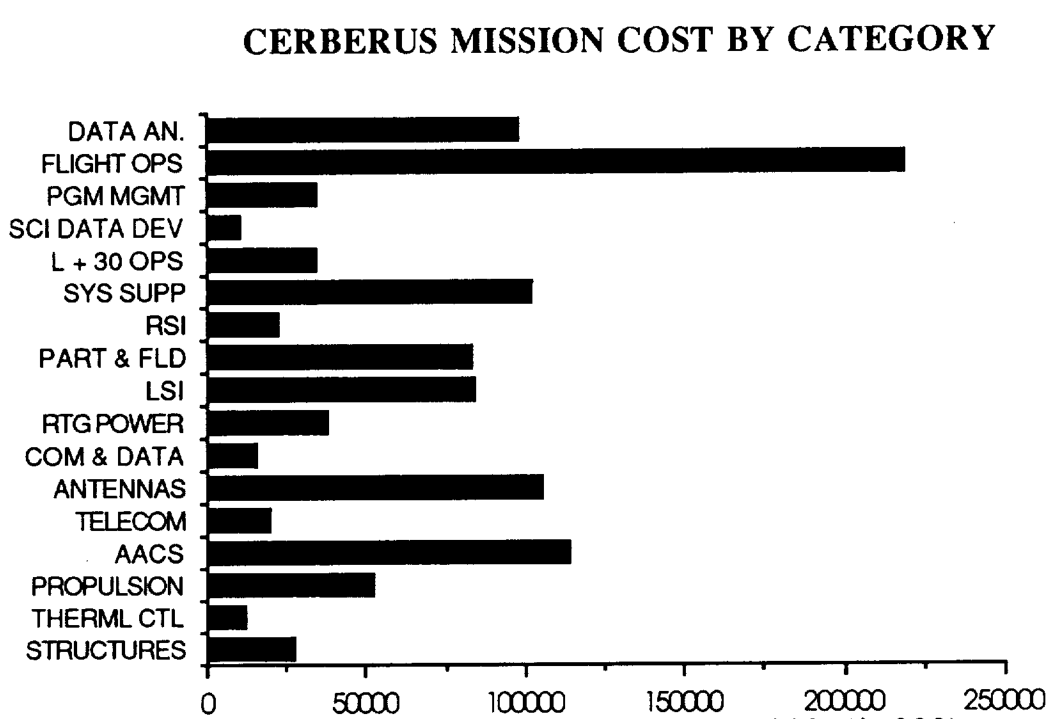

Figure 1-3: Category Cost - 1990 ($1000)

Number of Spacecraft: 4

DEVELOPMENT PROJECT - FLIGHT HARDWARE

FLIGHT HARDWARE TOTAL COST: $572892850.01

STRUCTURES AND DEVICES

MASS (kg)	168.00	ADJUSTED
DLH	419.99	262.49
RLH	178.15	178.15

SYSTEM SUBTOTAL: 440.64

THERMAL CONTROL, CABLING, AND PYROTECHNICS

MASS (kg)	82.00	ADJUSTED
DLH	276.86	69.22
RLH	126.47	126.47

SYSTEM SUBTOTAL: 195.68

PROPULSION

MASS (kg)	90.00	ADJUSTED
DLH	596.57	551.82
RLH	201.13	201.13

SYSTEM SUBTOTAL: 752.96

ATTITUDE ARTICULATION AND CONTROL

MASS (kg)	85.00	ADJUSTED
DLH	1202.55	1112.36
RLH	656.88	656.88

SYSTEM SUBTOTAL: 1769.24

TELECOMMUNICATION

MASS (kg)	10.00	ADJUSTED
DLH	189.44	189.44
RLH	130.37	130.37

SYSTEM SUBTOTAL: 319.80

ANTENNAS

MASS (kg)	40.00	ADJUSTED
DLH	1147.07	1147.07
RLH	534.24	534.24

SYSTEM SUBTOTAL: 1681.31

COMMAND AND DATA HANDLING

MASS (kg)	15.00	ADJUSTED
DLH	186.03	186.03
RLH	90.33	90.33

SYSTEM SUBTOTAL: 276.37

Table 1-6: Cerberus Mission Costing Spreadsheet Values (page 1 of 3)
Note: Labor Hours listed in 1000's of hours
Dollar amounts given in February 1990 producer dollars

MITG POWER

MASS (kg)		50.00	ADJUSTED
	DLH	547.61	342.26
	RLH	348.15	348.15

SYSTEM SUBTOTAL: 690.40

LINE SCAN IMAGING

MASS (kg)		29.70	ADJUSTED
	DLH	761.60	761.60
	RLH	450.03	450.03

SYSTEM SUBTOTAL: 1211.64

PARTICLE AND FIELD INSTRUMENTS

MASS (kg)		28.00	ADJUSTED
	DLH	692.84	692.84
	RLH	577.58	577.58

SYSTEM SUBTOTAL: 1270.42

REMOTE SENSING INSTRUMENTS

MASS (kg)		27.00	ADJUSTED
	DLH	305.49	305.49
	RLH	40.20	40.20

SYSTEM SUBTOTAL: 345.69

TOTAL HARDWARE ADJUSTED DLH	5620.61
TOTAL HARDWARE ADJUSTED DLH + RLH	8954.15

DEVELOPMENT PROJECT - SUPPORT FUNCTIONS

SUPPORT FUNCTIONS TOTAL COST: $180416000.00

SYSTEM SUPPORT AND GROUND EQUIP.
DLH 1732.94

LAUNCH + 30 DAYS OPERATIONS AND GROUND SOFTWARE

DLH 551.27

SCIENCE DATA DEVELOPMENT
DLH 108.25

PROGRAM MANANGEMENT/MA&E
DLH 601.40

Table 1-6: Cerberus Mission Costing Spreadsheet Values (page 2 of 3)
Note: Labor Hours listed in 1000's of hours
Dollar amounts given in February 1990 producer dollars

FLIGHT PROJECT

FLIGHT PROJECT TOTAL COST:	$180416000.00

FLIGHT OPERATIONS
DLH 3544.50

DATA ANALYSIS
DLH 1506.41

TOTAL MISSION DLH	13665.39
TOTAL MISSION DLH + RLH	16998.93
MISSION TOTAL COST:	$1069152990.01

Table 1-6: Cerberus Mission Costing Spreadsheet Values (page 3 of 3)
Note: Labor Hours listed in 1000's of hours
Dollar amounts given in February 1990 producer dollars

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Appendix A-1: Acronyms

AU	Astronomical Unit (149.6E6 km)
Δv	change in velocity (km/s)
DLH	Direct Labor Hours
km	kilometer
L	Launch
MMPC	Mission Management, Planning, and Costing
MULIMP	Multiple Impulse Optimizing Program
RFP	Request for Proposal
s	second
TOF	Time of Flight
VEEGA	Venus - Earth - Earth Gravity Assist

Appendix A-2: Equations and Conversion Factors

Hohmann Transfer

r_{Venus} = .723 AU

r_{Earth} = 1 AU

r_{Mars} = 1.52 AU

$r_{Jupiter}$ = 5.203 AU

$r_{Pluto(average)}$ = 39.4 AU

$r_{Pluto(perihelion)}$ = 29.6 AU

$\mu_{Sun} = 1.327E11\ km^3/s^2$

$R = r_2/r_1$

$\Delta v_1/v_{c1} = (2R/(1+R))^{.5} - 1$

$v_{c1} = (\mu_{Sun}/r_1)^{.5}$

$\Delta v_2/v_{c1} = R^{(-.5)} - (2/(R(1+R)))^{.5}$

$a = (r_1 + r_2)/2$

$t_{Hohmann} = \pi(a^3/\mu_{Sun})^{.5}$

Costing

Purchasing Power of Dollar

	FY77	1982/1984	February 1990
Consumer	1.649	1.00	.782
Producer	1.546	1.00	.852

Consumer Dollar Conversion Factor: 2.1088

Producer Dollar Conversion Factor: 1.8150

Section 2: STRUCTURES

Section 2-1: LISTING OF REQUIREMENTS

Easily the most difficult part of designing any spacecraft is dealing with the vague, and often contradictory, requirements of the mission. Requirements such as the ones listed in the Request for Proposal (RFP) identify the objectives of the design, and it is up to the analyst to achieve the optimum solution. Listed at the beginning of this proposal are the requirements that must be met by all subsystems. Concepts such as minimizing cost, keeping the design simple and reliable, etc., must be on the mind of the analyst at all times. The most important part of preliminary design is meeting as many, if not all, of the requirements outlined by the RFP.

As well as meeting the overall objectives of the mission, each subsystem must also satisfy many derived requirements. These derived requirements are based on the objectives outlined in the RFP, but they are specific to the subsystem.

For the Structures Subsystem, the derived requirements have a great deal to do with the overall design of the spacecraft (spacecraft). Below is a listing of the derived requirements for the Structures Subsystem. All of them are objectives specific to this subsystem, but they are based on concepts outlined in the RFP.

1) Overall design of the Cerberus spacecraft
2) Design to maximize science performance
3) Calculation of inertia properties of spacecraft
4) Material selection for the various components
5) Thermal control considerations
6) Verification of Launch Vehicle Compatibility
7) Identification of any On-Orbit Assembly (OnOA)
8) Structural analysis of truss bays to meet launch conditions
9) Identification of subsystem interaction

The RFP requirements kept in mind when designing the spacecraft as well as the overall mission plan agreed on by the group. In essence, Cerberus is meant to be a conservative project. With the Moon/Mars initiative the centerpiece of both NASA's and Congress' space commitment, it was believed that this was the best approach. Nothing was to be done that would overshadow these important advances. From the beginning, Cerberus was meant to be an inexpensive mission. This meant that as much off-the-shelf hardware should be used. Fortunately, with the Mariner Mark II (MMII) program just beginning, it was believed that a good amount of off-the-shelf hardware would be available.

This conservative, off-the-shelf approach is well reflected by the structural design methodology. Older missions such as Voyager and Galileo were studied to understand what problems they had and the solutions to those problems. The more recent MMII program gave valuable insight into new methods of spacecraft design. The most important part of studying these previous missions was understanding how the mission to Pluto differed. With a mission lifetime of 18.7 years, and a safety margin of approximately 5 years, it is easy to see the major differences between the projects. Only Voyager has come close to having a mission lifetime of this magnitude. The conservative basis of Cerberus is well-suited to meeting this difficult requirement. Material selection, spacecraft configuration, and thermal design all reflect this overall mission plan.

The most important requirement for the structural designer was ensuring Cerberus ability to carry out its mission. The objective of a Pluto probe is to gather as much scientific data as possible, in the most efficient and cost-effective way. This must be kept in mind at all times when designing the spacecraft. It is believed that this design, with its conservative and off-the-shelf approach, is the most effective and cost-efficient way to successfully complete the long journey to Pluto.

Section 2-2: MAJOR DESIGN FEATURES

The most important part of the Structures subsystem is determining the overall layout of Cerberus. There is a great deal of system interaction that occurs for an effective design. Not only must the requirements of the RFP be satisfied, but any constraints imposed by the different subsystems must be taken into consideration.

Figure 2-1 represents a view of Cerberus from the bottom of the spacecraft (see next page). Some of the major design features are visible from

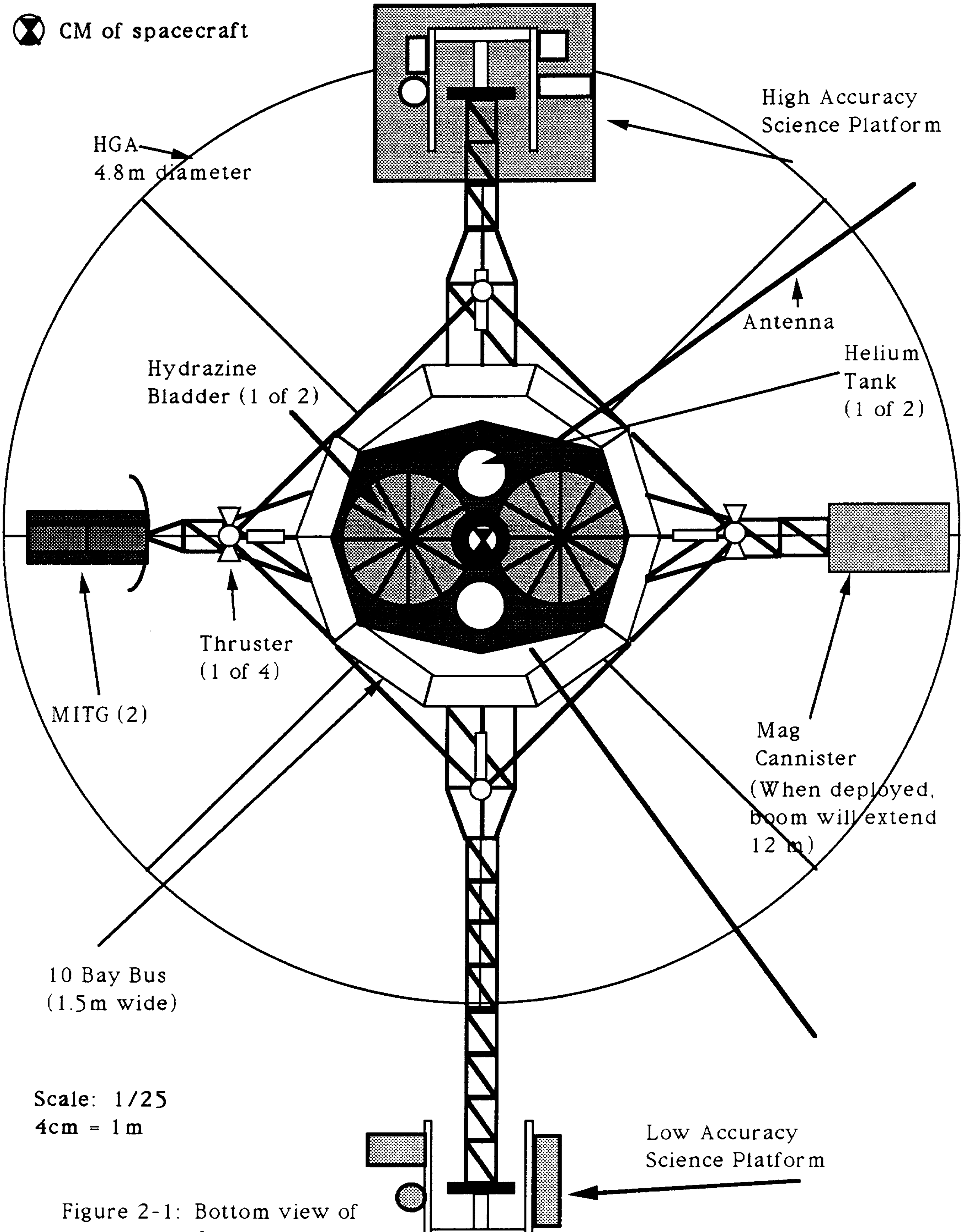

Figure 2-1: Bottom view of Cerberus

this angle. A 10 bay bus is the major structural component of the spacecraft. It contains the electronics that control the mission to Pluto. The following is a list of the major design features, grouped by subsystem.

Table 1-1: Summary of major design features

Science	• Cerberus contains two science platforms, 1 for High Accuracy science (HAP), the other for Low Accuracy science (LAP). Both platforms have a good field of view, maximizing the science performance of the instruments. • The HAP is protected by a radiation shield. This will minimize the effects of the space environment by protecting the instruments from both radiation and micrometeroid impact. • The MITGs have been placed opposite the magnetometer to eliminate interference. Over 14.5 m separates the two components. • A dipole antenna has been placed on Cerberus for radio science. The two antennas were placed at a ,90 angle to each other to satisfy science requirements.
CCC	• A 4.8 m High Gain Antenna (HGA) is the component that is responsible for telemetry. It has been placed on top of the 10 bay bus. It can be folded for launch.

Power & Propulsion	• Two MITGs are the main power source for Cerberus. They have been placed on a boom .8 m away from the bus. • A propulsion module that will provide necessary Δv's for the spacecraft is attached at the bottom of the bus. It's consists of 2 bladders, 2 tanks and a support structure. Two large bladders (r=.335 m) are for the Hydrazine propellant, two small aluminum tanks (r=.110 m) contain the Helium pressurant. A plate made of titanium is the material for the support structure.
Attitude, Articul-ation and Control	• There are 4 sets of thrusters that control the attitude of the Cerberus spacecraft. They are placed along the principal axes of the craft. • A star tracker and sun sensor have been placed on the HAP for inertial reference.
Mission Manage-ment and Planning	• The dimensions of the spacecraft have been sized to conform to the launch vehicle.

Since the scientific instruments will be fully operational during the whole flight, there is nothing in the design that precludes it from performing several possible missions.

The overall design approach reflects the conservative nature of the mission. Most of the equipment is off-the-shelf hardware, with inheritance from both Galileo and the more recent MMII program. Cerberus is simpler than Galileo, with no spun sections, and only one bus. Most of the structural design reflects the newer MMII program [1]. The 10 bay bus was chosen to take advantage of any extras from this program, since the Comet Rendezvous and Asteroid Flyby (CRAF) uses this configuration. The two scan platforms also reflect the CRAF inheritance, as does the short MITG boom and the long (12m) magnetometer truss structure. All of these design features represent compliance with the requirement to maximize use of off-the-shelf hardware.

Figure 2-2 represents a side view of the Cerberus spacecraft. This angle shows two features that differ from the CRAF configuration. The first is the Galileo type antenna (4.8m diameter) that will provide telemetry for the mission. Again, even this feature is inherited from the Galileo project, so it does not represent a major redesign. The propulsion module also represents a change from MMII. It is smaller, fitting inside the 1.5m space inside the bus. Although different from MMII, it is not much different than modules used for other interplanetary missions.

As mentioned before, the 10 bay bus contains the electronics that will control the mission to Pluto. Each bay contains electronics for one subsystem, although each subsystem was assigned more than one bay. Figure 2-3 is a graphical representation of each of the bay assignments.

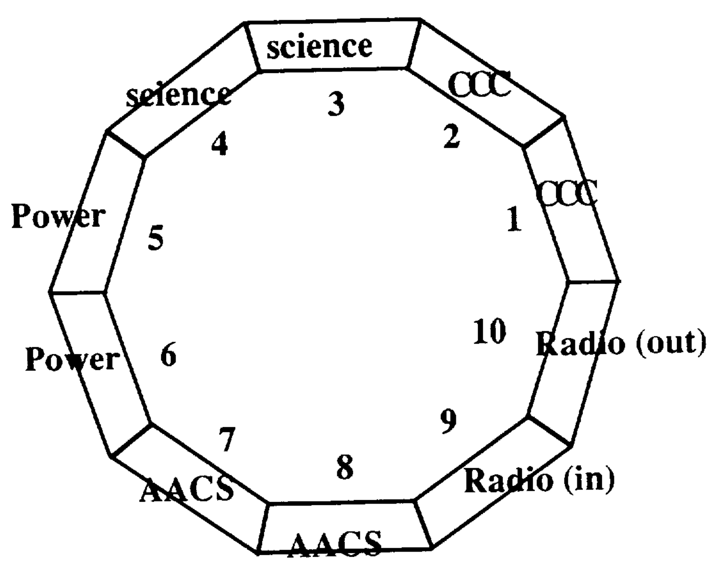

Figure 2-3: Electronics contained in each bay

The bay assignments were selected to minimize cabling and also to even out the inertia properties of the electronics stored in the bus.

Section 2-3: LAYOUT OF COMPONENTS/INERTIA PROPERTIES

Now that the major design features of the Cerberus have been discussed, the exact layout of the components and the inertia properties will be studied.

The location of the components is directly related to the inertia properties of the spacecraft. In turn, the problem of attitude control is heavily dependent on the inertia properties. To make the attitude control problem as simple as possible, there were certain objectives of the design. These objectives are listed below.

1) The center of mass of the whole structure should be kept near the middle of the bus. A good location was selected at (0,0,-.40).

2) The off-diagonal terms of the inertia tensor should be kept at a minimum. This means that the principal directions of the spacecraft lie along the directions chosen for the initial layout (See Figures 2-1 and 2-2 for these axes).

An initial configuration for Cerberus was drawn up. In the preliminary phase, close attention was paid to the symmetry of the spacecraft. The two science platforms were placed on opposite ends of the bus since it was believed that their inertia properties would even out. The magnetometer and the RTGs were then placed at 90° angles to these platforms. Constraints imposed on the design were kept in mind at all times. Most importantly, the RTGs were kept far from the magnetometer, and the HAP and the LAP were given a good field of view. The size of the propulsion module enabled it to be placed inside the 1.5m wide electronics bus.

Two iterations resulted in the final placement of the spacecraft components. The inertia matrices for each of these iterations, as well as the center of mass (CM) and principle directions, are given in Appendix B-1. Only the final inertia tensor will be given here. A summary of how the configuration changed is below in Table 2-2.

Table 2-2: Summary of Configuration Changes

Initial Design	The off-diagonal terms of the inertia matrix were small and the eigenvectors were close to the chosen axes. The CM was 25 cm too far in the x-direction and 12 cm in the y-direction.
After 1st iteration	The LAP was moved .5m further away from the spacecraft and the HAP was moved .6m closer to the bus. The inertia matrix and eigenvectors were still good. The CM was still 11cm too far in the x-direction and 9cm too far in the y-direction.
After 2nd iteration	The MITGs were moved .5m closer to the bus. 5 kg of mass (considered thermal protection) was added to the LAP. This resulted in the final inertia properties.

The propellant weight was added to the bladders and the total inertia of Cerberus plus Hydrazine was obtained. The principle inertia matrix is given below.

Principle Inertia Matrix	$\begin{bmatrix} 2192 & 0 & 0 \\ 0 & 886 & 0 \\ 0 & 0 & 2800 \end{bmatrix}$

The location of the CM is shown in Figures 2-1 and 2-2. Once the CM and the eigenvectors were found, the thrusters were placed. The thruster placement is also shown the above mentioned figures.

The final inertia properties of Cerberus represent a satisfactory configuration. Both the principal directions and the CM are well-placed, allowing for easy attitude control. Also, the principal inertias are not outside the range of normal thruster sizes. The final configuration of the spacecraft allows for precise attitude control, thereby increasing the science performance of the flyby mission.

One configuration problem that needed to be addressed was "center of mass migration" (author's term). This problem is the result of propellant loss during the mission. As the Hydrazine bladders empty, the CM will "migrate" away from its original position. This could cause an attitude and control problem during the flight or, more importantly, at Pluto rendezvous. This effect was studied by calculating the inertia properties of Cerberus at varying stages of propellant loss. The beginning of the mission was assumed to have 0% loss, the end 100%. It was found that the principal inertias changed by 2.4%, which is not a significant amount. A more important change occurred in the location of the CM. The results are plotted below in Figure 2-4.

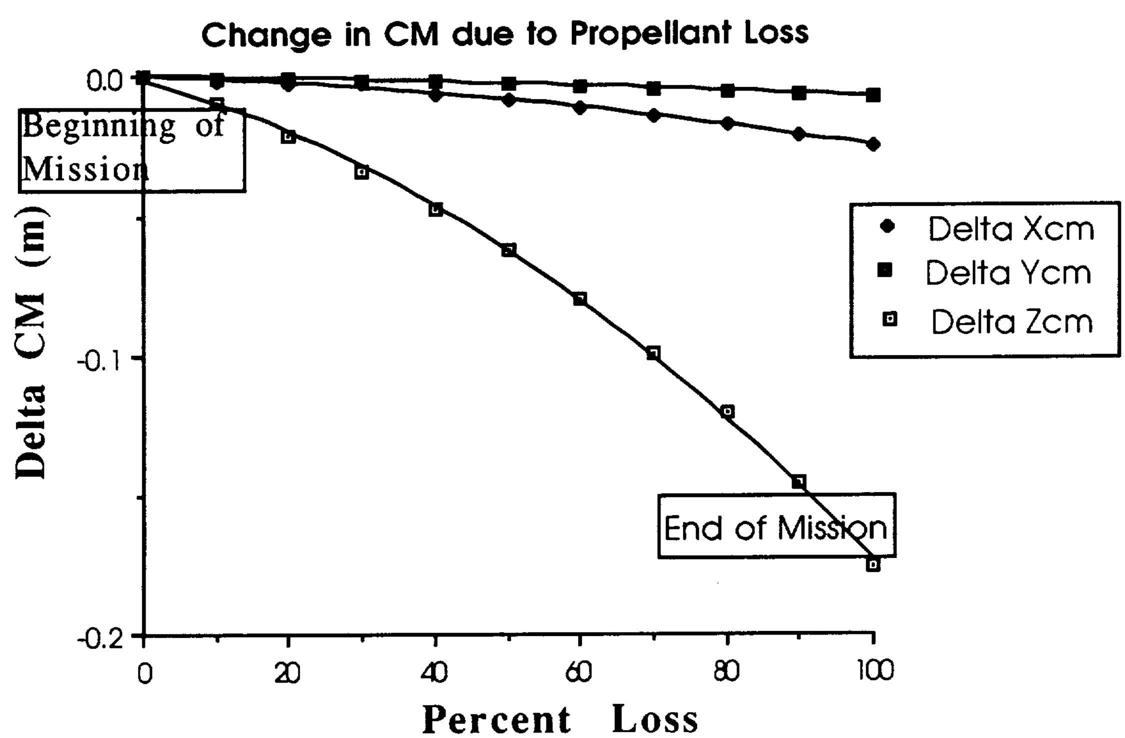

Figure 2-4: Center of Mass "Migration"

Figure 2-4 shows that the parameter most affected by propellant loss is the z component of the center of mass. The negative sign in ΔZcm appears because of the chosen direction for the z-axis. The CM is moving 'up' the spacecraft, by as much as 17.53cm for empty bladders. This could cause attitude control problems as the fuel is expended.

A simple solution to this problem exists. Since all four of the booms have the ability to move up and down, due to the launch requirements (see Section 2-6), they can be used to counteract this mass loss. By moving both the MITG and magnetometer booms .5m in the positive z direction, the CM of mass can be placed only 6cm away from its original position. This also assumes that 30% of the propellant is left in the bladders, which is slightly over the excess allotted for the mission. It may be possible to do even better by lowering the science platforms, but this is not desirable. Moving the science platforms might cause a loss in pointing accuracy, which in turn would degrade science performance. A 6cm movement in the CM is much better performance than a 17.5cm shift. This is not expected to cause any attitude and control problems.

Section 2-4: MATERIAL SELECTION

One of the most important requirements of spacecraft design is weight minimization. Weight minimization is dependent on the materials selected for the spacecraft. Ideally, a designer will choose the lightest available materials. Unfortunately, this is not the only factor involved. The materials must be space-proven and reliable, thereby maximizing the chances for a successful mission. Depending on the application, the material must have a high yield strength, so it will not fail at launch. All these requirements must be taken into account when selecting the materials for the mission to Pluto.

The first stage to selecting materials was to gather important properties. Table 2-3 is a list of commonly used space materials and their relevant properties. [2]

Table 2-3: Properties of materials for space use

Material	density (kg/m^3)	yield strength (MPa)	$\frac{\rho}{\sigma_y}$
Al-2024	2801.55	375.20	7.47
Al-2219	2829.29	369.33	7.66
Beryllium	1858.45	381.06	4.88
Mg-Hm21A	1775.24	193.46	9.18
Stainless Steel	7933.10	1084.55	7.31
Ti-6Al-4V	4549.05	785.57	5.79
Graphite Epoxy	1941.67	1055.24	1.84
Graphite Al	3051.19	586.25	5.20

Table 2-3 displays the varying material properties. These properties are not the only factor that must be taken into account when selecting materials. Given the conservative, low cost nature of the Cerberus mission, it is very important to look at the reliability factor. The most commonly used material for space applications is aluminum. It is easy to form and has proven its worth.[2] Titanium is also space-proven, as well as steel. Many of the other materials in Table 2-3 do not have the reliability and ease of fabrication that the above materials provide. The composites are very expensive and are subject to degradation in UV environment.[2] Berylliums are difficult to form, and are toxic. Magnesiums are difficult to weld, which increases fabrication cost.

In keeping with the overall Cerberus objective of designing a conservative, inexpensive spacecraft, the materials selected are all space-proven and relatively easy to fabricate. Table 2-4 summarizes these decisions.

Table 2-1: Material Comparison

Aluminum	The material that has been the mainstay of spacecraft. Will be used for the 10 bay bus and most of the truss elements. Also used for science platforms and the radiation and MITG shields.
Titanium	Given its lower weight to strength ratio, titanium will be used wherever load is carried. This includes the support plate for the propulsion module and some of the truss members.
Steel	Although heavier than aluminum and without the excellent weight to strength ratio of titanium, steel will still be used for pins, springs, etc.

The most important trade-off involves selecting a material for the 10 bay bus. Figure 2-5 displays the relative masses of a 1.5m wide,10 bay bus.

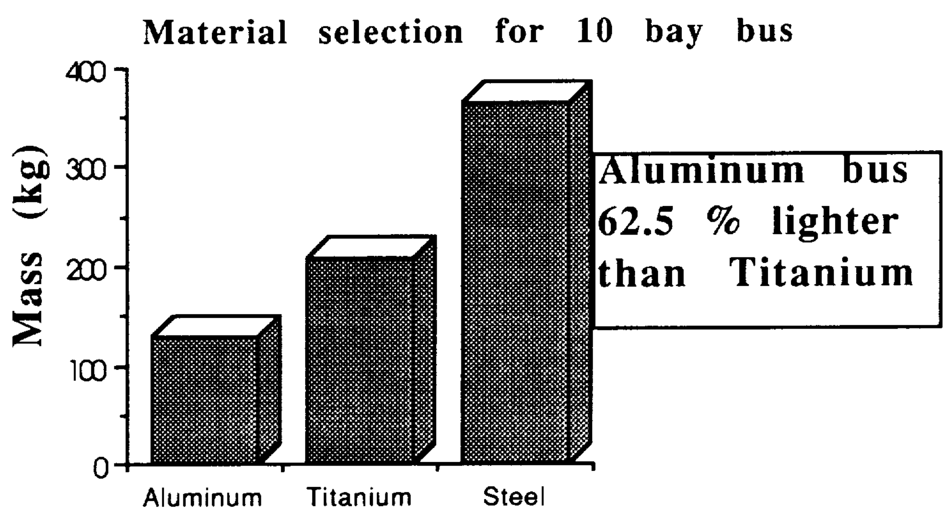

Figure 2-5: Material Selection for 10 bay bus

The aluminum bus definitely satisfies the minimum weight requirement, being 62.5% lighter than a titanium bus of the same configuration. Steel is not even a consideration due to its weight.

Selection of aluminum is also in keeping with the requirement for simplicity and reliability. It is both space proven and easy to fabricate. It can easily be considered an off-the-shelf item, which again decreases cost of the overall mission.

In material selection for the truss members, the decision was made to use both aluminum and titanium. Aluminum will be used in the members that do not have to carry the high launch loads, while titanium will be the material for the truss elements that do. For a complete discussion of this analysis, see Appendix B-2.

Section 2-5: THERMAL CONSIDERATIONS

Given the length of Cerberus' mission to Pluto, the thermal control problem is one that needs to be studied closely. In general, there are two types of control that could be used: passive and active. Passive involves the use of insulating blankets and louvers to reduce the escape of heat from a component. Active control comes in two forms, electric heaters and Radioisotope Heating Units (RHUs). Table 2-4 summarizes the disadvantages and advantages of these three types of thermal control.

Table 2-4: Design Trade-Offs for Thermal Control

Type	Advantages	Disadvantages
Louvers	Used to emit heat from electrical components.	Only good for emitting heat.
Mylar Insulation	Insulation used to reduce heat loss from a component. Also good for micrometeroid protection.	Passive control system, might not last the complete mission.
Electric Heaters	Can control temperature over relatively large range.	Uses electrical power.
RHU	Uses no electrical power.	Supplies only 1 Watt of power. Need to be used in quantities.

The thermal control of Cerberus will involve a combination of all the different types listed in Table 2-4. The problem has been broken down into three areas: the electrical bus, the science platforms, and the propulsion module. Table 2-5 lists the thermal control for each area.

Table 2-5: Thermal Control for Cerberus

10 bay bus	A combination of heaters and Mylar insulation will be used. Louvers will be placed on the boom side of the bus to enable heat emissivity from the electrical components. For bays with no heat storage, Mylar insulation will be used to reduce heat loss.
LAP and HAP	Three types of control are necessary. When the instruments are not operating, electrical heaters will be used for temperature control. Mylar insulation will be used to reduce heat loss and for micrometeroid protection. Finally, louvers are placed on the 'inside' of the platform to allow heat to escape during periods of high instrument use.
Propulsion Module	Mylar blankets can be placed over the bladders and tanks to reduce heat loss. RHUs will be used (approx. 50 of them) for thermal control during flight. These will be placed on the support structure of the module.

This thermal control design is again reflective of the overall nature of the mission. Inheritance from MMII program can be seen [3], thereby increasing the use of off-the-shelf hardware. The system is redundant, especially in the important area of the science platforms. This is needed, given the length of the mission to Pluto.

One design change was looked into. This involved the use of waste heat from the RTGs for thermal control of the propulsion module. Two factors excluded this design. The first was that the propulsion module was small enough to fit inside the bus, thereby making it more efficient to use RHUs. The second problem was the fact that the RTGs will be moving down during the mission for inertia control. This was discussed in Section 2-3.

Section 2-6: LAUNCH VEHICLE COMPATIBILITY AND ON-ORBIT ASSEMBLY

To ensure that the Cerberus spacecraft would be compatible with the launch vehicle, three measures had to be taken. First, the bus could not be overly large. Second, the antenna must be foldable, like Galileo's. Finally, the booms must be movable to allow them to be folded down.

Figure 2-6 (see next page) shows Cerberus ready for launch. In this configuration, the spacecraft measures slightly under 3.4m from side to side and 4.4m from top to bottom. These dimensions are acceptable for modern launchers. The magnetometer boom has been retracted and the trusses have been folded down. These will be extended after the spacecraft begins its journey to Pluto. The truss that mates Cerberus with the launch vehicle will use explosive bolts to be jettisoned from the spacecraft after disengaging with the upper stage. It will not be carried along to Pluto.

It is obvious from the diagram that no on-orbit assembly (OnOA) is required. This is a major simplification to the overall mission. Although the deployment of the Space Station Freedom in the coming decade makes OnOA a possibility, there is no need for it. Adding the burden of OnOA only makes the mission more expensive and complicated. Making Cerberus compatible with existing launch vehicles and eliminating the need for OnOA makes the design reliable, more simple, and less expensive.

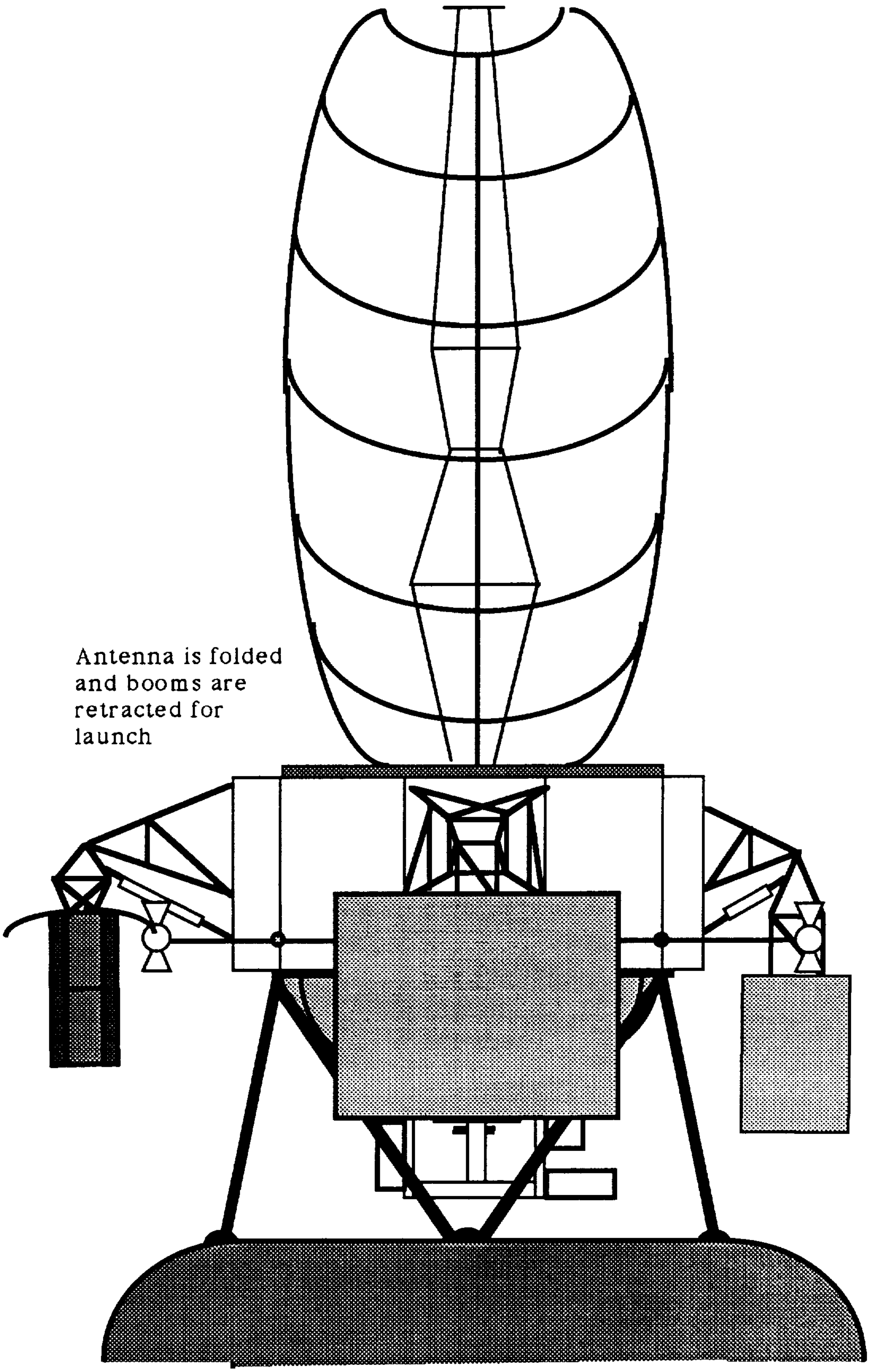

Figure 2-6: Cerberus in launch configuration

Numbered References

1. Draper, Ronald, "Comet Rendezvous Asteroid Flyby First MarinerMark II", Advances in Astronautical Sciences, v64 Part I: pp 399-418.

2. Koepke, Andy, "AAE 241 Lecture Notes from 2/22/90", Table of Material Properties

3. Tsuyuki, G. et al, "Thermal Design for the Comet Rendezvous Asteroid Flyby Spacecraft", AIAA Paper 89-1753, from AIAA 24th Thermophysics Conference.

4. "JPL, Division 35 Mass Estimates", Interoffice Memorandum, Nov. 24 1980, Jet Propulsion Laboratory.

5. McGill, David, and King, Wilton, **Engineering Mechanics: an Introduction to Statics and Dynamics**, PWS Engineering (Boston), 1985.

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Galileo: Exploration of Jupiter's System, NASA SP-479

Avila, A, et al, "Thermal Design of the Galileo Bus and Retro Propulsion Module", AIAA Paper 89-1749, AIAA 24th Thermophysics Conference.

Appendix B-1:
ITERATIONS OF THE SPACECRAFT CONFIGURATION

As mentioned in Section 2-3, there were two iterations done on the initial configuration. The data for those calculations will be given here. A complete summary of the location of the CM, the principal inertias, and the eigenvectors is included.

All CM values in m

All inertias in kg - m^2

After initial design:

x_{cm} = -.2552 y_{cm} = -.1196 z_{cm} = -.3901

Principal Inertias $\begin{bmatrix} 980 & 0 & 0 \\ 0 & 883 & 0 \\ 0 & 0 & 1664 \end{bmatrix}$

Principal Directions $\begin{bmatrix} .9995 & .03050 & 0 \\ -.0305 & .9994 & 0 \\ 0 & 0 & 1 \end{bmatrix}$

After 1st iteration:

x_{cm} = -.1136 y_{cm} = -.0905 z_{cm} = -.3909

Principal Inertias $\begin{bmatrix} 1336 & 0 & 0 \\ 0 & 810 & 0 \\ 0 & 0 & 1948 \end{bmatrix}$

Principal Directions the same

After 2nd iteration:

x_{cm} = -.0571 y_{cm} = -.0180 z_{cm} = -.3917

Principal Inertias $\begin{bmatrix} 2085 & 0 & 0 \\ 0 & 845 & 0 \\ 0 & 0 & 2733 \end{bmatrix}$

Principal Directions the same

The final inertia matrix of Cerberus plus propellant is given in Section 2-3 of the report.

The inertia matrices and CM locations were presented here to display how these quantities changed during the optimization process. The values for the CM locations are relative to an origin placed in the middle of the thruster nozzle (See Figure 2-1). The final position of the CM is shown in these figures.

Minimization of Truss Mass

The most intense loading on the structure will occur during launch form Earth. The acceleration of the spacecraft upward inside the launcher will produce forces well above the normal 1g felt by the stationary craft. Since the 10 bay bus and antenna are off-the shelf items, it was assumed that they would be able to handle the launch loads without damage. The only analysis left to do is on the trusses that will be stressed during launch.

The fact that all four booms will be stowed during launch introduces the necessity for a stress analysis of the supporting truss structure. The configuration and labels are shown in Figure 2-7. A simple truss analysis yields the maximum force in the structure [5].

$$F_{max} = \frac{P\ b}{a\ \cos\ \theta} \qquad \text{(Eq. B2-1)}$$

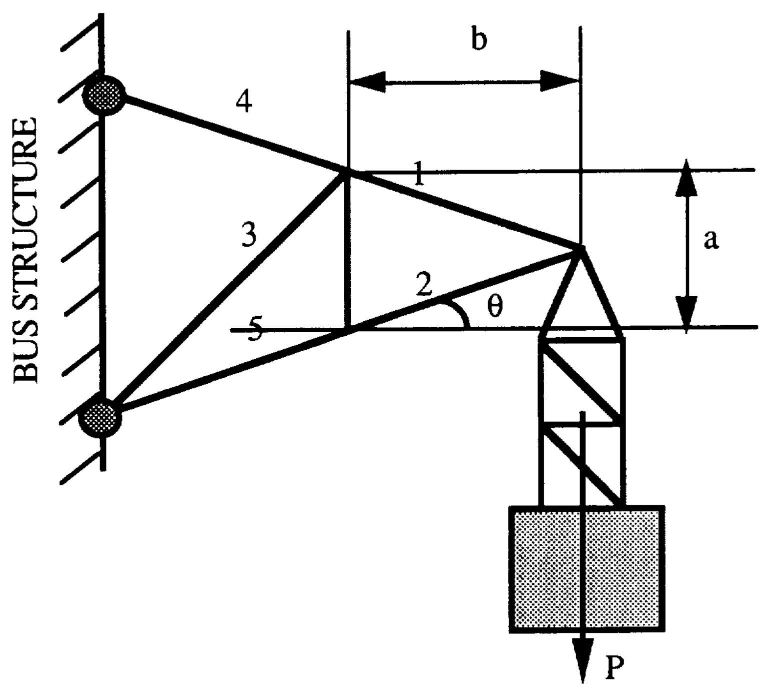

Figure B2-7: Truss configuration during launch

P is considered to be the mass multiplied by the quasistatic load factor. For this analysis, the load factor was taken to be 10g [4]. The maximum stress in the structure is Eq. 2.1 divided by the area of the element. This stress must be less than the yield strength divided by the safety factor. Using this concept and the geometry of the structure, the mass of an element can be found.

$$Mass_{element} = \frac{P\ SF\ L}{2\ \sin\theta}\left(\frac{\rho}{\sigma_y}\right) \qquad \text{(Eq. B2-2)}$$

SF = Safety Factor

L = Length of element

The major finding of Eq. B2-2 is that the the mass of the truss elements will be minimized by the lowest weight to strength ratio. This assumes equivalent geometry. For this reason, the truss elements that must carry the load during launch will be made of titanium. The other truss elements, especially the members that are retracted in the mag cannister, can be made of aluminum, since they are lighter and don't need to carry as much structural load. This will save weight on the overall spacecraft configuration.

Section 3: POWER AND PROPULSION

Section 3-1: INTRODUCTION

In the integration of Cerberus' power and propulsion systems, there were key requirements that drove each design. The method of attack was to first identify the requirements for each system. Of secondary importance is the identification of interfaces with other subsystems of the spacecraft. Finally, to meet the requirement of feasibility and cost, the power and propulsion system was discretized into three components: power, propulsion module and Earth Launch Vehicle (ELV)/ upperstage transfer vehicle.

The power systems main concern was lifetime, since it will take approximately 19 years to reach Plutonian space. Lifetime of this system will not preclude it from fulfilling its mission or other possible missions. The reliability is governed by the space worthiness of the off-the-shelf items used. By interfacing with all the other subsystems, the peak power usage was determined for a worst case scenario.

For the propulsion module, the Δv's needed for the mission were the driving factors. Simplicity, reliability and space worthiness, of course, played an important role in the final design. Tight integration with the mission planning group produced reasonable Δv's that minimized weight and complexity of the overall design. Masses from all groups were then needed to get a spacecraft dry weight. This was used to calculate total amount of fuel needed and final module weight. A final integration with the structure group was then done.

The driving factor for ELV/upperstage transfer vehicle was the minimization of on-orbit assembly. It is felt here that for the proposed flyby no on-orbit assembly should need to be done, thus reducing risk to manpower and increasing cost savings. Also, the reliability and availability of these vehicles were critical considerations.

This, again, enhanced the need for reasonable Δv's, especially at Earth. Integration with all groups was then completed in order to fit Cerberus into an existing or near existing transfer vehicle with no need for on-orbit assembly.

Each sub-system is discussed in detail below. Final consideration in the method of attack of feasibility and cost is then addressed.

Section 3-2: POWER

For a peak power estimate, a worst condition case was asked of each subsystem. This scenario allowed for the highest level of power each system could need. These powers were then added in series simulation an all systems on condition. Table3-1 gives these results for each system and the total for the spacecraft.

Table 3-1: Peak power estimates by subsystem

Subsystem	Power (W)
Science	30
CCC	60
AACS	60
Propulsion	15
TOTAL	165

For accessories and propulsion this took into account the heating of valves and catalyst-beds for the system. Structural thermal control was done by separate units supplying their own heat.

This total number is a conservative power estimate. To allow for error, a 30% safety factor was added, bringing this estimate to 214.5 Watts. Since the length of time to Plutonian space is 19 years, it is the necessary to find a power source that can provide this peak power for the duration of the flight. This will ensure successful completion of the mission. Also, there is a chance that the power source will be operational after the rendezvous with Pluto.

The existing tested and flight proven sources are batteries, solar cells, and Radio Isotope Thermoelectric Generators (RTGs). Of these, use of RTGs is the only viable option for a mission of this duration. Batteries will not last 19 years and solar cells will be useless at 40 A.U. Other sources are presently being developed (e.g. heat stirling engines and small nuclear reactors) but are not yet reliable for long-term use. RTGs, however, have a proven track record.

Those employed in Voyager have been operating for over 14 years and have shown excellent performance [5]. The latest RTGs are built around the General Purpose Heat Source (GPHS). Figure 3-1 displays a typical GPHS/RTG[1].

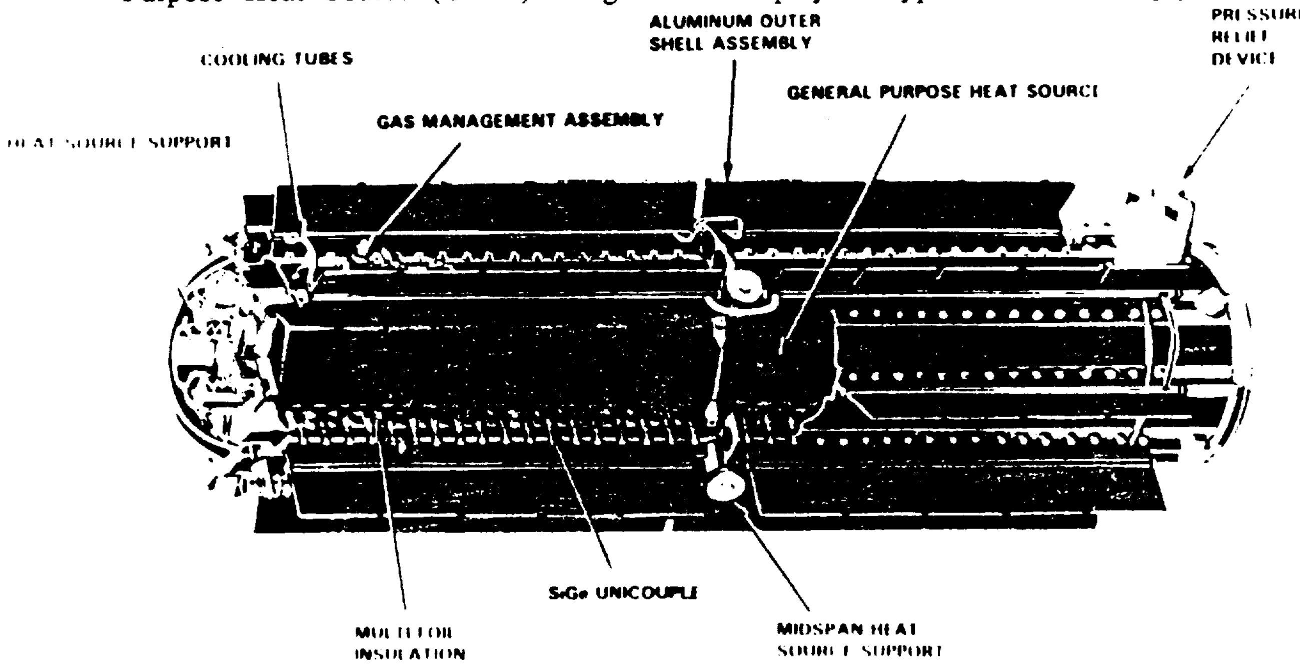

Figure 3-1: GPHS/MITG

A new design that uses the GPHS shown in Figure 3-2 is called the Modular Isotopic Thermoelectric Generator (MITG) incorporates better features than the GPHS/RTG[8]. The advancements are due to better design and new materials that increase the conversion efficiency. These features are listed in Table 3-2. A typical MITG "slice", shown in Figure 3-3, produces approximately 24 Watts at 28 Volts with an initial thermal load of 250 Watts.

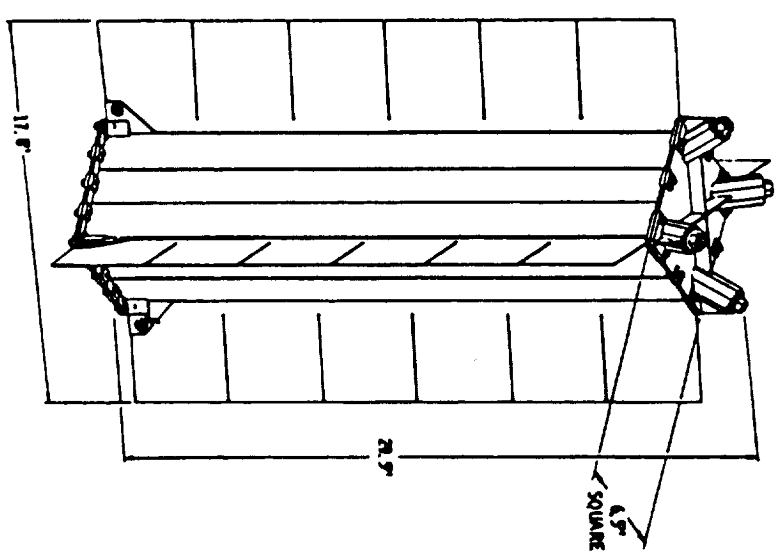

Figure 3-2: Showing the MITG unit

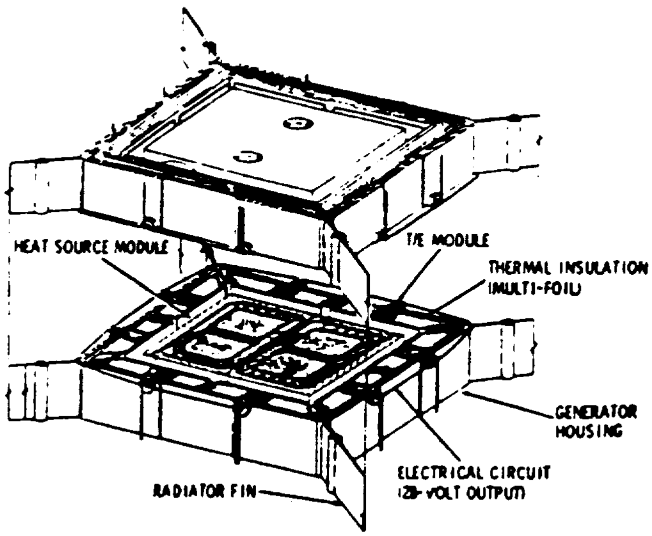

Figure3-3: Showing a typical MITG slice

Table 3-2: Advancements for MITG

1	Adaptable to wide range of power since it is available in standard 24 Watt slices
2	When varying number of slices, only the redesign of housing is necessary
3	Performance of each slice can be checked individually
4	The series parallel circuit permits high redundancy
5	Lighter weight (higher specific power)

A comparison was done by the producer of the MITG (Fairchild Space and Electronic Company) between a typical 290 Watt GPHS/RTG and a similar

282 Watt 12 slice MITG[8]. The results of weight differences is given in Table 3-3.

Table 3-3: Showing weight advantage of MITG over a typical RTG

[illegible]			Δ	[illegible]		
[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	[illegible]
[illegible]			[illegible]			
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	
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[illegible]		[illegible]	[illegible]	[illegible]		[illegible]

The figures show that for approximately the same power output the MITG design weighs about half as much. This weight savings relates to a cost savings while still using a reliable and proven GPHS, therefore satisfying mission requirements. For Cerberus, two 11- slice MITGs are slated to be used. Each MITG is capable of providing the spacecraft with the necessary power, thus achieving 100% redundancy. As mentioned before, a 30% safety factor has been incorporated. This has been done for sources of errors in predicting performance such as effect of fuel decay on power transfer, uncertainty in the amount of dopant precipitation in the thermal electric material, loss due to oxygen diffusion in $^{238}PuO_2$ pellets, and uncertainty in establishing power profile throughout the mission [2]. For 11 slices, Figure 3-4 shows that an optimum can be found at the elbow of this curve. At MITG weight of 24.95 kg (55lb), the resulting power supply for Cerberus is then given in Table 3-4. The equations for the calculations are shown in Appendix C.

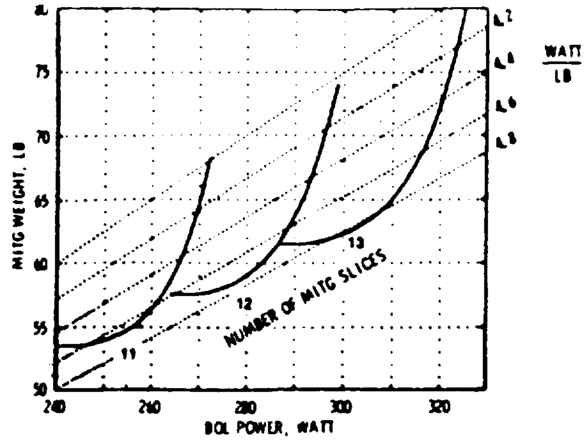


Figure 3-4: Performance curves for MITG

Table 3-4: Cerberus Power Supply (MITG)

	Thermal Load/MITG (Watts)	Output/MITG (Watts)	% Above Peak (Watts)
At Loading	2750	259	57
After 19 Years	2369	222	35

Specific power 10.36 Watts/kg. Total weight = 49.9kg

NOTE: The 11 slice system saves a total of 2.21 kg/MITG over the 12 slice system while easily satisfying power requirements

It can be seen that the 11 slice MITG slightly exceeds the 30% safety factor. If exact power was desired, the MITGs could be fined tuned simply by adjusting the radiator fin lengths. The power conditioner along with the computer will regulate and condition power according to the needs of the spacecraft. This will be done by autonomous sensing and programming that will periodically review the system.

In conclusion, with the conservative approach taken, it is felt that this is a feasible and cost effective system due to its savings in weight and use of off-shelf items. If necessary, this system could easily be up or down-sized for the possibility of other missions such as the measurement of heliopause or a change in launch date.

Section 3-3: PROPULSION MODULE

After the spacecraft is delivered on to the transfer orbit, the propulsion module's function is to carry out the necessary Δv sequence up to and including the Jupiter assisted flyby. It must also carry of all of the necessary fuel for both propulsion and AACS. The Δv sequence is listed below in Table 3-5.

Table 3-5: Cerberus Δv sequence

Location	Δv(km/s)
Earth	.001
Midcourse	.395
Jupiter	.391

Unmanned reconnaissance spacecraft of the past, such as Voyager, have weighed between 200-800 kg [4]. This is the weight class that Cerberus was designed for. Table 3-6 show Cerberus' dry weight breakdown. For this payload mass and the required Δv's, the fuel of choice is Hydrazine (N_2H_4). It has an excellent track record with 20 years experience and a large data base [7]. It's I_{sp} of 235s provides adequate thrust times for this type of spacecraft.

Table 3-6: Cerberus' dry weight breakdown

Payload Type	Mass (kg)
Science	90
MITG'S	50
Antenna	40
Propulsion Module	90
AACS	85
CCC	35
Structure	250
TOTAL	640

Other fuels were considered , such as cold gas and the bipropellant the N_2O_4/MMH. Cold gas offered a simpler design and is less expensive but it's I_{sp} of 50 is only good for low thrust pulsing. The bipropellant has a higher I_{sp} of 285 but needs a more complex system of metering the fuel. For a mission of this lifetime the simpler the system is more reliable. Therefore, the Cerberus propulsion system will be based around the monopropellant hydrazine. A more complete breakdown of hydrazine's advantages are shown below in Table 3-7 and were found in [7].

Table 3-7. Hydrazine advantages for this mission

1	Simple and reliable (20 years experience)
2	Lowest cost propulsion system, other than cold gas
3	Space storable for long periods (> 12 years demonstrated
4	Low thrust capability
5	Moderate thrust levels

To get the amount of fuel needed a semi-dry mass was worked backward using the rocket equation. By semi-dry mass it is meant that part of the AACS fuel will still be left after the Jupiter flyby. A 20% redundancy was then built in to the calculations. The equations are shown in Appendix C.

These calculations yielded the propellant structure weight along with the amount of propellant needed. The total amount of N_2H_4 needed, including the AACS requirements and a 20% redundancy, is shown below.

392 Propulsion for Δv's + 20%
+81 Attitude and control (ACCS)

TOTAL = 473 kg

A two tank configuration was then chosen. Two rubber bladders carry the hydrazine and two smaller helium pressurant tanks are needed to pressure feed the propellant on demand to thrusters. This configuration was determined to be better than a 3 or 4 tank set up because of its simpler design. Although simpler, redundancy was still achieved.

The size of the tanks or rubber bladders was done through a simple conversion of propellant mass to volume through the density of hydrazine. The radius of the tank was found by equating this quantity to the equation for the volume of a sphere. As an approximation, the helium tanks were taken to have 1/3 the radius of the rubber bladders.

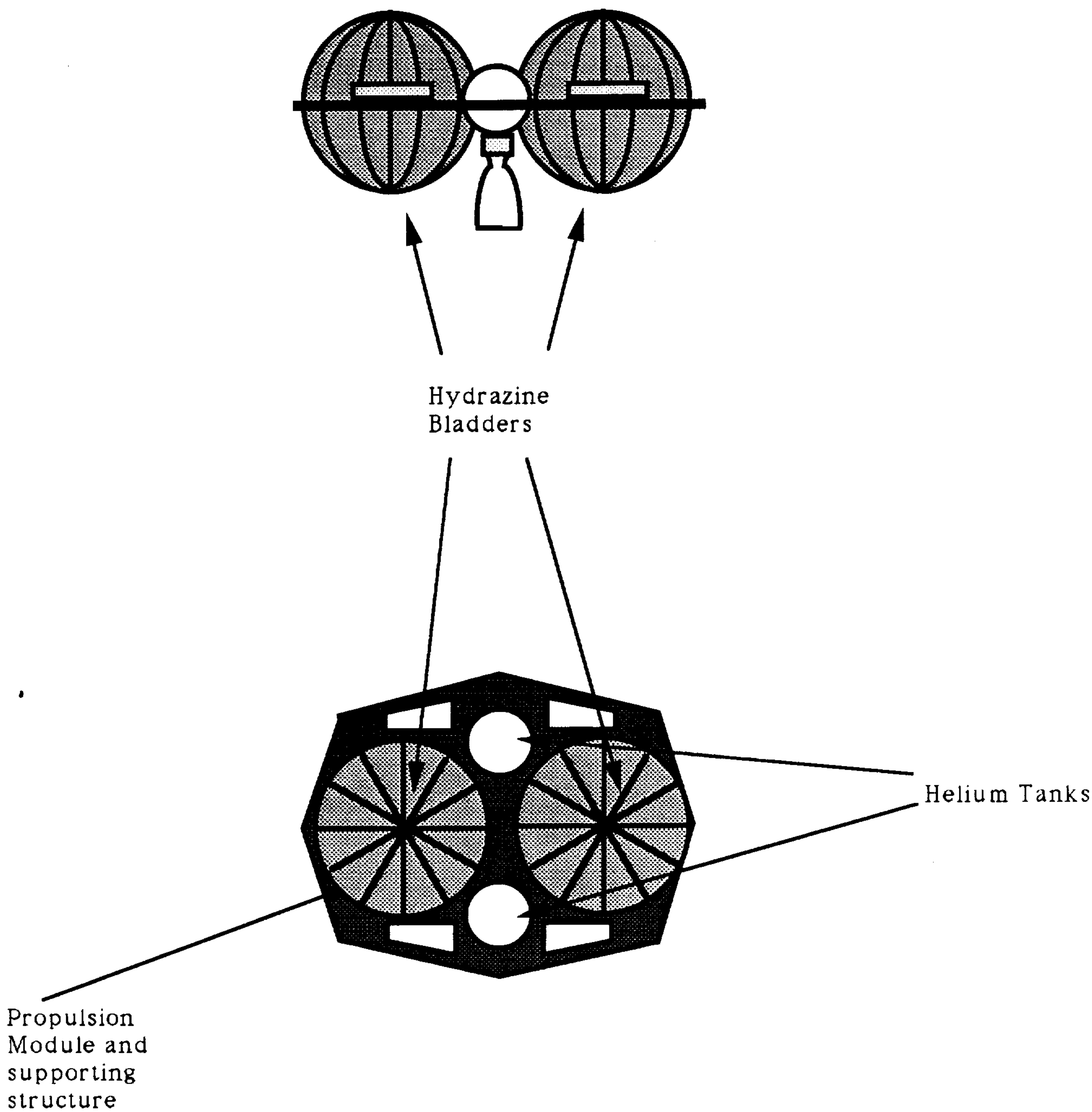
A 400 N main engine was selected to deliver the Δv's. Rough conservative calculations of the thrust times needed to perform each Δv are given in Table 3-6. The equations and an explanation of the estimates are given in Appendix C. These times are still feasible for approximate impulsive maneuvers. Furthermore, such engines have been used in past and are present missions. They have proved themselves to be reliable in situations with similar Δv's [4].

Table 3-6: Times of thrust for Δv's using 400N engine

Δv(km/s)	Time
.001	2.3sec
.395	17min 59sec
.391	14min 39sec

The complete design of Cerberus' propulsion module is given in Figure 3-5 and is shown in 1/25th scale. The final integration into the overall spacecraft structure is shown in Figures 2-1 and 2-2. The propulsion module is shown fitting into the 1.5m wide main bus of the structure. The side view shows that half the module will be up in the bus. The system configuration and valve network is shown in Figure 3-6. Since thrusters will not be redundant, 2 valves and 2 lines are assigned to each thruster providing a feed redundancy from each tank.

Table 3-7 provides the weight breakdown for the final propulsion module. It should be noted that the weights for components are rough estimates based on findings in reference [6]. While the weights are not exact there is room for expansion of the overall design if needed. Also, the weights for AACS thrusters and plumbing were not taken into consideration here but instead are accounted for in the AACS weight of 85kg.



1/25th Scale
4 cm = 1 m

Figure 3-5: Propulsion Module
Top and Side View

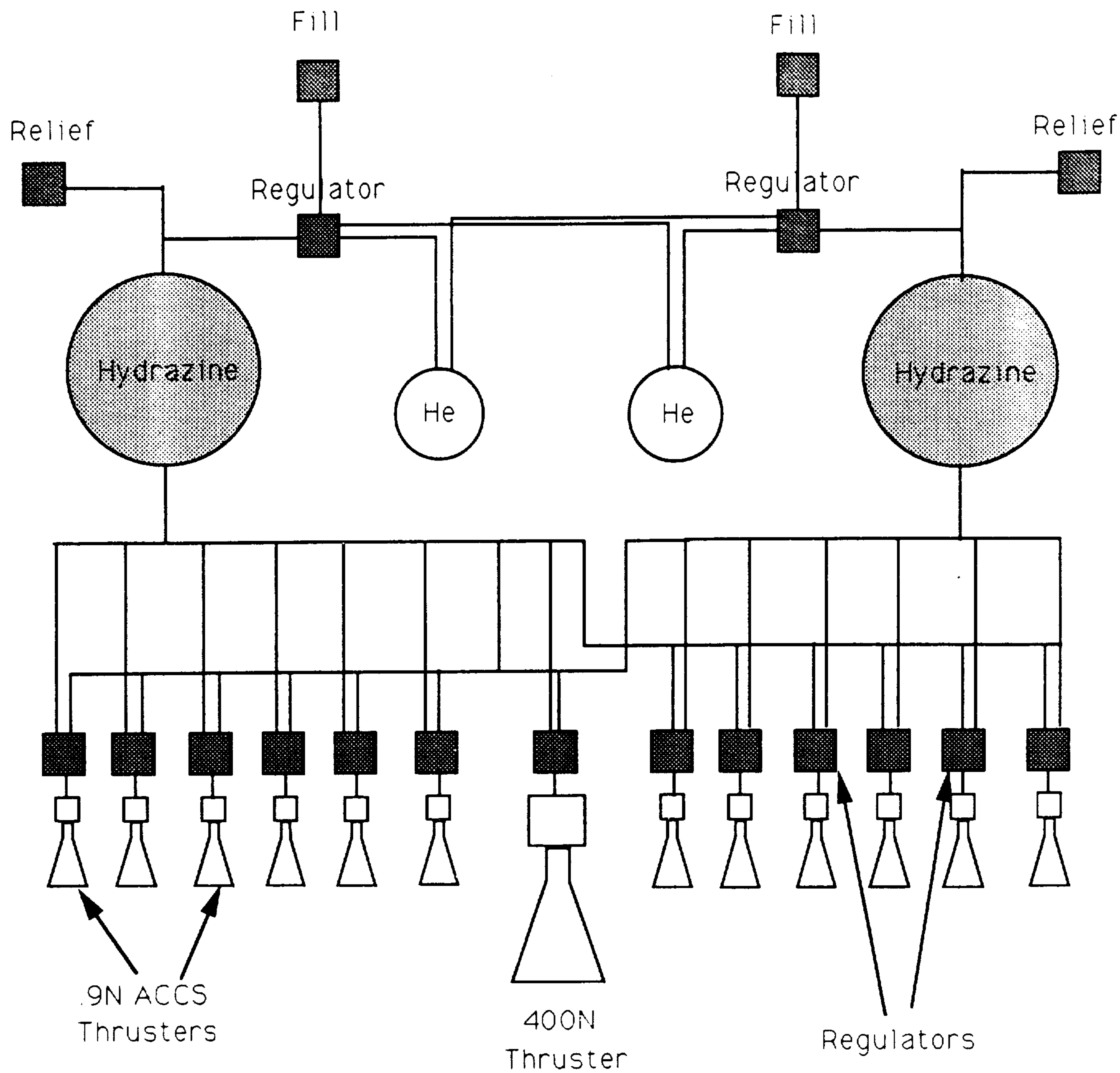


Figure 3-6. System configuration

Table 3-7: Dry weight breakdown for propulsion module

Component	Weight(kg)
Tanks (2 bladders, 2 He tanks)	27
PMDS Management devices	7
400 N engine	6
Heaters	5
Structure	44
Residuals	1
TOTAL	90

The cost of the system has been kept to a minimum by its light weight and simple design. In addition, the propellent to be used, hydrazine, has been flight-tested and is reliable. With a redundancy in the fuel (20%) and in valve configuration, this module will last the the mission lifetime while not precluding it from being utilized for other missions.

Section 3-4: ELV/UPPERSTAGE TRANSFER VEHICLE

Once the final propulsion sizing was finished, the fully loaded (wet weight) of the spacecraft was determined. This was part of the calculation for the propellant need done in the Appendix. The wet weight of Cerberus craft is 1093 kg. The Δv needed to insert Cerberus into its transfer orbit is 5.192 km/s. Since most upperstages utilize the solid propellant ammonium perchlorate, it was used for the calculations of amount of propellant needed to do this burn with this payload. The burn required 6401 kg of fuel making the total weight of the upperstage 6957 kg. This number plus the wet weight of the craft determined the total integrated takeoff payload for the ELV. This came to 8050 kg. Therefore, the upperstage must have 6401 kg of fuel and the ELV must be able to lift 8050 kg into Low Earth Orbit (LEO).

Table 3-8 shows the available upperstages while Table 3-9 shows the available ELVS that can meet these requirements[9]. It is noted also in the tables the approximate percentage of downloading that needs to be done for each vehicle.

Table 3-8: Possible upperstages

Vehicle	Contractor	Weight (kg)	% Download
IUS	Boeing	14,660	57
TOS	Orbital Sciences	10,894	36
TOS/AMS	Orbital Sciences	16,016	57

Table 3-9: Possible ELV vehicles

Vehicle	Contractor	Performance to orbit (kg)	Approximate % Download
Commercial Titan	Martin Marrietta	14,519	45
Titan 4 NUS (Type1)	Martin Marrietta	17,740	55
Titan 4 NUS (Type2)	Martin Marrietta	17,015	53

The tables show that the optimal ELV/upperstage combination would be the Commercial Titan with the TOS upperstage. The IUS upperstage was eliminated because it could not be downloaded the necessary amount [3]. Others were then considered for the least amount of downloading, i.e., the least alteration to the existing vehicle. Vehicles requiring the least amount of downloading will cost less and be easier to get flight ready. Again the requirements were met for the cost effectiveness by the use of off-the-shelf items.

Section 3-5: CONCLUSION

Through the use of off-the-shelf items and a conservative cost effective approach, feasible power and propulsion systems were conceptualized for the

Cerberus spacecraft. It is believed that these systems will not preclude Cerberus from successfully completing its mission along with possibly performing others. It is also believed, from a cost standpoint, that these systems will not cause Cerberus to overshadow other missions of the same era.

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Appendix C: EQUATIONS

EQUATIONS FOR POWER CALCULATIONS

Using the power law equation:

$$p=ce^{kt} \qquad \text{eq. 1}$$

and using conditions found in ref.[2] for $^{238}PuO_2$

t(0)= 4460.6 watts

1.5493 years later t(1.5493)=4406.7 watts

the constant (k) for $^{238}PuO_2$ was found to be -7.8468E-3

Now plugging the conditions of c=initial thermal loading of 258.5 watts from Figure 3-4 for an eleven slice MITG and t=19, the power after 19 years is 222.69 watts.

EQUATIONS FOR PROPELLANT CALCULATIONS

Using the rocket equation:

$$\Delta v=(I_{sp}*g)*\ln(m_{init}/m_{final}) \qquad \text{eq.2}$$

and the propellant mass fraction:

$$\text{mass frac}=\frac{m_{prop}}{m_{prop\ struc.+prop}} \qquad \text{eq.3}$$

masses were found.

For Hydrazine: I_{sp}=235 mass frac.=.9 unusable=2.5%

For Ammonium Perchlorate (70%):

I_{sp}=2 mass frac.=.92 unusable=2%

EQUATIONS FOR CALCULATIONS OF TIME OF BURNS

A simple linear approach was taken using the following:

$$F=ma \qquad \text{eq.4}$$

to get the acceleration (a) and the linear velocity equation:

$$\Delta v=at \qquad \text{eq.5}$$

The change in mass was accounted for at the end of each burn, then subtracted from old to get new mass for next burn.

*It is noted here, that in reality (a) is not constant throughout the burn since the mass is also changing as fuel is expelled from the spacecraft. Iterating numerically would yield smaller times since the rocket effect would occur. Therefore, this linear approach is a rough but conservative estimate.

Section 4: ATTITUDE, ARTICULATION, AND CONTROL (AACS)

Section 4-1: INTRODUCTION

The function of the Attitude and Articulation Control Subsystem (AACS) is to determine the orientation of the spacecraft and control its motion. This includes orienting the axes of the spacecraft; controlling the valves, heater, and firing of the thrusters; firing the engine for trajectory correction maneuvers (TCM); and controlling the science platform [1].

The design of the AACS was determined largely by the requirements in the Request for Proposal (RFP). The requirements that applied specifically to the design of this subsystem were satisfied. First, it was required that the spacecraft's performance, weight, and cost were optimized. Second, the spacecraft was designed to be simple, reliable, and easy to operate. Third, off-the-shelf hardware and technology available by 1999 were used as much as possible. Fourth, the spacecraft was designed to be able to perform several possible missions. Fifth, the spacecraft will have a design lifetime sufficient to carry out its eighteen year mission plus a reasonable safety margin. (A 20% safety margin would result in a design lifetime of 21.6 years.)

In addition to those in the RFP, there were design requirements dictated by the other subsystems. The Command, Control, and Communication (CCC) Subsystem required that the high-gain antenna must be pointed at the Earth with 0.1° pointing accuracy. The Science Subsystem needed to be able to point remote sensing instruments at specific locations for extended periods of time with 0.1° pointing accuracy. The Mission Planning Subsystem required that windows must be identified in which the ΔV maneuvers could be executed, and the Power and Propulsion Subsystem defined limits for the power consumption and mass of the AACS.

Finally, the AACS was also designed to follow the overall objective of the Cerberus mission. Cerberus was designed to sell and work, which means that at every turn, measures were taken to design an AACS that was cost effective and truly feasible.

Section 4-2: MAJOR FEATURES OF THE AACS

After careful consideration between three-axis, spin, and dual-spin control, three axis stabilization was selected as the control method for Cerberus. Spin stabilization met many of the requirements. It is the simplest and least expensive of the three [2]. It is very reliable and has a long lifetime, which is very important considering the length of this mission (eighteen years). In addition, it is the lightest and requires the least power [3]. Nonetheless, spin stabilization was unacceptable for one reason. The Cerberus mission requires that the remote sensing science instruments be inertially fixed for extended periods of time. This would require the spacecraft to undergo a complicated despinning process, which makes spin control infeasible. On the other hand, both three-axis and dual-spin provide the neccesary fixed inertial orientation. Dual-spin has certain advantages over three-axis. The former provides scanning science capabilities and has low sensitivity to disturbances, whereas the latter has neither [3]. The deciding factor between the two, however, was the fact that dual-spin is much more expensive and complex than three-axis. Although dual-spin offered certain conveniences, it did not meet the requirements of optimizing cost and maintaining simplicity. Therefore three-axis stabilization was selected. Table 4.1 compares the advantages and disadvantages of the three control methods discussed above.

Table 4-1: Comparison of Control Methods [2, 3]

Types of Control	Advantages	Disadvantages
Three-Axis	•High accuracy •Good maneuverability •Adaptable to changing mission requirements •Allows inertial remote sensing science	•High weight and power •Costly hardware •Extensive fault detection/correction for back-up
Spin	•Simple, low cost •High reliability, long life •Low weight and power •Inherent science scan motion •Low sensitivity to disturbances	•Poor maneuverability •Must despin to do some imaging science, which is complicated process

Dual-spin	•Provides both scanning and inertial science •Low sensitivity to disturbances •Fixed inertial orientation	•Expensive and complex •Articulated elements require balance compensation

After the decision was made to use three-axis control, the next step was to decide how to best implement such control. There were two options. The first was to use momentum wheels or control moment gyros for stability and small turns. A reaction control system using gas thrusters would also be needed to dump momentum from the momentum wheels or control moment gyros and to do large turns and maneuvers. The second option was to only use gas thrusters. A trade study was conducted to see if the addition of momentum wheels or control moment gyros would save enough fuel to offset the additional weight. First, it was concluded that momentum wheels are lighter than control moment gyros [2], so momentum wheels were used as the basis of comparison. From three different sources [2, 4, 5], the mass of a system of four momentum wheels was estimated at 100 kg. Next, the mass of the fuel that would be saved by using momentum wheels was calculated to be 80.6 kg. See equation (1) in Appendix D for details of this calculation. Therefore, even without considering the additional fuel needed for momentum dumping, using a system with only thrusters would be lighter than one that used both thrusters and momentum wheels. See Table 4.2 for a summary of this trade study.

Table 4-2: Trade Between Momentum Wheels and Thrusters

	System with thrusters only	System with both momentum wheels and thrusters
mass (kg)	80.6	100.0*

*Does not include fuel needed for momentum dumping

In addition to being heavier, momentum wheels would also add unnecessary complexity and would decrease the lifetime due to wear. Therefore, three-axis control will be implemented using a configuration of only thrusters because such a selection satisfies the requirements of simplicity, sufficient lifetime, and low weight.

The last major feature in the design configuration of the AACS is the articulation control of the science scan platforms. The Science Subsystem requires both a high accuracy scan platform with two degrees of freedom and a low accuracy platform with one degree of freedom. Two options were explored. The first was the traditional method, in which scan platforms are articulated using two step motor actuators without any momentum compensation [6]. In addition, the AACS electronics, star tracker, and gyros are all placed on the spacecraft itself. This was the technology used for Voyager. The second option involves a new technology being developed for the Mariner Mark II (MMII) project, and it is called the Integrated Platform Pointing and Attitude Control Subsystem (IPPACS). If the second option were used, the star tracker, gyros, and AACS electronics would all be placed on the scan platform [7]. This second option using IPPACS was selected for the following reasons. First, the scan platform momentum compensation decouples the scan platform dynamics from those of the spacecraft [7]. Second, this decoupling of the dynamics ensures dynamic stability of the spacecraft [8]. This satisfies the requirements of reliability and ease of operation. Third, the high accuracy sensors and controls are rigidly attached to the high accuracy science instruments, which greatly reduces many errors found in systems like that on Voyager [7]. See Table 4.3 for a comparison of errors between Voyager and MMII IPPACS. This meets the requirement to optimize the spacecraft's performance.

Table 4-3: Comparison of Errors Between Voyager and MMII IPPACS in Terms of Scan Platform Pointing Control (Adapted from [7])*

Error Source	Voyager	MMII IPPACS
Limit Cycle	1.52	0.00
Sun Sensor	0.69	0.00
Star Sensor	0.59	0.02
Scan Platform Control	1.20	0.34
Structural Misalignment	0.58	0.58
Dynamic Stability	0.33	0.33
Gyro Drift (2 hours)	0.00	0.11
Total 3σ Scan Platform Pointing Accuracy	2.26	0.76

*All numbers in mrad

Because IPPACS has such high accuracy, it will easily be able to perform several possible missions. Although IPPACS has yet to be flight tested, it will have flown aboard a Mariner Mark II mission before 1999, and can be considered off-the-shelf hardware. The biggest disadvantage of IPPACS is that it shifts the weight of many of the attitude control components away from the center of mass to the scan platform, slightly increasing the moments of inertia of the spacecraft [7]. Nonetheless, the advantages far outweigh this disadvantage, and IPPACS was selected to control the articulation of the high accuracy platform. Figure 4.1 shows a schematic of the IPPACS configuration, and Figure 4.2 shows the actuator orientation for two articulation degrees of freedom.

The question still remained of how to control the articulation of the low accuracy platform. One option was to use IPPACS on it also, and therefore have a fully redundant AACS. This, however, would certainly not optimize weight and cost. Instead, it was decided to use a single uncompensated step motor actuator and measure the angular displacement with a simple optical sensor. Momentum compensation wasn't necessary because the low accuracy platform will not be used for determining the attitude of the spacecraft or conducting imaging science. Instead, it will be used merely to scan free space. The absence of momentum compensation will, however, lead to some disturbance of the spacecraft which must be corrected by firing the thrusters. This will still be lighter and less expensive than including a momentum compensation wheel and thus meets the RFP requirements of optimizing cost and weight.

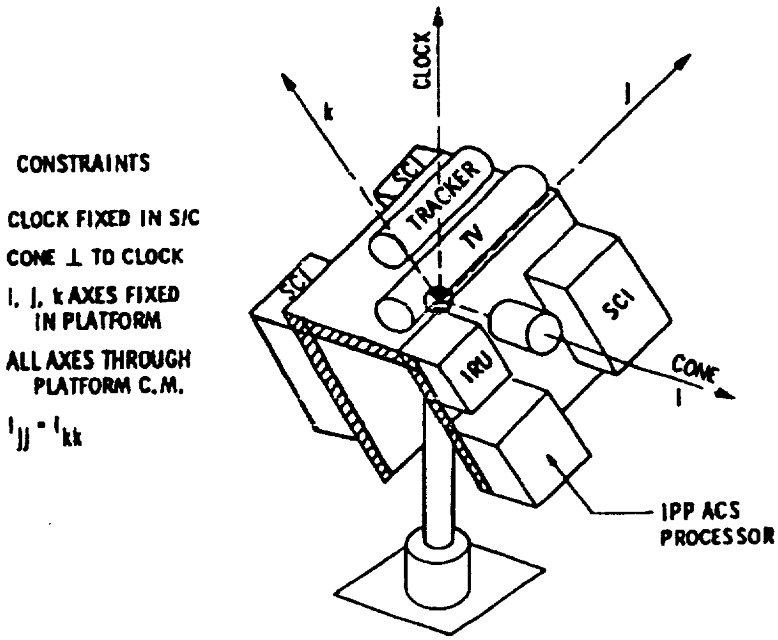

Figure 4-1: IPPACS Configuration (Adapted from [8])

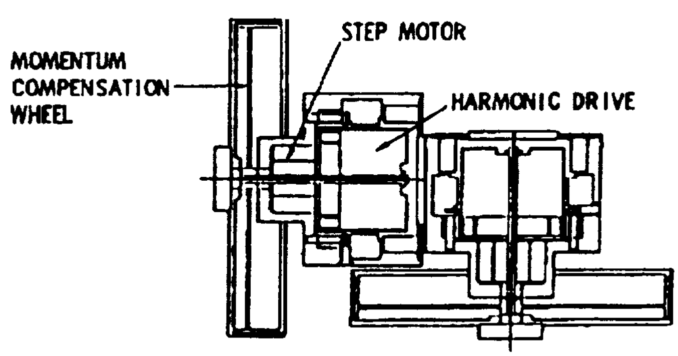

Figure 4.2: Actuator Orientation for Two Articulation Degrees of Freedom (Adapted from [8])

Section 4-3: HARDWARE SELECTION AND PLACEMENT

After the decision was made to use thrusters for attitude control, the next phase was to select thruster size and placement. With the use of IPPACS technology, turning rates of less than 1.3 $\frac{deg}{sec}$ are desired. From preliminary data on the moments of inertia and thruster lever arm distances, 0.9 N thrusters were found to produce a turning rate of 1.261 $\frac{deg}{sec}$, which meets the specification. See equation (2) of the Appendix. The 0.9 N thrusters were selected because in addition to meeting turning rate specifications, they are off-the-shelf hardware, being previously used on both Voyager [9] and MMII [7]. Hydrazine was selected as the fuel so that these thrusters and the 400 N engine could be supplied by the same fuel source. This satisfies the requirement of simplicity.

The thrusters were placed in such a way as to provide a torque couple around each axis in both directions of spin. This is required for three-axis control. Such a configuration requires four thrusters per axis, or twelve thrusters total. Figure 4.3 illustrates the placement of the thrusters with respect to the principal axes of the spacecraft. Only the minimum number of thrusters necessary was used for this configuration, which appears to sacrifice reliability and redundancy for the sake of optimizing weight and cost. This, however, is not actually the case. The thrusters themselves have proved very reliable on past missions; most problems occur in the plumbing, valves, and heaters which are redundant on Cerberus. The details of this can be found in the Power and Propulsion Subsystem in Section 3. Also, the thrusters were carefully placed to ensure that

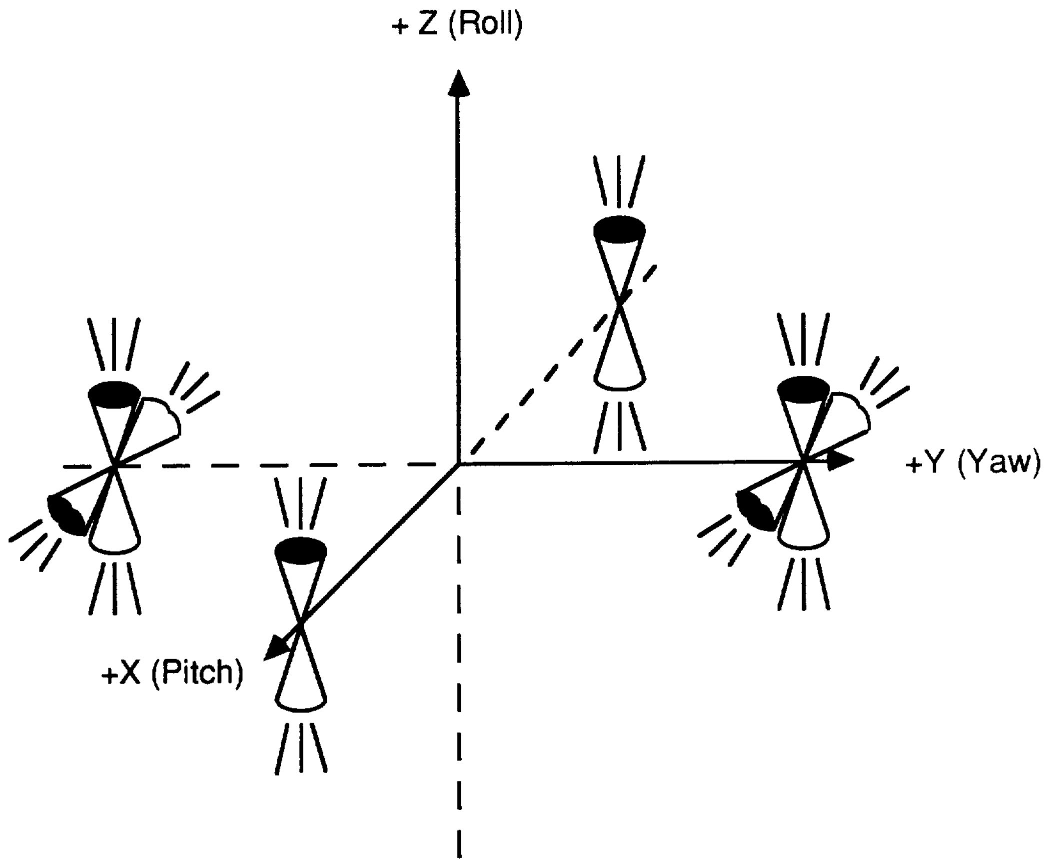

Figure 4-3: Placement of the Thrusters With Respect to the Principal Axes of the Spacecraft

their exhaust would not damage any of the science instruments. Refer to the Structures Subsystem in Section 2 for a picture showing thruster locations relative to the scan platforms.

The Planetary ASTROS (Advanced Star/Target Reference Optical Sensor) was selected as the star tracker for Cerberus. Its accuracy of 4 arcsec (0.001°) exceeds the required pointing accuracy of 0.1° [7]. The Planetary ASTROS has the additional benefit of being able to track other targets, so it will also be used to locate Pluto and Charon. In addition, it has the lowest weight and power requirements of any star tracker currently available (the Voyager Canopus Tracker is no longer manufactured) [7]. There is, however, one problem area. Although the Planetary ASTROS is internally redundant and has the longest lifetime of any star tracker currently available, its design lifetime of ten years falls dreadfully short of the required 21.6 years [7]. Nonetheless, the best plan of attack is to select the Planetary ASTROS and continue to work on increasing its lifetime through design modification. A comparison of the characteristics of the Planetary ASTROS, Canopus Tracker, and the Digital Standard Star Tracker (DSST) are given in Table 4.4.

Table 4-4: Comparison of Star Tracker Characteristics (Adapted from [7])

Parameter	Canopus Tracker	DSST	Planetary ASTROS
Field-of-View (degrees)	9x36	8x8	11x11
Spin (drift) Rate (deg/sec)	N/A	<0.3	<0.5
Calibrated Accuracy (arcsec)	180	10	4
Multistar Measurement	NO	NO	3 stars simultaneously
Internal Redundancy	NO	NO	YES
Mass (kg)	4.3	8.2	8
Power (W)	4.5	20.0	11

The Planetary ASTROS will be placed on the high accuracy scan platform as necessitated by IPPACS. It will be boresighted with the science instruments to enable accurate pointing of these instruments. Furthermore, placement of the star tracker on the high accuracy scan platform allows the Planetary ASTROS to track stars and other objects without requiring the rotation of the entire spacecraft. This will save AACS fuel and optimize weight and cost. Because the Planetary ASTROS is already being developed for the MMII, Cerberus will reap the benefit of a recurring cost as opposed to spending money to develop a new technology. In addition to meeting the requirement of optimizing performance, weight, and cost, the Planetary ASTROS also qualifies as off-the-shelf hardware.

In selecting the components for the Inertial Reference Unit (IRU), there were found to be two general choices: flight-tested mechanical gyros or a new gyro based on fiber optics called the Fiber Optic Rotation Sensor (FORS). One source states, "FORS is attractive for space missions because it promises performance comparable to or better than that of mechanical gyros as well as significant improvements in lifetime, weight, power consumption and cost" [10]. As with the Planetary ASTROS star tracker, the FORS lacks flight experience and a sufficient lifetime. Nonetheless, it is still the best available, having a longer lifetime than any mechanical gyro because there are no moving parts in the FORS [7]. Clearly, an optical gyro such as FORS best meets the requirements. The question remained, however, of which optical gyro was best. Of the three optical gyros studied, FORS best optimized performance (lowest drift and rate noise, and best angular resolution), lifetime, power, and weight. Table 4.5 gives the numerical data for these parameters.

Table 4-5: Optical Gyro Characteristics (Adapted from [7])

Parameter	FORS	DRIRU II	CG-1300 Laser Gyro
Residual Drift Rate (deg/hr)	0.2E-3	3.0E-3	7.0E-3
Rate Noise (deg/sec)	1.0E-5	1.0E-5	40E-5
Angular Resolution (arcsec)	0.005	0.05	1.4
MTBF (yr)	10	3	3
Power (W)	< 10*	22	18
Mass (kg)	10	11	18
Volume (in^3)	1000	990	350

*For three units

In addition to meeting the above stated requirements, the FORS will be off-the-shelf before 1999 because it is tentatively scheduled for use on the MMII project. For these reasons, Cerberus will use FORS for its IRU. Two identical sets of three (one for each axis) will be placed on the high accuracy platform, making the IRU redundant [7]. This is to fulfill the requirement of reliability.

A sun sensor will be mounted behind the high-gain antenna and will be pointed at the sun through a hole in the reflector. As with Voyager, it will be boresighted at an offset of 5-6° off the Z (roll) axis of the spacecraft, so that when the sun sensor points at the sun, the antenna points at the Earth, as required by CCC [6].

At this point in the preliminary design, there was no way to calculate the precise ΔV required for a minimum maneuver scheme. Instead, a calculation was performed based on an average of AACS maneuvers per month in past missions. This calculation is given in equation (1) of the Appendix. The result is that 80.6 kg of hydrazine will be required for AACS maneuvers. This does not include TCM's.

To conclude the discussion on hardware selection and placement, Table 4.6 gives a summary of the power and mass requirements for the AACS.

Table 4-6: Power and Mass Requirements for the AACS

Component	Quantity	Power (W)	Mass (kg)
Planetary ASTROS Star Tracker	1	11	8
FORS IRU (3-axis, redundant)	1	10	10
Digital Sun Sensor	1	5	4
Microstep Scan Actuator	2	19	21
Low Accuracy Platform Actuator	1	7	6
Low Accuracy Platform Position Sensor	1	1	0.5
Heating valves and thrusters	12	7	15
Total*		60	64.5

*Does not include computer, propulsion system, or fuel

Section 4-4: SCANNING AND POINTING REQUIREMENTS IMPLEMENTATION

By pointing the sun sensor at the sun and the star tracker at a star orthogonal to the roll axis, the high-gain antenna can be pointed at the Earth with 0.1° accuracy as required by CCC. From this orientation, the low accuracy scan platform can perform particle field science around its one degree of freedom; the Science Subsystem did not deem it necessary to perform such experimentation around all three axes. From this same celestial lock orientation, much of the science on the high accuracy scan platform can be accomplished. The star tracker will be used in this situation to track the targets of the science instruments. If an object cannot be sighted from celestial lock, the entire spacecraft will be rotated using the 0.9 N thrusters, with the attitude controlled by the IRU. With this configuration, the remote sensing and imaging science can remain fixed on a target in any direction for extended periods of time (approximately three hours) as required by the Science Subsystem.

Section 4-5: ATTITUDE CONTROL MODES

After Cerberus separates from the launch vehicle, the AACS will enter the deployment mode. During this time the AACS will control the deployment of the magnetometer, RTG's, science scan platforms, and high-gain antenna

[9]. All the hardware and instruments will be checked out and calibrated in order to optimize performance.

When the checkout and calibration is complete, Cerberus will enter the cruise mode. During this mode, the AACS will execute a preset series of commands to measure fields and particles of interplanetary space [9] after every A.U., as required by the Science Subsystem. Other procedures executed during this mode are attitude determination, high-gain antenna pointing [1], and maintenance of all three axes within a deadband limit of 0.05° [6]. This mode would also include a special routine in both the computer RAM and ROM to point the antenna back to the Earth from any other orientation. This enhances the spacecraft's reliability and ease of operation.

Finally, Cerberus will enter its flyby mode when it encounters Earth, Jupiter, and finally Pluto. At Earth and Jupiter this will involve the AACS recognizing a box-shaped window defined by certain stars in which it will control the firing of the 400 N engine. During the approximately fifty day encounter near Jupiter and Pluto, the AACS will control the pointing and slewing of the high accuracy scan platform.

Section 4-6: CONCLUSION

The AACS of Cerberus will use a three-axis stabilized design controlled with 0.9 N hydrazine thrusters. The high accuracy scan platform will be controlled using the new IPPACS concept which offers the tightest pointing and the most reliable control. All components were selected to optimize performance, weight, and cost and will be readily available by 1999. The AACS has also been shown to meet the requirements imposed upon it by the other subsystems. The critical problem areas left to be solved are extending the lifetime of the components to at least 21 years (20% safety margin) and designing the details of the attitude control modes and the exact sequence of AACS commands.

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Appendix D

m_1 = mass for small AACS maneuvers

$$= 0.373 \frac{kg}{month} \qquad \{\text{from } [6]\}$$

$$mass_{tot} = \left(0.373 \frac{kg}{month}\right)\left(\frac{12 \text{ months}}{1 \text{ year}}\right)(18 \text{ years})$$

$$= 80.6 \text{ kg} \qquad \textbf{eqn. (1)}$$

$$\theta_c = \frac{2FL}{I_v}\left(\frac{t}{2}\right)^2 \qquad \{\text{from } [3]\} \qquad \textbf{eqn. (2)}$$

Section 5: COMMAND, CONTROL AND COMMUNICATION

Section 5-1: INTRODUCTION

For the Cerberus mission, communication between the spacecraft and Earth becomes the primary driver for the Command, Control and Communication (C^3) subsystem. Due to a time lag as high as 11 hours (round-trip), the craft has been programmed for a high degree of autonomy. Using as much off-the-shelf hardware as possible, it will use the latest in computer technology, and utilize a spare 4.8m Galileo antenna. The receiving antenna system will be the existing Deep Space Network (DSN). Decisions were reached based on previous missions and the requirements specific to the proposal.

Section 5-2: COMMAND AND CONTROL

Consisting primarily of the computer, Command is divided into three main subsystems - Computer Command Subsystem (CCS), Attitude And Articulation Control Subsystem (AACS), and Flight Data Subsystem (FDS). Each subsystem will utilize expert systems when available, as well as sophisticated fault protection algorithms, (FPA). Due to the distance involved and the length of the mission (18.7 years) the spacecraft is designed to be almost fully autonomous.

Command will be performed by two computers in parallel redundancy. Each computer will consist of five central processing units (cpu) with math coprocessors, all accessing a common memory chunk of 64 megabytes (MB). One cpu - designated as the monitor - will break each assignment into subparts and allocate them to the other four cpu's. The monitor will also serve as a fault detector should one of the cpu's fail. In this case, an FPA will be used to exclude that cpu. In the event that an entire computer fails, the cpu's in the remaining computer can be reprogrammed so that they are working in redundant pairs. In a worst case scenario, the whole spacecraft could be run from one cpu.[17]

The maximum bit-rate for Cerberus is 145.5 kbps. Science instrumentation may require up to 1000 kbps. Real-time transmission, then, is not always feasible. Data must be stored until transmission is possible. External memory is needed.

Three options exist for external memory - magnetic tape, removable hard drives or optical disks.[6] Although not proven as space technology, optical disks provide a compact, cost efficient, error-free method for storing data. It will have read-write capabilities, with a maximum storage capacity of 600 MB. See Table 5-1.

Table 5-1: Comparison of External Memory Options

External Memory	Disadvantages	Advantages
Magnetic Tape	Bulky, error-prone, heavy.	Proven technology, inexpensive.
Removable Hard drive	Not proven tech., high power cost, heavy.	Small, low error rate.
Optical Disk	Not proven tech.	Small, lightweight, low error rate.

Three options also exist for the computer programming language. Assembly is the lowest level language of the three, and as such is difficult to program in. C is an industry standard. Many people are familiar with it and programming is much simpler than with Assembly. For this mission, ADA was selected. The standard for the Defense Department, it is ideal for situations where prioritizing is needed. Semaphores, which act as flags, signal important incoming information, allowing new information or more important processes to take precedence over existing tasks. This is ideal for sequencing.[17]

Each computer will be able to access the optical drive. The computers and the optical drive will be fully independent and redundant, in case of failure. In this way, a computer could be reprogrammed from either the optical drive or the other functional computer. The computers and optical disk will be redundantly programmed for the three subsystems CCS, AACS, and FDS.[1]

Section 5-2.1: COMPUTER COMMAND SUBSYSTEM

As overseer of the craft, CCS is primarily concerned with issuing commands to the other subsystems. Other responsibilities include sequencing, FPA's and real-time command processing.

Sequencing will function throughout the duration of the mission, but becomes more important when science instrumentation is active. It assigns each function a value based on priority. This priority is determined before the mission's onset. Considerations include, but are not limited to, whether science cannot be achieved from Earth, capability of the instrument(s) required, the number of observations needed, duration and time of the observations, power needed, and tolerance of location and duration. A program such as SEQTRAN will be implemented.[22] This program converts input into mnemonic commands and checks for constraint violations. Two constraints are spacecraft physical limitations and mission rules established for safe operation of the spacecraft. Maximum efficiency of sequencing space is achieved by overlap. This allows one sequence to start before a prior sequence is finished.

Expert systems will be used wherever possible. These allow on-line changes to be made in the structure of craft operations without input from ground control. Each system must consist of knowledge representation, knowledge utilization, and a computational model.[29] These will be further broken down into a data handler and trend analyzer.[7] The data handler receives data from telemetry and stores it appropriately, according to significance at that particular time. The trend analyzer calculates, plots and posts trend information. Use of expert systems will provide better fault protection, allowing quick response to potential problems.

Improved fault detection will enhance the craft's ability to successfully achieve mission goals. FPAs will consist of five major modules - the main controller, status monitor, fault diagnosis module, knowledge base, and interface handler. These five modules will work together as an expert system to detect errors early on and take preventative actions when possible. The craft's autonomy will be increased by reducing the amount of ground control intervention.

CCS is responsible for enabling all telemetry commands. It will also process real-time commands and send them to AACS or FDS as needed, monitoring them as well as other craft operations.

Section 5-2.2: ATTITUDE AND ARTICULATION CONTROL SUBSYSTEM

Utilizing an expert system such as APPS, AACS will hold the craft to its chosen trajectory. In order to maintain communication with Earth, the high gain antenna (HGA) must always point to Earth. This will be done by attitude maintenance, antenna pointing, and gyro control. Antenna pointing control will be divided into a main reflector and a subreflector drive.[13] More information is contained in the Attitude and Articulation Subsection.

Section 5-2.3: FLIGHT DATA SUBSYSTEM

Responding to commands from the CCS, the Flight Data Subsystem contains routines that control science instrumentation and the optical drive. Some data processing is also handled through FDS.

FDS first collects engineering and science instrument data. This is used to control the operation of the instruments. It is then formatted for either storage or real-time transmission. Analog data must be converted to digital form before it can be sent. This is also handled by the FDS. FDS must provide data modes, rates and formats. FDS also provides frequency references for the other subsystems.[19]

A system will be utilized using movable blocks of observations. These groups are controlled relative to a single adjustable starting time, which allows the computer to compensate for inability to determine the time of closest approach in time for effective trajectory control maneuvers.[8]

Section 5-3: COMMUNICATIONS

Communications will consist of two subdivisions - Telemetry and the Radio Frequency Subsystem (RFS). Ground commands will be processed through RFS and passed on to CCS. Information is also relayed back to Earth through the RFS by telemetry.

Section 5-3.1: TELEMETRY

Telemetry is made up of information from three different sources. Science data is generated by instrument observations. While only small amounts of science data are created, it requires the highest quality transmission accuracy. Engineering data, for daily craft operation, requires a moderate quality transmission of moderate volume. Imaging data, due to its high redundancy, has a very high volume with the lowest quality standard.

In order to minimize data rates, all data that can be compressed shall be. Both convolution and Reed-Solomon (RS) coding will be used, reducing bit errors down to 10^{-6}.[6] Although RS coding requires more processing at the ground end, it is more effective and efficient than the Golay coding originally used for the Voyager missions.[14] An RS system consists of one chip for the encoder and seven chips for the decoder. It operates at a rate up to 80 Mbps.[3] All, or nearly all pertinent data is retained.

Loss of information can also be reduced through multiple playbacks. The memory capacity of the computers is high enough that the computer can wait for data confirmation from Earth. If a high loss has occurred, data is merely retransmitted.

Information will be sent in telemetry packets. Packets will be constructed by individual subsystems. The CCS will add RS code bits to the packets, providing error-free transmission.[6] Due to the large quantity of memory, data loss can be further minimized.

Section 5-3.2: RADIO FREQUENCY SUBSYSTEM

Similar to the Cassini mission, Cerberus takes advantage of the more powerful Ka-band (32 GHz). Although the antenna surface tolerances are lower for higher frequencies, technology exists that will compensate for this. This will include the addition of Ka feeds, waveguides and amplifiers. Ka-band transmission simplifies hardware by decreasing size and power requirements. Bit error rates and doppler tracking accuracy are also improved.[6]. A net gain of 8 db can be realized by upgrading from X-band to Ka, as antenna gain increases in proportion to the square of the link frequency.[11].

For further efficiency, the command detection unit (CDU) will be under RFS jurisdiction. Its uplink command will be performed by the Ka-band transponder. Technology exists allowing the telemetry modulation unit (TMU) functions to be accomplished with a few chips. This too will come under the RFS umbrella.[6]

Optical communications were also considered. Unfortunately, under the proposal time limit, optical communication is not possible.[31] If developed, it would have many advantages over the traditional RF system. Its wide bandwidth would allow gigabits of information to be transmitted via a small laser antenna with low transmitting power. However, the pointing accuracy requirement alone, 10^{-4} degrees, disqualifies it from use - the best accuracy achieved today is 0.1 degrees. Output powers must be increased, as well as improved beam quality. Materials must be found that are radiation tolerant. Space debris is an additional factor, as it can damage the optics and will cause an inferior signal quality.[4] For all these reasons, a traditional RF system was judged to be best (Table 5-2).

Section 5-3.3: SPECIFICATIONS

For an RFS to meet the challenges of this mission, new techniques must be employed. The Galileo antenna had the advantage of being made of a light-weight mesh material. Its folding capabilities allow many different launch possibilities. However, in the ten years since the antenna was developed, new technologies have provided different ways to improve upon its design.

Before we can use the Galileo antenna, it must be modified. New technology allows us to transmit at the higher Ka-band frequency. Offset subreflectors must also be added, to allow X- and S-band transmission. These subreflectors will have frequency selective surfaces (FSS), allowing transmission of several different frequencies from one antenna with the addition of multiple feeds.[15,33] The antenna will be attached to a three-way gimballed joint to allow it to be pointed in any direction. Sitting "on top" of the craft, it will always be pointed at Earth, except in cases where the spacecraft must be turned to perform science experimentation. The low gain antenna (LGA) will also transmit at Ka-band. It will be used to communicate with Earth until Jupiter is reached. The high gain antenna will be deployed at that time.

Table 5-2: Communication Options

Communication Mode	Disadvantage	Advantage
High gain antenna and Low gain antenna	Higher weight and cost. Relatively high power requirement.	Higher redundancy, allows delayed deployment of HGA. Proven technology.
Optical Communications	Not yet developed.	Extremely lightweight and power efficient.

Although bandwidths are traditionally 5-10% of the transmitting frequency, the antenna is designed with a 21.85KHz bandwidth. This is approximately $7E10^{-5}\%$ of the 32 GHz transmitting frequency. Additional feeds must be used, as well as amplifiers and waveguides.

The antenna system must also be designed around a number of other factors. DC to RF conversion (L_t or transmitter system losses) is currently 21%.[11] Attenuation, which increases with inclement weather, must also be considered (Figure 5-1). Using Goldstone as the receiving station, one has an atmospheric attenuation (L_a) of 92%.[27] Antenna efficiency, μ,can be pushed as high as 80%.[33] Other losses include receiver system losses (L_r), pointing losses of both the transmitter and the receiver (L_{tp} and L_{rp}, respectively), free space loss (L_s) and polarization loss between antennas (L_p). A transmission power (P_t) of 10 W results in a receiving power of $9.05*10^{-17}$ W.[30]

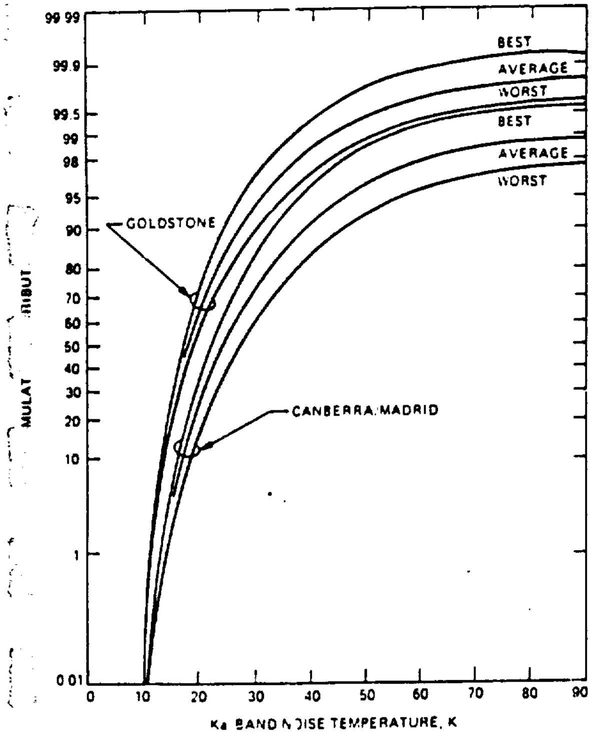

Fig.6-1: Ka-band atmospheric noise temperature statistics: all sites, 30-deg elevation angle.[27]

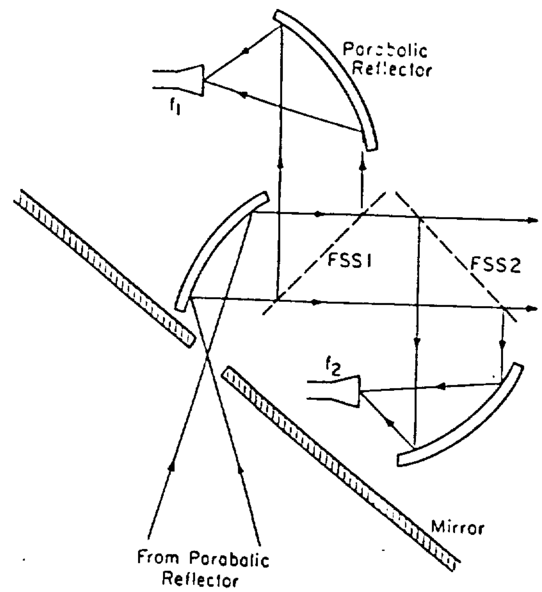

Fig.6-2: Multiple feeds for different frequency bands can be arranged with beam waveguides, and frequency selective surfaces.[15]

DSN will be the receiving antenna. This requires a signal-to-noise ratio (SNR) of at least 10, with higher values providing better transmission.[30] All my calculations are based on the current 70m antenna with an SNR of 20. Vast improvements in receiving power will be realized when the proposed array of 35m antennas is deployed. Other improvements could be made by orbiting a receiving dish. Gravity effects would not be felt and atmospheric attenuation would be eliminated. The dish could either be an orbiting satellite or a multi-deployable dish on the space station.

Section 5-4: CONCLUSION

As few as ten years ago, the idea of a mission to Pluto would have been ludicrous. It is only with recent advances that this mission has become feasible. Without the ability to transmit at a higher frequency or use an upgraded DSN, the antenna size alone would have precluded a successful venture. Recent advances in materials such as Kevlar or optical disks further decrease the size and weight of the craft. All of this adds up to a smaller, more cost-efficient operation, able to explore not just Pluto, but other planets as well.

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Appendix E-1

Equations:

eqtn.5.1 $\frac{P_r}{P_T}=\mu\frac{16A_RA_T}{\lambda^2L^2}$

eqtn.5.2 $G_t=\frac{4\pi A_T}{\lambda^2}$

eqtn.5.3 $SNR=P_r/P_n$

eqtn.5.4 $L_s=(\lambda/4\pi L)^2$

eqtn.5.5 $P_r=P_tL_tG_TL_{TP}L_SL_AL_PL_{RP}G_RL_R$

eqtn.5.6 $P_n=KTW$

eqtn.5.7 $B=W\log_2(P_r/P_n+1)$

P_r - Power received

P_t - Power transmitted

A_R - Area of receiving antenna

A_T - Area of transmitting antenna

G_T - Gain for transmitting antenna

P_n - power of noise

SNR - signal to noise ratio

μ - efficiency for antenna

L - distance between spacecraft and ground antenna

L_A - loss due to atmospheric attenuation

L_{TP} - transmitter pointing loss

L_{RP} - receiver pointing loss

L_R - receiving system losses

L_T - transmitting system losses

B - Bit rate of data

Values:

SNR=20

L_{Pluto}=40AU*149.6E9m

=5.984E12m

$L_{Jupiter}$=5.203AU*149.6E9m
=7.79E11m
λ=9.375E-3m
L_R*G_R*L_{RP}=.89
L_P=1.0
L_A=.92
L_{TP}=.89
L_T=.21
P_t=10 W
T=3^{o}K

transmission
L_S - space loss
L_P - polarization loss between antennas
G_R - receiving antenna gain
K - Boltzmann's constant
T - temperature (in ^{o}K)
W - Bandwidth (Hz)

Equation 5.1 gives a rough estimate of the antenna receiving power. However, combining equations 5.2,.4 and .5 gives a more exact number:

HGA Pluto
P_r=9.05E-17 W

LGA at Jupiter
P_r=6.976E-15

Using this and an SNR of 20, one can calculate bandwidth W from equations 5.3 and 5.6. This, in turn, can be used to determine P_n:

W_{HGA}= 9.05E-17W/{(1.38E10^{-23})3^{0}K(100)}
=21.85 kHz
Pn=1.38E-23*3^{o}K*21.85 kHz
= 9.05E-19 W
W_{LGA}=1685 kHz
P_{nLGA}= 6.976E-17 W

Using equation 6.7, one gets the information capacity:

B=21.85 kHz $\log_2$(100 +1)
=145.5 kbps

B=1685 kHz $\log_2$(100+1)
=11219.1 kbps

Section 6: SCIENCE SUBSYSTEM

Section 6-1: RFP REQUIREMENTS

The request for proposal (RFP) serves as a basis for the entire Cerberus mission, yet the RFP includes only a few requirements that pertain to the science subsystem. The most fundamental of these requirements is one which states that the spacecraft must perform an unmanned study of Plutonian space. This requirement defines the purpose of the entire mission and implies that the science instruments aboard Cerberus must be suitable for studying Pluto, its satellite, Charon, and the space surrounding the system. A related RFP requirement demands that the spacecraft should be able to perform several possible missions. This means that Cerberus' scientific instruments should not be limited solely to a study of Pluto, but should also be useful for experiments conducted elsewhere along the spacecraft's trajectory. For this reason, Cerberus will gather data in interplanetary space, and if astronomers desire more information about Jupiter, after completion of the Galileo mission, then Cerberus will take measurements during its Jupiter flyby.

The RFP states that reliability, simplicity, and low cost must be emphasized in the spacecraft design and mission planning. There are several requirements which reflect this central objective. The first calls for optimization of spacecraft performance, weight, and cost. For the science subsystem, this requirement applies to the components selected for the mission. In an effort to reduce costs and ensure reliability, the request for proposal limits all components to off the shelf hardware available through 1999. To fulfill this requirement, Cerberus will primarily feature the scientific instruments, or derivatives of these instruments, used on the Voyager and Galileo missions. To further ensure mission reliability, each instrument must have a sufficient design lifetime so that the instruments will be functional throughout the mission and for a reasonable amount of time afterwards. Since this lifetime will be on the order of twenty years, while the Voyager mission is only thirteen years old and Galileo is just getting started, the instruments have not yet been tested for the lifetime of the Cerberus mission. These components, however, have undergone rigorous ground testing and by the proposed launch date in 2002, they will have experienced at least 12 years of flight testing, too. Finally, the RFP calls for artificial intelligence to be used where applicable, providing the spacecraft with rapid decision making capabilities while avoiding the long delays involved with communication to and from Earth. Science applications for artificial intelligence include automation of science instruments and data handling. While the above requirements must be fulfilled by the Cerberus mission, there is little restraint on the science which may be performed.

Section 6-2: METHOD OF ATTACK

Since the request for proposal sets so few standards for science objectives, the selection of experimentation is at the discretion of the science subsystem design engineer. In the past, studies of each planet have begun with a flyby of that planet. For instance, Jupiter and Saturn were first studied, up close, by flybys of Pioneers 10 and 11. Similarly, Uranus and Neptune were first explored by Voyager II flybys. These missions serve as a model for the experimentation to be carried out at Pluto. The information received from Cerberus can then be used as a basis for further exploration of Pluto, just as Galileo and Cassini will follow up where Pioneer and Voyager left off.

After it is determined which science will be performed, the selection process begins for finding the equipment that will run the experiments. For Cerberus, this procedure was accomplished by studying past, present, and planned missions. The parameters taken into account included instrument performance (spectral ranges, resolution capabilities, etc.), masses, power consumption, and data rates. Table 6-1 lists the three latter parameters for Cerberus' instruments. Amount of flight testing time was also considered in the process. With the numerous variables to take into consideration, it was difficult to perform numeric trade studies for each component, so the instruments were chosen for their practicality and compatibility with a mission of Cerberus' nature (inexpensive and reliable).

Table 6-1: Summary of Scientific Instruments

Insrument	**Mass (kg)**	**Power (W)**	**Data Rate (bps)**	**Temp. Constraints**
SSI Camera	29.7	15.5 (ave.) 20.0 (max)	34.180 - 888,686	CCD -70deg C (max)

Imaging Spectrometer	(18.0)	(8.0)	(500 - 10,000)	Focal Plane 80 K (max)
Photo-polarimeter/ Radiometer	3.6	7.5 (Photo.) 4.5 (Radio.)	180	-50 to +40 deg C
Ultraviolent Spectrometer	5.33	5.33	-	-18 to +6 deg C
Magnetometers	5.6	2.2	-	-
Plasma	9.9	8.1	-	-
Cosmic Ray	(10)	(10)	-	-
Radio Astronomy /Plasma Wave	1.4	6.7 (Radio) 1.1-1.6 (Plasma)	266 - 115,200 (Radio) 32 - 115,200 (Plasma)	-20 to +70 deg C
Radio Science	0	0	0	-

( ) Estimate - No data available

Ideally, it is desirable to perform as much science as possible on this mission to Pluto. Due to limitations in power supply and instrument endurance, though, the lengthy flight time will restrict the amount of data which can be collected . For this reason, the science sequence must be optimized by giving priority to the most important experiments. Highest priority will go to the remote sensing experiments at Pluto. Next in line are the particles and fields studies at Pluto followed by the exploration of interplanetary space. Final priority goes to the study of Jupiter. This type of method will help ensure that the main objective of the mission, the exploration of Pluto, will be carried out successfully.

Section 6-3: SCIENCE OBJECTIVES

Most of the information regarding the bodies of Pluto and Charon, themselves, will be obtained from a series of remote sensing experiments. An imaging device will take high resolution pictures from which studies will be made providing information about the structure and motion of Pluto's atmosphere (if it exists), as well as information regarding size, shape, color, albedo, surface texture, and spin state. From these images, theories stating that Pluto is covered with a methane haze can be tested and it can be determined whether Pluto has a ring system. Spectroscopy studies will aid in the determination of atmospheric and surface compositions. A photopolarimetry experiment will reveal information about atmospheric particles and the reflective properties of the surfaces of Pluto and Charon. Finally, a radiometer will measure the visible and infrared radiation that is emitted and reflected by the bodies so that the balance of energy between Pluto, its satellite, the sun, and other sources may be studied.

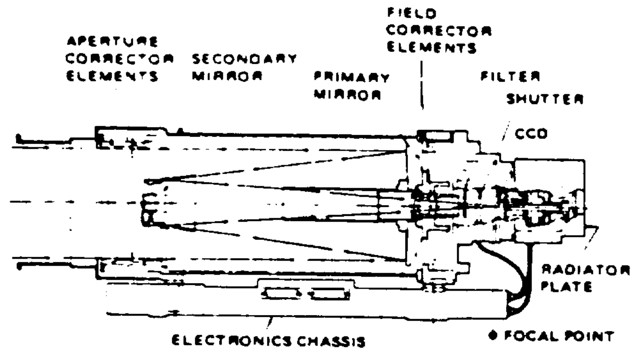
Particles and fields experiments will probe the space around Pluto as well as interplanetary space to enhance our knowledge of the solar wind and its interaction with Pluto. Four magnetometers will measure magnetic field intensity along the spacecraft trajectory allowing scientists to estimate the shape and properties of Pluto's magnetic field. A plasma instrument will identify and sample the energies and velocities of low energy ions and electrons for studies of the solar wind and Pluto's magnetosphere. Properties of cosmic rays will be tested by an instrument that measures the energies and distribution of the high energy particles that make up these rays. The results of this experiment should help scientists determine the origin and motion of cosmic rays. Complimenting the plasma studies, a plasma wave experiment will study the propagation of disturbances through plasma.

Other investigations include radio astronomy and radio science experiments. Radio astronomy will involve the transmission to Earth of various radio signals that Cerberus will encounter during its journey, including those emitted by Pluto. The radio science experiment will utilize the communication system aboard Cerberus in an effort to estimate the mass and size of Pluto and Charon. The radio signals sent back from Cerberus, during Earth occultation, will contribute to the investigation of atmosphereic density and composition.

Section 6-4: COMPONENTS

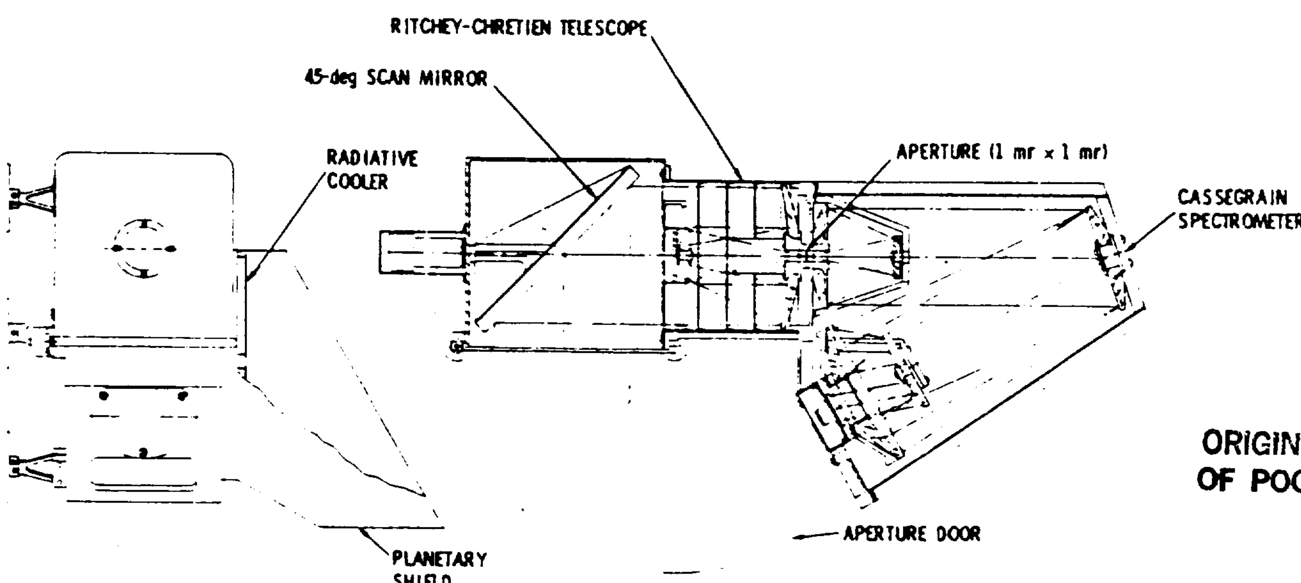
With the science objectives determined, it is necessary to select the best off the shelf hardware to run the experiments while conforming to mass, size, cost, and power constraints. To perform the imaging, the solid state imaging device (SSI) from Galileo will be inherited (See Fig. 6-1.). This system utilizes an 800 x 800 element, charged-coupled device (CCD) with a silicon image sensor array. Chosen for its high resolution capabilities, the SSI camera is more than one hundred times as sensitive as the comparable vidicon-tube camera used on Voyager.[1] This camera features an eight position filter wheel with filters centered in the 727nm. and 889nm. methane absorption bands. It has four exposure times and four repetition rates ranging from 2 1/3s. to 60 2/3s.[1] The angular resolution of the SSI device is 8,128 mrad., and to minimize the amount of data space necessary for each image, the camera has the ability to compress data at a ratio of about 2.5:1.[1] The CCD is sensitive to radiation and operates at a maximum temperature of about -70 deg. C., so sensor shielding and radiative cooling will be necessary.[1]

Visible and infrared spectroscopy measurements will be taken by the imaging spectrometer developed for the Mars Geoscience / Climatology Orbiter (MGCO) (Fig. 6-2). This instrument was selected because, in spite of the fact that it is scaled from Galileo's imaging spectrometer, it utilizes superior sensor technology without additional weight or power requirements. A silicon detector will sense visible and near infrared light ranging from 400 to 1000nm., and an indium antimonide sensor will be used for infrared radiation



Parameter	Value
Angular resolution	10.16 μr/pixel
Shortest exposure	4-1/6 ms
Longest exposure	51.2 s
Active CCD area	12.19 × 12.19 mm
Array aspect ratio	1 to 1
Pixel aspect ratio	1 to 1
Active lines per frame	800
Active pixels per line	800
CCD full well capacity	1×10^5 electrons
Dark current	<10 electrons/s/pixel
Bits/picture element	8 raw 3.24 compressed
Readout noise	≈30 electrons root-mean-square/pixel
Number of filters	8
Gain states	4 (1, 4, 10, 40)
Mass	29.7 kg
Average power	15.5 W
Peak power	20.0 W
Volume	L 90 cm W 25 cm H 30 cm

Figure 6-1: SSI camera and parameters [1]



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Parameter	Value
Design Altitude, km	300
Ground IFOV, m	300
Spectral Coverage, μm	0.4 to 3.5
Spectral Sampling Interval, nm	20
Field of View, deg	7.7
Swath Width, km	40
Data Rate	Variable (Mode Dependent)
Aperture Diameter, cm	16.2
Instrument Temperature, K	200
Signal-to-Noise Ratio at 3.5 μm	105
Signal-to-Noise Ratio at 2.0 μm	500
Focal Plane:	
Visible and Near Infrared (0.4 to 1.0 μm)	Silicon 32-Element Line Array
Short Wavelength Infrared (1.0 to 3.5 μm)	Indium Antimonide 128-Element Line Array
Focal Plane Temperature, K	80

Figure 6-2: MGCO imaging spectrometer and parameters [2]

between 1000 and 3500nm.[2] The MGCO spectrometer operates at a temperature of 200K and the focal plane must be kept below 80K. so a temperature control system must be implemented.[2]

A Photopolarimeter/Radiometer (PPR) device identical to that aboard Galileo will be used on Cerberus (Fig. 6-3). This configuration is advantageous because two experiments share the same equipment which reduces spacecraft weight and complexity. The PPR must be kept at about 223K. and may not be pointed toward the sun.[3]

The combination of Galileo's recent technology and flight experience generally make its components the most attractive for use aboard Cerberus, and like a majority of the other instruments, Cerberus' ultraviolet spectrometer (UVS) will be inherited from Galileo. The UVS will examine light ranging from 115nm to 430nm.

This range exceeds $2\frac{1}{2}$ times that of Voyager's ultraviolet spectrometer.[4] This instrument has two separate fields of view and covers its entire spectral range in 4 1/3s.[4] This, and all of the remote sensing instruments, will be located, together, on a platform of high pointing accuracy.

Four triaxial fluxgate magnetometers will be responsible for magnetic field readings. These magnetometers were selected because their capabilities have already been proven aboard Voyager.

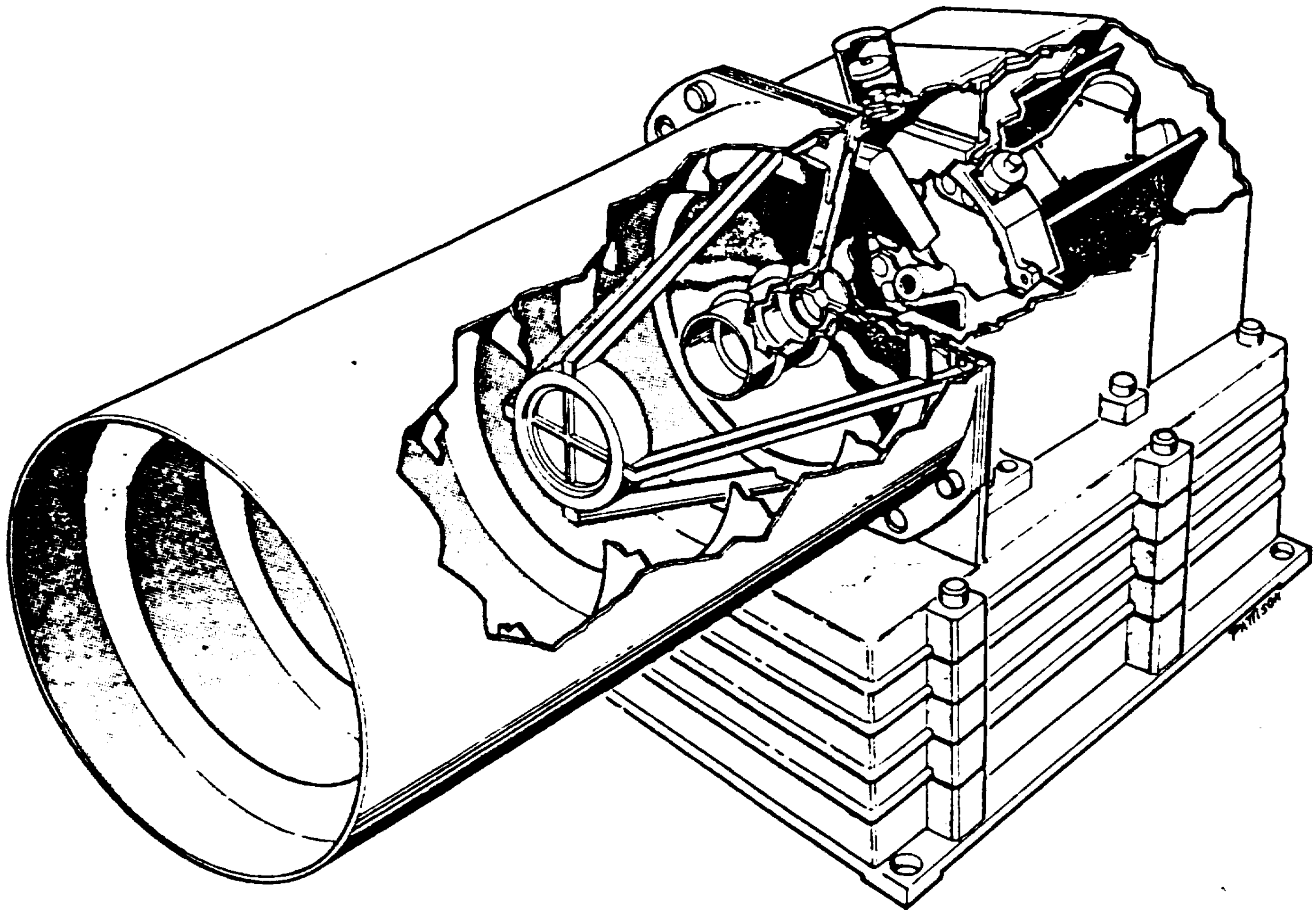
The system will include two high field and low field magnetometers. Due to their sensitivity, the low field magnetometers will be

suspended on a long boom, to avoid magnetic interference from the spacecraft's RTGs and other components.

The superior energy range of Galileo's plasma instrument

(Fig. 6-4) makes it the best candidate for plasma studies aboard Cerberus. Its range of 1.2 to 50,400V. is roughly ten times that of Pioneer's and Voyager's instruments, plus its three mass spectrometers allow it to identify several positive ions.[4] The instrument will be mounted on a rotating, low accuracy platform so that it can analyze particle velocity distributions in all directions.

The energies of cosmic ray particles will be measured by the same type of instrument as used on Voyager. The Voyager instrument was selected because it was designed for a study of the



PARAMETER	CHARACTERISTIC
TELESCOPE	CASSEGRAINIAN TYPE 10-cm APERTURE DIAMETER, 50-cm EFFECTIVE FOCAL LENGTH
COMMANDS	THREE 8-BIT PARAMETER BYTES CONTAINING: MODE (3 BITS); F/R WHEEL POSITION SELECT (5 BITS); GAIN (4 BITS); CAL LAMP ON (1 BIT); NUMBER OF SAMPLES (2 BITS); DC-RESTORE (1 BIT); BOOM START (1 BIT); TELESCOPE COVER STOW (1 BIT); SPACE COVER STOW (1 BIT); TEMPERATURE RANGE SELECT (1 BIT); UNASSIGNED (4 BITS).
INTERFACE SIGNALS	INTERFACE BUS SIGNALS (WORD SYNCH, BIT SYNCH, REAL TIME INTERRUPT, COMMAND BYTES, DATA BYTES); 30 VDC AND RETURN; REPLACEMENT HEATER AND RETURN.
DATA FORMAT	36 BITS ENGINEERING STATUS PLUS 84 BITS SCIENCE AND SAMPLE RELATED DATA.
DATA RATE	120 BITS (BUFFER FILL) EVERY 10 RTIs (2/3 sec) OR 180 BPS.
LOCATION	INSTRUMENT TO BE LOCATED ON SCAN PLATFORM.
VIEWING AND POINTING REQUIREMENTS	INSTRUMENT OPTICAL AXIS TO BE ALIGNED PARALLEL TO THAT OF THE IMAGING AND INFRARED INSTRUMENTS: SCIENCE CALIBRATION TARGET AND SCIENCE BLACKBODY TO BE VIEWABLE OCCASIONALLY: DIRECT VIEWING OF THE SUN TO BE AVOIDED EXCEPT ON A TRANSIENT BASIS.
TEMPERATURE LIMITS	+40°C TO -50°C OPERATING OR NON-OPERATING.
SIZE	43.2 X 19.1 X 24.0 cm (INCLUDING 27 cm TELESCOPE BAFFLE).
MASS	3.6 kg (7.9 lb).
POWER	7.5 watts (CYCLE MODES); 4.5 watts (FIXED APERTURE MODES); REPLACEMENT POWER (4.5 watts).

Figure 6-3: Photopolarimeter / Radiometer and parameters [3]

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outer planets and space beyond the solar system and heliosphere. The extended energy range of 1-500MeV will allow the cosmic ray instrument to study particles of higher energy that may exist in deep space.[5] The instrument will accompany the plasma instrument on the low accuracy platform.

Like Voyager, the design of Cerberus includes two long, perpendicular antennas that will be used for both the plasma wave and astronomy experiments. The electronics for the two experiments will be incorporated into one component, thus conserving space and mass (Fig. 6-5). For the plasma wave studies, the antennas will act as a dipole antenna, and for radio science, they will serve as two monopoles.

The last experiment, radio science, requires no instrumentation. Instead, it will use Cerberus' high gain antenna and communication system. An analysis of perturbations in the spacecraft trajectory will yield mass estimates for Pluto and Charon, while disturbances in the radio transmission to Earth as Cerberus enters Earth occultation will reveal atmosphereic properties. The length of occultation time will help astronomers better estimate the size of Pluto.

Section 6-5: SCIENCE TIMELINE

The period of time dedicated to studying Pluto will last 50 days, centered around the closest approach date. This period will begin with remote sensing measurements taken at a low rate. At this time, the SSI camera will operate at its slowest exposure time. About two weeks prior to closest approach, the particles and fields instruments will begin to take readings at a much higher rate than in interplanetary space as Cerberus approaches the magnetosphere of the Pluto system. At about one week before closest approach, the frequency of readings will have increased to a level which will necessitate a steady rate of data transmission. The peak experimentation period will take place 24 hours before and after closest approach. The camera will collect images at it highest rate while operating at its fastest exposure time to avoid distortions due

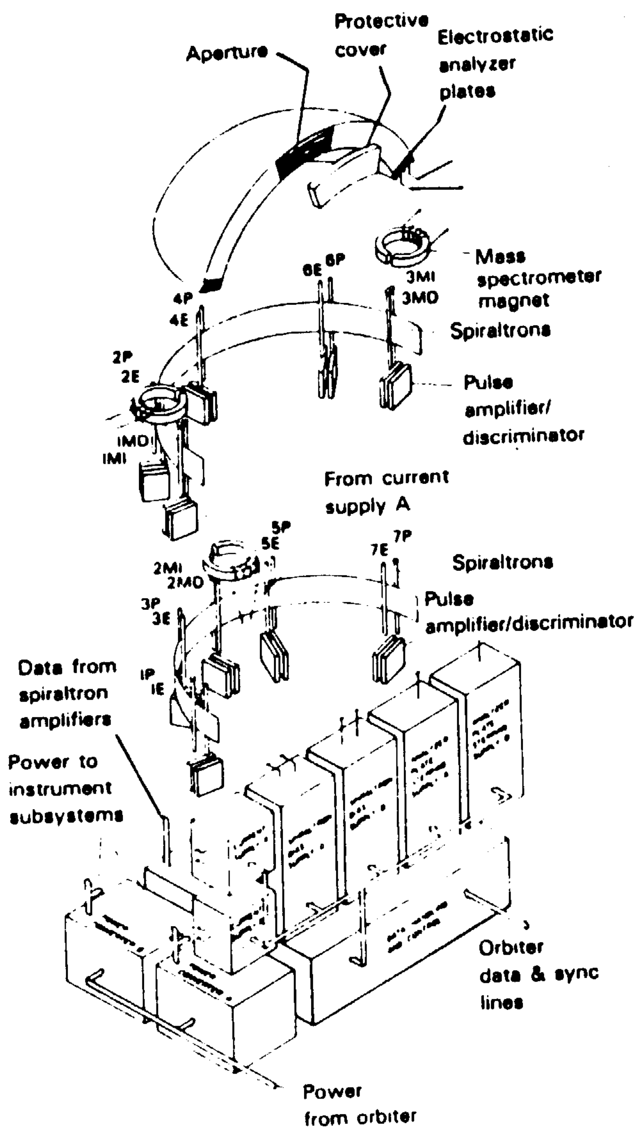

Figure 6-4: Plasma instrument - exploded view [4]

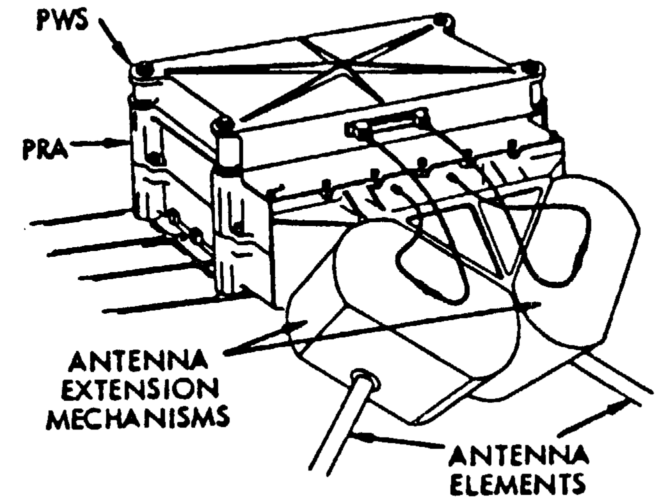

Figure 6-5: Plasma wave and radio astronomy instrument [5]

to the motion of the spacecraft. For a 12 to 18 hour interval during this busy period, Cerberus' attitude will be adjusted so that the high accuracy platform will point towards Charon and a detailed investigation will take place. The precise timing for this maneuver will be determined when the configuration of Pluto, Charon, Cerberus, and the sun, during encounter, is better known. The positions of all of the previously mentioned bodies also effect the sequencing of science events. For example, emitted radiation can only be measured on the dark side of the body of interest. Earth occultation is another subject which must be taken into consideration. When Cerberus enters occultation, it will be impossible to transmit data to Earth, so all data will be recorded for later transmission. A signal will still be sent by the spacecraft, however, for radio science purposes. Following the 48 hour peak period, the amount of data taking will be minimal, allowing time for the transmission of recorded data.

The proposed flyby of Jupiter provides an excellent opportunity for further experimentation. For a period of about 120 days, Cerberus will be in the vicinity of Jupiter and could follow a science scheme similar to that at Pluto. Galileo, however, is already on its way to an extensive exploration of Jupiter, therefore, the results of the Galileo mission will dictate what science will take place during the Jupiter flyby.

In interplanetary space, particles and fields readings will be taken at regular intervals for solar wind studies. Prior to the Pluto encounter, data will be collected every 1 A.U. Afterwards, the rate will be increased to once every 0.5 A.U. Using artificial intelligence, unknown bodies in Cerberus' path can be explored as suspicious patterns in the particles and fields data will cause the sensing instruments to search for large objects in deep space.

Section 6-6: INTERACTION WITH OTHER SUBSYSTEMS

The design of a science subsystem, for a complex project as Cerberus, requires an extensive amount of communication between design team members. The mission planning person must provide information on planet and spacecraft dynamics, so that science sequencing can be arranged, and submit details about space environment, so that the components may be sufficiently protected. Since the mission revolves around the experimentation, however, the science subsystem department is generally responsible for providing information. The structures person requires the masses of each component and their desired locations so that the spacecraft design will maximize science performance. The science subsystem must work with the AACS representative to ensure that instruments will be pointed with the necessary amount accuracy. The CCC person must be informed about the data taking scheme, including data rates, so that the communication system will be able to process and transmit data with minimal losses. Automation of science instruments is another concern shared by the science and communications subsystems. Finally, the power and propulsion representative must know about all of the component power requirements in order to create a sufficient power system.

Section 6-7: FUTURE CONCERNS

This preliminary design sets up the basic concepts of the science subsystem of Cerberus, but there are several details to be dealt with in the following design phases. The extended flight time of the mission introduces the question of design lifetime, yet by the proposed launch date in 2002, Voyager's scientific instruments will have been operating in flight for 25 years and Galileo's instruments will have collected 12 years of experience. Power and cost restrictions, as well as the constraints on data transmission, will set a limit on the quantity of science that can be performed. These limitations will have major effects on the details of the science timeline. Another topic for the next design phase is the fulfillment of specific instrument requirements. Heating, cooling, and shielding devices must be implemented into to the spacecraft's design. These are just a few of the many details which still need to be worked out. With the science objectives and instruments intact, however, the preliminary design of Cerberus is a sound one with no major "show-stoppers".

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N91-16049

AAE 241
SPRING 1990
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.

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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)

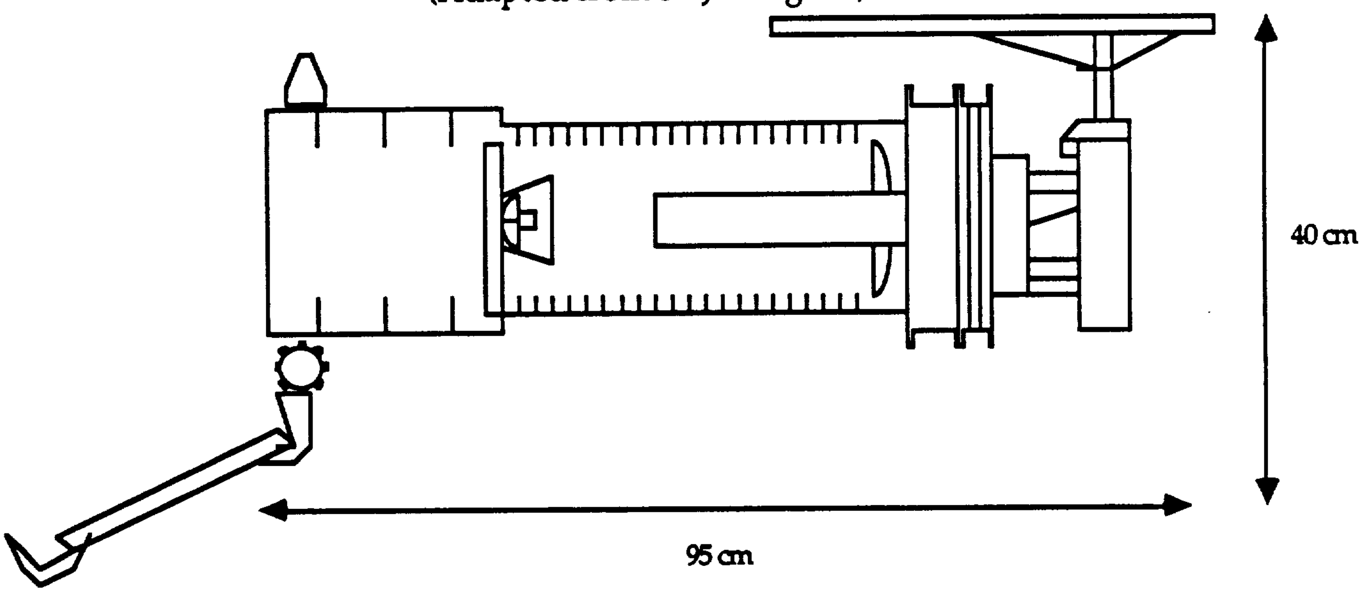
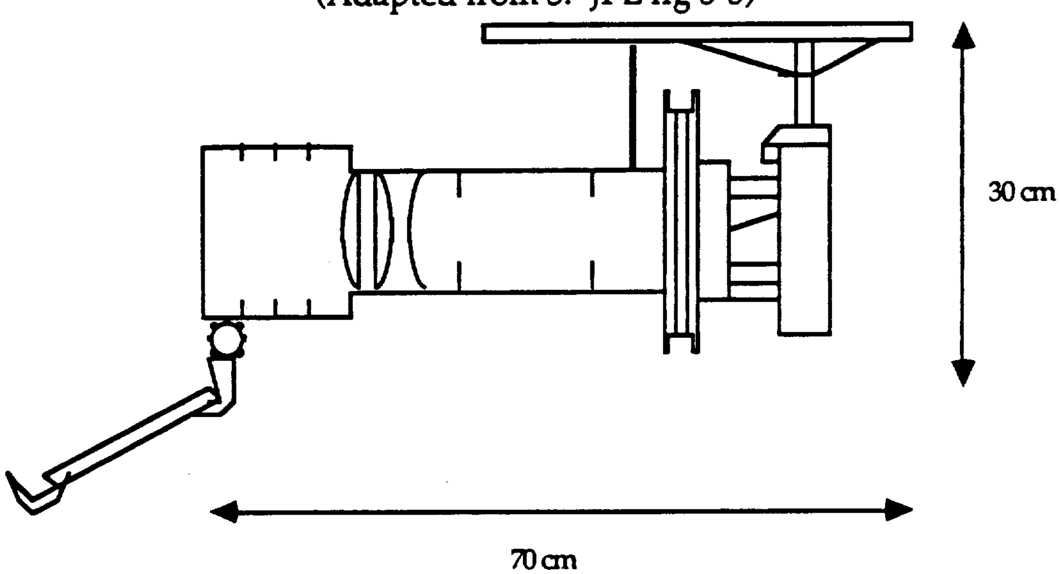


FIGURE 1.2

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.

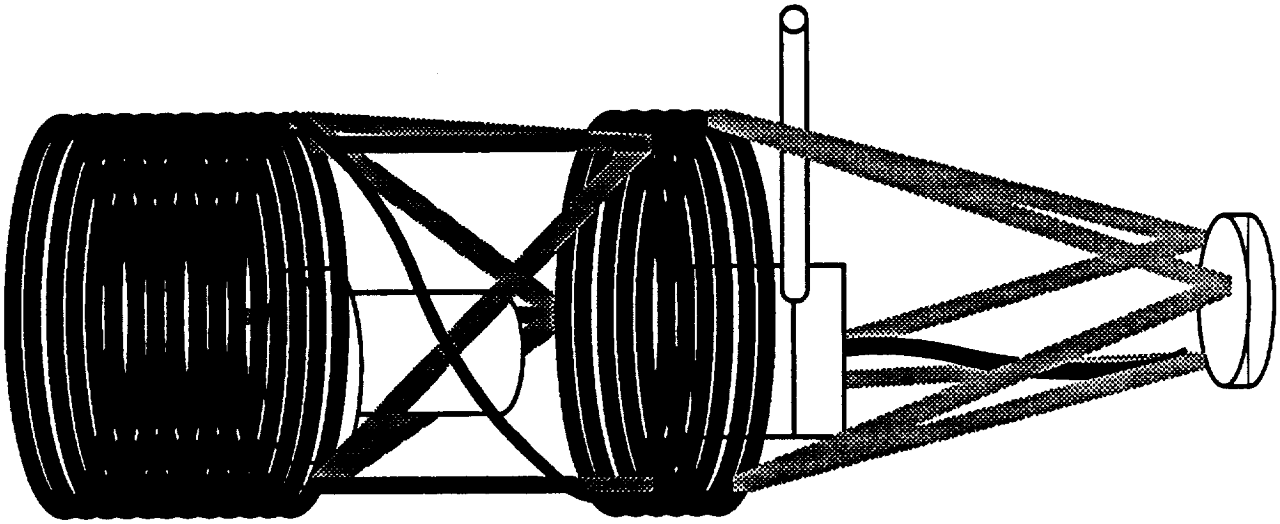
MAGNETOMETER

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

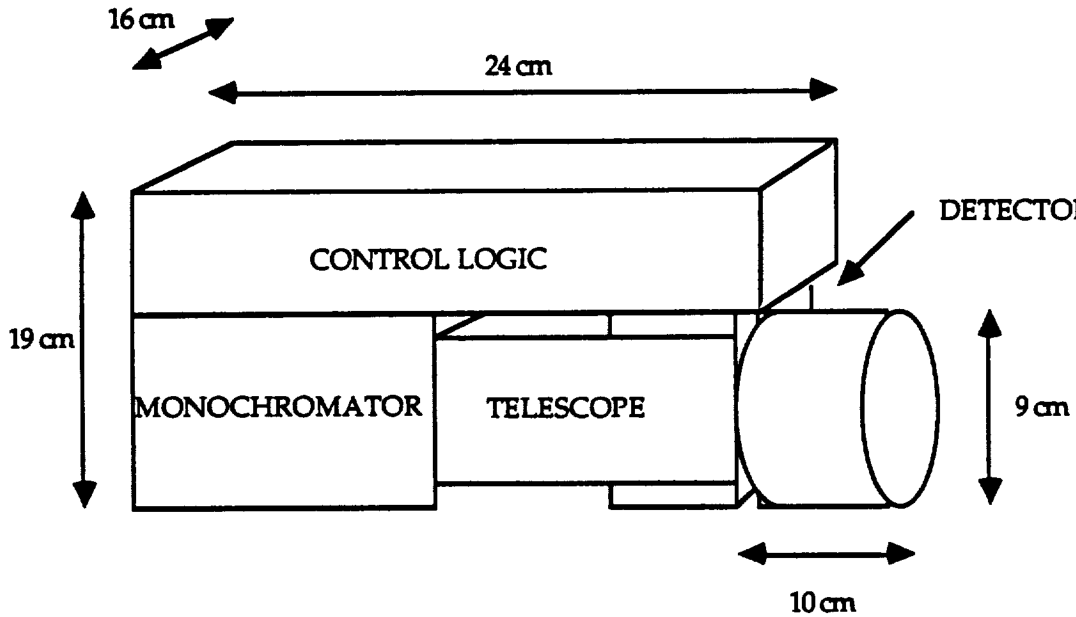
This instrument was also originally designed for the Galileo probe. For Galileo's mission it was used to analyze the Jovian atmosphere and its satellites' surface makeup. Without the highly significant atmosphere of Jupiter's system to analyze, OPTIC's UV spectrometer will be utilized to perform detailed 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:

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

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

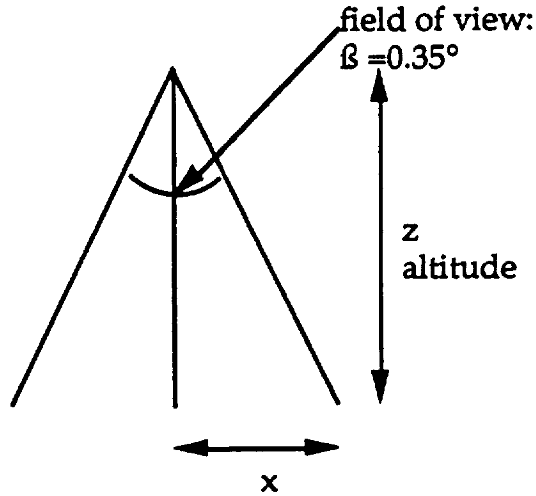
Mapping occurs from f = 270° and f = 90°

These equations yield a mapping pass time of 5.4973 hours.

APPENDIX 1.1 (cont)

Figure 1.5

MAPPING GEOMETRY



x = ztanß

=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^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°)

1.5 BIBLIOGRAPHY

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2.0 MISSION MANAGEMENT, PLANNING AND COSTING

INTRODUCTION

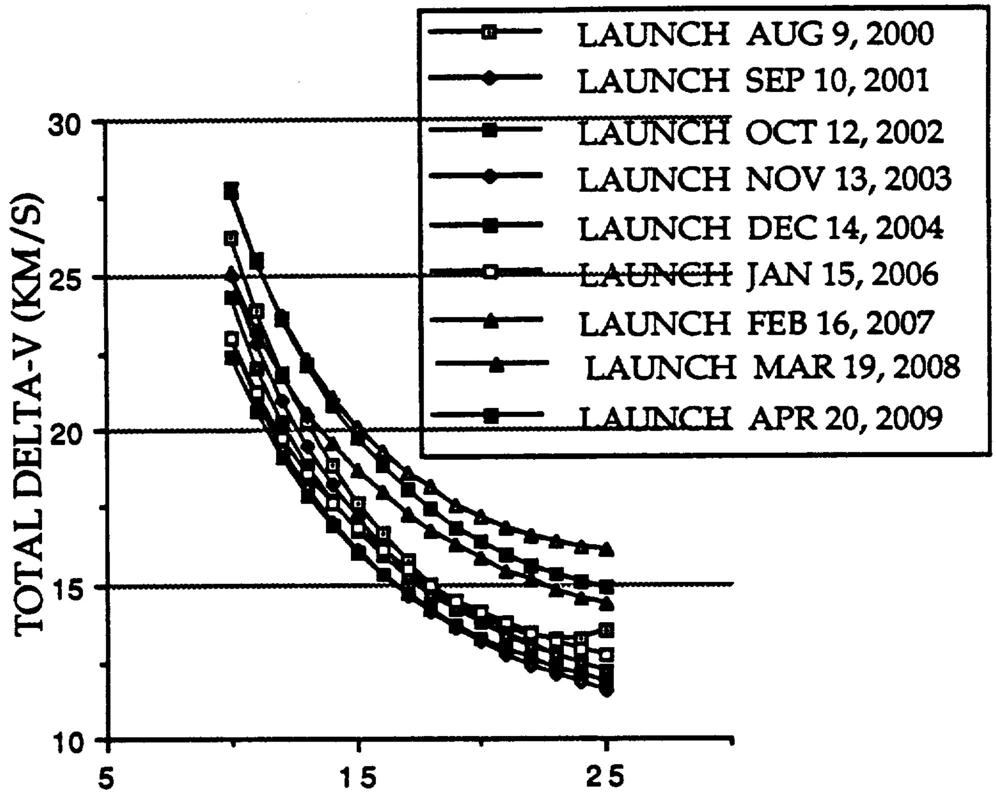
The primary requirement and problem for an orbiter is the need to keep the velocity at Pluto a minimum. If the velocity is too high, the amount of fuel needed for orbit insertion becomes unrealistic. Also of concern is the need to keep the delta-v at Earth departure at a minimum while reducing mission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

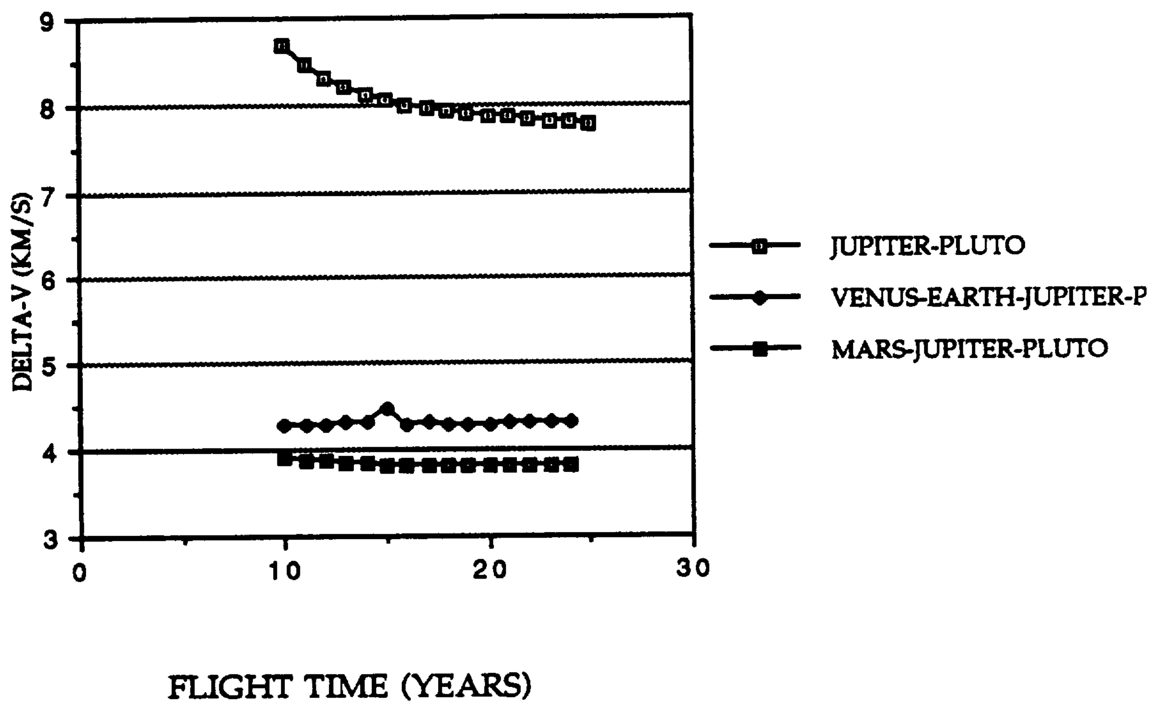


FIGURE 2.3
COMPARISON OF FINAL DELTA V's FOR PLUTO MISSIONS WITH ORBIT INSERTION

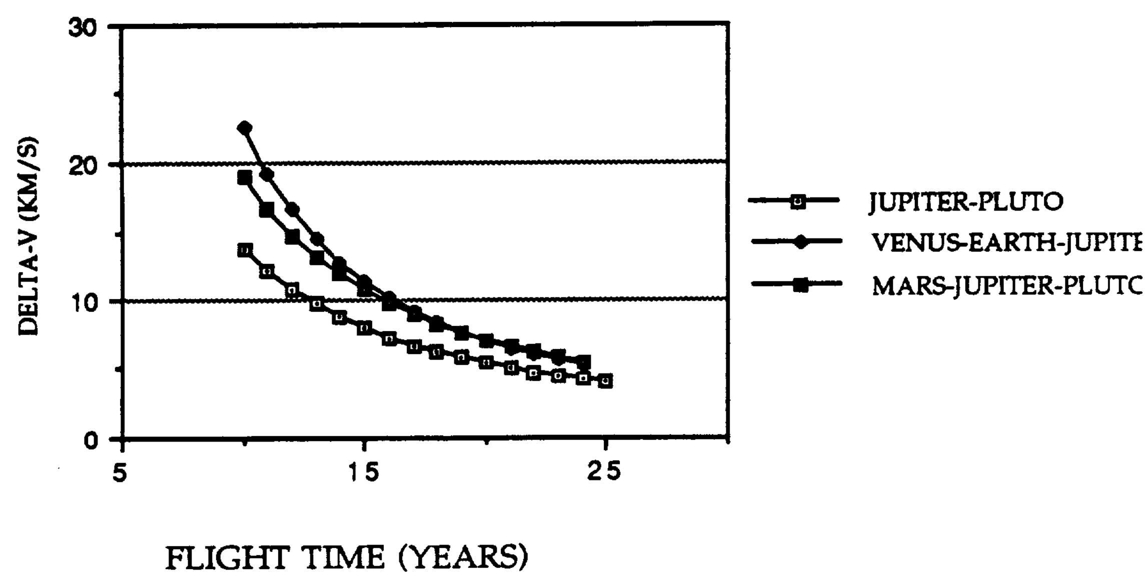
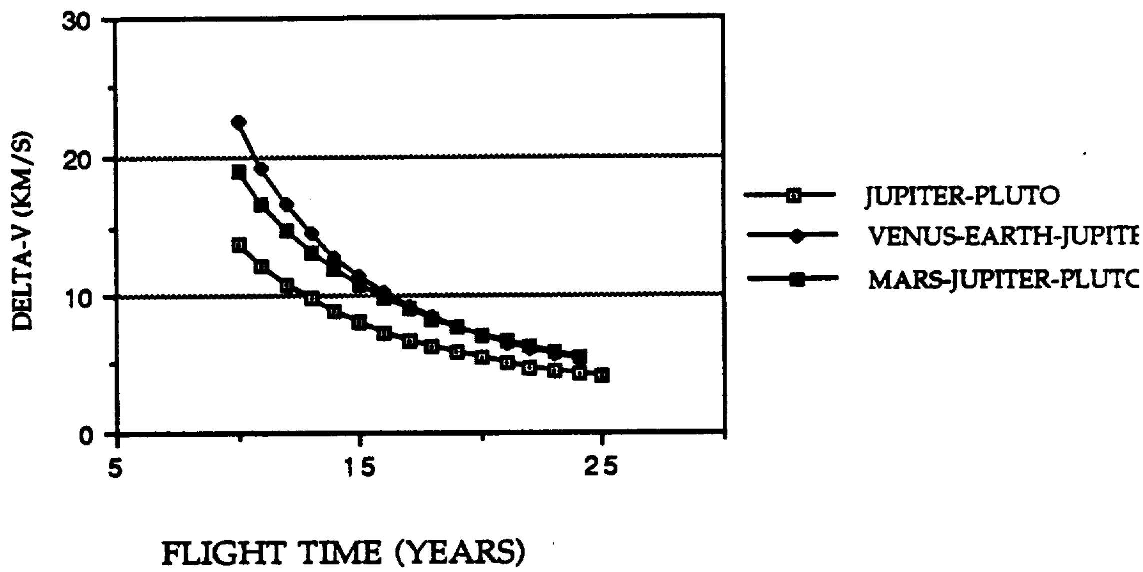


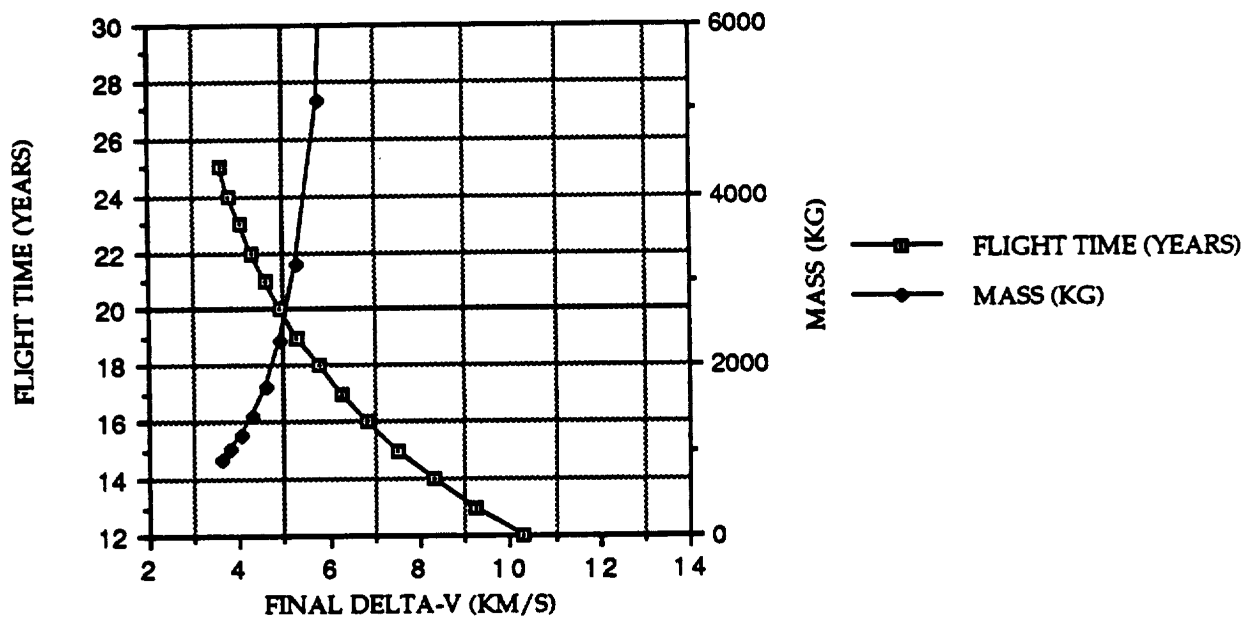
FIGURE 2.4
COMPARISON OF TOTAL DELTA V's
FOR PLUTO MISSIONS
WITH ORBIT INSERTION



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

FORMULAS USED FOR COSTING(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

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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).

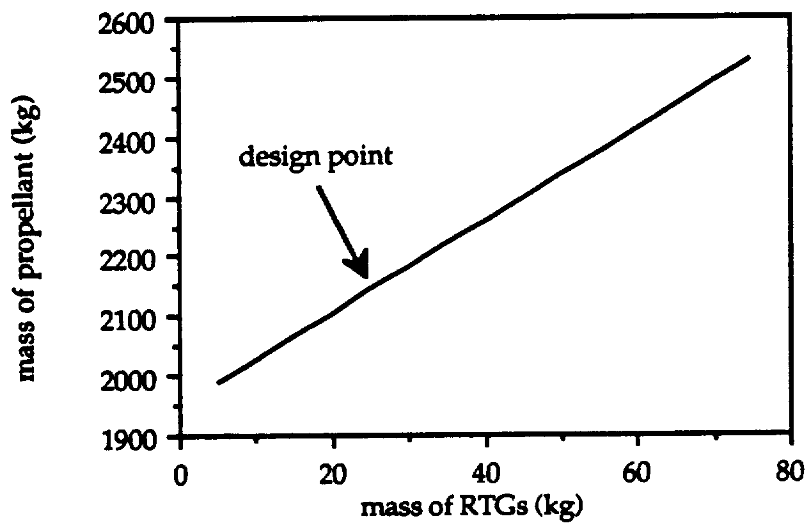
Although nuclear reactors will have superior specific power, other factors must also be weighed. Our mission duration is over 21 years. This figure does not include any time in storage or grounded because of various reasons. The time at which the demand on the power supply will be the greatest is when 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.

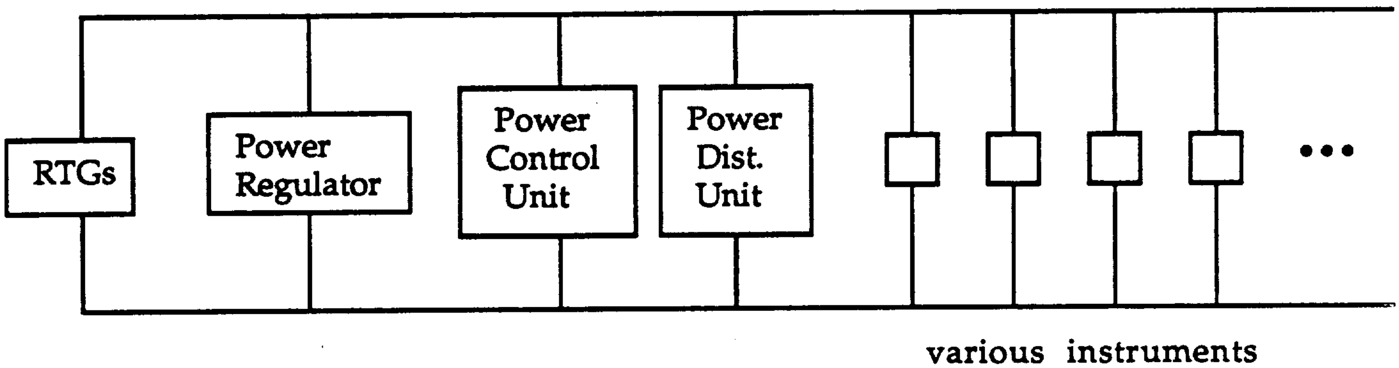
Using the total peak power of 236.38 W, dividing by 12 W/kg and then dividing by .77, the mass of the RTGs that need to be installed initially into OPTIC is 25.582 kg. This is a starting power of about 307 W. The power that will be left on OPTIC, once in Plutonian space, will be able to keep every instrument 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



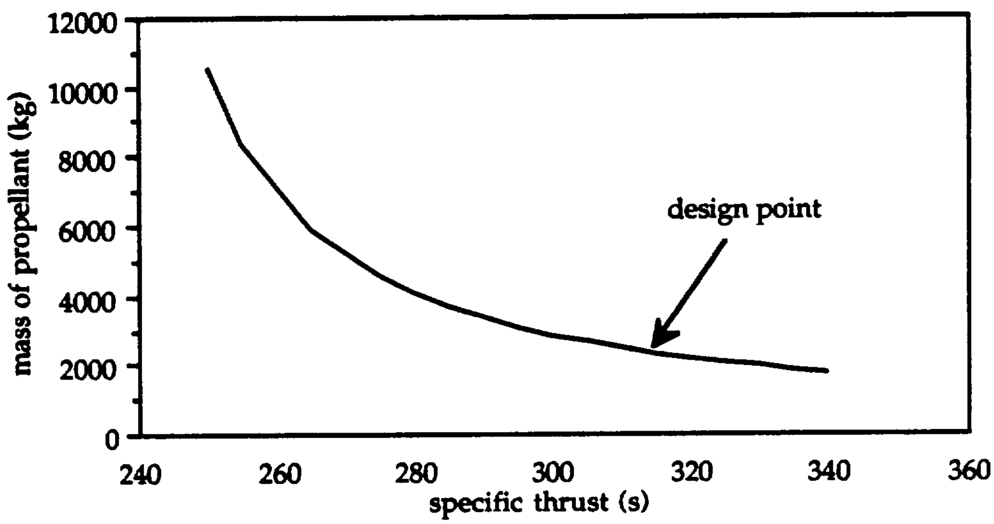
3.2 PROPULSION SUBSYSTEM

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.

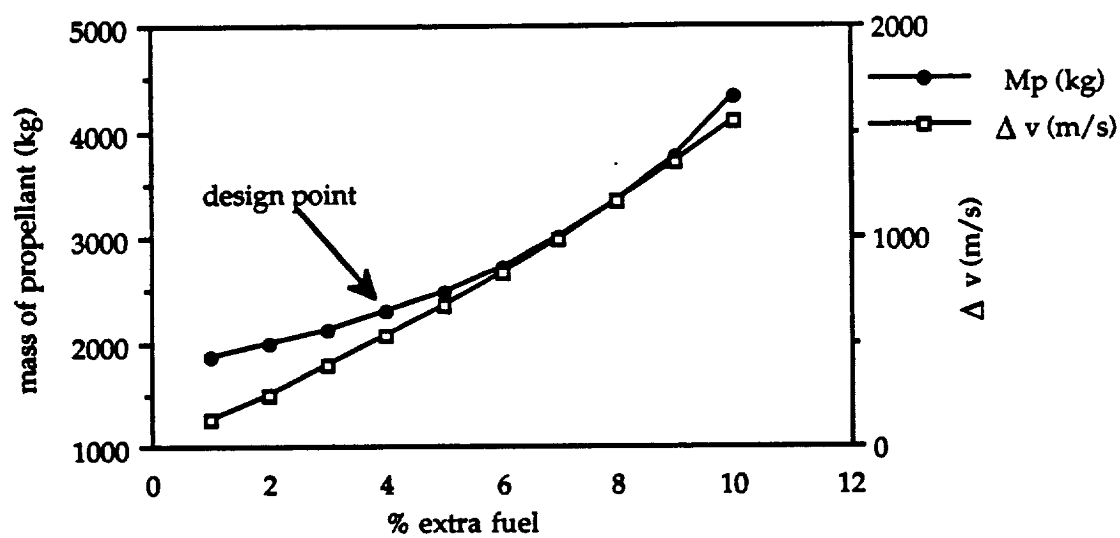
Because of its size, one would assume that the LEROS engine is a poor option. However, since reliability has been the main factor stressed throughout the design process, this makes the LEROS an excellent selection. In order to make the Δv at Pluto of 4927 m/s, it will require a long burn. There will 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

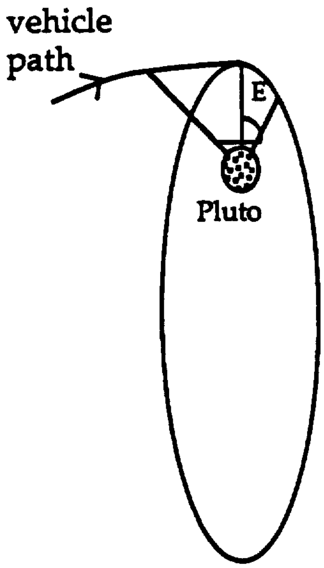
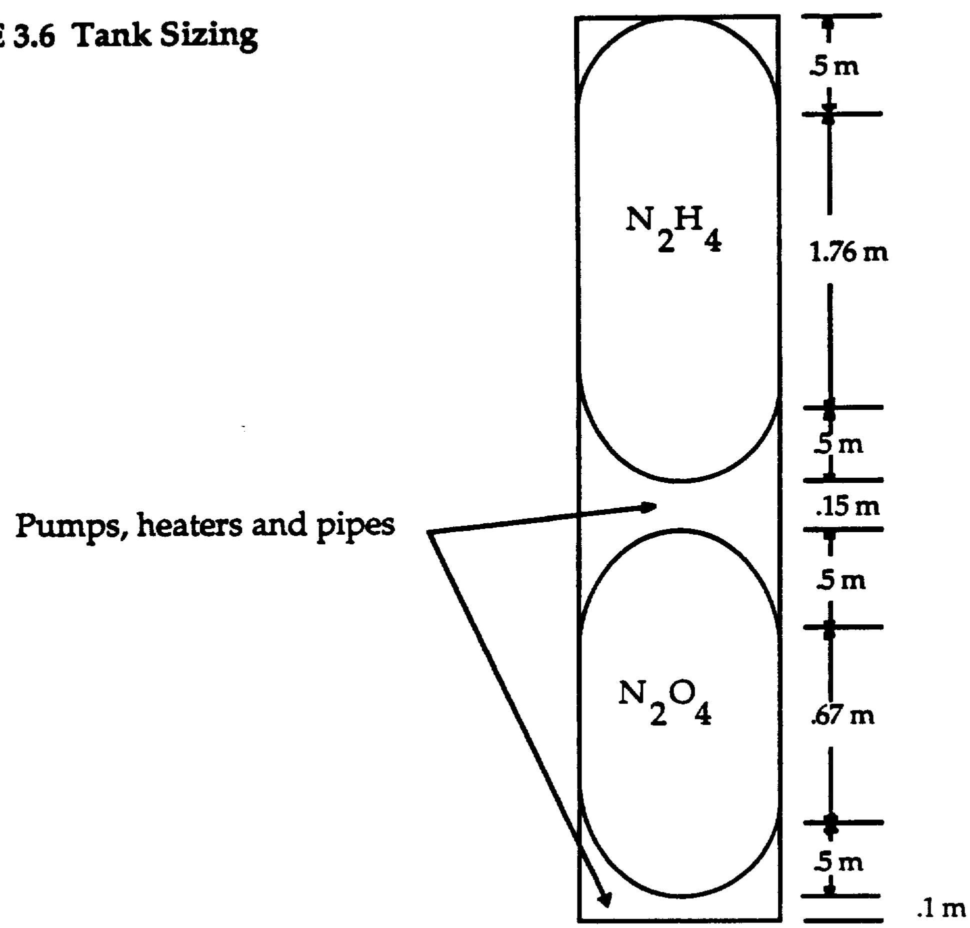


TABLE 3.3 Engine Configurations

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

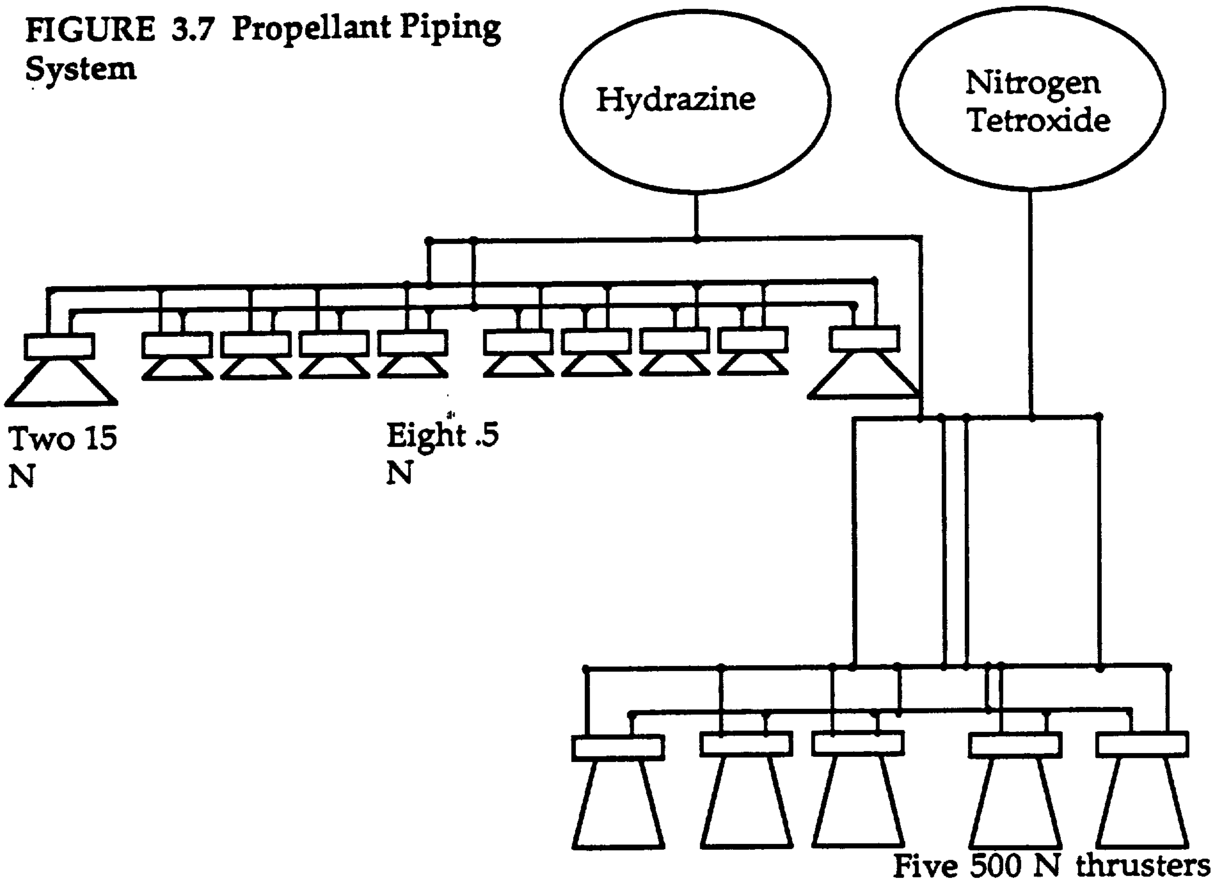


the proposed path and angle (E) that OPTIC will travel past periapse of its orbit about Pluto. Table 3.3 displays results for different engine configurations. Using the impulsive burn approximation it is assumed that the burn will take place at periapse. In this case, the burn will be spread to both 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:

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

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

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

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

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

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

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

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

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

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

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:

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

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

$$\text{where } Mp_{tot} = \text{Mp(from above)} + 95 + 20\text{(attitude control) kg}$$

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

solve for $Mp_{TCM} = 95.968$ 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: $Mp_{TCM} = 439.109$ kg, $M_{TP} = 271.361$ kg

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

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

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.

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

Δv capability once around Pluto:

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

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

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

$\Delta v_{req.} = 47\,\frac{m}{s}$ $\quad \therefore$ 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. $\Delta v = g_0 I_{SP} \times \ln(\frac{M_{ISTV}+M_P+M_{SC}}{M_{ISTV}+M_{SC}})$

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

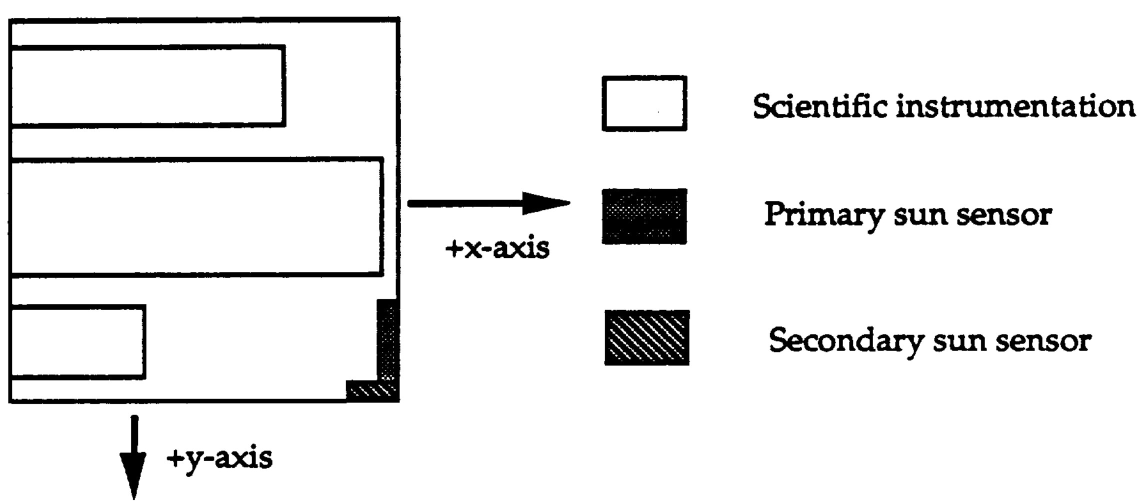
As stated in the request for proposal, the objective of this project is to develop a conceptual design for a spacecraft to study Plutonian space. The optimization of performance, weight and cost are very important. Our choice of an orbiting spacecraft has made the optimization of weight substantially 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

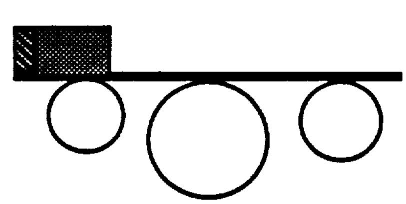
The attitude of this spacecraft will be determined with the use of a fixed head star tracker, a two-axis sun sensor/horizon sensor, two secondary sun sensors, and a state of the art fiber optic rotation sensor. This combination provides reliability, simplicity, and again relatively low weight.

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

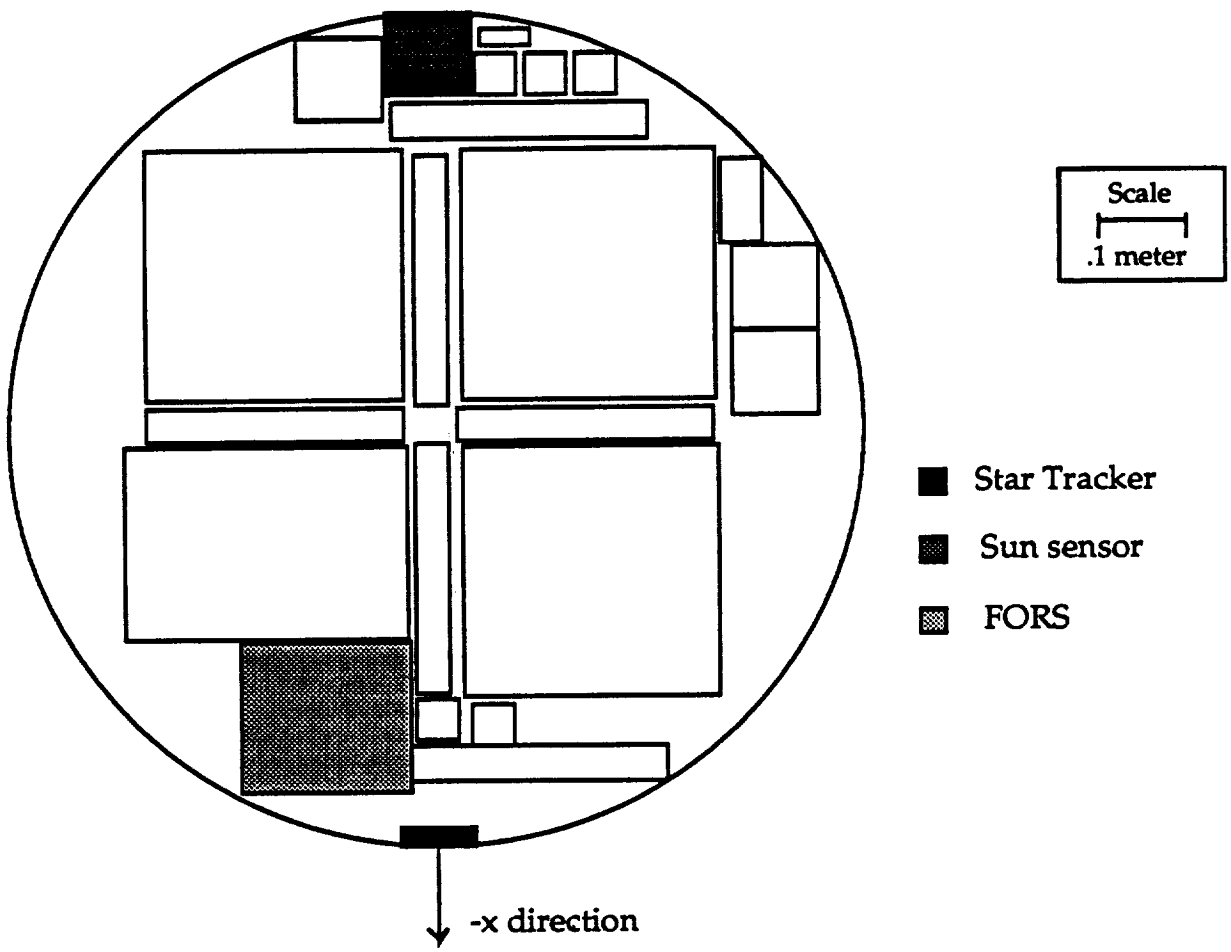


The additional sun sensor will be located on the electronics bus opposite the boom of the scan platform (figure 4.2). The secondary sun sensors will primarily be used during sun acquisition and as a backup for the primary sun sensor. Mass estimates for these sun sensors are based on sensor found in NASA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

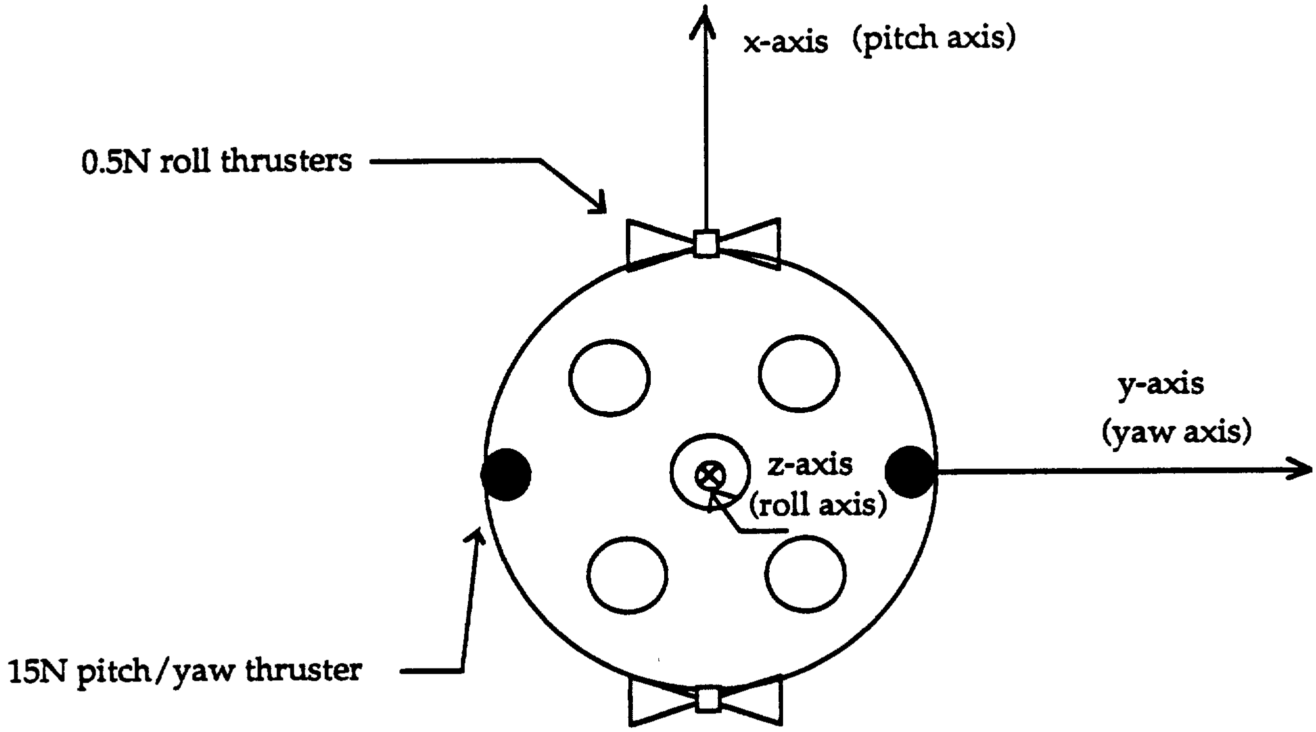
The two 15N jets will serve the additional role on this mission of settling the fuels in their respective tanks before major burns of the primary propulsion system. To assure fuel supply to the 15N thrusters a 1kg capacity nitrogen pressurized hydrazine fuel source has been provided for each jet. Once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

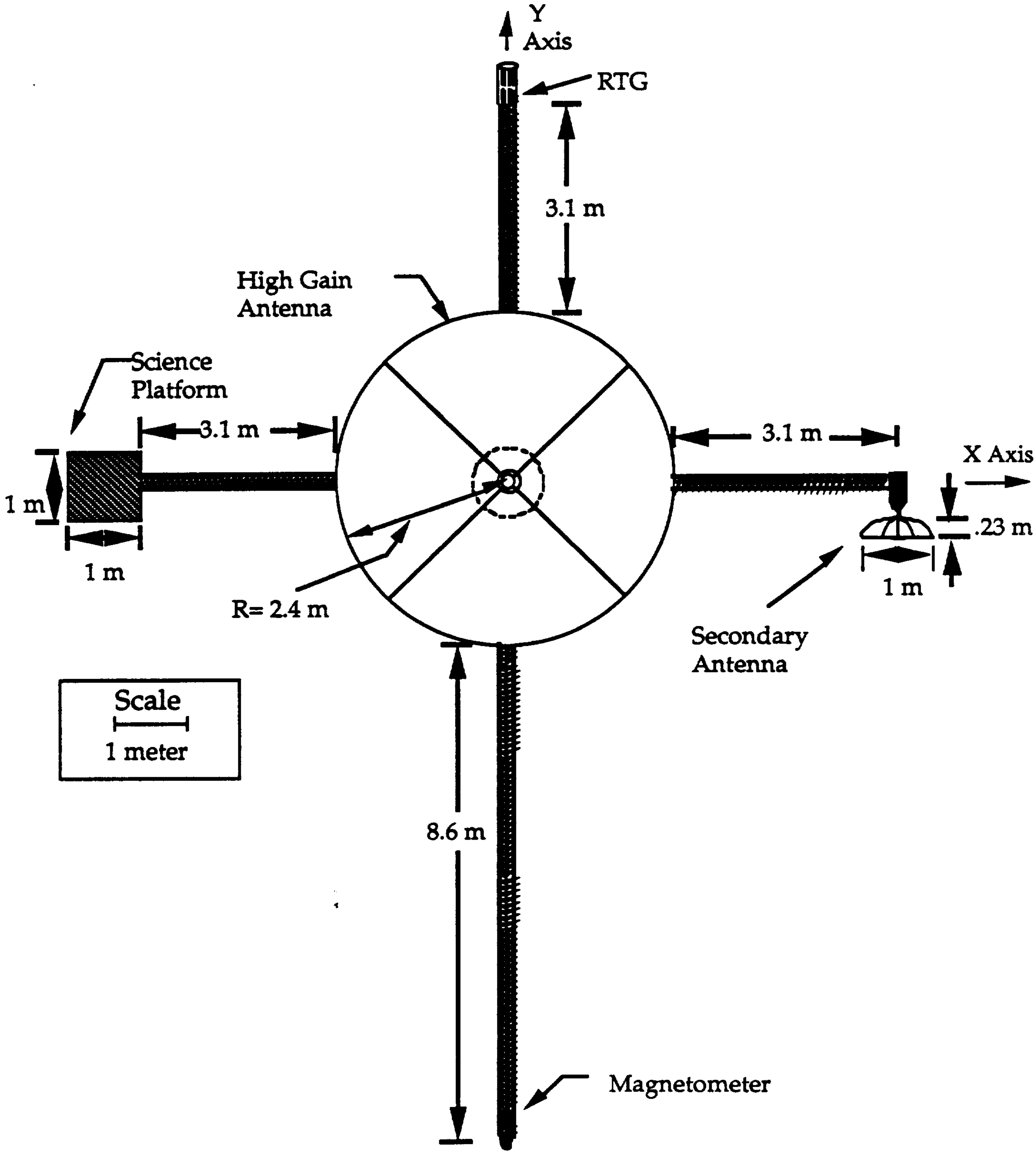
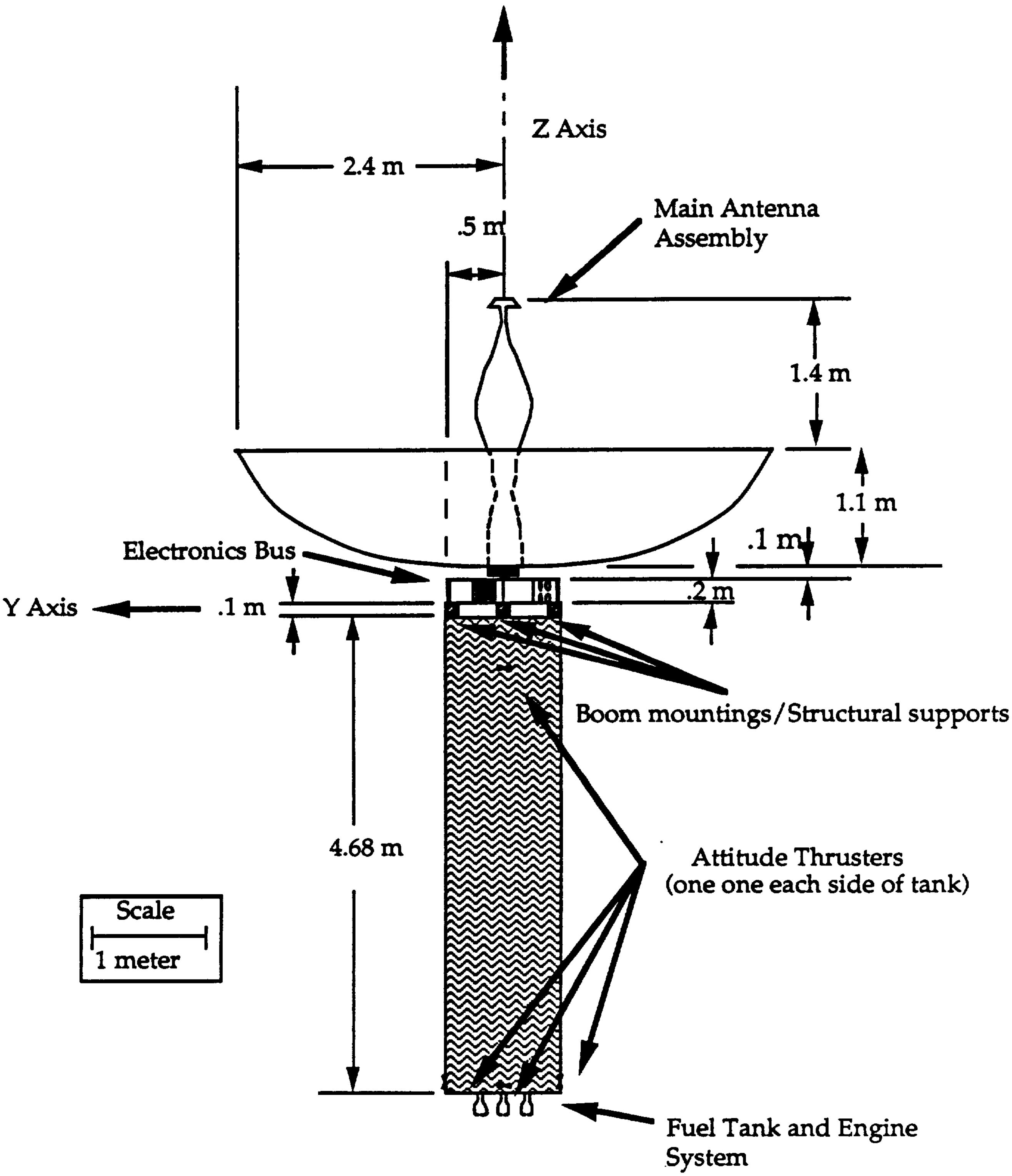


FIGURE 5.2: Side View of the OPTIC Spacecraft



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.

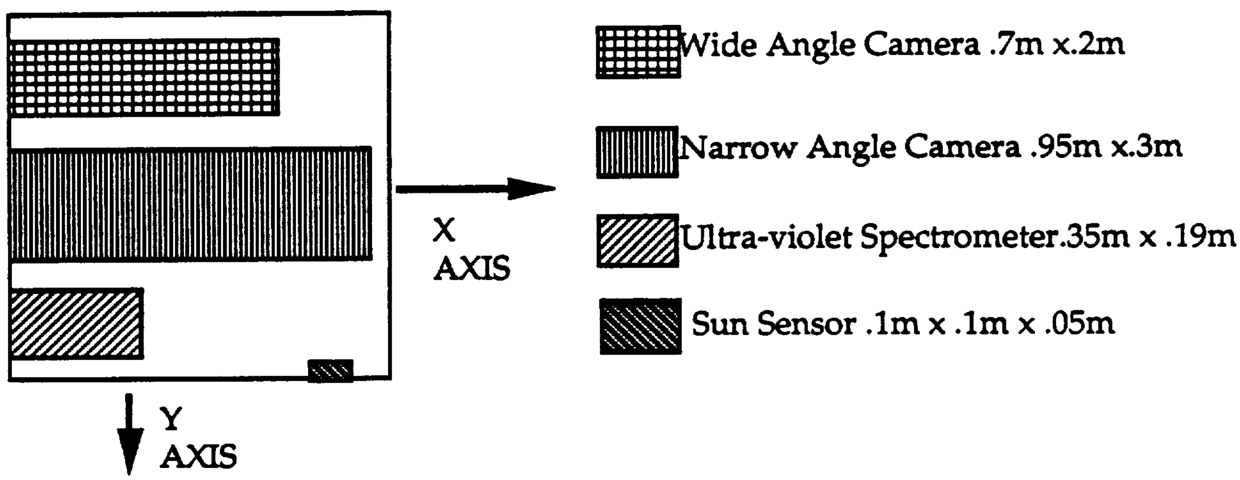
This left the science platform and secondary antenna mounted on five meter booms that are perpendicular to the Y axis. The direction toward the secondary antenna was arbitrarily designated as the positive X direction. See Figure 5.1.

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.

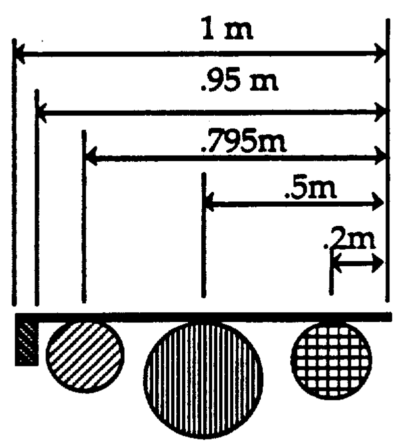
The science platform itself is mounted with actuators that allow it to rotate independently of the spacecraft. This reduces the amount of spacecraft pointing, and therefor reduces the amount of required fuel and total mass of the mission.

FIGURE 5.3 The Science Platform

Science Platform Viewed from Below



Science Platform Viewed from the Side



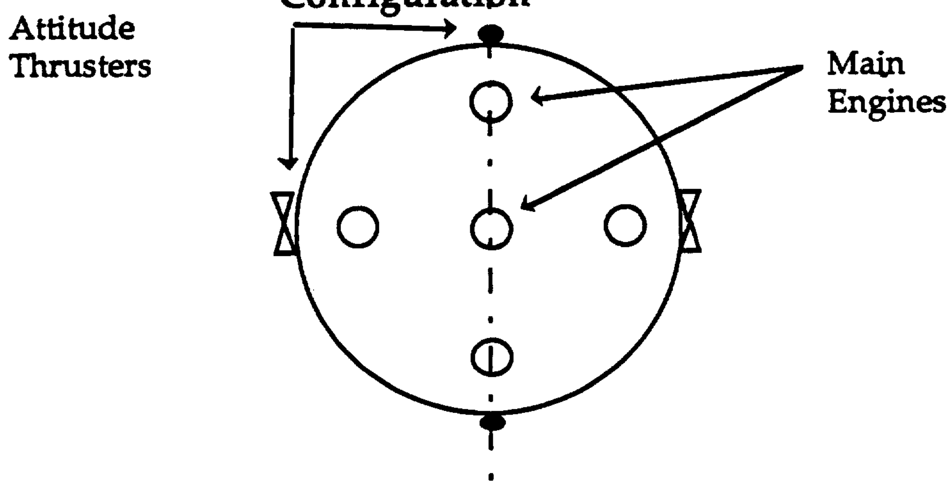
The secondary antenna is also mounted on actuators that allow independent pointing. This function will be used in the mapping portion of the mission.

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

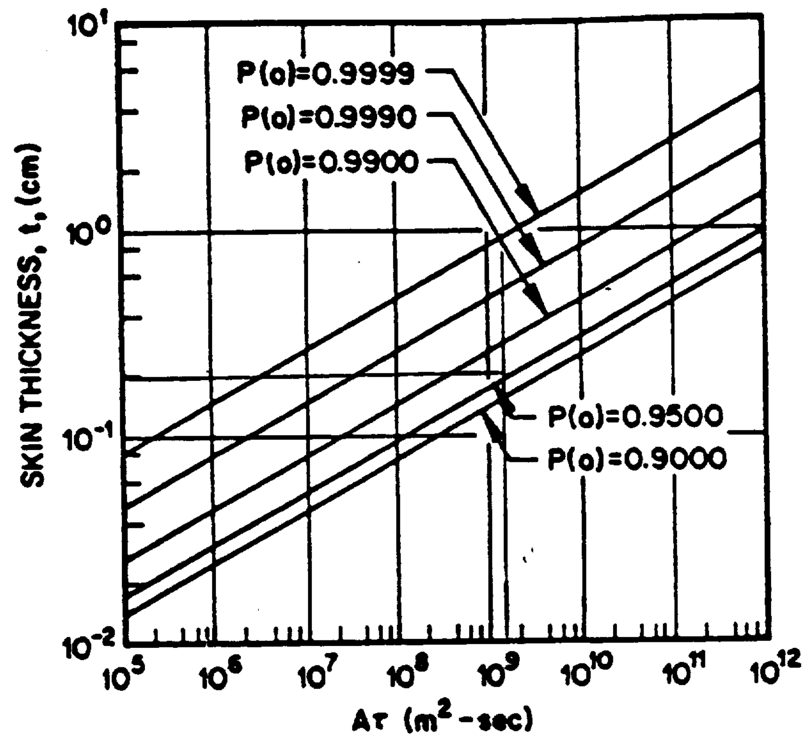


FIGURE 5.5: Sheet Thickness of Al as a Function of the Surface Area-Lifetime Product Required for Various Probabilities of No Meteoroid Puncture (Ref. 5, p. 503).

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

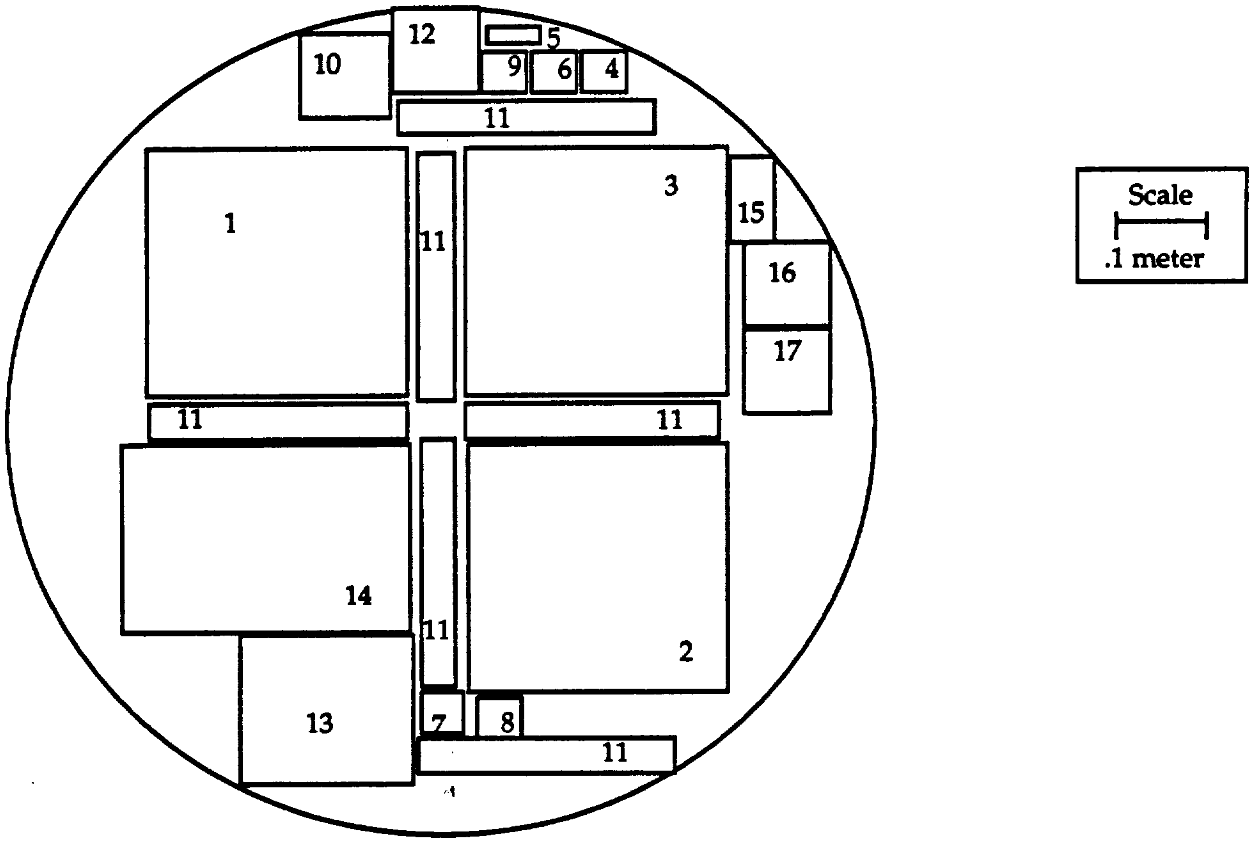


TABLE 5.3: Electronics Compononent Masses

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)
$\alpha = 3.3 \times 10^{-15}$
β = 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.

APPENDIX 6.1

ANTENNA SIZING

$P_r = P_t{*}L_t{*}G_t{*}L_{tp}{*}L_s{*}L_a{*}L_p{*}L_{rp}{*}G_r{*}L_r$ (7. Koepke, p.12)

P_r = Power Received

P_t = Power Transmitted

L_t = System Losses in Transmitter

G_t = Transmitting Antenna Gain

L_{tp} = Pointing Loss of Transmitter

L_s = Free Space Loss

L_a = Atmospheric Attenuation

L_p = Polarization Loss between Antennas

L_r = System Losses in Receiver

L_{rp} = Pointing Loss of Receiver

G_r = Receiving Gain

$L_r{*}G_r{*}L_{rp} = .89$ (4. Hansen, p.113)

$L_t = L_p = 1.0$

$L_{tp} = .89$ (4. Hansen, Hansen, p.111)

$L_a = .92$ (11. Slobin p.140)

Pt = 6.3 watts HGA or 2.1 watts LGA

$G_t = .8{*}(\pi D_t/\lambda)^2$

D_t = Transmitter Antenna Diameter = 4.8m HGA or 1.0m LGA

$G_r = .8{*}(\pi D_r/\lambda)^2$

D_r = Receiver Antenna Diameter = 70m

$L_s = (\lambda/4\pi r)^2$

λ = wavelength = .009375m

r = distance between antennas = 40 AU

Combining these equations yields

$P_r = P_t{*}\pi^2{*}70^2{*}4.8^2{*}.80^3{*}.89^2{*}.92/(16{*}(40{*}1.49{*}10^{11})^2{*}(.009375)^2)$

$P_r = 9.04635{*}10^{-18}{*}P_t$

$P_r = 5.6992{*}10^{-17}$ watts for the HGA at Pluto

DSN is capable of receiving signals as weak as $4{*}10^{-21}$ watts

$P_r = 4.8789{*}10^{-17}$ watts for the LGA at Jupiter

DATA TRANSMISSION ESTIMATES

Shannon's Law (7. Koepke, p.15)

$B = W\log_2(1+P_r/P_n)$ B = Bit Rate

W = Bandwidth P_n = Background Noise

$P_n = K*T*W$ $SNR = P_r/P_n$

K = Boltzmann's Constant   T = Background Noise Temperature = 8°K

$W*SNR = P_r/KT$

W*SNR = 5162331.88 for the HGA at Pluto

Since the SNR needs to be about 25 for good communications

W = 20649.3 Hz   B = 97060.7 bits per second before compression

These numbers can be further optimized if more information about SNR can be obtained.

At Pluto, Science sends information at no more than 250 kb per second. Data can be collected for 5.5 hours out of the 31.5 hours in one orbit. This means that 619 MB can be collected. If 48.5 kB (97000*4/8) per second can be transmitted, it will take 3.54 hours to send all of this data. Since 250 kb per second is an absolute maximum, there should be no problem in transmitting this data in the 26 hours (31.5 - 5.5) of orbital time there is for communication.

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