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Group 1
AAE Final Design Report
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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

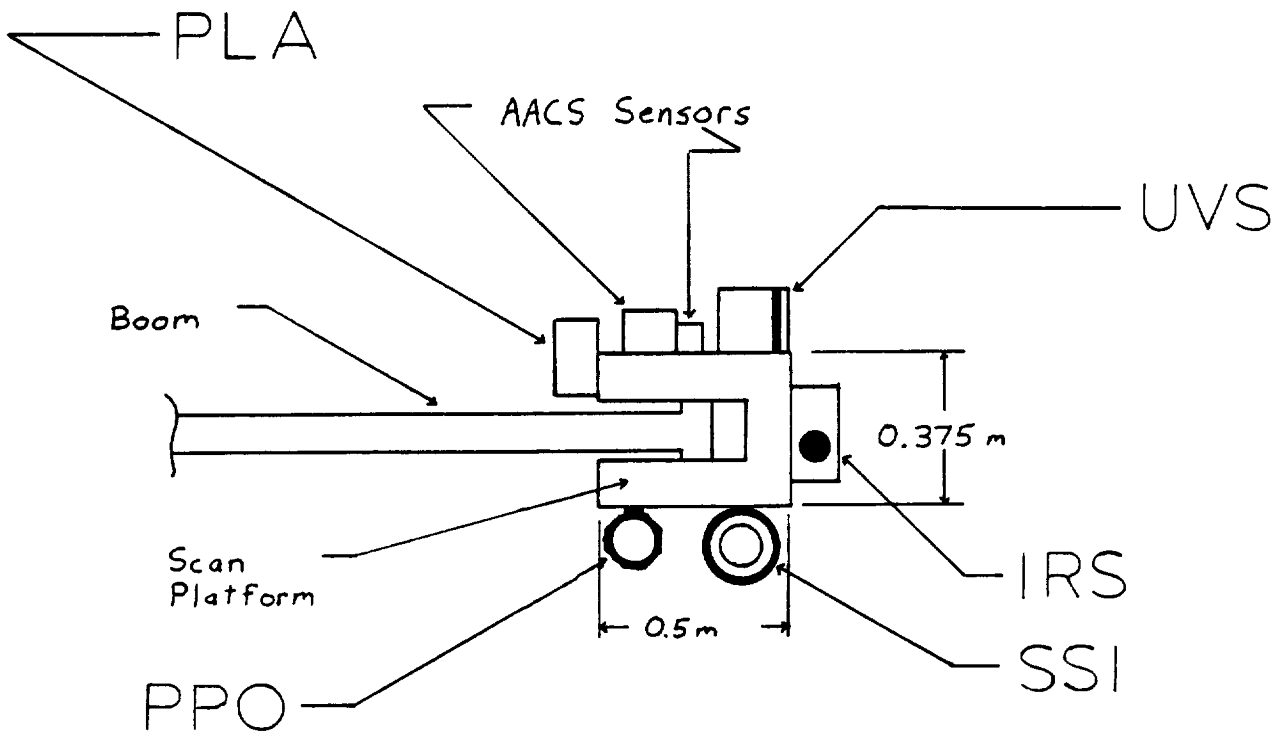
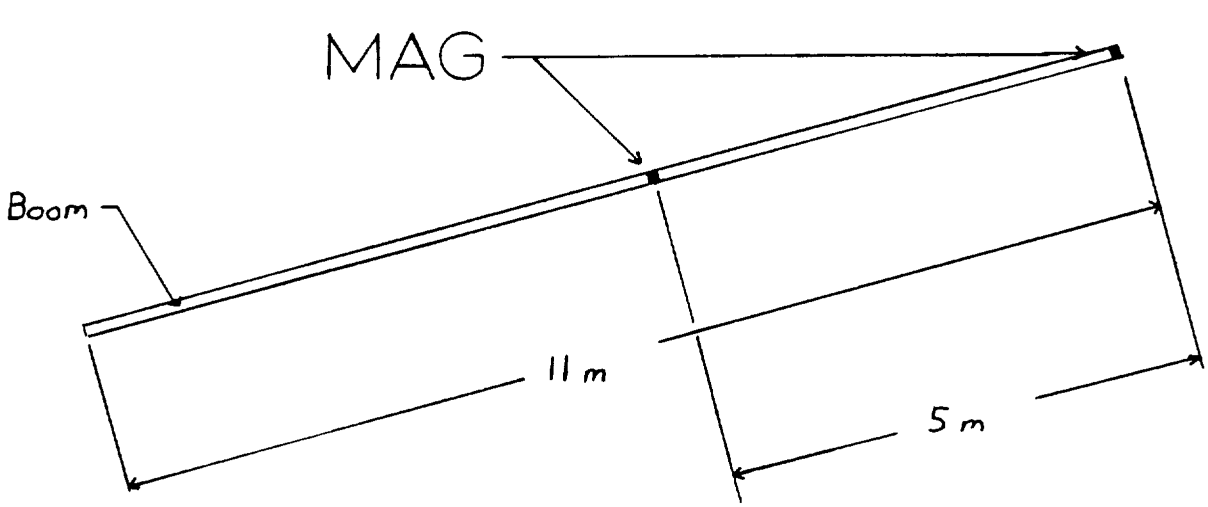


FIGURE SI-2 : Magnetometer Boom



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

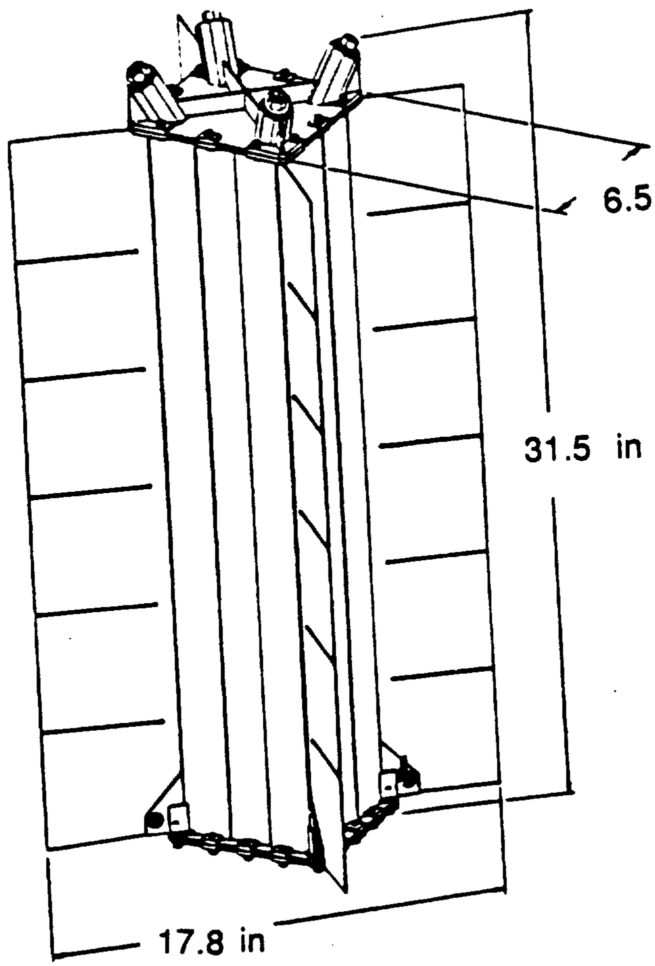

Figure PP-1

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.

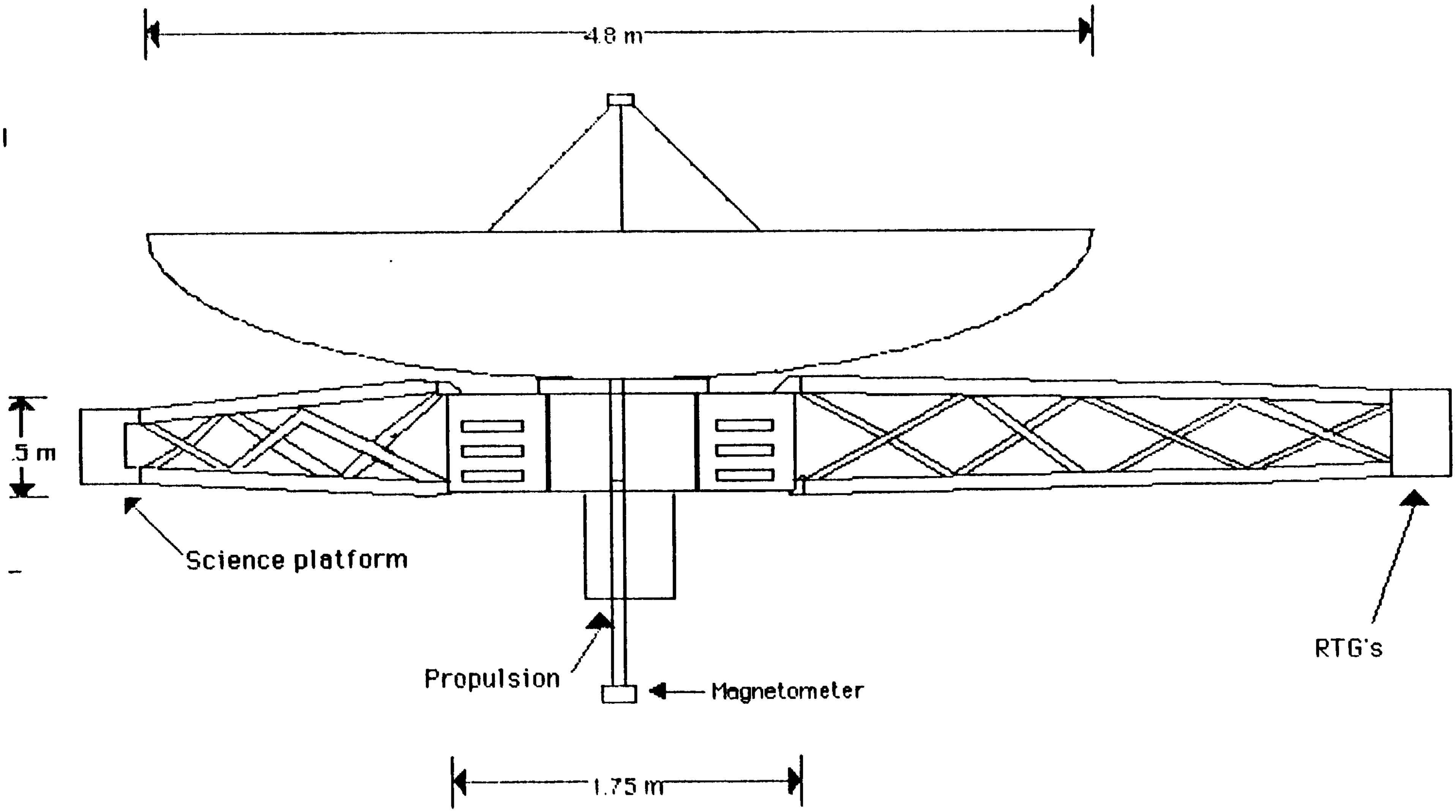
A sample drawing of this craft has been included on the next page. The observant reader will note the similarities in design among Group One's choice and previous craft flown by NASA, such as Voyager, Mariner, and Galileo (Dumas, p. 535). This is not just coincidence. Not only do these designs make good 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.

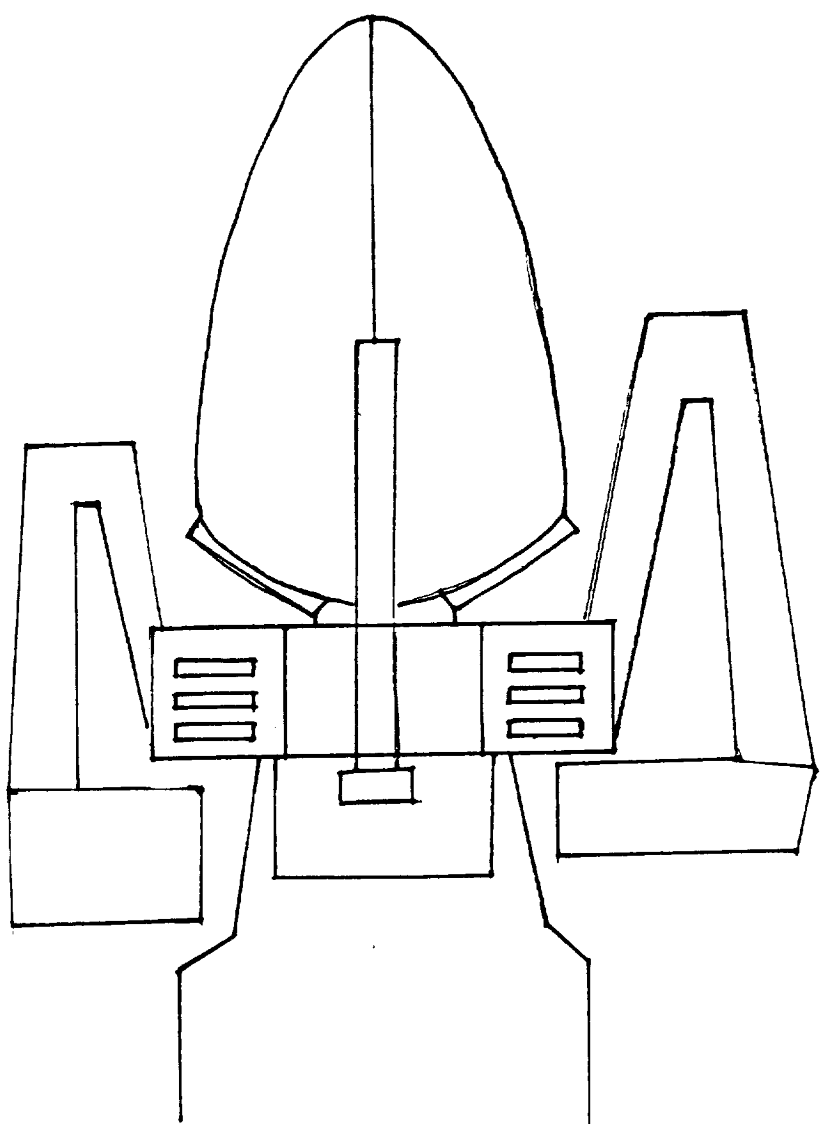
The spacecraft will also have a "fourth boom". The midcourse booster will be attached to the bus on the opposite side of the high gain antenna with explosive bolts to allow it to be jettisoned after firing. This will provide the boost at an intermediate planet to attain the required velocity to make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

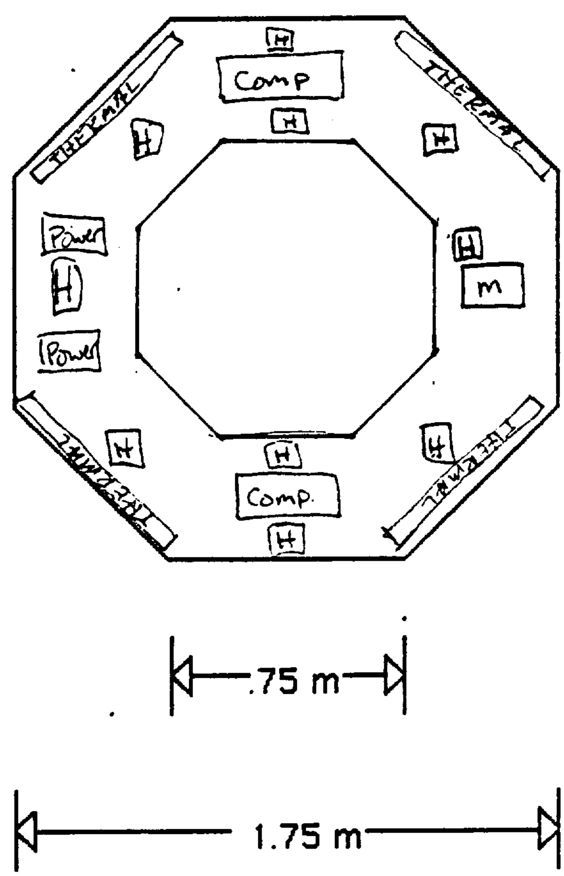
Finally, we must consider the layout of the craft before it reaches orbit. The PFERD mission will utilize the Titan IV to


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.

Jet Propulsion Laboratory Division 35; Mass Estimation Reference (JPL, Pasadena; 1980): Data on Thermal and Structural masses for Voyager and Galileo.

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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
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- - -			
17.68 Years		Total DV =	10.68

MARS-JUPITER GRAVITY ASSIST TRAJECTORY

Date (km/sec)	Location	Velocity (km/sec)	delta-V
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
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- - -			
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

References:

1) Diehl R., Belbruno E., Bender D., Myers M., Stetson D. "Low Launch Energy Trajectories to the Outer Solar System via Venus and Earth Gravity-Assist Flybys" AAS 87-419.

2) Friedlander, A.L. "(MULIMP) Multiple Impulse Trajectory and Mass Optimization Program" Science Application, Inc report No SAI 1-120-383-74, April 18, 1974.

3) Kolcum, E. H. "Air Force, Contractors Predict Long Life for Heavy Lift Vehicle" Aviation Week and Space Technology, July 17, 1989.