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Project Cerberus: Flyby Mission to Pluto

AAE 241: Spacecraft Design Proposal

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TABLE OF CONTENTS

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.

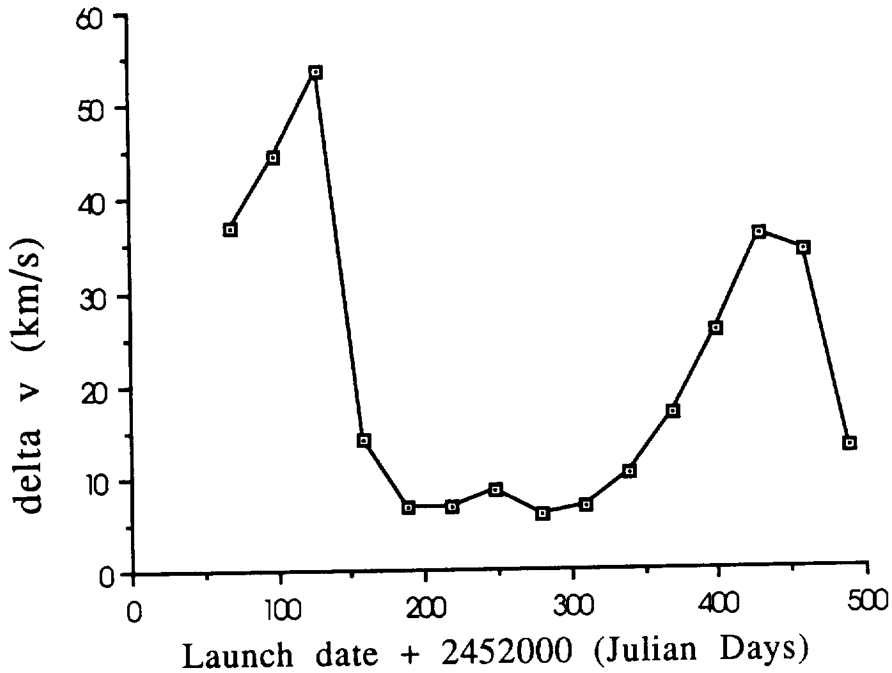

Figure 1-1: Comparison of Delta v versus Launch Date over 420 days

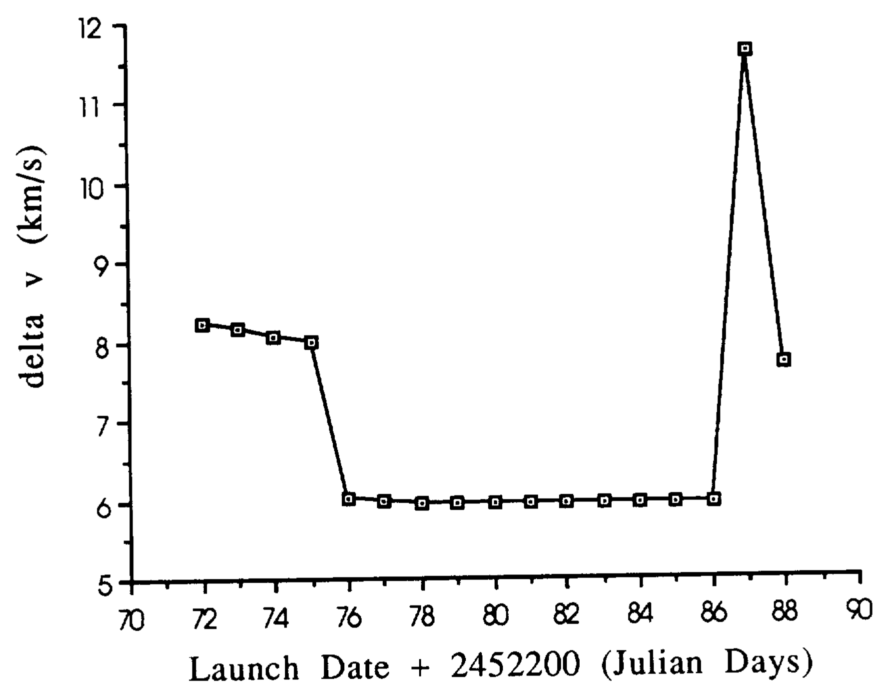

Figure 1-2: Comparison of Delta v versus Launch Date over 2 weeks

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.

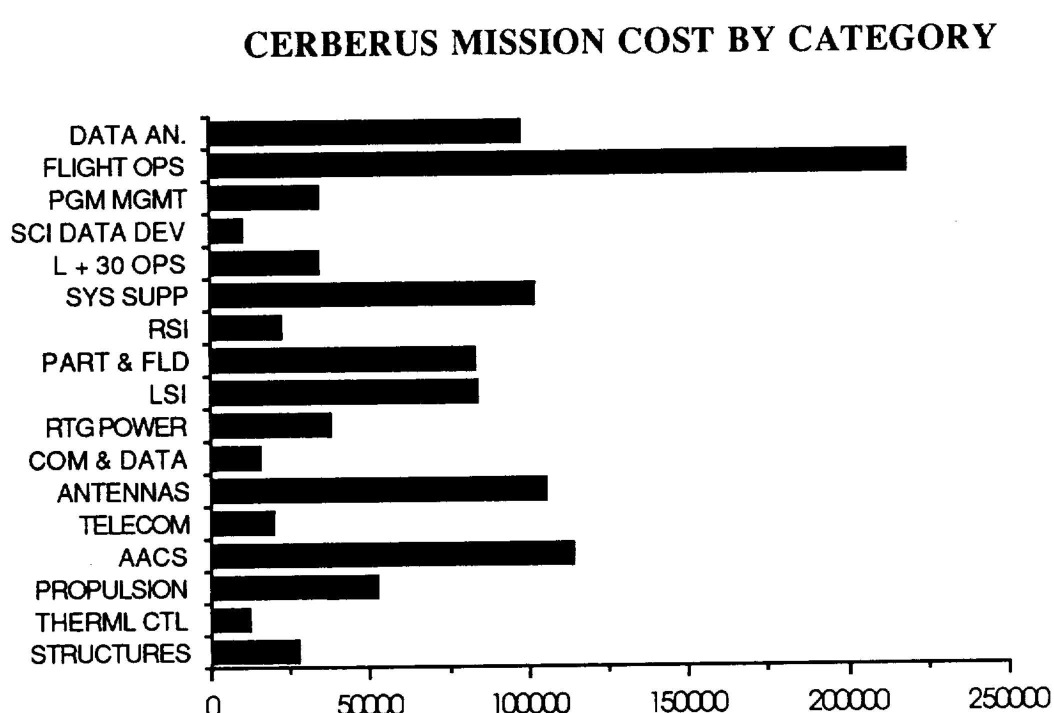

Figure 1-3: Category Cost - 1990 ($1000)

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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6. Kitchen, L. D., Manpower Cost Estimation Model for Automated Planetary Projects - 2, Report No. SAI 1-120-399-C2, Science Applications, Inc., September, 1973

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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

μ_{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

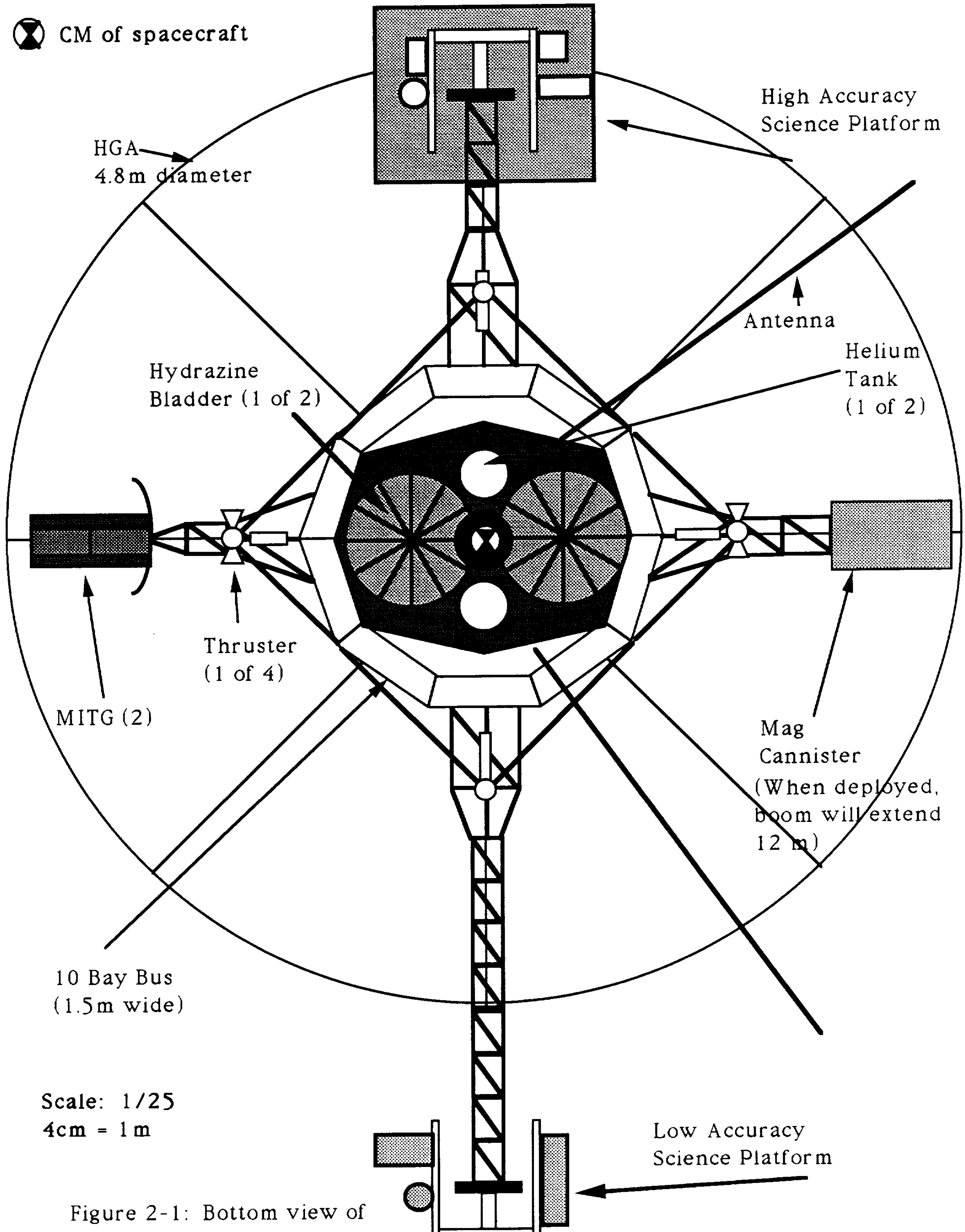

Figure 2-1: Bottom view of Cerberus

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.

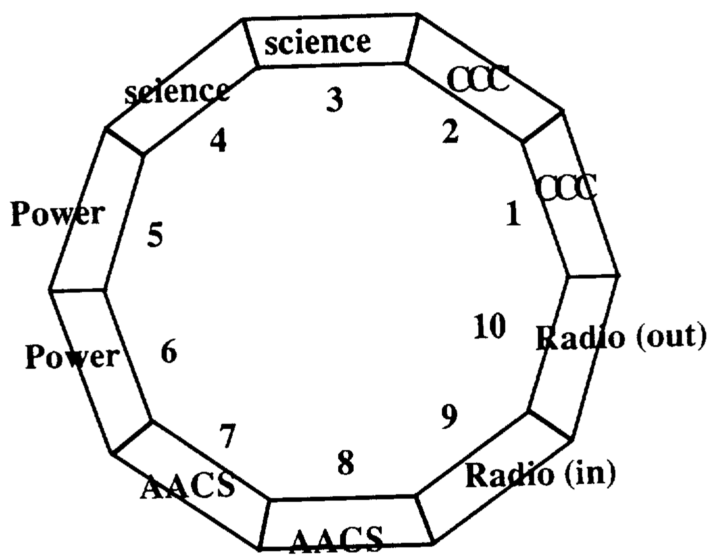

Figure 2-3: Electronics contained in each bay

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.

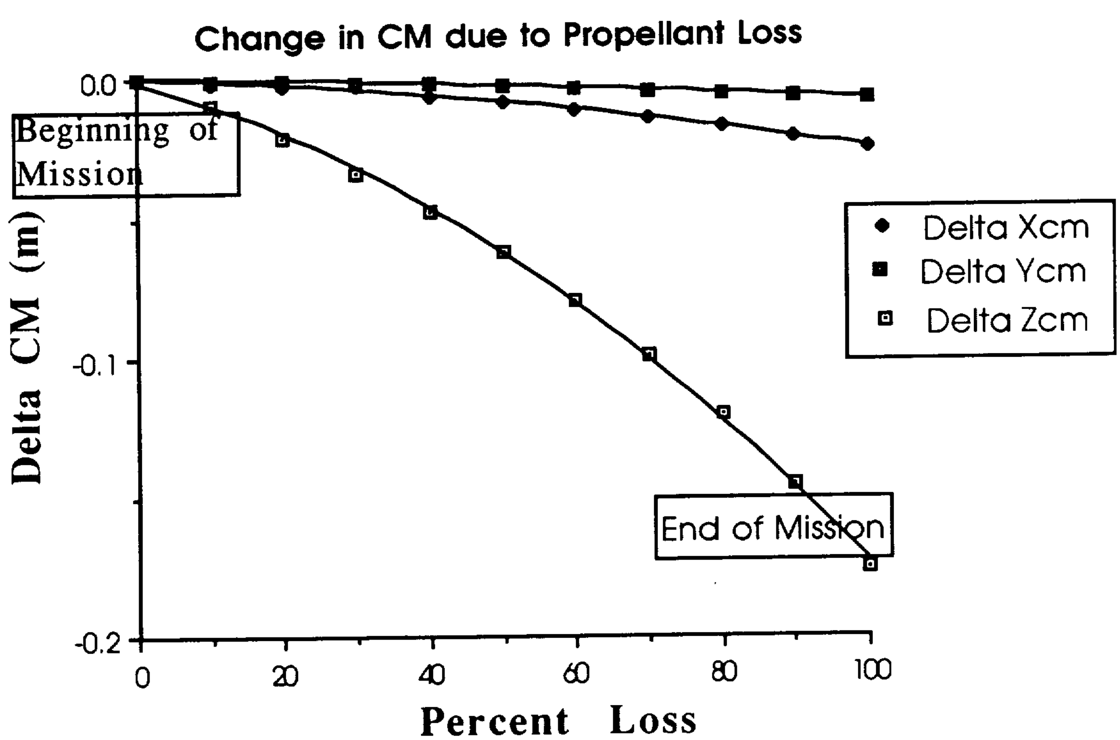

Figure 2-4: Center of Mass "Migration"

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.

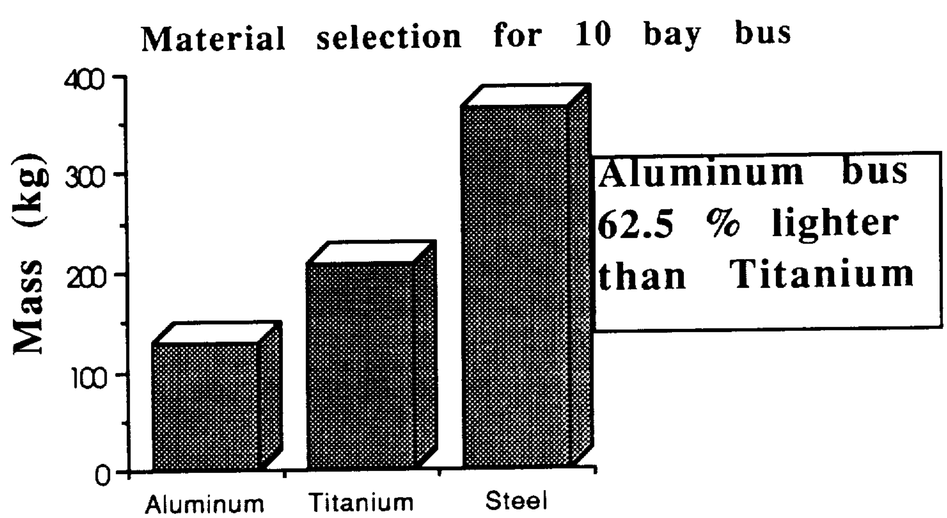

Figure 2-5: Material Selection for 10 bay bus

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 micrometeoroid 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.

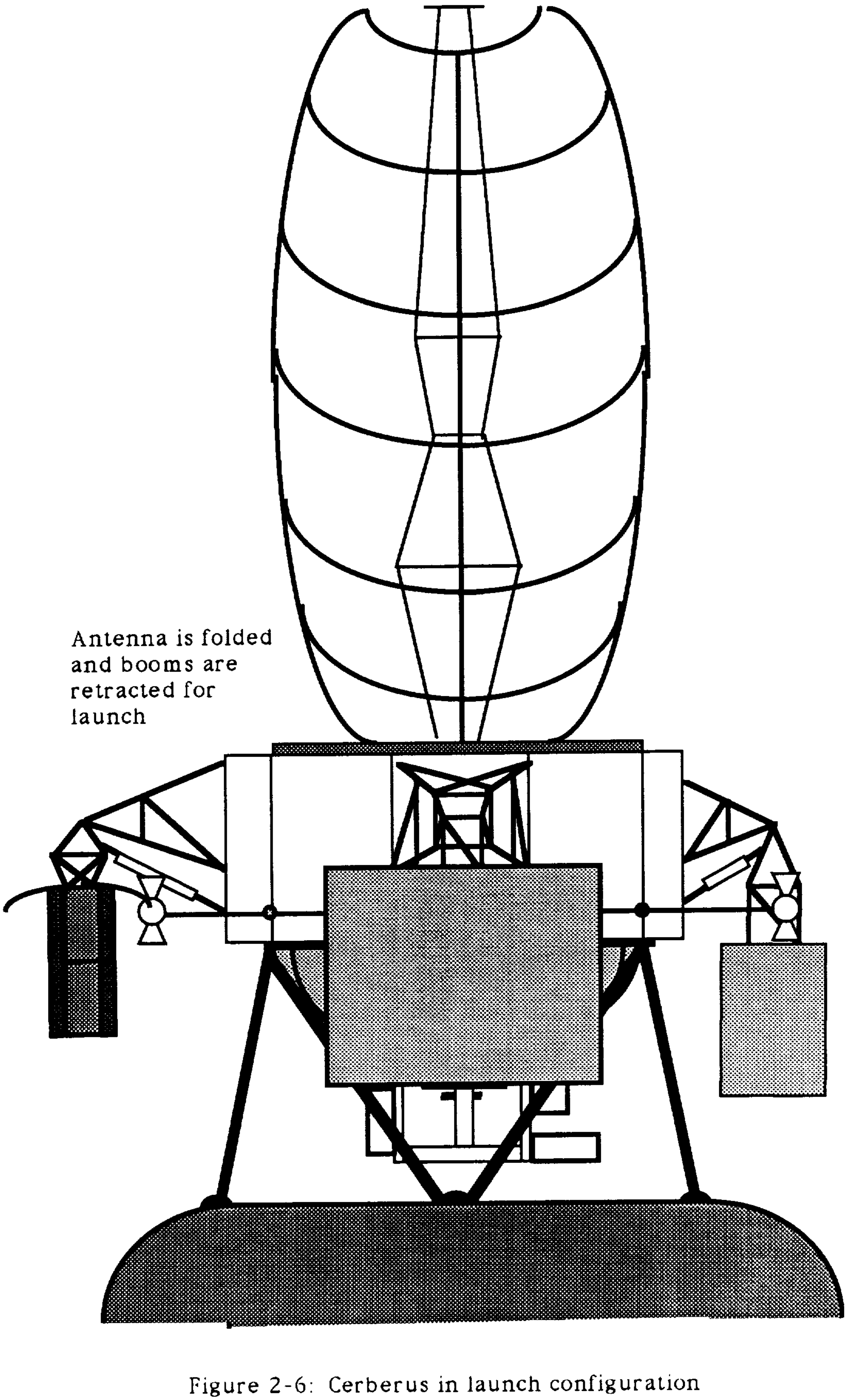

Figure 2-6: Cerberus in launch configuration

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.

Appendix B-2: 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].

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

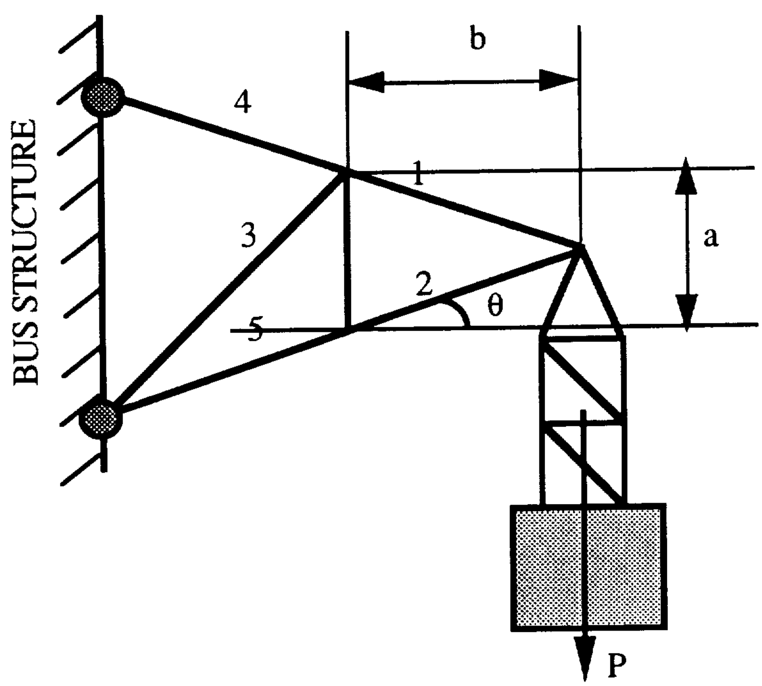

Figure B2-7: Truss configuration during launch

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.

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

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].

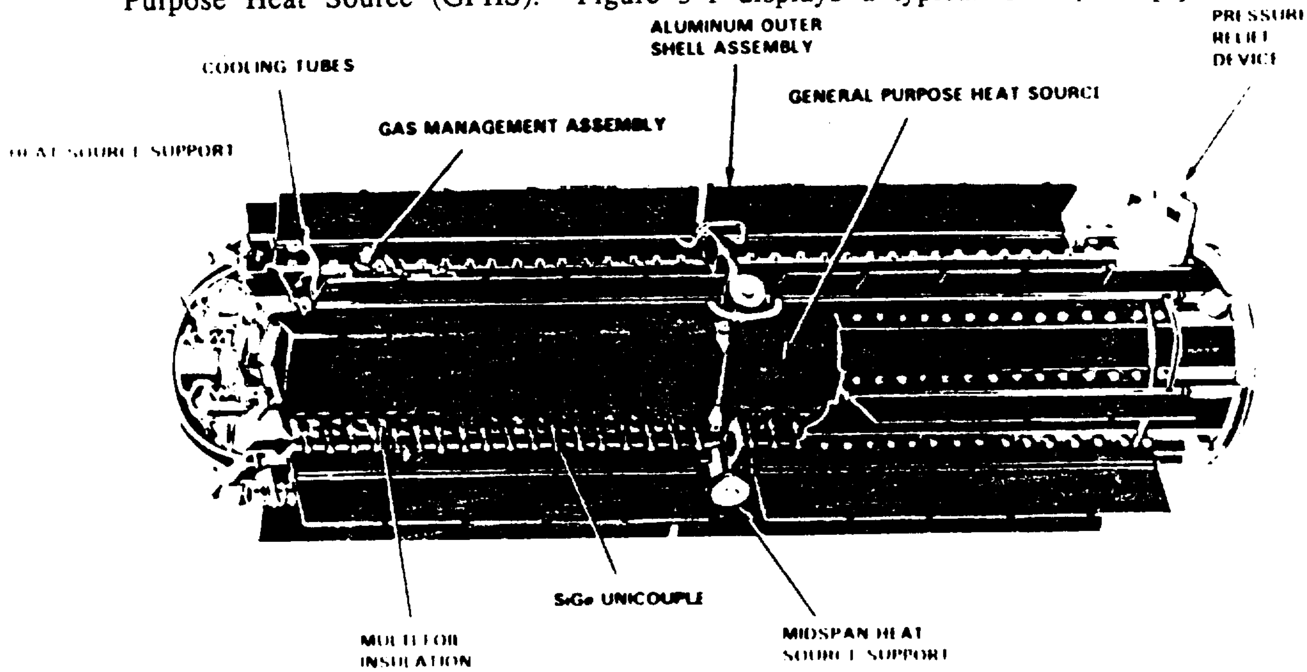

Figure 3-1: GPHS/MITG

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.

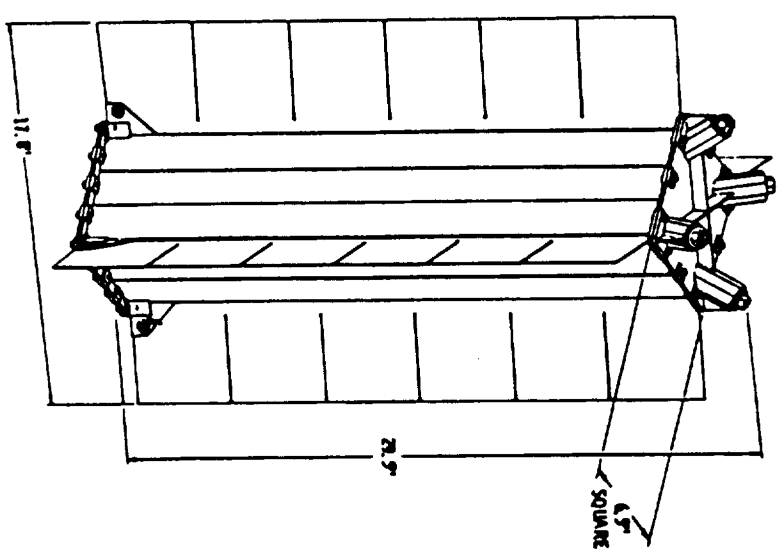

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Figure 3-2: Showing the MITG unit

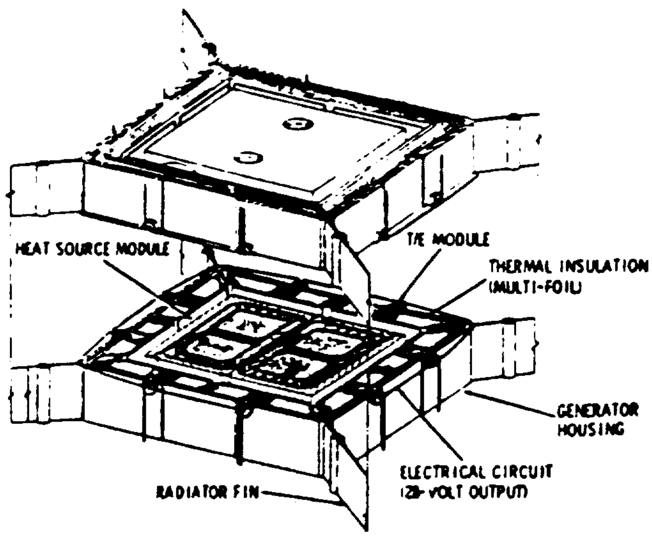

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Figure3-3: Showing a typical MITG slice

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

GPHS RTG (SiGe, 290 w)			Δ	MITG (SiGe/GaP, 282 w)		
[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	[illegible]
[illegible]			[illegible]			
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	
[illegible]			[illegible]			
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]			[illegible]			
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
Totals		[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.

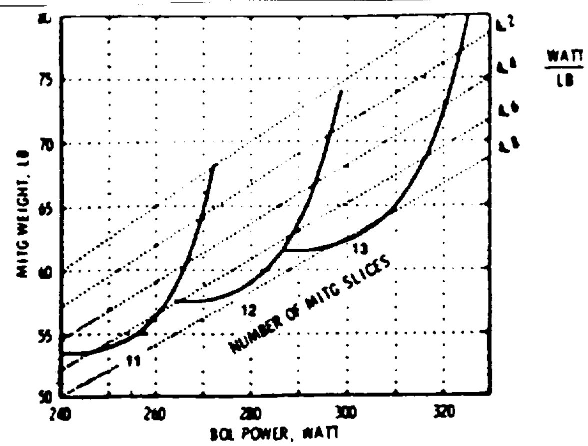


Figure 3-4: Performance curves for MITG

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.

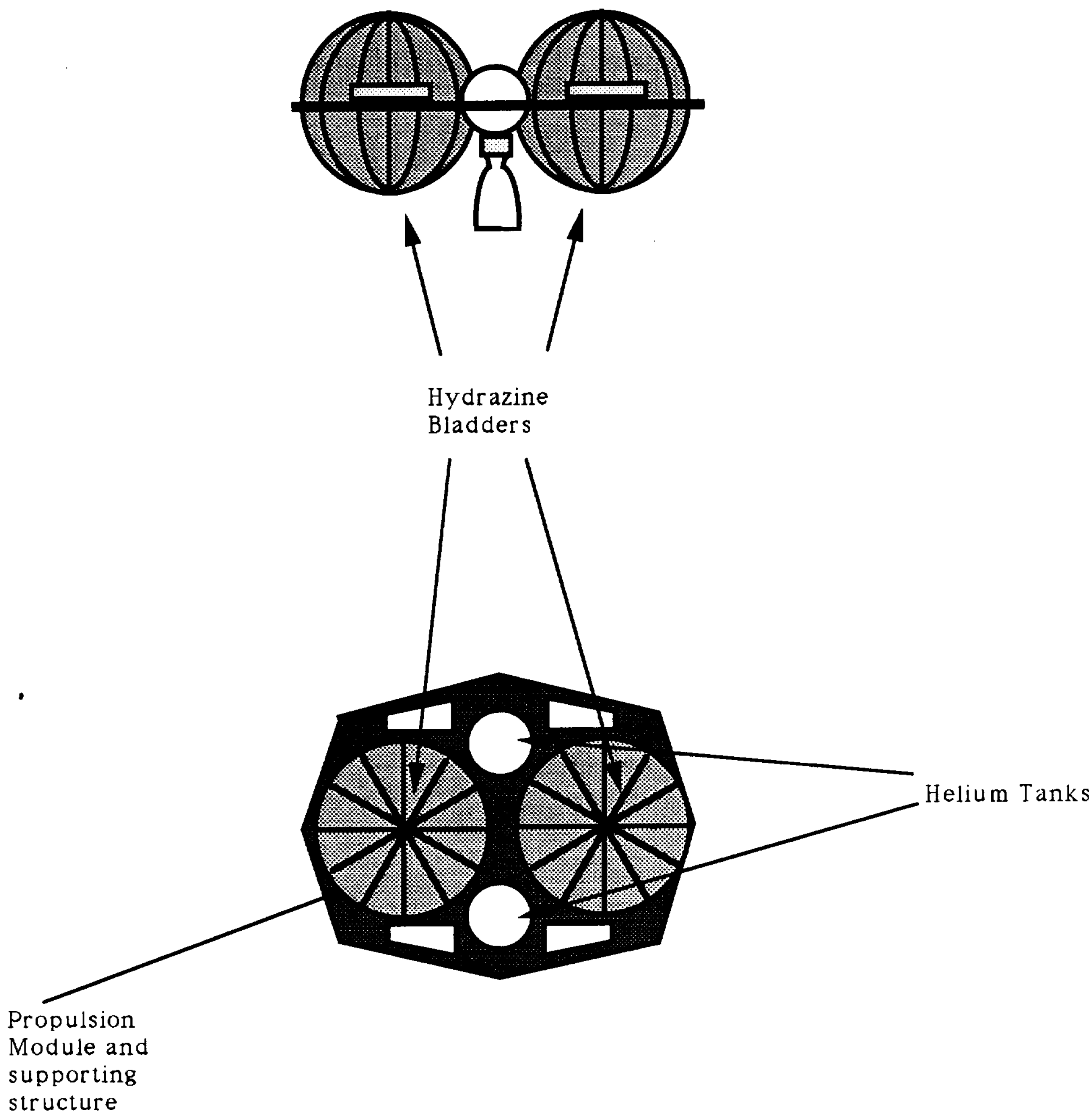
A 400 N main engine was selected to deliver the Δv's. Rough conservative calculations of the thrust times needed to perform each Δv are given in Table 3-6. The equations and an explanation of the estimates are given in Appendix C. These times are still feasible for approximate impulsive maneuvers. Furthermore, 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

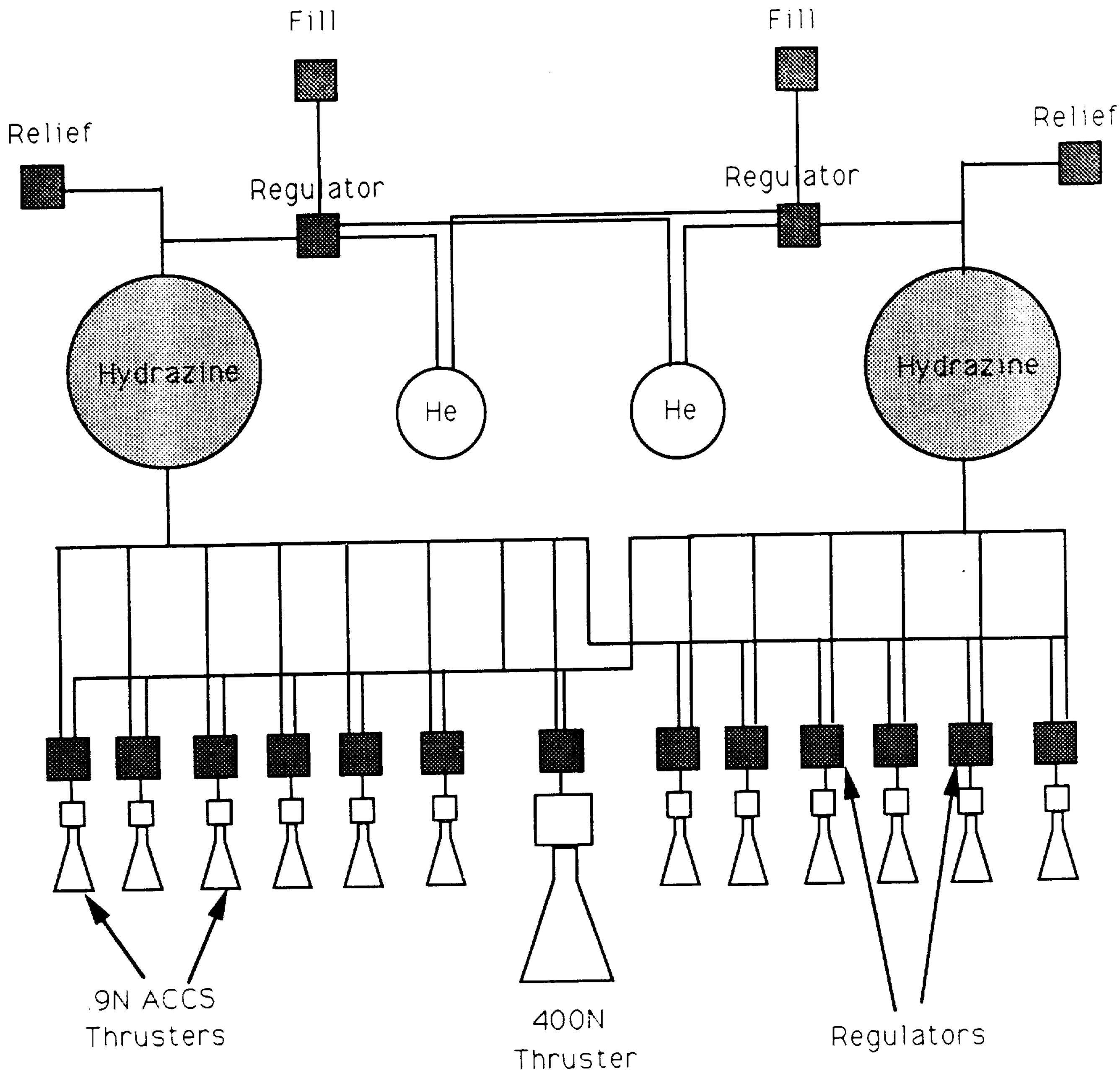


Figure 3-6. System configuration

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.

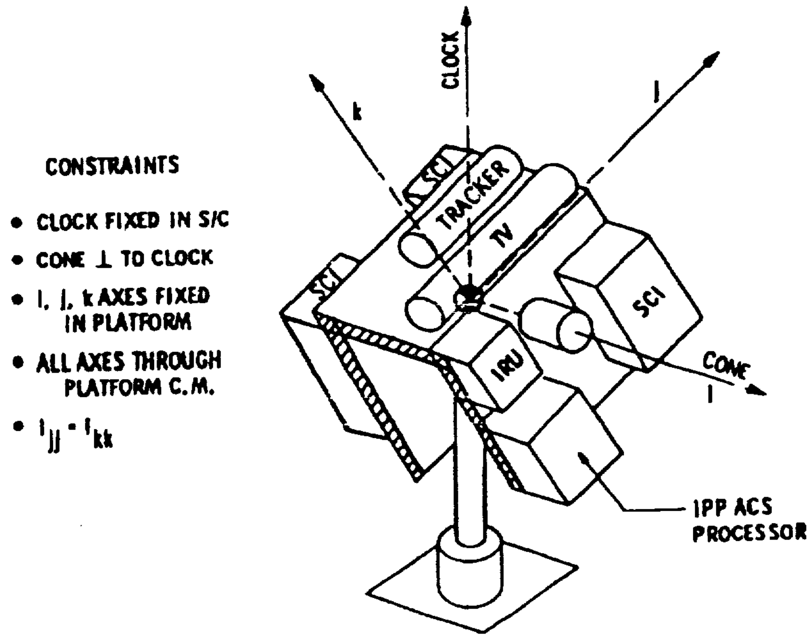

Figure 4-1: IPPACS Configuration (Adapted from [8])

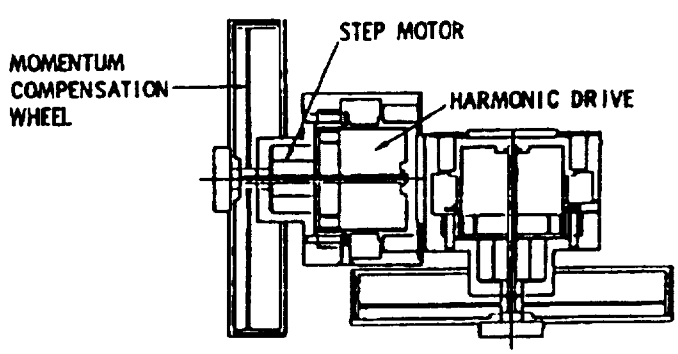

Figure 4.2: Actuator Orientation for Two Articulation Degrees of Freedom (Adapted from [8])

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

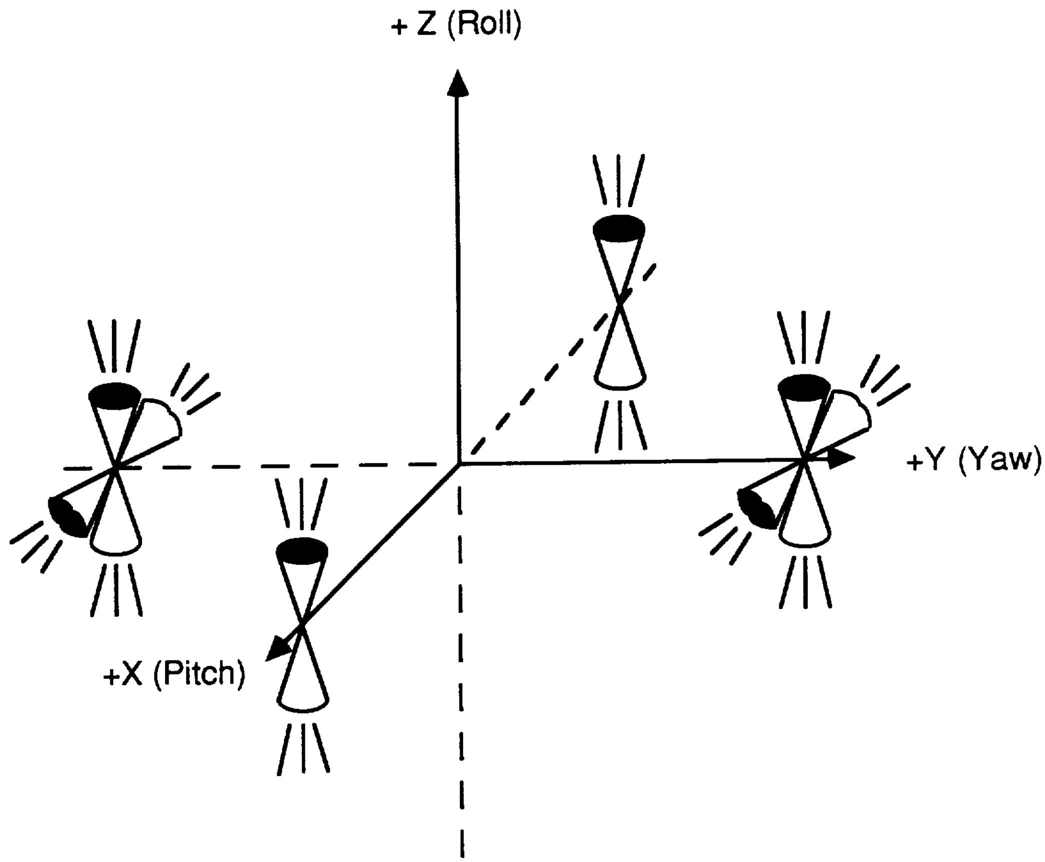

Figure 4-3: Placement of the Thrusters With Respect to the Principal Axes of the Spacecraft

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

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

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

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

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

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]

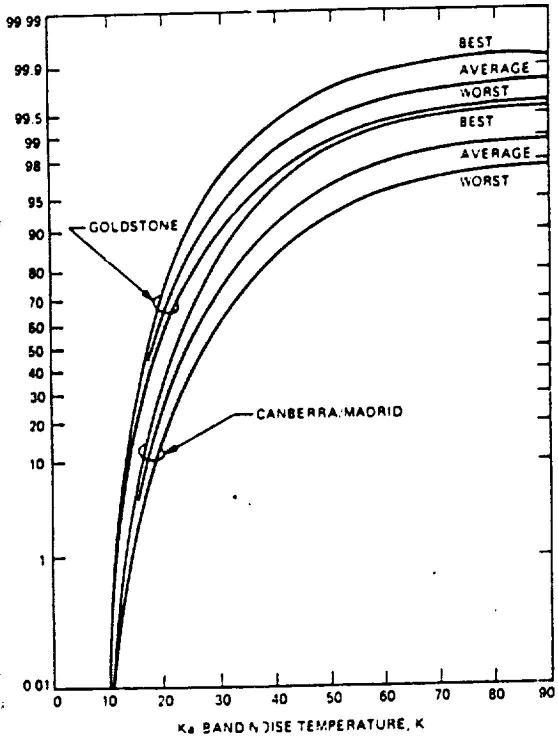

Fig.6-1: Ka-band atmospheric noise temperature statistics: all sites, 30-deg elevation angle.[27]

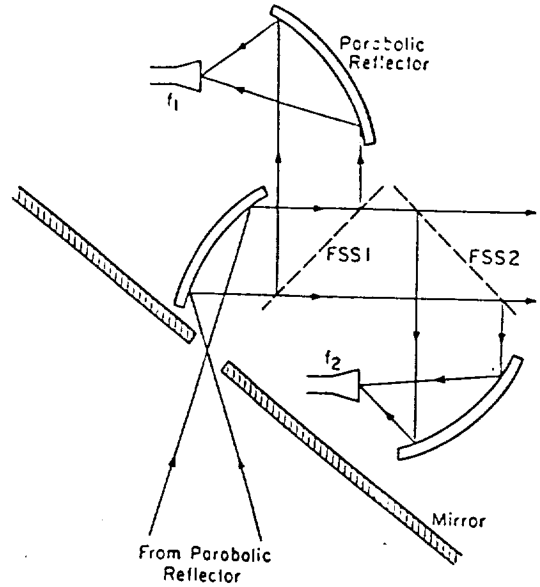

Fig.6-2: Multiple feeds for different frequency bands can be arranged with beam waveguides, and frequency selective surfaces.[15]

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 $$\frac{P_r}{P_T}=\frac{\mu 16 A_R A_T}{\lambda^2 L^2}$$

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 $$P_r=P_t L_t G_T L_{TP} L_S L_A L_P L_{RP} G_R L_R$$

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°K

transmission

L_S - space loss

L_P - polarization loss between antennas

G_R - receiving antenna gain

K - Boltzmann's constant

T - temperature (in °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	LGA at Jupiter
P_r=9.05E-17 W	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°K(100)}

=21.85 kHz

Pn=1.38E-23*3°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)	B=1685 kHz $\log_2$(100+1)
=145.5 kbps	=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.

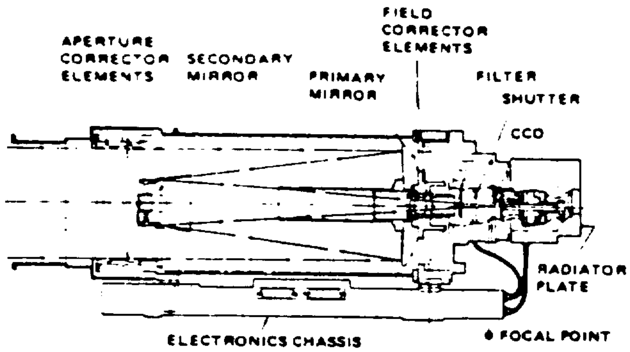
Particles and fields experiments will probe the space around Pluto as well as interplanetary space to enhance our knowledge of the solar wind and its interaction with Pluto. Four magnetometers will measure magnetic field intensity along the spacecraft trajectory allowing scientists to estimate the shape 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

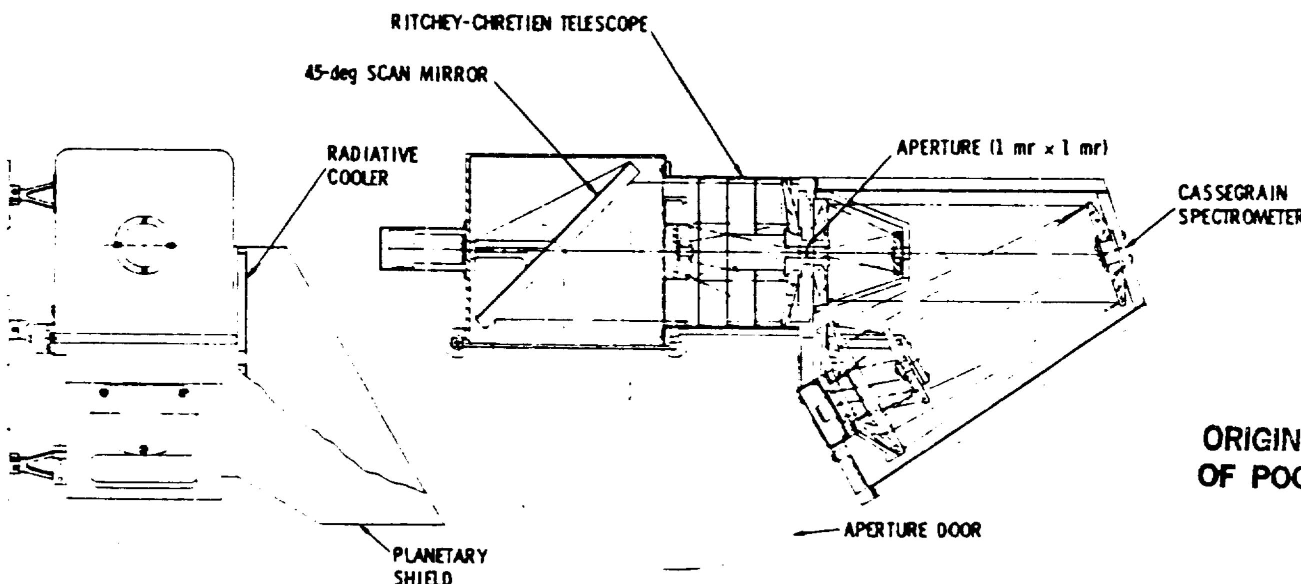
With the science objectives determined, it is necessary to select the best off the shelf hardware to run the experiments while conforming to mass, size, cost, and power constraints. To perform the imaging, the solid state imaging device (SSI) from Galileo will be inherited (See Fig. 6-1.). This system 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]

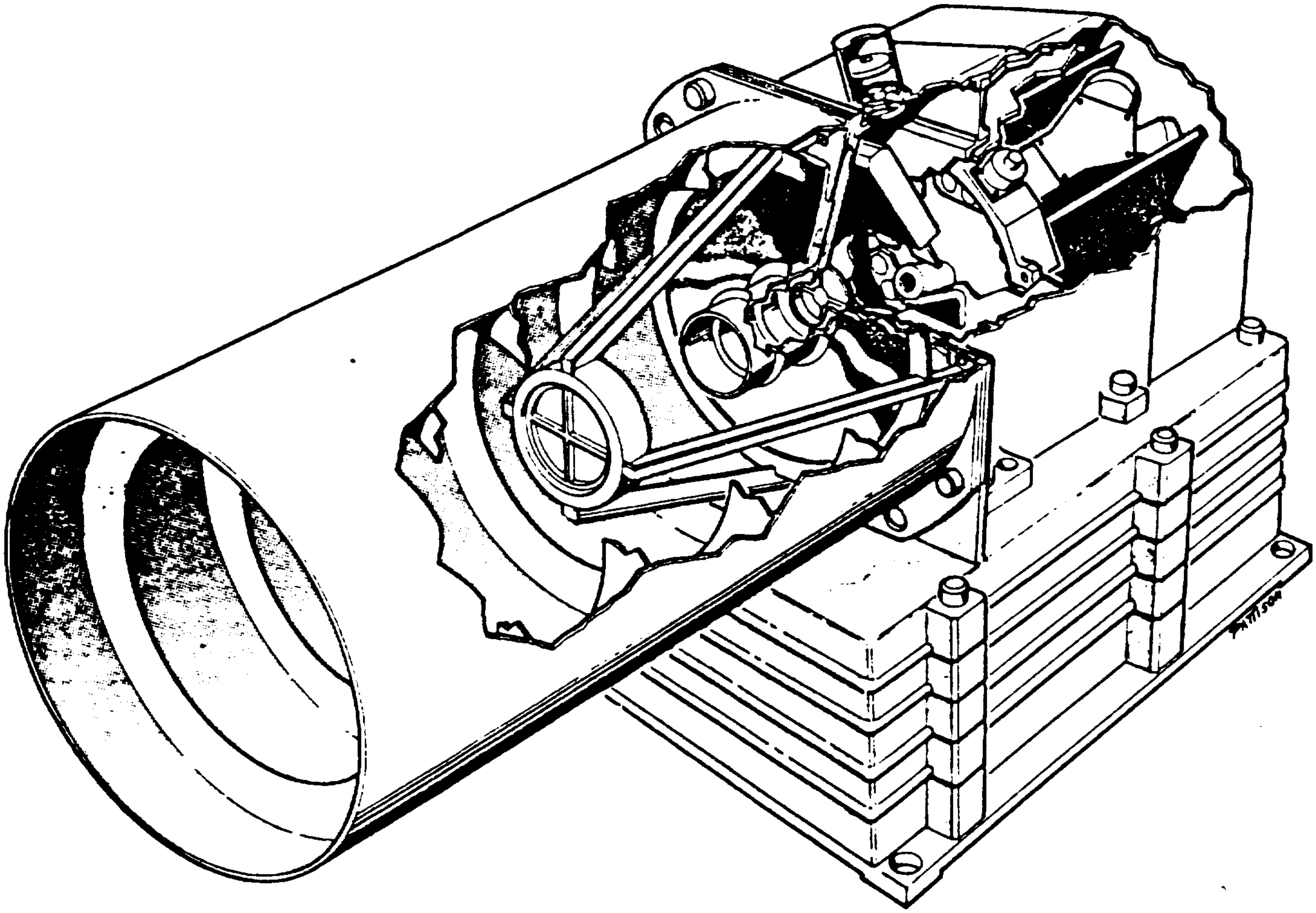
A Photopolarimeter/Radiometer (PPR) device identical to that aboard Galileo will be used on Cerberus (Fig. 6-3). This configuration is advantageous because two experiments share the same equipment which reduces spacecraft weight and complexity. The PPR must be kept at about 223K. and may not be pointed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

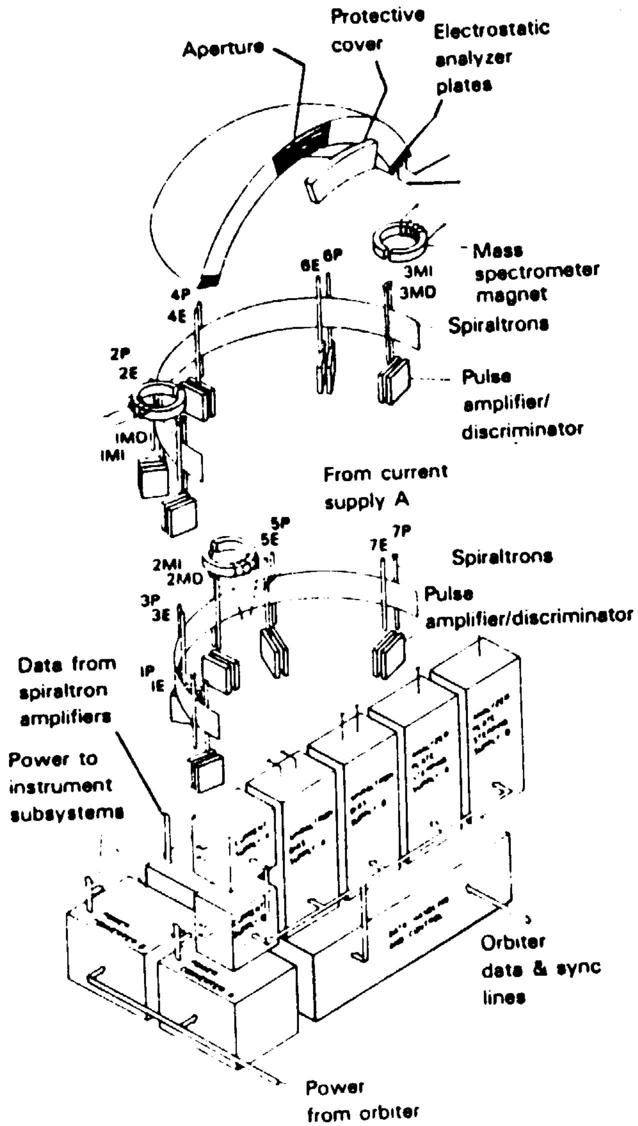

Figure 6-4: Plasma instrument - exploded view [4]

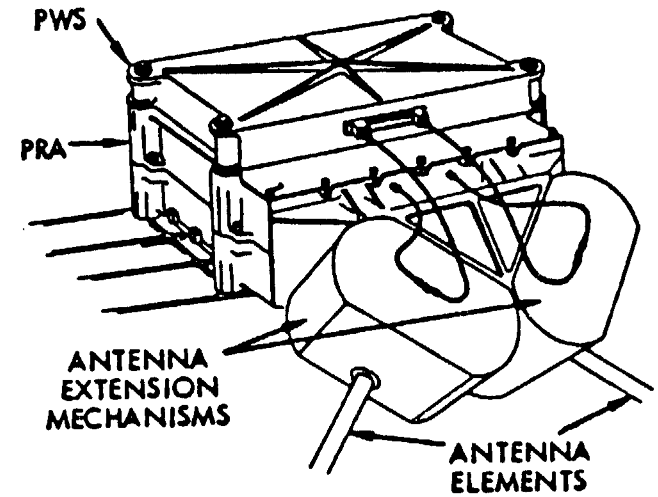

Figure 6-5: Plasma wave and radio astronomy instrument [5]

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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